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STUDY OF LIQUID OXYGEN/LIQUID HYDROGEN AUXILIARY PROPULSION SYSTEMS FOR THE SPACE TUG

FINAL REPORT

by

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16. Abstract

This report considers several design concepts that permit use of a liquid-liquid (as opposed to gas-gas) oxygen/hydrogen thrust chamber for attitude control and auxiliary propulsion thrusters on the Space Tug. The best of the auxiliary propulsion system concepts are defined and their principal characteristics, including cost as well as operational capabilities, are established. Design requirements for each of the major components of the systems, including thrusters, are developed at the conceptual level. The competitive concepts considered use both dedicated (separate tanks) and integrated (propellant from main propulsion tanks) propellant supply. The integrated concept is selected as best for the Space Tug after comparative evaluation against both cryogenic and storable propellant dedicated systems. A preliminary design of the selected system is established and recommendations for supporting research and technology to further the concept are presented.

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FOREWORD

This report presents the complete results of a Study of Liquid Oxygen/Liquid Hydrogen Auxiliary Propulsion Systems for the Space Tug. It covers both conceptual and preliminary design phases of study and culminates in preliminary design specifications and technology development plans for the system.

The study was conducted by the Space Division of Rockwell International Corporation under Contract NAS3-18913 with the National Aeronautics and Space Administration, Lewis Research Center (LeRC). Contract technical direction was provided by J. P. Wanhainen, LeRC Program Manager. Space Division Study Manager was J. F. Nichols. Study effort on auxiliary propulsion engines was performed under a subcontract by Aerojet Liquid Rocket Company, J. D. Sims, Program Manager.

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1. SUMMARY

This study considered several possible methods for applying the liquid-liquid cryogenic oxygen/hydrogen auxiliary propulsion concept to the Space Tug. This concept, which is based on maintaining the propellant in the liquid phase from storage to thruster inlet, is much simpler than previous oxygen/hydrogen systems which converted the propellant to vapor for distribution to the thrusters.

The study revealed that the best liquid-liquid concept is an integrated design which uses the Tug main propulsion tanks as the propellant source--as opposed to a dedicated design which uses its own separate tanks. The selected integrated design utilizes capillary reservoirs for the propellant source during zero-gravity. The reservoirs are refilled on demand during main or auxiliary engine velocity maneuvers. During sustained auxiliary propulsion velocity maneuvers, propellant flows through the reservoirs, so that they are not depleted. The selected concept also uses nonredundant pumps, accumulators, and a thruster feed manifold system, all actively cooled by hydrogen bleed.

The characteristics of the selected integrated system were compared in detail with dedicated cryogenic and storable propellant systems. The integrated concept was found to have superior (5 percent) mission payload performance. More significally, it also provides inherent vehicle versatility through the interchangeable availability of main and auxiliary propellants. This versatility has a pervasive influence on the Tug program. It permits the planning of Tug missions with auxiliary propulsion impulse requirements limited only by the total propellant loads and the reduced specific impulse of the liquid-liquid thrusters (86 percent of that for the main engine). Such versatility will permit a larger portion of as yet unforeseen Tug missions to be accommodated without payload penalty or resizing of the auxiliary propulsion propellant system.

Integrated system versatility also permits the auxiliary system to provide abort backup capability in the event of a main engine failure. Allowing for the additional gravity losses due to the lower thrust of the auxiliary system, Tug vehicle and payload recovery can be accomplished for up to 60 percent of the main engine duty cycle without any special provision. This capability results in Tug program cost savings which more than offset the higher development costs of the liquid-liquid auxiliary propulsion system. These savings are realized in reduced main engine development and scheduled maintenance costs, as well as in the probable abort recovery of the Tug vehicle and payloads.

A brief preliminary design effort resulted in further definition of the integrated auxiliary propulsion system. This phase of the study also resulted in several improvements which increase system reliability and decrease its weight without increasing cost.

It is recommended that technology development in support of the integrated system concept be pursued. To this end, a detailed plan, enumerating technology background, goals, and development approach is established for each of the four critical technology elements of the system: thruster, pumps, zero-g reservoir, and system thermodynamic control.

It is considered that these recommended technology developments are no more elusive or formidable than is justified by the potential Tug improvements. All of the technology needed is based on firm prior art--at least in building-block form, if not as a unit. The only requirement is to incorporate the technology into pre-prototype hardware so that design concepts may be verified experimentally.

2. INTRODUCTION

Auxiliary propulsion system (APS) design studies for several versions of the Space Tug have been conducted by industry and government organizations over the last three years. These studies have invariably been only a small part of Tug vehicle studies intended to define the Tug and its program as an element of the Space Transportation System. The definition of a full-capability reusable Space Tug emerged from these studies, and the resulting vehicle satisfied the original objective in terms of performance and utilization. The full-capability Tug is programmed for initial use in 1984-87 and, as its name implies, could beneficially use what might be considered as a full-capability auxiliary propulsion system. This study considers Tug APS designs in more depth than was possible on the vehicle studies in the expectation that a superior APS can be evolved.

The earlier Tug studies emphasized vehicle performance and versatility and hence focused on high-performance main engines and gaseous oxygen/gaseous hydrogen APS. More recent studies have been conducted with less stringent performance goals and emphasis on reducing DDT&E costs. As a result, the focus in these later studies shifted to lower performing main engines and either bipropellant or monopropellant APS of restricted functional capability. Future events may cause the reassertion of performance and versatility as Tug driver requirements. The intent of this study is (1) to determine if another type of cryogenic APS concept--the liquid-liquid oxygen/hydrogen (liquid-liquid O/H) system--is advantageous to the Space Tug mission, and (2) to develop for the concept a preliminary system design at the same level of maturity as other candidate APS designs intended for use with a full-capability Space Tug.

Central to the study is the feasibility and potential advantages of a thruster which accepts oxygen and hydrogen in liquid form at the thruster inlet without requiring gas generation as an intermediate step. This concept originally was conceived for application to the Shuttle reaction control system (RCS). The Aerojet Liquid Rocket Company (ALRC) participated in that development, beginning in 1969, through a series of projects on the much larger 1500-lb thrust liquid-liquid O/H engine needed. Although this concept was not selected for Shuttle, the engine utilized a torch igniter which by itself was suitable with modifications as a 111-N (25-lb) thrust engine for a Tug APS employing the liquid-liquid concept. As a subcontractor in this study, ALRC has developed the parametric design and programmatic data for the Tug APS thruster. Their work with the Shuttle engine igniter forms the experimental basis for the current effort.

2.1 OBJECTIVES

The objective of the first phase of this study is to provide a data base leading to preliminary design of a liquid-liquid O/H auxiliary propulsion system for the full-capability reusable Space Tug. This includes the analysis of baseline conceptual designs at two total impulse levels, variation of important design parameters, integration with other vehicle systems, and comparison with earth-storable bipropellant and hydrazine monopropellant auxiliary propulsion systems. The end product of the conceptual design phase is a cryogenic system defined in detail sufficient to determine its advantages, if any, over comparable storable propellant designs.

The objective of the preliminary design phase is to define the best cryogenic APS by establishing component requirements, operating ranges, and refined weight estimates. In addition, the preliminary design analyses concentrate on improvements in key design areas of the single selected cryogenic system. These improvements provide a total design description which, together with a plan for technology development, raise the level of maturity of the design to that of the storable propellant APS designs for Tug.

2.2 STUDY DESCRIPTION

The performance of the study follows the diagram shown in Figure 2-1. In Task I, two baseline dedicated (separate propellant tanks) liquid-liquid O/H APS are defined with separate impulse ranges. Using the baseline systems as

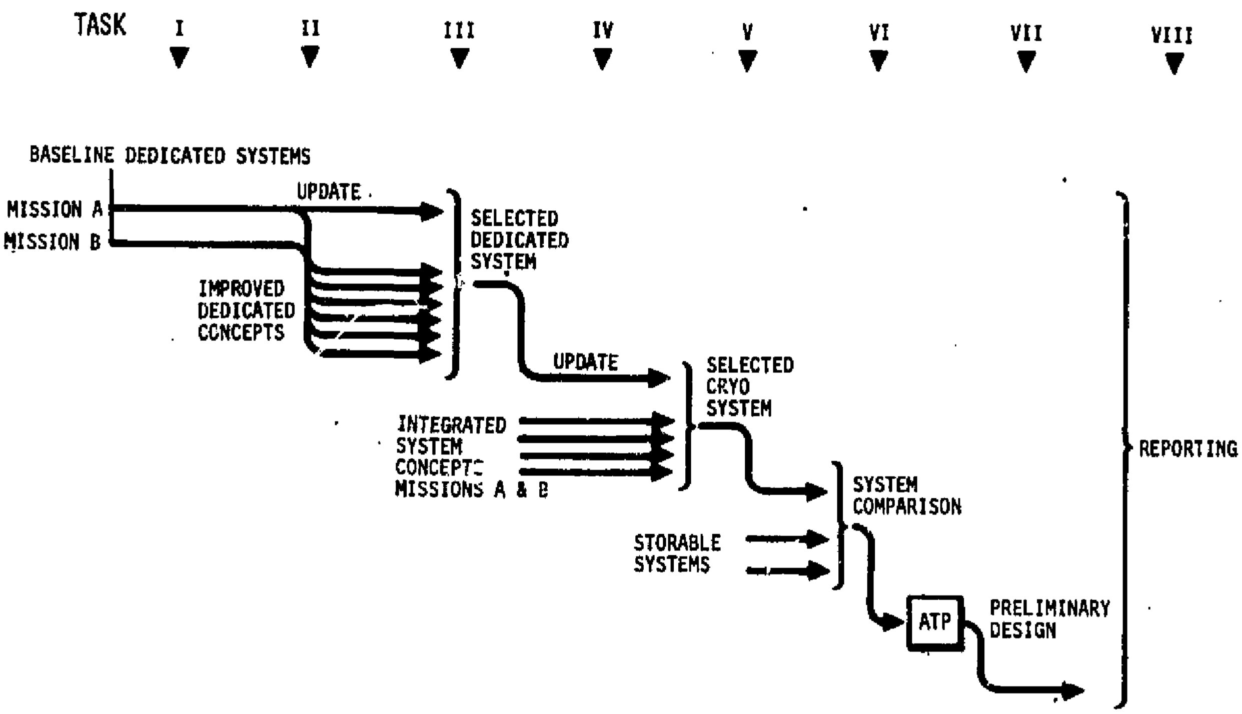

Figure 2-1. Study Flow Diagram

departure points, multiple concepts are synthesized in Task II, each with specific changes that show potential for improved performance. The most promising of these candidates are selected in Task III on the basis of best weight, cost, and reliability. In Task IV integrated APS concepts (propellant from main tanks) are synthesized. The surviving integrated concepts are compared with the dedicated concepts in Task V, again on the basis of best weight, cost, and reliability. The selected concepts are then compared with earth-storable monopropellant and bipropellant APS of References 1 and 2 in Task VI to determine relative advantages. The best system is shown to be a liquid-liquid O/H system, and the study concludes with preliminary design of the APS.

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3. STUDY REQUIREMENTS AND APPROACH

This section presents the requirements used in the study as taken from the contract statement of work, the two system studies described in References 1 and 2, and the MSFC Baseline Space Tug Requirements Document (Reference 3).

3.1 GENERAL REQUIREMENTS

1. The Tug is launched and returned to earth by the Shuttle.
2. The Tug is capable of synchronous equatorial orbit payload retrieval and multiple deployment.
3. The Tug is a reusable, full-capability vehicle.
4. The maximum operating time per mission is seven days.
5. The Tug life is 20 missions over a 10-year period, with interim replacement/refurbishment of components.
6. The Tug mission model involves 243 flights by a fleet of 17 vehicles.
7. The technology level is 1977-1980.
8. The Tug shall be designed for a successful mission completion probability of 0.97. This is interpreted to mean that the Tug will leave the Orbiter, perform its mission, and return to the Orbiter on 97 percent of its missions. This reliability figure does not account for any degradation which might be caused by Shuttle or payload failures. Tug recovery probability is 0.99.
9. The payload center of gravity shall be defined as being at the geometrical center of the maximum 4.57-m (15-ft) diameter, 7.62-m (25-ft) long payload envelope.
10. All Tug propulsion systems shall be required to be in a safe condition before reentry from orbit in the Orbiter.
11. The factors of safety for yield and ultimate strength are 1.1 and 1.4 times the limit load, respectively, for structure other than pressure vessels.

12. The factors of safety for pressure vessels are proof at 1.5 times limit pressure and ultimate (burst) at 2.0 times limit pressure.

13. The Tug shall be designed to vent propellant boiloff gases safely while on the launch pad, during launch and flight, in orbit, and during reentry while in the payload bay.

3.2 AUXILIARY PROPULSION SYSTEM GUIDELINES

1. All APS interfaces with the Tug are compatible.
2. The APS propellant tanks may be separate or integrated with the main propulsion tanks.
3. Helium pressurant may be stored in the main propellant tanks.
4. The APS geometry utilizes 16 engines in 4 quads spaced 90 degrees apart equally around the Tug circumference.
5. Rotational control of the Tug about three axes is by APS engine pairs.
6. Translational control of the Tug along three axes may be by 2 or 4 engines per axis.
7. The reliability goal for the APS is 0.996. This is based on an equal apportionment of the Tug reliability goal of 0.97 among 7 vehicle systems.
8. The APS is required to be fail-safe in the vicinity of the Shuttle.

3.3 REFERENCE MISSION

The Tug mission for the baseline conceptual APS design is described as a triple payload deployment mission which begins and ends at the Shuttle orbit. The mission results in the placement of three equal-weight payloads in synchronous equatorial orbit. Table 3-1 describes the mission timeline for propulsion events. Certain orbit maneuver burns totaling 370 m/sec (1220 ft/sec) are performed by the MPS in Mission Profile A and by the APS in Mission Profile B. The APS functional requirements for both profiles also show a total of 47 m/sec (155 ft/sec) for normal APS operations, including (1) Tug release and recovery by the Shuttle Orbiter, (2) payload release and surveillance, (3) roll axis steering during main engine burns, and (4) control of all three axes during coast periods and APS delta-V maneuvers. If propellant settling for MPS burns is performed by the APS, approximately 24.4 m/sec (80 ft/sec) additional velocity is required.

Table 3-1. Mission Timeline for Propulsion Events (Mission A or B)

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM										
1.75	3	0 7	DEPLOY TUG					3					
6.39	4	4 38	COAST NO 1										
6.43	5	0 3	PHASING ORBIT INSERTION		163								
7.74	6	1 19	COAST NO 2										
7.99	7	0 15	TRANSFER ORBIT INSERTION		2384								
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)			15							
13.13	9	5 6	COAST NO 3										
13.26	10	0 8	MISSION ORBIT INSERTION		1786								
13.30	11	0 2	ORIENT PAYLOAD					3					
13.30	12	0 0	DEPLOY PAYLOAD 1										
13.41	13	0 7	PAYLOAD SURVEILLANCE					9					
37.16	14	23 45	COAST NO 4										
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)			85							
89.34	16	52 9	COAST NO 5										
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)			85							
89.41	18	0 2	ORIENT PAYLOAD					3					
89.41	19	0 0	DEPLOY PAYLOAD 2										
89.51	20	0 6	PAYLOAD SURVEILLANCE					9					
95.34	21	5 50	COAST NO 6										
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)			85							
147.52	23	52 9	COAST NO 7										
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)			85							
147.58	25	0 2	ORIENT PAYLOAD					3					
147.58	26	0 0	DEPLOY PAYLOAD 3										
147.67	27	0 5	PAYLOAD SURVEILLANCE					9					
150.19	28	2 31	COAST NO 8										
150.27	29	0 5	TRANSFER ORBIT INSERTION		1783								
155.44	30	5 10	COAST NO 9										
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)			15							
155.54	32	0 4	PHASING ORBIT INSERTION		1693								
157.63	33	2 5	COAST NO 10										
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS		762								
163.56	35	5 53	COAST NO 11										
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK					8					
163.59	37	0 0	SHUTTLE DEORBIT										
164.29	38	0 42	TOUCHDOWN										
TOTALS					8571	370		47					

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM										
1.75	3	0 7	DEPLOY TUG					10					
6.39	4	4 38	COAST NO 1										
6.43	5	0 3	PHASING ORBIT INSERTION		536								
7.74	6	1 19	COAST NO 2										
7.99	7	0 15	TRANSFER ORBIT INSERTION		7820								
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)			50							
13.13	9	5 6	COAST NO 3										
13.26	10	0 8	MISSION ORBIT INSERTION		5859								
13.30	11	0 2	ORIENT PAYLOAD					10					
13.30	12	0 0	DEPLOY PAYLOAD 1										
13.41	13	0 7	PAYLOAD SURVEILLANCE					30					
37.16	14	23 45	COAST NO 4										
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)			280							
89.34	16	52 9	COAST NO 5										
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)			280							
89.41	18	0 2	ORIENT PAYLOAD					10					
89.41	19	0 0	DEPLOY PAYLOAD 2										
89.51	20	0 6	PAYLOAD SURVEILLANCE					30					
95.34	21	5 50	COAST NO 6										
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)			280							
147.52	23	52 9	COAST NO 7										
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)			280							
147.58	25	0 2	ORIENT PAYLOAD					10					
147.58	26	0 0	DEPLOY PAYLOAD 3										
147.67	27	0 5	PAYLOAD SURVEILLANCE					30					
150.19	28	2 31	COAST NO 8										
150.27	29	0 5	TRANSFER ORBIT INSERTION		5851								
155.44	30	5 10	COAST NO 9										
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)			50							
155.54	32	0 4	PHASING ORBIT INSERTION		5555								
157.63	33	2 5	COAST NO 10										
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS		2500								
163.56	35	5 53	COAST NO 11										
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK					25					
163.59	37	0 0	SHUTTLE DEORBIT										
164.29	38	0 42	TOUCHDOWN										
TOTALS					28121	1220		155					

3.4 VEHICLE DESCRIPTION

The baseline vehicle selected for this study is shown in Figure 3-1. It generally conforms to the Program 2 version defined in the General Dynamics Convair Aerospace (GDCA) Cryogenic Tug Systems Study (Reference 1). The overall vehicle weight breakdown is shown in Table 3-2. All checked weights are influenced by the APS and are different for each candidate concept. Furthermore, all MPS-related weights changed during the study with the choice of engine type and propellant settling method.

Brief descriptions of each major Tug system are presented in the following subsections.

MAIN PROPULSION SYSTEM (MPS)

A simple representation of the relative location of the MPS is shown in Figure 3-2. Figure 3-3 describes the main engine characteristics in terms of the actual data used in the study analyses.

The baseline MPS was changed during the study as a result of current studies by MSFC and LeRC. At the outset of the study, the MPS defined by McDonnell Douglas Astronautics Company (MDC) in Reference 2 for the Option 2 Tug was used. It incorporates a Pratt and Whitney Category IIA RL-10 main engine with tank head and pumped idle capability and operates at a mixture ratio of 6.0. The engine has a zero NPSH requirement (permits use of self-pressurized propellant tanks) through use of a low-speed inducer which provides two-phase pumping capability. It also utilizes tank head idle mode self-settling for engine start.

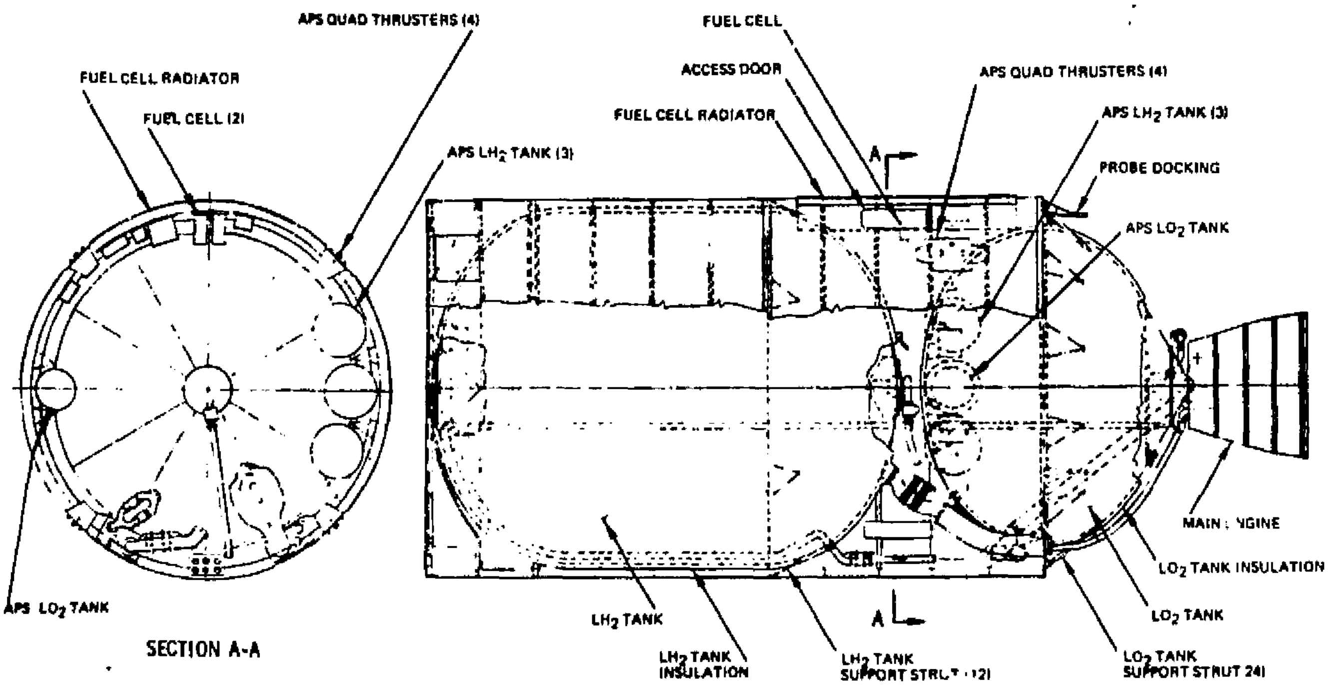


Figure 3-1. Baseline Vehicle

Table 3-2. Vehicle Weight Summary

Description		System A kg(lb)	System B kg(lb)
Structure	√	965(2127)	965(2127)
Thermal Control System		179(394)	179(394)
Astrionics		459(1012)	459(1012)
Propulsion		725(1599)	1078(2377)
Main propulsion		550(1213)	550(1213)
Auxiliary propulsion	√	175(386)	528(1164)
Dry Weight	√	2328(5132)	2681(5910)
Contingency (13%)	√	303(667)	348(768)
Dry Weight With Contingency	√	2631(5799)	3029(6678)
Nonusable Fluids	√	324(715)	351(774)
APS trapped propellant	√	4(8)	23(50)
APS trapped gas	√	3(7)	20(44)
MPS trapped propellant		52(115)	52(115)
MPS pressurant	√	150(331)	150(331)
MPS reserve (FPR)	√	115(254)	106(234)
Burnout Weight	√	2955(6514)	3380(7452)
Expended Fluids	√	23123(50977)	24039(52998)
Usable APS propellant	√	156(343)	1153(2541)
APS LH_2 bleed	√	10(21)	29(65)
Usable MPS propellant	√	22859(50396)	22759(50175)
Main tank boiloff vented		59(130)	59(130)
Fuel cell reactants	√	39(87)	39(87)
Gross Tug Weight at Tug/EOS Separation		26018(57361)*	27361(60320)*
Tug Chargeable Interface Provisions		1181(2603)	1181(2603)
Payload Weight	√	2225(4906)	883(1947)
Gross Weight at EOS		29483(65000)	29483(65000)
Mass Fraction (ΔV Propellant/1st Ignition Weight)	√	0.885	0.874

*Does not include 58.97 kilograms (130 pounds) vented prior to Tug/EOS separation

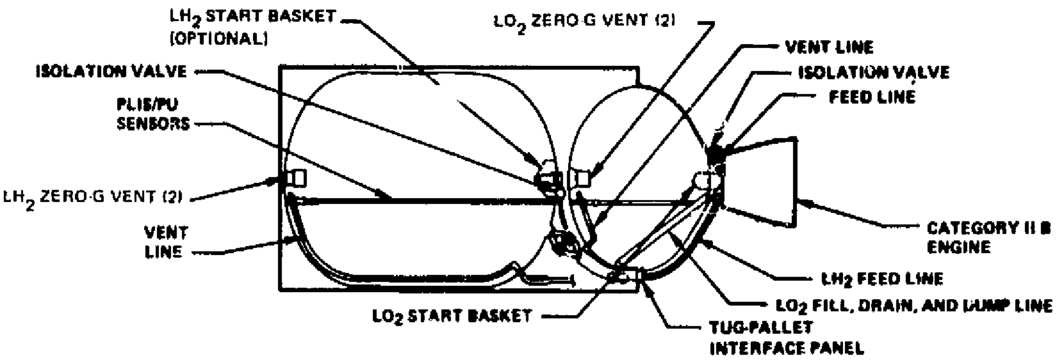
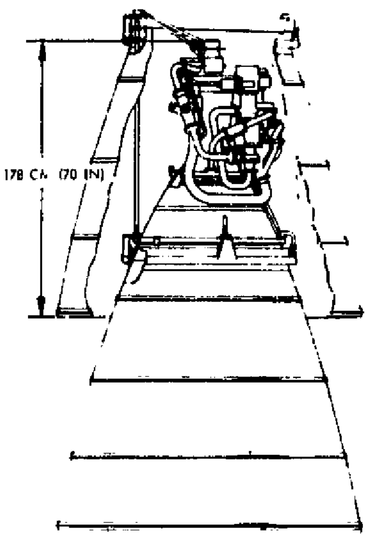


Figure 3-2. Main Propulsion System



<table>
<tr><th></th><th>INITIAL BASELINE CATEGORY IIA RL-10</th><th>STUDY BASELINE CATEGORY IIB RL-10</th></tr>
<tr><td>THRUST, KG (LB)</td><td></td><td></td></tr>
<tr><td>TANKHEAD IDLE</td><td colspan="2">71.2 (157)</td></tr>
<tr><td>MANEUVER (PUMPED IDLE)</td><td colspan="2">1701 (3750)</td></tr>
<tr><td>FULL</td><td colspan="2">6804 (15000)</td></tr>
<tr><td>MIXTURE RATIO</td><td></td><td></td></tr>
<tr><td>TANKHEAD IDLE THRUST</td><td colspan="2">4.0</td></tr>
<tr><td>MANEUVER THRUST</td><td colspan="2">6.0</td></tr>
<tr><td>FULL THRUST</td><td>6.0</td><td>5.6</td></tr>
<tr><td>CHAMBER PRESSURE, N/CM^2 (PSIA)</td><td></td><td></td></tr>
<tr><td>TANKHEAD IDLE</td><td>3.5 (5.1)</td><td>3.6 (5.20)</td></tr>
<tr><td>MANEUVER</td><td colspan="2">70.3 (102)</td></tr>
<tr><td>FULL</td><td colspan="2">275.8 (400)</td></tr>
<tr><td>SPECIFIC IMPULSE, N-SEC/KG (SEC)</td><td></td><td></td></tr>
<tr><td>TANKHEAD IDLE</td><td>3795.1 (387)</td><td>3579.4 (365)</td></tr>
<tr><td>MANEUVER</td><td colspan="2">4290 4 (437.5)</td></tr>
<tr><td>FULL</td><td>4503.2 (459.2)</td><td>4540.5 (463)</td></tr>
<tr><td>AREA RATIO</td><td>66.2/262</td><td>66.3/262</td></tr>
<tr><td>OPERATION</td><td></td><td></td></tr>
<tr><td>MANEUVER THRUST
FULL THRUST</td><td colspan="2">SATURATED PROPELLANTS
2/15 O_2/H_2 NPSH</td></tr>
<tr><td>CONDITIONING</td><td colspan="2">TANK HEAD IDLE</td></tr>
<tr><td>WEIGHT, KG (LB)</td><td>232.7 (513)</td><td>214 (474)</td></tr>
<tr><td>LIFE</td><td colspan="2">190 FIRINGS/5 HOURS</td></tr>
<tr><td>ENVELOPE, CM (IN)</td><td></td><td></td></tr>
<tr><td>LENGTH</td><td colspan="2">177/323 (70/127)</td></tr>
<tr><td>NOZZLE EXIT DIAMETER</td><td colspan="2">102/202 (40/79.6)</td></tr>
</table>

Figure 3-3. Main Engine

The present baseline MPS is autogenously pressurized, using the derivative IIB RL-10 engine with a 2/15 (oxidizer/fuel) minimum NPSH capability. The IIB engine also operates at zero NPSH in pumped idle mode (PIM), and at the start of PIM, engine bleed vapor is available and is used to supply prepressurization (bootstrap autogenous pressurization) prior to engine buildup to full thrust. The minimum tank pressure is set at 11 N/cm^2 (16 psia). The baseline MPS utilizes APS thrust for the pre-start ullage (settling) maneuver. As a study alternative, main-tank start baskets (capillary) are also considered and Tug performance is established for both cryogenic and storable APS.

The MPS includes the following subsystems:

1. Main Engine TVC - Apollo service propulsion system electromechanical actuators.

2. Propellant Utilization - Closed loop with capacitance probes.

3. Engine and Feedline Conditioning - Conditions feedline and engine while operating main engine in tank head idle mode.

4. Feed - LH_2: 7.6-cm (3.0-in.) multilayer insulation (MLI)-wrapped ducting to new 7.6-cm (3-in.) prevalve. Ducting transition to 8.1 cm (3.2 in.) prior to engine interface. LO_2: 10.2-cm (4.0-in.) insulated ducting and Parker 10.2-cm (4-in.) prevalve. Ducting transition to 11.7 cm (4.7 in.) prior to engine interface.

5. Fill and Drain - LH_2: 5.1-cm (2.0-in.) vacuum-jacketed ducting and Parker 5.1-cm (2-in.) valve. LO_2: 5.1-cm (2.1-in.) insulated ducting and Parker 5.1-cm (2-in.) valve.

6. Vent (Type for LH_2 and LO_2) - Four-valve configuration, two Calmec vent and relief valves and two Calmec flight vent isolation valves. Vent ducting through Tug-Orbiter interface 5.1 cm (2.0 in.). Flight vent 2.5 cm (1 in.).

7. Propellant Orientation - Alternatives include APS settling (baseline), main engine tank head idle mode self-settling, and start basket. Settling time is variable depending on quantity of LH_2 in tank.

8. Main Stage and PIM Pressurization - Autogenous-engine bleed vapor.

9. Pneumatics - S-IVB derivative valves and controls, Pressure Systems, Inc. 0.028-m^3 (1-ft^3) bottle.

THERMAL CONTROL

The thermal control/insulation system characteristics are shown in Figure 3-4.

ASTRIONICS

The significant characteristics of the astrionics system are listed in Table 3-3.

INITIAL BASELINE AUXILIARY PROPULSION SYSTEM

The initial baseline APS is a dedicated system with pressure-fed propellant. Separate tank capacity versions are defined for Missions A and B. The capacity for the Mission A version is 156 kg (343 lb) of propellant for a dry weight of 175 kg (386 lb), while the capacity for the Mission B version is 1153 kg (2541 lb) of propellant for a dry weight of 582 kg (1164 lb). The system mechanical operation is similar to that of the updated baseline system shown in Figure 4-1.

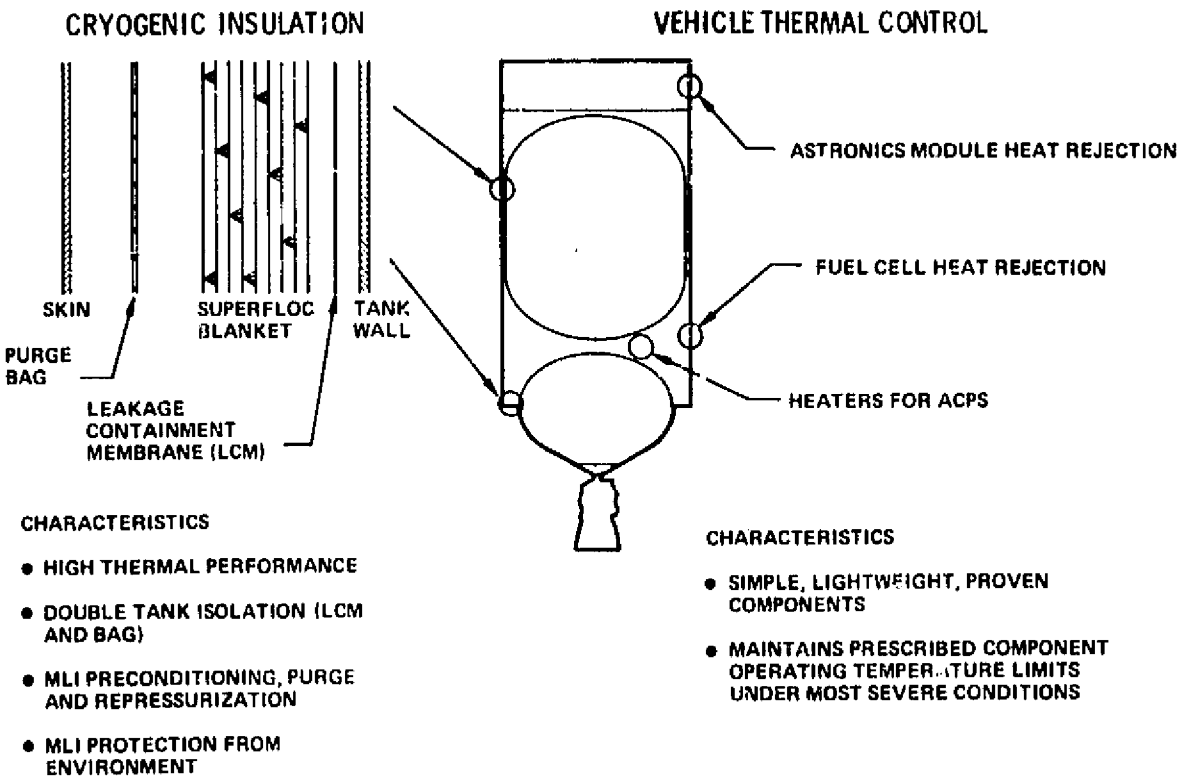


Figure 3-4. Insulation and Thermal Control

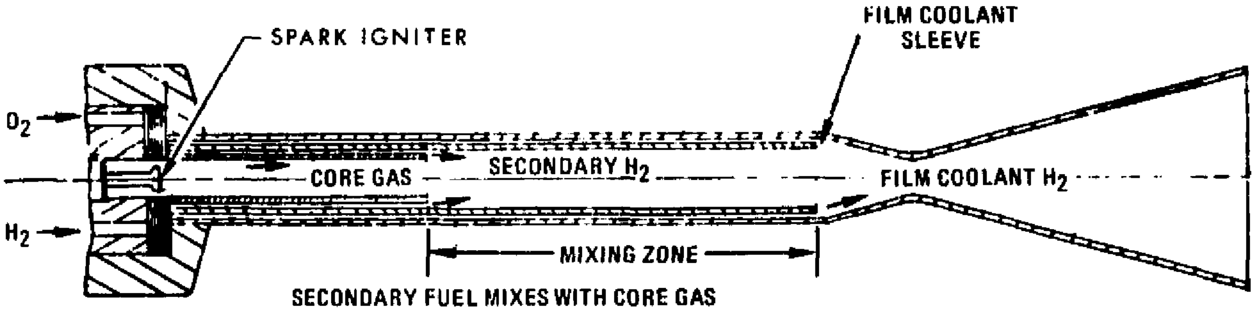
Table 3-3. Astrionics System Characteristics

Subsystem	Characteristics
Data Management	65K memory, data bus tape recorder
Guidance and Navigation	Electrostatic gyro
Flight Control	Electromechanical
Guidance Update	Level I autonomy star sensor and horizon sensor
Rendezvous and Docking	Laser radar
Communications	USB (NASA), Tug/ground, Tug/payload SGLS (DOD), secure, Tug/ground, Tug/payload
Instrumentation	Uses data bus
Electrical Power	Fuel cell with emergency battery and boost pump battery
Power Distribution and Control	Solid state and hybrid, boost pump inverters

Thruster Description

The baseline thruster design is depicted in Figure 3-5. For the initial baseline APS the specific impulse was assessed at 3680 N-sec/kg (375 sec). Later in the study, the dedicated APS candidates utilized a reassessed specific impulse of 3740 N-sec/kg (381.7 sec).

The igniter is a spark type with ignition being achieved at the forward end of the igniter in a highly oxygen-rich environment. For the baseline thrusters, all of the oxygen and 8 percent of the hydrogen is injected around the spark plug and ignition occurs at a mixture ratio of 50:1. The remaining 92 percent of the hydrogen is split between the two sleeves in the combustion chamber to cool the chamber wall and inner sleeve. The combustion chamber and nozzle extension material is columbium. The selection of columbium permits



SECONDARY FUEL MIXES WITH CORE GAS

FILM COOLANT SLEEVE: CONVECTIVE COOLING FOR MIXING ZONE

PROTECTS, INJECTS THROAT FILM COOLANT

THRUST, N (LB)	111	(25)
CHAMBER PRESSURE, N/CM^2 (PSIA)	103	(150)
MIXTURE RATIO	4	
NOZZLE AREA RATIO	50	
VACUUM SPECIFIC IMPULSE, N-SEC/KG (SEC)	3740.	(381.7)
THROAT AREA, M^2 (IN^2)	635×10^{-5}	(.0984)
THROAT DIAMETER, M (IN)	.009	(.354)
CHAMBER DIAMETER, M (IN)	.0188	(0.74)
CHAMBER LENGTH, M (IN)	.181	(7.14)
NOZZLE LENGTH (100% BELL) M (IN.)	.08	(3.2)
NOZZLE EXIT DIAMETER, M (IN)	.066	(2.6)
TOTAL THRUSTER LENGTH, M (IN)	.4064	(16.0)
THRUSTER ASSEMBLY WEIGHT, KG (LB)	3.0*	(6.7*)
LOX INLET TEMP, NOM, K (R)	91.7	(165)
LH_2 INLET TEMP, NOM, K (R)	27.8	(50)
PROPELLANT INLET PRESS, N/CM^2 (PSIA)	152	(220)
MATERIAL COLUMBIUM		

*ASSUMES REDUNDANT VALVES

Figure 3-5. APS Thruster Chamber - Initial Baseline operation at high gas-side wall temperatures 1644 K (2500 F) and results in higher thruster performance. The use of columbium at high operating temperatures has been demonstrated successfully on engine components such as the Apollo SPS engine nozzle extension.

The two-sleeve design was selected as a result of studies which showed that the single-sleeve design, a direct application of prior igniter technology, is very low performing. Performance was estimated to be approximately 10 percent lower than the two-sleeve design. In the single-sleeve design, all but 8 percent of the fuel is carried in the cooling sleeve and is released as film coolant upstream of the throat. This film coolant then mixes with the core gases (MR = 50:1) to provide additional thrust and specific impulse. To avoid chamber burnout at the throat, the film coolant must be injected at a short distance: 3.8 cm (1.5 in.) or less from the throat. The poor performance of the single-sleeve design results because of the poor mixing of the core gases with the film coolant.

The two-sleeve design adopted splits the fuel between the inner and outer sleeves in a manner dictated by the thermal analysis. Some of the fuel cools the chamber wall and is injected just upstream of the throat as film coolant. The remaining fuel coolant flow is injected from the short innermost sleeve. This fuel then mixes with the core gases (MR = 50:1) over a mixing length of approximately 11.4 cm (4.5 in.) thereby producing high performance.

Propellant System Description

For Mission Profile A, the LOX is stored in one 0.635-m (25-in.) diameter sphere and the LH_2 in three 0.762-m (30-in.) spherical tanks. For Mission B, a toroidal LH_2 tank of 0.660-m (26-in.) minor diameter and 1.676-m (66-in.) major diameter is required to avoid major Tug vehicle redesign. LOX is stored in nine 0.5715-m (22.5-in.) diameter spherical tanks. Figures 3-6 and 3-7 show the location of these tanks for Mission Profiles A and B.

The MLI insulation for the APS tanks is the same configuration as is used on the main propellant tanks. Polyurethane foam covered with goldized Kapton has been selected as the baseline design for the propellant line insulation.

Pressurization System Description

The pressurization system includes a 0.57-m (22.5-in.) diameter helium sphere located in the main LH_2 tank, a regulator to maintain the ullage pressure, and a LOX tank pressurant heater to avoid the adverse cooling effects of the helium, which is below the oxygen triple point. Each of the propellant tankage systems has a relief valve to protect it from overpressurization, primarily during periods of regulator lockup, and a solenoid-controlled vent valve for ground operations.

Thermodynamic Control Description

The thermodynamic control system maintains the propellant feed and storage system at an acceptable temperature level during the periods when

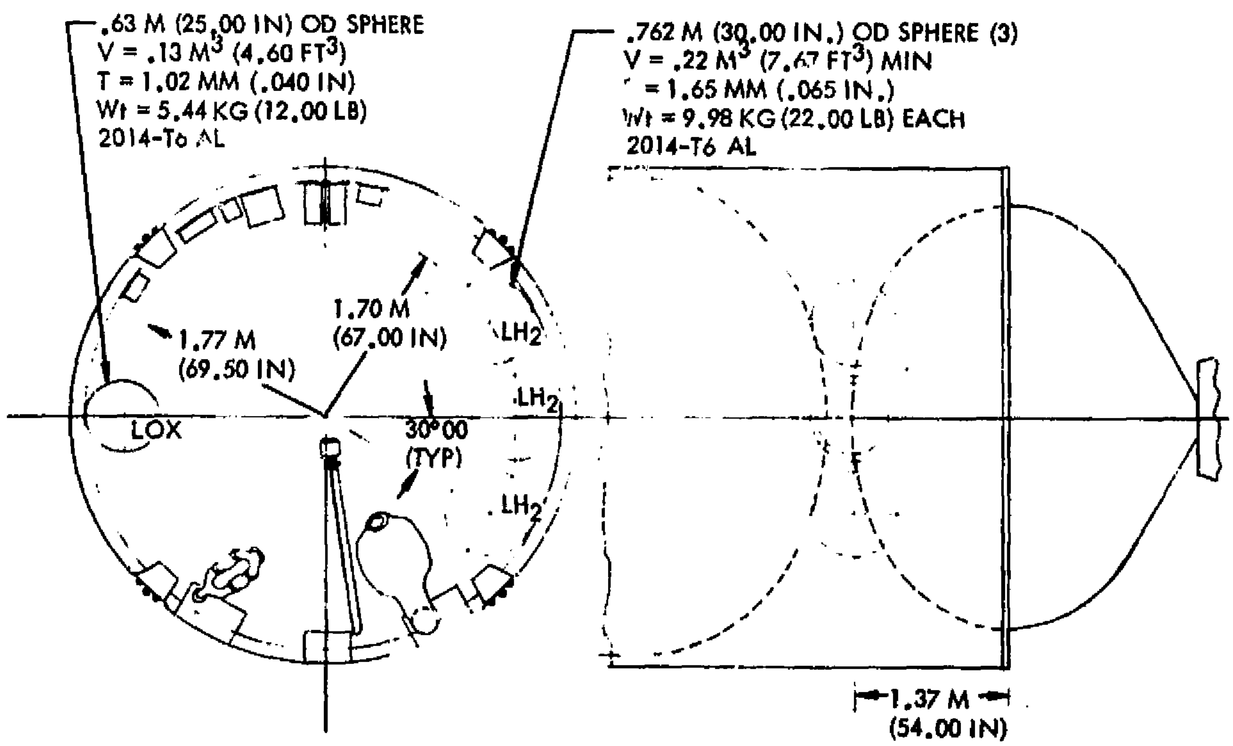


Figure 3-6. APS Propellant Tanks - Initial Baseline for Mission A

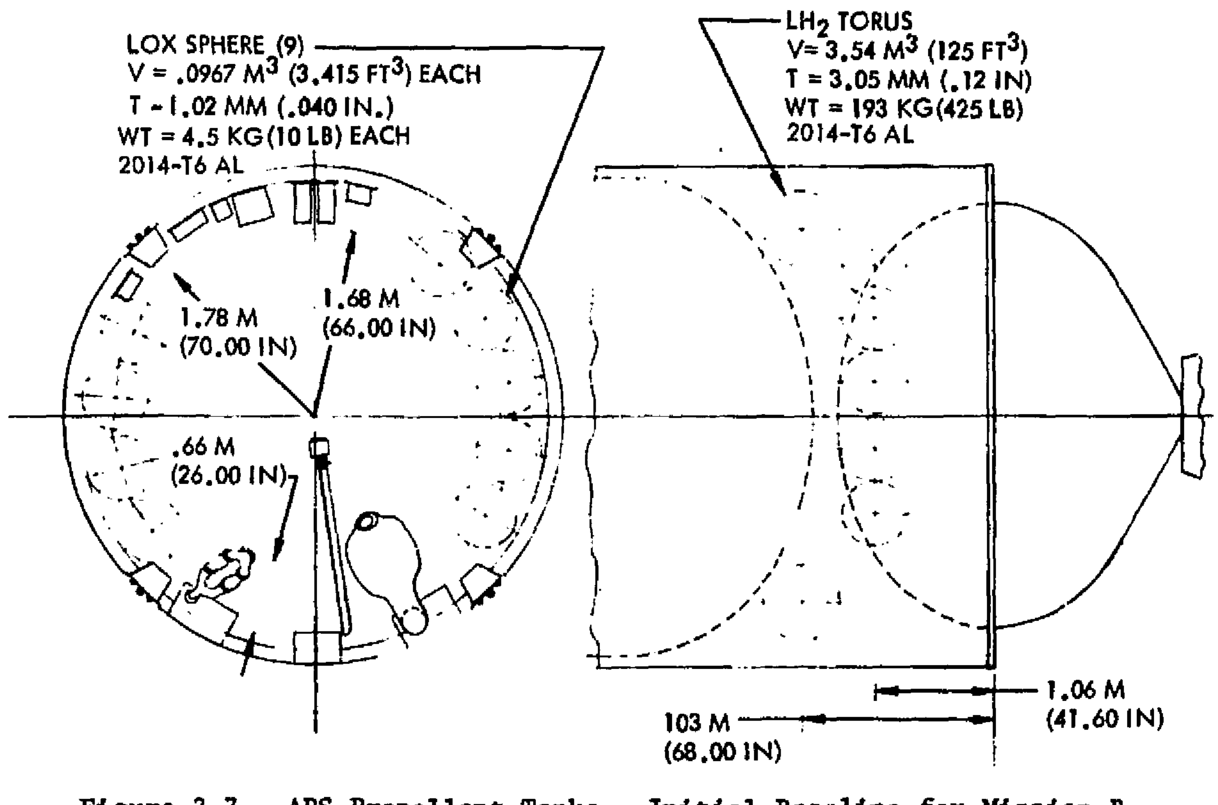


Figure 3-7. APS Propellant Tanks - Initial Baseline for Mission B

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the thrusters are not firing. This is accomplished by maintaining a small bleed coolant flow through the feedlines and the propellant tanks. In the hydrogen system, the LH_2 is drawn from the feedlines through a Joule-Thompson expansion valve. The resulting low-temperature, two-phase fluid is routed to the APS hydrogen tank through a line in thermal contact with the feedlines. The coolant enters the tank through tubing that is thermally attached to the internal propellant acquisition device as a finned heat exchanger. This heat exchanger cools the liquid and the ullage in the tank and acquisition device. After exiting the tank, the hydrogen bleed is routed through an oxygen bleed heat exchanger before being vented overboard through a nonpropulsive vent. The oxygen bleed flow is not expanded but instead is cooled as it passes through the bleed heat exchanger. It then cools the LOX lines and tank in a manner similar to that for the hydrogen system. Because of its pressure level, the LOX bleed is not lost but is returned to the main LOX tank.

Propellant Control Description

The totally passive propellant acquisition device within the APS propellant tanks is a separate can (volume is approximately 20 percent of tank volume). It has a screened slot at the bottom to allow propellant to flow into the can during positive g conditions to replace that used by the engines. The can contains a series of full horizontal retention screens, with narrow vertical wicking channels connecting the screens near the outside edge of the can. Additional wicking channels are located outside the can in front of the screened slot to form a liquid accumulator region. A central standpipe prevents the loss of liquid through the inlet screen. The size of the can is a function of the amount of gas expected to enter during vehicle maneuvering. Only the last (sump) tank of the three series-connected LH_2 tanks of Mission A require the can type of acquisition device; the two upstream tanks require only a series of centrally located wicking channels and retention sump. The nine-LOX-tank configuration for Mission B has three three-in-series tank units, each with a sump tank and two upstream tanks.

4. DEDICATED CONCEPT DEVELOPMENT

The dedicated APS concepts were generated through a series of parametric studies which provided improvements to an initial baseline system concept. Compatible combinations of the improvements, together with changes which increased reliability and reduced cost and weight, were employed in each candidate concept.

Two of the parametric studies investigated the effects of thrust level and angle of application on the dynamics of vehicle operations. These studies resulted in reduction of the minimum impulse bit size from 50 to 25 msec.

The third parametric study was devoted to determining the influence of engine design parameters on engine and system characteristics. The basic engine design and performance data resulted from parametric analyses by the ALRC subcontractor. This study resulted in recommended mixture ratios and area ratios to provide optimum performance. These three studies are discussed in detail in the Design Analysis section.

In the fourth parametric study, alternate propellant storage and feed system concepts were synthesized. Initially, preliminary screening was made of possible combinations of liquid acquisition, thermodynamic control, and propellant feed techniques. These combinations form the salient differences between the candidate dedicated system concepts and are discussed in this section.

The parametric studies also revealed the inability of dedicated systems to perform large total impulse tasks efficiently. The APS for Mission B is defined to have the capability of performing all attitude control, propellant settling, and delta-V maneuvers totaling at least 419 m/sec (1375 ft/sec). The impulse requirement is approximately seven times that of Mission A. In evaluating the Mission B capability some disadvantages were found: the mission payload capability decreases drastically and propellant usage increases sharply and creates serious volume and thermal control problems for the APS tank designs.

Since these shortcomings are a function of total impulse only, system configuration changes which could evolve later in the study would have no effect on the conclusion that a dedicated Mission B is not a viable concept. In consequence, no further effort was expended on dedicated Mission B concepts. The requirement for Mission B capability is retained, however, for the integrated APS concepts.

4.1 BASELINE THRUSTER

The configuration and performance characteristics of the liquid-liquid O/H thruster used in the dedicated APS concept development studies is described in the Study Requirements and Approach, Section 3.

4.2 PRELIMINARY CANDIDATE SCREENING

The development of candidate APS concepts began with the survey, identification, and preliminary screening of potential combinations of design concepts for providing propellant orientation, thermodynamic control, and expulsion. Several alternatives for providing each of these functions are listed in Table 4-1. As can be seen, the number of possible combinations is too great to allow a detailed study of each. For this reason, a preliminary screening was made to select the combinations which have the greatest potential and which represent the widest variety of approaches to APS tankage and feed system design. The results of the screening are shown in the table.

At the same time, several different approaches to thermodynamic control and expulsion were chosen to be used with the baseline capillary technique for propellant orientation so that the effect of each single variation on weight, cost, and reliability could be isolated.

Table 4-1. Screening Matrix for Dedicated APS Options

Element	Combination No. 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32
Propellant Orientation																																
None	X	X	X	X	X	X																										
Linear Acceleration							X	X	X	X	X	X	X	X	X	X	X	X														
Capillary																			X	X	X	X	X	X	X	X						
Bellows or Bladder																											X	X	X	X	X	X
Thermodynamic Control																																
No Venting	X						X	X	X										O								O					
Direct Venting				O						X	X	X								O								O				
Internal Heat Exchanger					X								X	X	X						X	X	X						O			
Tank Wall-Mounted Tubes						X										X	X	X						X	X	X				X	X	X
Expulsion																																
Pump	X				X	X	X			X			X			X					X			X						O		
Pressure - Stored Gas		O						X			X			X			X					X			X						X	
Pressure - Vapor Pump			O						X			X			X			X					⊗			⊗						⊗
Incompatible combination		✓	✓	✓															✓	✓							✓	✓	✓	✓		
Low potential combination																							✓			✓						✓
High potential combination	✓				✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓			✓			✓	✓						✓	
Selected for Tank II study (Candidate No.)	2															7					1	B		6	36 4						56 7	

Code:
X, O, ⊗ : Considered combinations
O : Incompatible element of combination
⊗ : Potentially incompatible element
B : Initial Baseline

The combinations of Table 4-1 which appear viable are as follows:

1. The combination of no zero-g liquid propellant orientation, no venting, and pump feedout has design simplicity but only minimum control of thruster inlet propellant conditions. Low APS weight and cost would result if the thruster and vehicle control system designs could be made compatible with the wide range of propellant densities and thrust levels. For this reason, this combination was selected for further study.

5,6. These two combinations, which use either an internal or an external heat exchanger without propellant orientation, are similar to Combination 1.

7-18. All of the thermodynamic control and expulsion concepts identified are compatible with linear translation for propellant orientation. However, for the reference Tug mission, approximately 25 percent of the APS propellant is consumed for short attitude control pulses which may be required after long coast periods. The attitude control propellant may be loaded into a separate tank provided with a metallic bellows for propellant orientation. The remaining propellant will be self-oriented during Tug linear translation maneuvers. The combination selected for further study utilizes tank wall-mounted tubes for propellant thermodynamic control and a pump for propellant expulsion.

21,22. The combination of capillary devices for propellant orientation and an internal heat exchanger for propellant thermodynamic control appears to have high potential when considered with either pump or pressure expulsion and thus both were selected for further evaluation. This approach uses the expansion of liquid withdrawn from the storage tank to provide a low-temperature heat sink for cooling the remaining bulk propellant. The internal heat exchanger provides the heat transfer area between the stored propellant and the colder vent flow.

24,25. Tank wall-mounted tubes serve the same function as an internal heat exchanger in absorbing the heat leak through insulation and structural supports. For the former, the tank wall itself provides increased heat transfer area, whereas for the latter, a bulk mixer or extensive fin arrangement may be required for low-g propellant temperature destratification. Both combinations were selected for further evaluation.

31. The only propellant thermodynamic control concept found to be compatible with the use of solid barriers such as metallic bellows or nonmetallic bladders was tank wall-mounted tubes. The other approaches either physically interferred with the barrier or required further separation of the liquid and vapor

on the liquid side of the barrier. Pressure expulsion is required to assure displacement of the bladder or bellows during feedout. Two combinations were selected for further study, one using a bladder for expulsion of all the APS propellant and one using a bellows for orientation of only the attitude control propellant. The latter is used in conjunction with linear acceleration or propellant settling and pump feed for the APS propellant consumed during Tug linear translation.

4.3 CANDIDATE SYSTEM CHARACTERISTICS

The selected combinations comprise eight candidate systems, including an updated version of the initial baseline system. These are discussed next, with emphasis on the concepts which were eventually selected: Candidates 3 and 6.

UPDATED BASELINE - PRESSURE FEED, INTERNAL HEAT EXCHANGER

The baseline APS tankage and feed system uses capillary devices for propellant orientation, an internal heat exchanger for propellant thermodynamic control, and stored helium gas for propellant expulsion. A mechanical flow diagram is presented in Figure 4-1. The numbers in circles are component identification numbers for correlation with the weight tables.

The variations from the initial baseline are as follows:

1. The results of a thermodynamic control analysis showed the need for destratification fins in each propellant tank to assure reasonably uniform cooling of the propellant in zero gravity.

2. Triply-redundant regulators and isolation solenoid valves have been added to the helium pressurization system to improve first mission reliability.

3. The functions of emergency relief and ground fill venting have been combined into one vent/relief valve for each propellant tank. This change also improves reliability by eliminating the parallel leakage path previously suffered by the use of two separate valves.

4. The specific impulse and thruster flow rate values have been revised to reflect the results of parametric studies.

5. The LOX/LH_2 heat exchanger used to chill the LOX bleed flow was relocated to represent more graphically the original intent for LOX feedline tracing by the bleed flow of chilled LOX.

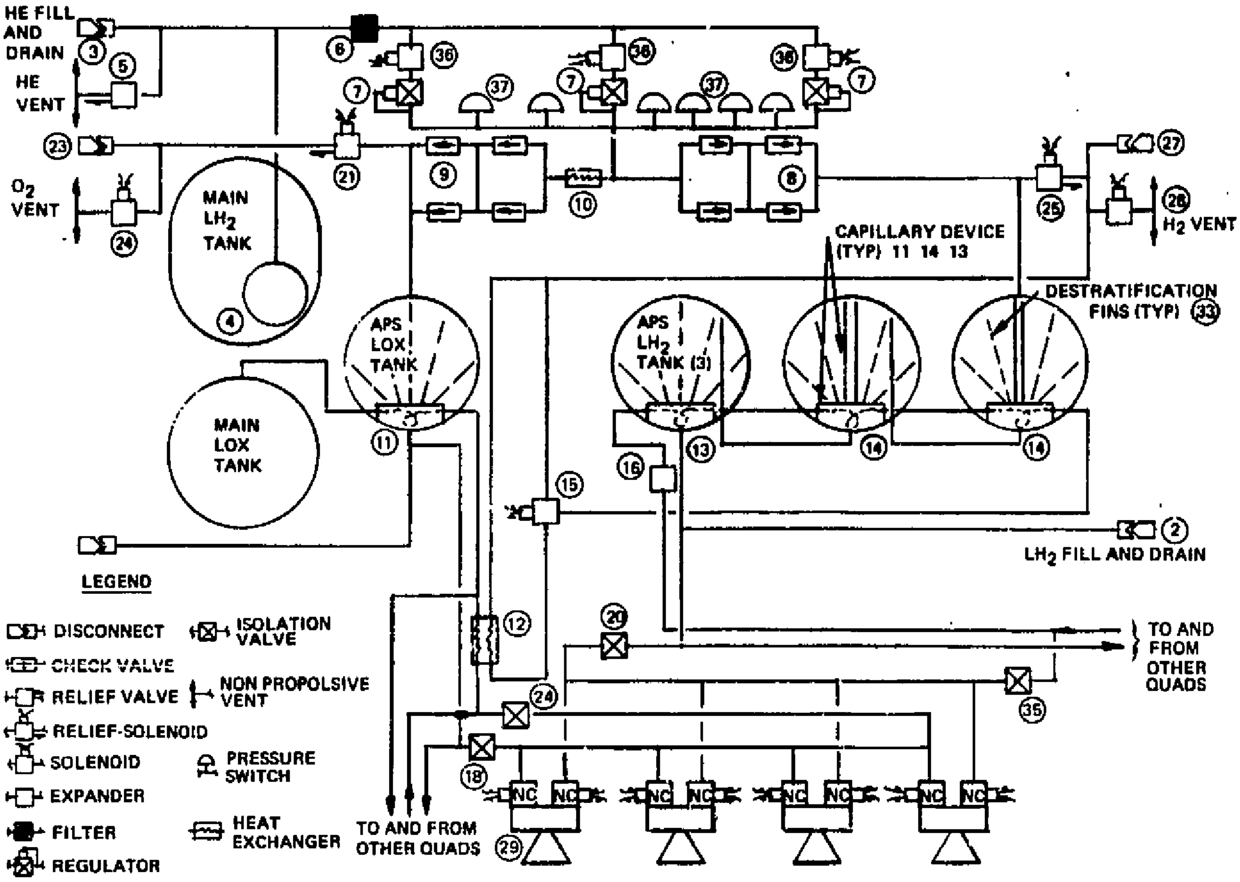


Figure 4-1. Mechanical Flow Diagram for Updated Baseline

CANDIDATE 1 - PUMP FEED, INTERNAL HEAT EXCHANGER

This concept is similar to the updated baseline with the exception that expulsion is provided by an electrical motor-driven pump rather than by pressurizing the storage tanks. A mechanical flow diagram is presented in Figure 4-2. A small accumulator is provided downstream of both the LOX and LH_2 pumps to provide a pressure reservoir during pump start transients and to avoid frequent cycling of the pumps during short APS burns. Approximately 80 percent of the accumulator volume is charged with helium gas at 172 N/cm^2 (250 psia) which expands to 138 N/cm^2 (200 psia) upon depletion of the liquid propellant retained at the accumulator outlet by a capillary screened basket. At the signal of a pressure switch, the pump refills the accumulator until the trapped helium is repressurized to 172 N/cm^2 (250 psia).

The potential advantages of Candidate 1 over the baseline tankage and feed system are reduced weight due to elimination of the stored pressurization gas system and reduction in the propellant storage tank wall thickness. The LOX bleed flow can be returned to the APS LOX tank rather than being recovered in the Tug main LOX tank as is done for the baseline. This allows a commensurate reduction in LOX tank volume and weight. Potential disadvantages are increased complexity and power and reduced reliability associated with the accumulator, pump, and motor.

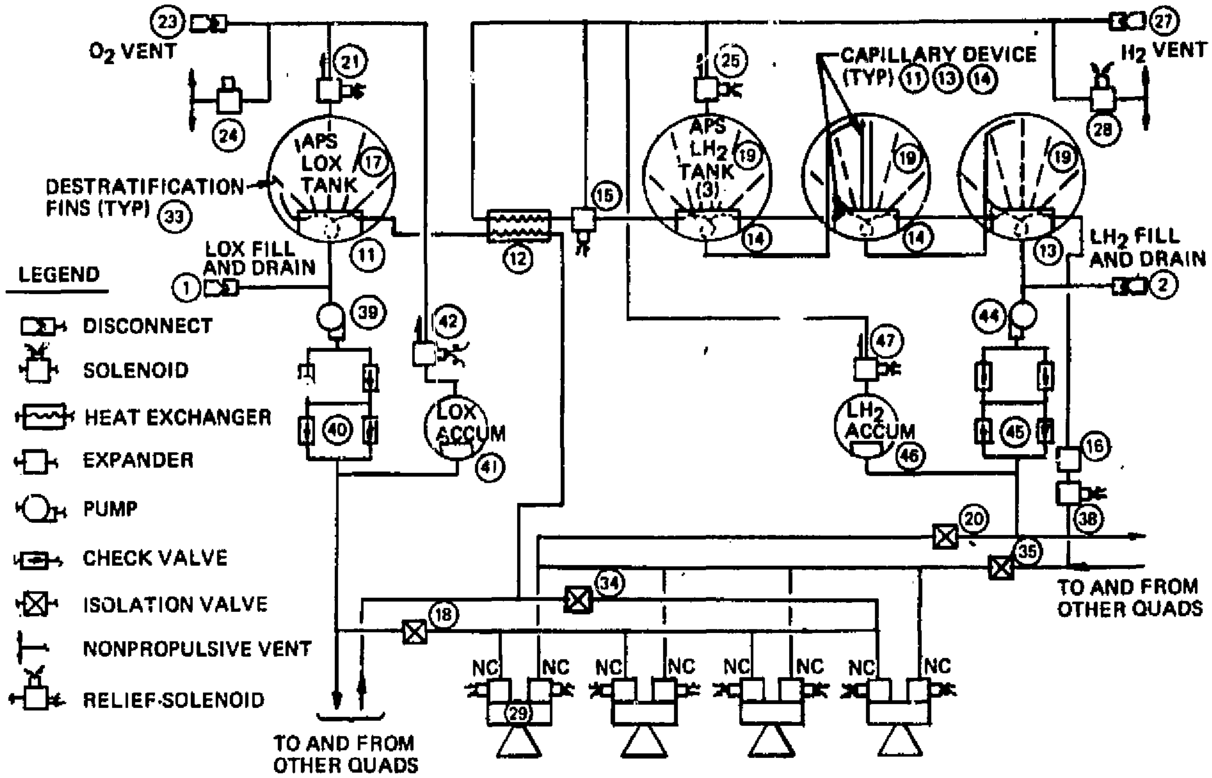


Figure 4-2. Mechanical Flow Diagram for Candidate 1 - Pump Feed, Internal Heat Exchanger

CANDIDATE 2 - MIXED PHASE PROPELLANT

This concept is the simplest conceivable tankage system design, having no provisions for liquid propellant orientation and minimum provisions for propellant thermodynamic control.

Thermodynamic control is provided without venting through the use of internal conductive fins for temperature destratification. The influence of heat leakage on tank pressure rise is partially offset by thruster propellant consumption. Although feasible, this method of thermodynamic control allows a wide variation in propellant temperatures, depending on propellant consumption and heating environments.

The electrically driven positive displacement pumps are designed to pump liquid or gas. The power requirement for pumping liquid hydrogen to supply four thrusters simultaneously was found to be 1600 watts, while the power for the same mass flow rate with vapor was found to be 34,000 watts. Although the storage tank outlet could be located so that liquid flow would exist most of the time, the electrical power supply system would have to be designed for the worst case.

Because of the severe power requirements and poor control of propellant inlet conditions, Candidate 2 was not considered further.

CANDIDATE 3 - PRESSURE FEED, TANK WALL-COOLED

Mechanical and process flow diagrams for Candidate 3 are shown in Figures 4-3 and 4-4. The system weight and cost, stage weight, and mission timeline are itemized in Tables 4-2, 4-3, and 4-4. The cost data are discussed later in this section and in Section 6. This candidate is basically the same as the updated baseline with the exception of the propellant thermodynamic control. Both use the same capillary devices and helium pressure expulsion, as well as a hydrogen vent system which expands liquid bleed from the feed line and then returns to cool the line. Unlike the baseline, however, which uses a heat exchanger internal to the LH_2 storage tank, the hydrogen bleed flow for Candidate 3 is routed through tank wall-mounted tubes to absorb the storage tank heat load. These cooling coils can be concentrated at structural support and insulation penetrations to minimize the temperature stratification effects of heat shorts. Thus, the internal destratification forms of the baseline are not required. The hydrogen storage temperature is controlled by opening and closing the solenoid valve (No. 38) upstream of the expander (No. 16).

After cooling the LH_2 tanks, the hydrogen bleed is electrically heated to a temperature above the freezing point of LOX and routed over the LOX feed lines and storage tank. This approach saves the weight of the LOX bleed and its associated storage volume, but has the potential safety hazard of close

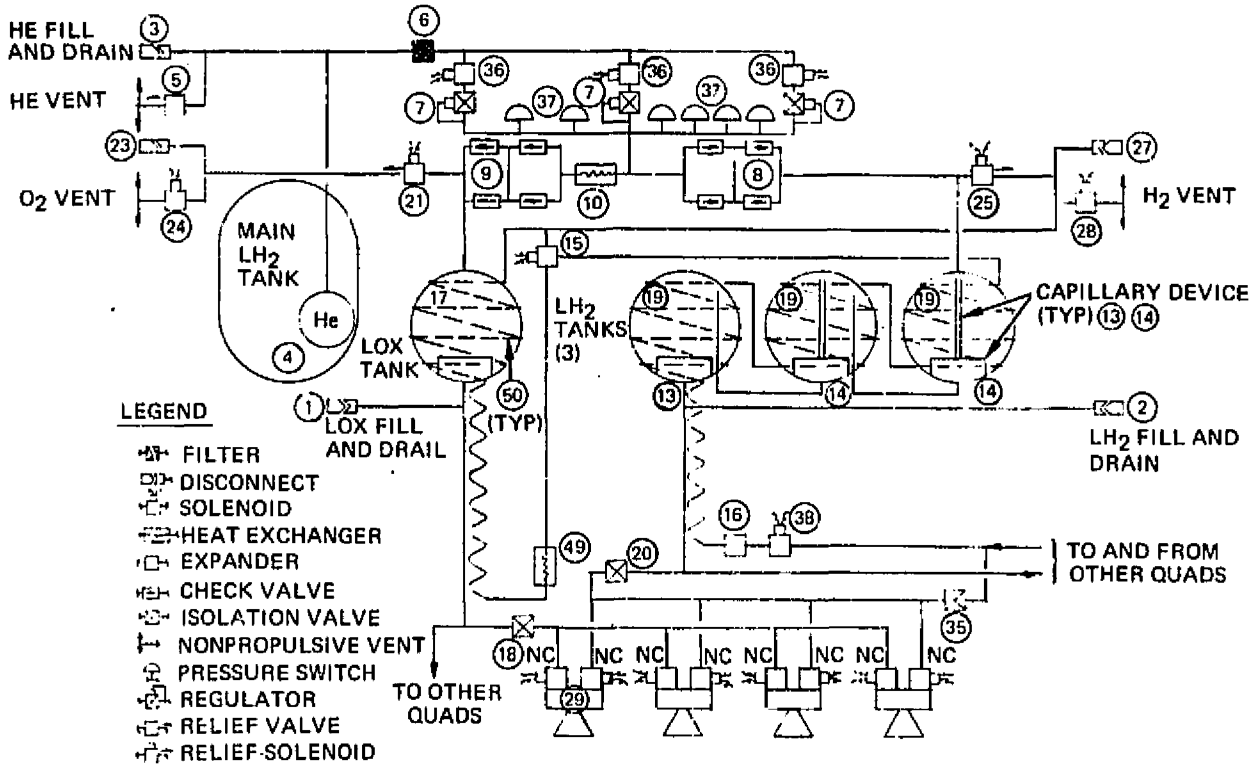
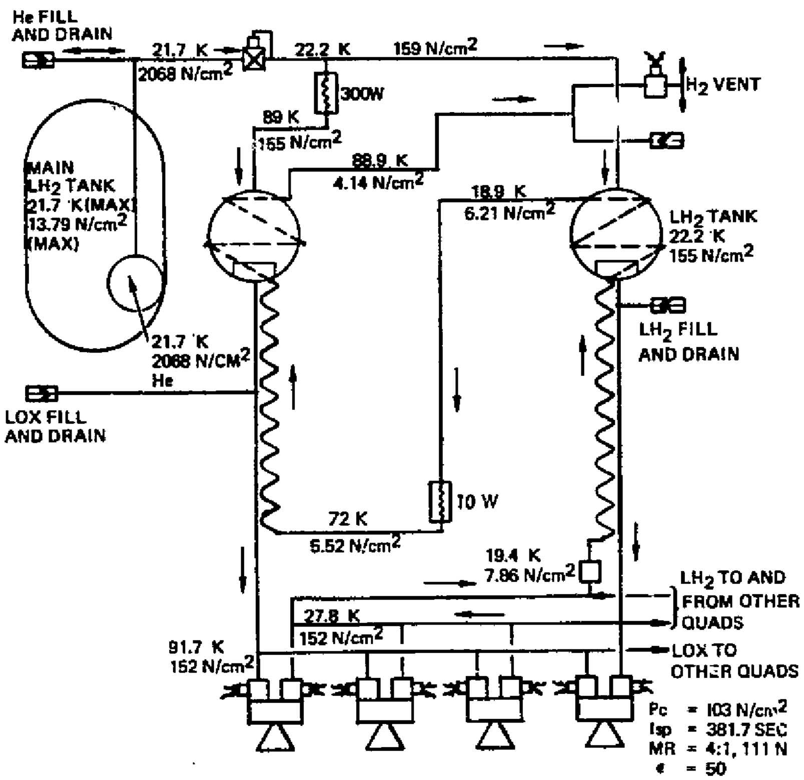


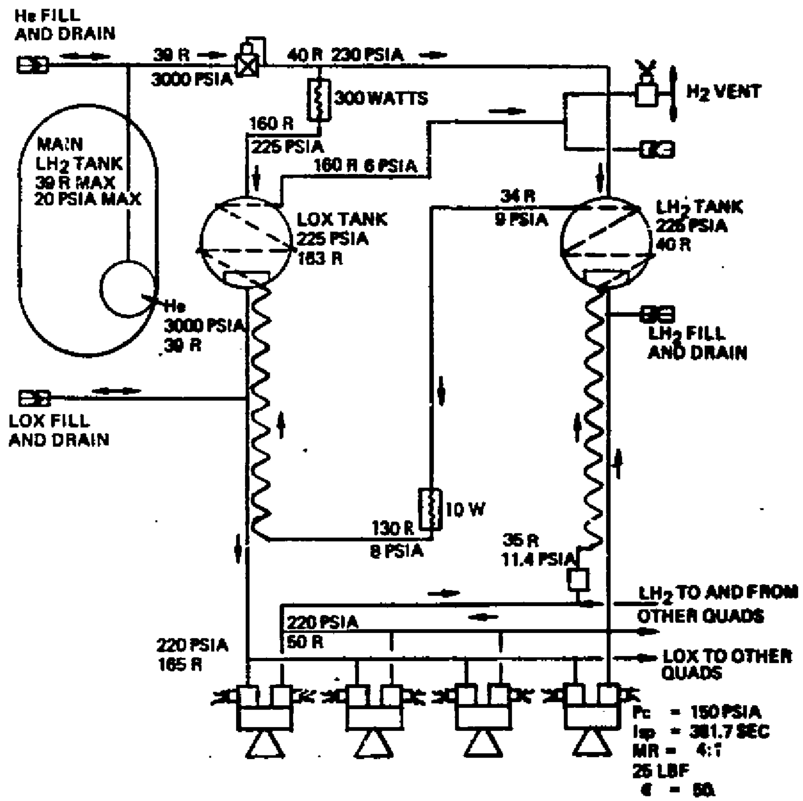
Figure 4-3. Mechanical Flow Diagram for Candidate 3 - Pressure Feed, Tank Wall Cooled

FLOW CHARACTERISTICS	MISSION A	
	LOX	LH_2
FLOW/ENGINE (KG/SEC)	.0239	.00594
BLEED FLOW (KG/HR)	0	.0603
HE FLOW AT 22.2 K (KG/SEC)		.0113
AT 88.9 K (KG/SEC)	.00072	



Metric Units

FLOW CHARACTERISTICS	MISSION A	
	LOX	LH_2
FLOW/ENGINE (LB/SEC)	.0526	.0131
BLEED FLOW (LB/HR)	0	.133
HE FLOW AT 40 R (LB/SEC)		.0250
AT 160 R (LB/SEC)	.00159	



English Units

Figure 4-4. Candidate 3 Process Diagram

Table 4-2. Weight and Cost Summary for Candidate 3

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG	DDT&E $1000.	1ST UNIT $1000.	REFURB $1000.
FILL & DRAIN SYSTEM		(2)			(2.0)	(0.9)	(29.0)	(5.1)	(0.0)
LOX FILL & DRAIN DISC	1	1	1.0	0.5	1.0	0.5	29.0	2.5	0.0
LH2 FILL & DRAIN DISC	2	1	1.0	0.5	1.0	0.5	0.0	2.5	0.0
PRESSURIZATION SYSTEM		(25)			(102.9)	(46.7)	(302.6)	(121.6)	(503.9)
HELIUM FILL DISCONNECT	3	1	1.5	0.7	1.5	0.7	7.4	2.1	0.0
HELIUM TANK	4	1	81.0	36.7	81.0	36.7	0.0	7.9	0.0
HELIUM TANK RELIEF VLV	5	1	0.5	0.2	0.5	0.2	80.0	8.5	0.0
HELIUM FILTER	6	1	0.5	0.2	0.5	0.2	0.0	0.6	3.2
HELIUM REGULATOR	7	3	1.5	0.7	4.5	2.0	100.0	32.0	330.6
LOX SYSTEM HE CHECK VLV	9	4	0.3	0.1	1.2	0.5	0.0	15.2	79.6
LH2 SYSTEM HE CHECK VLV	8	4	0.3	0.1	1.2	0.5	33.5	15.2	79.6
LOX SYSTEM HE HEATER	10	1	0.5	0.2	0.5	0.2	25.6	0.8	0.0
HE REGULATOR ISO VALVE	36	3	2.0	0.9	6.0	2.7	23.4	25.1	8.7
PRESSURE SWITCH	37	6	1.0	0.5	6.0	2.7	32.7	13.3	2.2
PROPELLANT CONTROL SYSTEM		(12)			(33.6)	(15.2)	(238.0)	(25.3)	(17.4)
LOX TANK CAPILLARY DEV	11	1	3.6	1.6	3.6	1.6	94.5	0.0	0.0
LH2 TANK SUMP CAP DEVICE	13	1	4.2	1.9	4.2	1.9	94.5	0.0	0.0
LH2 TK UPSTRM CAP DEV	14	2	2.4	1.1	4.8	2.2	0.0	0.0	0.0
LH2 BLEED RETURN SOL VLV	15	1	1.5	0.7	1.5	0.7	36.0	8.7	8.7
LH2 BLEED EXPANDER VLV	16	1	1.5	0.7	1.5	0.7	13.0	0.8	0.0
LH2 BLEED SHUTOFF VALVE	38	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2	0.0	0.8	0.0
TK EXT COOLING COIL	50	4	4.0	1.8	16.0	7.3	0.0	6.3	0.0
PROPELLANT FEED SYSTEM		(16)			(159.4)	(72.3)	(290.3)	(361.8)	(127.2)
LOX TANK	17	1	16.4	7.4	16.4	7.4	92.0	31.6	0.0
LOX TANK INSULATION	0	1	2.6	1.2	2.6	1.2	62.7	23.7	23.7
LOX FEED ISO VALVE	18	4	3.0	1.4	12.0	5.4	0.0	34.8	5.7
LH2 TANK	19	3	35.5	16.1	106.5	48.3	75.0	142.2	0.0
LH2 TANK INSULATION	0	3	3.3	1.5	9.9	4.5	60.5	94.8	94.8
LH2 FEED ISOLATION VALVE	20	4	3.0	1.4	12.0	5.4	0.0	34.8	0.0
VENT SYSTEM		(6)			(9.5)	(4.3)	(173.5)	(85.6)	(80.6)
LOX TANK VENT/RELIEF V	21	1	2.0	0.9	2.0	0.9	123.5	31.6	31.6
GOX VENT DISCONNECT	23	1	1.0	0.5	1.0	0.5	13.9	2.5	0.0
GOX IN-FLT VENT SOL VLV	24	1	1.5	0.7	1.5	0.7	36.0	8.7	8.7
LH2 TANK VENT/RELIEF V	25	1	2.0	0.9	2.0	0.9	0.0	31.6	31.6
GH2 VENT DISCONNECT	27	1	1.5	0.7	1.5	0.7	0.0	2.5	0.0
GH2 IN-FLT VENT SOL VLV	28	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
THRUSTER QUAD (50 LB)		(68)			(71.6)	(32.5)	(5810.2)	(1238.7)	(183.3)
THRUST CHAMBER & NOZZLE	29	16	2.6	1.2	41.6	18.9	5810.2	733.1	0.0
THRUSTER VALVES	0	32	0.6	0.3	19.2	8.7	0.0	252.8	173.9
IGNITER	0	16	0.2	0.1	3.2	1.5	0.0	75.8	9.5
EXCITER	0	4	1.9	0.9	7.6	3.4	0.0	177.0	0.0
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)	(19.6)	(54.4)	(6.3)
COMPONENT TOTAL		172			396.2	179.7	6863.3	1892.4	918.7
LINES					36.0	16.3			
INSULATION					16.2	7.3			
COMPONENT MOUNTINGS					19.8	9.0			
DRY SYSTEM					468.2	212.4			

proximity of hydrogen and oxygen leakage. This condition is considered acceptable, however, since leakage is into the well-vented intertank structure. Control of the oxygen storage temperature is provided by a hydrogen bleed flow bypass valve (No. 15).

CANDIDATE 4 - PRESSURE FEED, MODULAR

Candidate 4 is the same as Candidate 3 with the exception that the APS storage vessels and components are arranged into four separate modules, one for each thruster quad. The mechanical flow diagram is presented in Figure 4-5. The modules are identical and have interconnecting manifolds to facilitate common interfaces with the Shuttle cargo bay for propellant fill, drain, and vent. The major advantage of this approach is the simplified development, qualification, and acceptance testing of a single compact unit. Tug maintenance operations also can be simplified through the fault identification, removal, and replacement of a complete modular system.

Table 4-3. Vehicle Weight Summary for Candidate 3

DESCRIPTION	WEIGHT	
	LB	KG
STRUCTURE	(2089.)	(949.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1590.)	(721.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	468.	212.
DRY WEIGHT	5085.	2307.
CONTINGENCY (13%)	661.	300.
TOTAL DRY SYSTEM	5746.	2606.
NONUSABLE FLUIDS	(614.)	(279.)
APS TRAPPED PROPELLANT	8.	4.
APS PRESSURANT	86.	38.
MPS TRAPPED PROPELLANT	373.	168.
MPS PRESSURANT	154.	70.
FLIGHT RESERVES	(368.)	(167.)
APS RESERVE (10%)	48.	22.
MPS RESERVE	320.	145.
BURNOUT WEIGHT	6730.	3053.
EXPENDED FLUIDS	(51192.)	(23220.)
APS USABLE PROPELLANT	477.	216.
APS LH2 BLEED OVERBOARD	26.	12.
MPS USABLE PROPELLANT	50405.	22863.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	134.	61.
PAYLOAD	(4475.)	(2030.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

Module manifold isolation valves (No. 51 and 53) have been included to take advantage of the reduced heat loads made possible by the shorter feedlines. This advantage is somewhat offset by a greater number of propellant tanks. The total hydrogen bleed required can be reduced by 10 kg (22 lb) total to 8.6 kg (19 lb). The reliability of Candidate 4 is comparatively poor, since each of the modules must function for complete mission success.

CANDIDATE 5 - BLADDER FEED

The mechanical diagram for Candidate 5 is presented in Figure 4-6. This candidate is the same as Candidate 3 except that liquid propellant orientation is provided by a bladder. Although several promising bladder materials were identified, no test-proven material or laminate was found for cryogenic service. Thus, the use of a cryogenic bladder must be penalized by requiring supporting research and technology (SR&T) cost and risk.

Table 4-4. Mission Timeline for Propulsion Events, Candidate 3

MISSION TIME HR	EVENT NO	DURATION HR	DURATION MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0	0	LIFTOFF										
	1			SHUTTLE BURNOUT										
1.63	2	1	38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0	7	DEPLOY TUG	APS				3	95148	2495	29303	15672	464568
6.38	4	4	38	COAST NO 1							2677	28276	15669	464247
6.43	5	0	3	PHASING ORBIT INSERTION	MAIN	163			2	4105	8389	28272	15668	464237
7.74	6	1	19	COAST NO 2							2471	27267	15541	453436
7.99	7	0	15	TRANSFER ORBIT INSERTION	MAIN	2383			2	4363	8409	27265	15541	453385
8.02	8	0	2	MIDCOURSE CORRECT (DV 1)	PIM		14	220814	4	5443	5951	16126	14128	329023
13.13	9	5	6	COAST NO 3							2011	16064	14120	328296
13.26	10	0	8	MISSION ORBIT INSERTION	MAIN	1784			4	5440	6044	16059	14120	328238
13.33	11	0	2	ORIENT PAYLOAD	APS				3	36417	1419	10833	13457	262637
13.30	12	0	0	DEPLOY PAYLOAD 1 (677 KG)								10822	13456	262498
13.41	13	0	7	PAYLOAD SURVEILLANCE	APS				9	102240	1173	10146	11688	212164
37.16	14	23	45	COAST NO 4							2629	10119	11684	211832
37.19	15	0	2	PHASING ORBIT INSERT (DV 2)	MAIN		83	831817	6	4899	3848	10097	11681	211575
89.34	16	52	9	COAST NO 5							4410	9904	11657	209254
89.37	17	0	2	MISSION ORBIT INSERT (DV 3)	MAIN		83	812107	7	4860	3797	9857	11651	208699
89.41	18	0	2	ORIENT PAYLOAD	APS				3	32504	1121	9669	11627	206436
89.41	19	0	0	DEPLOY PAYLOAD 2 (677 KG)								9659	11626	206296
89.51	20	0	6	PAYLOAD SURVEILLANCE	APS				9	90520	846	8983	9858	149990
95.34	21	5	50	COAST NO 6							1253	8958	9855	149726
95.37	22	0	2	PHASING ORBIT INSERT (DV 4)	MAIN		83	737583	7	4649	2740	8953	9854	149667
147.52	23	52	9	COAST NO 7							4771	8781	9832	147811
147.55	24	0	2	MISSION ORBIT INSERT (DV 5)	MAIN		83	719569	7	4608	2698	8734	9827	147302
147.58	25	0	2	ORIENT PAYLOAD	APS				3	28800	800	8567	9805	145495
147.58	26	0	0	DEPLOY PAYLOAD 3 (677 KG)								8559	9804	145399
147.67	27	0	5	PAYLOAD SURVEILLANCE	APS				9	79431	486	7882	8036	80842
150.19	28	2	31	COAST NO 8							698	7861	8034	80622
150.27	29	0	5	TRANSFER ORBIT INSERTION	MAIN	1781			7	4386	1559	7859	8033	80597
155.44	30	5	10	COAST NO 9							883	5303	7709	54397
155.47	31	0	2	MIDCOURSE CORRECT (DV 6)	THIM		12	64435	9	3788	1039	5298	7709	54338
155.54	32	0	4	PHASING ORBIT INSERTION	MAIN	1690			9	3782	1070	5275	7706	54104
157.63	33	2	5	COAST NO 10							485	3630	7497	37233
157.68	34	0	3	CIRCULARIZE FOR RENDEZVOUS	MAIN	758			12	3232	752	3628	7497	37213
163.56	35	5	57	COAST NO 11							967	3064	7425	31429
163.59	36	0	2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	25694	209	3059	7425	31373
163.59	37	0	0	SHUTTLE DEORBIT								3052	7424	31373
164.29	38	0	42	TOUCHDOWN										
TOTALS						8559	358	3386323	123	728974	81510			

MISSION TIME HR	EVENT NO	DURATION HR	DURATION MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0	0	LIFTOFF										
	1			SHUTTLE BURNOUT										
1.63	2	1	38	CIRCULARIZE AT 160 NM								65009		
1.75	3	0	7	DEPLOY TUG	APS				10	21390	561	62397	11560	342652
6.38	4	4	38	COAST NO 1							602	62339	11557	342445
6.43	5	0	3	PHASING ORBIT INSERTION	MAIN	534			2	4105	1886	62329	11557	342438
7.74	6	1	19	COAST NO 2							555	60113	11463	334419
7.99	7	0	15	TRANSFER ORBIT INSERTION	MAIN	7817			2	4363	1890	60109	11462	334434
8.02	8	0	2	MIDCOURSE CORRECT (DV 1)	PIM		45	49641	4	5443	1338	35551	10421	242678
13.13	9	5	6	COAST NO 3							452	35415	10415	242142
13.26	10	0	8	MISSION ORBIT INSERTION	MAIN	5854			4	5440	1359	35404	10414	242099
13.30	11	0	2	ORIENT PAYLOAD	APS				10	8187	319	23882	9926	193713
13.30	12	0	0	DEPLOY PAYLOAD 1 (1492 LB)								23860	9925	193611
13.41	13	0	7	PAYLOAD SURVEILLANCE	APS				30	22984	264	22368	8621	156487
37.16	14	23	45	COAST NO 4							591	22307	8618	156241
37.19	15	0	2	PHASING ORBIT INSERT (DV 2)	MAIN		273	187000	6	4899	865	22259	8616	156052
89.34	16	52	9	COAST NO 5							991	21834	8598	154340
89.37	17	0	2	MISSION ORBIT INSERT (DV 3)	MAIN		273	182569	7	4860	854	21731	8594	153923
89.41	18	0	2	ORIENT PAYLOAD	APS				10	7307	252	21315	8576	152239
89.41	19	0	0	DEPLOY PAYLOAD 2 (1492 LB)								21296	8575	152158
89.51	20	0	6	PAYLOAD SURVEILLANCE	APS				30	20350	190	19804	7271	110629
95.34	21	5	50	COAST NO 6							282	19759	7269	110433
95.37	22	0	2	PHASING ORBIT INSERT (DV 4)	MAIN		273	165815	7	4649	616	19738	7268	110390
147.52	23	52	9	COAST NO 7							1072	19363	7252	109021
147.55	24	0	2	MISSION ORBIT INSERT (DV 5)	MAIN		273	161766	7	4608	606	19256	7248	108646
147.58	25	0	2	ORIENT PAYLOAD	APS				10	6475	180	18887	7232	107306
147.58	26	0	0	DEPLOY PAYLOAD 3 (1492 LB)								18870	7231	107242
147.67	27	0	5	PAYLOAD SURVEILLANCE	APS				30	17857	109	17378	5927	59627
150.19	28	2	31	COAST NO 8							157	17339	5925	59465
150.27	29	0	5	TRANSFER ORBIT INSERTION	MAIN	5843			7	4386	351	17325	5925	59446
155.44	30	5	10	COAST NO 9							199	11691	5686	40114
155.47	31	0	2	MIDCOURSE CORRECT (DV 6)	THIM		40	14486	9	3788	234	11683	5686	40078
155.54	32	0	4	PHASING ORBIT INSERTION	MAIN	5545			9	3782	241	11633	5683	39936
157.63	33	2	5	COAST NO 10							109	8004	5530	27462
157.69	34	0	3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2408			12	3232	169	7993	5529	27447
163.56	35	5	53	COAST NO 11							217	6756	5477	23181
163.59	36	0	2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	5776	47	6744	5476	23140
163.59	37	0	0	SHUTTLE DEORBIT								6729	5476	23140
164.29	38	0	42	TOUCHDOWN										
TOTALS						28081	1177	761276	231	163880	18324			

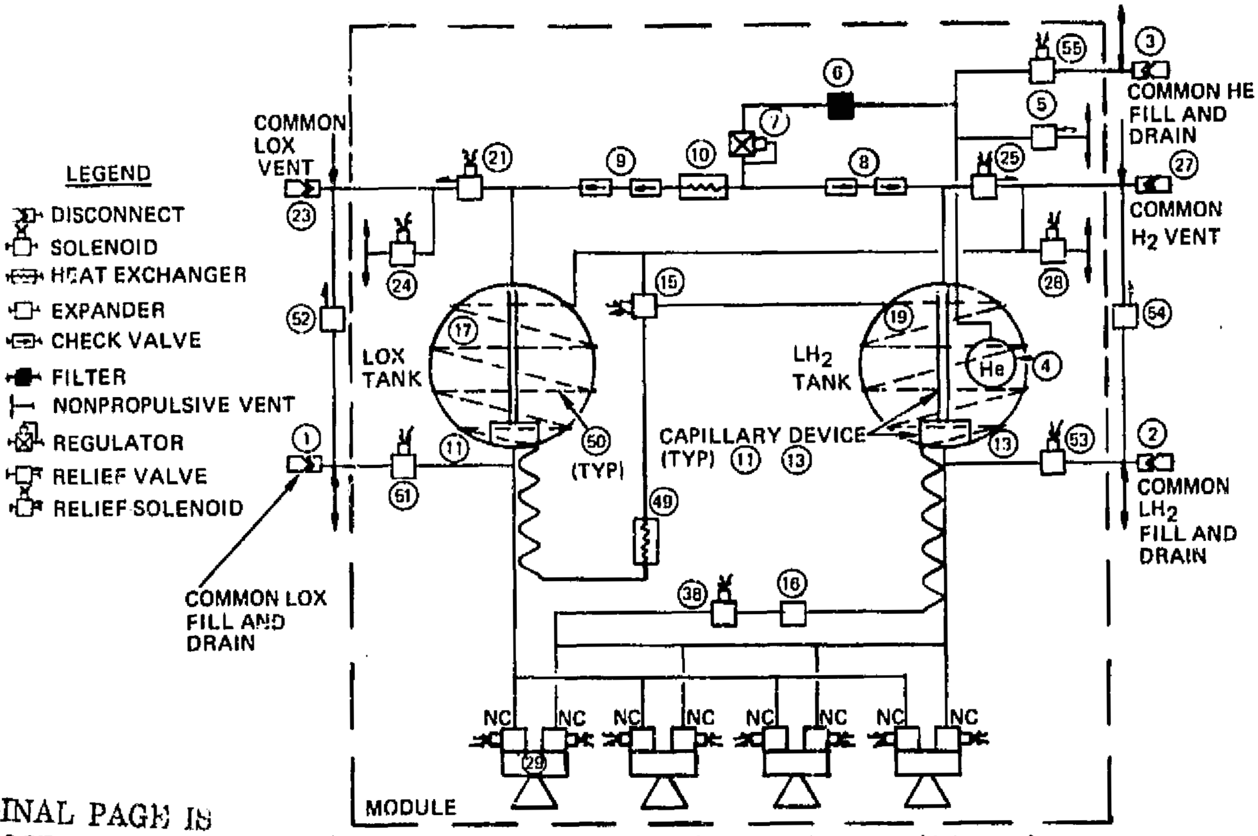


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Figure 4-5. Mechanical Flow Diagram for Candidate 4 - Pressure Feed, Modular

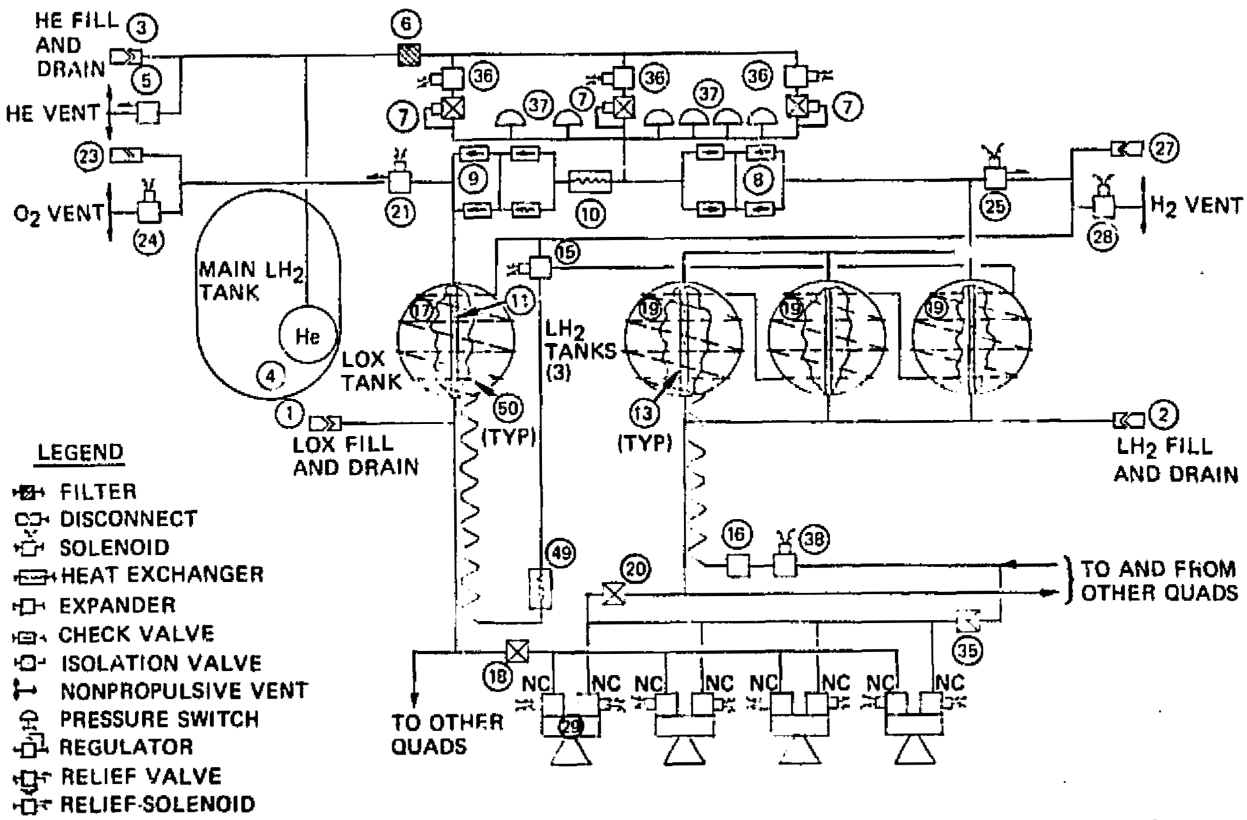

Figure 4-6. Mechanical Flow Diagram for Candidate 5 - Bladder Feed

CANDIDATE 6 - PUMP FEED, TANK WALL-COOLED

The mechanical and process flow diagrams for Candidate 6 are shown in Figures 4-7 and 4-8. The system weight and cost, and stage weights are listed in Tables 4-5 and 4-6. The cost data are discussed later in this section and in Section 6. This candidate is the same as Candidate 1 with two exceptions. The primary difference is that both the LOX and LH_2 tanks for Candidate 6 are cooled by the bleed flow of hydrogen through externally mounted tank wall tubes. The operation is the same as that previously described for Candidate 3. Although there is no LOX bleed for Candidate 6, the major portion of the weight penalty for LOX bleed could have been saved by the APS tank recovery of LOX bleed made possible by the pump pressure differential. Thus it can be seen that the weight advantage of tank wall cooling over internal heat exchanger cooling is not as dramatic for a pump-feed system as it is for a pressure-feed system.

The second difference between this candidate and Candidate 1 is the method of pressurizing and venting the accumulator. The Candidate 1 accumulator is charged with helium on the ground by back pressurizing through the accumulator vent and relief valve. Reliability is impaired by the additional leakage path introduced by these valves. An improved accumulator helium charge technique is used for Candidate 6. The probability of helium leakage

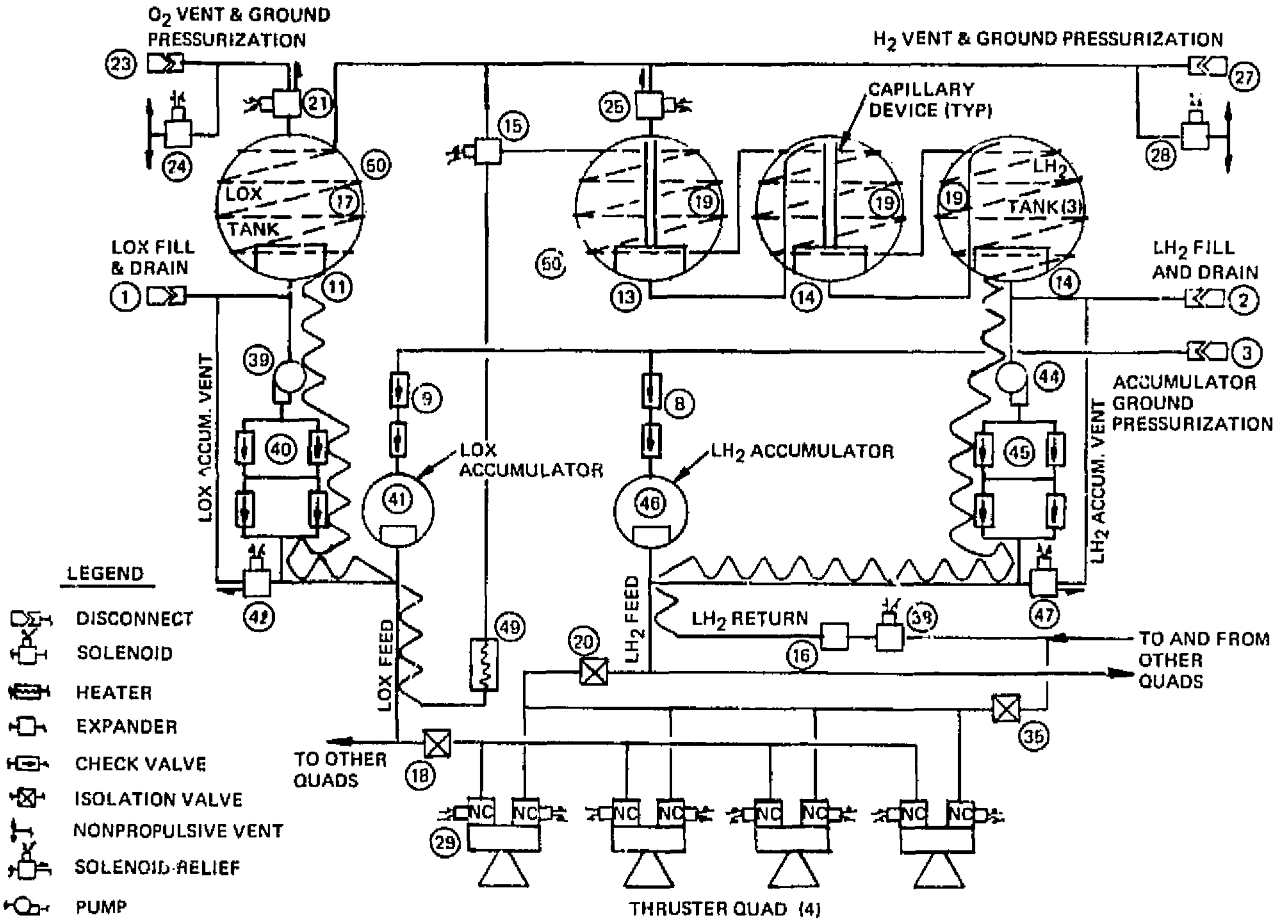

Figure 4-7. Mechanical Flow Diagram for Candidate 6 - Pump Feed, Tank Wall Cooled

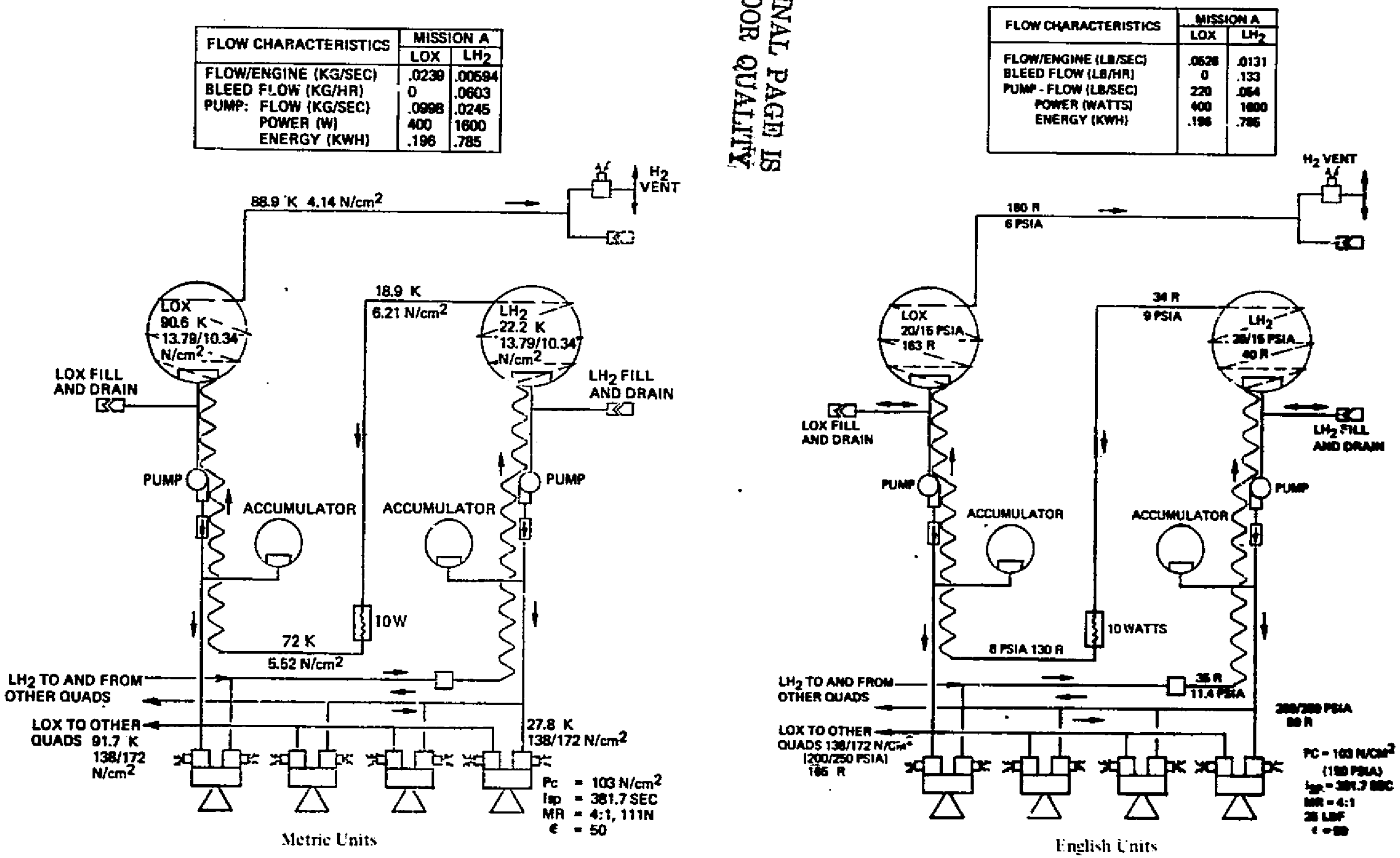

Figure 4-8. Candidate 6 Process Diagram

Table 4-5. Weight and Cost Summary for Candidate 6

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG	DDT&E $1000.	1ST UNIT $1000.	REFURB $1000.
FILL & DRAIN SYSTEM		(2)			(2.0)	(0.9)	(29.0)	(5.1)	(0.0)
LOX FILL & DRAIN DISC	1	1	1.0	0.5	1.0	0.5	29.0	2.5	0.0
LH2 FILL & DRAIN DISC	2	1	1.0	0.5	1.0	0.5	0.0	2.5	0.0
PRESSURIZATION SYSTEM		(5)			(2.7)	(1.2)	(40.8)	(17.2)	(7.6)
HELIUM FILL DISCONNECT	3	1	1.5	0.7	1.5	0.7	7.4	2.1	0.0
LOX SYSTEM HE CHECK VLV	9	2	0.3	0.1	0.6	0.3	0.0	7.6	3.8
LH2 SYSTEM HE CHECK VLV	8	2	0.3	0.1	0.6	0.3	33.5	7.6	3.8
PROPELLANT CONTROL SYSTEM		(12)			(33.6)	(15.2)	(264.5)	(25.3)	(17.4)
LOX TANK CAPILLARY DEV	11	1	3.6	1.6	3.6	1.6	94.5	0.0	0.0
LH2 TANK CAPILLARY DEV	13	1	4.2	1.9	4.2	1.9	94.5	0.0	0.0
LH2 TK UPSTRM CAP DEV	14	2	2.4	1.1	4.8	2.2	0.0	0.0	0.0
LH2 BLEED RETURN SOL VLV	15	1	1.5	0.7	1.5	0.7	36.0	5.7	5.7
LH2 BLEED EXPANDER VLV	16	1	1.5	0.7	1.5	0.7	13.0	0.8	0.0
LH2 BLEED SHUTOFF VALVE	38	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2	13.0	0.8	0.0
TK EXT COOLING COIL	50	4	4.0	1.8	16.0	7.3	13.4	6.3	0.0
PROPELLANT FEED SYSTEM		(30)			(105.0)	(47.6)	(1697.0)	(554.9)	(505.3)
LOX TANK	17	1	6.4	2.9	6.4	2.9	85.7	31.5	0.0
LOX TANK INSULATION	0	1	2.3	1.0	2.3	1.0	67.7	23.7	71.1
LOX FEED ISO VALVE	18	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LH2 TANK	19	3	9.3	4.2	27.9	12.7	75.0	142.2	0.0
LH2 TANK INSULATION	0	3	2.3	1.0	6.9	3.1	60.5	94.8	94.8
LH2 FEED ISOLATION VALVE	20	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LOX PUMP	39	1	13.5	6.1	13.5	6.1	351.4	9.5	9.5
LOX PUMP CHECK VALVE	40	4	0.0	0.0	0.0	0.0	27.7	15.2	79.6
LOX ACCUM & CAP DEVICE	41	1	1.0	0.5	1.0	0.5	75.3	23.7	23.7
LOX ACCUM VENT/RELIEF V	42	1	2.0	0.9	2.0	0.9	0.0	31.6	31.6
LH2 PUMP	44	1	17.0	7.7	17.0	7.7	702.9	19.0	19.0
LH2 PUMP CHECK VALVE	45	4	0.0	0.0	0.0	0.0	0.0	15.2	79.6
LH2 ACCUM & CAP DEVICE	46	1	2.0	0.9	2.0	0.9	56.1	47.4	47.4
LH2 ACCUM VENT/RELIEF V	47	1	2.0	0.9	2.0	0.9	0.0	31.6	31.6
VENT SYSTEM		(6)			(9.5)	(4.3)	(173.5)	(85.6)	(80.6)
LOX TANK VENT/RELIEF V	21	1	2.0	0.9	2.0	0.9	123.5	31.6	31.6
GOX VENT DISCONNECT	23	1	1.0	0.5	1.0	0.5	13.9	2.5	0.0
GOX IN-FLT VENT SOL VLV	24	1	1.5	0.7	1.5	0.7	36.0	8.7	8.7
LH2 TANK VENT/RELIEF V	25	1	2.0	0.9	2.0	0.9	0.0	31.6	31.6
GH2 VENT DISCONNECT	27	1	1.5	0.7	1.5	0.7	0.0	2.5	0.0
GH2 IN-FLT VENT SOL VLV	28	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
THRUSTER QUAD (50 AR)		(68)			(71.5)	(32.5)	(5810.2)	(1238.7)	(183.3)
THRUST CHAMBER & NOZZLE	29	16	2.6	1.2	41.6	18.9	5810.2	733.1	0.0
THRUSTER VALVES	0	32	0.6	0.3	19.2	8.7	0.0	252.8	173.8
IGNITER	0	16	0.2	0.1	3.7	1.5	0.0	75.9	9.5
EXCITER	0	4	1.9	0.9	7.6	3.4	0.0	177.0	0.0
INSTRUMENTATION	0	(42)	0.4	0.2	(16.8)	(7.6)	(19.6)	(53.1)	(6.3)
PUMP PWR SUPP - APS CHARGE		(2)			(29.2)	(13.2)	(41.9)	(12.6)	(94.8)
INVERTER	0	1	11.3	5.1	11.3	5.1	41.9	7.9	0.0
BATTERY	0	1	17.9	8.1	17.9	8.1	0.0	4.7	94.8
COMPONENT TOTAL		167			270.4	122.7	7876.5	1992.5	895.2
LINES					28.3	12.8			
INSULATION					8.5	3.9			
COMPONENT MOUNTINGS					13.5	6.1			
DRY SYSTEM					320.7	145.5			

from each accumulator is reduced by the redundant fill check valves (No. 8 and 9). Emergency pressure relief is provided by relief valves No. 42 and 47 which in a failure mode would return liquid upstream of the pump where it could be recovered.

CANDIDATE 7 - ACCELERATION SETTLED, PUMP AND BELLOWS FEED

The mechanical flow diagram for Candidate 7 is presented in Figure 4-9. This concept provides minimum development risk to the problem of liquid propellant orientation. Approximately 75 percent of the APS propellant is stored in tanks without propellant orientation devices where pumped feedout occurs only during the APS translation maneuvers. All such maneuvers are constrained to +X translations so that propellant is positioned at the tank

Table 4-6. Vehicle Weight Summary
for Candidate 6

DESCRIPTION	WEIGHT	
	LB	KG
STRUCTURE	(2089.)	(948.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1443.)	(655.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	321.	146.
DRY WEIGHT	4938.	2240.
CONTINGENCY (13%)	642.	291.
TOTAL DRY SYSTEM	5580.	2531.
NONUSABLE FLUIDS	(534.)	(242.)
APS TRAPPED PROPELLANT	10.	5.
MPS TRAPPED PROPELLANT	370.	168.
MPS PRESSURANT	154.	70.
FLIGHT RESERVES	(367.)	(166.)
APS RESERVE (10%)	47.	21.
MPS RESERVE	320.	145.
BURNOUT WEIGHT	6481.	2940.
EXPENDED FLUIDS	(50777.)	(23032.)
APS USABLE PROPELLANT	473.	215.
APS LH2 BLEED OVERBOARD	27.	12.
MPS USABLE PROPELLANT	49993.	22676.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	134.	61.
PAYLOAD	(5139.)	(2331.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TLG CHARGEABLE INTERFACES	(2623.)	(1191.)
GROSS LIFTOFF WEIGHT	65020.	29493.

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outlet. Liquid propellant feed during the initial settling period is supplied by a helium-pressurized metallic bellows. After the propellant has settled, the LOX and LH_2 pumps are turned on. These pumps provide flow for four thrusters plus enough to replenish the accumulator for the next settling maneuver. The accumulators will provide all of the propellant for attitude control.

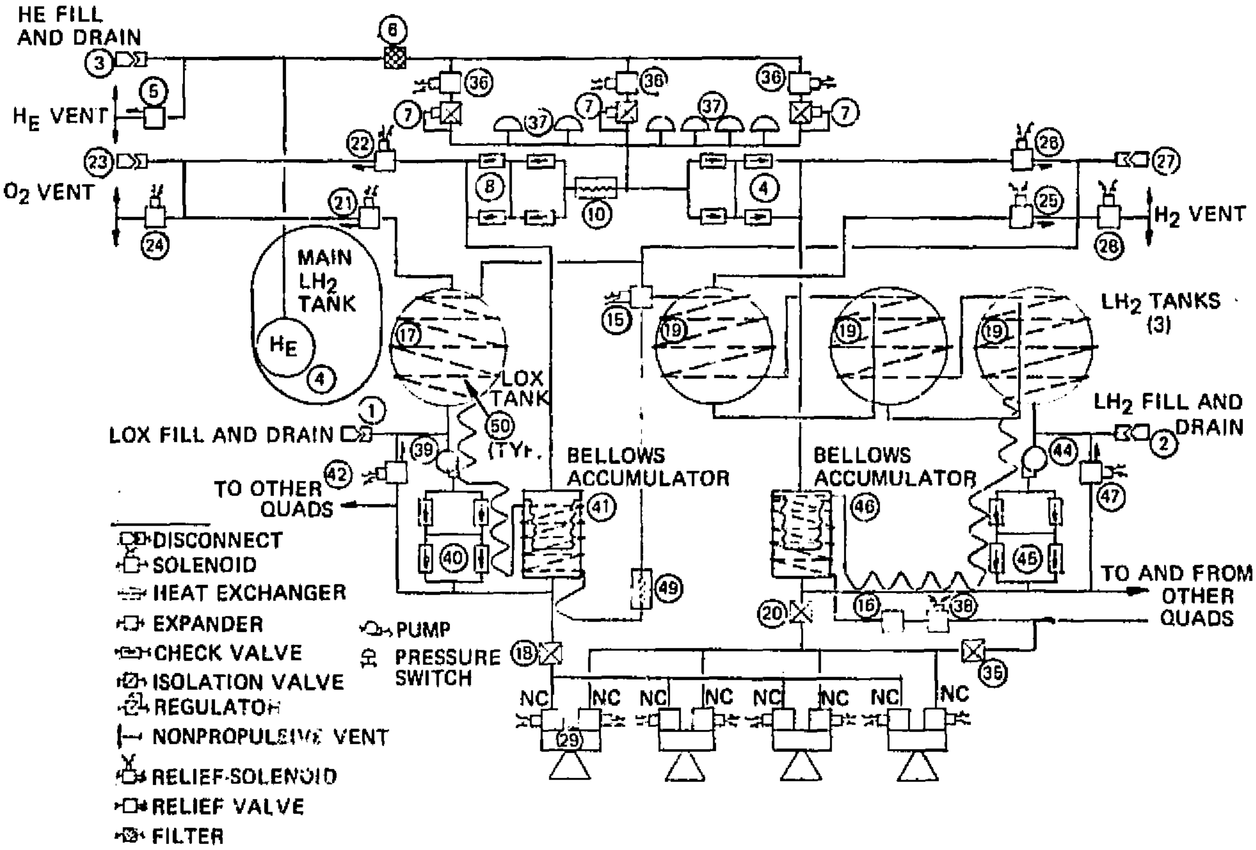

Figure 4-9. Mechanical Flow Diagram for Candidate 7 - Acceleration-Settled, Pump and Bellows Feed

CANDIDATE SYSTEM WEIGHT AND COST SUMMARY

A major criterion for comparing system options is the payload capability which can be obtained with a Tug employing the various systems. To generate data for this comparison, dry weights were estimated for each system at the component level in a form similar to the statements shown previously in Tables 4-2 and 4-5. The system dry weights were then used as inputs to a timeline computer program, which resized the tank weights according to the propellant and pressurant needed to fly the mission. The output of the computer program is a compatible set of APS, Tug, and payload weight data which were used to readjust the tank weights in the component weight statement for each candidate system. These early data are summarized by weight category in Table 4-7. The weight and performance data later were updated, as shown in Tables 4-2 and 4-5, but were sufficient for initial evaluations.

It can be seen that Candidate 6 provides the greatest payload capability and that pump-feed systems, in general, provide greater payload capability than the pressure-feed systems.

The dedicated candidate systems also were compared by cost and reliability criteria; again, using data generated early in the study.

Table 4-7. Interim Data for Dedicated Candidate APS Weight Comparison

WEIGHT CATEGORY (KG)	DEDICATED APS CANDIDATE						
	UPDATED BASELINE PRESSURE FEED	PUMP FEED (NO. 1)	TK WALL COOLED PRESS FEED (NO. 3)	MODULAR TW COOL PRESS FEED (NO. 4)	TW COOL BLADDER FEED (NO. 5)	TK WALL COOLED PUMP FEED (NO. 6)	PUMP AND BELLOWS FEED (NO. 7)
FIXED DRY	96.7	110.5	89.7	121.4	91.4	103.7	157.3
VARIABLE DRY*	71.5	15.1	69.2	65.1	49.3	14.4	24.1
COMPONENT TOTAL	168.2	125.6	158.9	186.5	140.7	118.1	181.4
LINES	16.4	12.7	24.5	13.7	24.5	12.8	24.5
INSULATION	4.9	3.8	7.3	4.2	7.3	3.9	7.3
COMPONENT MOUNTING	8.5	6.3	8.1	10.2	8.1	5.9	9.1
DRY SYSTEM	198.0	140.4	198.8	214.4	200.6	140.7	222.3
CONTINGENCY (13%)	25.7	19.3	25.8	27.9	26.1	18.3	28.9
TOTAL DRY SYSTEM	223.7	167.7	224.6	242.3	226.7	159.0	251.2
USABLE PROPELLANT	151.0	151.0	151.0	151.0	151.0	151.0	151.0
RESERVE PROPELLANT	15.1	15.1	15.1	15.1	15.1	15.1	15.1
TRAPPED PROPELLANT	3.6	4.5	3.6	4.5	3.6	4.5	6.7
BLEED - LOX	24.5	.0	.0	.0	.0	.0	.0
- LH_2	10.0	10.4	10.0	8.6	10.0	10.4	11.3
PRESSURANT	28.2	.0	27.5	26.1	27.5	.0	13.2
LIFTOFF WEIGHT	456.1	348.7	431.8	447.3	433.9	340.0	448.5
BURNOUT WEIGHT	270.6	187.3	270.8	287.7	272.9	178.4	286.2
PAYLOAD	2003	2232	2003	1957	1998	2251	1960

*LOX, LH_2, AND He TANK WITH INSULATION

NOTE: REVISED WEIGHTS FOR CANDIDATES 3 AND 6 ARE GIVEN IN THEIR RESPECTIVE CANDIDATE SYSTEM CHARACTERISTICS DISCUSSIONS AND IN SECTION 6.

WEIGHT CATEGORY (LB)	DEDICATED APS CANDIDATE						
	UPDATED BASELINE PRESSURE FEED	PUMP FEED (NO. 1)	TK WALL COOLED PRESS FEED (NO. 3)	MODULAR TW COOL PRESS FEED (NO. 4)	TW COOL BLADDER FEED (NO. 5)	TK WALL COOLED PUMP FEED (NO. 6)	PUMP AND BELLOWS FEED (NO. 7)
FIXED DRY	213.3	243.7	197.8	267.7	201.6	228.6	346.7
VARIABLE DRY*	157.6	33.2	152.6	143.4	152.6	31.8	53.2
COMPONENT TOTAL	370.9	276.9	350.4	411.1	354.2	260.4	399.9
LINES	36.1	28.0	54.0	30.0	54.0	28.3	54.0
INSULATION	10.8	8.4	16.2	9.2	16.2	8.5	16.2
COMPONENT MOUNTING	18.7	13.9	17.7	22.4	17.9	13.1	20.1
DRY SYSTEM	436.5	327.2	438.3	472.7	442.3	310.3	490.2
CONTINGENCY (13%)	56.7	42.5	57.0	61.5	57.5	40.3	63.7
TOTAL DRY SYSTEM	493.2	369.7	495.3	534.2	499.8	350.6	553.9
USABLE PROPELLANT	333.0	333.0	333.0	333.0	333.0	333.0	333.0
RESERVE PROPELLANT	33.0	33.0	33.0	33.0	33.0	33.0	33.0
TRAPPED PROPELLANT	8.0	10.0	8.0	10.0	8.0	10.0	14.7
BLEED - LOX	54.0	.0	.0	.0	.0	.0	.0
- LH_2	22.0	23.0	22.0	19.0	22.0	23.0	25.0
PRESSURANT	62.1	.0	60.6	57.6	60.6	.0	29.2
LIFTOFF WEIGHT	1005.3	765.7	951.9	986.8	956.4	746.6	988.8
BURNOUT WEIGHT	596.3	412.7	596.9	634.8	601.4	393.6	630.8
PAYLOAD	4416	4920	4416	4314	4404	4962	4320

*LOX, LH_2, AND He TANK WITH INSULATION

NOTE: REVISED WEIGHTS FOR CANDIDATES 3 AND 6 ARE GIVEN IN THEIR RESPECTIVE CANDIDATE SYSTEM CHARACTERISTICS DISCUSSIONS AND IN SECTION 6.

Itemized nonrecurring (DDT&E) and first unit recurring (hardware) costs were computed for the updated baseline and Candidate 6 concepts, since these represented typical pressure and pump-feed systems. Equipment costs included design and development support and the estimated markup over vendor costs for system contractor support, profit, and overhead. The DDT&E estimates included system contractor component costs as well as subsystem development costs.

Test and maintenance costs were based on Rockwell cost estimating relationships (CER's) and Global Positioning System (a current Rockwell satellite program) data. The nonrecurring costs associated with assembly and checkout, and with acceptance testing, fund the preparation of procedures and specifications as well as the initial support.

The test hardware costs included one complete vehicle set plus 10 percent spares of all flight hardware, with the exception of the thrusters. Only 4 thrusters (one quad) are necessary for DDT&E system tests.

The cost of the initial submittal of all data (DRL) required by the contract also was included in DDT&E costs. DDT&E system test programs were costed to cover the system-level contractor qualification program. The engineering cost associated with this item provides for system definition, specification preparation at the system and component levels, and the definition of reliability and safety procedures.

The cost of maintaining hardware and software is contained within the first unit recurring cost for tooling maintenance, peculiar support equipment, facilities maintenance, and data.

Table 4-8 summarizes the dedicated concept weight, cost, and reliability data used in the dedicated system selection. Data for the selected concepts are later refined for comparison with integrated systems.

Table 4-8. Summary of Interim Data for Dedicated APS Selection

SELECTION CATEGORY	DEDICATED APS CANDIDATE						
	UPDATED BASELINE PRESSURE FEED	PUMP FEED (NO. 1)	TK WALL COOLED PRESS FEED (NO. 3)	MODULAR TW COOL PRESS FEED (NO. 4)	TW COOL BLADDER FEED (NO. 5)	TK WALL COOLED PUMP FEED (NO. 6)	PUMP AND BELLOWS FEED (NO. 7)
DRY WEIGHT, KG (LB)	224 (493)	168 (370)	225 (495)	242 (534)	227 (500)	159 (351)	251 (554)
LIFTOFF WEIGHT, KG (LB)	456 (1005)	349 (766)	432 (952)	447 (987)	434 (956)	340 (747)	449 (989)
BURNOUT WEIGHT, KG (LB)	271 (596)	187 (413)	271 (597)	288 (635)	273 (601)	179 (394)	286 (631)
PAYLOAD, KG (LB)	2003 (4416)	2232 (4920)	2003 (4416)	1957 (4314)	1998 (4404)	2251 (4962)	1960 (4320)
DDT&E COST ($M)	11.638	12.968	11.441	11.283	12.261	12.604	12.489
FIRST UNIT COST ($M)	1.517	1.625	1.436	1.871	1.490	1.536	1.614
RELIABILITY	.9599	.9472	.9411	.8801	.9349	.9472	.9314

NOTE: REVISED WEIGHT, COST, AND RELIABILITY FOR CANDIDATES 3 AND 6 ARE GIVEN IN THEIR RESPECTIVE DISCUSSION SECTIONS AND SECTION 6.

4.4 DESIGN ANALYSIS

PERFORMANCE ANALYSIS BASIS

The Tug design constraints and the triple payload deployment mission requirements were programmed on a digital computer to provide rapid and accurate computation of performance effects of the APS. One type of program output is shown in the propulsion timeline tables of this report.

Payload capability and propellant usage of a given APS concept is determined by considering all of the operations which consume expendables during each event of a mission. The events are categorized into four types: coast, APS velocity change (ΔV), MPS ΔV, and payload separation. Although the last event does not consume propellant, the resulting change in total weight when a payload is separated changes the mass and inertia to be maneuvered by the APS and MPS.

Coast operations require that the Tug be held on-attitude within a specified deadband by thruster couples in each axis. Minimum impulse limit cycling is assumed for the deadband operation and the moment arm of each thruster is the radius to the thruster location, diminished by the cosine of the cant angle. The APS specific impulse is degraded for this short pulse operation in accordance with design estimates of this effect.

An attitude maneuver sequence of 90 degrees in roll and then in pitch is assumed to occur once each coast period. Since no time constraints were identified, a full 5 minutes is allowed for the maneuver in each axis. A shorter maneuver time would appreciably increase propellant consumption.

APS ΔV maneuvers are performed using the four aft thrusters (one of each cluster), which are pulsed for pitch and yaw steering. Roll steering is performed using lateral engines in couples. The net propulsion force which contributes to delta-velocity is the total force reduced by the cosine of the cant angle.

An MPS ΔV event begins by settling the propellant by APS, by tank head idle mode (THIM), or by a passive means such as a start basket. APS settling is shown in the timelines for the dedicated Candidate 3 and the integrated I-5 APS. The settling time is based on the acceleration capability and the amount of propellant remaining in the main hydrogen tank, and accounts for 4.5 free falls.

After propellant settling, the main engine begins its start sequence: THIM (for chilldown), pump idle mode (PIM), and mainstage. If the required delta-V during an event is not large, the MPS may not reach full steady-state thrust. To reflect practical operations, it was assumed that each succeeding thrust buildup phase woul not be entered unless the engine could be operated in steady-state in that phase for at least 5 seconds. The total impulse and the average specific impulse during each buildup phase is used to compute the propellant usage with the characteristic velocity equation. Roll control during main engine operation is performed by the APS assuming a constant propellant consumption rate which is based on records of RCS propellant consumption during Apollo CSM SPS engine burns.

Fuel cell reactant consumption, and propellant loss from main tank boiloff and APS bleed venting, are assumed to be constant rates throughout the mission, except that the boiloff ceases during MPS operation.

Propellant and pressurant tankage weight, as well as the quantity of pressurant, is resized for the dedicated APS concepts based on the propellant actually expended during the mission. The program initially computes two cases for assumed payloads to determine stage burnout weights for a fixed gross weight. These data are used to predict the correct payload, considering tanks and pressurant which have been resized from known basic input data. A burnout weight correct to within 0.45 kg (1 lb) results from the third iteration.

THRUSTER DESIGN POINT SELECTION

Thruster Design Analysis

The effect of thruster design parameters on thruster performance was examined in detail using the data from trade studies performed by Aerojet Liquid Rocket Company. The ALRC data are summarized in Figure 4-10.

To obtain these data, a parametric analysis of performance and design parameters was conducted over the following ranges using detailed mathematical models:

Thrust	111 - 444 N (25 - 100 lb)
Chamber pressure	69 - 345 N/cm^2 (100 - 500 psia)
Mixture ratio	2.0 - 6.0
Area ratio	40 - 200

The thruster specific impulse and characteristic exhaust velocity were analyzed using a one-dimensional equilibrium and kinetic performance model.

Nozzle divergence loss (which was less than 1 percent at all design points) was evaluated using a model that derives nozzle divergency efficiency and length from specified area and length ratios. All nozzles were designed to have a length 20 percent greater than the miminum for a particular design condition.

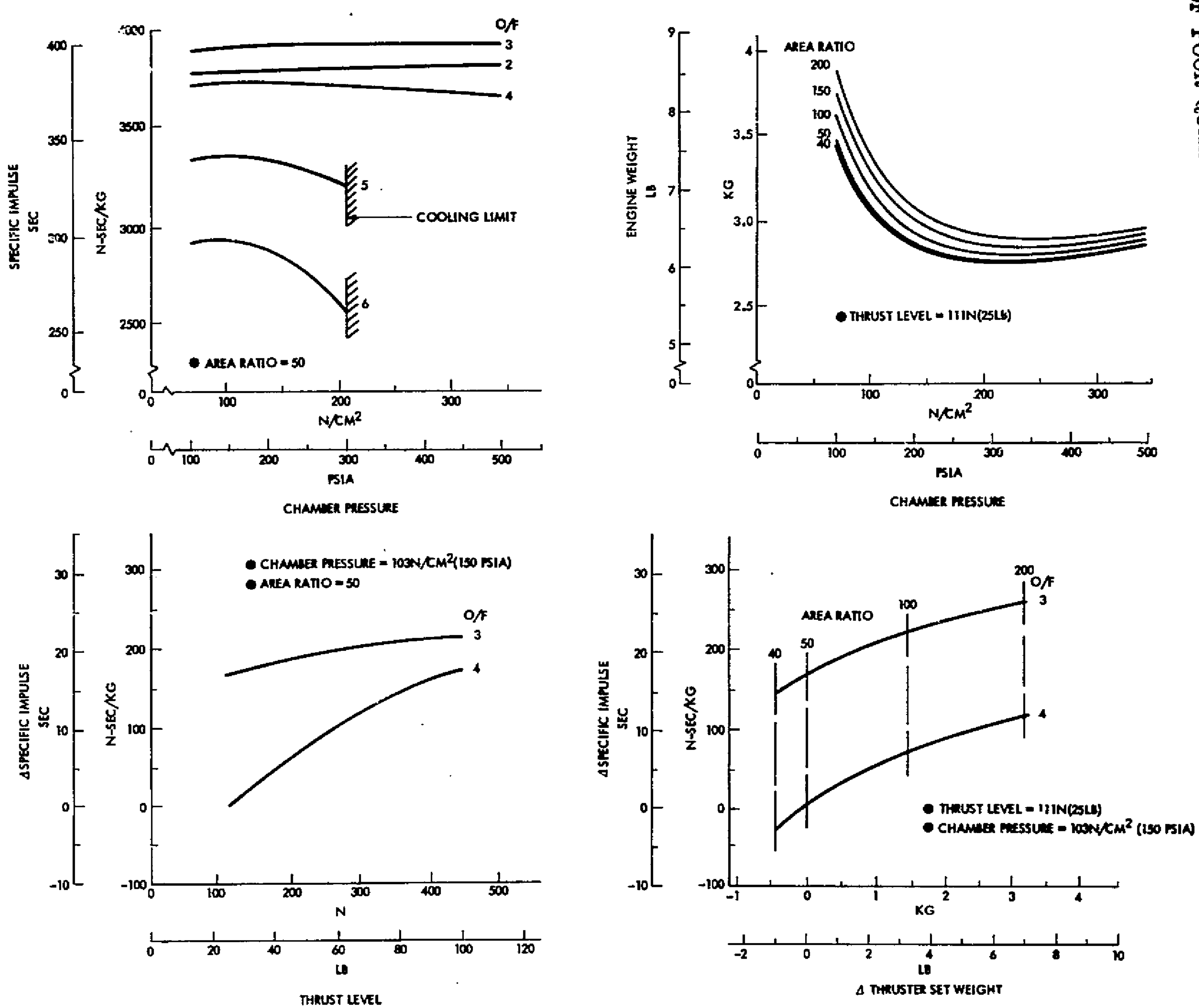

Figure 4-10. Thruster Parametric Data

Boundary layer performance losses were evaluated from a model using nozzle characteristics, chamber pressure, characteristic exhaust velocity, and wall temperature ratios. The boundary layer loss varied from approximately 2 to 4 percent over the thruster parametric range.

Thruster film cooling losses were derived, using a simple mass-weighted stream tube model, to account for the performance reduction resulting from the use of fuel-rich barrier cooling. The fuel film (barrier) cooling performance loss varied from less than 1 percent to approximately 40 percent over the parametric range. Extremely large cooling losses resulted at high operating mixture ratios (5-6) and at low thrust and high chamber pressure conditions. The performance loss is large under these conditions because of the gross level of film cooling required and because the core flow (flow inside of the fuel-rich barrier) performance decreases significantly as the overall mixture ratio is increased. In general, these points are considered to be unfeasible.

The energy release performance loss due to incomplete mixing effects were analyzed. By the nature of the thruster design, mixing is inhibited so that the desired barrier cooling characteristics are retained. As a result, nonuniform mixture ratio distribution may persist in both the core and barrier stream tubes. The degree of mixture ratio maldistribution and its effect on performance has been estimated but will require experimental evaluation and verification. For the purpose of this study, an energy release efficiency of 95 percent was selected as a reasonable value to achieve during a thruster development program which would trade off performance with the benefits of thruster durability and operational versatility.

This efficiency is typical of that obtained from a single-element coaxial injector which the thruster design concept approximates.

The flow split between the secondary fuel and the throat film coolant was analyzed from a heat transfer standpoint. The composition of gases at the throat also was determined.

From the analysis of specific operating conditions it is possible to construct a "feasibility map" which defines regions of acceptable combinations of thrust, chamber pressure, and mixture ratio for a given wall temperature. Such a map shows the usual trend of increased cooling margin (i.e., lower wall temperatures, with increasing thrust and decreasing chamber pressure). The initial baseline design lies clearly in the acceptable area on the feasibility map.

A number of assumptions were made to facilitate the analyses:

1. Throat diameter was scaled assuming constant C_f.

2. Contraction ratio was kept constant.

3. Both fuel sleeves were assumed to be 0.2 cm (0.080 in.) thick.

4. Rectangular coolant channels with convection on all sides were assumed in both sleeves. The channels were not optimized for wall temperature or coolant pressure drop.

5. Wall heat transfer coefficients were based on turbulent flow correlations. The ox-rich core gases were assumed to be fully mixed at their injection point, and the secondary fuel was assumed to be fully mixed with the ox-rich core at the throat film coolant injection point.

6. Fuel injection is at a unity velocity ratio (injected fluid to core), to minimize mixing of the stream and to lower downstream temperature.

7. The barrier film cooling analysis was performed using an ALRC computer model which is based on empirical data.

8. The throat was assumed to be the location of limiting wall temperature. Radiation losses from the throat wall were neglected, lending a degree of conservation, perhaps 27.8 K (50 F) to the results.

The peak specific impulse in Figure 4-10 occurs at a mixture ratio of 3 and falls off severely with mixture ratios above 4. This is unusual when compared with large O/H engines and is apparently due to the adverse scale effect on cooling limits with the small film-cooled engine. Also, specific impulse is not very sensitive to chamber pressure variations and engine weight is not very sensitive to area ratio.

Thruster and System Performance Analysis

The results of the thruster parametric studies were used in systems analyses to determine Tug payload performance as a function of thruster design point. Both pump- and pressure-feed generic types were subjected to the performance analyses.

APS characteristics which affect payload are shown in Figure 4-11. The figure presents two types of information: (1) APS performance requirements, and (2) the performance capability lines for the pump- and pressure-feed types of APS. This information is shown on one chart for Missions A and B by plotting total impulse and payload against APS burnout weight. The total impulse required to perform either mission increases slightly as the burnout weight increases. At the same time, higher burnout weight permits less payload. Payload is affected to a lesser extent by the APS specific impulse. Since APS capability can be measured in terms of total impulse, and the capability depends on system size, the capability lines are plots of total impulse versus burnout weight.

The total impulse required for Mission A increases approximately 49 N-sec for each kg (5 lb-sec for each lb) of APS burned weight. This value is related to the impulse increment necessary to impart the Mission A momentum of 104 m/sec (155 ft/sec) to a unit of mass. The payload decrease with burned weight is 2.7 units of payload per unit of burned weight for either reference mission.

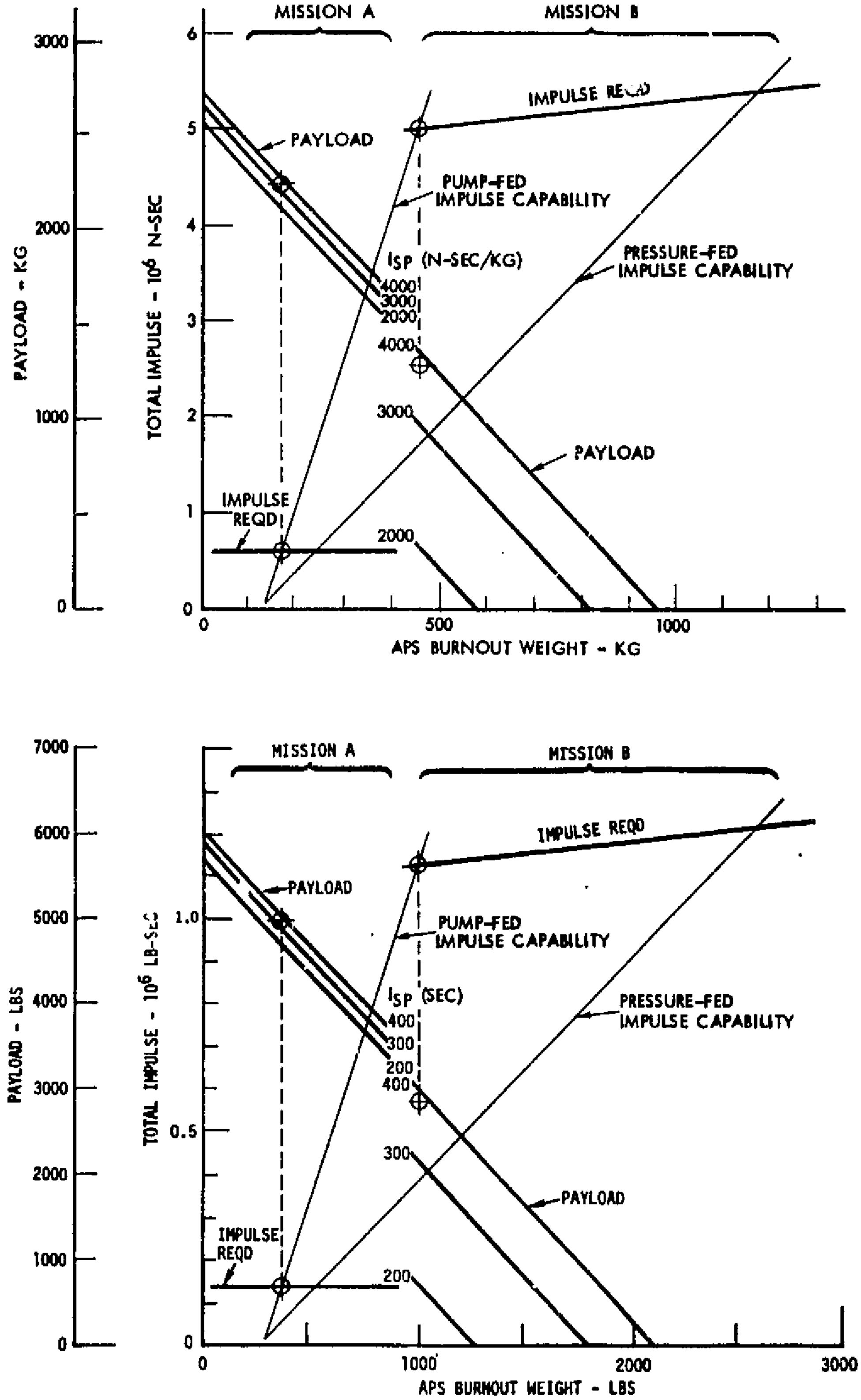

Figure 4-11. Dedicated APS Performance Capability

The APS specific impulse influence on payload is low compared to that of the MPS: 0.0462 kg per N-sec/kg (1 lb/sec) and 0.323 kg per N-sec/kg (7 lb/sec) for Missions A and B, respectively. The MPS value is nearly 4.62 kg per N-sec/kg (100 lb/sec) in comparison. The Mission B payload is more sensitive to specific impulse simply because more total impulse is required.

The required size of a given APS type is found at the intersection of its impulse capability line with the mission impulse requirement line in Figure 4-11. Its payload capability is then found at that burned weight. The design points for the capability lines shown include a mixture ratio of 4, a chamber pressure of 103 N/cm^2 (150 psia), and an area ratio of 50.

The results of the performance optimization analysis for Mission A with both system types are shown in Figure 4-12. Because of unique thruster parametric performance characteristics, a mixture ratio higher than the baseline value did not prove superior for the pressure-feed system, as could be expected based on the results of similar O/H optimizations for main engines.

For the pressure-feed systems, the lowest chamber pressure of 69 N/cm^2 (100 psia), which reduces tank, helium, and helium vessel weights, is optimum and increases payload about 45 kg (100 lb). Mixture ratios of 3, 4, and 5 are very nearly equal in performance.

An area ratio of 50 provides the best performance for both systems and also retains thruster installation suitability.

The pump-feed system showed no performance gain at chamber pressures higher than 103 N/cm (150 psia). This is due to thruster performance characteristics, which is this case are not unique to this engine concept. There is little gain in specific impulse performance with chamber pressure in the low Pc region. The specific impulse gain was offset by increases in pump power (battery) and accumulator weight. Pump-feed system performance with mixture ratio is even flatter than for the pressure-feed concept.

Thrust Level Analysis

The thrust level of the APS engines influences subsystem weight, cost, reliability, and propellant consumption and is influenced in turn, by the modes in which the APS is operated and, in particular, by the parameters and constraints inherent in these modes. These influences were investigated with the objective of recommending the most desirable thrust level. Since the results were to be used for comparison, the scope of the analysis included only those influences which identify distinctions between APS concepts.

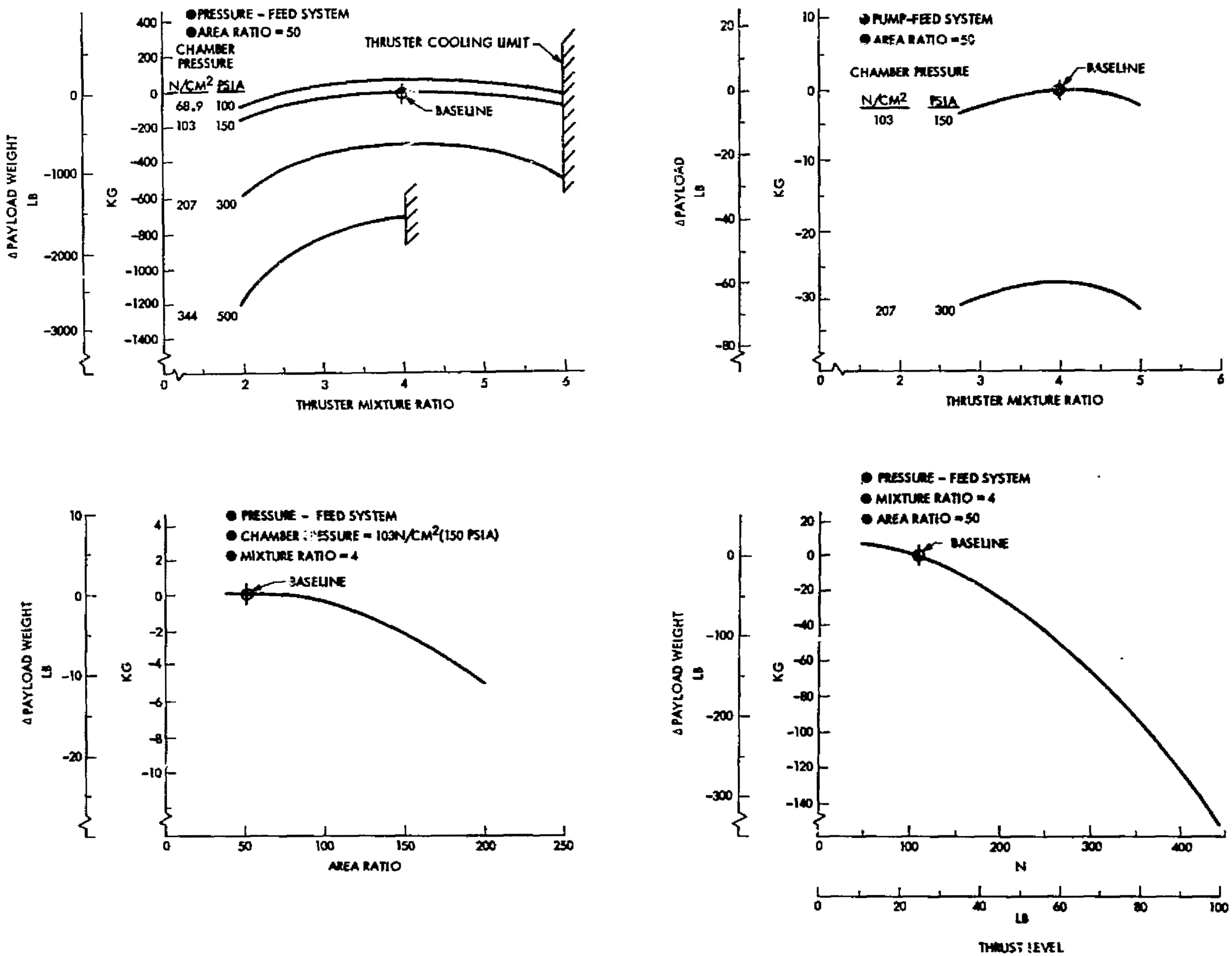

Figure 4-12. Thruster Design Point Selection Data

Different levels of thrust produce varied performance effects according to the mode of APS operation. For example, a high thrust level is advantageous in ΔV maneuvers since it reduces propellant gravity losses; however, it may increase the weight of engines, valves, and lines. In contrast, a low thrust level (or more precisely, a low minimum impulse bit) conserves propellant in the attitude hold coast mode. The total effect of thrust level on payload is shown in Figure 4-12, which describes high performance in the thrust range which includes the 111-N (25-lb) level.

Additional analysis of all Tug flight modes, including deployment and retrieval missions, indicates a clear preference for the baseline thrust level and also shows that the Tug payload capability is increased as the miminum impulse bit size is decreased. The baseline minimum pulse duration was decreased from 0.05 to 0.025 seconds on the basis of the analysis.

Thruster Cant Angle Analysis

A range of cant angles was studied to determine if the baseline angle of 25 degrees is appropriate. Since the Tug must fit within the 4.57-m (15-ft) diameter envelope dictated by the Orbiter, the APS engines are recessed in the Tug outer shell, requiring that the Tug structure be insulated from engine plume heating. At low cant angles, the engine exhaust impinges on the Tug structure (or insulation) and decreases efficiency by producing a negative force and a deflected exhaust stream. At high cant angles, the effective thrust varies as the cosine of the cant angle, producing a loss in efficiency. The effects of insulation weight and variable effective thrust due to impingement and cosine loss were analyzed in terms of payload weight for the baseline Mission A. The results show that, although the 25-degree cant angle is a good choice, a larger cant angle would increase payload performance. However, the model assumed for thermal protection should be more thoroughly investigated in the context of Tug systems before cant angle requirements are changed. The problems of APS installation interfaces are common to all of the cryogenic and earth-storable propellant APS concepts and thus will not reveal advantages for a single concept.

Cluster Location

The efficiency of the four aft thrusters, which are used for translational velocity changes, could be increased nearly 10 percent if they were oriented at zero cant angle. Relocation of these thrusters, or the entire clusters, aft of the oxygen tank midpoint would permit this. Although the increase in payload capability afforded by the design change would be small for Mission A, considering changes in insulation weight, the performance would be greatly improved for Mission B and for APS backup of an MPS failure.

In addition, the failure mode tolerance of the APS could be improved by changing to a more aft location. The moment arm parallel to the longitudinal axis of the Tug from the cluster station to the center-of-gravity station could then be used to generate pitch and yaw control torques by firing the proper roll thrusters.

Thruster Design Point Conclusions

The optimization studies yielded no reason for changing the dedicated APS design choices on the basis of chamber pressure, mixture ratio, area ratio, or thrust level. Future studies of cant angle and APS thruster location should be considered in the context of all Tug systems.

PROPELLANT ACQUISITION

A primary driver in the propellant storage and feed system design is to provide for zero-g and low-g space operation. The capability may be obtained by the use of capillary, bladder, or bellows devices, or by linear acceleration. The methods considered are shown in Figure 4-13.

The capillary devices include Concepts A through F. All the concepts depend on the level, direction, and duty cycle of acceleration by the vehicle's engines.

Concept A relies on open channel capillary pumping forces to relocate propellant after an adverse maneuver; its retention capability can withstand accelerations of 10^{-4} g, whereas Tug APS maneuvers create accelerations on the order of 10^{-2} g. Thus, this concept cannot provide positive propellant control. It is not suitable for the baseline LOX tank or LH_2 sump tank. However, this general type of design is suitable for the upstream LH_2 tanks.

Concept B utilizes screened acquisition channels to acquire propellant from anywhere within the tank. Available screen material is sufficiently fine to retain propellant easily in a 10^{-2} g acceleration field. However, the high thrust-to-weight ratio for the Tug MPS culminating in 3 g's at the end of the mission exceeds the retention capability of even the finest mesh screen available. Thus, this concept does not warrant further consideration.

Concept C, a compartmented approach which decreases the tank dimension over which the main engine acceleration acts, would be adequate if compartment dimensions were sufficiently small and fine mesh screen were used. This concept has the following problems: capillary barriers for compartmentation must be attached to or extremely close to the pressure vessel wall, weight is high, and fabrication and inspection are difficult. Thus, this concept is eliminated from further consideration.

Concept D is a screened basket, refillable during main engine thrust. Gas which flows into the basket during feedout under adverse APS maneuvers is purged (burped) out of the basket back into the tank under the hydrostatic head created by main engine burn. Fairly coarse screen is required for the basket and the bubble purge tube to permit gas expulsion. However, coarse screen breaks down readily under MPS burn accelerations near the end of the mission, permitting propellant to escape from the basket. Hence, this concept is rejected for the dedicated APS application.

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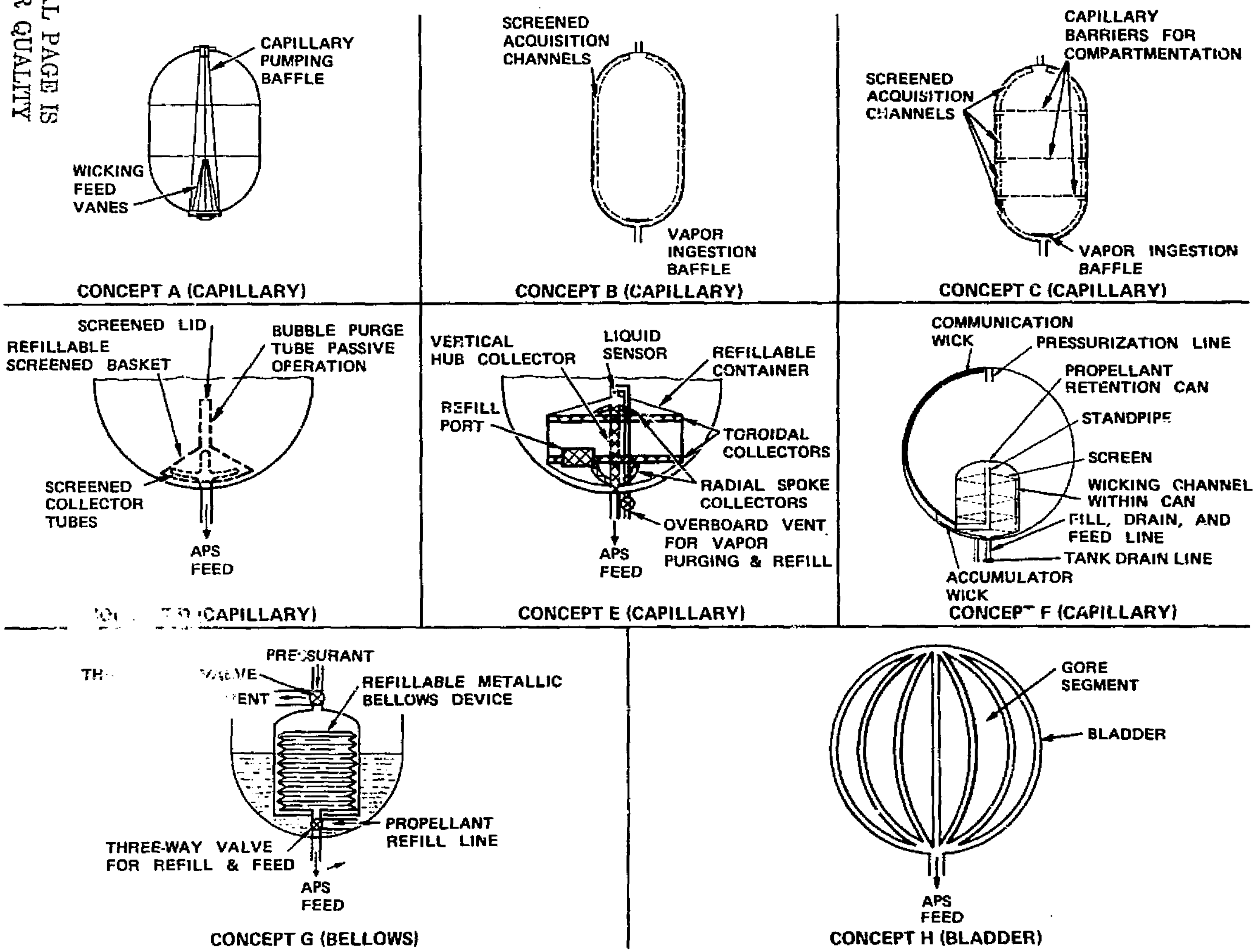

Figure 4-13. Alternative Acquisition and Orientation Devices

Concept E utilizes refill during MPS and APS settling burns with gas vented to space. This eliminates the problem of screen breakdown and propellant loss encountered with Concept D, but adds the complexity of an overboard vent valve. If the screened refill port is replaced by a valved refill port, the container can be pressure-isolated from the tank. Designs of this type have been studied by MDAC (Reference 4). This concept is heavy, complex, and less reliable than Concept F, but is reconsidered for the integrated APS.

Concept F was selected for the LOX and LH_2 sump tank design. It is passive, utilizing redundant screens of coarse mesh within the can. It uses an open-channel communication wick to pump propellant by capillary action to the accumulator wick continguous with the can entry port.

Since volume constraints dictate that the hydrogen be stored in three tanks, series feed was chosen. Thus, only the tank nearest the engines need be used as a sump tank to provide gas-free liquid at the required temperature to the engines and to the thermal control system. The upstream tanks need not always provide liquid as the sump tank acts as a gas accumulator; however, the amount of gas transferred before liquid transfer is complete must be held to a manageable amount. Ultimately, the upstream tanks must be essentially depleted of liquid to provide good expulsion efficiency. Following upstream tank depletion, thermal control can be relaxed to allow gas heatup, thereby reducing helium pressurant requirements.

The LH_2 sump tank employs a 0.044 m^3 (1.57 ft^3) capacity propellant retention can to assure gas-free liquid feed to the engine. Flow out of the can to the engine is replaced by flow into the can through a screened slot at the bottom of the can (Figure 4-14). The capacity of the can was established by the amount of gas which could enter the can during the mission. This is determined by the frequency, duration, and flow rates during adverse maneuvers (those which tend to dislocate liquid from the liquid accumulator region at the entry slot to the can), and the ability of the accumulator wicks to retain propellant during these maneuvers.

The major limitation of the design is its inability to retain propellant during sustained -X, $\pm$Y, and $\pm$Z maneuvers. During such maneuvers, gas will flow prematurely to the downstream tank; however, the retention can and wick accumulator in the sump tank serve to prevent gas passage to the engines. The expulsion efficiency is expected to be at least 0.65 percent.

Since only one LOX tank is used, the design is similar to, but smaller than, that of the LH_2 sump tank. The scale effect favors the retention and rapid reacquisition of propellant to the screened slot of the can.

Concept H of Figure 4-13 describes a bladder system which can be refurbished after each mission. Although a bladder configuration has been used successfully in the Apollo service module for storable propellants, the use of bladders for cryogenic propellants requires additional development and thus is not a good competitor.

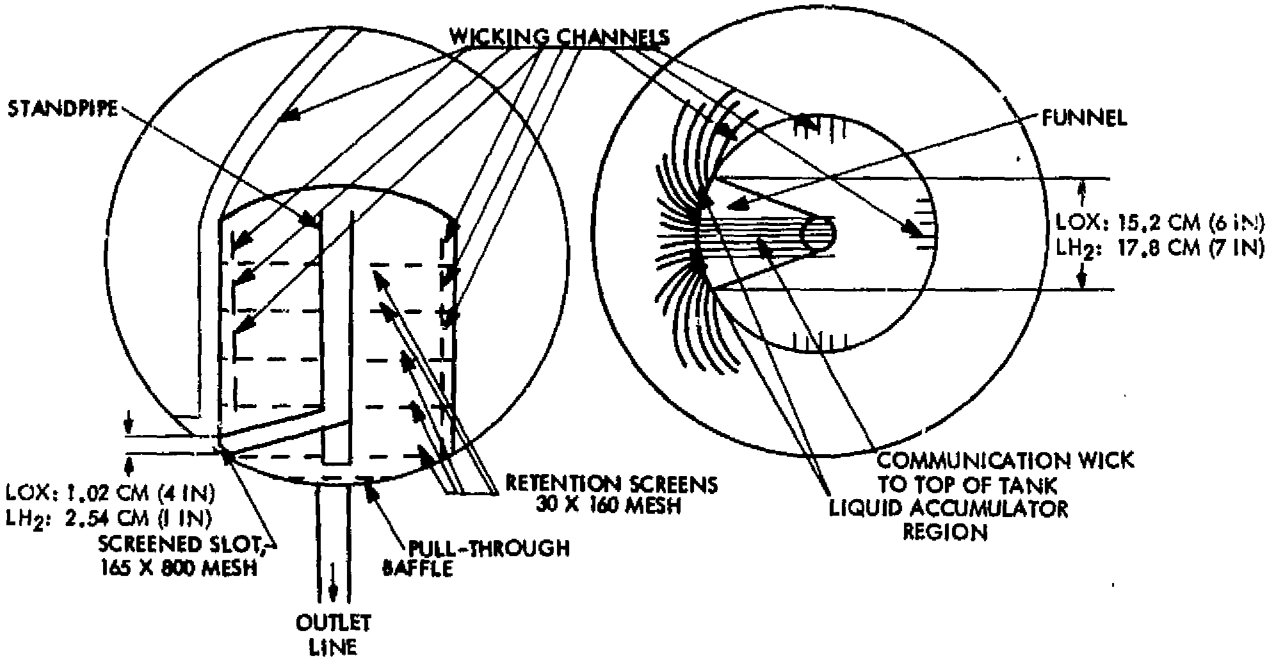

Figure 4-14. Capillary Propellant Acquisition Device

The bellows (Concept G) can be used either as a refillable container in the propellant tank (dedicated or integrated), as an accumulator separately pressurized and located internally or externally in the integrated concept, or as a separate propellant storage positive expulsion device in the case of a dedicated system. However, the bellows concept has the disadvantage of high weight.

To minimize the weight penalty of a metallic bellows device, an APS tankage system can be considered that uses the bellows only during Tug attitude control maneuvers and relies on vehicle acceleration to settle the propellant in conventional storage tanks during Tug linear translation maneuvers. Dedicated APS Candidate 7 reflects this approach. During the initial phase of the linear translation, propellant is drawn from the bellows accumulator tank until propellant in the main storage tank has been settled.

PUMP FEED

Several of the candidate APS tankage and feed system concepts use electric motor-driven pumps for propellant feed. Candidates 1 and 6 use zero-g propellant acquisition devices to assure all-liquid supply at the pump inlets. Candidate 2 was conceived as a method of eliminating the development effort associated with cryogenic zero-g propellant acquisition by capitalizing on the projected capability of the liquid-liquid thruster to accept mixed phase propellants and by incorporating a pump with a similar mixed phase capability.

The pump analysis considered mixed phase flow as well as all-liquid flow. The type of flow greatly affects pump design, speed, efficiency, weight, number of stages, development effort, and related factors. Basically, there are four types of pumps which can be considered as candidates for system operation, but naturally no single type can be expected to operate at peak efficiency for both the liquid oxygen and liquid hydrogen system under all types of fluid flow. The four types are centrifugal, vane, piston, and gear, each requiring the use of an inverter to prevent excessive dc motor brush wear and arcing.

All of the candidate tankage and feed system concepts incorporating pumps use a downstream accumulator for pressure oscillation control. The design of a combined pump and accumulator combination with trapped helium ullage was investigated. The ullage will expand and contract as propellant is alternately withdrawn and returned to the accumulator. Liquid-gas interface control can be provided by a capillary system or by a small bellows or bladder.

Assuming the helium acts as an ideal gas, the ratio of the initial volume of helium to the final volume of helium (or the total volume of the accumulator) is inversely proportional to the ratio of initial and final accumulator pressures. Thus, the thruster inlet pressure could be controlled between 138 and 172 N/cm^2 (200 and 250 psia) by a minimum helium-to-accumulator volume ratio of 0.80. This inlet pressure variation will result in a thrust variation from 103.5 to 122 N (23.3 to 27.5 lb). A review of docking and separation maneuvers indicates that such a variation occurring on all thrusters simultaneously is acceptable.

A tradeoff between pump flow and accumulator liquid capacity has been conducted. The minimum accumulator capacity results when all of the propellant flow for the thrusters is provided by the pump. As the pump flow rate is reduced, more of the propellant flow must be provided by the accumulator. After thrust termination, pump operation is continued to replenish the accumulator. The largest single demand on the accumulator is 123,000 N/sec (27,665 lb-sec) total impulse at 37.19 hours. This is equivalent to 33.1 kg (73 lb) of propellant, or 6.6 kg (14.6 lb) of LH_2 and 26.5 kg (58.4 lb) of LOX. The propellant is consumed over a 275-sec period by four thrusters burning simultaneously to provide translation. Thus, as the LH_2 pump flow approaches zero, the required accumulator capacity approaches 6.6 kg (14.6 lb) of LH_2.

The results of a weight trade study for the LH_2 accumulator and pump are presented in Figure 4-15 in which the LH_2 accumulator weight (a function of capacity) is plotted versus LH_2 pump power (a function of flow rate). Pump and inverter weights also are shown. As can be seen, minimum weight occurs at the maximum pump flow and minimum accumulator capacity.

From a reliability standpoint, it is desirable to limit the number of pump cycles per mission to approximately 100. Therefore, accumulator design points of 0.45 kg (1 lb) of LH_2 and 1.36 kg (3 lb) of LOX were selected. These capacities are more than adequate to handle pump start transients up to 5 sec.

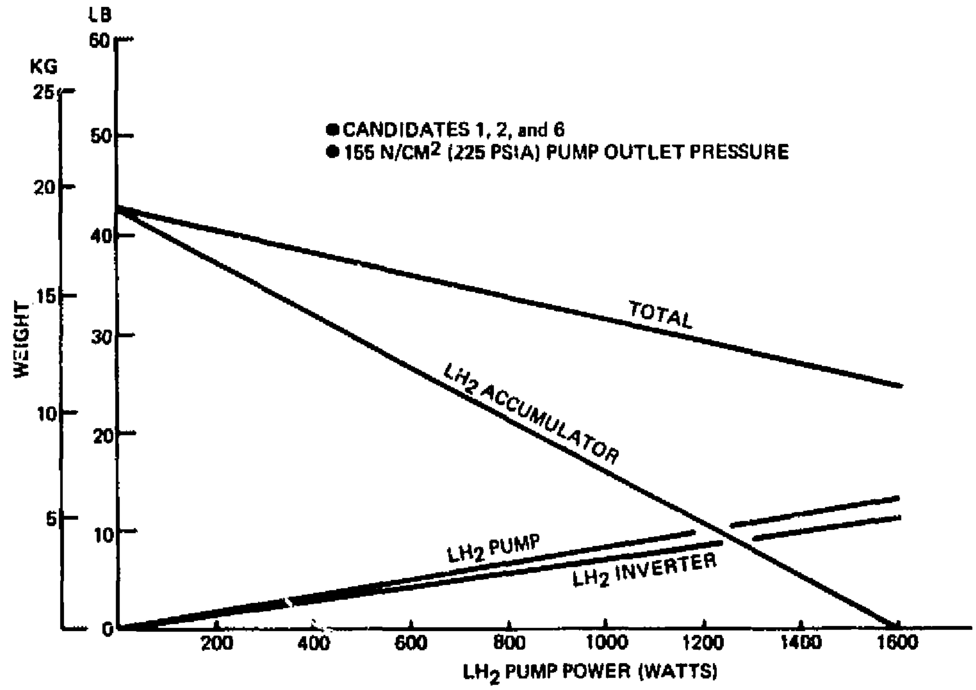

Figure 4-15. LH_2 Accumulator, Pump, and Inverter Weight as Function of Pump Power

Short-duration attitude control pulses occurring during sustained four-thruster linear translation maneuvers also will be supplied by the accumulators. The pump sizes selected for minimum system weight are 0.1 kg/sec (0.22 lb/sec) LOX and 0.0245 kg/sec (0.054 lb/sec) LH_2, which are sufficiently greater than the flow of four thrusters to provide for accumulator replenishment.

The total power requirement for both pumps based on four-thruster flow rate is 2000 watts, or a total of 0.98 kwh for a complete mission. Although the additional reactants consumed by the existing Tug fuel cells would be negligible - (approximately 0.40 kg (0.88 lb) - it is estimated that two additional fuel cell modules weighing 13.6 kg (30 lb) each would be required to handle the extra 2000-watt load during peak power periods. In comparison, a silver-zinc primary battery with a 7-day life can be added for only a 5-kg (11-lb) weight penalty. Because of this weight savings, the battery approach was selected for the pump power supply.

THERMAL CONTROL

One of the most critical design considerations for a cryogenic APS is thermodynamic control of the propellants. Proper thermodynamic control of LOX and LH_2 is necessary to avoid tank overpressurization, loss of propellant through venting, and excessively warm propellants at the thruster inlet. In addition, for a tankage system using capillary screened compartments for zero-g propellant orientation, thermodynamic control is required to prevent vaporization or drying out of the capillary wicks.

Five control concepts were considered, including no venting, direct overboard venting, and three types of thermodynamic venting with heat exchangers: internal, internal with bulk mixer, and wall-mounted.

Each of these concepts is based on thermal isolation of the storage tanks and feed lines through the use of multilayer insulation and low conductivity supports. Heat transfer of the insulation was analyzed and the thermodynamic control system heat loads established. Each candidate concept was evaluated and performance predictions are presented for critical operational characteristics.

Effective thermal conductivity values of 0.312 J/hr-m-K (4×10^{-5} Btu/hr-ft-R) and 0.249 J/hr-m-K (5×10^{-5} Btu/hr-ft-R) were taken from Reference 5 for the LOX and LH_2 insulation, respectively.

Insulation thicknesses of 1.27 to 2.54 cm (0.5 to 1 in.) were assumed and the heat leak values so calculated were doubled to account for structural supports and insulation layup effects. The resultant heat loads are:

	LOX	LH_2
Tank, J/hr (Btu/hr)	5590 (5.3)	17900 (17.0)
Feedline, J/hr (Btu/hr)	5490 (5.2)	7800 (7.4)
Total	11080 (10.5)	25700 (24.4)

Because of the strong sensitivity of cryogenic system design to heat load magnitudes, these values were recomputed by using data from another source (Reference 6) and applying the radiation heat transfer equation. Close agreement between the two methods was reached. Because of the importance of minimizing heat leak, an effective emittance value of 0.002 was selected as a design point. Therefore, these values - 25700 and 11080 J/hr (24.4 and 10.5 Btu/hr) total for LH_2 and LOX - are used for all dedicated concepts.

If heat load to storage volume ratios are low and if propellant withdrawal is evenly distributed over the storage period, a closed-tank or no-venting concept can be considered. Tank pressure rise may be completely or partially offset by thruster propellant consumption. This concept was analyzed for the baseline System A mission profile and heat loads previously presented. Using the open system general energy equation, the ideal condition of a mixer system was assumed to isothermalize the liquid and vapor propellant. The propellant was assumed to be saturated at atmospheric pressure at Shuttle liftoff.

Propellant temperature and vapor pressures were found to be excessive for the 164-hour mission, reaching 31.8 K (57.4 R) and assuming 105 N/cm^2 (153 psia) in the LH_2 tank for the last APS burn. For this reason, no-venting has been eliminated as a viable thermodynamic control concept, this applies to either pump-feed or pressure-feed initial conditions.

After concluding that some form of venting is required for thermodynamic control of the propellants, the simplest method to be considered is direct overboard venting of the ullage to relieve tank pressure and reduce propellant temperature by vaporization of the bulk liquid. This approach is ideal for ground operations or for space vehicles which can provide an artificial gravity by rotation or linear translation.

Direct overboard venting in a zero-g or low-g environment, however, will result in excessive loss of liquid propellant. Capillary screen barriers which work well to block the passage of gas through a liquid film will not adequately retard the flow of liquid when either wet or dry. Locating the vent outlet at the ullage also is difficult because of the variable liquid orientation resulting from the multi-axis thrusting of the APS. Even after locating the ullage, significant liquid entrainment into the vent flow can result for high liquid mass levels.

Periodical propellant settling and venting also has been considered whereby the propellant temperature and pressure would be allowed to rise between settled vents. This approach has the serious drawback of not providing the subcooling necessary to preclude vapor entrapment within the zero-g capillary compartments or the thruster feed manifold. For these reasons, direct overboard venting was not considered further.

Thermodynamic venting can be used to remove energy from the propellant tanks in zero-g without the high weight penalty of releasing liquid to space. This method of thermodynamic control works by withdrawing liquid from the tank, expanding it to a low-pressure and -temperature two-phase fluid, and then vaporizing the liquid by heat transfer from the stored propellant. Although the propellant orientation system normally assures liquid withdrawal, the system can still work efficiently if unusual conditions allow gas into the vent system.

Heat exchange between the cold expanded vent fluid and the stored propellants can be accomplished in several different ways. If heat exchangers inside the tanks are used, expanded hydrogen can be passed through an LH_2 tank internal heat exchanger and then through a hydrogen/oxygen external heat exchanger before it is vented overboard. The chilled oxygen is routed through a LOX tank internal heat exchanger and then is recovered in the Tug main propulsion LOX tank. For this system, heat transfer from the bulk propellant and ullage volume depends on fluid convection and conduction as well as conduction through structural members and heat exchanger fins. Natural convection was found to be a significant contributor to the total heat transfer at 10^{-5} g, but only a small contributor at 10^{-7} g. For Mission A, the vehicle will be in gravity environments less than 10^{-8} g for up to 52 hours at a time while coasting at geosynchronous orbit. At this low acceleration level, conduction is by far the predominant heat transfer mode.

A temperature stratification analysis of the no-venting case showed that stratification in zero g caused unacceptable pressure rises. APS propellant usage during these periods is not sufficient to maintain a cold environment.

Internal heat exchanger and propellant bulk mixers have been evaluated as a part of several study programs (References 7 and 8). The mixers reduce thermal stratification and bring the bulk fluid in contact with the heat exchanger. Although such systems perform well, they have the disadvantage of complexity and the hazardous requirement for electrical power in the LOX tank. For thermodynamic control concepts utilizing expended GH_2 to chill the LOX system, an additional external hydrogen/oxygen heat exchanger is required to preclude the potential hazard of hydrogen leakage within the LOX tank. It is believed that these disadvantages can be circumvented by the use of tank wall-mounted heat exchanger tubes. The fabricability and performance of tank wall-mounted heat exchangers have been demonstrated successfully in two experimental programs (References 33 and 34).

By concentrating external tank wall-mounted heat exchanger tubes at heat shorts under the insulation, the tank or feed line heat load can be essentially intercepted before it reaches the propellant and temperature stratification is virtually eliminated. This approach eliminates the need for bulk mixers or internal conduction fins.

During this study, 13 different tube attachment designs were evaluated from the standpoint of thermal performance and productivity. From these, a concept utilizing local brazing to contain a tube within a chem-milled channel has been selected for weight, cost, and reliability assessment because of its favorable heat transfer and producibility characteristics.

Various techniques have been studied for providing the thruster inlet temperature requirements. All of these concepts involve bleeding LH_2 from the tank to the thruster inlets and then expanding the hydrogen flow through a Joule-Thompson valve to a lower temperature to act as a heat sink.

For the baseline concept, the expanded hydrogen bleed is used to cool the LH_2 feed lines and storage tanks and then to cool the oxygen bleed flow in an external hydrogen/oxygen heat exchanger. This bleed flow is then used to trace the LOX feed lines. If the lines are not traced, the LOX feed lines are cooled only by their internal flow. The oxygen feed line heat load is such that either the bleed flow rate or the temperature rise are excessive. For example, an 11.1 K (20 R) temperature rise dictates a 0.27-kg/hr (0.6-lb/hr) bleed flow rate, resulting in a 44.5-kg (98-lb) propellant loss for a 164-hour mission.

As another alternative, Joule-Thompson expansion of the oxygen bleed was considered. The cold oxygen would be routed around the LOX feed lines and then would cool the LOX storage tank. This approach has the advantage of eliminating the weight, cost, and potential hazard associated with the hydrogen/oxygen external heat exchanger. The additional heat sink provided by the expanded oxygen reduced the LOX bleed flow to 0.05 kg/hr (0.117 lb/hr) or a total loss of 8.6 kg (19.2 lb) for a feed line temperature rise of 11.1 K (20 R).

A third alternative was evaluated which eliminates LOX bleed altogether. After cooling the LH_2 tank, the expanded LH_2 bleed fluid is routed under the multilayer insulation around the LOX feed lines and storage tank, absorbing the complete heat load of the LOX system. The hydrogen bleed is warmed with a 10-watt electrical heater to a temperature of 72.2 K (130 R) to preclude freezing of the oxygen. As the design flow rate of 0.0605 kg/hr (0.133 lb/hr), 26000 J/hr (24.6 Btu/hr) are available for cooling, while the LOX system heat load is only 11100 J/hr (10.5 Btu/hr). To avoid over-cooling, a control valve would be provided to bypass the electrical heater and LOX tankage system. Although this concept saves the weight of the hydrogen/oxygen heat exchanger and the LOX bleed, it does increase the potential hazard associated with close proximity leakage of hydrogen and oxygen. This hazard is avoided while in the Shuttle cargo bay, however, since cooling is not needed during the applicable flight phases.

RELIABILITY ANALYSIS

The inherent subjectivity of reliability assessment was recognized early in the study and a procedure was devised to permit a fair comparison of APS concepts. First, the storable bipropellant APS reliability was computed using the data in Reference 2. The purpose was to verify that the allocated goal of 0.996 could be reached using the quoted component failure rates with the 144-hour, single payload deployment mission used in Reference 2. It was also necessary to understand the success path logic in order to rectify any differences in the liquid-liquid O/H APS system logic. Since it was a more analogous case and more detailed data were available on the storable bipropellant system, it was chosen for this purpose over the storable monopropellant system.

Concurrently, initial analyses of the liquid-liquid O/H APS reliability centered around the refinement of failure rate data and the definition of operating times, and culminated in the description given in Table 4-9.

Next, as shown in Table 4-10, the storable bipropellant APS reliability was reassessed by using the refined failure rate data and then by imposing the triple payload placement mission with refined criteria. The change from the single to the triple payload placement mission produced the more pronounced effect on reliability. The storable monopropellant APS was then evaluated.

Finally, the reliability of each liquid-liquid O/H candidate concept was computed, as shown in the table. The reliability analysis was closely coordinated with the design effort to assess component operating life characteristics and to define component redundancy needs on a rational basis. All of the cryogenic APS candidate failure rates were adjusted by including environmental degradation (K factors). The effect of the degradation also is shown in the table for the updated baseline concept.

At this point in the reliability study, the data were sufficiently compatible to permit concept selection, although none of the candidates actually achieved the reliability goal. The reliability increase necessary in each case was assumed to be available through the SR&T development of critical components.

Table 4-9. Liquid-Liquid O/H APS Failure Rates

Component	λ = F.R. x 10^{-6}	Source	K Factor	λ Total
Solenoid valve	4.8	Shuttle	LH 5	24.0
(thruster)			LOX 3	14.4
Igniter	0.6	Planning Research	-	0.6
Isolation valve	6.5	RI (TA66)	LH 5	32.5
(dual coil)			LOX 3	19.5
Orifice	0.15	AVCO	LH 5	0.75
Capillary device	0.023	RI (72-2)	LH 5	0.115
			LOX 3	0.069
Tank-He	0.0114	RI (72-2)	-	0.0114
Tank-cryogenic	0.2976	Apollo	-	0.2976
Bladder	3.6	MACDAC	-	3.6
Solenoid valve	.75 Operate	RI (TA66)	LH 5	3.75
(3-position)	1.0 Leak			5.0
Check valve	7.65 Leak	MACDAC	LH 5	38.25
	1.35 Operate			6.75
Heat exchanger	1.0	RI (TA66)	LH 5	5.0
Relief valve	9.0	JPL	LH 5	45.0
			LOX 3	27.0
Burst disc/relief valve	10.0	MACDAC	LH 5	50.0
			LOX 3	30.0
Vent valve	9.0	JPL	LH 5	45.0
			LOX 3	27.0
Regulator	3.6295 Leak	Apollo	LH 5	18.1475
			LOX 3	10.8885
	1.725 Return			8.625
				5.175
	0.595 Operate			2.975
				1.785
Filter	0.0114	JPL	LH 5	0.057
Heater	0.0228	JPL	LH 5	0.114
Fill valve and cap	0.171	JPL	LH 5	0.855
Lines and fittings	0.02	JPL	LH 5	0.1
Nozzle/chamber	0.16	MACDAC	-	0.16
Pump-electric	8.7	Saturn	LH 5	43.5
			LOX 3	26.1
Accumulator	6.2	Autonetics	LH 5	31.
			LOX 3	18.6
	0.06 Lead	Apollo		0.3
				0.18

80% of valve FR is in leakage; 90% of remaining FR is in unenergized state (i.e., failure to return); 10% is in energized state (i.e., fail to operate)

t_1 = 164 hours - mission time

t_2 = 2,300 cycles - single thruster mission operation

t_3 = 82 hours - assumed failure point

t_4 = 1 cycle - isolation

t_5 = 50 cycles - 3-position valve operation

t_6 = 10 cycles - vent

t_7 = 32.8 hours - 20% duty cycle on heater

t_8 = 5,060 cycles - regulator

t_9 = .25 hours - max burn on any thruster

t_{10} = 230 cycles - assumed standby operation

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Table 4-10. System Reliability Values

Configuration	Condition			Reliability
	Mission	Failure Rate	K Factor	
Storable Propellant APS Analyses				
Bipropellant	*	*	1	.995980
Bipropellant	*	R	1	.993345
Bipropellant	R	R	1	.988443
Monopropellant	R	R	1	.9955[illegible]8
Baseline Environment Comparison	R	R	1	.9919930
Candidate Comparison				
0. Baseline	R	R	R	.959887
1. Pump feed	R	R	R	.913014
2. Mixed phase	R	R	R	-
3. Pressure feed, tank wall-cooled	R	R	R	.941070
4. Modular	R	R	R	.880081
5. Bladder feed, tank wall-cooled	R	R	R	.934882
6. Pump feed, tank wall-cooled	R	R	R	.947204
7. Pump & bellows feed, acceleration settling	R	R	R	.931415

* - McDonnell Douglas data
R - Study data

5. INTEGRATED CONCEPT DEVELOPMENT

This section describes the development of an integrated cryogenic APS concept. Initial candidate screening evaluations and a summary of their trade study basis are presented and followed by a detailed description of the selected integrated concept. The design, operation, weight, performance capability, cost, and an initial appraisal of supporting research and technology (SR&T) requirements for the APS and its major components are defined and discussed. The impact of the APS on other vehicle subsystems also is presented. Design analyses and assumptions made in developing the integrated APS concept and defining its characteristics are included at the end of this section.

5.1 CANDIDATE SCREENING

Parametric studies and candidate screening evaluations applied to dedicated APS concepts identified potential combinations of design concepts for providing propellant acquisition, thermodynamic control, and propellant feed for a cryogenic APS utilizing such a tankage system. After the elimination of operationally incompatible and low potential combinations, weight, reliability, and cost trade studies resulted in the selection of the following concepts for dedicated systems:

Function	Design
• Propellant acquisition	• Capillary devices
• Thermodynamic control	• Expanded H_2 bleed feedline tracing and tank wall-mounted cooling coils
• Propellant feed	• Pump feed with pressure feed alternate

The principal characteristic of an integrated cryogenic APS is that its propellants are drawn directly from the MPS tanks without the necessity for separate storage. This difference in design has its most significant impact on the approach to propellant acquisition. The functional requirements for propellant thermodynamic control and propellant feed can still be satisfied using techniques similar to those applied to the dedicated concepts. The following section presents the results of trade study and screening evaluations for several propellant acquisition concepts. This is followed by a review of the pressure versus pump alternatives for propellant feed as they apply to an integrated APS.

PROPELLANT ACQUISITION

Figure 5-1 presents schematic representations of the concepts considered for propellant acquisition for an integrated system. The concepts considered are restricted to capillary systems since other techniques were eliminated during the analysis of dedicated systems because of excessive weight (bellows), technology risk (bladders), or inability to satisfy thruster inlet requirements (mixed phase).

The capillary systems evaluated for an integrated APS can be classified as either nonvented (non-refillable) or vented (refillable). The vented systems can be further subdivided as either internally vented or overboard vented. As these concepts evolved during the study, candidate numbers were assigned (I-1 through I-5) as shown on Figure 5-1. For this phase of the study, all the candidate concepts were considered to be installed within the main propellant tanks. Subsequent evaluation of the factors involved with location of the zero-g reservoir are presented in Section 5.3, Design Analyses. The most favorable installation was found to be external to the main tank.

Two nonvented systems were considered. The first, based on the principle of vapor accumulation (Candidate I-3), consists of a zero-g reservoir or container incorporating a screened inlet port and internal compartments

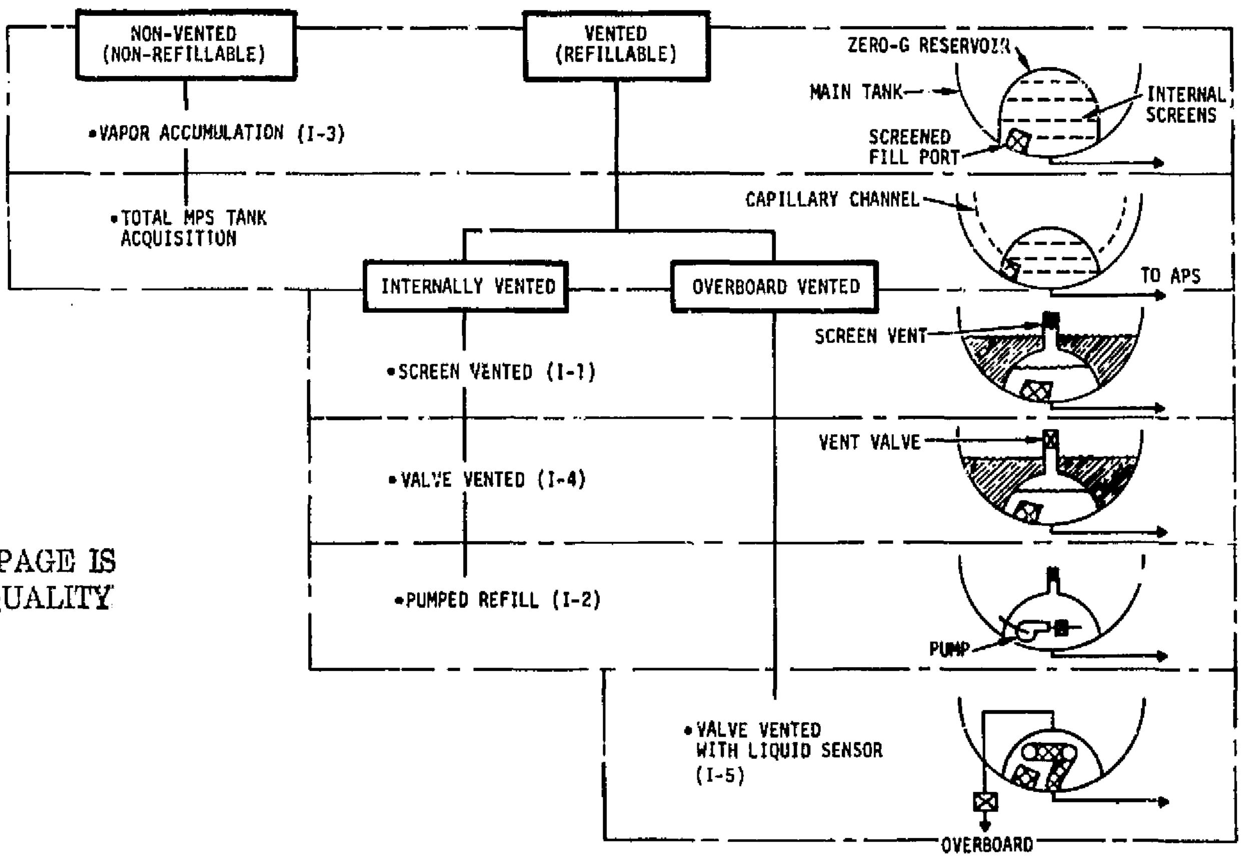


Figure 5-1. Integrated Cryogenic APS Capillary Systems Considered

separated by self-wicking screens. The screened inlet port prevents the loss of liquid during adverse acceleration while the internal screens assure vapor-free feed to the APS pumps and thrusters. The internal design is basically the same as the one for dedicated APS tanks shown on Figure 4-14.

When propellants in the MPS tanks are settled, APS operation would draw liquid propellant directly through the zero-g reservoir without the entrance of vapor. When propellants in the MPS tanks are not settled, APS operation would draw vapor into the zero-g reservoir where it would be trapped by the compartment screens. On orbit, Tug acceleration levels are not great enough to allow venting of the vapor through the screens by liquid displacement. Thus, the zero-g reservoir must be large enough to accumulate any vapor which might enter during adverse acceleration maneuvers. The volumes computed were 0.062 m^3 (2.2 ft^3) and 0.48 m^3 (17 ft^3) for the LOX and LH_2 reservoirs, respectively, with either Mission A or B. Although simple and completely passive, the vapor accumulation concept was not selected because it is heavy and lacks the mission flexibility characteristics of a refillable design.

The second nonvented capillary system is similar to the vapor accumulation concept except that vapor is not allowed to enter the zero-g reservoir. This would be accomplished by extending capillary channels throughout the MPS tanks to transport vapor-free liquid to the reservoir. Performance analysis of this concept showed it to be impractical, however. At best, the extensive channel system is heavy. If the channels are made large enough to provide an adequate flow rate, their capillary retention force is not great enough to prevent draining during adverse acceleration maneuvers. If smaller channels or screened ducts are used, then an excessive number or an impractically fine mesh size are required to provide a satisfactory flow without ingesting vapor.

To consider a lighter weight approach to propellant acquisition, refillable vented concepts were synthesized and evaluated. The most obvious approach to zero-g reservoir refill is to vent the accumulated vapor back into the MPS tank (internal venting). The advantage of internal venting as compared to overboard venting is in recovery of the vapor and any entrained liquid or liquid overflow.

The simplest refillable design uses a passive screened vent (Candidate I-1, Figure 5-1). When propellant is settled by Tug +X velocity maneuvers, vapor is forced through the screened vent by hydrostatic head. A standpipe can be used to provide additional head pressure. During adverse acceleration, the wetted screens at the reservoir inlet and vent prevent the entrance of vapor and thus the loss of liquid. A detailed analysis of this concept, including predicted static head levels, screen mesh sizes, and refill flow rates, is presented in Section 5.3. In that analysis, comparison of the predicted and required refill rates reveals that insufficient time is available for refill after propellants have been settled during planned Tug linear translations.

Analysis of a valve vented concept (Candidate I-4, Figure 5-1) showed higher but still inadequate refill rates for reasonable valve sizes. The addition of a pump to increase the refill flow as shown for Candidate I-2 provides an adequate refill time but adds significantly to the system complexity in terms of either an extra pump or additional valving to utilize the APS feed pump. For these reasons, internally vented refill of the zero-g reservoir was not selected for the integrated APS.

The best way to accompiish reservoir refill was found to be by use of an overboard vent (Candidate I-5). The amount of vapor lost is minimal, only 0.95 kg (2.1 lb) per mission. The higher, flow-driving, pressure differential allows complete refill during any programmed APS or MPS burn of either Mission A or B. In the event of a more extreme mission for which attitude control demands are greater, or zero-g coast periods are longer, the mission profile can be modified to include a special settling maneuver for the specific purpose of refilling the APS reservoir. A more detailed design and operational description of this concept is presented in the next section.

PROPELLANT FEED

After selecting an overboard-vent refillable capillary system for propellant acquisition, attention was directed toward the options available for propellant feed to the thrusters. Trade studies conducted for the dedicated cryogenic systems narrowed the choice down to two basic conepts: pump feed and pressure feed. Analysis conducted in support of integrated cryogenic APS included a third alternative: a hybrid concept utilizing pump feed for LH_2 and pressure feed for LOX. The hybrid concept was considered to have potential merit because previous analysis of the dedicated APS revealed that the major weight penalty for pressure feed is attributable to the low density of hydrogen propellant, and the high density of helium at liquid hydrogen temperature and the engine feed pressure, 152 N/cm^2 (220 psia). Similar hybrid concepts have been proposed for Tug by others (References 9 and 10).

Mechanical schematics and detailed weight statements were prepared for each of the three propellant feed systems under consideration. The I-7 schematic is shown in Figure 5-2. Detailed weights for the pressure feed concept (I-6) and the hybrid concept (I-7) are presented for both Missions A and B in Tables 5-1 through 5-4.

The schematics and detailed weights for the pump feed concept (I-5) are presented in the next section.

The results of the weight and performance comparison are summarized in Table 5-5. These data include a 13 percent contingency for weight and growth. Power supply options shown in table are discussed in Section 5.2. The pump feed concept (I-5), using either power option, is the selected integrated cryogenic APS design.

As can be seen from the data, the pressure feed (I-6) concept is considerably heavier than the pump feed (I-5) concept, especially for the higher total impulse requirement of Mission B. The hybrid concept (I-7) is shown schematically on Figure 5-2. Its design and operation are the same as for Candidate I-5 on the hydrogen side and Candidate I-6 on the LOX side. For either concept, I-6 or I-7, pressure feed dictates pressure isolation valves between the pressure-fed zero-g reservoir and the corresponding MPS tank. Moreover, the pressure-fed accumulator must be large enough to sustain APS operation during reservoir refill. Since direct through flow from the MPS tanks to the thrusters is not possible, the pressure-fed zero-g reservoirs must be larger than for pump feed. Mission-required APS impulse also controls the size of pressure-fed reservoirs and helium tanks. For this reason the pressure-fed refillable concepts, although somewhat better than dedicated

Table 5-1. APS Weight Summary for Candidate I-6 (Mission A)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG
PRESSURIZATION SYSTEM		(30)			(129.4)	(57.1)
HE TANK	4	1	84.9	38.5	84.9	38.5
HE REG ISO VALVE	36	3	2.0	0.9	6.0	2.7
HE REGULATOR	7	3	1.5	0.7	4.5	2.0
PRESSURE SWITCH	37	6	1.0	0.5	6.0	2.7
LOX RESERVOIR HE SOL	61	4	1.5	0.7	6.0	2.7
LOX ACCUMULATOR HE SOL	62	4	1.5	0.7	6.0	2.7
LH2 ACCUMULATOR HE SOL	64	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR HE SOL	63	4	1.5	0.7	6.0	2.7
LOX SYSTEM HE HEATER	10	1	0.5	0.2	0.5	0.2
PROPELLANT CONTROL SYSTEM		(8)			(16.8)	(7.6)
LOX RESERVOIR CAP DEV	11	1	1.3	0.6	1.3	0.6
LH2 RESERVOIR CAP DEV	13	1	5.6	2.5	5.6	2.5
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7
LH2 BLEED SHUTOFF VLV	38	1	1.5	0.7	1.5	0.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2
LOX COOLING COIL	50	1	1.1	0.5	1.1	0.5
LH2 COOLING COIL	50	1	3.8	1.7	3.8	1.7
PROPELLANT FEED SYSTEM		(34)			(60.3)	(27.4)
LOX RESERVOIR	17	1	2.8	1.3	2.8	1.3
LOX RESERVOIR INSULATION	17	1	0.9	0.4	0.9	0.4
LH2 RESERVOIR	19	1	7.9	3.6	7.9	3.6
LH2 RESERVOIR INSULATION	19	1	2.8	1.3	2.8	1.3
LOX RESERVOIR FILL SOL	65	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR FILL SOL	66	4	1.5	0.7	6.0	2.7
LOX QUAD ISO VALVE	18	4	3.0	1.4	12.0	5.4
LH2 QUAD ISO VALVE	20	4	3.0	1.4	12.0	5.4
LOX FEED CHECK VALVE	40	4	0.3	0.1	1.2	0.5
LH2 FEED CHECK VALVE	45	4	0.3	0.1	1.2	0.5
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0
LOX ACCUM RELIEF SOL	42	1	2.0	0.9	2.0	0.9
LH2 ACCUM RELIEF SOL	47	1	2.0	0.9	2.0	0.9
LOX ACCUM RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1
LH2 ACCUM RELIEF ORIFICE	58	1	0.3	0.1	0.3	0.1
OVERBOARD VENT SYSTEM		(14)			(14.6)	(6.6)
LOX ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LH2 ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LOX RESERVOIR VENT VALVE	24	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR VENT VALVE	28	4	1.5	0.7	6.0	2.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1
THRUSTER QUAD (50 AR)	29	(4)	17.9	8.1	(71.6)	(32.5)
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)
COMPONENT TOTAL		133			306.4	139.0
LINES					27.3	12.4
INSULATION					8.2	3.7
COMPONENT MOUNTINGS					15.3	6.9
DRY SYSTEM					357.2	162.0

Table 5-2. APS Weight Summary for Candidate I-6 (Mission B)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG
PRESSURIZATION SYSTEM		(30)			(479.5)	(217.5)
HE TANK	4	1	438.5	198.9	438.5	198.9
HE REG ISO VALVE	36	3	2.0	0.9	6.0	2.7
HE REGULATOR	7	3	1.5	0.7	4.5	2.0
PRESSURE SWITCH	37	6	1.0	0.5	6.0	2.7
LOX RESERVOIR HE SOL	61	4	1.5	0.7	6.0	2.7
LOX ACCUMULATOR HE SOL	62	4	1.5	0.7	6.0	2.7
LH2 ACCUMULATOR HE SOL	64	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR HE SOL	63	4	1.5	0.7	6.0	2.7
LOX SYSTEM HE HEATER	10	1	0.5	0.2	0.5	0.2
PROPELLANT CONTROL SYSTEM		(8)			(53.8)	(24.4)
LOX RESERVOIR CAP DEV	11	1	5.7	2.6	5.7	2.6
LH2 RESERVOIR CAP DEV	13	1	23.7	10.5	23.2	10.5
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7
LH2 BLEED SHUTOFF VLV	38	1	1.5	0.7	1.5	0.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2
LOX COOLING COIL	50	1	4.4	2.0	4.4	2.0
LH2 COOLING COIL	50	1	15.5	7.0	15.5	7.0
PROPELLANT FEED SYSTEM		(34)			(150.3)	(69.2)
LOX RESERVOIR	17	1	23.7	10.8	23.7	10.8
LOX RESERVOIR INSULATION	17	1	3.6	1.6	3.6	1.6
LH2 RESERVOIR	19	1	66.1	30.0	66.1	30.0
LH2 RESERVOIR INSULATION	19	1	11.0	5.0	11.0	5.0
LOX RESERVOIR FILL SOL	65	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR FILL SOL	66	4	1.5	0.7	6.0	2.7
LOX QUAD ISO VALVE	18	4	3.0	1.4	12.0	5.4
LH2 QUAD ISO VALVE	20	4	3.0	1.4	12.0	5.4
LOX FEED CHECK VALVE	40	4	0.3	0.1	1.2	0.5
LH2 FEED CHECK VALVE	45	4	0.3	0.1	1.2	0.5
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0
LOX ACCUM RELIEF SOL	42	1	2.0	0.9	2.0	0.9
LH2 ACCUM RELIEF SOL	47	1	2.0	0.9	2.0	0.9
LOX ACCUM RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1
LH2 ACCUM RELIEF ORIFICE	58	1	0.3	0.1	0.3	0.1
OVERBOARD VENT SYSTEM		(14)			(14.6)	(6.6)
LOX ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LH2 ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LOX RESERVOIR VENT VALVE	24	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR VENT VALVE	28	4	1.5	0.7	6.0	2.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1
THRUSTER QUAD (200 AR)	29	(4)	19.6	8.9	(78.4)	(35.6)
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)
COMPONENT TOTAL		133			793.8	360.1
LINES					27.3	12.4
INSULATION					8.2	3.7
COMPONENT MOUNTINGS					39.7	18.0
DRY SYSTEM					869.0	394.2

Table 5-3. APS Weight Summary for Candidate I-7 (Mission A)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG
FILL & DRAIN SYSTEM		(1)			(2.0)	(0.9)
LH2 DRAIN VALVE	0	1	2.0	0.9	2.0	0.9
PRESSURIZATION SYSTEM		(26)			(39.0)	(17.7)
HE TANK	4	1	4.0	1.8	4.0	1.8
HE REG ISO VALVE	36	3	2.0	0.9	6.0	2.7
PRESSURE SWITCH	37	6	1.0	0.5	6.0	2.7
HE REGULATOR	7	3	1.5	0.7	4.5	2.0
LOX RESERVOIR HE SOL	61	4	1.5	0.7	6.0	2.7
LOX ACCUMULATOR HE SOL	62	4	1.5	0.7	6.0	2.7
LH2 ACCUMULATOR HE SOL	36	2	2.0	0.9	4.0	1.8
LH2 ACCUMULATOR PRESS SW	37	2	1.0	0.5	2.0	0.9
LOX HE HEATER	10	1	0.5	0.2	0.5	0.2
PROPELLANT CONTROL SYSTEM		(8)			(14.4)	(6.5)
LOX RESERVOIR CAP DEV	11	1	1.3	0.6	1.3	0.6
LH2 RESERVOIR CAP DEV	13	1	4.2	1.9	4.2	1.9
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7
LH2 BLEED SHUTOFF VLV	35	1	1.5	0.7	1.5	0.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2
LOX COOLING COIL	50	1	1.1	0.5	1.1	0.5
LH2 COOLING COIL	50	1	2.8	1.3	2.8	1.3
PROPELLANT FEED SYSTEM		(29)			(65.2)	(29.6)
LOX RESERVOIR/INSULATION	17	1	2.7	1.2	2.7	1.2
LH2 RESERVOIR/INSULATION	19	1	5.6	2.5	5.6	2.5
LOX RESERVOIR FILL SOL	65	4	1.5	0.7	6.0	2.7
LOX QUAD ISO VALVE	18	4	3.0	1.4	12.0	5.4
LH2 QUAD ISO VALVE	20	4	3.0	1.4	12.0	5.4
LOX FEED CHECK VALVE	40	4	0.3	0.1	1.2	0.5
LH2 PUMP CHECK VALVE	45	4	0.3	0.1	1.2	0.5
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0
LOX ACCUM RELIEF SOL	42	1	2.0	0.9	2.0	0.9
LH2 ACCUM RELIEF SOL	47	1	2.0	0.9	2.0	0.9
LOX ACCUM RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1
LH2 ACCUM RELIEF ORIFICE	58	1	0.3	0.1	0.3	0.1
LH2 PUMP	44	1	17.0	7.7	17.0	7.7
OVERBOARD VENT SYSTEM		(14)			(14.6)	(6.6)
LOX ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LH2 ACCUM HE VENT VLV	5	2	0.5	0.2	1.0	0.5
LOX RESERVOIR VENT VALVE	24	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR VENT VALVE	28	4	1.5	0.7	6.0	2.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1
THRUSTER QUAD (50 AR)	29	(4)	17.9	8.1	(71.6)	(32.5)
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)
PUMP PWR SUPP - APS CHARGE		(2)			(23.9)	(10.8)
INVERTER	0	1	9.7	4.4	9.7	4.4
BATTERY	0	1	14.2	6.4	14.2	6.4
COMPONENT TOTAL		127			247.9	112.4
LINES					27.3	12.4
INSULATION					8.2	3.7
COMPONENT MOUNTINGS					12.4	5.6
DRY SYSTEM					295.8	134.2

Table 5-4. APS Weight Summary for Candidate I-7 (Mission B)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG
FILL & DRAIN SYSTEM		(1)			(2.0)	(0.9)
LH2 DRAIN VALVE	0	1	2.0	0.9	2.0	0.9
PRESSURIZATION SYSTEM		(26)			(57.4)	(26.0)
HE TANK	4	1	22.4	10.2	22.4	10.2
HE REG ISO VALVE	36	3	2.0	0.9	6.0	2.7
PRESSURE SWITCH	37	6	1.0	0.5	6.0	2.7
HE REGULATOR	7	3	1.5	0.7	4.5	2.0
LOX RESERVOIR HE SOL	61	4	1.5	0.7	6.0	2.7
LOX ACCUMULATOR HE SOL	62	4	1.5	0.7	6.0	2.7
LH2 ACCUMULATOR HE SOL	36	2	2.0	0.9	4.0	1.8
LH2 ACCUMULATOR PRESS SW	37	2	1.0	0.5	2.0	0.9
LOX HE HEATER	10	1	0.5	0.2	0.5	0.2
PROPELLANT CONTROL SYSTEM		(8)			(22.1)	(10.0)
LOX RESERVOIR CAP DEV	11	1	5.7	2.6	5.7	2.6
LH2 RESERVOIR CAP DEV	13	1	4.2	1.9	4.2	1.9
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7
LH2 BLEED SHUTOFF VLV	38	1	1.5	0.7	1.5	0.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2
LOX COOLING COIL	50	1	4.4	2.0	4.4	2.0
LH2 COOLING COIL	50	1	2.8	1.3	2.8	1.3
PROPELLANT FEED SYSTEM		(29)			(81.2)	(36.8)
LOX RESERVOIR/INSULATION	17	1	18.7	8.5	18.7	8.5
LH2 RESERVOIR/INSULATION	19	1	5.6	2.5	5.6	2.5
LOX RESERVOIR FILL SOL	65	4	1.5	0.7	6.0	2.7
LOX QUAD ISO VALVE	19	4	3.0	1.4	12.0	5.4
LH2 QUAD ISO VALVE	20	4	3.0	1.4	12.0	5.4
LOX FEED CHECK VALVE	40	4	0.3	0.1	1.2	0.5
LH2 PUMP CHECK VALVE	45	4	0.3	0.1	1.2	0.5
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0
LOX ACCUM RELIEF SOL	42	1	2.0	0.9	2.0	0.9
LH2 ACCUM RELIEF SOL	47	1	2.0	0.9	2.0	0.9
LOX ACCUM RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1
LH2 ACCUM RELIEF ORIFICE	59	1	0.3	0.1	0.3	0.1
LH2 PUMP	44	1	17.0	7.7	17.0	7.7
OVERBOARD VENT SYSTEM		(14)			(14.6)	(6.6)
LOX ACCUM HE VENT VLV	54	2	0.5	0.2	1.0	0.5
LH2 ACCUM HE VENT VLV	54	2	0.5	0.2	1.0	0.5
LOX RESERVOIR VENT VALVE	24	4	1.5	0.7	6.0	2.7
LH2 RESERVOIR VENT VALVE	28	4	1.5	0.7	6.0	2.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1
THRUSTER QUAD (200 AR)	29	(4)	19.6	8.9	(78.4)	(35.6)
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)
PUMP PWR SUPP - APS CHARGE		(4)			(46.0)	(20.9)
INVERTER	0	1	9.7	4.4	9.7	4.4
FUEL CELL	0	2	13.5	6.1	27.0	12.2
FUEL CELL RADIATOR	0	1	9.3	4.2	9.3	4.2
COMPONENT TOTAL		129			318.9	144.6
LINES					27.3	12.4
INSULATION					8.2	3.7
COMPONENT MOUNTINGS					15.9	7.2
DRY SYSTEM					370.3	168.0

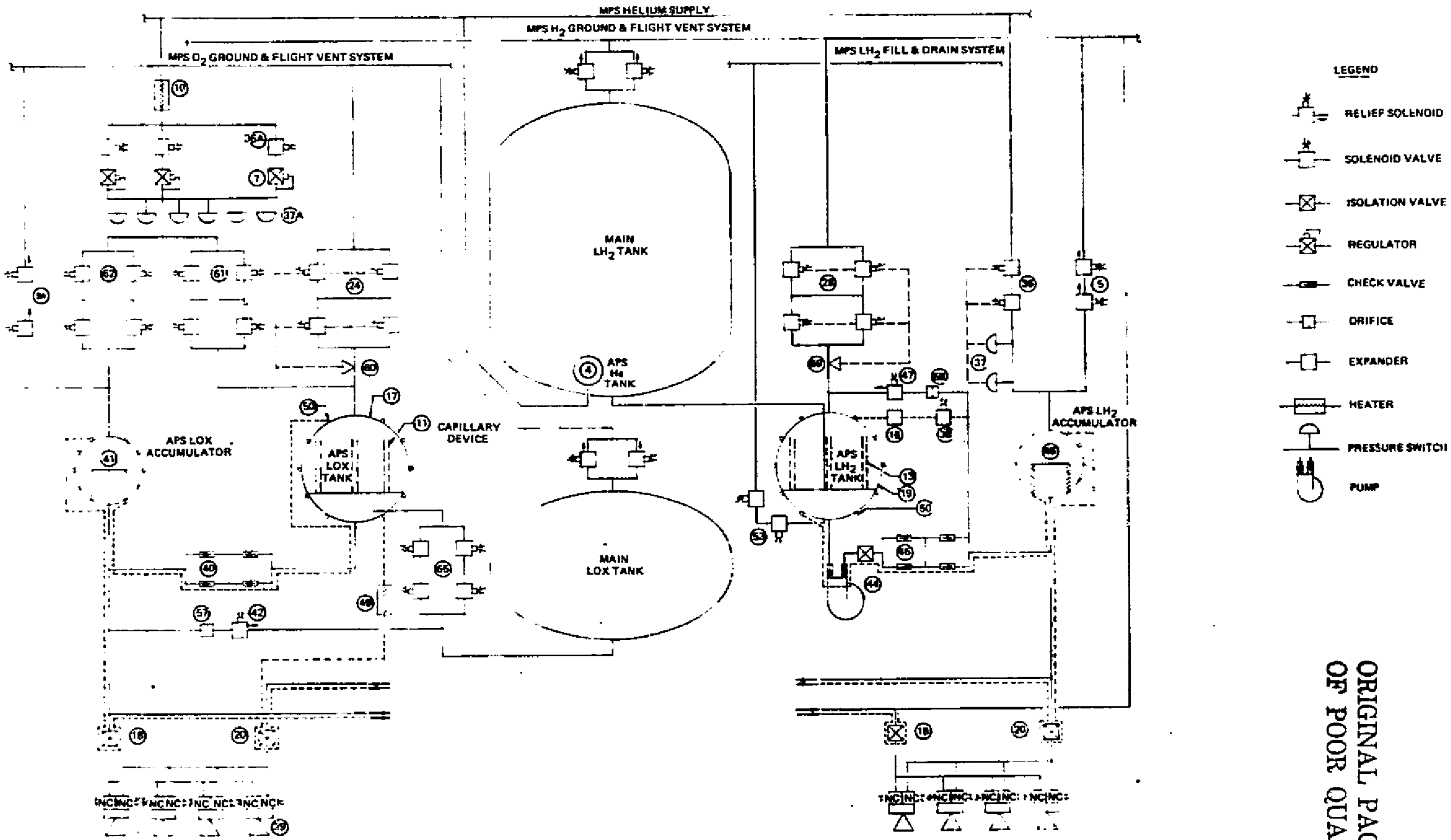


Figure 5-2. LH_2 Pump Feed, LOX Pressure Feed (I-7) Mechanical Diagram

Table 5-5. Integrated Cryogenic APS Propellant Feed Concepts Considered

Candidate	Propellant Feed	Power Supply	Δ Dry Weight Mission A [kg (lb)]	Δ Dry Weight Mission B [kg (lb)]	Evaluation
I-5	Pump	Primary battery	0 (0)	28 (61)	Selected (Mission A baseline)
		Fuel cell	23 (52)	24 (54)	Selected (Mission B baseline)
I-6	Pressure (both LH_2 and LOX)	NA	36 (80)	298 (658)	Heavy, low mission flexibility
I-7	Hybrid (pump LH_2, pressure LOX)	Primary battery	5 (10)	55 (121)	Heavy for Mission B, low mission flexibility
		Fuel cell	14 (32)	42 (93)	

concepts, have little mission flexibility. These influences are shown graphically on Figure 5-3. The pump feed concept (I-5) has the lowest dry weight at total impulse levels above 3.6×10^5 N-sec (0.8×10^5 lb-sec) for a primary battery pump power supply system and above 18.7×10^5 N-sec (4.2×10^5 lb-sec) for a fuel cell pump power supply system.

5.2 INTEGRATED CONCEPT DESCRIPTION

THRUSTER

The thruster selected for the integrated concept is esentially the same as the baseline thruster described in Section 3.4. The only major change in the engine operating point is a reduction in nominal steady-state mixture ratio from 4.0 to 3.0. In addition, for Mission B designs, an area ratio of 200 was selected. The characteristics of the thruster are listed in Table 5-6.

Performance sensitivities to inlet conditions were determined for the new baseline engine, using the same techniques as for the dedicated system analysis. These data show that with a nominal inlet pressure of 152 N/cm^2 (220 psia), safe engine starts can be achieved with hydrogen inlet temperatures up to approximately 39 K (70 R) for oxygen inlet temperatures over a range of

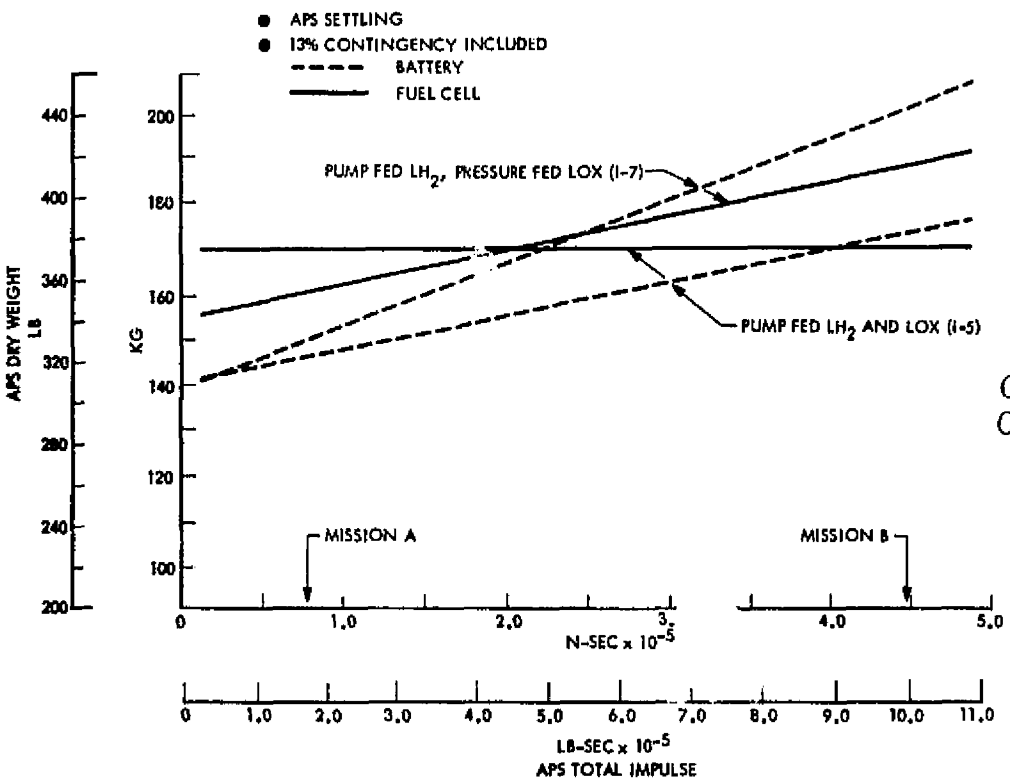


Figure 5-3. APS Dry Weight as a Function of APS Impulse

90 to 111 K (160 to 200 R). The data also indicate that from a thruster viewpoint, it is desirable to maintain or bias the tank pressure controls so that the oxygen inlet pressure is 152 N/cm^2 (220 psia) maximum and the fuel inlet pressure is that same value as a minimum. In doing so, performance variations are minimized. In particular, engine MR is shifted in a favorable direction. The option of implementing this bias should be reconsidered after completion of preliminary thruster development tests.

Pre- and post-burn thermal analysis and impulse bit analysis were also conducted for the new baseline and are presented in the thruster design point discussion in Section 5.3. Those analyses show that a pulsing performance level of 80 percent of steady state is achievable at a minimum impulse bit of 2.23 N-sec (0.5 lb-sec).

The generic failure rates assumed for the thruster components are based on ALRC data and experience for igniters on cryogenic propellants at a higher thrust scale, and on valves for storable propellant service. Experience with the igniter for the 5560 N (1250 lb) Shuttle RCS thruster Contracts NAS3-15850 (ITA) and NAS3-16775 (ETR) has demonstrated that a safe, reliable thruster can be developed to meet the system goals. One thruster/igniter tested at ALRC achieved more than 44,000 cycles while a second thruster delivered to NASA/LeRC was fired successfully 51,000 times.

ALRC experience with small solenoid-actuated propellant valves indicates that they are very reliable. For example, on the 5-lb thruster program, a storable bipropellant valve was used which experienced no failures in

Table 5-6. Thruster Characteristics for Integrated Concepts

Item	Mission A	Mission B
Thrust, N (lb)	111 (25)	
Chamber pressure, N/cm^2 (sec)	103 (150)	
Mixture ratio	3	
Nozzle area ratio	50	200
Specific impulse, N-sec/kg (sec)		
Steady state	3910 (398.7)	4002 (408.1)
Pulse train (cold)	3069 (313.0)	3202 (326.5)
Minimum bit, N-sec (lb-sec)	2.2 (0.5)	
Steady state flow rate, kg/sec (lb/sec)	0.0284 (0.0627)	
Throat diameter, cm (in.)	0.909 (0.358)	
Chamber diameter, cm (in.)	1.90 (0.748)	
Chamber length, cm (in.)	18.2 (7.18)	
Nozzle length, cm (in.)	8.15 (3.21)	18.7 (7.35)
Nozzle exit diameter, cm (in.)	6.60 (2.6)	12.9 (5.1)
Nominal propellant inlet temp, K (R)		
Fuel	28 (50)	
Oxidizer	92 (165)	
Nominal propellant inlet pressure, N/cm^2 (psia)	157 (220)	
Quad weight, kg (lb)		
Thrust chamber assemblies (4)	4.7 (10.4)	5.8 (12.8)
Valves (8)	2.2 (4.8)	2.2 (4.8)
Redundant power supply (1)	1.2 (2.7)	1.2 (2.7)
Total	8.1 (17.9)	9.2 (20.3)

3,425,000 cycles. The upper 50 percent confidence limit on failure rate for this valve is 2×10^{-7} failures per cycle. The following discussion provides the basis for this failure rate.

During the 5-lb thruster program, testing subjected three different engines to 300,000, 50,000, and 50,000 cycles with no failures. The 300,000-cycle series corresponded to the maximum design goal. Additional test data were obtained on the Moog bipropellant valve. These data include one valve cycled 1,000,000 times on the Minuteman program and vendor tests of 27,000 cycles on each of 75 valves. These tests, including the cited engine tests, represent 3.425×10^6 cycles without failure.

DDT&E and first unit costs have been estimated at $5.8 $\bar{M}$ and $75,000, respectively. DDT&E cost is treated as a government-furnished procurement (GFP) in subsequent cost comparisons since a program of that magnitude would ordinarily be purchased in that manner.

The thruster SR&T program required is estimated by this system contractor at $1.8 M over an 18-month program. Little test experience has been obtained for a liquid-liquid O/H thruster and none in the APS thrust level range. Accordingly, a pre-prototype concept confirmation test program which also explores concept potential and defines achievable design characteristics is necessary. The program, as a minimum, should include thrust chamber, igniter, and valve development tests to verify steady state and minimum bit performance, life, and cooling and inlet temperature condition design limits at a baseline chamber pressure. A more extensive SR&T plan would culminate in experimental development, design verification, and peripheral (design limit) testing of a complete thruster quad assembly. Additional specific objectives identified by ALRC include: columbium chamber machining and coating materials experimental investigations; testing/development of thrust chamber cooling; determination of thermal transients and impulse bit; and valve and igniter life testing.

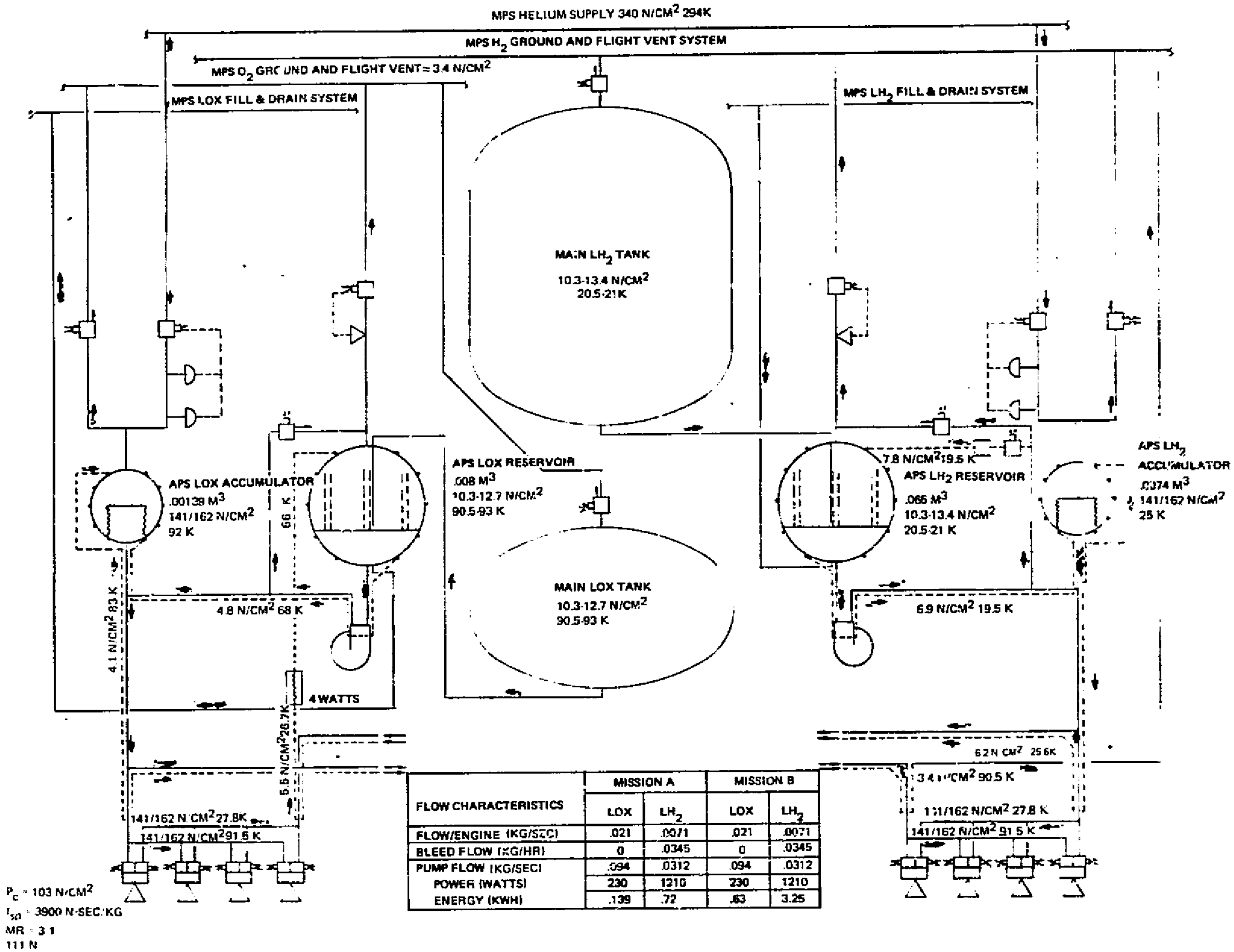
Thruster refurbishment requirements were based on failure rate data. To avoid system reliability degradation, the average replacement quantities per vehicle over 20 missions are computed to be 22 out of the 32 thruster valves and 2 of the 16 igniters (see Tables 5-7 and 6-2). Thrust chamber life exceeds the 20 mission requirements for either Mission B (54 hours) or a main engine backup abort (17 hours).

PROPELLANT SYSTEM AND ELEMENTS

The integrated cryogenic APS concept selected for comparison with the dedicated cryogenic and storable APS has the following basic design characteristics:

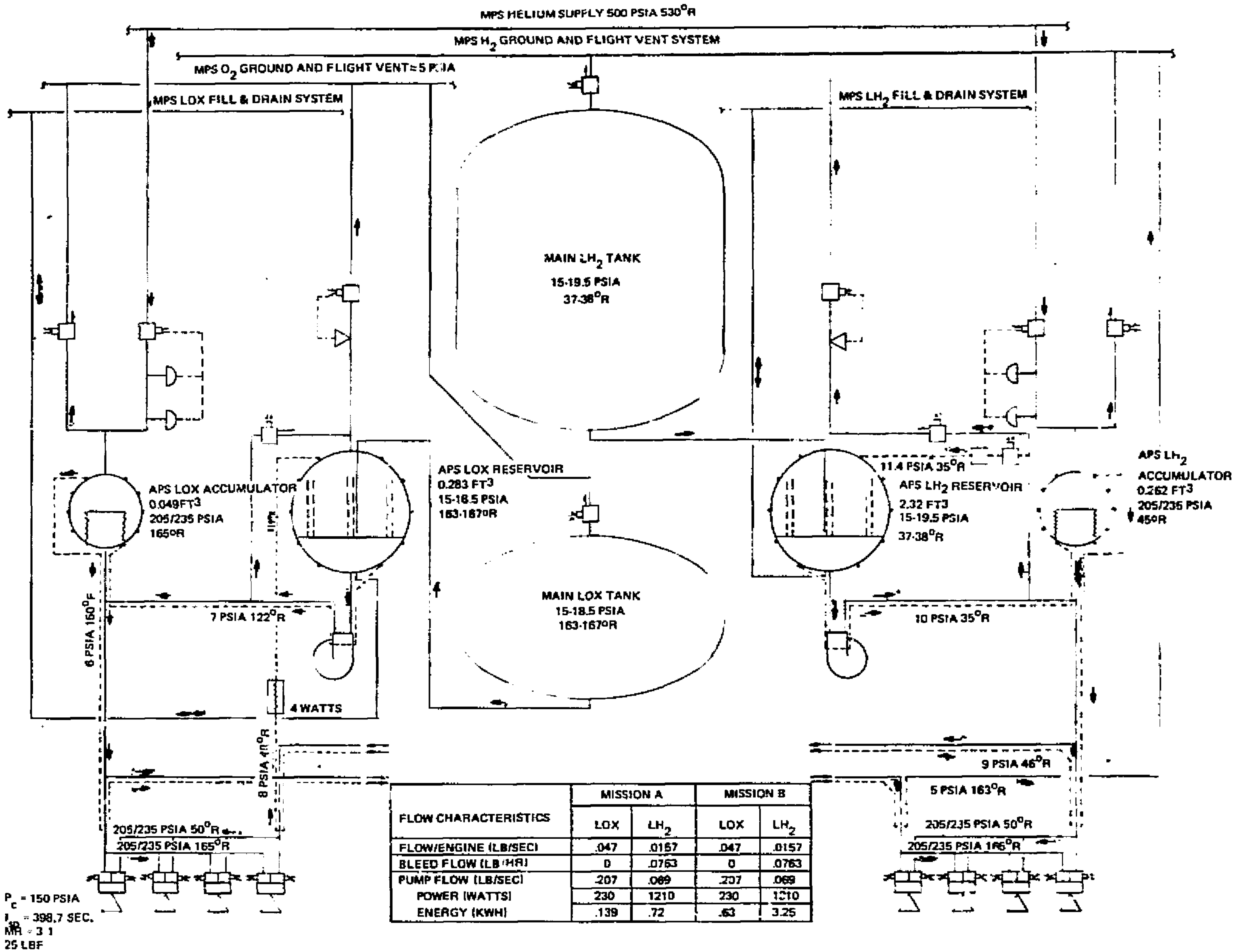
Function	Design
• Propellant acquisition	• Capillary device with overboard vent refill
• Propellant feed	• Pump and accumulator
• Thermodynamic control	• Expanded H_2 bleed feed line tracing and tank wall-mounted cooling coils

This concept has been designated I-5 to distinguish it from the other integrated concepts which were evaluated but eliminated. Process flow and mechanical flow diagrams for Candidate I-5 are shown in Figures 5-4 and 5-5. The system consists of a zero-g reservoir connected to the MPS tanks and containing capillary screens and collector tubes to supply vapor-free propellant to a thruster feed pump. An accumulator is provided downstream of the pump to minimize pump short cycles. The entire system is insulated with multilayer insulation (MLI) to provide a radiation barrier and minimize heat leaks to the cold feedlines, pumps, and tanks. The system heat loads are absorbed by hydrogen bleed flow which is tapped off downstream of the pump and expanded through a Joule-Thompson expander to a pressure of 7.9 N/cm^2 (11.4 psia) and a temperature of 35 K. This cold hydrogen bleed is first routed through cooling coils mounted on the outside of the zero-g reservoir. The bleed then traces the hydrogen feed line manifold, absorbing heat through saddleblock segments brazed between the chill line and feed line. After leaving the hydrogen system, the bleed is electrically heated above the freezing temperature of oxygen and routed along the oxygen system in a manner similar to that for the hydrogen system, after which it is vented overboard through the MPS vent system.



FLOW CHARACTERISTICS	MISSION A		MISSION B	
	LOX	LH_2	LOX	LH_2
FLOW/ENGINE (KG/SEC)	.021	.0071	.021	.0071
BLEED FLOW (KG/HR)	0	.0345	0	.0345
PUMP FLOW (KG/SEC)	.094	.0312	.094	.0312
POWER (WATTS)	230	1210	230	1210
ENERGY (KWH)	.139	.72	.63	3.25

Figure 5-4. Pump Feed, Overboard Vent Refill (I-5) Process Diagram (Metric Units)



FLOW CHARACTERISTICS	MISSION A		MISSION B	
	LOX	LH_2	LOX	LH_2
FLOW/ENGINE (LB/SEC)	.047	.0157	.047	.0157
BLEED FLOW (LB/HR)	0	.0763	0	.0763
PUMP FLOW (LB/SEC)	.207	.069	.207	.069
POWER (WATTS)	230	1210	230	1210
ENERGY (KWH)	.139	.72	.63	3.25

Figure 5-4. Pump Feed, Overboard Vent Refill (I-5) Process Diagram (English Units)

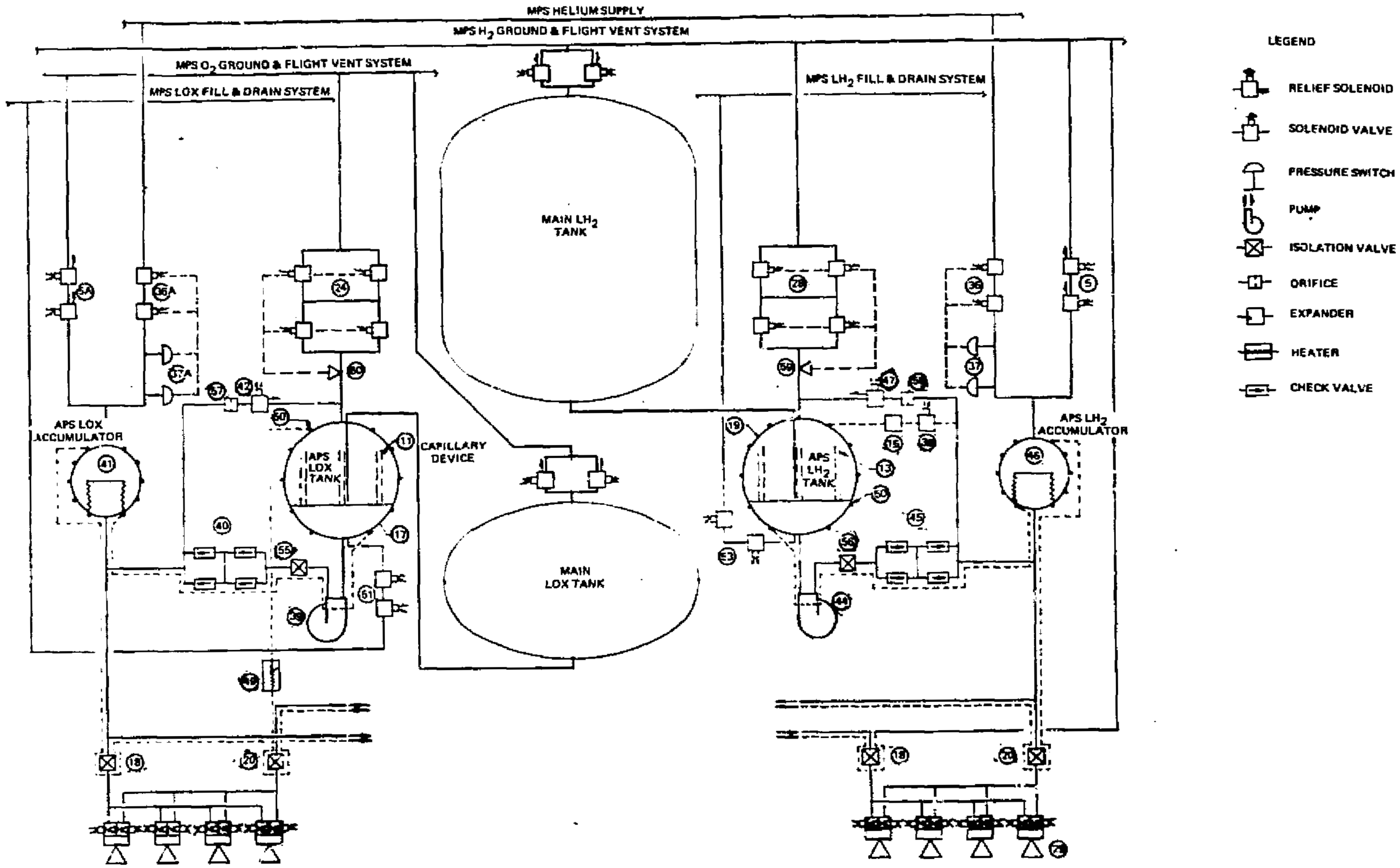


Figure 5-5. Pump Feed, Overboard Vent Refill (I-5) Mechanical Diagram

Each component in the mechanical diagram has a designated identification number shown in an adjacent circle. These numbers allow correlation between the schematic and the APS weight and cost tables presented in Tables 5-7 and 5-8, for Missions A and B, respectively. The tables present the dry weight, DDT&E cost, first unit cost, and single vehicle refurbishment cost estimated for each component. The weight and cost values shown are influenced by APS pump power supplied by primary batteries for Mission A and an increased Tug fuel cell capability for Mission B. These options were selected on the basis of maximum payload capability (discussed later in this section). The DDT&E, first unit, and refurbishments costs were estimated using the same estimating techniques applied to the dedicated systems (see Section 4.3).

Table 5-7. Candidate I-5 Component Weights and Costs (Mission A)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT		SYSTEM WEIGHT		DDT&E $1000.	1ST UNIT $1000.	REFURB $1000.
			LB	KG	LB	KG			
FILL AND DRAIN SYSTEM		(2)			(4.0)	(1.8)	(10.0)	(17.4)	(0.0)
LOX DRAIN VALVE	0	1	2.0	0.9	2.0	0.9	5.0	8.7	0.0
LH2 DRAIN VALVE	0	1	2.0	0.9	2.0	0.9	5.0	8.7	0.0
PRESSURIZATION SYSTEM		(8)			(12.0)	(5.4)	(40.5)	(69.5)	(17.4)
HELIUM ISO VALVE - LOX	36	2	2.0	0.9	4.0	1.8	23.4	17.4	8.7
HELIUM ISO VALVE - LH2	36	2	2.0	0.9	4.0	1.8	0.0	17.4	8.7
PRESSURE SWITCH - LOX	37	2	1.0	0.5	2.0	0.9	17.1	17.4	0.0
PRESSURE SWITCH - LH2	37	2	1.0	0.5	2.0	0.9	0.0	17.4	0.0
PROPELLANT CONTROL SYSTEM		(8)			(13.3)	(6.0)	(326.2)	(54.5)	(17.4)
LOX TANK CAPILLARY DEV	11	1	0.7	0.3	0.7	0.3	102.7	15.8	0.0
LH2 TANK CAPILLARY DEV	13	1	4.2	1.9	4.2	1.9	134.2	15.8	0.0
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7	36.0	8.7	8.7
LH2 BLEED SHUTOFF VLV	38	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7	13.4	1.6	0.0
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2	13.0	0.8	0.0
LOX EXT COOLING COILS	50	1	0.6	0.3	0.6	0.3	13.4	1.6	0.0
LH2 EXT COOLING COILS	50	1	2.8	1.3	2.8	1.3	13.4	1.6	0.0
PROPELLANT FEED SYSTEM		(30)			(77.7)	(35.2)	(1432.6)	(308.4)	(309.4)
LOX TANK	17	1	1.2	0.5	1.2	0.5	75.3	23.7	0.0
LOX TANK INSULATION	0	1	0.5	0.2	0.5	0.2	58.6	15.8	15.8
LH2 TANK	19	1	5.1	2.3	5.1	2.3	66.8	31.6	0.0
LH2 TANK INSULATION	0	1	0.5	0.2	0.5	0.2	50.1	23.7	23.7
LOX TANK ISOLATION VALVE	55	1	3.0	1.4	3.0	1.4	0.0	8.7	0.0
LH2 TANK ISOLATION VALVE	56	1	3.0	1.4	3.0	1.4	0.0	8.7	0.0
LOX FEED & ISOLATION VLV	18	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LH2 FEED & ISOLATION VLV	20	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LOX PUMP & DRIVE	39	1	13.5	6.1	13.5	6.1	351.4	9.5	9.5
LH2 PUMP & DRIVE	44	1	17.0	7.7	17.0	7.7	702.9	19.0	19.0
LOX PUMP CHECK VALVE	40	4	0.3	0.1	1.2	0.5	27.2	15.2	79.6
LH2 PUMP CHECK VALVE	45	4	0.3	0.1	1.2	0.5	0.0	15.2	79.6
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3	50.1	23.7	23.7
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0	50.1	23.7	23.7
LOX RELIEF SOLENOID	42	1	2.0	0.9	2.0	0.9	0.0	8.7	8.7
LH2 RELIEF SOLENOID	47	1	2.0	0.9	2.0	0.9	0.0	8.7	8.7
LOX RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1	0.0	1.6	0.0
LH2 RELIEF ORIFICE	58	1	0.3	0.1	0.3	0.1	0.0	1.6	0.0
OVERBOARD VENT		(14)			(14.6)	(6.6)	(157.5)	(205.4)	(90.1)
LOX-HELIUM VENT VALVE	5	2	0.5	0.2	1.0	0.5	123.5	63.2	31.6
LH2-HELIUM VENT VALVE	5	2	0.5	0.2	1.0	0.5	0.0	63.2	31.6
LOX VENT SOLENOIDS	24	4	1.5	0.7	6.0	2.7	0.0	34.8	8.7
LH2 VENT SOLENOIDS	28	4	1.5	0.7	6.0	2.7	0.0	34.8	8.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1	0.0	4.7	4.7
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1	34.0	4.7	4.7
THRUSTER QUAD (50 AR)		(68)			(71.6)	(32.5)	(5810.2)	(1238.7)	(183.3)
THRUST CHAMBER & NOZZLE	29	16	2.6	1.2	41.6	18.9	5810.2	733.1	0.0
THRUSTER VALVES	0	32	0.6	0.3	19.2	8.7	0.0	252.8	173.8
IGNITER	0	16	0.2	0.1	3.2	1.5	0.0	75.8	9.5
EXCITER	0	4	1.9	0.9	7.6	3.4	0.0	177.0	0.0
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)	(19.6)	(54.4)	(6.3)
PUMP PWR SUPP - APS CHARGE		(2)			(28.5)	(12.9)	(41.9)	(12.6)	(94.8)
INVERTER	0	1	11.3	5.1	11.3	5.1	41.9	7.9	0.0
PRIMARY BATTERY	0	1	17.2	7.8	17.2	7.8	0.0	4.7	94.8
COMPONENT TOTAL		175			238.9	108.4	7838.6	1960.9	718.5
LINES					27.3	12.4			
INSULATION					8.2	3.7			
COMPONENT MOUNTINGS					11.9	5.4			
DRY SYSTEM					286.3	129.9			

Table 5-8. Candidate I-5 Component Weights and Costs (Mission B)

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG	DDT&E $1000.	1ST UNIT $1000.	REFURB $1000.
FILL AND DRAIN SYSTEM		(2)			(4.0)	(1.8)	(10.0)	(17.4)	(0.0)
LOX DRAIN VALVE	0	1	2.0	0.9	2.0	0.9	5.0	8.7	0.0
LH2 DRAIN VALVE	0	1	2.0	0.9	2.0	0.9	5.0	8.7	0.0
PRESSURIZATION SYSTEM		(8)			(12.0)	(5.4)	(40.5)	(69.5)	(17.4)
HELIUM ISO VALVE - LOX	36	2	2.0	0.9	4.0	1.8	23.4	17.4	8.7
HELIUM ISO VALVE - LH2	36	2	2.0	0.9	4.0	1.8	0.0	17.4	8.7
PRESSURE SWITCH - LOX	37	2	1.0	0.5	2.0	0.9	17.1	17.4	0.0
PRESSURE SWITCH - LH2	37	2	1.0	0.5	2.0	0.9	0.0	17.4	0.0
PROPELLANT CONTROL SYSTEM		(8)			(13.3)	(6.0)	(326.2)	(54.5)	(17.4)
LOX TANK CAPILLARY DEV	11	1	0.7	0.3	0.7	0.3	102.7	15.8	0.0
LH2 TANK CAPILLARY DEV	13	1	4.2	1.9	4.2	1.9	134.2	15.8	0.0
LH2 BLEED RETURN SOL	15	1	1.5	0.7	1.5	0.7	36.0	8.7	8.7
LH2 BLEED SHUTOFF VLV	38	1	1.5	0.7	1.5	0.7	0.0	8.7	8.7
LH2 BLEED EXPANDER	16	1	1.5	0.7	1.5	0.7	13.4	1.6	0.0
LH2 BLEED HEATER	49	1	0.5	0.2	0.5	0.2	13.0	0.8	0.0
LOX EXT COOLING COILS	50	1	0.6	0.3	0.6	0.3	13.4	1.6	0.0
LH2 EXT COOLING COILS	50	1	2.8	1.3	2.8	1.3	13.4	1.6	0.0
PROPELLANT FEED SYSTEM		(30)			(77.7)	(35.2)	(1432.6)	(300.4)	(309.6)
LOX TANK	17	1	1.2	0.5	1.2	0.5	75.3	23.7	6.0
LOX TANK INSULATION	0	1	0.5	0.2	0.5	0.2	58.6	15.8	15.8
LH2 TANK	19	1	5.1	2.3	5.1	2.3	66.8	31.6	0.0
LH2 TANK INSULATION	0	1	0.5	0.2	0.5	0.2	50.1	23.7	23.7
LOX TANK ISOLATION VALVE	55	1	3.0	1.4	3.0	1.4	0.0	8.7	0.0
LH2 TANK ISOLATION VALVE	56	1	3.0	1.4	3.0	1.4	0.0	8.7	0.0
LOX FEED & ISOLATION VLV	18	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LH2 FEED & ISOLATION VLV	20	4	3.0	1.4	12.0	5.4	0.0	34.8	8.7
LOX PUMP & DRIVE	39	1	13.5	6.1	13.5	6.1	351.4	9.5	9.5
LH2 PUMP & DRIVE	44	1	17.0	7.7	17.0	7.7	702.9	19.0	19.0
LOX PUMP CHECK VALVE	40	4	0.3	0.1	1.2	0.5	27.2	15.2	79.6
LH2 PUMP CHECK VALVE	45	4	0.3	0.1	1.2	0.5	0.0	15.2	79.6
LOX ACCUMULATOR/BELLOWS	41	1	0.7	0.3	0.7	0.3	50.1	23.7	23.7
LH2 ACCUMULATOR/BELLOWS	46	1	2.2	1.0	2.2	1.0	50.1	23.7	23.7
LOX RELIEF SOLENOID	42	1	2.0	0.9	2.0	0.9	0.0	8.7	8.7
LH2 RELIEF SOLENOID	47	1	2.0	0.9	2.0	0.9	0.0	8.7	8.7
LOX RELIEF ORIFICE	57	1	0.3	0.1	0.3	0.1	0.0	1.6	0.0
LH2 RELIEF ORIFICE	58	1	0.3	0.1	0.3	0.1	0.0	1.6	0.0
OVERBOARD VENT		(14)			(14.6)	(6.6)	(157.5)	(205.4)	(90.1)
LOX-HELIUM VENT VALVE	5	2	0.5	0.2	1.0	0.5	123.5	63.7	31.6
LH2-HELIUM VENT VALVE	5	2	0.5	0.2	1.0	0.5	0.0	63.2	31.6
LOX VENT SOLENOIDS	24	4	1.5	0.7	6.0	2.7	0.0	34.8	8.7
LH2 VENT SOLENOIDS	28	4	1.5	0.7	6.0	2.7	0.0	34.8	8.7
LOX VENT LIQ POINT SENS	60	1	0.3	0.1	0.3	0.1	0.0	4.7	4.7
LH2 VENT LIQ POINT SENS	59	1	0.3	0.1	0.3	0.1	34.0	4.7	4.7
THRUSTER QUAD (200 AR)		(68)			(81.2)	(36.8)	(5010.7)	(1251.4)	(183.3)
THRUST CHAMBER & NOZZLE	29	16	3.2	1.5	51.2	23.2	5010.7	745.8	0.0
THRUSTER VALVES	0	32	0.6	0.3	19.2	8.7	0.0	252.8	173.8
IGNITER	0	16	0.2	0.1	3.2	1.5	0.0	75.8	9.5
EXCITER	0	4	1.9	0.9	7.6	3.4	0.0	177.0	0.0
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)	(19.6)	(54.4)	(6.3)
PUMP PWR SUPP - APS CHARGE		(4)			(54.7)	(24.8)	(41.9)	(24.5)	(0.0)
INVERTER	0	1	11.3	5.1	11.3	5.1	41.9	7.9	0.0
FUEL CELL	0	2	16.2	7.3	32.4	14.7	0.0	15.8	0.0
FUEL CELL RADIATOR	0	1	11.0	5.0	11.0	5.0	0.0	0.8	0.0
COMPONENT TOTAL		177			274.7	124.6	7839.0	1985.4	623.8
LINES					27.3	12.4			
INSULATION					8.2	3.7			
COMPONENT MOUNTINGS					13.7	6.2			
DRY SYSTEM					323.9	146.9			

Tables 5-9 and 5-10 present Tug stage weight statements for Missions A and B. The payload weights are based on performance as shown in the mission timelines of Tables 5-11 and 5-12. Together with reliability assessments, this weight and cost data comprise the primary basis for comparison with the selected dedicated systems and storable APS designs.

Table 5-13 summarizes the major reliability drivers for the I-5 integrated cryogenic APS concept. The table lists failure modes, the components involved, their generic failure rate, and a definition of their redundancy arrangement for each of the components causing a significant contribution to the total number of failures in the APS per Tug mission. The summation of the failure contributions (X 10^{-6}) subtracted from 1.0 yields a 0.9935 predicted

Table 5-9. Concept I-5 Stage Weight Statement (Mission A)

DESCRIPTION	WEIGHT LB	WEIGHT KG
STRUCTURE	(2096.)	(951.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1408.)	(639.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	286.	130.
DRY WEIGHT	4910.	2227.
CONTINGENCY (13%)	638.	289.
TOTAL DRY SYSTEM	5548.	2517.
NONUSUABLE FLUIDS	(536.)	(243.)
APS TRAPPED PROPELLANT	10.	5.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	372.	169.
FLIGHT RESERVES	(321.)	(146.)
APS RESERVE*	3.	1.
MPS RESERVE	318.	144.
BURNOUT WEIGHT	6405.	2905.
EXPENDED FLUIDS	(50619.)	(22960.)
APS USUABLE PROPELLANT	452.	205.
APS LH2 BLEED OVERBOARD	13.	6.
MPS USABLE PROPELLANT	49871.	22621.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	133.	60.
PAYLOAD	(5373.)	(2437.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

Table 5-10. Concept I-5 Stage Weight Statement (Mission B)

DESCRIPTION	WEIGHT LB	WEIGHT KG
STRUCTURE	(2123.)	(963.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1444.)	(655.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	322.	146.
DRY WEIGHT	4973.	2256.
CONTINGENCY (13%)	646.	293.
TOTAL DRY SYSTEM	5619.	2549.
NONUSUABLE FLUIDS	(543.)	(246.)
APS TRAPPED PROPELLANT	10.	5.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	379.	172.
FLIGHT RESERVES	(312.)	(142.)
APS RESERVE*	4.	2.
MPS RESERVE	308.	140.
BURNOUT WEIGHT	6474.	2937.
EXPENDED FLUIDS	(51084.)	(23171.)
APS USUABLE PROPELLANT	2466.	1119.
APS LH2 BLEED OVERBOARD	13.	6.
MPS USABLE PROPELLANT	48318.	21917.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	137.	62.
PAYLOAD	(4839.)	(2195.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

*Includes RSS with MPS reserve of 10% of APS translate and attitude control propellant, and 2% of orbit maneuver delta-V (Mission B).

Table 5-11. Mission Timeline for Propulsion Events - Concept I-5, Battery Option (Mission A)

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0 7	DEPLOY TUG	APS				3	95150	2664	28303	16685	496136
6.38	4	4 38	COAST NO 1							2835	28278	16682	495861
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	163			2	4105	8962	28273	16681	495809
7.74	6	1 19	COAST NO 2							2636	27268	16554	484752
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	2383			2	4363	8986	27266	16553	484732
8.07	8	0 2	MIDCOURSE CORRECT (DV 1)	PIM		14	220526	4	5443	6437	16126	15141	355896
13.13	9	5 6	COAST NO 3							2138	16065	15133	355131
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	1784			4	5440	6533	16060	15132	355071
13.30	11	0 2	ORIENT PAYLOAD	APS				3	36421	1535	10834	14470	284172
13.30	12	0 0	DEPLOY PAYLOAD 1 (812 KG)								10824	14468	284024
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				9	100892	1255	10011	12346	227922
37.16	14	23 45	COAST NO 4							2631	9985	12342	227586
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		83	821001	6	4867	4133	9965	12340	227323
89.34	16	52 9	COAST NO 5							4312	9775	12316	224873
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		83	801617	7	4828	4079	9730	12310	224295
89.41	18	0 2	ORIENT PAYLOAD	APS				3	32086	1203	9544	12286	221876
89.41	19	0 0	DEPLOY PAYLOAD 2 (812 KG)								9535	12285	221765
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				9	87908	885	8723	10163	157532
95.34	21	5 50	COAST NO 6							1279	8700	10160	157279
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		83	716334	7	4582	2877	8695	10153	157221
147.52	23	52 9	COAST NO 7							4683	8529	10138	155370
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		83	696379	7	4541	2835	8484	10132	154871
147.58	25	0 2	ORIENT PAYLOAD	APS				3	27977	840	8322	10112	153062
147.58	26	0 0	DEPLOY PAYLOAD 3 (812 KG)								8314	10111	152979
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				9	75602	467	7502	7988	76940
150.19	28	2 31	COAST NO 8							681	7482	7986	76740
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	1781			8	4278	1494	7480	7985	76716
155.44	30	5 10	COAST NO 9							878	5048	7677	51768
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		12	61337	10	3687	994	5043	7676	51721
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	1690			10	3680	1027	5021	7673	51499
157.63	33	2 5	COAST NO 10							480	3456	7475	35440
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	758			12	3141	722	3454	7475	35421
163.56	35	5 53	COAST NO 11							973	2917	7407	29918
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	24458	201	2912	7406	29864
163.59	37	0 0	SHUTTLE DEORBIT								2906	7405	29864
164.29	38	0 42	TOUCHDOWN										
TOTALS					8559	358	3317493	126	716043	84800			

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM								65000		
1.75	3	0 7	DEPLOY TUG	APS				10	21391	599	62397	12306	365936
6.38	4	4 38	COAST NO 1							637	62342	12304	365733
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	534			2	4105	2075	62332	12303	365695
7.74	6	1 19	COAST NO 2							593	60116	12209	357540
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	7017			2	4363	2020	60112	12209	357525
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)	PIM		45	49644	4	5443	1447	35553	11167	262499
13.13	9	5 6	COAST NO 3							481	35410	11162	261935
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	5854			4	5440	1469	35407	11161	261891
13.30	11	0 2	ORIENT PAYLOAD	APS				10	8188	345	23884	10672	209597
13.30	12	0 0	DEPLOY PAYLOAD 1 (1791 LB)								23863	10671	209488
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				30	22681	282	22072	9106	168109
37.16	14	23 45	COAST NO 4							592	22014	9103	167861
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		273	184568	6	4867	929	21969	9101	167667
89.34	16	52 9	COAST NO 5							969	21549	9084	165840
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		273	180211	7	4828	917	21451	9079	165434
89.41	18	0 2	ORIENT PAYLOAD	APS				10	7213	270	21041	9062	163650
89.41	19	0 0	DEPLOY PAYLOAD 2 ([illegible] LB)								21022	9061	163567
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				30	19763	199	19231	7496	116191
95.34	21	5 50	COAST NO 6							288	19181	7493	116005
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		273	161038	7	4582	647	19169	7493	115962
147.52	23	52 9	COAST NO 7							1053	18802	7477	114597
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		272	156552	7	4541	637	18704	7473	114229
147.58	25	0 2	ORIENT PAYLOAD	APS				10	6289	189	18346	7458	112894
147.58	26	0 0	DEPLOY PAYLOAD 3 (1791 LB)								18330	7457	112833
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				30	16996	105	16539	5892	56749
150.19	28	2 31	COAST NO 8							153	16496	5890	56601
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	5843			8	4278	336	16491	5890	56584
155.44	30	5 10	COAST NO 9							197	11128	5662	38182
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		40	13789	10	3687	223	11118	[illegible]	38149
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	5545			10	3680	231	11070	[illegible]	37986
157.63	33	2 5	COAST NO 10							108	7616	5513	26140
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2487			12	3141	162	7614	5513	26126
163.56	35	5 53	COAST NO 11							219	6425	5463	22066
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	5498	45	6420	5462	22027
163.59	37	0 0	SHUTTLE DEORBIT								6406	5462	22027
164.29	38	0 42	TOUCHDOWN										
TOTALS					28080	1176	745892	234	160973	19064			

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Table 5-12. Mission Timeline for Propulsion Events - Concept 1-5, Fuel Cell Option (Mission B)

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
---		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0 7	DEPLOY TUG	APS				3	95151	2564	28303	16082	477481
6.38	4	4 38	COAST NO 1							2742	28278	16079	477216
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	163			2	4222	8626	28274	16078	477164
7.75	6	1 19	COAST NO 2							2539	27269	15951	466249
9.00	7	0 15	TRANSFER ORBIT INSERTION	MAIN	2383			2	4469	8659	27267	15951	466229
8.30	8	0 18	MIDCOURSE CORRECT (DV 1)	APS		15	270667			1884	16127	14538	340151
13.12	9	4 50	COAST NO 3							2050	16059	14529	339327
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	1784			5	5482	6252	16054	14529	339272
13.30	11	0 2	ORIENT PAYLOAD	APS				3	36408	1467	10830	13866	271660
13.30	12	0 0	DEPLOY PAYLOAD 1 (732 KG)								10820	13865	271525
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				9	101670	1207	10088	11954	218702
36.14	14	22 44	COAST NO 4							2568	10063	11950	218384
37.19	15	1 3	PHASING ORBIT INSERT (DV 2)	APS		85	935695			1463	10043	11948	216140
88.35	16	51 10	COAST NO 5							4307	9808	11918	215234
89.37	17	1 1	MISSION ORBIT INSERT (DV 3)	APS		85	909670			1436	9764	11912	214684
89.41	18	0 2	ORIENT PAYLOAD	APS				3	32056	1150	9535	11883	211831
89.41	19	0 0	DEPLOY PAYLOAD 2 (732 KG)								9527	11882	211727
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				9	88636	857	8795	9971	152206
94.45	21	4 57	COAST NO 6							1194	8773	9968	151960
95.37	22	0 55	PHASING ORBIT INSERT (DV 4)	APS		85	816932			1126	8768	9967	151911
146.64	23	51 16	COAST NO 7							4671	8563	9941	149665
147.54	24	0 34	MISSION ORBIT INSERT (DV 5)	APS		85	793669			1108	8518	9936	149177
147.58	25	0 2	ORIENT PAYLOAD	APS				3	27968	808	8319	9910	146987
147.58	26	0 0	DEPLOY PAYLOAD 3 (732 KG)								8312	9910	146907
147.66	27	0 5	PAYLOAD SURVEILLANCE	APS				9	76393	471	7580	7998	77744
150.19	28	2 31	COAST NO 8							685	7561	7996	77546
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	1781			7	4264	1511	7559	7995	77523
155.37	30	5 6	COAST NO 9							871	5101	7684	52313
155.47	31	0 6	MIDCOURSE CORRECT (DV 6)	APS		15	85532			347	5096	7683	52267
155.53	32	0 4	PHASING ORBIT INSERTION	MAIN	1690			10	3672	1039	5075	7680	52046
157.62	33	2 5	COAST NO 10							480	3692	7480	35818
157.67	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	758			12	3136	730	3491	7479	35799
163.56	35	5 53	COAST NO 11							973	2948	7411	30236
163.58	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	24720	203	2943	7410	30182
163.58	37	0 0	SHUTTLE DEORBIT								2937	7409	30181
164.28	38	0 42	TOUCHDOWN										
TOTALS					8559	370	3812165	85	595300	59127			

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM								65000		
1.75	3	0 7	DEPLOY TUG	APS				10	21391	576	62397	11862	352177
6.38	4	4 38	COAST NO 1							616	62343	11859	351981
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	534			2	4222	1939	62333	11859	351943
7.75	6	1 19	COAST NO 2							571	60117	11765	343892
8.00	7	0 15	TRANSFER ORBIT INSERTION	MAIN	7817			2	4469	1947	60113	11765	343878
8.30	8	0 18	MIDCOURSE CORRECT (DV 1)	APS		50	60848			423	35554	10723	250886
13.12	9	4 50	COAST NO 3							461	35403	10717	250278
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	5854			5	5482	1406	35393	10716	250237
13.30	11	0 2	ORIENT PAYLOAD	APS				10	8185	330	23875	10227	200369
13.30	12	0 0	DEPLOY PAYLOAD 1 (1613 LB)								23854	10227	200269
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				30	22856	271	22241	8817	161308
36.14	14	22 44	COAST NO 4							577	22185	8814	161074
37.19	15	1 3	PHASING ORBIT INSERT (DV 2)	APS		280	210353			329	22141	8812	160894
88.35	16	51 10	COAST NO 5							968	21623	8790	158751
89.37	17	1 1	MISSION ORBIT INSERT (DV 3)	APS		280	204502			323	21525	8786	158345
89.41	18	0 2	ORIENT PAYLOAD	APS				10	7206	258	21021	8765	156241
89.41	19	0 0	DEPLOY PAYLOAD 2 (1613 LB)								21003	8764	156164
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				30	19926	193	19390	7354	112263
94.45	21	4 57	COAST NO 6							268	19340	7352	112082
95.37	22	0 55	PHASING ORBIT INSERT (DV 4)	APS		280	183654			253	19331	7352	112046
146.64	23	51 16	COAST NO 7							1050	18878	7332	110389
147.54	24	0 54	MISSION ORBIT INSERT (DV 5)	APS		280	178424			249	18780	7328	110029
147.58	25	0 2	ORIENT PAYLOAD	APS				10	6288	182	18341	7310	108413
147.58	26	0 0	DEPLOY PAYLOAD 3 (1613 LB)								18325	7309	108355
147.66	27	0 5	PAYLOAD SURVEILLANCE	APS				30	17174	106	16712	5899	57342
150.19	28	2 31	COAST NO 8							154	16669	5897	57196
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	5843			7	4264	340	16664	5897	57179
155.37	30	5 6	COAST NO 9							196	11245	5667	38585
155.4	31	0 6	MIDCOURSE CORRECT (DV 6)	APS		50	19228			78	11235	5667	38551
155.53	32	0 4	PHASING ORBIT INSERTION	MAIN	5545			10	3672	233	11188	5665	38388
157.62	33	2 5	COAST NO 10							108	7699	5517	26418
157.67	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2488			12	3136	164	7695	5517	26404
163.56	35	5 53	COAST NO 11							219	[illegible]	5466	22301
163.58	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	5557	46	6488	5465	22261
163.58	37	0 0	SHUTTLE DEORBIT								6474	5465	22261
164.28	38	0 42	TOUCHDOWN										
TOTALS					28081	1220	857009	193	133829	15540			

Table 5-13. Reliability Summary

SYSTEM	FAILURE MODE	COMPONENT	GENERIC FAILURE RATE (10^{-6} HR)	REDUNDANCY	FAILURE CONTRIB PER 10^6 MISSIONS
CRYO I-5	THRUST LOSS	THRUSTER (16)	0.76	14 OF 16	
	PROP LOSS	THRUSTER VALVE (32)	4.8	QUAD ISOLATION	694
		ISOLATION VALVE (8)	6.5	(STANDBY)	
		ACCUMULATOR (2)	3.13(1.56)	NONE	4496 (2248)
		TANK (2)	0.2976	NONE	98
		REFILL VENT (8)	9.0	QUAD CONNECTED	27
	PRESS LOSS	He RELIEF VALVE (4)	9.0	STANDBY	43
		He PRESS SWITCH (4)	8.0	NONE	160
	PROP OVER TEMP	BLEED SOLENOID (1)	6.5	NONE	240
		BLEED EXPANDER (1)	0.15	NONE	123
	PROP FEED LOSS	PUMP/MOTOR (2)	3.0	NONE	5
		FEED CHECK VALVE (8)	9.0	QUAD CONNECTED	109
	ALL OTHER		-	-	475
					6470, $\bar{R}$ = .9935 (.9955)

Note: Generic failure rates are from Table 4-9 unless updated in Section 5-2.

reliability compared to a goal of 0.996. Several potential reliability improvement techniques have been identified, however, that lead to the conclusion that the reliability goal is achievable. Some of these techniques are discussed in the following sections; for example, halving the accumulator failure rate by conducting SR&T life cycle tests would result in a reliability of 0.9955.

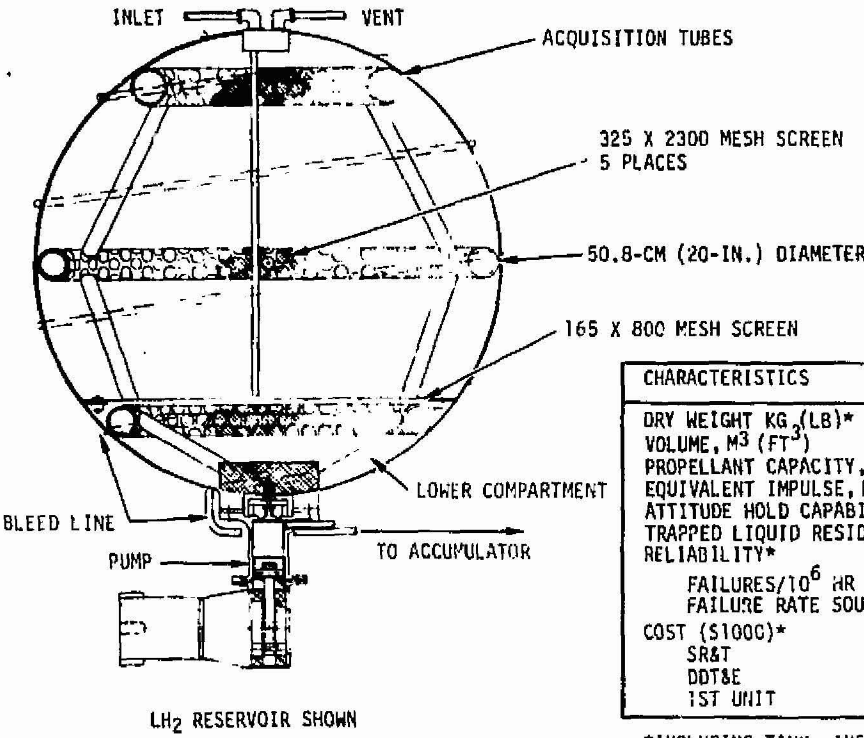
The objective of the APS tankage and feed system is to satisfy continuously the following thruster inlet requirements:

	LOX	LH_2
Pressure, N/cm (psia)		
Minimum	134 (195)	134 (195)
Nominal	151 (220)	151 (220)
Maximum	169 (245)	169 (245)
Temperature, K (R)		
Minimum	90.5 (163)	20.6 (37)
Nominal	91.8 (165	27.8 (50)
Maximum	111 (200)	30.6 (55)

Four functional areas of the system are influenced by these requirements: the zero-g reservoir, pump, accumulator, and the thermodynamic control system. Each of these areas is discussed in terms of its design, operation, performance, reliability, cost and refurbishment requirements, and SR&T goals.

Zero-G Reservoir

A conceptual design of the zero-g reservoir is presented on Figure 5-6 along with a summary of its weight, reliability, and cost characteristics for both LOX and LH_2. The LH_2 reservoir is shown as representative since the



CHARACTERISTICS	LOX	LH_2
DRY WEIGHT KG (LB)*	1.4 (3.0)	5.7 (12.6)
VOLUME, M^3 (FT^3)	0.008 (.28)	0.065 (2.3)
PROPELLANT CAPACITY, KG (LB)	9.1 (20.1)	4.5 (10.0)
EQUIVALENT IMPULSE, N-SEC (LB-SEC)	47150 (10600)	
ATTITUDE HOLD CAPABILITY (HR)	100	
TRAPPED LIQUID RESIDUAL, KG (LB)	0.4 (0.9)	0.3 (0.6)
RELIABILITY*		
FAILURES/10^6 HR	.426	.5126
FAILURE RATE SOURCE	APOLLO,	P72-2, JPL
COST ($1000)*		
SR&T	500	
DDT&E	250.0	264.5
1ST UNIT	56.9	72.7

*INCLUDING TANK, INSULATION, COOLING COILS, AND CAPILLARY DEVICES

Figure 5-6. Integrated Cryogenic APS Zero-G Reservoir

design of the LOX reservoir is similar. The reservoir is divided into two compartments separated by a 165 x 800 mesh capillary barrier. The upper compartment has the capability of being refilled (purged of vapor) during each of the APS translational maneuvers and MPS burns. The lower compartment offers a degree of redundancy. The design is such that under normal operating conditions, no vapor will enter the lower compartment until depletion of the upper compartment. However, should an off-design condition occur, the lower compartment can accumulate some vapor before any vapor is drawn into the acquisition tubes and on to the APS pump. As the thermal control system condenses this vapor, the capillary acquisition tubes will replenish the lower compartment with more liquid.

The acquisition tubes provide a communication path for liquid from the upper to the lower compartment. They are arranged to be in contract with liquid under any adverse acceleration that might be imposed. The 325 x 2300 Dutch twill self-wicking screen covers are designed to be wetted during the entire mission. In this way these screens will preferentially pass liquid while blocking vapor flow due to the bubble pressure at the liquid/vapor interface.

Figure 5-7 presents a timeline profile over the reference mission for the liquid propellant level in the LH_2 zero-g reservoir. This profile is based on worst-case propellant orientation within the MPS tank. Thus any APS propellant demand during periods when liquid is not positively settled will result in a reduction of reservoir liquid level as vapor is drawn from the MPS tank.

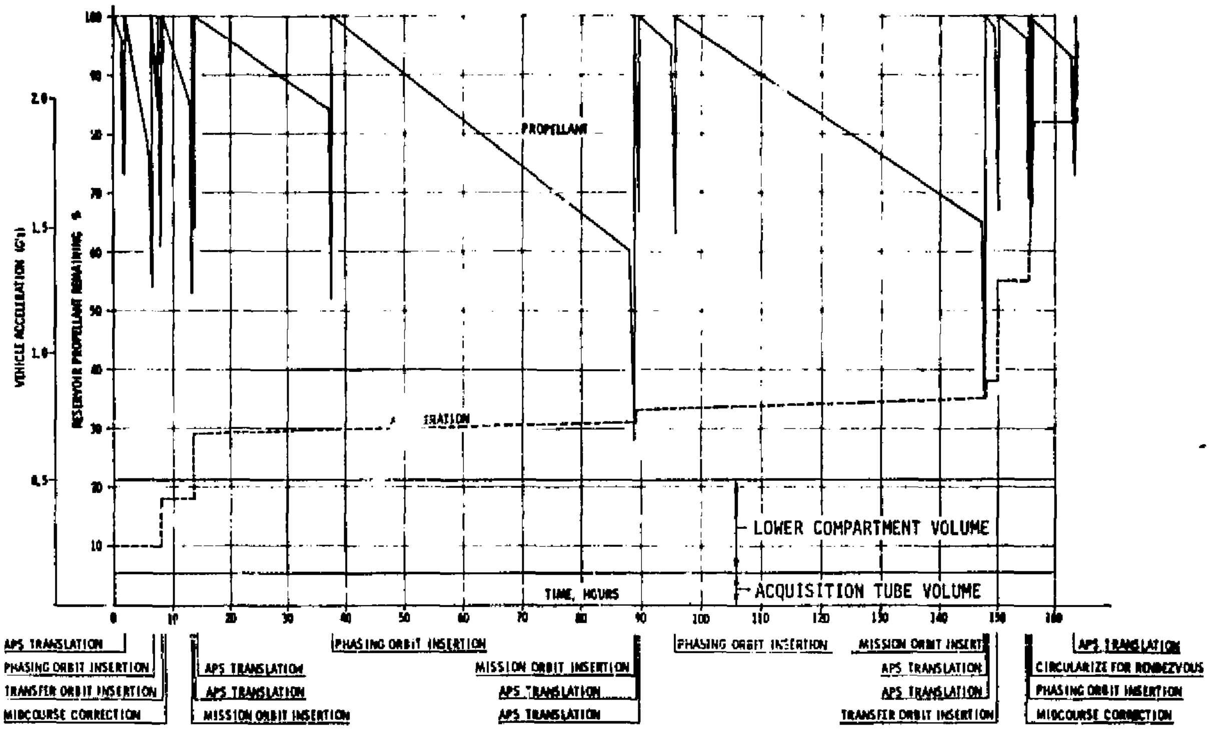


Figure 5-7. Hydrogen Reservoir Propellant Profile

The rapid drops in liquid level shown in the figure result from APS propellant withdrawal during the initial propellant settling period of a translational maneuver, while the slower drops are due to attitude control operation and hydrogen bleed for thermal control.

After the propellants are settled during either an MPS or APS translation maneuver, the refill overboard vent valves are opened and the zero-g reservoir is replenished to 100 percent liquid. Baffles in the tank vent exit minimize losses due to liquid entrainment above which liquid point sensors are installed to activate closure of the vent valves. APS propellant demand for the remainder of the linear translation is provided by a direct flow of liquid from the MPS tank through the reservoir and into the APS pump. Thus the reservoir size is not affected by extended APS linear translation burns. In addition, since APS operation was assumed for propellant settling prior to any MPS burn, vapor accumulation in the zero-g reservoir is the same for either Mission A or B.

Typical liquid propellant orientations within the APS zero-g reservoir are illustrated in Figure 5-8 for zero-g coast and for MPS burn phases. To avoid liquid draining and gas entry into the collector tubes during MPS burn, the capillary screens must be sized to provide a bubble pressure greater than the static head (Dimension X in the figure). The maximum vehicle axial acceleration that can be imposed without draining of the 325 x 2300 mesh screen-covered collector tubes is shown in Figure 5-9 as a function of hydrogen liquid fraction remaining within the reservoir. Also shown is the upper envelope of acceleration and liquid fraction conditions predicted from

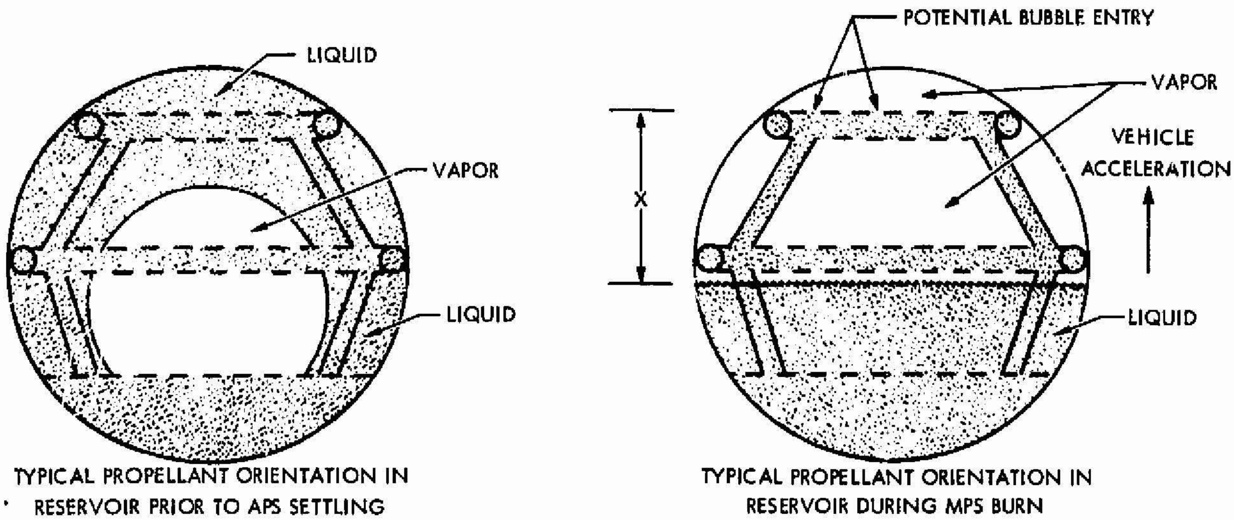


Figure 5-8. Stability o. Capillary Collectors During MPS Firing

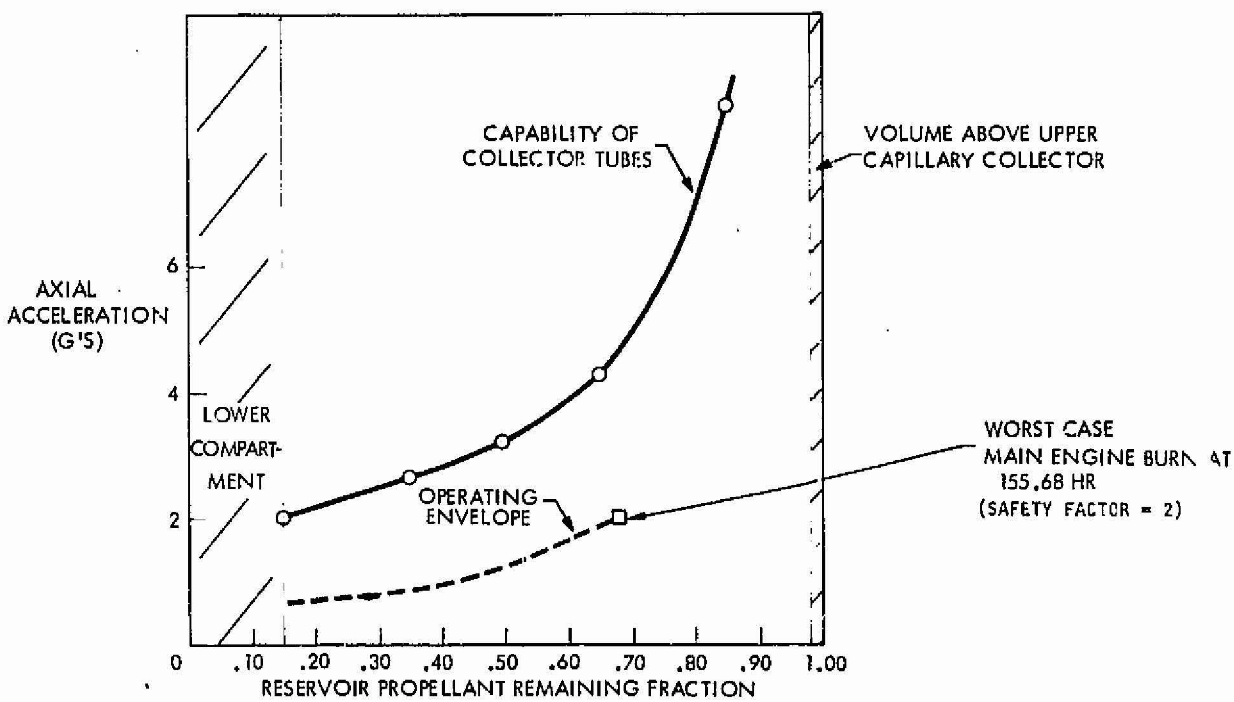


Figure 5-9. Capability and Operating Envelope for Screened Collector Tubes

the mission profile of Figure 5-7. The collector tube screens have been designed for a minimum safety factor of 2 in preventing vapor entry.

The total impulse capacity of the LOX and LH_2 zero-g reservoirs is equivalent to 47151 N-sec (10600 lb-sec), of which 78 percent is contained in the upper compartment. For the maximum propellant consumption period shown on Figure 5-7, this capacity provides a margin of 2825 N-sec (635 lb-sec), or 6 percent of the reservoir volume, before vapor would break into the lower compartment and 8016 N-sec (1802 lb-sec), or 17 percent of the reservoir volume, before vapor could be drawn into the pump.

Although there are currently no indications that other Tug missions would impose more severe impulse or static head requirements on the design of the zero-g reservoir, additional capability can be provided by an extra zero-g reservoir refill operation accomplished during an additional APS linear translation burn.

The reliability of the LOX and LH_2 zero-g reservoirs has been estimated to be 0.426 and 0.5126 failures per million hours, respectively, based on generic failure rates for cryogenic tanks, capillary devices, and lines and fittings of 0.2976, 0.023, and 0.02 failures per million hours, respectively. The generic failure rates were multiplied by cryogenic K factors of 3.0 for LOX and 5.0 for LH_2. The sources of reliability data are the Apollo program for the tanks, P72-2 Spacecraft heat pipe suppliers for the capillary devices, and JPL for the lines and fittings.

SR&T requirements for the zero-g reservoir are dictated primarily by the internal capillary devices. Although similar devices are operational with storable propellant, no flight experience exists for cryogenic propellants. However, theoretical analyses, fluid properties research, and scale model drop and pushover flight tests (short duration zero-g tests), accomplished in a series of research programs carried out for more than a decade, have placed cryogenic capillary device technology on a sound basis. More recently, subscale and full-scale test programs have been carried out at minus one g, (references 11, 12, and 13) and additional work is currently in progress (Reference 14). The objective of Tug APS SR&T in the capillary device area is to apply these well-developed simulation and experiment analysis techniques to a full-scale APS prototype design. No technology advance is needed, but concept confirmation is required. The experimental program would take the form of prototype hardware subjected to operational thermal conditions and negative one g demonstration of retention and feedout capability. The test program is considered to involve validation of thermal control aspects as much as capillary phenomena influences. This is because the two factors are intimately related. Capillary device performance is tied to the design's ability to avoid vapor formation and retain suppression head subcooling at critical points within the device and in the operational cycle.

Propellant refill of the APS propellant reservoir with venting of vapor to space also involves technological problems requiring SR&T. The vent line must have a flow restrictor so that during steady flow, reservoir pressure does not drop enough to cause propellant flashing. As long as the flashing does not occur within the capillary collectors, the loss of propellant is small, and effects inconsequential. To assure that liquid is not vented to space, a liquid sensor at the top of the reservoir, which activates the vent valve, is provided. Propellant sloshing may wet the sensor and could cause premature valve closure if appropriate design provisions are not explored in SR&T. Another problem which must be faced during refill is thermal and thermodynamic state control. The contents of the reservoir are subcooled with respect to the propellant in the main tank due to the cooling of the thermodynamic vent system. The introduction of large quantities of warmer liquid during refill temporarily overloads the thermodynamic vent system. Several hours are required to cool the reservoir's contents to steady state level. The thermal isolation and thermodynamic cooling systems must be designed to prevent boiling of propellant within the collector tubes during this period. SR&T testing analysis will be necessary to define the map of acceptable characteristics.

The estimated cost of an SR&T program for concept validation of both a LOX and LH_2 zero-g reservoir ranges from a low of $300,000 to a high $700,000 with a mean value of $500,000 and $130,000 one sigma uncertainty.

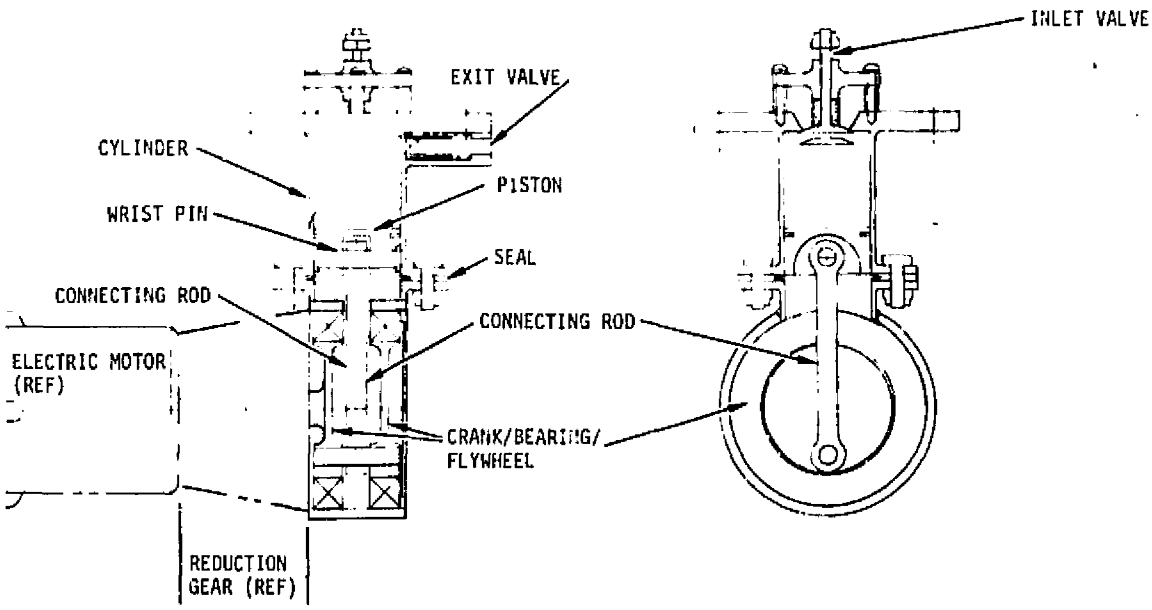
Pump

A conceptual design of the baseline APS pump is presented in Figure 5-10. A positive displacement piston pump was selected as baseline while a vane pump, shown in Figure 5-11, is identified as a competitive alternate. Positive displacement pumps were selected for the integrated cryogenic APS because of their superior efficiency at reasonable design speeds. This choice is considered in more detail in Section 5.3.

The design requirements and operating conditions for the APS pump are listed in Table 5-14. Table 5-15 presents a breakdown of the power, efficiency, and weight characteristics for the piston pump, harmonic drive gear reducer, ac induction motor, and variable frequency inverter. The complete pump and drive system for both LOX and LH_2 (including inverter) requires a power input of 1.44 kw, has an overall efficiency of 58 percent, and weighs 19 kg (42 lb).

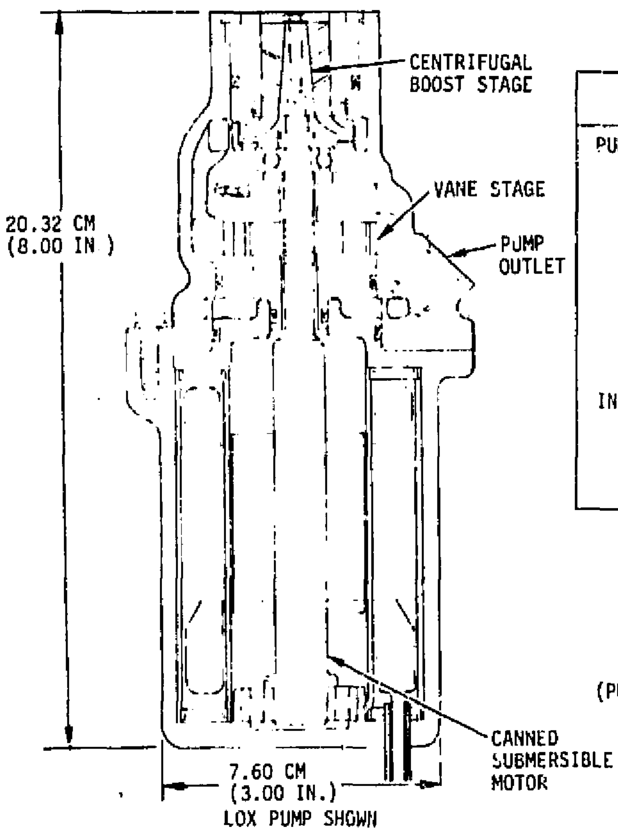
The generic failure rate assumed to compute the reliability of the piston pump and motor combination is 3.0 failures per million hours of operation based on AIRCO field usage data for more than 1000 units with more than 10,000 hours of operation for each unit. Cryogenic K factors of 3.0 and 5.0 were applied for LOX and LH_2 use, respectively, to the pump complement of the generic failure rate (1.0 per 10^6 hr).

The resultant failure contribution per million missions for both the LOX and LH_2 pump/motor combinations is only 5 out of a total of 6470 for the entire APS. Thus, the pump reliability is not a significant factor in system reliability and redundant pumps are not required to achieve an overall APS reliability goal of 0.996. The requirements for the APS to fail operational



PUMP CHARACTERISTICS	LOX	LH_2
DISPLACEMENT, CC/REV (IN./REV)	16.6 (1.014)	53.9 (3.29)
BORE, CM (IN.)	2.3 (0.9)	3.0 (1.2)
STROKE, CM (IN.)	2.4 (0.95)	3.7 (1.45)

Figure 5-10. Piston Pump Preliminary Design



CHARACTERISTICS	LOX	LH_2
PUMP/MOTOR		
FLOW RATE, KG/SEC (LB/SEC)	0.094 (0.2069)	0.031 (0.069)
PRESSURE RISE, N/CM^2 (PSI)	141 (205)	141 (205)
SPEED (RPM)	4,000	12,000
DISPLACEMENT, CC/REV (IN^3/REV)	9.5 (0.136)	1.21 (0.074)
MAJOR RADIUS, CM (IN.)	1.47 (0.580)	1.40 (0.552)
MINOR RADIUS, CM (IN.)	1.31 (0.517)	1.31 (0.517)
EFFICIENCY (%)	58	58
INPUT POWER (KW)	0.233	1.22
WEIGHT, KG (LB)	3.7 (8.1)	3.9 (8.6)
INVERTER (VARIABLE FREQUENCY)		
EFFICIENCY (%)	90	
INPUT POWER (KW)	1.6[illegible]	
WEIGHT, KG (LB)	5.85 (12.9)	

(PUMP AND MOTOR DESIGN AND DATA PROVIDED BY SUNDSTRAND CORPORATION)

Figure 5-11. Vane Pump and Motor Characteristics

Table 5-14. Pump Requirements and Conditions

I. PROPELLANT	LIQUID OXYGEN	LIQUID HYDROGEN
II. PUMP INLET CONDITIONS		
A. FLUID TEMPERATURE, °K (°R)		
(1) MAXIMUM	92.433 (166.38)	21.267 (38.28)
(2) MINIMUM	89.667 (161.40)	20.333 (36.60)
B. PROPELLANT DENSITY, Kg/m^3 (lbs_m/ft^3)		
(1) MAXIMUM	1142.9173 (71.35)	70.7215 (4.415)
(2) MINIMUM (100% LIQUID)	1129.3016 (70.50)	69.6002 (4.345)
C. NPSP, N/cm^2 (psi)		
(1) MAXIMUM	1.37895 (2)	0.68948 (1)
(2) MINIMUM*	0 (0)	0 (0)
III. PUMP DISCHARGE REQUIREMENTS		
A. FLUID TEMPERATURE, °K (°R)		
(1) MAXIMUM	100.000 (180)	22.222 (40)
(2) MINIMUM	89.667 (161.4)	20.333 (36.6)
B. FLUID PRESSURE, N/cm^2 abs (psia)		
(1) MAXIMUM	172.369 (250)	172.369 (250)
(2) MINIMUM	137.895 (200)	137.895 (200)
IV. FLOW REQUIREMENTS		
A. MASS FLOWRATE, Kg/sec (lbs_m/sec)		
(1) MAXIMUM	0.09385 (0.2069)	0.03125 (0.0689)
(2) MINIMUM	0.07508 (0.1655)	0.0250 (0.5512)
B. VOLUMETRIC FLOWRATE, liters/min (gpm)		
(1) MAXIMUM	4.9865 (1.3173)	26.9435 (7.11771)
(2) MINIMUM	3.9892 (1.05384)	21.5548 (5.6942)
V. INTERNAL PUMP RISE REQUIREMENTS		
A. PRESSURE, N/cm^2 (psi)		
(1) MAXIMUM	162.7163 (236)	162.0268 (235)
(2) MINIMUM	125.1398 (181.5)	124.4504 (180.5)
B. HEAD, meters (ft)		
(1) MAXIMUM	146.8389 (481.7539)	2316.8191 (7601.097)
(2) MINIMUM	114.6347 (376.0973)	1754.3598 (5755.762)

*20 PERCENT MAXIMUM VAPOR FRACTION FOR EACH FLUID

while in the vicinity of the Space Shuttle can be satisfied by the propellant accumulators located downstream of each pump. This is discussed at greater length in the next section.

DDT&E and first unit costs have been estimated to be \$351,000 and \$9,500, respectively, for the LOX pump and \$703,000 and \$19,000, respectively, for the LH_2 pump.

SR&T requirements are dictated by the fact that neither oxygen nor hydrogen pumps currently exist for the APS pump-fed designs since the pressure, flow,

Table 5-15. Piston Pump System Weight and Power Summary

Item	Oxider				Fuel			
	Input Power Required [kw (HP)]	Overall Efficiency η	Component Weight [kg (lb)]	RPM	Input Power Required [kw (HP)]	Overall Efficiency η	Component Weight [kg (lb)]	RPM
Pump (AIRCO Cryogenics)	0.170 (0.228)	80	3.2 (7.0)	290	0.88 (1.18)	80	4.3 (9.4)	546
Harmonic drive reducer	0.187 (0.251)	90	1.8 (4.0)	--	1.04 (1.39)	90	2.3 (5.1)	--
AC induction motor (Western Gear Model 35YH81)	0.208 (0.279)	90	1.1 (2.5)	12,000	1.15 (1.54)	90	1.1 (2.5)	12,000
Total		0.648	6.1 (13.5)			0.648	7.7 (17.0)	

Power Summary:

Total power required from inverters (pump input power/η_{drive} η_{motor}

Oxidizer, Kw (HP)	0.21 (0.28)
Fuel, Kw (HP)	1.08 (1.46)
	1.29 (1.74)

Power required into inverter (pump system power/$\eta_{inverter}$ Assume η = 0.9, Kw (HP) 1.44 (1.93)

Weight Summary:

Total pump system weight, kg (lb)	13.8 (30.5)
Variable frequency Inverter weight (General Motors Type, 8 lb/kw), kg (lb)	5.23 (11.5)
Total system weight (inverter + pump system), kg (lb)	19.0 (42.0)

and suction requirements are unique. In particular, the flow rates are orders of magnitude smaller than that of flight-weight pumps previously developed for O_2/H_2 rocket engines. The pressure requirement is moderate but high enough to demand special design provisions with positive displacement machines. The low suppression head suction requirement has been previously satisfied, but only for high-flow centrifugal/axial rocket engine pumps. For these reasons, it is considered that the APS pump SR&T involves relatively straightforward development of custom designs using well-established concepts scaled from other applications. The SR&T program would include the objectives of substantiating NPSP capability, supporting material selection studies, verifying cryogenic clearances, and investigating the safety problems associated with an oxygen pump installation. Cryogenic flow tests of full-scale prototype LOX and LH_2 pumps would be conducted. The SR&T costs are estimated to be approximately $200,000 expended over a 9-month period.

It is expected that the pump can be designed for a life expectancy of over 100 hours and 2000 cycles and thus no scheduled replacement would be required over the 20-mission, 10-year life of the Tug. The required operating life per mission is 0.6 and 2.7 hours for mission profiles A and B, respectively.

Accumulator

A cross-sectional view and table of preliminary design characteristics for the LOX and LH_2 accumulators are presented on Figure 5-12. Weight trade studies presented in Section 5.3 show that the lightest weight APS can be achieved at the smallest possible accumulator volume. The accumulators are sized, then, to limit pump short cycles to 100 cycles per mission to provide reasonable pump reliability and minimize pump replacements. The accumulator,

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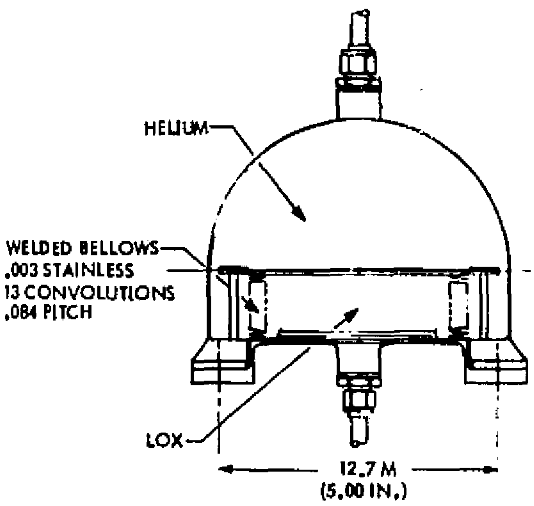


CHARACTERISTICS	LOX	LH_2
DRY WEIGHT, KG (LB)	0.3 (0.7)	1.0 (2.2)
TOTAL VOLUME, M^3 (FT^3)	0.0014 (0.05)	0.0074 (0.26)
PROPELLANT CAPACITY, KG (LB)	0.200 (0.45)	0.07 (0.15)
EQUIVALENT IMPULSE, N-SEC (LB-SEC)	1063 (239)	
ATTITUDE HOLD CAPABILITY (HR)	1.5	
TRAPPED LIQUID RESIDUAL, KG (LB)	0.01 (.03)	0.003 (0.006)
PRESSURE, MAX/MIN, N/CM^2 (PSIA)	162/141 (235/205)	
RELIABILITY		
FAILURES/10^6 HOURS	9.4	15.7
FAILURES/10^6 CYCLES	1.5	2.5
FAILURE RATE SOURCE	BELSAD & METAL BELLOW CO.	
COST, K $		
SR&T	100	
DDT&E	50.1	50.1
1st UNIT	23.7	23.7

Figure 5-12. Accumulator

which has enough propellant capacity for 1.5 hours of Tug attitude hold, also provides backup propellant feed during the critical phase of Tug rendezvous and docking with the Space Shuttle.

Each accumulator is a trapped ullage device. The helium on the gaseous side of the bellows expands and contracts as liquid propellant leaves or enters the expandible liquid volume. The gaseous and liquid volumes have been selected to control the thruster inlet pressure within a band of 141 to 162 N/cm^2 (205 to 235 psia). Positive stops are provided to prevent over-expansion of the bellows. Pump start, stop, and speed are controlled by a combination of accumulator pressure switches and bellows position switches. Propellant for short APS pulses is supplied from the accumulator until it is nearly depleted, whereupon the lower position switch activiates the pump. During extended APS burns, propellant is fed directly from the pump to the thruster, replenishing and then bypassing the accumulator.

The primary problem associated with the accumulator design is the impact of the currently predicted bellows failure rate on overall APS reliability. Generic life and cycle failure rates have been developed based on data provided by two bellows suppliers, Belsad Corp. and Metal Bellows Company. The resultant failure contribution per Tug mission is 4496 out of a total of 6470 for the entire APS. This results in an overall APS reliability of 0.9935 as compared to a goal of 0.996. It is considered that the bellows failure rate prediction could be reduced by a factor of 2 through SR&T life cycle testing, and the overall APS reliability could then be raised to 0.9955. The estimated SR&T cost to achieve this goal is $100,000 expended over a 12-month period.

Thermodynamic Control

The objectives of the thermal control system are:

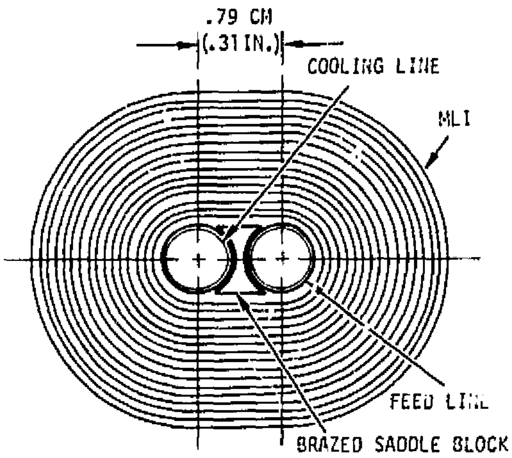
1. To satisfy thruster propellant inlet temperature requirements.

2. To provide subcooled propellant within the zero-g reservoir capillary acquisition tubes.

3. To satisfy feed pump net positive suction pressure (NPSP) requirements as necessary to assure full pump flow (refer to Table 5-14).

The thermal control system consists of insulation and low conductivity supports to minimize tankage and feedline heat loads, and an active hydrogen bleed system to dissipate these heat loads and control propellant temperature. MLI consisting of 1.27 to 2.54-cm (0.5 to 1.0-in.) thick blankets made up of 30 to 60 layers of single aluminized Mylar or single goldized Kapton will be used to provide a radiation barrier. Thermal isolation supports designed for maximum strength-to-conductivity ratio will be provided by using titanium struts or cylindrical posts consisting of axially oriented S-glass rods. Ultimately, the materials and designs of the MLI and supports will be common to the MPS tank insulation.

A detailed breakdown of the predicted heat loads for the APS thermal isolation system is presented in Figure 5-13. Also listed are the heat



HEAT LOADS (E_{eff} = .002)

COMPONENT	PROPELLANT	SURFACE AREA M^2 (FT^2)	HEAT LOAD W (BTU/HR)
ZERO G RESERVOIR	LOX	0.19 (2.08)	0.16 (0.55)
	LH_2	1.00 (10.8)	0.91 (3.1)
PUMP & ACCUM	LOX	0.06 (0.647)	0.05 (0.17)
	LH_2	0.18 (1.98)	0.17 (0.57)
LINES	LOX	2.42 (26.0)	1.52 (5.2)
	LH_2	2.42 (26.0)	2.17 (7.4)
THRUSTERS	LOX	*	0.73 (2.5*)
	LH_2	*	1.46 (5.0*)
TOTAL	LOX	—	2.46 (8.4)
	LH_2	—	4.71 (16.1)

*BASED ON CONDUCTION THROUGH 0.318-CM (0.125-IN.) DIAMETER X 3.8-CM (1.5-IN.) BELLOWS WITH 0.013-CM (0.005-IN.) STAINLESS STEEL WALL

LH_2 BLEED REQUIREMENTS	LOX SYSTEM	LH_2 SYSTEM
H_2 BLEED TEMPERATURE, INLET/OUTLET, K (R)	66/89 (119/160)	19/27 (35/48)
H_2 BLEED ENTHALPY AVAILABLE, J/LB (BTU/LB)	384,000 (165)	491,000 (211)
H_2 BLEED FLOW REQUIRED, KG/HR (LB/HR)	0.023 (0.0509)	0.035 (0.0763)
HEATER POWER FOR TEMPERATURE CONTROL (WATTS)	4	ON/OFF FLOW CONTR
H_2 BLEED EXPENDED PER MISSION, KG (LB)	5.67 (12.5)	

Figure 5-13. Liquid-Liquid O/H APS Thermodynamic Control

transfer surface areas computed for the feed lines and other major components. Since all of the APS except the thrusters is installed within the Tug vehicle skin, direct solar and albedo heat loads do not have to be included. The assumed environmental temperature of 294K (530 R) is considered conservative since a major portion of the exposure will be to the colder main tank insulated surfaces. An effective emittance ($\bar{\epsilon}$) of 0.002 for the overall thermal isolation system, including insulation and supports, was used to compute the listed heat loads. This value is based on anticipated 1978 technology incorporating insulation advancements, primarily in vacuum-deposit metallizing techniques and insulation layup methods. The thruster heat loads were based on conductive heat transfer from an uninsulated thrust chamber to the cryogenically cooled thruster inlet valves through 0.32-cm (0.125-in.) diameter bellows having a 3.8-cm (1.5-in.) developed length and 0.013-cm (0.005-in.) stainless steel wall.

The resultant heat loads are dissipated by the flow of hydrogen bled from downstream of the APS pump and expanded through a Joule-Thompson expander to 7.86 N/cm^2 (11.4 psia) and 19.4 K (35 R). The cold hydrogen bleed is routed through cooling coils mounted on the external wall of the zero-g reservoir and then through cooling lines which trace the hydrogen feed manifold. Heat is transferred via saddleblock segments brazed between the feed and cooling lines as shown on Figure 5-13. After passing through an electrical heater, the hydrogen bleed is then routed around the LOX system in a manner similar to that for the hydrogen system.

Also listed in the figure are the hydrogen bleed inlet and outlet temperatures and the corresponding enthalpy, or heat absorption capability. As can be seen, the hydrogen bleed flow rate required to carry away the heat load is less for the oxygen than for the hydrogen system. Thus, a 4-watt electrical heater in the hydrogen bleed line is required to control the oxygen propellant temperature above freezing. Local temperature control will be provided by the design of thermal contact zone areas and conductivities and verified by cold vacuum chamber development tests. The quantity of hydrogen expended to provide steady state thermal control of both the LOX and LH_2 APS propellant is 5.7 kg (12.5 lb) per mission.

Transient chilldown characteristics are presented in Figure 5-14 for the APS LOX system. During the Shuttle pre-launch and boost phases, the APS feed lines will be inerted to minimize the hazard of leakage into the cargo bay. After the cargo bay doors are opened on-orbit and prior to release of the Tug, the APS must be activated by filling and chilling the thruster feed manifolds. This chill operation must be completed in approximately 30 minutes to allow Tug deployment within the first complete orbit revolution. Figure 5-14 presents the LOX feedline temperature as a function of time for several different hydrogen bleed flow rates. Although hydrogen and oxygen system chilldown will occur simultaneously, steady state temperature is expected to be attained first for the hydrogen system. As can be seen, the APS thermodynamic control system must provide a higher than nominal flow rate capability for initial

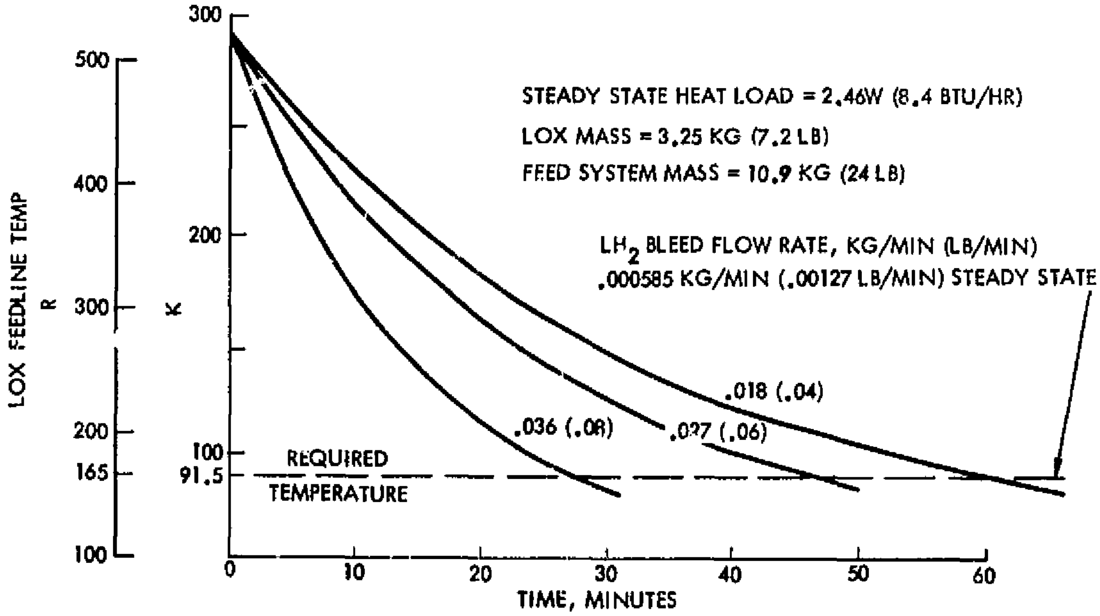


Figure 5-14. Integrated Cryogenic APS LOX Feed System Chill Transients

chill. The additional high flow control valve and Joule-Thompson expander necessary to provide this capability are shown in Figures 9-2 and 9-3. A total of 1 kg (2.2 lb) of hydrogen will be expended for initial system chill. This is equivalent to only a 0.36 kg (0.8 lb) payload penalty.

DDT&E, first unit, [illegible] urbishment costs for the various components of the thermodynamic control system are shown in Tables 5-7 and 5-8. Refurbishment costs include scheduled inspections of the insulation and localized repair as required.

Reliability analysis reveals that two thermodynamic control system components make a significant contribution to the total number of predicted failures per mission. As seen on Table 5-13, these are the bleed solenoid and bleed expander, neither of which are redundant, which contribute 240 and 123 failures per million missions, respectively, out of an APS total of 6470 failures per million missions. Redundant solenoid valves or the use of an electrical heater could be considered as potentially more reliable temperature control methods.

SR&T requirements for the thermodynamic control system can best be satisfied by thermal-vacuum testing of a prototype system including thermally representative segments of the entire APS. System-level SR&T is considered necessary because the two critical elements - the cryogenic capillary device and the liquid-liquid thruster - are both SR&T items and their interactions in a new concept system design need to be investigated experimentally prior to DDT&E. It is also certain that thruster feed system manifolds, the thermal arrangement of the manifold, and thruster interfaces will have a strong influence on performance of the system and the capillary device and, in turn, on thruster design requirements. For these reasons, a system-level demonstration test program using a prototype configuration is considered necessary. It would involve a complete thruster, a representative insulated segment of the feed system, and the capillary device mockup (from previous capillary device SR&T test program). The system would be set up in a vacuum chamber (only to the pressure level necessary to obtain MLI thermal performance) with thruster exhaust to sea level pressure. System thermal environment would be simulated during mission duty cycle firing profiles to obtain performance and operational characteristics data. It is expected that this SR&T program would cost approximately $1.2 M and cover a 12-month period.

SYSTEM PERFORMANCE

The integrated concept (I-5) performance depends on the electrical power option selected as a source for the pump drive. As noted in a preceding section, the power source can be either a fuel cell or a primary battery sized to match the planned mission. Table 5-16 shows the reference mission performance of the integrated concept with these power options.

Battery Power Options

The power option of I5-1 provides the highest performance since it uses the lightest power source, a minimum size battery capable of supporting only the Mission A profile. I5-3 has the same power source but utilizes a higher area ratio thruster. The I5-3 payload is lower than I5-1 since, on the low APS impulse Mission A profile, the weight penalty of higher area ratio offsets the specific impulse gain. I5-4 also uses high area ratio but has a larger battery to permit operation of the Mission B profile. This added battery weight reduces the I5-4 payload on the Mission A profile below that of either I5-1 or I5-3. If the final Tug power system design permits battery size options to be exercised for each flight, then the high area ratio thruster could be preferred. This would take advantage of the integration feature, and either Mission A or B profiles could be flown as appropriate at only a 4.5 kg (10 lb) payload penalty on Mission A profile due to the greater thruster weight. As noted in a following paragraph, the higher thruster performance (for +X thrusters) also enhances vehicle flexibility through APS backup capability of the main engine. For any of the battery options, switchover to fuel cell power during APS abort propulsion is possible due to the absence of main engine power demand.

Fuel Cell Power Option

The I5-2 option is provided with dedicated power by augmenting the Tug fuel cell. The fuel cell weight penalty is between that of the Mission A and B profile batteries, and the payloads vary correspondingly. With this power option, either Mission A or B flight profiles may be flown without changing the configuration of any vehicle subsystem. This power option permits not only main engine backup power, but on-demand unlimited energy for the APS at any point in a mission. With negligible penalty due to fuel cell reactant consumption, any APS total impulse (up to MPS tank capacity) may be provided for

Table 5-16. Integrated Concept Performance

	Power Option			
	I5-1	I5-2	I5-3	I5-4
Power source, kg (lb)				
Fuel Cell Δweight	-	19.7 (43.4)	-	-
Primary battery weight	12.9(28.5)	-	12.9(28.5)	35.2(77.6)
Thruster				
Expansion area ratio -	50	200	200	200
Specific impulse, N-sec/kg (sec)	3910(398.7)	4002(408.1)	4002(408.1)	4002(408.1)
Reference Mission Payload, kg (lb)				
Mission A	2437(5373)	2389(5267)	2433(5363)	2357(5196)
Mission B	-	2195(4839)	-	2163(4768)
MPS/APS Impulse Interchange, % of MPS				
Pulse mode--MR = 3.0	0	53	0	0
ΔV Mode-- MR = 5.6	0	88	0	0
Main Engine Backup Abort Capability, % of MPS				
Approximate Burn Time	60	60	60	60

a mission. With respect to the opportunity to plan missions which use this additional APS capability, Tug vehicle versatility is enhanced. For missions which are modified during flight to use this capability for contingencies such as main engine failure or unanticipated events, Tug vehicle flexibility is enhanced.

The capability to interchange propellant between APS and MPS is limited by tank capacity and the APS mixture ratio as shown in Table 5-16. The MPS loaded mixture ratio is 5.6 for all except the full-tank retrieval mission when the mixture ratio is 6.0. In pulse mode (mixture ratio of 3), only 52 percent of the MPS impulse capacity is interchangeable due to the fuel load limitation. However, this is undoubtedly more than ample for that type of maneuvering. In a velocity mode, the concept's capability for steady state operation of thrusters at a mixture ratio of 5.6 eliminates the propellant outage effect and the impulse interchange fraction is in the ratio of APS to MPS specific impulse values--0.86. The battery powered options are listed in the table as having no impulse interchange capability since they require switchover to fuel cell power and the main engine power allocation. This mode is considered, at present, to be applicable only to operations after a main engine failure.

The APS/MPS impulse interchange capability of 0.86 can also be used for abort backup of the main engine. After allowing for gravity losses as well as the impulse difficiency (see Section 5.3), the main engine backup capability, without payload jettison or abort propellant reserve, covers approximately 60 percent of the main engine duty cycle on synchronous equatorial missions. A similar coverage value could be expected for all other types of Tug-recovered earth-orbital missions since a large fraction of the outbound leg of any mission is the phase where backup capability exists, and this leg, in general, requires much more than 60 percent of the main engine burn time.

Propulsion Crossovers

Small velocity maneuvers can be performed with the APS or any one of the three main engine operating modes: tank head idle (THIM); pumped idle (PIM), and mainstage. For this study, a main engine operating rule assumed was that the engine would not enter the next higher thrust mode unless the maneuver was large enough to require 5 seconds of steady state burning in the higher mode. This rule was made to reflect practical requirements to eliminate short cycling of engine controls and to minimize engine wear where little performance penalty was entailed. The result, for the reference mission, is shown in the mission timelines of this report (see Table 5-11, for example) where the main engine mode is identified for each maneuver. In Figure 5-15, the best performance domains of each of the four possible propulsion modes is shown in terms of propellant required to provide a particular maneuver total impulse. THIM has less specific impulse than the APS and therefore has no best performance domain. The crossover to PIM occurs at its minimum (5-second steady state operation) practical capability and so does the crossover to main stage.

TUG VEHICLE IMPACT

The application of the integrated cryogenic APS to the Tug vehicle affects vehicle design, development, and operations. The significant impacts are discussed in this section.

Structure Design

The installation of the integrated APS requires less space allocation for tanks than either storable or cryogenic dedicated systems. Lines and manifolds are more complex than for a storable system but are still readily installed. Thruster quad installation is similar for all systems except that the O/H system requires better surface thermal protection since its exhaust stagnation temperature is higher, particularly when compared to a monopropellant system. If the augmented fuel cell power option is used for the APS, Tug radiators will be increased 2.2 m^2 (22 ft^2) and 5 kg (11 lb).

Power System Design

The power system required for the integrated cryogenic APS involves added power source and conditioning. The ac pump driver require variable frequency inverters for speed control. Added power can be obtained by augmentation of both of the baseline vehicle's redundant fuel cells from 1.75 kw to 2.60 kw or by the addition of silver-zinc primary batteries. The batteries may be sized for either mission profile A at 0.86 kw-hr or B at 3.9 kw-hr. The respective battery weights are 12.9 kg (28.5 lb) and 35.2 kg (77.6 lb). The difference in APS and Tug performance capability for these power options has been described previously. Their design basis is discussed in Section 5.3.

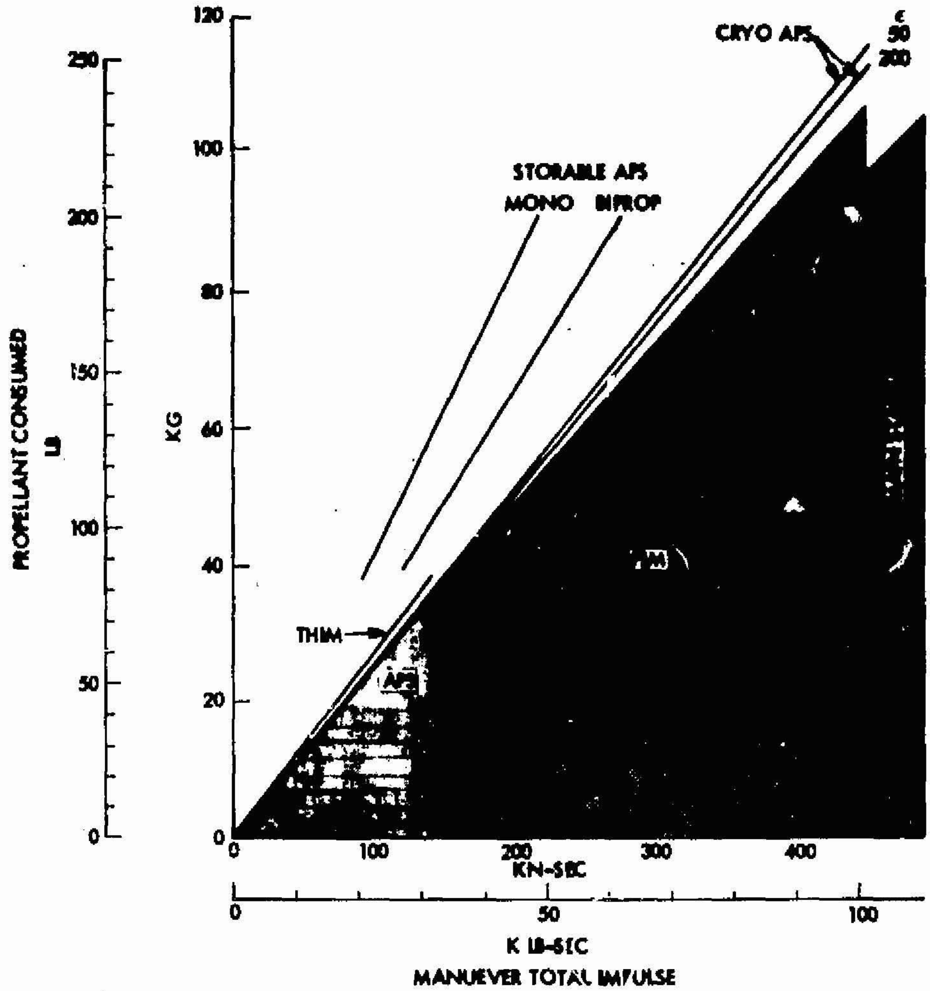


Figure 5-15. Propulsion Crossovers

Main Engine Design

The integrated APS permits elimination of self-settling (but not THIM childown) and PIM. However, since these features are described by the engine manufacturer as being inherent in the RL-10 IIB design, no configuration changes are involved.

Main Engine Development

The elimination of THIM self-settling and PIM functional requirements on the main engine reduces development cost and time. The impact of eliminating THIM self-settling has not been estimated by the engine manufacturer, but it would probably appear as an engineering cost (both engine and systems contractor) as well as engine development test cost reductions. Current engine manufacturer plans to cover the anticipated range of inlet conditions (from vapor to liquid for either propellant) in 30 engine development tests would be simplified to liquid-inlet-only tests. Design and testing involving the engine's oxygen heat exchanger would similarly be simplified since accommodation of mixed or slugging flow would be eliminated.

In this study, no attempt is made to quantify the favorable impact of the integrated concept regarding engine start propellant acquisition. The integrated cryogenic APS can provide MPS settling for engine start at no penalty, since the APS specific impulse is higher than that of THIM. A Tug design using storable APS can use either a start basket (as in Reference 1) or THIM self-settling or incur a severe payload penalty by providing APS settling. Reference 1 estimates the SR&T cost for a start basket at $3.2 M. No estimate for start basket DDT&E is identified.

Elimination of PIM is estimated by the engine manufacturer to reduce engine development by $2.83 $\bar{M}$ and two months. The test program is reduced by two engine sets and 50 tests. On the study reference mission, PIM is used only once. This raises the question as to whether the storable APS vehicles also can eliminate PIM. Reference to Figure 5-15, propulsion crossovers, reveals that the penalty is higher, 11.8 kg (26 lb) of propellant per maximum PIM maneuver, for storables (using THIM) and only 5.5 kg (12 lb) for the integrated cryogenic APS. It has not been determined how many PIM maneuvers are best for other Tug missions, only that the payload penalty for eliminating it is high for storables and nearly negligible for the cryogenic integrated APS.

An additional impact on main engine development results from APS ability to provide backup propulsion for 60 percent of the main engine duty cycle. This reduces the criticality of the main engine which otherwise is a single-point failure through all Tug operations. Whether this feature ultimately produces an actual cost saving in engine development, the APS backup capability must be viewed as a virtual cost saving somewhere in the Tug program. The existence of a backup would be pervasive during both vehicle and engine development and during initial flight operations. To assign a value to this potential cost saving, it is estimated that engine testing can be reduced by 150 tests at a saving of 3.6\bar{M}$. This estimate is obtained by noting that the engine manufacturer's planned number of tests of 750 corresponds to a demonstrated reliability of 0.996 at 95 percent confidence (Figure 5-16). This particular combination of factors is not unusual and can be viewed as typical of a single-point failure engine program objective. However, with APS partial backup, it is assumed that program reliability demonstration objectives can be relaxed to a value of 0.995. This corresponds to the reduction of 150 tests.

Vehicle Operations

The integrated APS provides several advantages to Tug operations. Foremost among these is the added versatility provided by APS/MPS propellant interchange. As noted previously, with fuel cell power or with the proper battery capacity and battery-to-fuel cell switchover, the APS total impulse capability is 86 percent of the MPS.

For the study reference mission, the Mission B profile using the APS instead of the MPS for low velocity changes can be flown at will when missions are not payload-limited. Engine reliability degradation and overhaul costs would then be reduced since 5 out of 11 main engine rotating starts would be supplanted by the APS. The payload limit would be 195 kg (428 lb) less than maximum rated, but engine overhaul cost would be reduced by $9000 per flight. For the baseline mission model of 243 flights, it is estimated that this profile could be used on 100 flights, yielding a total potential cost reduction of $900,000.

The added duty placed on the APS by Mission B does not measurably increase its maintenance cost because the refurbishment frequencies of the affected components are limited by random failures rather than by life. The added duty adds only 5 pump/accumulator cycles to the 100 cycles of Mission A. Similarly, the thruster start cycles are increased by 5 from 2300. The increase in operating hours is substantial, from 0.5 to 1.7 hours per mission, but neither the pump nor the thrust chamber is life-limited and thus their useful life is not appreciably affected by the addition of 24 hours of operation.

An even greater benefit of APS/MPS impulse interchange is that the severe penalty of providing APS impulse margin in the design of a dedicated propellant supply is avoided. To illustrate the need for margin, Figure 5-17 shows the current impulse requirements spectrum for the Tug vehicle in terms of rotational and linear impulse. It is apparent that no firm maximum requirement is readily identifiable. To avoid the penalty of margin, APS dedicated systems are frequently undersized in early vehicle design phases. This occurred on the Apollo program where the SM RCS propellant capacity was doubled late in the program. Undersizing also occurred on the two most recent Tug system studies (Reference 1 and 2) where the (so far) controlling triple placement mission was not examined for APS impulse requirements. Reference 1 notes that the 14-day

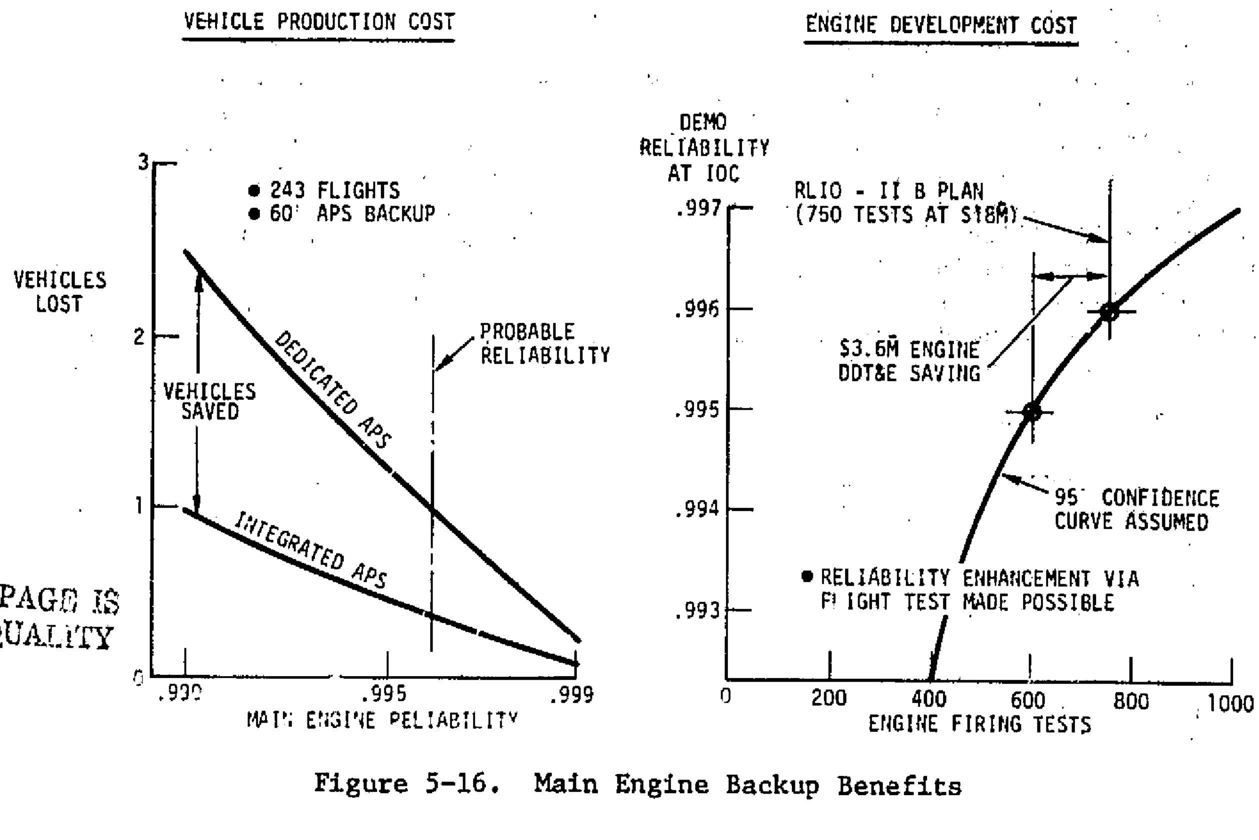


Figure 5-16. Main Engine Backup Benefits

Tug service mission requires additional APS impulse but the amount is not determined. On the figure, the shaded area for the 14-day service mission represents only a rough estimate. The determination of required APS impulse for any Tug mission is an extensive task and has not been attempted for all planned Tug missions. To summarize, future missions and payloads are difficult to predict, missions are evidently ultimately planned around capability, and the Tug is a multipurpose vehicle with a probable operational program span of at least 20 years; therefore it is most beneficial to obviate the need to make a firm early prediction on the APS impulse requirement or to change the vehicle as new requirements arise.

Main engine backup capability by the APS also has operational cost benefits. Figure 5-16 also shows the number of vehicle losses for the baseline 243-flight Tug program as a function of main engine reliability. In Reference 1, the attrition rate is assumed as 1 percent. If the main engine reliability is assumed at 0.996, then 60 percent of that attrition rate is attributable to the main engine--a not unreasonable correlation. With integrated APS backup, 0.6 of a unit is the mathematical expectation for the number of vehicles and Tug payload losses to be avoided. (This neglects the retrieval missions where payload loss is not involved in the outbound leg.) Assigning $10.5 M and $20 M unit costs to the Tug and typical payload yields potential (expected value) cost savings of $6.3 $\bar{M}$ and $12.4 $\bar{M}$, respectively, due to the backup capability.

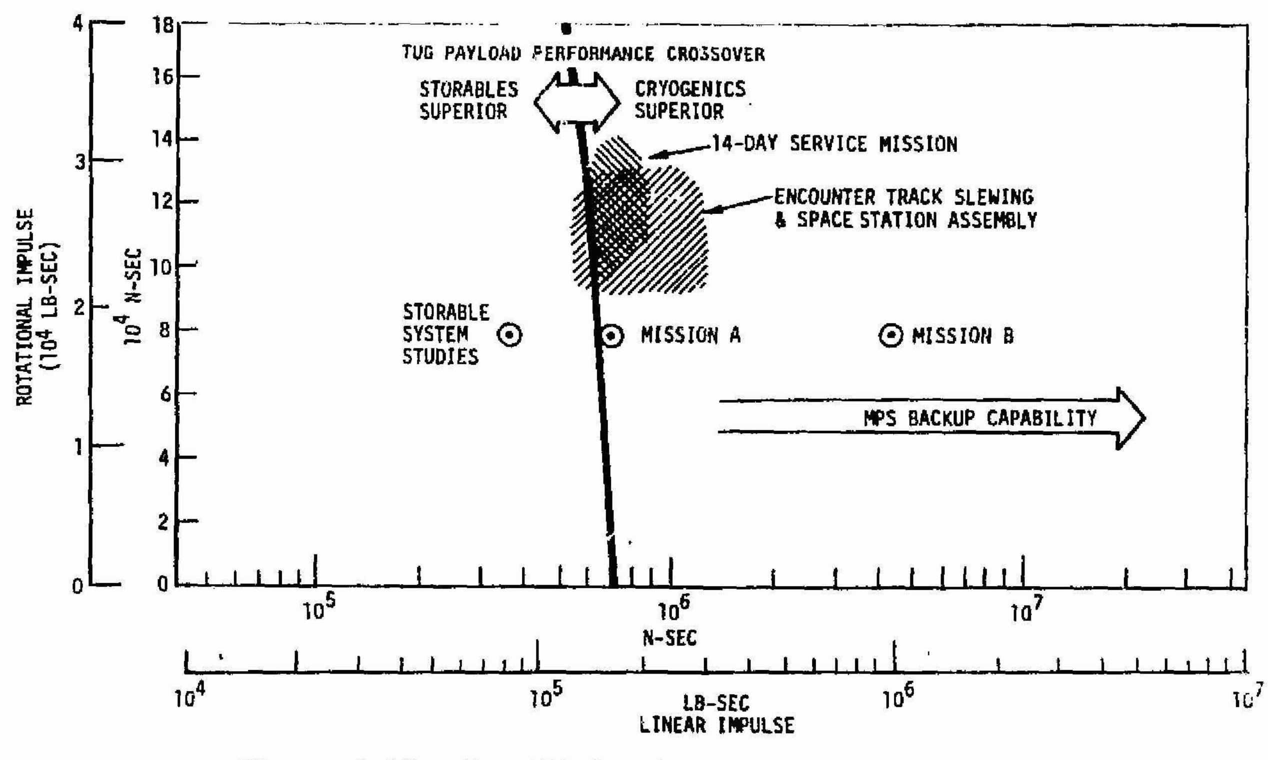


Figure 5-17. Tug APS Impulse Requirements Spectrum

5.3 DESIGN ANALYSES

PERFORMANCE ANALYSIS BASIS

Performance analysis of integrated concepts involved the same model as used for dedicated concepts plus additions to account for (1) the added MPS tankage required for storage of APS propellant and (2) the comingling of MPS and APS flight performance reserve propellant allocations.

The added MPS tankage weight penalty was computed by using the increase in total MPS tank propellant load over the baseline (dedicated APS) value of 22767 kg (50193 lb). This increase was computed for both oxidizer and fuel, and inert weight fractions of 0.0127 and 0.114 (burnout weight to usable propellant in main tanks) for oxidizer and fuel, respectively, were applied. These inert weight fractions account for residual vapor weight increase as well as mid-tank structure and insulation inert weights. For the selected concept (I-5), the total tank weight penalty attributable, including 13 percent dry weight growth, was 4.75 kg (10.5 lb) for Mission A and 21.76 kg (48.0 lb) for Mission B.

The dedicated APS concept flight propellant reserves (FPR) for MPS and APS are physically separated. Accordingly, each allocation is loaded and the performance analysis treats them as burned weight items since, on a nominal mission, they are actually present at final cutoff. The MPS FPR is nominally equal to 3 percent of the velocity budget for the tank-size-controlling retrieval mission. The APS reserve, for all concepts, is 10 percent of the APS usable propellant. Both of these quantities represent the probable limit to the accumulated magnitude of a multiplicity of possible trajectory and propulsion performance deviations during operation of the respective systems. Just as the FPR for each system is found by obtaining the root sum square (RSS) of the contributing deviations (Reference 15), the comingled FPR is the RSS of the two contributing FPR. All of the deviations from nominal propellant consumption including the two FPR values are random, independent, continuous variables and are thus combined by the RSS method. This method also retains the same probability level for the limit to accumulated deviations.

In this case, three sigma is assumed. For the integrated concept (I-5), the stage weight statement (Table 5-9) shows only 3 lb for the APS FPR. This is a misleading but convenient simplification resulting from listing the MPS FPR of 144 kg (318 lb) at its baseline value. The sum of the MPS and APS FPR provides for the three-sigma probability that the MPS and APS deviations will not exceed 3 percent of the MPS velocity budget plus 10 percent of the APS usable propellant.

The MPS FPR is available to it at final cutoff except for missions (not the study reference mission) where the APS reservoir capacity of 11.8 kg (26 lb) exceeds the APS requirement after final MPS cutoff. In that event, the excess in the APS reservoir can be used by the APS to perform the remaining MPS velocity at 86 percent effectiveness or the refill of the APS reservoir can be programmed to avoid this small MPS FPR shortage.

THRUSTER DESIGN POINT SELECTION AND SENSITIVITIES

Thruster Design Point

The application of engine parametric performance data to the integrated APS resulted in performance variations with engine design point characteristics as shown in Figure 5-18. The influences of mixture ratio, chamber pressure, and area ratio follow the same pattern on Mission A as previously described for dedicated concepts. The trend for Mission B is different since considerably more APS impulse is provided. At the higher total impulse, performance is more sensitive to specific impulse and the dropoff with mixture ratio is more pronounced.

An increase in area ratio increases specific impulse and is most beneficial on Mission B. Higher chamber pressure does not improve the Mission B payload since pump, drive and power source weight overcome any specific impulse gain.

After preparation of performance trends, system and thruster susceptibility to propellant temperature variations and thrust cooling limitations were combined to make the thruster design point selection for integrated concepts. The only major change was to shift the operating mixture ratio from 4.0 to 3.0. As noted in the following section, this permits a wider thruster inlet temperature variation. As shown in Figure 5-18, the payload variation was minimal and not a factor in the selection.

The chamber pressure was retained at 103 N/cm^2 (150 psia) since it permits a lower risk approach from a thruster cooling viewpoint. A higher pressure would be desirable to accept greater propellant temperature variations, but the reduced throat size adversely affects thruster cooling. The payload loss at higher chamber pressures is another contributing factor.

The area ratio for both A & B mission profiles was selected for best performance. No change in the dedicated system choice of 50 for Mission A was required. For Mission B systems, a 200 area ratio was selected.

Thruster Sensitivity to Propellant Inlet Conditions

The purpose of this analysis was to evaluate the performance sensitivity of the thruster to propellant valve inlet conditions. Engine performance was parametrically evaluated as a function of propellant temperature and pressure by calculating the corresponding variation in engine flowrates, chamber pressure, and mixture ratio. The integrated APS engine baseline design (mixture ratio of 3.0) was used in this study.

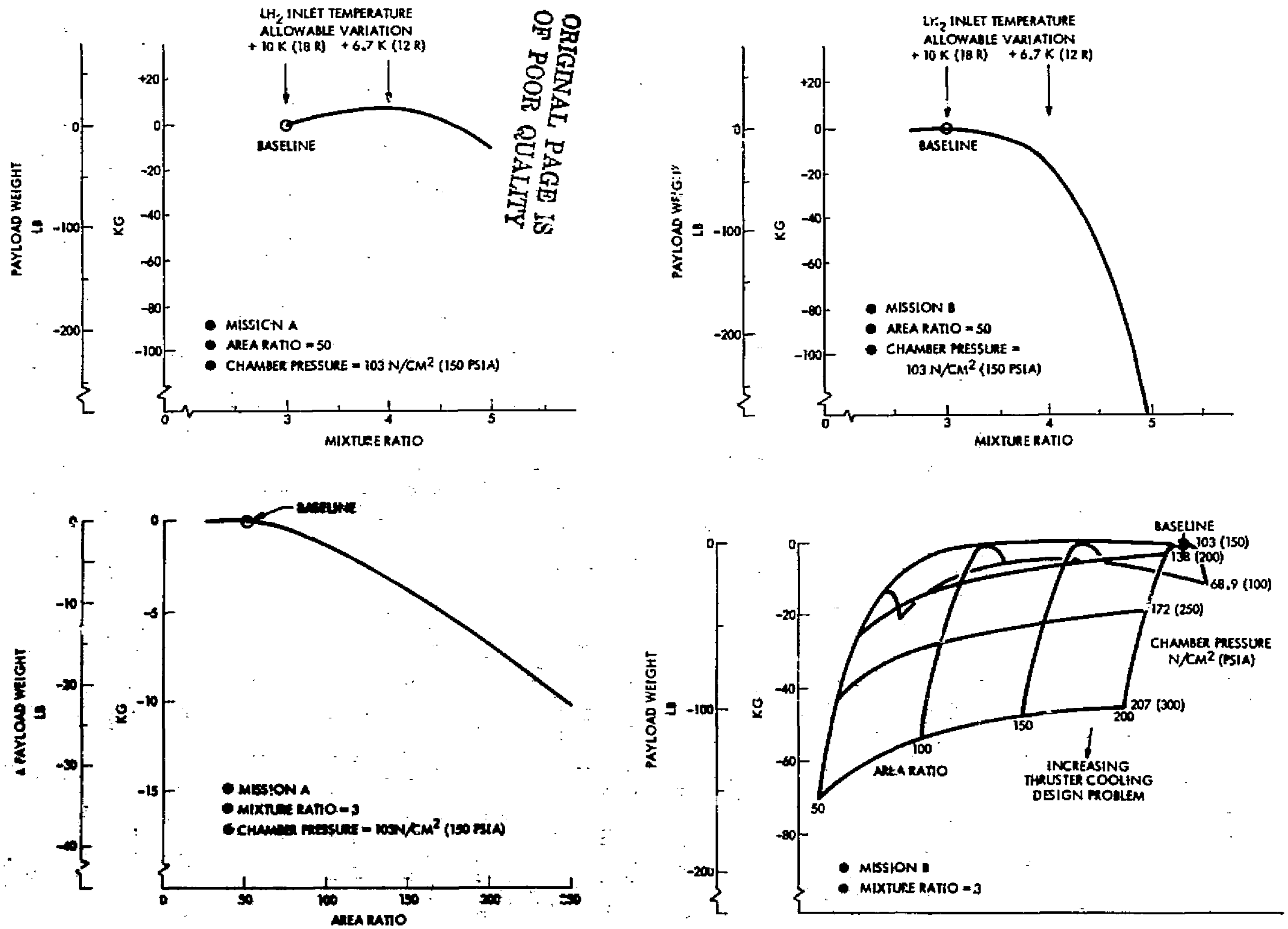


Figure 5-18. Thruster Parametric Design Data

The analysis was initiated by selecting a parametric range of propellant inlet pressures and temperatures. The range for both hydrogen and oxygen inlet pressures was 134 and 169 N/cm^2 (195 and 245 psia). The oxygen and hydrogen inlet temperature ranges were varied between 89 and 111 K (160 to 200 R), and 22 to 50 K (40 to 90 R), respectively.

The variation in engine performance with inlet conditions was established by iterative solution of the equations for flow path pressure loss, chamber pressure, and characteristic exhaust velocity (C*). With constant inlet pressure and flow path pressure loss coefficient, flow rate and mixture ratio are functions of inlet propellant density and chamber pressure. In turn, chamber pressure is a function of flow rate and C* and C* was determined as a function of mixture ratio. The results allowed calculation of corresponding values for specific impulse and thrust.

Figure 5-19 shows the effect of propellant valve inlet temperatures on mixture ratio, chamber pressure, specific impulse, and thrust at three sets of valve inlet pressures. For the purpose of the study, the maximum mixture ratio is limited to 7.0. For both hydrogen and oxygen, the nominal case is 152 N/cm^2 (220 psia).

In the nominal case, since oxygen is a saturated liquid in the applicable ranges, it has a small effect on engine performance. Therefore, the hydrogen inlet temperature was chosen as the independent variable. It is evident that the operating characteristics of the thruster change greatly above a 33 K (60 R) hydrogen valve inlet temperature.

Also shown in the figure are extreme cases of injection pressure imbalance with maximum and minimum inlet pressures at 169 and 135 N/cm^2 (245 and 195 psia), respectively. With the high mixture ratio extreme case, thruster performance starts to degrade rapidly for hydrogen temperatures above 27.7 K (50 R), while with the low mixture ratio extreme case, thruster performance does not change significantly until the hydrogen temperature exceeds 36 K (65 R).

The system design could be made to take advantage of this effect by biasing the pressures so that the hydrogen pressure is at all times higher than the oxidizer pressure. That feature is not implemented in this study but is recommended for application during or after thruster SR&T test data are obtained.

Pre- and Post-Burn Thermal Analysis

A preliminary post-fire thermal analysis of the thruster operating at the baseline conditions was performed to assess thermal feasibility, particularly with regard to the start and restart propellant flow transients. It was found that, with proper attention to thermal isolation of the system and insulation of specific manifold passages, it will be possible to start the engine and restart it with non-detrimental excursions in mixture ratio.

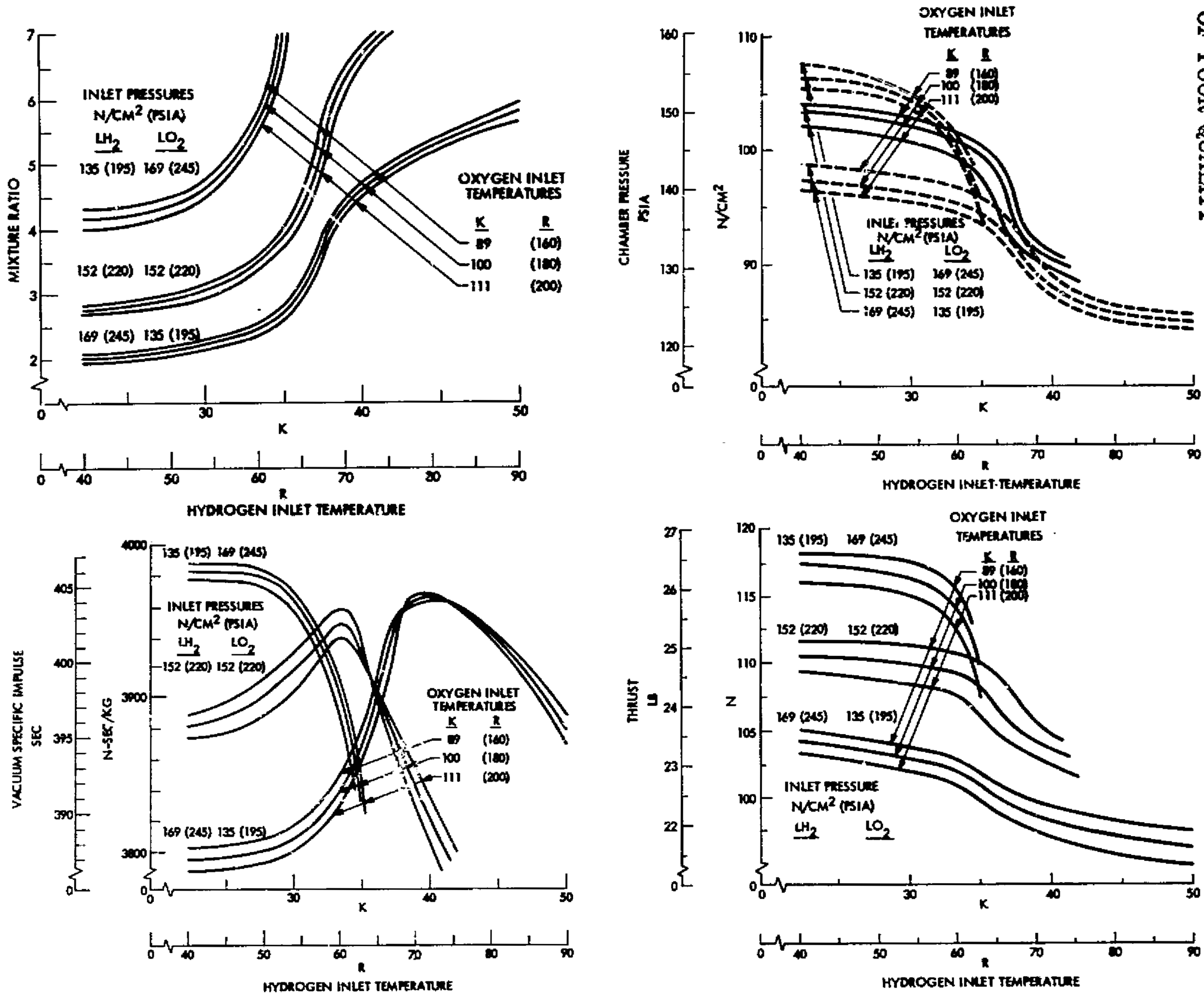


Figure 5-19. Parametric Propellant Inlet Sensitivities

Thermal isolation of the system is achieved using a thin wall bellows with low thermal conductance between the valves and manifolds. Design values correspond to stainless steel bellows having a mean diameter of 0.318 cm (0.125 in.), a wall thickness of 0.013 cm (0.005 in.), and a developed length of 3.8 cm (1.5 in.). Between the manifold and the power source (exciter) a thin wall stainless steel tube will provide the insulation. This tube will be 1.5 cm (0.6 in.) long and 1.5 cm (0.6 in.) in diameter, with a wall thickness of 0.0254 cm (0.010 in.), and will house the power lines and necessary electrical insulation.

It was assumed that the propellant lines at the inlet to the valve would be pre-conditioned to 27.7 K (50 R) and 92 K (165 R) for the fuel and oxidizer, respectively. It was also assumed that the only other significant thermal connection with the vehicle would be by thermal radiation.

A thermal model of the system was constructed using these assumptions, and the results of this model may be seen on Figure 5-20. The temperature transients for three major components, the throat, the thrust chamber wall, and the manifold are shown on this figure as a function of time from shutdown. At shutdown, the temperature of the throat, thrust chamber, and manifolds are 1644 K, 255 K and 61 K (2500 F, 0 F, and -350 F), respectively. These are, of course, mean temperatures and do not represent local variations. The minimum coast time after pulsing is 0.09 hr or 324 sec, at which time the three temperatures are 394 K, 378 K, and 144 K (250 F, 220 F, and -200 F), respectively. Between short pulses, there is no minimum restart time because the

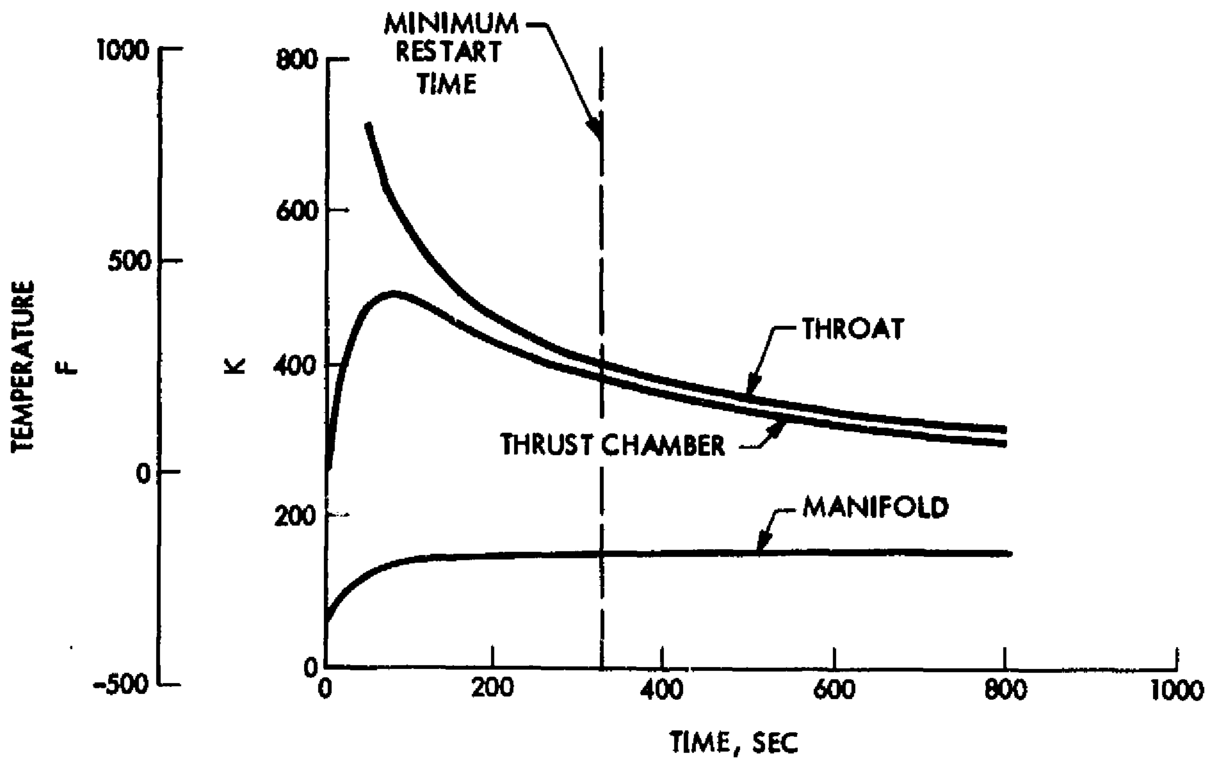


Figure 5-20. Post-Fire Thermal Transients - Uninsulated Manifolds

manifolds are chilled. As was seen on previous figures, when the fuel enters the thrust chamber at temperatures much in excess of 39 K (70 R), excessive (>6) mixture ratio transients will occur leading to possible cooling problems at the throat. High mixture ratio will occur after a long coast since the fuel inlet temperature is near thermal equilibrium with the manifold at 144 K (-200 F). High mixture ratio can cause damage unless the coast time has been long enough to allow the throat to cool to a point where its heat sink capacity will accommodate the high mixture ratio transient period while the manifold is cooling down. This difficulty will be alleviated by providing low effective mass insulators on the manifold walls. This technique, which has been demonstrated on the recent extended temperature range (ETR) thruster with restarts at ambient temperatures, consists of photo-etched platelets bonded to the inside of the manifolds.

By using this concept restarts are permissible at any time after pulsing, because the low effective manifold mass permits cooldown immediately to a temperature of 39 K (70 R) or less.

A critical variable in a conditioned cryogenic thruster is the heat load imposed on the conditioning system. In the present system, the heat load has been minimized to an acceptable level, with capability of reducing it even more. The maximum steady state heat loads for all 16 thrusters are 1.5 and 0.73 watts (5.0 and 2.5 Btu/hr) for the fuel and oxidizer circuits, respectively.

Impulse Bit Analysis

The impulse bit analysis is based on the results of testing conducted for Contract NAS3-16775 (Reference 16) with LO_2/LH_2 propellants. Data from igniter tests on that contract and preliminary design analyses show that the 80 percent specific impulse goal can be achieved at 111-N (25-lb) thrust for an electrical pulse width of 25 msec. This corresponds to a minimum impulse bit of 2.2 N-sec (0.5 lb-sec) at a nominal thrust level of 111 N (25 lb).

The test data obtained for the contractual effort mentioned also showed that there was no significant difference in total impulse between the first and subsequent pulses. However, a difference in pulsing performance was noted. Figure 5-21 shows the variation of pulse performance as a function of impulse bit for a nominal thrust of 111 N (25 lb). This figure was derived from the test data cited. Pulsing performance is higher when manifolds are chilled down, which is typical of a pulse train. The lower performance is typical of the first pulse or pulses preceded by long coast periods resulting in thermal heat soakback to manifolds. The differences in performance are due to a greater shift in the oxidizer-rich mixture ratio in the first pulse when manifolds are warm than in subsequent pulses when manifolds are colder.

INTEGRATED APS PROPELLANT SUPPLY CONDITIONS

To aid in the design and performance analysis of the integrated APS concepts, the main tank propellant conditions were defined. These conditions include the tank pressure and temperature profiles, as well as APS subcooling and supply line requirements.

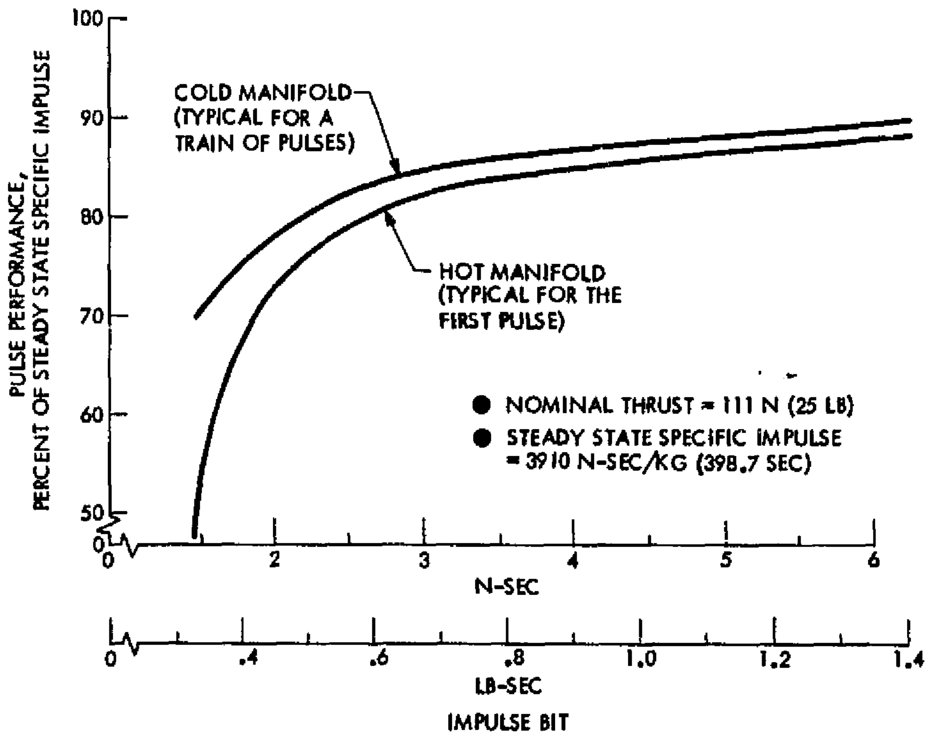
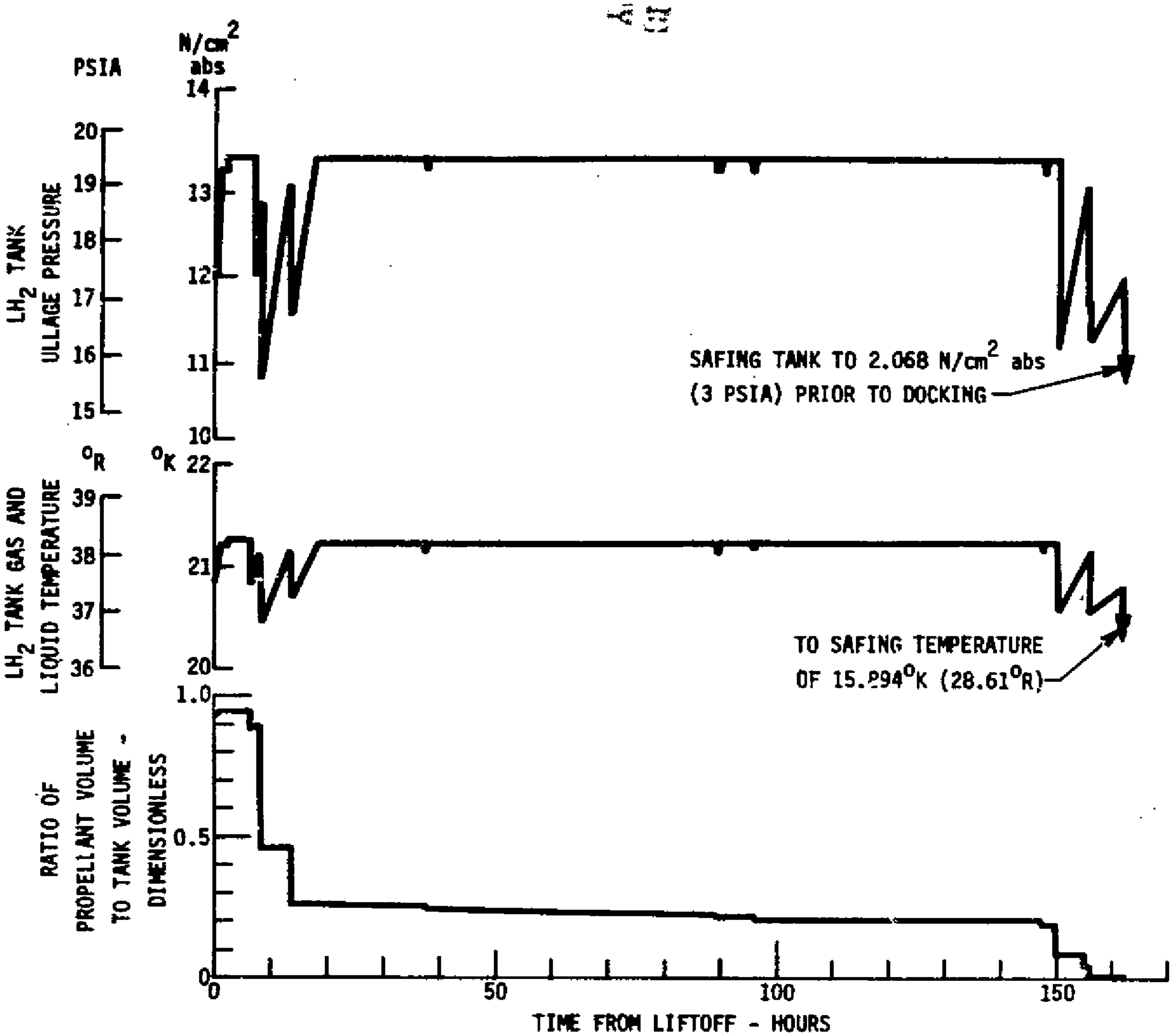


Figure 5-21. Effect of Impulse Bit on Pulse Performance

The initial MPS pressurization system concept used for this study was a zero NPSH blowdown method utilizing a Category IIA RL-10 engine. This concept used the tank head idle mode of the engine to settle the propellants. Figures 5-22 and 5-23 present the resulting tank conditions utilizing the mission timeline and propellant usages from Table 4-4.

Because the RL-10 IIA engine requires a minimum of 11 N/cm^2 (16 psia) at engine start, the zero NPSH blowdown concept is mission-dependent. Self-pressurization during a sustained MPS burn could result in LOX and LH_2 propellant pressures at engine cutoff less than the 11 N/cm^2 (16 psia) minimum required for engine restart. Thus, mission flexibility would be constrained due to the waiting period required for propellant repressurization by tank insulation heat leaks. THIM thrusting for propellant settling would further reduce the tank pressure if the propellant were adversely located and ullage gas escaped through the engine.

To improve mission flexibility, an autogenously pressurized Category IIB RL-10 engine was selected which requires a 2/15 (oxygen/fuel) minimum NPSH at mainstage start. This engine also requires a minimum pressure of 11 N/cm^2 (16 psia). Prior to mainstage, the IIB derivative engine is capable of operating in THIM similar to the IIA derivative; thus, it can settle the propellants in this mode. In addition, the IIB derivative has a pumped idle mode which can pressurize the ullage prior to mainstage operations in a so-called bootstrap manner. The ullages continue to be pressurized autogenously during mainstage operations.



HEAT LOADS JOULES/HR (BTU/HR)

1. FIRST HOUR: 1.28816×10^7 (12,200)
2. SUBSEQUENT HOURS: 3.04091×10^5 (288)
3. DURING ENGINE FIRING: 3.16761×10^5 (300)

NOTES

1. TOTAL TANK VOLUME: 54.45330 m^3 (1923 FT^3)
2. INITIAL LIQUID MASS: 3583.38 Kg (7900 LBS_m)
3. GAS AND LIQUID AT SATURATED CONDITION AT ALL TIMES
4. RELIEF VALVE MODULATES AT 13.4448 N/cm^2 abs (19.5 PSIA)

Figure 5-22. MPS Hydrogen Tank Properties Profile

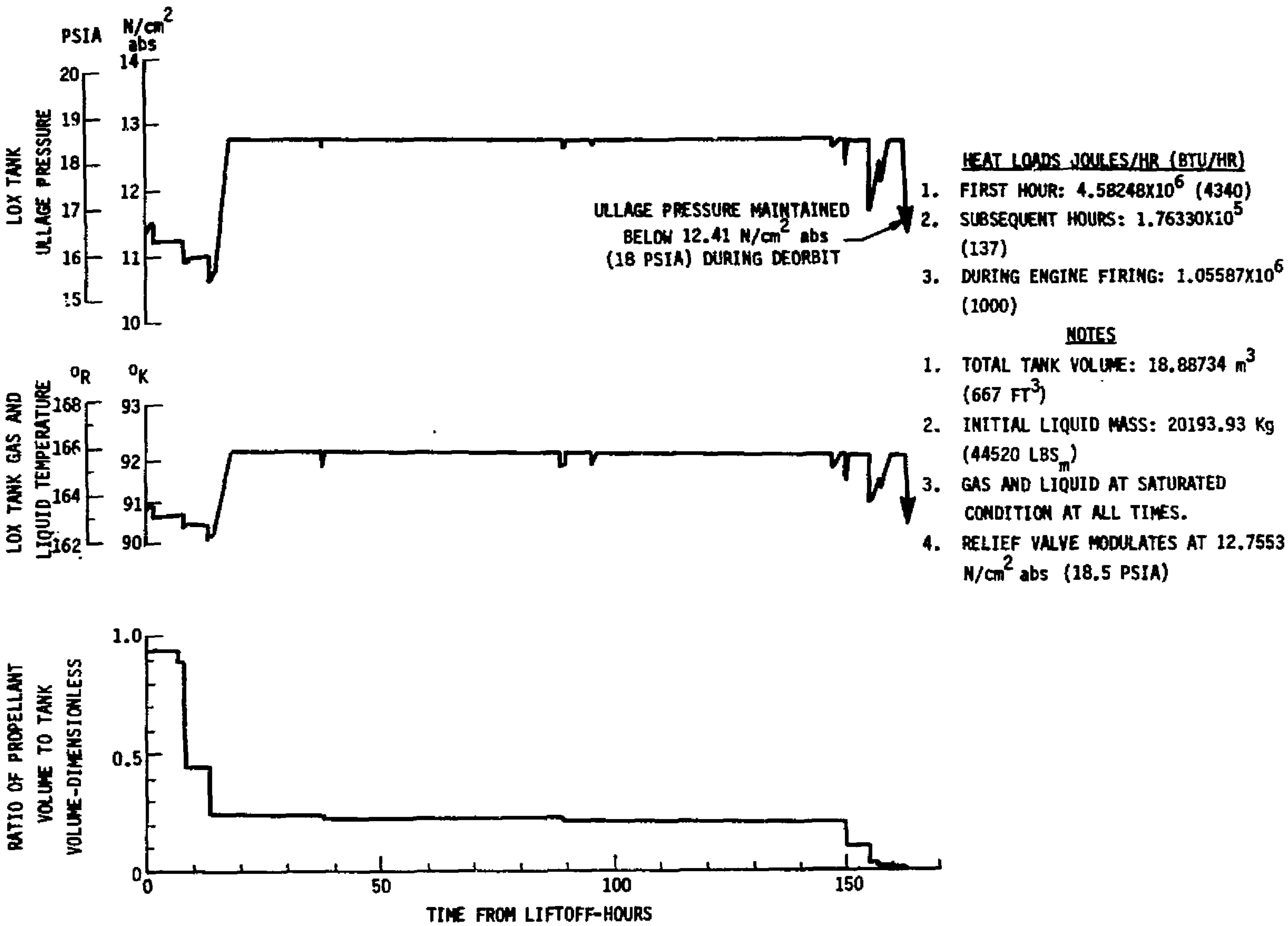


Figure 5-23. MPS Oxygen Tank Properties Profile

Table 5-17 summarizes the methods used to settle propellants by several Tug system investigators. The decision to utilize the APS for propellant settling gave the design the added dimension of having a redundant means of settling the propellants. In addition, the table presents the various systems used to control ullage pressure which were part of those designs.

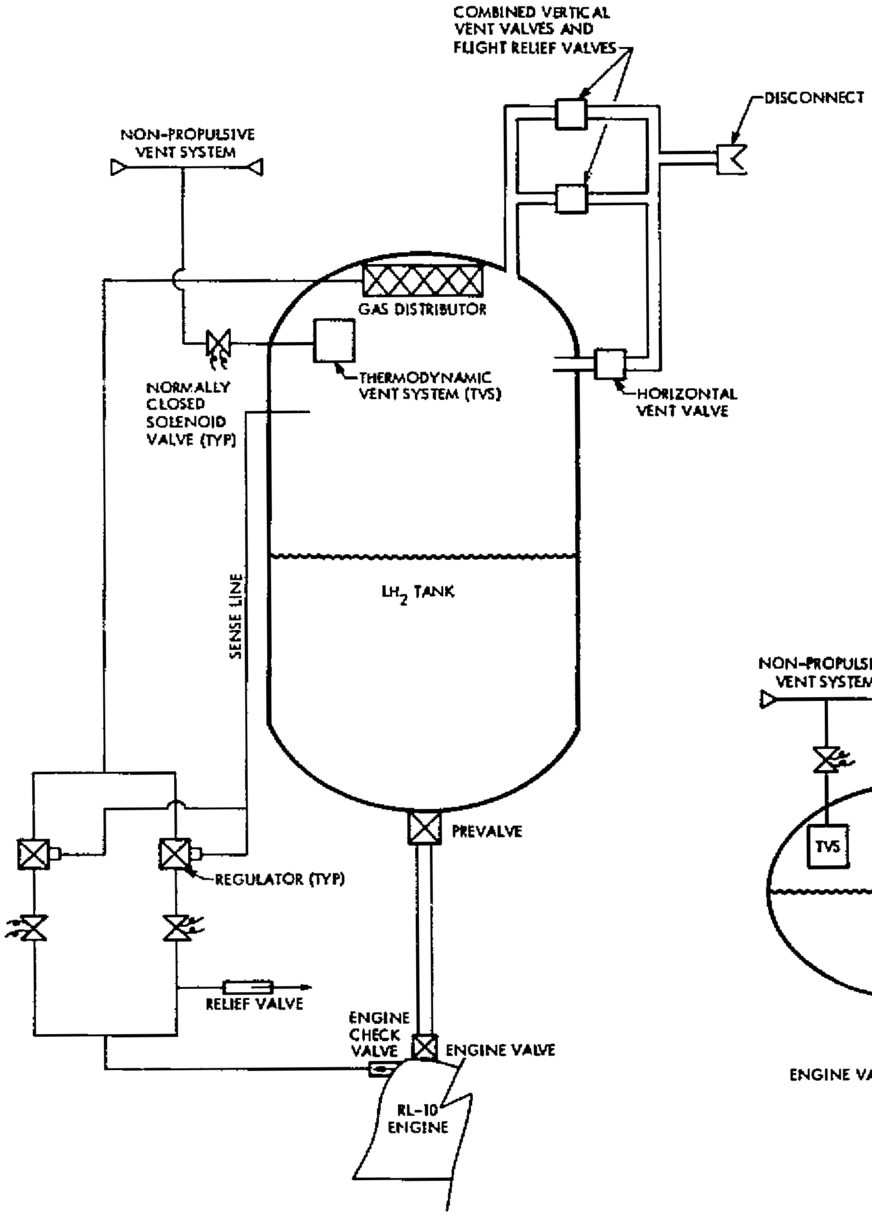
The autogenous pressurization system selected is similar to the system presented in the MSFC report (Reference 17). Figure 5-24 presents a schematic representation of the LH_2 and LOX tank MPS pressurization and vent subsystems. Redundancy is provided in the regulation of both ullage pressurization systems by two regulators pneumatically in parallel protecting against the failed-closed regulator mode, and a shutoff valve in each leg that will protect against the failed-open mode. Pneumatically in parallel relief valves protect against system overpressurization. Control of GH_2 and GO_2 flow from the main engine tapoff is by single nonredundant solenoid valves.

The LH_2 and LOX tank vent and relief systems are composed of two subsystems. The primary vent systems are functional during the loading, ascent, and positive acceleration periods of Tug operation. On the LH_2 tank, it is also utilized in the event of an aborted mission requiring that the Orbiter land with the Tug LH_2 tank full. The LOX tank is dumped prior to an abort landing and horizontal venting is not required.

The secondary tank vent system is required to vent the LH_2 and LOX tanks with minimal liquid losses during periods of zero or low acceleration when propellants are not settled and venting of either gas, liquid, or both is a possibility. A zero-thermodynamic vent system (TVS) is baselined in both

Table 5-17. Main Propulsion System Design Comparison

	SYSTEM DESIGNS					
				CRYO APS STUDY		
	MACDAC	CONVAIR	MSFC	BASELINE	ALTERNATES	
ENGINE TYPE						
RL-10 CATEGORY IIA	●					
RL-10 CATEGORY IIB		●	●	●	●	●
PRESSURIZATION - MAINSTAGE						
BLOWDOWN + BOOST PUMP		●				
BLOWDOWN + H_E 1ST FLIGHT	●					
AUTOGENOUS			●	●	●	●
PREPRESSURIZATION						
NONE	●	●				
H_E			●			
BOOSTRAP AUTOGENOUS (PIM)				●	●	●
ENGINE CHILLDOWN						
THIM	●	●	●	●	●	●
PROPELLANT ACQUISITION						
THIM SELF-SETTLING	●		●			●
START BASKET		●			●	
APS SETTLING				●		



COMPONENT OPERATING LIMITS	LH_2 TANK	LOX TANK
FLIGHT RELIEF VALVES		
• CRACK, N/cm^2 abs (PSIA)	13.10 - 13.44 (19 - 19.5)	13.44 - 13.79 (19.5 - 20)
• RESEAT, N/cm^2 abs (PSIA)	12.76 - 13.10 (18.5 - 19)	13.10 - 13.44 (19 - 19.5)
REGULATOR		
• SET POINT, N/cm^2 abs (PSIA)	12.41 (18)	12.76 (18.5)
• BANDWIDTH, N/cm^2 abs (PSI)	±0.345 (±0.5)	±0.345 (±0.5)
THERMODYNAMIC VENT SYSTEM CONTROL RANGE		
• NOMINAL, N/cm^2 abs (PSIA)	11.38 (16.5)	11.38 (16.5)
• TOLERANCE, N/cm^2 (PSI)	±0.345 (±0.5)	±0.345 (±0.5)

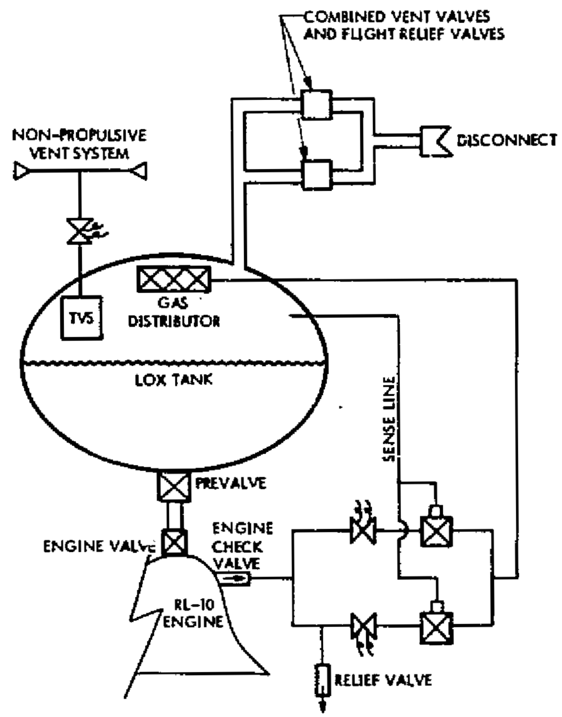


Figure 5-24. MPS Pressurization and Vent Systems Schematic

tanks due to its light weight, good technology base, and mission flexibility (i.e., it is self-compensating for variations in heat leak). The thermodynamic vent requires electrical power for mixing, and vents only gas even though gas and liquid may be present. Technology and a practical demonstration of the thermodynamic vent system already has been accomplished. Redundancy in the LO_2 and LH_2 thermodynamic vent systems is provided through the use of dual valving.

As seen in Figure 5-24, TVS control range is set such that the minimum ullage pressure satisfies the minimum engine inlet pressure requirement. The pressurization system regulator minimum control level was determined by the engine NPSH requirements. Rapid pressurization of the LOX tank ullage 0.69 N/cm^2 (1 psi) above the TVS pressure level results in slightly more than 0.6m (2 ft) of NPSH. Similarly, pressurizing the LH_2 tank ullage 0.345 N/cm^2 (0.5 psi) above the TVS pressure level results in slightly more than 4.6 m (15 ft) of NPSH. In addition, it was considered necessary to allow the regulator a tolerance of $\pm$ 0.345 N/cm^2 ($\pm$ 0.5 psia) about its setpoint; the regulator limits reflect this choice. Finally, the flight relief valve modulation band was determined by the maximum regulator control pressure and the structural limits of the tank. The 13.79 N/cm^2 (20 psia) maximum LOX tank ullage pressure represents an increase from the self-pressurized system concept maximum level of 12.76 N/cm^2 (18.5 psia).

In summary, the MPS pressurization and vent system will maintain the propellant and ullage fluids within the limits presented in Table 5-18.

Table 5-18. MPS Pressure and Temperature Limits During Mission

Parameter	LH_2 Tank	LOX Tank
Pressure, N/cm^2 (psia)		
Maximum	13.45 (19.5)	13.79 (20)
Minimum*	11.03 (16)	11.03 (16)
Ullage Temperature, K (R)		
Maximum	94.44 (170)	138.89 (250)
Minimum*	20.55 (37)	91.00 (163.8)
Propellant Temperature, K (R)		
Maximum	20.77 (37.38)	91.56 (164.8)
Minimum*	20.55 (37)	91.00 (163.8)

*From liftoff until the heat leak into the tanks increase the pressure and temperature into the TVS control range, the fluids are colder. Liftoff conditions: LOX tank 10.1 N/cm^2 (14.7 psia), saturated; LH_2 tank 10.5 N/cm^2 (15.2 psia), saturated.

The calculated LH_2 and LOX main tank pressure and temperature profiles adequately meet all MPS engine requirements; however, the fluid extracted from the tanks to support the APS must be subcooled to enable the APS pumping system to operate properly. The subcooling is accomplished in a zero-g reservoir with a separate TVS. With the assumption that the LH_2 pump inlet NPSP requirement is 0.7 N/cm^2 (1 psi), the required minimum subcooling is 0.2 K (0.4 R) during APS usage. If the LOX pump inlet NPSP requirement is 114 N/cm^2 (2 psia), the minimum required subcooling is 1.1 K (2 R) during APS usage. The subcooling requirement assures negligible pressure loss from heating or frictional flow in the suction line from the zero-g reservoir to the pump inlet. If the pressure loss in not negligible, additional subcooling will be required.

To assure that the pump suction line does not cause an appreciable pressure loss due to friction or heating, the pump inlet should be inside the accumulator tank and the suction line diameter should be no less than 1.5 cm (0.6 in.). In this way, the suction line is no warmer than the propellant bulk temperature and the frictional pressure loss is minimal.

PROPELLANT ACQUISITION

This section presents the detailed analytical results supporting the selection and conceptual design of an overboard vent refill capillary device for the APS zero-g reservoir. The topics include concept selection trade analysis, zero-g reservoir sizing, capillary screen design analysis, zero-g reservoir location, and MPS start basket evaluation.

Concept Selection Trade Analysis

The concepts evaluated (Section 5.1) can be classified into two basic categories: nonvented (nonrefillable) and vented (refillable) capillary devices. The nonrefillable concepts were eliminated for being heavy and lacking mission flexibility. The refillable concepts can be further classified as either internally vented or overboard vented. These two categories are illustrated diagramatically in Figure 5-25.

The internal screen vented concept is the simplest of the refillable concepts studied, as it has no valves. It must be designed to vent vapor back into the tank during engine firing, while being hydrodynamically stable during orbit operation. A typical design schematic is shown in Figure 5-26. The narrow central vent tube is often used to hasten refilling and to minimize trapped residual.

However, analysis shows that the vent tube diameter must be carefully chosen. If the diameter is too small, dynamic and dissipative forces are created which can impede refill. The height of the vent tube is limited by weight, liquid level in the main tank during the final burns, and stability requirements during orbital maneuvers. The orbital maneuver which determines screen mesh size is the +X translational maneuver with four engines firing. For the smallest vehicle weight, this yields an acceleration level of only 0.0151 g. Total vertical height of the reservoirs is about 0.76 m (2.5 ft) for LH_2 and 0.38 m (1.25 ft) for LOX. As the acceleration level is low, a coarse screen such as a 24 x 110 mesh plain Dutch twill can be used. To supply APS and thermodynamic vent propellant, capillary collectors as shown in Figure 5-6 are required within the reservoir. Thermal control provisions are required to maintain the propellant within the reservoir subcooled compared to the contents of the main tank.

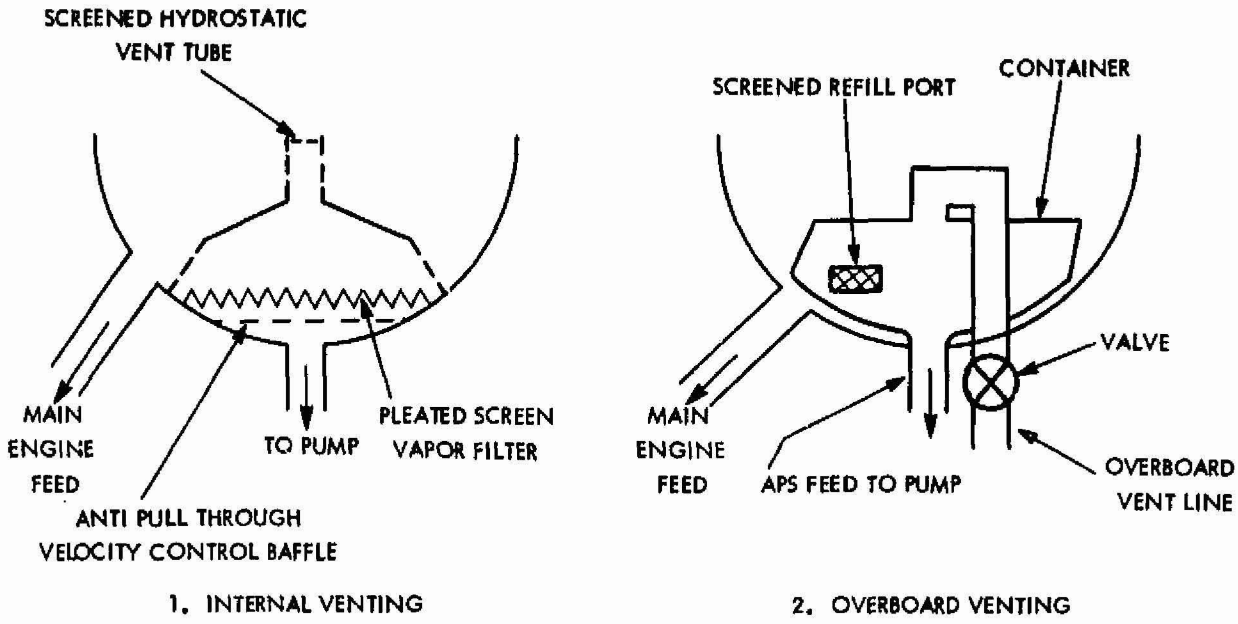


Figure 5-25. Refillable Designs, Generic Types

An analysis has been conducted of refill rates for the LOX and LH_2 reservoirs for APS and MPS burns. The basic equations are developed below. Consider the streamtube path from Position 1 to Position 2 as identified in Figure 5-26 using the generalized Bernoulli equation:

$$\int_1^3 \frac{\partial u}{\partial t}\, ds + \frac{u_3^2 - u_1^2}{2} + g\,(x_3 - x_1) + \frac{P_3 - P_1}{\rho} + F' = 0 \qquad (1)$$

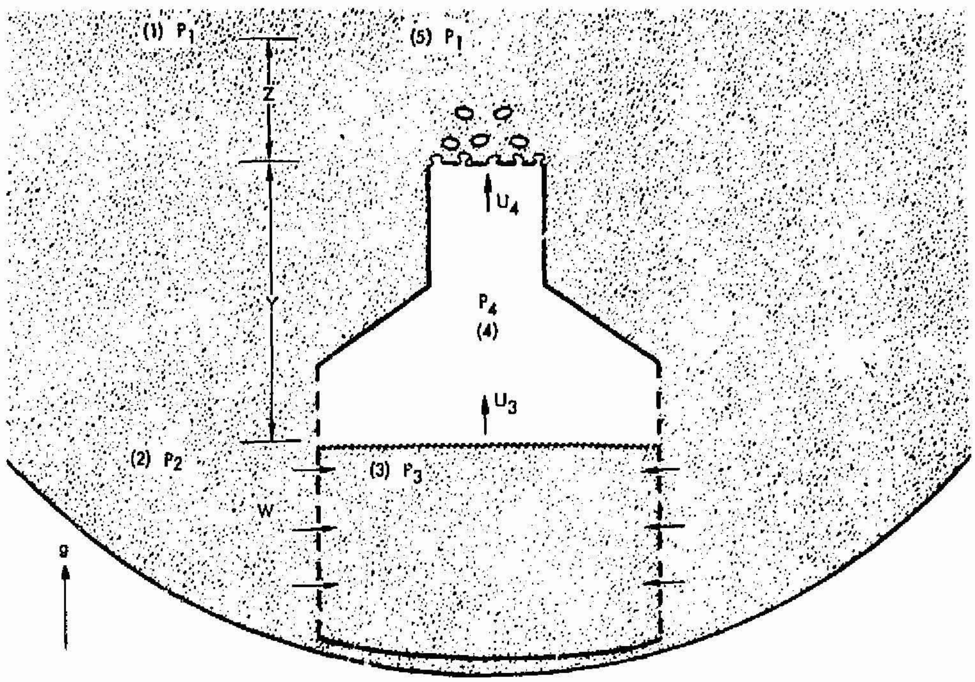


Figure 5-26. Schematic of Refill Flow for Internally Vented APS Reservoir

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where

u = velocity

x = vertical coordinate

s = the streamtube coordinate in the direction of motion

g = the acceleration

P = pressure

t = time

ρ = density of the liquid

F' is a dissipative term due to flow of liquid through the screen between Positions 2 and 3. From References 18 and 19:

$$F' = E\omega + F\omega^2 \tag{2}$$

where ω = velocity through the screen

E = the coefficient of the laminar pressure loss term

F = the coefficient of the turbulent pressure loss term

The velocity at Position 1 is essentially zero, and

$$u_1 = 0 \tag{3}$$

The transient term is important initially because it represents the pressure required to accelerate the flow from rest to steady or quasi-steady velocity. Its inclusion will result in a slower filling. If it is neglected and it is found that filling is still too slow, then it is fruitless to proceed further. Therefore, it was assumed that

$$\int_1^3 \frac{\partial u}{\partial t} \, ds = 0 \tag{4}$$

The velocity at Position 3 can be related to the velocity at Position 4 by continuity. The symbol A refers to the flow area at the station sbuscripted.

$$u_3 = \frac{A_4}{A_3} u_4 \tag{5}$$

Similarly, ω can be related to u_4 by

$$\omega = \frac{A_4}{A} u_4 \tag{6}$$

where $A\omega$ is the wetted screen area through which liquid flows into the reservoir. Substituting Equations 2 through 6 into 1 and solving for pressure difference, there results

$$\frac{P_3 - P_1}{\rho} = -\left(\frac{A_4}{A_3}\right)^2 u_4^2 + g\ (z + y) - \left[E\left(\frac{A_4}{A\omega}\right) u_4 + F\left(\frac{A_4}{A\omega}\right)^2 u_4^2\right] \tag{7}$$

In most derivations of this type, the hydrostatic term is taken as gy. The addition of the term gz is explained below.

The pressure difference, $P_3 - P_1$, can also be evaluated by considering the dynamics of the gas leaving the reservoir. Position 5 is taken to be sufficiently high above the vent tube that bubble dynamics do not affect the pressure field. Thus, the pressure at Position 5 is the same as that at Position 1, namely P_1. The pressure losses from Position 3 to Position 4 is the contraction loss in entering the vent tube; this is generally small compared to the losses from Position 4 to Position 5. The mechanism of the flow from 4 to 5 is not completely understood. However, enough is known to develop limiting conditions. Testing done at Rockwell Space Division (Reference 20) for a different problem with similar fluid mechanics has shown that for a wide range of flow rates the pressure drop across a wetted screen is somewhat greater than the bubble pressure. It is expected that for flow rates that are very high, the screen pressure drop is that for vapor flow alone (the screen is dry for this case). The bubbles passing through the screen create a dynamic pressure field above the screen; this is the "virtual mass effect" associated with accelerating the liquid above. The pressure drop due to this effect has a upper limit of about one momentum head ($\Delta P_{max} = \rho\ u_4^2$), where ρ is the liquid density. Thus,

$$P_3 - P_1 \cong P_4 - P_5 = \left\{\begin{matrix} \rho_v\ (Eu_4 + Fu_4^2) \\ \Delta P_B \end{matrix}\right\} + K\rho u_4^2 \tag{8}$$

where the bracketed expression infers that the pressure loss has a lower limit of the bubble pressure and an upper limit of vapor flow alone and K has an upper limit of unity. Here ρ_v is the vapor density and ΔP_B is the bubble pressure. Equating Equations 7 and 8:

$$\rho\left(\frac{A_4}{A_3}\right)^2 u_4^2 + \rho g\ (z + y) + \rho\left[E\left(\frac{A_4}{A}\right) u_4 + F\left(\frac{A_4}{A_5}\right)^2 u_4^2\right] \tag{9}$$

$$= \left\{\begin{matrix} \rho_v\ (Eu_4 + Fu_4^2) \\ \Delta P_B \end{matrix}\right\} + K\rho u_4^2$$

The terms g z and K ρu_4^2 tend to cancel each other. As bubble dynamics increase so does the wake region above the screen. Thus, it may be argued that these terms tend to cancel each other, with the inertial term always larger than the hydrostatic head term.

This rationale has been applied to evaluate the internally vented APS reservoir. It follows immediately that refill during APS firing is a practical impossibility. This is because the hydrostatic head "driving force" term, ρ gy, is less than the bubble pressure. If this were not the case, the screen would be destabilized during APS maneuvers. Thus, the only possibility to refill during APS burns using internal venting is to replace the screen at the top of the vent tube with a valve. The valve is opened after settling is completed and closed when refill is completed. For hydrogen, the hydrostatic head driving pressure is on the order of 0.00028 N/cm^2 (0.0004 psi). Consider the shortest time available for refill after complete propellant settling, which occurs during the 64-sec APS translation following mission orbit insertion at 147.55 hours. Refill can be at least partially accomplished if a 5 to 7.6-cm (2 to 3-in.) diameter vent is used. A valve with a 5 to 7.6-cm (2 to 3-in.) orifice is probably excessively heavy and complex.

Considering main engine burns, the delta velocity maneuvers shown in Table 5-11 are all long enough to accomplish refill for the screened reservoir of Figure 5.26. The most difficult refill operations are the very short Mission A orbit maneuver burns, particularly the midcourse correction at 8.02 hours. The time available for refill and the maximum acceleration for this burn are:

	Time (sec)	Acceleration (g)
APS settling	54.43	0.0028
THIM burn	13.23	0.105

The hydrogen reservoir propellant profile on Figure 5-7 shows that 38 percent of the reservoir (0.02 m^3 or 0.847 ft^3) must be refilled during this burn. Analyses conducted for this case show that the refill flowrate is highly dependent on the height and diameter of the vent tube; even a 15 percent refill appears to be marginal for this case.

Reservoir refill by internal screen venting could be accomplished by limiting the refill operation to main engine delta velocity maneuvers when the acceleration level and duration would be great enough to provide the necessary refill propellant flow. This would mean that the LOX and LH_2 reservoirs would have to be large enough to supply all the propellant required by the APS during unsettled operation. Reservoir capacity would have to increase by approximately 380 percent, however, resulting in a 20-kg (44-lb) increase in APS dry weight. By contrast, the overboard vented refillable approach results in only a 0.9-kg (2-lb) vent loss for the entire mission.

Thus, the internally vented basket, while having size and weight advantages to the passive (nonrefillable) designs, has some major drawbacks. It cannot be refilled during the APS burns because the hydrostatic head driving force is less than the screen vent tube bubble pressure. Refill can be accomplished during main engine delta velocity burns, but is problematic for at least one of the MPS orbit maneuver burns because it is very short--13.23 sec (the midcourse correction at 8.02 hours). Use of a large vent valve, rather than screen material, at the top of the vent tube will hasten refilling for APS and MPS burns, but is prohibitive in weight. Overboard vent has been selected as the most weight-effective approach to zero-g reservoir refill.

Zero-g Reservoir Sizing

For the propellant acquisition system in the zero-g reservoir, a conservative design and sizing approach was used. Propellant is drawn from the retention reservoir for attitude control, APS translations, propellant settling prior to main engine burns, and thermodynamic venting (LH_2 only). Capillary devices are used within the reservoir to provide vapor-free liquid for all these cases. However, under the zero- and low-gravity conditions of space, the liquid fed out of the retention reservoir is not necessarily displaced by liquid. If vapor is present outside of the reservoir, it will flow into the reservoir in preference to liquid.

In sizing the reservoir a worst-case propellant orientation was assumed: propellant adversely located during attitude control maneuvers and thermodynamic venting flow, and initially for APS translations and MPS settling. Initial APS operation is sustained using propellant from the retention reservoir until propellant settling is accomplished. For the APS translations, after settling is accomplished, propellant flows into the retention reservoir replacing propellant flowing to the APS engines until the APS translation is completed. For MPS settling maneuvers, the APS is fired until main propellant is settled, at which time the MPS firing is initiated and APS firing is terminated.

For attitude control and thermodynamic venting, all propellant used is displaced by vapor. For APS translations and MPS settling maneuvers, vapor displaces liquid in the reservoir until settling is accomplished. The hydrodynamics of linear acceleration settling has been the subject of considerable investigation in the last decade. For the purposes of this study, the empirical "four free fall" rule has been found to be sufficiently accurate for conceptual design. That is, settling time, t_s, is given by:

$$t_s = \tau\sqrt{\frac{2h}{g}} \tag{10}$$

where

h = the distance of the liquid interface from the tank bottom

g = the settling acceleration

τ = an empirical constant with a value of four in this case

The volume of vapor which enters the retention cans is given as follows.
For LH_2:

$$V_{GH_2} = \left(\frac{W}{\rho_{LH_2}}\right) \left(\frac{1}{(M + 1)}\right) t_s \tag{11}$$

For LOX:

$$V_{GOX} = \left(\frac{W}{\rho_{LOX}}\right) \left(\frac{M}{M + 1}\right) t_s \tag{12}$$

where

V_{GH2} = the volume of hydrogen vapor which enters the LH_2 reservoir during settling

V_{GOX} = the volume of oxygen vapor which enters the LOX reservoir during settling

W = the mass flow rate of LH_2 + LOX during the APS settling burn

ρ_{LH_2} = the density of liquid hydrogen

ρ_{LOX} = the density of liquid oxygen

= the mixture ratio

t_s = the settling time

As geometry and size are different for the LH_2 and LOX main tanks, the settling time is different for each. The mass flow rate is that required to fire four +X APS engines with a thrust of 111 N (25 lb) each. Settling can be accomplished with only two +X APS engines; indeed, while settling time is longer, total propellant utilized and hence total gas entry into the reservoir is less. A two-level settling process has been considered in which two engines are used initially, thus reducing the liquid momentum and bubble formation propensity of the settling flow. All four engines are used to complete the settling process, hastening bubble rise and reducing slosh amplitude. These refinements can be investigated in subsequent program phases. It was considered that four-engine settling was the simplest, most reliable approach--with two-engine settling as backup in the event of a single APS engine failure.

The baseline design is a refillable reservoir containing capillary devices. Refill can be accomplished during any programmed APS or MPS burn. To obtain the required pressure difference to accomplish refill in the available time (duration of engine firing subsequent to settling) overboard venting is used. The size of the reservoir is determined by the maximum vapor entry between two sequential engine burns. For both the LOX and LH_2 systems, maximum vapor entry occurred between the second and third orbit maneuver burns (between the first phasing orbit insertion and mission orbit insertion burns).

For the LOX system, this vapor consists of the sum of that which entered during attitude control maneuvers (0.0014 m^3 or 0.049 ft^3) and during the APS settling preceeding the orbit insertion burn (0.0029 m^3 or 0.104 ft^3), for a total of 0.0043 m^3 (0.153 ft^3). It should be noted that this quantity is independent of whether the orbit insertion burn is accomplished with the APS or MPS. Thus, reservoir size is the same for Missions A and B. For the LH_2 system, the vapor ingested into the reservoir is the sum of that due to attitude control maneuvers (0.0034 m^3 or 0.12 ft^3), APS settling (0.0207 m^3 or 0.73 ft^3), and thermodynamic venting (0.0210 m^3 or 0.74 ft^3), for a total of 0.0451 m^3 (1.59 ft^3). Again, this quantity is the same for Missions A and B.

The reservoir is made up of two compartments, an upper refillable compartment and a lower compartment for redundancy. The compartments were sized as follows:

	Volume, m^3 (ft^3)	
	LOX	LH_2
Maximum vapor ingested	0.0043 (0.153)	0.0450 (1.59)
Margin (20%)	0.0009 (0.031)	0.0093 (0.33)
Hardware	0.0006 (0.020)	0.0014 (0.05)
Lower compartment	0.0021 (0.075)	0.0099 (0.35)
Total	0.0079 (0.279)	0.0656 (2.32)

The 20 percent margin allowance accounts for design uncertainties, ullage volume at end of refill, and fabrication and ground loading tolerances. A timeline for worst-case propellant quantity in the LH_2 reservoir is shown in Figure 5-7. The greatest depletion occurs between the first phasing orbit insertion and mission orbit insertion burns, as discussed previously. The steep inverted spikes represent gas ingested during settling; the lines with the moderate negative slopes represent gas ingestion during attitude control and thermodynamic venting. The next worst case of depletion, almost as bad as the worst case, occurs between the second phasing orbit insertion and mission orbit insertion burns. The timeline for propellant quantity in the LOX reservoir is similar to that of the hydrogen with one exception. There is no flow of LOX for thermodynamic venting; the LOX reservoir, accumulator, and lines are cooled by hydrogen heated above the LOX melting point.

Capillary Screen Design Analysis

Figure 5-6 in Section 5.2, Integrated Concept Description, presents a conceptual design of the LOX and LH_2 zero-g reservoirs. The toroidal acquisition tube design used for the upper compartment provides a communication path for liquid from the upper compartment to the lower one. The upper and lower compartments are separated by a 165 x 800 mesh capillary screen. The design of these screens is dictated by the functional requirements for the following APS operational phases:

- **Ground loading and drainage**
- **Feedout during APS operation**
- **Liquid retention during propulsive maneuvers**
- **Orbital refill**

Ground Loading and Drainage

Ground loading is accomplished through the isolation valve at the bottom of zero-g reservoir not shown on Figure 5-6. Prior to loading, the reservoir is inerted with helium. The helium is then purged with GOX or GH_2, for the oxygen and hydrogen reservoir, respectively. In each case, venting is accomplished through the overboard vent valve. Following this, LOX and LH_2 are loaded through the isolation valve, with venting through the overboard vent. As the capillary collector is covered with self-wicking 325 x 2300 mesh dutch twill screen, it is possible that although the reservoir is filled with liquid, some vapor is trapped within the collector tubes. Loading then will involve three phases: cooldown, tank fill, and collector fill. The last phase is accomplished by condensation of vapor within the collector and displacement by liquid. Analytic procedures to calculate fill times have been developed for the Space Shuttle supercritical cryogenic tankage and can be modified to include the capillary device condensation process. These results will be confirmed by developmental testing.

Drainage can be accomplished through the isolation valve, by back-flowing through the fill system. Propellant vapor or helium is drawn in through the vent line to displace the liquid. Drainage can be hastened by introducing the vapor or helium at elevated pressure, within the constraints of tank structural limitations. Due to the bubble pressure and wicking characteristics of the collector tube screen, the liquid within the tubes will be last to drain. This can be hastened by using warm (relative to the cryogen) pressurant to destabilize the collector tube screen.

Feedout During APS Operation

The screened collector tubes when wetted will pass liquid rather than vapor. For vapor to enter, the local pressure drop must exceed the screen bubble pressure. In general, vapor will not enter if:

$$\Delta P_B \geq \Delta P_E + \Delta P_D + \Delta P_F + \Delta P_H + \Delta P_T \qquad (13)$$

where

ΔP_B = the bubble pressure of the screen

ΔP_E = the entry pressure loss due to flowing through the screen

ΔP_D = the pressure loss due to the need to turn the entering flow

ΔP_F = the frictional pressure losses

ΔP_H = the hydrostatic head pressure loss

ΔP_T = the pressure loss due to start up or shutdown transients

These various losses have been treated in many studies of capillary devices; however, it is only recently that enough information existed to determine the magnitude of the various terms. The entry loss term is the best understood and has been experimentally evaluated by various investigators (References 18 and 19). A model which accounts for the pressure loss in flowing through the screen as the sum of a laminar and a turbulent term is widely accepted. The pressure loss due to turning the flow entering the collector and imparting a velocity to it at right angles to the original flow direction is equal to two velocity heads as shown in Reference 21 by employing the conservation of linear momentum equation. Frictional losses due to flow within collectors has been unestimated by many investigators. It was shown first by Hines (Reference 21) and more recently in an extensive treatment by Cady (Reference 19) that the screen wires constitute roughness elements for flow parallel to the screens. Depending on the screen, friction factors for the externally unwetted portion of collector tubes were found to be 50 to 500 percent higher than for smooth channels. Hines suggests that friction factors in the unwetted portion of a collector are highly dependent on the type and alignment of the screen material and on the flow regime. Cady further conjectures that the flow through the screen at the beginning of the channel flow annulus, though small, could act as a turbulence generator, thus triggering transition to turbulent flow at fairly low R ($<10^3$). The transient pressure loss at startup and shutdown has been studied by Gluck (Reference 20) and Warren (Reference 22). References 18 through 22 were used to determine the pressure losses and to size the capillary collector design of Figure 5-6.

The 325 x 2300 mesh screen was dictated by head considerations during main engine firing. It has a fairly high ΔP_E loss, given by Cady as

$$H = a\ V + b\ V^2 \tag{15}$$

where H is the hydrogen pressure loss at 1 g and saturated pressure conditions. V is the velocity through the screen. However, it compares favorably with 200 x 1400 mesh screen which has a slightly lower entry pressure loss. Table 5-19 lists the coefficients from which the difference may be seen.

From Cady, the bubble pressure at 1g and saturated hydrogen pressure, and the flow loss data, were determined as shown in Table 5-19. Furthermore, the

Table 5-19. Screen Pressure Comparison

Item	Dutch Twill Screen Mesh	
	325 x 2300	200 x 1400
Entry Pressure Loss Coefficients:		
a, 1/sec	1.14	.885
b, sec^2/m (sec^2/ft)	7.45 ([illegible].27)	6.59 (2.01)
Saturated pressure, N/cm^2 (psia)	34 (50)	34 (50)
Bubble Pressure, m (ft)	0.48 (1.580)	0.34 (1.108)

frictional loss term, ΔP_F, is approximately 50 percent less for the 325 x 2300 mesh screen than for the 200 x 1400 mesh screen, due to the lesser roughness of the finer wire (Reference 19).

The acquisition system is somewhat oversized to assure that the pressure losses are small during all four-engine APS burns. At APS depletion at the end of the mission, the flow losses finally exceed the bubble pressure. Eventually, as the propellant quantity and concomitant wetted collector area decrease in the upper compartment, the entry flow loss term, ΔP_E, dominates the loss terms. This is due to the higher entry velocity needed to satisfy the four-engine mass flow rate requirement. Liquid contained within the collector system as gas enters through the screen is taken as unavailable. The LH_2 residuals in the upper compartment are 0.16 kg (0.35 lb) trapped within the collector system and 0.08 kg (0.17 lb) trapped within the compartment for a total of 0.24 kg (0.52 lb) or 5.2 percent of the reservoir volume.

Examination of Figure 5-6 shows that propellant fed to the lower compartment can enter through either of the two toroidal collectors or through the 800 x 165 mesh compartment barrier directly. The toroids, made of perforated tubing covered with 325 x 2300 mesh screen, are connected to each other and to the lower compartment by unscreened tubing. A toroidal configuration was chosen over a curved vertical tube to facilitate overboard venting, since it is further removed from the vent tube and thus is less affected by the venting dynamics.

The lower compartment contains a single screened toroidal collector connected by tubing to a screened sump. Together they provide good communication between propellant and outlet. The lower compartment residual is 0.08 kg (0.18 lb) trapped in the collectors and 0.04 kg (0.08 lb) trapped in the lower compartment for a lower compartment total of 0.12 kg (0.26 lb) or 2.6 percent of the hydrogen reservoir volume. Total residual for upper and lower compartments is 0.35 kg (0.78 lb) or 7.8 percent of the reservoir volume. This residual quantity is largely due to the conservative sizing of the collector tube system. It is anticipated that a weight-optimized design would have a residual between 4 to 5 percent with somewhat lower collector system weight.

Similar considerations were used in the design of the collector system for oxygen. Here, the greater density of oxygen warrants the use of additional collector system weight to decrease residual volume. As volumetric flow rate is much less for LOX, the collector system tube diameters for LOX are about half those for hydrogen

Liquid Retention During Propulsive Maneuvers

Choice of the fine screen (325 x 2300 mesh) was dictated by the need to maintain propellant in the collector during the high accelerations of the MPS. This situation is a function of propellant quantity in the reservoir and vehicle acceleration. As each MPS burn is proceeded by an APS settling maneuver, the propellant in the reservoir is itself settled before MPS thrust levels are achieved. This is illustrated in Figure 5-8. During MPS burns, the screen must have the bubble pressure capability to withstand the static head, (Dimension X in the figure). The value of g during the mission is given in Figure 5-7, while the value of X can be calculated from the propellant volume data of the same figure. These data were combined with screen bubble pressure data to develop the two curves of Figure 5-9. The upper one is the acceleration capability (the acceleration level the screen can withstand without destabilizing) of the collector screen as a function of propellant remaining in the reservoir. The lower curve is the loci of worst-case events for the reference mission. The majority of the MPS firings are at acceleration levels below that of the "envelope". The worst of the worst cases is the main engine ΔV burn at 155.68 hours. Even for this event, the screen has a safety factor of two.

Orbital Refill

One of the most critical aspects of acquisition system operation is refill of the reservoir in orbit. During this operation, vapor in the reservoir is displaced by liquid propellant and is vented to space. This operation must be preceeded by settling of main tank propellants. Two basic cases exist: filling during MPS burns and filling during APS translations. To be compatible with the former, the feedline from the main tank to the reservoir tank must be located away from the MPS feed line to avoid pressure loss in the APS due to MPS dynamic flow effects. For the longer main engine burns, refill can await buildup to full thrust. For the shorter MPS orbit maneuver burns, refill must be initiated during the engine start sequence. For example, for the two midcourse correction burns a large portion of the burn is tank head idle mode operation. Each restart requires cooldown with about an 89-sec THIM thrust level. During this period, thrust builds up from about 34 to 71 kg (75 to 157 lb). Even for the worst case, settling is accomplished with THIM in 58 sec, thus allowing 41 sec for reservoir refill. A more critical refill case, however, occurs for the shorter APS translation burns. The worst of these is the 64-sec APS translation following Mission Orbit Insertion at 147.55 hours. As main tank LH_2 settling requires 46 sec for this case, refill time is 64-46 = 18 sec. Figure 5-7 shows that approximately 65 percent of the hydrogen zero-g reservoir volume must be vented for this case.

Another concern during refill is the pressure differential and line size between main tank and reservoir. The line size pressure drop relationship is determined by the refill mass flow requirements. To supply four APS engines, a hydrogen flow rate of 0.0283 kg/sec (0.0625 lb/sec) is required. To this must be added the refill rate, which for the worst case is 0.0755 kg/sec (0.167 lb/sec) for an unfilled reservoir volume of 0.02 m^3 (0.70 ft^3). The total rate is therefore 0.104 kg/sec (0.230 lb/sec). Flow loss analyses show that the 2.24-m (88-in.) long line between the main LH_2 tank and reservoir will experience a 0.3 N/cm^2 (0.4 psia) pressure loss if the line diameter is 1.6 cm (5/8 in.). This pressure loss corresponds to about a 0.08 K (0.14 R) drop in saturation temperature.

To refill the LOX reservoir during the same 64-sec APS translation 0.0026 m^3 (0.092 ft^3) must be displaced in 18 sec. Four-engine LOX flow rate is 0.085 kg/sec (0.188 lb/sec). Total rate during refill is 0.250 kg/sec (0.552 lb/sec). Proceeding as with LH_2 for the longer line between the main LOX tank and the reservoir, a 0.14 N/cm^2 (0.2 psia) pressure drop is experienced if the line diameter is again 1.6 cm (5/8 in.). This pressure loss corresponds to about a 0.1 K (0.2 R) drop in saturation temperature.

Zero-G Reservoir Location

The different factors that must be considered in the decision to locate the zero-g reservoir internally or externally to the main propulsion tanks have been identified and are listed in Table 5-20. In order to provide subcooled propellant within the reservoir to meet pump NPSP requirements and to preclude boiling and vapor disruption of the capillary devices, insulation is required for either an internal or external installation. An internal installation would require the development of a flight-qualified LOX-compatible

Table 5-20. Zero-G Reservoir Location

CONSIDERATION / LOCATION	INTERNAL	EXTERNAL
PROPELLANT SUBCOOLING REQUIRED~LOX/LH_2, K (R)	1.2/0.3 (2.2/0.5)	1.2/0.3 (2.2/0.5)
INSULATION WEIGHT, KG (LB)		
MLI	-	0.5 (1.0)
FOAMSIL	1.2 (2.6)	-
REINFORCED POLYURETHANE FOAM	0.6 (1.4)	-
ADDED MPS TANK WEIGHT, KG (LB)	0.6 (1.4)	0
ADDED LH_2/LOX HEAT EXCHANGER, VALVES AND LINES, KG (LB)	1.4 (3.0)	-
LH_2 BLEED COOLANT, KG (LB)	1.1 (2.4)	1.1 (2.4)
PRELAUNCH & BOOST PROPELLANT BOILOFF, LOX/LH_2, KG (LB)	0	0.3/0.8 (0.7/1.8)
BOILOFF PERCENT OF RESERVOIR CAPACITY, LOX/LH_2 (%)	0	3.5/18
TOTAL WEIGHT PENALTY, KG (LB)	4.9 (10.8)	2.7 (5.9)
ACCESSIBILITY FOR INSTALLATION, INSPECTION, AND MAINTENANCE	POORER	BEST
INSULATION DEVELOPMENT RISK AND COST	HIGHER	COMMON TO OTHER TUG INSUL

insulation such as Foamsil currently used in the laboratory. The insulation weight penalties listed in the table are based on providing equal bleed coolant flow for either the internal or external application. To avoid hydrogen coolant within the LOX tank, an O_2/H_2 heat exchanger and associated valves and lines would be required for internal location. Internal location would eliminate ground hold and boost boiloff losses; however, after such boiloff there would still be adequate propellant remaining in the reservoir to meet mission requirements prior to the first on-Orbit reservoir refill (refer to Figure 5-7). Finally, external location would provide the best accessibility for initial installation, inspection, and field maintenance of the zero-g reservoir.

A comparison of these factors shows that no major driver emerges to dictate a decision on locations. It does appear, however, that an external installation would be slightly more favorable in terms of weight, cost, and maintainability and thus it has been chosen as baseline for this study.

MPS Self-Settling Start Basket Evaluation

The General Dynamics Tug design (Reference 23) utilized an MPS start basket to eliminate the need for propellant settling at each MPS engine restart. General Dynamics has done appreciable work, including computer program development, on the hydrodynamics of start basket depletion and refilling (Reference 24). Space Division has used the analytic procedures described previously to evaluate the start basket design approach. As pointed out, at least one of the Mission A orbit maneuver burns (the Midcourse Correction at 8.02 hours) appears to be marginal with regard to refill. As the information on start baskets in Reference 23 was somewhat sparse, screen material and refill rates not being specified, it is difficult to evaluate the adequacy of the proposed design. In addition, thermal control provisions are not detailed. It is probable that, in addition to attached thermodynamic vent tubes, insulation may be required. Such insulation may decrease the open flow area for refill. In conclusion, there are unresolved questions about start basket design and performance. Further conceptual study is warranted, leading ultimately to experimental in-flight verification, in order to validate this start method.

MPS self-settling also has been considered through the use of the APS zero-g reservoir as a start basket for the main Tug engine. Since a minimum of 67 seconds of THIM operation is required for main engine thermal conditioning as compared to 55 seconds maximum for main tank propellant settling, liquid settling could be accomplished during this period without any additional propellant consumption penalty. The resultant savings in APS propellant is equivalent to approximately 95 kg (210 lb) of payload. The APS reservoir dry weight would have to be increased by approximately 13 kg (28 lb) total to provide the additional propellant capacity required for the main engine, resulting in a net payload gain of 59 kg (130 lb).

Use of the APS reservoir as an MPS start basket also would require redesign of the internal capillary devices to allow for the higher flow rate demand of the main engine. An additional isolation valve in each of the LOX and LH_2 main engine feed lines would be required to preclude vapor ingestion during the settling period. Additional SR&T and development tests would be necessary to validate the dual mode operation and performance of each reservoir. Because of these additional complexity and cost factors, this concept was not considered further.

PUMP SYSTEM SELECTION

Since the thruster inlet supply pressure requirement is 152 $\pm$ 14 N/cm^2 (220 $\pm$ 20 psia) and the maximum tank pressure is 13.4 N/cm^2 (19.5 psia), pumps are required to increase the fluid pressure. Also, accumulators downstream are used to limit the pump duty cycle as well as provide for APS functional capability in the event of pump failure during Tug recovery by the Shuttle.

Pumps

The purpose of this section is to present the system requirements and constraints which any pump configuration will have to meet, and to select the best type for each propellant system.

The APS pump requirements are presented in Table 5-14, as well as the inlet and discharge conditions any pump type will have to accept and satisfy. Figure 5-27 presents the power requirements needed to operate the pumps as a function of pump overall efficiency. Additional important items to be determined are the pump type, pump weight, and motor weight.

Candidate pumps are divided into two basic categories: dynamic pumps and positive displacement pumps. As seen in Figure 5-28 (from Reference 25), the specific speed (N_s) is higher for a dynamic pump (e.g., centrifugal or axial) than a positive displacement pump (e.g., piston, vane, drag, or roots) for the same specific diameter (D_s).

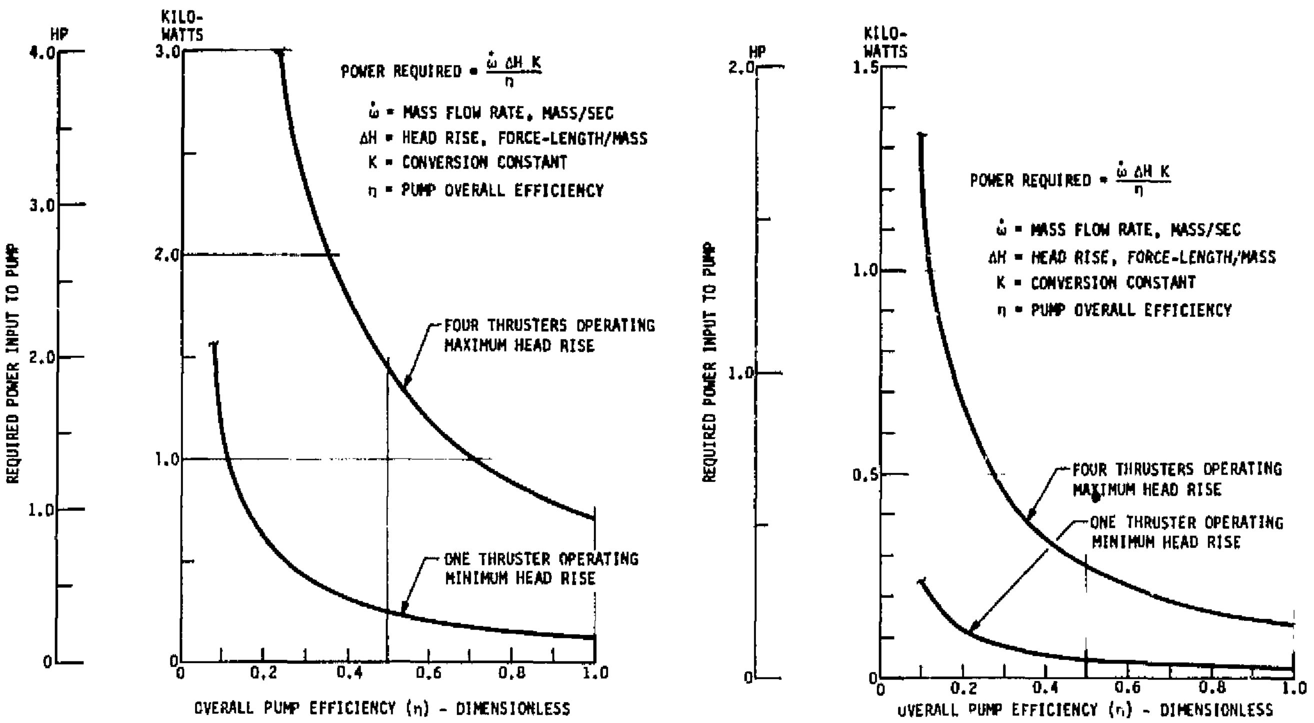
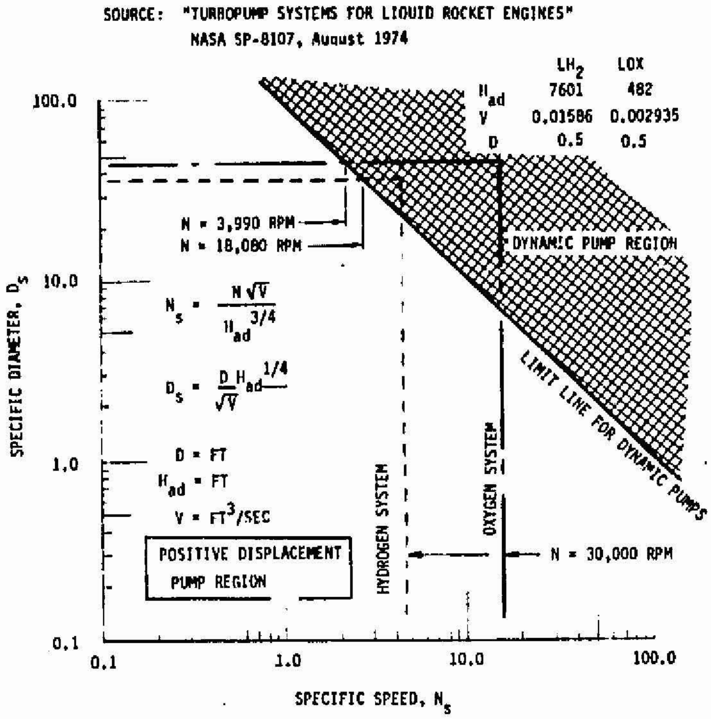


Figure 5-27. Pump Input Power Required as a Function of Efficiency



$$N_s = \frac{N\sqrt{V}}{H_{ad}^{3/4}} \qquad D_s = \frac{D\,H_{ad}^{1/4}}{\sqrt{V}}$$

Figure 5-28. Pump Specific Diameter and Speed

For the initial evaluation of candidate applicability, it is assumed that the maximum desired rotational speed is 30,000 rpm for any pump type. Further, the pump inlet diameter is limited to 15.2 cm (6 in.).

Using these limits and the design head rise and volumetric rates, pump operational envelopes can be superimposed onto Figure 5-28, which encompasses all design operations for either the hydrogen or oxygen system. As seen, the only appropriate choice for the hydrogen system is a positive displacement pump type. For the oxidizer system, the choice is not as forceful: a dynamic pump type could possibly be used for the oxidizer system; however, overall dynamic pump efficiency would tend to be low (50 percent or less). The required pump input power, from Figure 5-27, increases as pump efficiency decreases. For the oxygen system, a positive displacement pump is the first choice with a dynamic pump design as a possible alternate.

Of the common positive displacement pump types, the current baseline choice is a piston pump because of the availability of reliability data on field usage of such units. It also has high and constant hydraulic efficiency over a range of pump specific speeds. Hydraulic efficiency is the ratio of delivered head plus hydraulic losses where the hydraulic losses consist of the friction of the liquid flowing through the pump cylinder and the pressure loss through the pump valves. The piston pump is estimated to have an efficiency of 65 percent.

To reduce pump weight, a vane pump, with which there is virtually no cryogenic experience in this flow range, is an alternative. To maximize the mechanical efficiency of a vane pump, its specific speed from Reference 25, should be greater than 2 (see Figure 5-29). Even with this optimization, overall efficiences of 58 percent for both hydrogen and oxygen vane pumps are anticipated (Source: Sunstrand Co.).

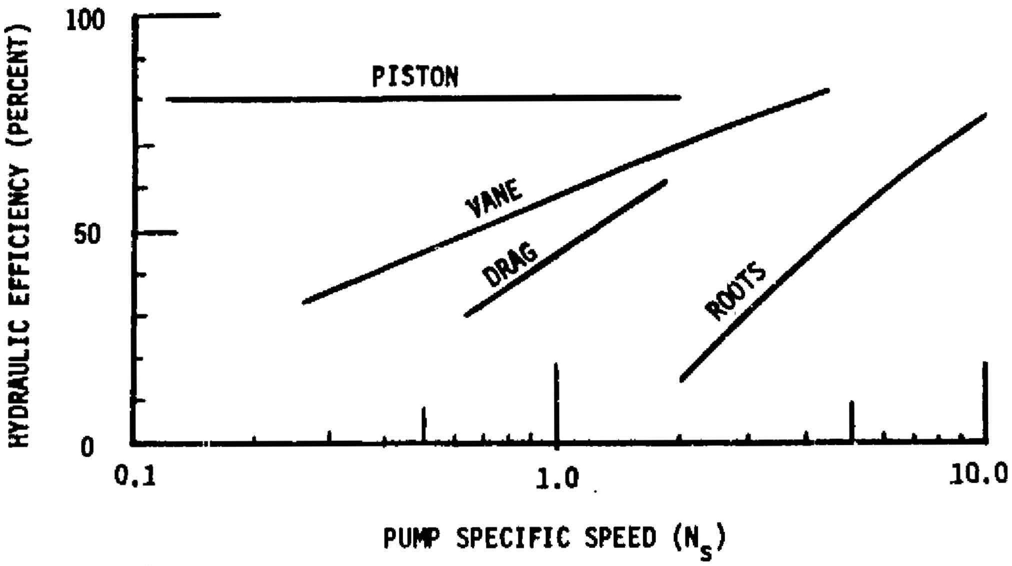


Figure 5-29. Positive Displacement Pump Efficiency

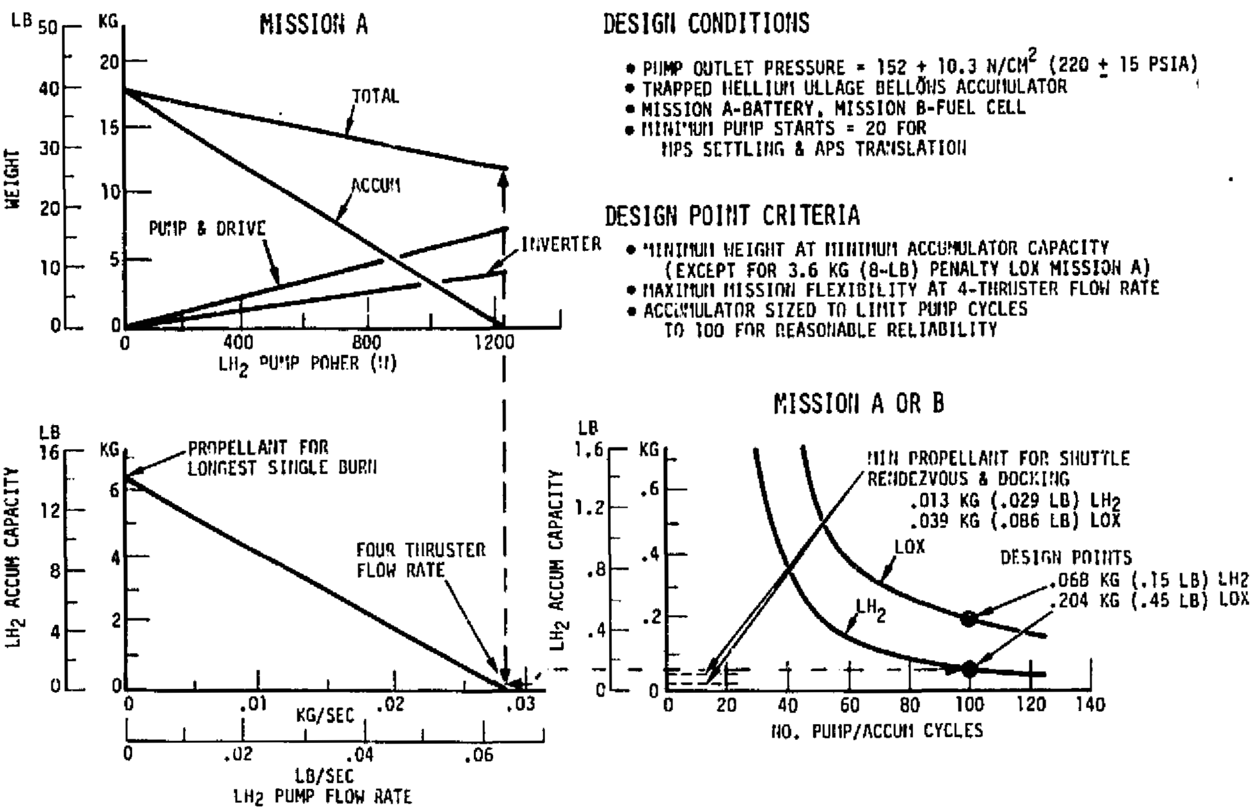


Figure 5-30. Pump/Accumulator Design Point Selection

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In the preliminary design phase of this study, the characteristics of the two pump types will be investigated in greater detail.

Accumulator Sizing

The rationale for sizing the integrated system accumulators is similar to that for the dedicated concept D-6 presented in Section 4.4. The sizing method is illustrated in Figure 5-30. Basically, the approach is to trade off accumulator weight versus pump and inverter weight. A trapped helium ullage bellows type of accumulator is assumed, providing a thruster inlet pressure ranging from 141 to 162 N/cm^2 (205 to 235 psia). A narrower control band than that assumed for the dedicated APS was used to provide additional margin in meeting thruster inlet requirements. For all cases, minimum weight occurs at the minimum accumulator volume with the exception of the LOX system for Mission A. Minimum accumulator volume occurs at a pump flow rate equivalent to four-thruster sustained flow. This pump size provides maximum system flexibility by allowing extended linear translation APS thrusting.

The design requirement dictating accumulator volume is the limit in pump start and stop cycles (assumed to be 100 for reasonable reliability and pump life). Application of this criterion results in accumulator liquid capacities of 0.068 kg (0.15 lb) for LH_2 and 0.204 kg (0.45 lb) for LOX. Incorporation of variable pump speed flow control during steady-state thruster operation eliminates accumulator cycling during all but pulse mode operations and thus allows a significant reduction in accumulator capacity from that developed previously for the dedicated systems.

In addition to limiting pump cycles, the accumulator also provides backup propellant feed capability in the event of a pump malfunction during Shuttle rendezvous and docking. The propellant requirements for this function are 0.013 kg (0.0288 lb) LH_2 and 0.0392 kg (0.0865 lb) LOX, which are well below the accumulator capacity requirements set by the cycle limit of 100 pump start and stops.

THERMODYNAMIC CONTROL

The baseline thermodynamic control system and its performance requirements were presented in Section 5.2. This system consists of insulation and low conductivity supports to minimize tankage and feed system heat loads, and an active hydrogen bleed system to dissipate these heat loads and control LOX and LH_2 temperatures. The elements of this system and their corresponding weights are listed on Table 5-21 along with the LH_2 bleed weight. These weights result in a payload loss of approximately 54.9 kg (121 lb), a penalty unique to a cryogenic type of APS. Because of this penalty, passive cooling of the APS has been evaluated.

The passive cooling approach considered utilizes the normal flow or propellant to the thrusters to absorb the system heat load. Review of the reference mission timeline reveals that the worst-case condition occurs during coast period No. 5 for which attitude control propellant consumption results in average thruster flow rates of 0.016 kg/hr (0.035 lb/hr) LOX and 0.005 kg/hr (0.011 lb/hr) LH_2.

Table 5-21. Thermodynamic Control System Weight
(Concept I-5)

Component	ID No.	Weight kg (lb)
LH_2 bleed return solenoid	15	0.68 (1.5)
LH_2 bleed shutoff valve	38	0.68 (1.5)
LH_2 bleed expander	16	0.68 (1.5)
LH_2 bleed heater	49	0.23 (0.5)
Tank external cooling coils	50	1.54 (3.4)
LOX tank insulation	17	0.23 (.5)
LH_2 tank insulation	19	0.23 (.5)
Bleed lines		7.71 (17.0)
Feed line insulation		3.72 (8.2)
Component mounting (5%)		0.77 (1.7)
Subtotal		16.47 (36.3)
Growth contingency (13%)		2.13 (4.8)
Total thermodynamic control system		18.60 (41.1)
Dry weight payload penalty (2.68:1)		49.94 (110.1)
Bleed propellant		5.67 (12.5)
Consumable payload penalty (.893:1)		5.08 (11.2)
Total payload penalty		55.02 (121.3)

Assuming the average flow rates are continuous, Figure 5-31 presents the resultant hydrogen and oxygen thruster inlet temperatures as a function of thermal isolation system effective emittance. Annotated on these curves are the nominal design temperatures and maximum design temperatures for LOX and LH_2 as well as the maximum LH_2 inlet temperature corresponding to the

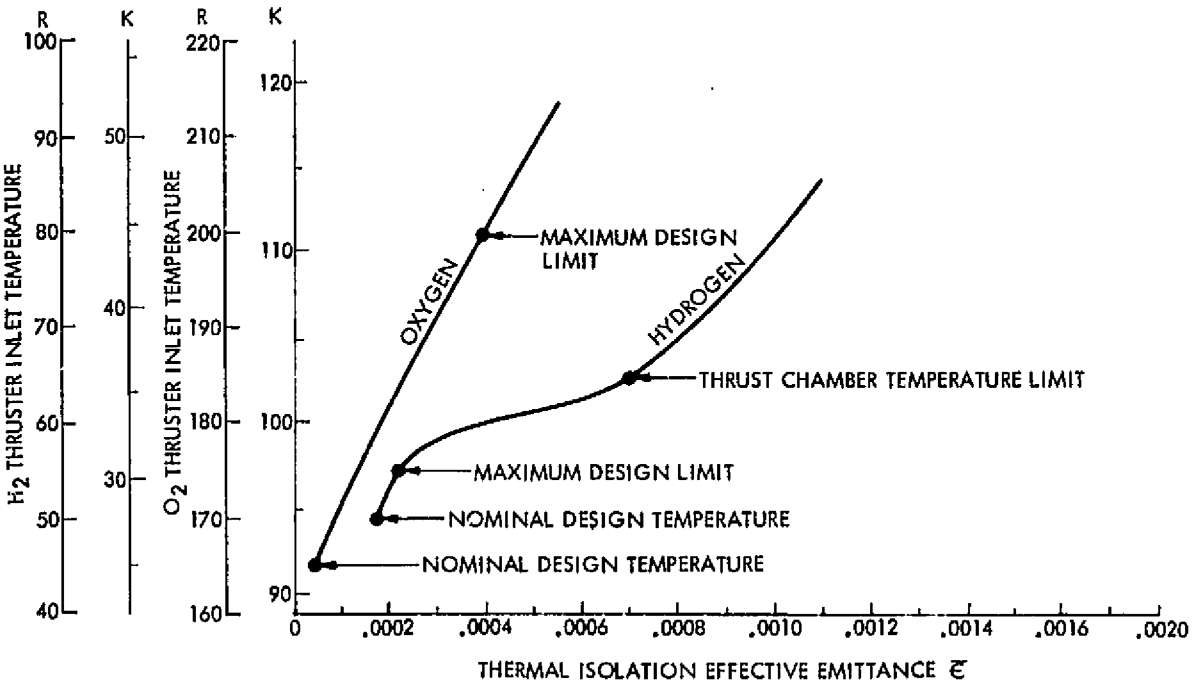

Figure 5-31. Passive APS Cooling

thrust chamber material upper temperature limit. As can be seen, satisfaction of these temperature limits would require thermal isolation system effective emittance values of 0.00004 to 0.00072 as compared to a value of 0.005 achieved to date for flight vehicles and 0.002 achieved for laboratory calorimeter tanks. Sufficient advancement in insulation technology to allow passive cooling is not expected in time to support the initiation of Tug vehicle development in 1978. Furthermore, since this analysis is based on average flow rates, quiescent periods between APS attitude control burns as well as more frequent operation of some thruster quads will result in localized transient temperatures far above those shown on Figure 5-31. For these reasons, it has been concluded that an active thermodynamic control system such as the one described earlier will be required for a cryogenic APS.

ELECTRICAL POWER SYSTEM

One of the most significant impacts that the use of an integrated cryogenic APS will have on the design of the Tug is on the electrical power subsystem. As shown in Table 5-15, simultaneous operation of the LOX and LH_2 APS pumps imposes an additional demand on the Tug power system of 1.44 kw. The total electrical energy required for a given mission depends on the pump operation time accumulated to provide APS total impulse and thermodynamic control hydrogen bleed. Assuming APS settling is used prior to MPS burns, mission pump electrical energy is 0.86 kw-hr and 3.9 kw-hr for Missions A and B, respectively.

APS Pump Power Evaluation

Two possibilities have been evaluated for providing APS pump power: the addition of silver-zinc primary batteries, and an increase in the size and capability of the Tug fuel cell. The latter will be discussed first.

Figure 5-32 presents a Tug power profile representative of the highest power demand flight phases. This profile was developed by starting with the baseline power requirements specified in Reference 1 as shown by the dotted line, subtracting the heater power required for the hydrazine monopropellant APS, and then adding the 1.44-kw pump power to the Tug APS linear translation, MPS burn, and Shuttle rendezvous and docking phases as shown by the solid line. The original baseline fuel cell and proposed fuel cell capability profiles are shown by the two dashed lines. Maximum predicted time periods for each phase also are identified. The baseline Tug fuel cell capability is 1.75-kw steady state and 2.5 kw peak. Overloading of the advanced Tug fuel cell is possible for short periods, with the magnitude of the overload a function of its duration up to a maximum overload capability of approximately 140 percent. Thus it can be seen from Figure 5-32 that the baseline fuel cell could meet the 2.4-kw pump power demand during the APS linear translation phase for only 11 minutes as compared to a required maximum of 3.8 minutes for Mission A and 35 minutes for Mission B.

To satisfy the 35-minute 2.4-kw pump power requirement, the Tug fuel cell must be designed for approximately 2.0 kw steady state or 0.25 kw above the current baseline capability. Similar evaluation of the other power profile phases reveals that the fuel cell design is dictated by APS pump operation during the high power demand period of MPS burn. If the APS accumulators were filled just before MPS burn and the APS pumps locked out during MPS burn, then the fuel cell design would be dictated by the rendezvous and docking phase of operation.

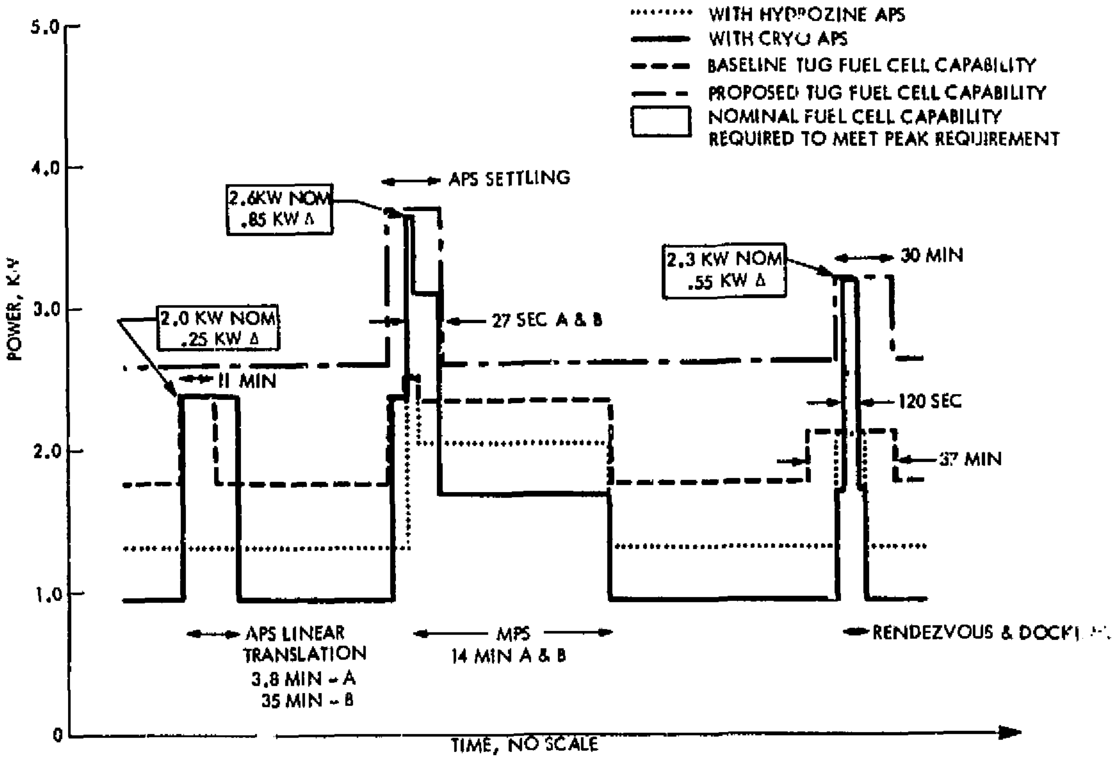

Figure 5-32. Worst-Case Tug Power Profile

Figure 5-33 presents Tug payload weight capability as a function of fuel cell steady state power required for APS pumping. Also shown is the associated fuel cell system weight penalty based on redundant fuel cell modules at 8.6 kg/kw (19 lb/kw) and additional radiator area at 5.9 kg/kw (13 lb/kw). As can be seen, approximately 18 kg (40 lb) of payload can be gained by APS pump lockout during MPS burns. Although not selected as baseline for this study because of reduced operational flexibility, this option should be re-evaluated after Tug operational and control functions are further defined.

Because of the substantial weight penalty for the fuel cell pump power supply, primary battery systems were considered. Figure 5-34 presents a weight comparison of the fuel cell system with a silver-zinc primary battery system as a function of total APS pump electrical energy demand. As can be seen, the weight of a fuel cell system is primarily power-dependent while the weight of a battery system is energy dependent. The crossover point occurs midway between the energy demands of Missions A and B. The lightest weight system for Mission A results from the use of a battery. However, the fuel cell system would provide the greatest mission flexibility for either Mission A or B. An unexpected increase in APS impulse would not be limited by the energy storage capability of the battery. The flexibility of the battery system could be improved, however, by designing the electrical power distribution system to allow switchover to the fuel cell during periods of low Tug power demand. For example, this switchover could be used to support mission abort operation of the APS in the event of a MPS failure.

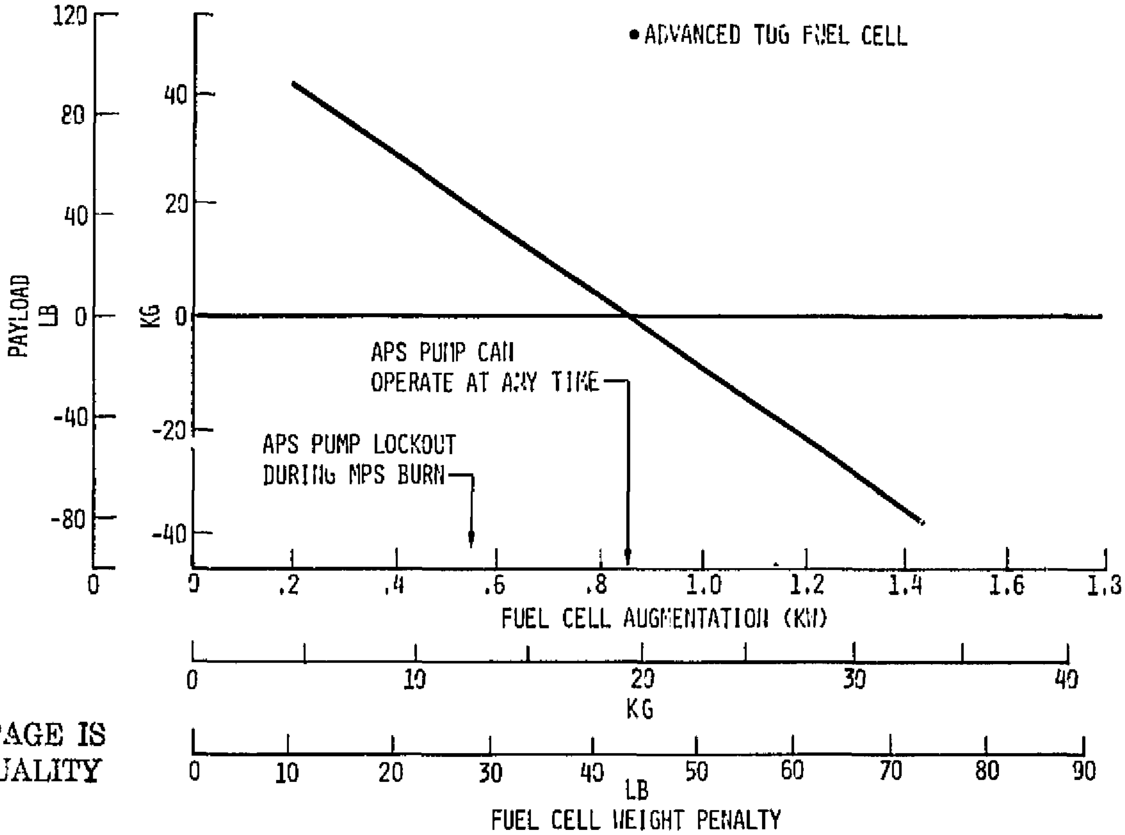

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Figure 5-33. APS Pump Power Supply - Fuel Cell Augmentation

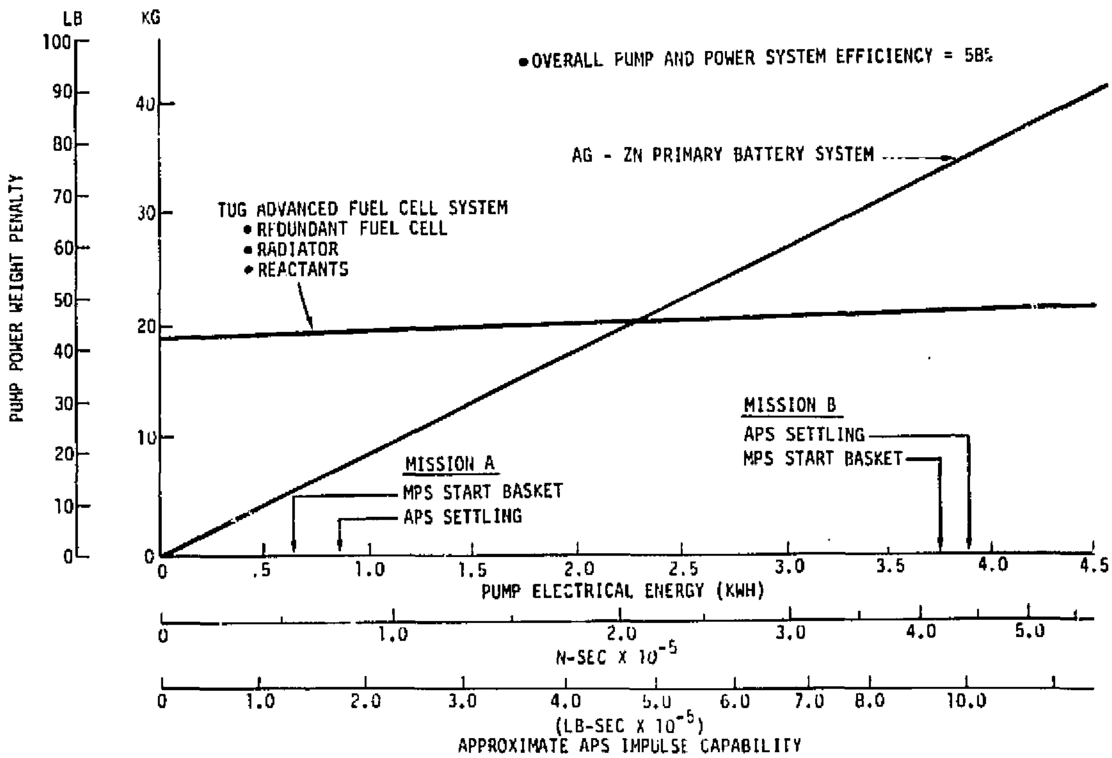

Figure 5-34. Pump Power Supply Options - Weight Comparison

Fuel Cell Reactant Storage

A comparative analysis has been made of three different approaches to the storage and supply of Tug fuel cell reactants:

- Dedicated system
- MPS integrated system
- APS integrated system

Figure 5-35 presents the estimated weight penalty for each approach as a function of Tug mission energy (kw-hr) and reactant storage capacity. Table 5-22 presents a detailed breakdown of the design considerations and their weight penalties as applicable to each storage approach. The weight penalty for the dedicated system is attributed to supercritical storage bottles of the type defined in Reference 2. For this system, the reactants are stored at a cryogenic supercritical state in dewar pressure vessels to avoid the problem of liquid-vapor interface control. As reactant fluid is withdrawn, pressure switches energize an electrical heater within the vessel to maintain constant pressure operation.

The MPS integrated concept is based on the system proposed in Reference 1. This system draws the fuel cell reactants directly from a zero-g capillary basket within the MPS tank through cleaners and filters used to provide the required reactant purity. Because the fuel cell must operate at a low main

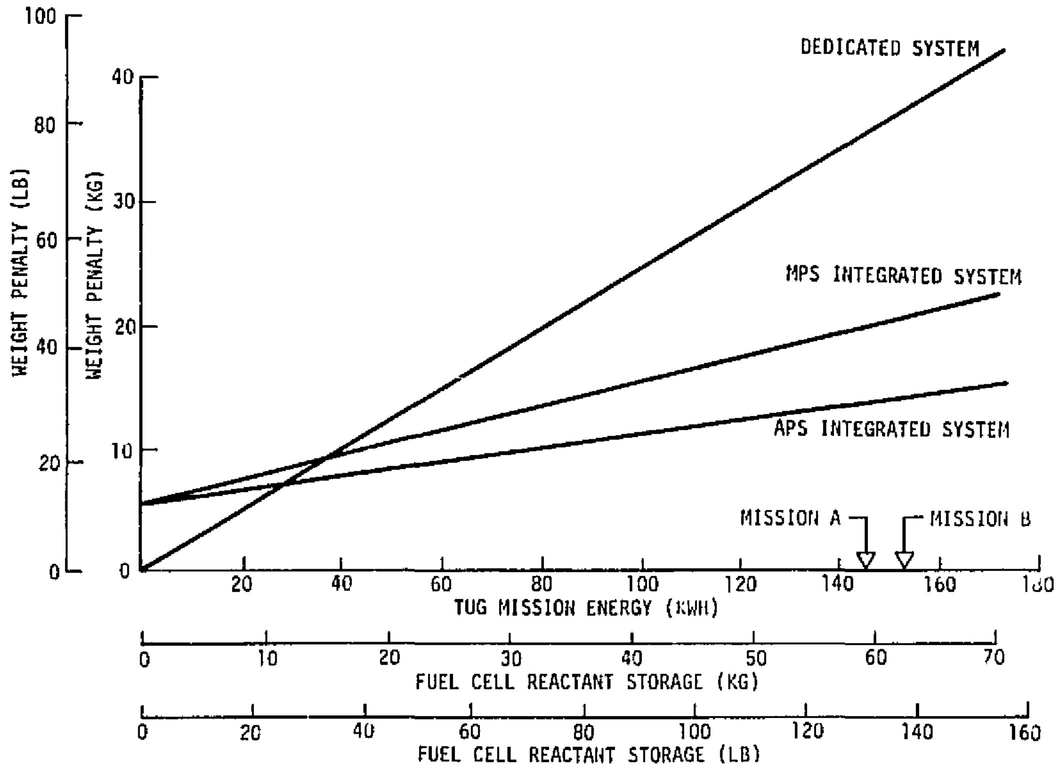

Figure 5-35. Integnated Versus Dedicated Fuel Cell Reactant Storage

tank pressure level of approximately 10.3 N/cm^2 (15 psia), as compared to 138 N/cm^2 (200 psia) for the two other concepts the fuel cell power density is reduced and results in an estimated weight penalty of 7.4 kg (16.4 lb) for a 2.6-kw system. For both integrated storage systems, liquid reactants are provided to avoid the temperature and flow oscillations that two-phase or slug flow could cause in the regenerative reactant supply heat exchangers. This liquid provides a useful heat sink, however, which can supplement fuel cell cooling and thus reduce the required radiator surface area. This advantage is equivalent to approximately 2.4 kg (5.4 lb) of radiator and is subtracted from the other weight penalties shown in Table 5-22.

Both integrated systems offer the advantage of mission flexibility; that is, additional electrical energy can readily be made available for special missions or in the event of on-orbit problems. Depending on the circumstances at the time, propellant refill of the capillary system by APS settling may be used to increase reactant supply.

The major technical problem in implementing an integrated fuel cell reactant storage concept is the design and development of the flight qualified

Table 5-22. Integrated Versus Dedicated Fuel Cell Reactant Storage

Weight Penalty Consideration	Weight Penalty, kg (lb)					
			Integrated			
	Dedicated		MPS		APS	
	O_2	H_2	O_2	H_2	O_2	H_2
Supercritical bottles	16.8(37.0	18.4(40.5)				
Additional main tank volume			0.7(1.5)	0.8(1.7)	0.7(1.5)	0.8(1.7)
Propellant cleaners and filters			2.7(6.0)	2.7(6.0)	2.7(6.0)	2.7(6.0)
MPS tank capillary basket			3.6(7.9)	3.9(8.6)		
Large APS zero-g reservoir					4.3(9.5)	4.6(10.1)
Reactant for reactant pump					0.0244(0.0538)	
Low pressure fuel cell penalty			7.4(16.4)			
Liquid reactant heat sink (radiator savings)			-2.4(-5.4)		-2.4(-5.4)	
Total	35.2(77.5)		19.4(42.7)		13.4(29.5)	

- Fuel cell nominal power rating = 2.6 kw
- Tug Mission A energy = 145 kw-hr
- Reactant storage requirement = 60 kg (132 lb)

cleaners and filters required when using propulsion grade hydrogen and oxygen. CO_2 and other harmful contaminants must be removed to provide the reactant purity necessary for sustained fuel cell operation. The investigation of these cleaners and filters should be the subject of a near-term SR&T effort.

As shown on Figure 5-35, the APS integrated system provides minimum weight for the missions under consideration. It is expected that the APS integrated concept also would result in DDT&E costs below those for the MPS integrated system by avoiding the development of a separate capillary zero-g propellant retention device. However, in order to simplify the subsequent preliminary design task and allow concentration of effort on APS problem areas, the MPS integrated concept has been selected as baseline for the remainder of this study.

RELIABILITY

The reliability of the I-5 integrated APS design was assessed following the same procedure used for the dedicated cryogenic and storable propellant designs. In general, the basic reliability of the integrated APS is higher than that of the dedicated cryogenic concepts, but lower than the storable propellant designs. However, with development of specific components to a level comparable to that of the storable propellant designs, the I-5 design will meet the reliability goal. In particular, it is believed that the generic failure rate of the accumulator may easily be reduced by half to 1.56 per million hours if life cycle testing is undertaken. The reliability of the design is itemized in Table 5-13 and shows that the goal of 0.996 can be attained by halving the accumulator generic failure rate.

Details of the reliability analysis of the I-5 design are described in the success path logic shown in Figure 5-36. In a few instances the operating times and cycles were revised from those used in the dedicated APS analysis. The complete list of times, together with valve failure rate assumptions is as follows:

Valve failure rate assumptions:

Leakage	80%
Failure to return (unenergized)	18%
Fail to operate (energized)	2%

Operating times and cycles:

t_1 - 164 hr	Mission time
t_2 - 2300 cycles	Single thruster mission operation
t_3 - 82 hours	Assumed failure point
t_4 - 1 cycle	Isolation after failure
t_5 - 50 cycles	LH_2 bleed shutoff valve operation
t_6 - 10 cycles	Vent
t_7 - 32.8 hr	LH_2 bleed heater operation

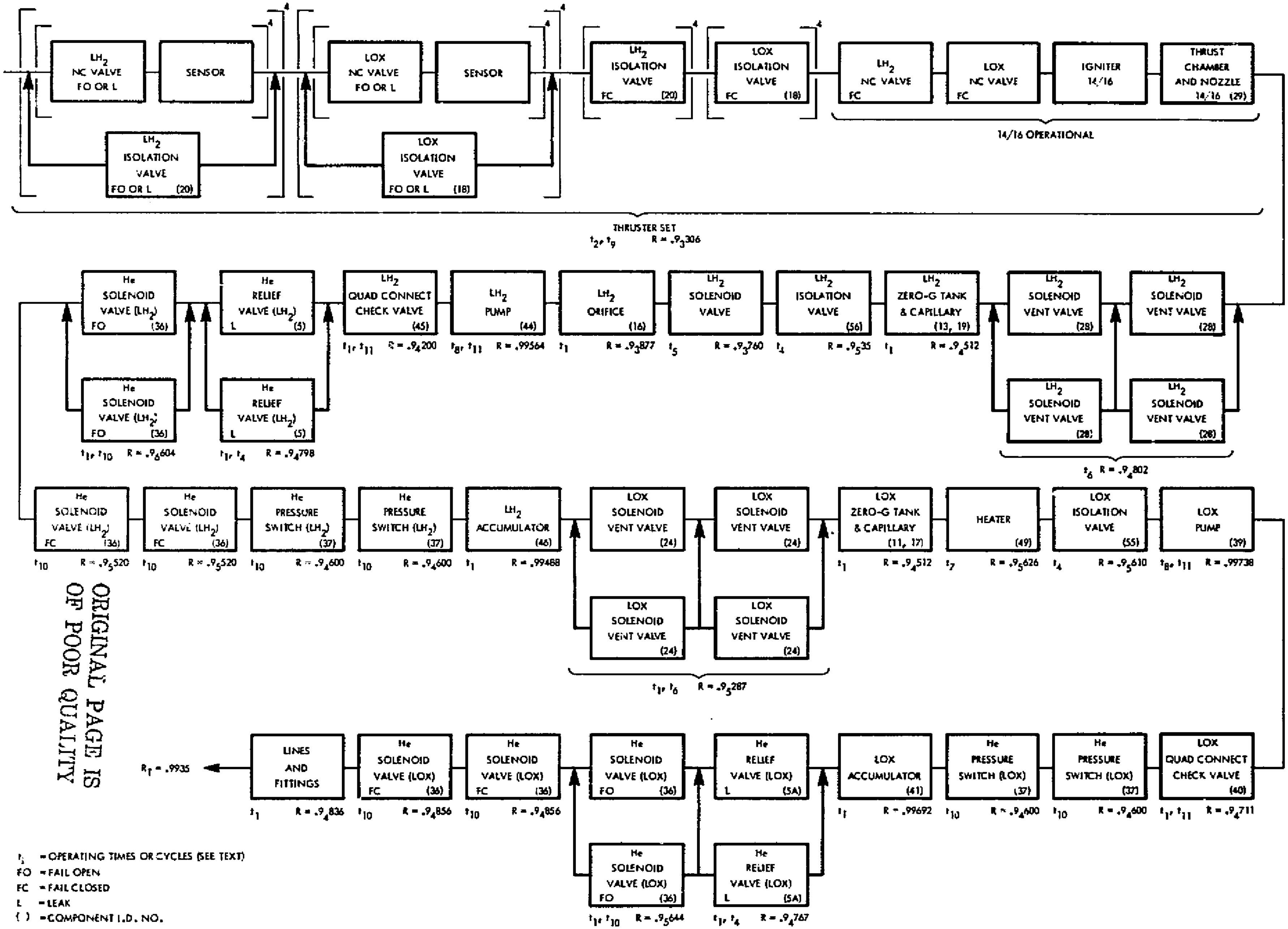

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Figure 5-36. Success Path Logic for Concept I-5

t_8	- 0.4/3.25 hr	Pump operating times, Mission A/B
t_9	- 0.25 hr	Single thruster operating time
t_{10}	- 5 cycles	Pressurization system operation
t_{11}	- 100 cycles	Pump operation

The failure rates used in the analysis are the same as those of the dedicated system analysis except for the following:

Accumulator	$3.13/10^6$ hr
He pressure switch	$8/10^6$ hr
Pump and motor	$3/10^6$ hr

All of the components which are charged to the APS as a stage system in the reliability sense are shown in the path logic. Other components which are utilized by the APS but are integral parts of other stage systems, such as the main tanks, rightfully are included in the reliability allocations of those systems.

Single propellant pumps in the I-5 design are permitted under the baseline requirements since they have inherently high reliability for their short operating periods. They do not constitute single points of failure leading to the loss of the Tug because the accumulators, being downstream of the pumps, provide sufficient propellant to afford rescue.

All of the reliability analyses conducted for this study are based on the application of predictive techniques to historical failure data. As system development progresses beyond this study, however, knowledge of the relative capability for success will grow firmer for all Tug systems and the APS reliability goal will become a requirement. At the same time, DDT&E test programs will yield improved reliability assessments and, even more important, actual reliability improvements through hardware maturity gains.

VEHICLE SYSTEMS IMPACT ANALYSIS

Main Engine Backup Capability Analysis

Backup for a Tug main engine failure can be provided by the integrated APS in the +X velocity mode. In this mode, the four aft firing thrusters provide 445 N (100 lb) of thrust and can be supplied with the total quantity of MPS propellant remaining. Since the loaded MPS mixture ratio is either 5.6 (for offloaded missions) or 6.0 (for the full tank retrieval missions), the APS mixture ratio for backup operations must be correspondingly increased from its normal value of 3.0 to use all of the oxygen as well as the hydrogen.

Thruster Capability

Thruster analyses indicate that the thrust chamber will operate satisfactorily at a mixture ratio of 5.6 with low density warm hydrogen at inlet. Assuming injector performance can be maintained with higher density (lower injector pressure drop), colder hydrogen at inlet, a specific impulse of 400 sec is obtained for the 200 area ratio thruster operating in steady state backup mode at the mixture ratio of 5.6. This assumption involves injector and thrust chamber design aspects which will require further study. Higher nominal (pulse mode) mixture ratio, a stiffer (higher pressure drop) fuel side injector or a valved, split-manifold fuel injector are some of the design possibilities for obtaining mixture ratio shift capability in the thruster. For the present conceptual level study, this capability is assumed. Although it would be beneficial if somewhat higher thrust resulted from the mixture ratio shift, abort performance analyses of this study assume constant thrust and 400 sec specific impulse at a mixture ratio of 5.6. Performance for retrieval missions accounts for the outage resulting from a 6.0 loaded ratio.

Propellant System Capability

During main engine backup operation by the APS, the propellant supply from the main tanks is unpressurized. This results in a blowdown mode but, unlike main engine blowdown conditions, the withdrawal rate is low and heat leak to the tank is sufficient to maintain absolute pressure above one atmosphere. Suppression head for pump suction is also maintained at the design condition by the active cooling of the reservoirs. The hydrogen bleed rate during this mode corresponds to a 24.5 N-sec/kg (2.5 sec) specific impulse loss.

The APS propellant system has the ability for mixture ratio shift by varying the speed of either pump up to about ± 40 percent through frequency variation in the power supply inverters. As shown in Table 5-23, at constant thrust and injector pressure loss coefficient, the oxidizer pump speed, head, and power demand increase after a mixture ratio shift while the converse is true of the fuel pump.

The total power demand drops to 59 percent of nominal, but fuel side injector pressure drop falls to an unacceptable 37 percent. If injector drop coefficient is still constant while chamber pressure and thrust are increased 25 percent, fuel injector drop reduces only to 57 percent and fuel cell power demand is constant.

The foregoing illustrates the range of feed system mixture ratio shift capability which can be balanced against thruster trades without significant power or propellant system weight penalties. The cryogenic integrated concept is therefore considered to have inherent capability to serve as a backup for a Tug main engine failure.

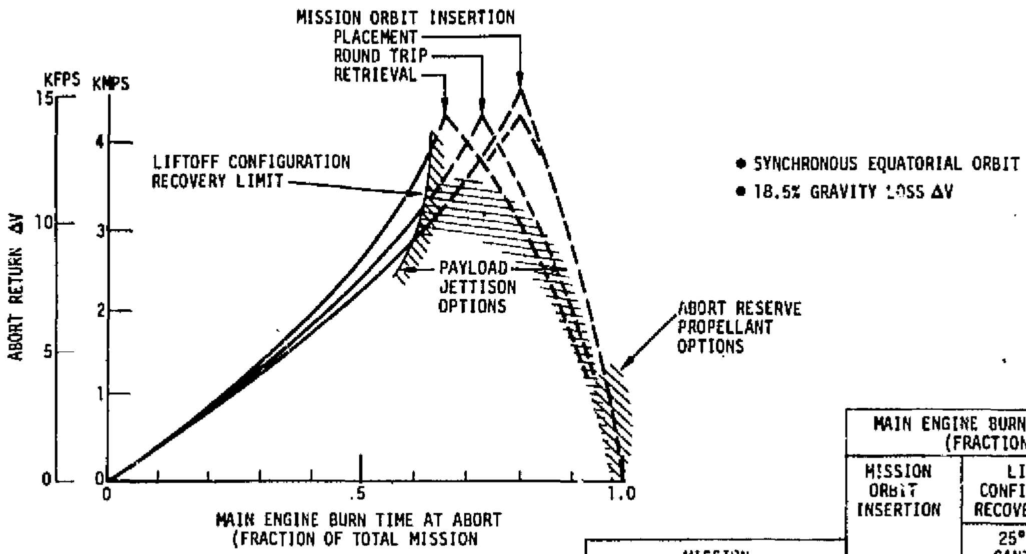
Table 5-23. Mixture Ratio Shift Values

Item	Parameter Values (% of Nominal)			
	Constant Thrust		Constant Power	
	Oxidizer	Fuel	Oxidizer	Fuel
Injector pressure drop coefficient	100	100	100	100
Thrust and chamber pressure	100	100	125	125
Pump flow and speed	113	61	141	76
Injector drop	128	37	200	57
Pump head	110	77	154	103
Pump power	124	47	216	78
Fuel cell power	59		100	

Backup Operations Analysis

With mixture ratio shift, the APS impulse capability is proportional to the specific impulse ratios of the two engine systems - 86 percent of the MPS impulse remaining. In Figure 5-37, the return velocity requirement is shown for synchronous equatorial missions as a function of main engine burn time. On the outbound leg, the return velocity is considered to be the Tug ideal velocity attained. On the inbound leg, it is the mission ideal velocity remaining.

The peak return velocity is therefore at mission orbit insertion (MOI) shown on the figure for the three types of missions: placement, round trip, and retrieval. The ability to return without jettisoning a payload is on the outbound leg up to the line labeled on the figure as the "liftoff configuration recovery limit." This point is, in general, during the mission orbit insertion burn. The values of main engine burn time at this limit and at MOI are tabulated on the figure for two APS aft firing thruster cant angles: 25 degrees and zero. As may be seen from the table, main engine backup capability for the liftoff configuration exists for very nearly 60 percent of the main engine duty cycle with the baseline 25-degree cant angle. For zero cant angle, the capability is not less than 60 percent. The method of computing the limit of outbound recovery accounts for both the lower specific impulse (400 sec) of the APS and the lower thrust, 445 N (100 lb). Lower thrust is accounted for by adding 18.5 percent to the ideal return velocity for gravity loss.



MISSION	MAIN ENGINE BURN TIME (FRACTION): MISSION ORBIT INSERTION	LIFTOFF CONFIGURATION RECOVERY LIMIT: 25° CANT	ZERO CANT
SINGLE PAYLOAD PLACEMENT	.80	.58	.60
TRIPLE PAYLOAD PLACEMENT	.77 TO .80	.59	.62
ROUND RIP	.72	.62	.64
RETRIEVAL	.65	.58	.62

Figure 5-37. Abort Capability - Main Engine Backup by APS

In a previous study (Reference 26), the effect of gravity losses on the total return ΔV using the APS for thrusting was investigated for a point partway through the MOI maneuver after an ideal velocity of 2896 m/sec (9500 ft/sec) had been obtained. The burn sequence involved a total of 43 APS apogee or perigee burns, each contributing a ΔV of about 61 m/sec (200 ft/sec). An APS engine cumulative burn time of 16.9 hours was required, with the longest individual burn lasting about 45 minutes. The total time to return to the 315 km (170 n mi) orbit was about 137 hours. Gravity losses for this flight profile amounted to 538 m/sec (1765 ft/sec). The previous study concluded that a curve representing required ΔV including gravity losses could be approximated by simply "pivoting" the ideal return ΔV curve counter-clockwise about the origin until it passes through the computed point. This amounts to adding 18.5 percent.

Return during the outbound leg involves mission abort. In addition, missions which do not carry a maximum payload can have the capability for mission completion by the APS in the event of a main engine failure near the end of its duty cycle. This could be accomplished by loading a small propellant reserve for such a contingency. For example, a 227-kg (500-lb) propellant margin will permit coverage of the last 916 m/sec (3000 ft/sec) of a placement mission. Other options considered in Reference 26 which extend the range of APS backup capability include (1) payload jettison which permits Tug recovery from a point about 80 seconds prior to completion of the MOI burn, and (2) a second Shuttle flight to recover the Tug at 926 km (500 n mi). This would reduce the return velocity requirement by over 305 m/sec (1000 ft/sec).

Use of APS for MPS Feedline Childown

This analysis considers the use of an APS propellant-supplied recirculation system for thermal pre-conditioning the MPS propellant feedline prior to main engine operation. The possibility investigated was that a weight saving or a preferred engine start transient could be obtained if the feedlines were pre-conditioned by the recirculation system rather than by initial THIM propellant flow. This analysis led to the conclusion that an APS propellant-supplied recirculation system is an inferior alternative to THIM cooldown.

The baseline vehicle propellant feedline dimensions from Reference 27 are presented in Table 5-24. The engine start transient analytical results established in Reference 27 are used directly for comparison with the results of the APS recirculation system analysis.

Table 5-24. MPS Propellant Feedline Data

Description	LH_2 System	LOX system
Material	Stainless steel	Stainless steel
Length, m (ft)	4.3 (14)	1.2 (4)
Inner diameter, cm (in.)	6.4 (2.5)	6.4 (2.5)
Wall thickness, cm (in.)	0.1 (0.02)	0.05 (0.02)
Weight, kg (lb)	1.7 (3.8)	0.50 (1.1)

A propellant feedline is considered to be thermally-conditioned if its temperature is within 2.22 K (4 R) of the propellant temperature. For a Tug with propellants at saturated conditions and 11 N/cm^2 (16 psia), propellant feedline thermal conditioning is complete when the LH_2 feedline temperature is no warmer than 23 K (41 R) and the LOX feedline temperature is 93.2 K (167.8 R).

Preliminary analysis showed that the best APS propellant-supplied recirculation system was one which extracted only liquid hydrogen from the APS. Such a system is schematically illustrated in Figure 5-38. Heat exchange between coolant line and feedline is obtained by thermal contact between the two. The recirculation system dry weight was estimated to be 8.6 kg (19 lb).

The recirculation system is activated by opening the two-way solenoid valve starting the LH_2 pump. Subsequent activation of the three-way solenoid valve permits coolant to bypass the MPS LOX propellant feedline once it has reached the desired temperature level. Because the pressure of the hydrogen coolant is greater than the LH_2 tank pressure, the warm hydrogen is dumped back into the LH_2 tank, thereby adding energy to the tank system and momentarily causing the hydrogen liquid in the tank to be subcooled.

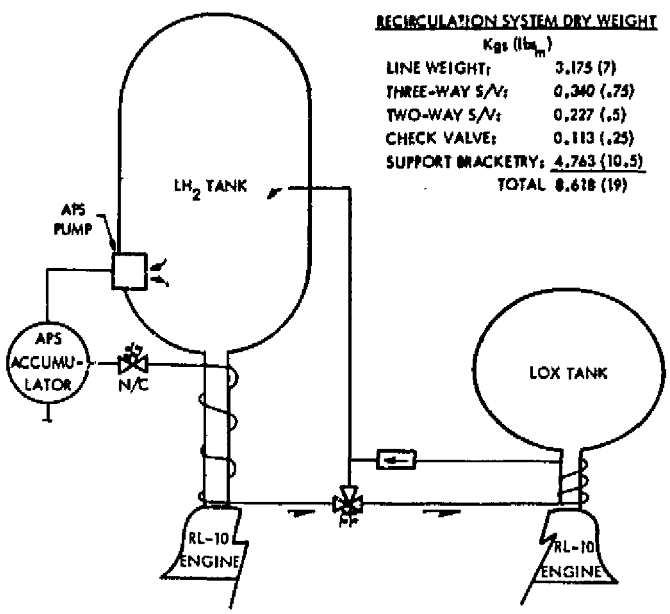


Figure 5-38. APS Propellant-Supplied Recirculation System Schematic

With some allowances for heat exchange efficiency, it was established that a maximum of 1.34 kg (2.96 lb) of hydrogen coolant initially at 155 N/cm^2. (225 psia) and 21.11 K (38 R) is required to thermally condition the MPS LH_2 and LOX propellant feedlines from 278 K (500 R) to the desired temperature levels. By assumption, the APS supplied the coolant at a mass rate of 0.0284 kg/sec (0.0627 lb/sec). Because of the different temperature levels and line masses, the oxygen feedline system is thermally conditioned within 37 sec while the fuel feedline requires 48 sec to be thermally-conditioned. A maximum of 278,000 Joule (263 Btu) of energy is extracted from the MPS propellant feedlines during each chilldown cycle.

In Reference 27 an analysis is presented with propellant feedlines assumed to be either preconditioned to the propellant saturation conditions of 11 N/cm^2 (16 psia) or to be unconditioned with an initial temperature of 278 K (500 R). The analysis showed that the use of colder feedlines increases the propellant consumption and the time required to condition the engine. This occurred because the colder lines allowed the oxidizer pump to cool down 12 sec earlier than with hot lines. Once the oxidizer pump was conditioned, oxidizer flow increased significantly, causing mixture ratio and chamber pressure to increase. The higher mixture ratio increased fuel heat transfer and the higher chamber pressure increased fuel system back pressure causing a reduction in fuel flow and a 4-sec longer time to condition the fuel pump. Since, in this case, the cooldown time was set by the fuel pump and the oxidizer pump was conditioned considerably earlier, total propellant consumption increased 2.9 kg (6.4 lb) with the colder suction lines. Aside from the weight disadvantage, to run the engine at a high mixture ratio involves a high engine damage risk due to the increased combustion temperature.

A comparison of the propellant masses used and the time required to chill the propellant feed lines and engine pumps with and without (from Reference 27) an APS recirculation system is presented in Table 5-25. The table also presents the logic used to conclude that an APS recirculation system is not mission-effective because its effective burned weight is more than double that of a system which permits initial propellant flow to condition the feedlines.

It is concluded than an APS propellant-supplied feedline recirculation system is an inferior alternative to permitting the feedlines to be thermally-conditioned by initial THIM propellant flow.

Table 5-25. Propellant Recirculation System Comparison

DESCRIPTION	NO RECIRCULATION OXIDIZER	NO RECIRCULATION FUEL	NO RECIRCULATION TOTAL	WITH RECIRCULATION OXIDIZER	WITH RECIRCULATION FUEL	WITH RECIRCULATION TOTAL	REMARKS
● PROPELLANT USAGE, Kgs (Lbs_m)							
• APS PROPELLANT REQUIRED FOR FEEDLINE DUCT CONDITIONING	N/A	N/A	N/A	0 (0)	1.34 (2.96)	1.34 (2.96)	MASS QUANTITIES AND TIMES PRESENTED ARE BASED ON THE ENGINE PUMPS AND FEEDLINES INITIALLY AT 277.77°K (500°R) AND THE TANK PRESSURES AT 11.03 N/Cm² abs (16 PSIA) AND ZERO NPSP. THIS IS THE WORST CASE FOR COOLDOWN TIME OR MASS USAGE FOR THE TUG.
• MPS PROPELLANT REQUIRED FOR ENGINE AND FEEDLINE DUCT CONDITIONING	7.17 (15.8)	2.22 (4.9)	9.39 (20.7)	9.98 (22)	2.31 (5.1)	12.29 (27.1)	
● TIME DURATION, SECONDS							
• APS FEEDLINE DUCT CONDITIONING	N/A	N/A	N/A	37	48	48	
• MPS ENGINE AND FEEDLINE DUCT CONDITIONING	59	87	87	47	91	91	
• COMBINED APS AND MPS CONDITIONING			87			139	
① APS AND MPS TOTAL PROPELLANT USAGE, Kgs (Lbs_m)			9.39 (20.7)			13.63 (30.06)	
② USEFUL TOTAL IMPULSE DURING THIM, Kg-Sec (Lbs_f-sec)			2459 (5421)			3545 (7816)	FOR PERFORMANCE, RELIABILITY AND COST, THE SYSTEM WITH NO APS RECIRCULATION IS THE BETTER CHOICE.
③ EQUIVALENT PROPELLANT AT 459.2 Lb_f-sec/Lb_m [② ÷ 459.2], Kgs (Lbs_m)			5.35 (11.80)			7.72 (17.02)	
④ EQUIVALENT LOST PROPELLANT PER START [① - ③], Kgs (Lbs_m)			4.04 (8.9)			5.91 (13.04)	
⑤ EQUIVALENT LOST PROPELLANT PER MISSION, Kgs (Lbs_m)							
• MISSION A - 12 STARTS			48.48 (106.8)			70.92(156.48)	
• MISSION B - 6 STARTS			24.24 (53.4)			35.46 (78.24)	
⑥ EQUIPMENT BURNED WEIGHT, Kgs (Lbs_m)			0 (0)			8.62 (19)	
⑦ LOST PROPELLANT EQUIVALENT BURNED WEIGHT, Kgs (Lbs_m)							
• MISSION A [⑤MISSION A ÷ 3]			16.15 (35.6)			23.66 (52.16)	
• MISSION B [⑤MISSION B ÷ 3]			8.07 (17.8)			11.83 (26.08)	
⑧ TOTAL EFFECTIVE BURNED WEIGHT, Kgs (Lbs_m)							
• MISSION A [⑥ + ⑦]			16.15 (35.6)			32.28 (71.16)	
• MISSION B [⑥ + ⑦]			8.07 (17.8)			20.45 (45.08)	

6. CRYOGENIC CONCEPT SELECTION

Competitive concepts were compared using a general methodology which was first developed to evaluate the seven dedicated APS designs. This same methodology, with additions to account for integration effects and refinements to include more depth of detail, was then used in evaluating the surviving integrated APS concepts against the selected dedicated concepts.

6.1 CONCEPT EVALUATION METHODOLOGY

The APS designs are evaluated by generating a matrix which ranks the designs in order to identify and eliminate the poorer candidates and to focus attention on the top two or three contenders. If the top contender excels in all criteria, the selection is obvious. If the criteria values for the top two or three contenders are virtually identical, then no significant difference exists among them. If the candidate preferences shift significantly among the criteria, then the selection is made on the basis of value judgments.

Cost-related criteria constitute the major basis for evaluation. Performance attributes constitute the second group of evaluation criteria, followed by schedule considerations.

No attempt was made to apply weighting factors because of (1) the highly subjective nature of such factors and (2) the incommensurability of the criteria.

COST

In compiling cost data, major attention is focused on costs at the APS component level, recognizing that in most systems, 80 percent of the cost comes from approximately 20 percent of the components (Pareto's rule). Vendor cost data were obtained and used (modified for additional costs) wherever feasible.

In generating both the nonrecurring and recurring cost estimates, incremental costing is used. Costs of peripheral items (such as APS ground support equipment) that are judged to be nearly constant across the alternatives being evaluated are omitted so as to place emphasis on the sources of cost differences or incremental costs. In this way, incremental life cycle costing (LCC) was developed and used as a primary criterion. To illustrate its significance, the LCC criterion can show that systems which incur relatively high DDT&E costs and even relative high production costs may be offset by low operational costs and vice versa.

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Because of the uncertainty of the magnitude of the buy, the APS fleet size was initially treated as a variable. In later comparisons involving the integrated APS, the fleet size of 17 from Reference 1 was used. All costs are determined in 1973 dollars.

Supporting Research and Technology Costs

Key technological deficiencies inhibiting the development of APS concepts were recognized during the study and the requirements for technology development programs were roughly defined in terms of cost and time. These SR&T program costs were not included as evaluation factors in the comparisons for two reasons: (1) when the final (Phase B) selection of the APS concept is made, the SR&T costs will have already been expended; and (2) subjects of SR&T funding, on basically new concepts such as the one considered here, usually have great potential for other programs.

DDT&E Costs

DDT&E cost estimates contain implicit probabilities of their attainment. The DDT&E costs shown are the "most likely" costs; these in themselves do not completely denote the intrinsic development risks associated with the DDT&E activities.

Production Costs

A first unit production cost was generated for each APS candidate. These are also based on modified vendor data wherever possible. A 90 percent learning curve was used to project the production costs for various quantities of systems produced.

Maintenance/Refurbishment Costs

The program model (taken from Reference 1) involves 243 flights with a fleet size of 17 and a maximum of 20 missions for each vehicle. This leads to an average of 12.15 vehicles refurbished for 20-mission capability. The spares required are determined by analysis of APS component life and replacement schedules. It is assumed that spares are produced during the production phase and are stored for future use. Accordingly, they are posted at the 17th unit level. Subsequent learning curve reductions are not taken in order to approximate compensation for storage and inventory costs. Refurbishment operations are taken at 50 percent of the cost of replacement units.

Potential Costs

Potential costs are those which [illegible] recommended changes or possible savings in the Tug vehicle or prog[illegible] strictly APS costs. They are identified and assigned a valu[illegible] establish "net" DDT&E or LCC costs.

PERFORMANCE

The performance criteria used in the evaluation consider both reference mission payload ("basic") performance and additional functional capability that would enhance operations. Two types of functional capability are considered. Added flexibility is considered to mean here the relative adaptability of an APS concept to provide main engine backup operational modes at any time. Vehicle versatility involves the capability to perform currently unplanned but nevertheless likely missions. More exactly, flexibility refers to the potential capability to provide added APS impulse instead of main engine impulse in the event of any mission contingency or opportunity. Similarly, versatility means an extremely large maneuvering capability may be obtained from the APS. Such a capability could be essential to complex orbital assembly operations and other potential missions.

SENSITIVITY ANALYSIS

The final step in the APS evaluation methodology is determination of the sensitivity of the key evaluation criteria values to the study assumptions and constraints.

6.2 DEDICATED SYSTEM EVALUATION

The seven cryogenic APS concepts which were defined in Section 4 were evaluated using most of the methodology described. Recognizing that the weight, cost, and reliability data of the dedicated concepts were preliminary, certain aspects of the methodology were relaxed to yield quick answers and to identify drivers which would permit the elimination of other criteria as nonessential.

EVALUATION ANALYSIS

Since cost (DDT&E plus production) is used as a primary evaluation criterion, the total cost at a buy of 17 reflects the relative ranking for this criterion. The least costly concept was found to be No. 3 (tank wall-cooled, pressure feed) while the most costly concept was No. 4 (modular tank wall-cooled, pressure feed). This ordering in increasing costs was used to establish the order of listing of the concepts in Table 6-1. The ordering was found to be independent of the buy size between 5 and 30. Next to cost, the second most important criterion was judged to be the additional payload capability and so these data are shown in Column 2 of the table. DDT&E cost, first unit cost, and reliability make up the balance of the evaluation table. Except for candidate 4, all reliabilities are acceptable.

In conducting the comparative evaluation, it was recognized that life cycle costs (LCC) should be considered. However, detailed data on maintenance requirements of the alternative concepts was not available at this point in the study schedule. Nevertheless, a sensitivity analysis showed that the incorporation of operational costs with the costs shown (so as to obtain LCC)

Table 6-1. Comparative Evaluation of Dedicated Cryogenic APS Concepts

Concept No.	Concept Description	Production Cost	Additional Payload Capability kg (lb)	Cost: DDT&E (M$)	Cost: 1 st Unit (M$)	Reliability
3	Tank Wall-Cooled, Pressure Feed	26.0	0	11.4	1.44	0.9411
5	Tank Wall-Cooled, Bladder Feed	26.5	- 5 (- 12)	11.2	1.49	0.9349
0	Updated Baseline, Pressure Feed	27.2	0	11.6	1.52	0.9599
6	Tank Wall-Cooled, Pump Feed	28.1	+248 (+546)	12.6	1.54	0.9472
7	Pump and Bellows Feed	28.8	- 44 (- 96)	12.5	1.61	0.9314
1	Pump Feed	29.5	+229 (+504)	13.0	1.62	0.9472
4	Modular Tank Wall-Cooled, Pressure Feed	30.8	- 46 (-102)	11.3	1.87	0.8801

would not change the relative standings of the top contenders. The bladder-fed system would require more costly maintenance than the pressure-fed system, and the pump-fed system would likely require even more costly maintenance. These increases are in the same direction as their cost standings of Table 6-1. Based on prior Tug system study estimates of LCC, the added maintenance costs also are considered to be less than the differentials.

The data in Table 6-1 show Concept 3 as the leading contender from a cost standpoint. There would be no substantial reason for choosing Concept 5 because not only does it cost more than Concept 3 but it also imposes a payload penalty of 5.4 kg (12 lb). Moreover, if SR&T costs were included in this evaluation, Concept 5 would be even more inferior. The bladder material does not exist even in prototype form. Therefore, Concept 5 represents a high risk/high SR&T cost concept.

Concept 0 is essentially the same as Concept 3 except for fine distinctions in cooling method and so should be carried along with Concept 3 into further analyses in subsequent tasks. Concept 6 costs approximately $2M more than Concept 3 and might be considered for elimination on that basis. However, it enables the Tug to increase its payload capability by approximately 248 kg (546 lb) over Concept 3. This could be a highly advantageous attribute and could result in cost savings in the payloads or in other subsystems that far exceed the $2M cost increase shown. Consequently, it is recommended that Concept 6 be included in subsequent analyses and evaluations.

Concept 7 costs more than Concept 6 and incurs a 44-kg (96-lb) payload penalty, and so can likely be eliminated from further consideration. As was the case in Concepts 3 and 0, Concept 1 is essentially the same as 6 except that it also differs in cooling method in the same manner. Additional refinements in the design could make it more attractive from a cost standpoint than Concept 6. Concept 4 can be eliminated because of cost, weight penalty, and intrinsically low reliability.

Based on evaluation of the data in Table 6-1, the pressure-fed Concept 3 and pump-fed Concept 6 were carried into subsequent analyses and evaluations, and APS Concepts 5, 7, and 4 were eliminated from further consideration. Concepts 0 and 1 were not rejected, being only minor cooling method variations of Concepts 3 and 6, respectively. However, for practical purposes of the study, they were excluded from subsequent evaluations in order to minimize the number of concepts considered.

EXPANDED CONCEPTS DATA

The two selected dedicated APS designs were subjected to further investigation in order to expand their descriptions. First unit and DDT&E costs were updated to be on the same basis as subsequent comparisons. Component replacement costs per vehicle were estimated for a 20-mission Tug lifetime. These data were used to develop fleet maintenance/refurbishment costs.

Replacement rates were based on two factors: (1) an estimate of the normal rate due to maintenance procedures usual for the component type, and (2) the life of the component without excess reliability degradation as implied by its failure rate and worst-case duty cycle.

The data are shown in Tables 6-2 and 6-3. The final entry (Table 6-3) is the incremental life cycle cost for the concept. It represents the life cycle total for the concept of all costs considered in this study and, as such, is the concept total cost measure to be used for subsequent evaluations. It includes only APS DDT&E, system production costs as required for 10-year Tug fleet operations, and component replacement costs necessary to maintain reliability at the design level.

Although SR&T requirements were defined as part of the expanded data, they were subsequently found to be similar for all cryogenic concepts. For this reason, as well as those previously cited, SR&T costs were not a selection factor. The SR&T requirements identified in the conceptual study phase are discussed in Section 5.2.

Table 6-2. Candidate D-3--Pressure Feed, Tank Wall Cooled--
Component Replacement Requirements

Component	Schematic ID No.	No. Per Tug	Failure Rate/ Million hr (cycles)	Worst-Case Exposure Cycles (hr)/ Mission	Predicted No. of Replacements for 20 Missions Over 10 Years		
					Per Component		Total per Vehicle
					Sched	Unsched	
Disconnect Valve, Liquid	1,2,23,27	4	32.5	(164)	0	0.1066	0
Disconnect Valve, Gas	3	1	32.5	2	0	0.0013	0
He Relief Valve	5	1	45	10	0	0.009	0
Helium Tank	4	1	0.3	(164)	0	0.0098	0
He Filter	6	1	0.057	(104)	5	2.0×10^{-5}	5
He Regulator	7	3	18	5060	10	0.182	31
He Check Valve	8,9	8	38	5060	5	0.385	42
He Heater	10	1	0.11	(131)	0	0.00029	0
He Solenoid Valve	36	3	32.5	(164)	0	0.107	1
Pressure Switch	37	6	45	(164)	0	0.147	1
LOX Tank Capillary Device	11	1	0.11	(164)	0	0.00036	0
LH_2 Tank (Sump) Capillary Device	13	1	0.11	(164)	0	0.00036	0
LH_2 Tank (Upstream) Capillary Device	14	2	0.11	(164)	0	0.00036	0
Solenoid Valve, Liquid	15,18,20,24, 28,35,38	16	32.5	(164)	1	0.1066	3
LOX Tank	17	1	0.3	(164)	0	0.00098	0
LH_2 Tank	19	3	0.3	(164)	0	0.00098	0
Bleed Expander	16	1	0.75	(164)	0	0.00246	0
Bleed Heater	49	1	0.11	(131)	0	0.00029	0
Cooling Coils	50	4	0.02	(164)	0	6.56×10^{-5}	0
Insulation	17,19	4	1.0	(164)	1	0.000164	4
Prop. Relief Valve	21,25	2	45	(164)	2	0.1476	2
Thruster Assembly	29	16			0		
Solenoid Valve		32	14.4	2,300	0	0.6624	22
Igniter		16	0.6	9,200	0	0.1104	2
Nozzle/Chamber		16	0.16	(.25)	0	3.2×10^{-6}	0
Instrumentation Set		43			0	10% Assumed	5

Table 6-3. Dedicated Concepts--Incremental Life Cycle Costs

Item	Pressure-Fed (D-3)	Pump-Fed (D-6)
First Unit		
Vehicle systems	0.0	.01
Thrusters	1.24	1.24
Feed system	.65	.74
Test, engineering, business management, etc.	.24	.24
Total	2.13	2.23
DDT&E		
Vehicle systems		.04
Thrusters (GFP)	5.81	5.81
Feed system components	1.05	2.02
System test hardware	1.06	1.16
System test	.76	.76
Engineering business management, etc.	2.84	2.84
Total	11.52	12.63
Production & Refurbishment		
Ship sets (17)	27.05	28.32
Replacement items	7.26	7.07
Replacement operations	3.63	3.54
Total	37.94	38.93
APS Incremental Life Cycle (17 ships)	49.46	51.56

6.3 INTEGRATED SYSTEMS EVAULATION AND FINAL SELECTION

Integrated APS data were generated in a manner similar to that described for the dedicated systems and assembled in matrix form for evaluation. The results of the comparison are shown in Table 6-4. The integrated concepts represent a single APS design, but have either a battery or a fuel cell power option.

The candidate concepts are listed in order of decreasing life cycle cost. An examination of the data reveals that the integrated cryogenic APS concepts are dominant in that they are superior to the dedicated concepts in every criterion.

The DDT&E costs display considerable uniformity with the exception of the dedicated pressure-fed design. The first unit costs also display relatively little variation, the highest being only 4 percent larger than the lowest.

The basic performance of the four competing designs in implementing Mission A was established and although all systems are adequate, the integrated concepts are superior. In determining payload capabilities for Mission B, the dedicated concepts were found to be inadequate. The basic difference in mission flexibility between the dedicated and integrated cryogenic concepts stems from the main engine backup capability of the integrated versions. Both the I5-1 and I5-2 concepts provide MPS backup capability for up to 60 percent of the main engine duty cycle without payload jettison or abort propellant margin. In addition, the fuel cell in I5-2 can provide 1.0 kw of additional electrical power over 98 percent of the mission. These data are also incorporated into the evaluation matrix.

It is not possible to identify unequivocally a preferred power option for the integrated concept. While I5-2 displays a slightly lower life cycle cost, its payload capability is 48 kg (106 lb) less than that of I5-1. Although this is only a 2 percent difference, it could exert a significant impact on the cost of payload placement because it occurs at the borderline of feasibility of dual placement of communication satellites.

The additional electrical power that can be provided by the I5-2 concept is of nebulous value at this point. It could conceivably be beneficial on some activities (e.g., increasing the power output of a target acquisition radar or providing additional power to a particular payload).

Because of the marginal advantages displayed by I5-1 and I5-2 in different criteria, both concepts will be compared against storable monopropellant and bipropellant systems in Section 8.

Table 6-4. Evaluation Matrix for Cryogenic APS Concepts

Concept	Cost ($M)					Payload To Orbit kg (lb)	MPS Backup Capability (Burn Time Fraction)	Additional Elec. Pwr (kw/Mission Fraction)
	Life Cycle	DDT&E	Production of 17	Operation (Maint.)	First Unit			
Mission A								
I5-2 Integrated Cryo (Fuel Cell)	48.19	12.56	28.23	7.40	2.22	2389 (5267)	0.60	1.0/0.98
I5-1 Integrated Cryo (Battery)	49.01	12.56	27.93	8.52	2.20	2437 (5373)	0.60	None
D3 Dedicated Cryo (Pressure-Fed)	49.46	11.52	27.05	10.89	2.13	2030 (4475)	None	None
D6 Dedicated Cryo (Pump-Fed)	51.56	12.63	28.32	10.61	2.23	2331 (5139)	None	None
Mission B								
I5-2 Integrated Cryo (Fuel Cell)	48.19	12.56	28.23	7.40	2.22	2195 (4839)	0.60	1.0/0.98
I5-4 Integrated Cryo Battery	49.01	12.56	27.93	8.52	2.20	2146 (4731)	0.60	None

7. STORABLE APS DESIGN DESCRIPTION

Storable propellant APS designs were analyzed by the Tug systems study contractors (References 1 and 2). These designs, which are monopropellant and bipropellant, provide reference points by which the relative advantages of the selected liquid-liquid O/H system may be measured. Before a fair comparison can be made, however, the storable designs must be redefined in a manner which is compatible with the cryogenic system in each of the comparison categories.

In Section 6, the criteria for cost, weight, and performance are developed. Descriptions of the storable designs are given in this section, and compared to the selected cryogenic system in Section 3.

7.1 MONOPROPELLANT SYSTEM DESCRIPTION

The N_2H_4 monopropellant APS is a pressure-regulated system utilizing helium as the pressurant. Two regulators, installed in parallel and operating simultaneously, maintain a constant pressure on the bladder of the single propellant tank. The mechanical schematic is presented in Figure 7-1 and significant operating and capacity characteristics are shown in the process diagram of Figure 7-2.

The system detailed weight and component level cost statement is given in Table 7-1. Table 7-2 includes the system dry weight in the weight summary for the total vehicle. Fluid weights were obtained by analyzing the mission timeline shown in Table 7-3.

A total payload capability of 2337 kg (5149 lb) is obtained for the reference mission using the storable monopropellant system. The MPS was assumed to include a start basket for propellant settling. If APS settling is used, the payload capability is 56 kg (124 lb) less, while THIM propellant settling will increase the payload capability by 50 kg (110 lb).

Critical items of reliability in the system are described in Table 7-4. These may be compared with the cryogenic system reliability elements given in Table 5-13.

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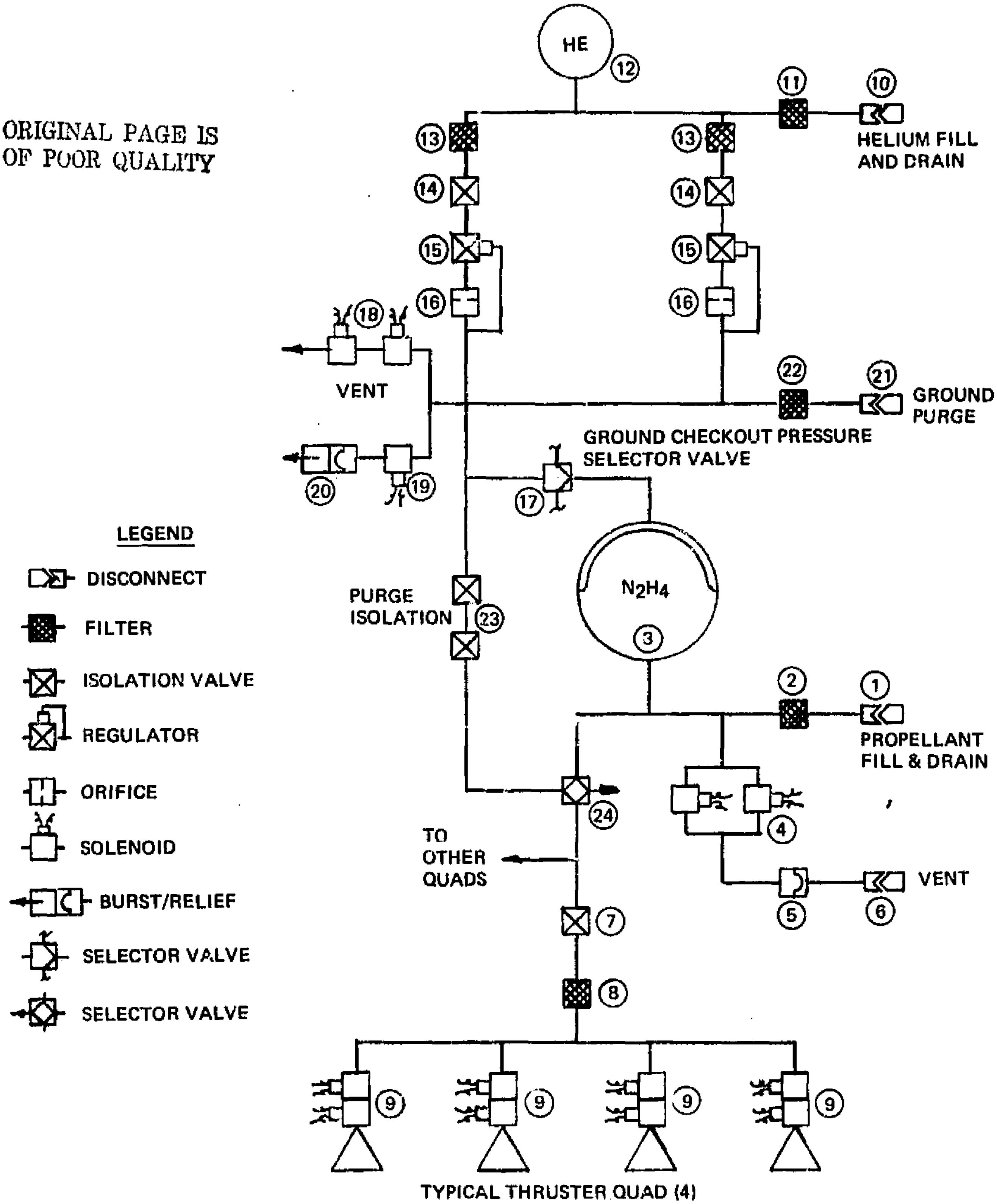

Figure 7-1. Monopropellant APS Mechanical Flow Diagram

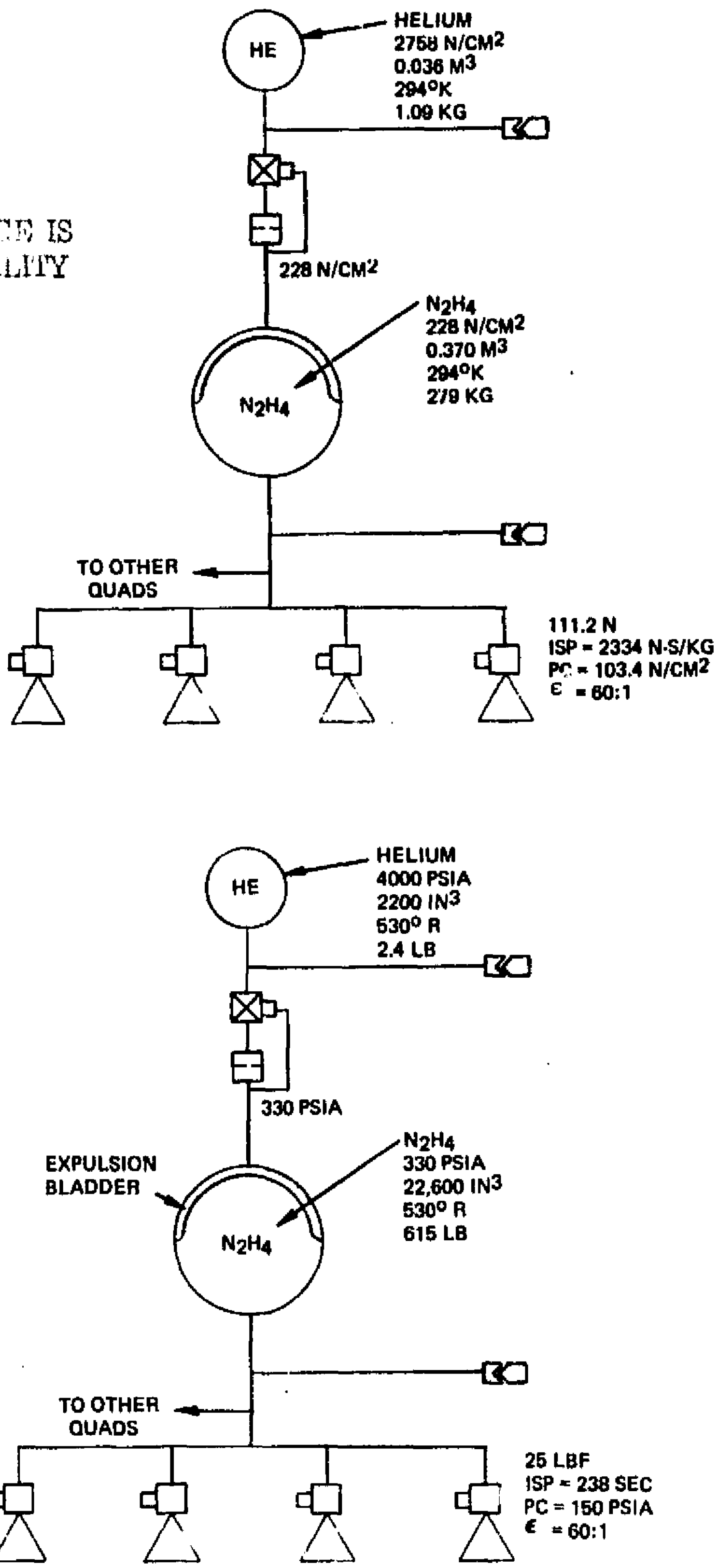

Figure 7-2. Monopropellant APS Process Diagram

Table 7-1. Weight Statement and Cost Summary for Storable Monopropellant System

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG	DDTCE $1000.	1ST UNIT $1000.	REFURB $1000.
FILL AND DRAIN SYSTEM		(2)			(1.5)	(0.7)	(17.1)	(2.7)	(4.6)
N2H4 FILL & DRAIN DISC	1	1	1.0	0.5	1.0	0.5	17.1	2.1	2.1
N2H4 FILL FILTER	2	1	0.5	0.2	0.5	0.2	0.0	0.6	2.5
PRESSURIZATION SYSTEM		(17)			(41.2)	(18.7)	(301.9)	(71.7)	(193.9)
HE FILL & DRAIN DISC	10	1	1.5	0.7	1.5	0.7	33.1	2.1	0.0
HE FILL FILTER	11	1	0.5	0.2	0.5	0.2	0.0	0.6	3.2
HE TANK	12	1	22.7	10.3	22.7	10.3	36.1	7.9	0.0
HE PRESSURIZATION FILTER	13	2	0.5	0.2	1.0	0.5	0.0	1.3	3.2
HE PRESSURE SOL VALVE	14	2	2.0	0.9	4.0	1.8	43.3	12.6	6.3
HE PRESSURE REGULATOR	15	2	1.5	0.7	3.0	1.4	116.1	15.8	165.9
HE PRESSURE ORIFICE	16	2	0.5	0.2	1.0	0.5	0.0	0.3	0.0
GROUND HE SELECTOR VLV	17	1	1.5	0.7	1.5	0.7	28.9	9.5	0.0
GROUND PURGE DISCONNECT	21	1	1.0	0.5	1.0	0.5	17.1	2.1	2.1
GROUND PURGE FILTER	22	1	0.5	0.2	0.5	0.2	0.0	0.6	0.6
PURGE SOLENOID VALVE	23	2	1.5	0.7	3.0	1.4	27.3	12.6	6.3
PURGE CONTROL VALVE	24	1	1.5	0.7	1.5	0.7	0.0	6.3	6.3
PROPELLANT CONTROL SYSTEM		(1)			(0.0)	(0.0)	(272.9)	(63.2)	(63.2)
N2H4 TANK BLADDER	0	1	0.0	0.0	0.0	0.0	272.9	63.2	63.2
PROPELLANT FEED SYSTEM		(17)			(45.3)	(20.5)	(313.1)	(86.3)	(16.4)
N2H4 TANK	3	1	27.3	12.4	27.3	12.4	180.5	39.5	0.0
QUAD ISO SOL VALVE	7	4	3.0	1.4	12.0	5.4	83.3	25.3	6.3
QUAD FEED FILTER	8	4	0.5	0.2	2.0	0.9	0.0	2.5	10.1
LINE & TANK HEATER	0	8	0.5	0.2	4.0	1.8	49.2	19.0	0.0
OVERBOARD VENT		(8)			(9.5)	(4.3)	(212.2)	(36.0)	(6.3)
N2H4 RELIEF ISO SOL VLV	4	2	1.5	0.7	3.0	1.4	43.3	12.6	6.3
N2H4 BURST DISCONNECT	5	1	0.5	0.2	0.5	0.2	40.4	0.8	0.0
N2H4 RELIEF DISCONNECT	6	1	1.0	0.5	1.0	0.5	37.1	2.1	0.0
HE VENT SOLENOID VALVE	18	2	1.5	0.7	3.0	1.4	35.3	12.6	0.0
HE RELIEF ISO SOL VALVE	19	1	1.5	0.7	1.5	0.7	35.3	6.3	0.0
HE BURST/RELIEF VALVE	20	1	0.5	0.2	0.5	0.2	24.8	1.6	0.0
THRUSTER QUAD		(64)			(80.0)	(36.3)	(1332.3)	(935.4)	(228.8)
THRUST CHAMBER & NOZZLE	9	16	5.0	2.3	80.0	36.3	1298.7	573.9	0.0
SOLENOID VALVE	0	16	0.0	0.0	0.0	0.0	0.0	177.0	44.2
CATALYST BED	0	16	0.0	0.0	0.0	0.0	0.0	134.0	134.0
CATALYST BED HEATER	0	16	0.0	0.0	0.0	0.0	33.6	50.6	50.6
INSTRUMENTATION	0	(25)	0.4	0.2	(10.0)	(4.5)	(16.7)	(31.6)	(3.8)
COMPONENT TOTAL		134			187.5	85.0	2466.0	1226.9	517.0
LINES					16.0	7.3			
					0.0	0.0			
COMPONENT MOUNTINGS					9.4	4.3			
DRY SYSTEM					212.9	96.6			

Table 7-2. Vehicle Weight Summary for Storable Monopropellant System

DESCRIPTION	WEIGHT LB	WEIGHT KG
STRUCTURE	(2127.)	(965.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1336.)	(606.)
MAIN PROPULSION	1122.	539.
AUXILIARY PROPULSION	214.	97.
DRY WEIGHT	4869.	2209.
CONTINGENCY (13%)	633.	287.
TOTAL DRY SYSTEM	5502.	2496.
NONUSUABLE FLUIDS	(540.)	(245.)
APS TRAPPED PROPELLANT	14.	6.
APS PRESSURANT	2.	1.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	370.	168.
FLIGHT RESERVES	(375.)	(170.)
APS RESERVE (10%)	55.	25.
MPS RESERVE	320.	145.
BURNOUT WEIGHT	6417.	2911.
EXPENDED FLUIDS	(50831.)	(23056.)
APS USABLE PROPELLANT	554.	251.
MPS USABLE PROPELLANT	49993.	22676.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	134.	61.
PAYLOAD	(5149.)	(2335.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

Table 7-3. Mission Timeline for Propulsion Events - Monopropellant System

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0 7	DEPLOY TUG	APS				3	95120	2622	28309	16432	488364
6.39	4	4 38	COAST NO 1							2795	28259	16427	487897
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	163					8793	28254	16426	487832
7.74	6	1 19	COAST NO 2							2595	27249	16299	476832
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	2384					8723	27247	16298	476839
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)	PIM		15	245167			6298	16113	14886	369213
13.13	9	5 6	COAST NO 3							2106	16051	14878	368453
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	1785					6368	16046	14878	368389
13.33	11	0 2	ORIENT PAYLOAD	APS				3	36375	1506	10823	14216	278927
13.33	12	0 0	DEPLOY PAYLOAD 1 (779 KG)								10806	14213	278676
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				9	100967	1234	10028	12179	223927
37.14	14	23 45	COAST NO 4							2629	9982	12173	223350
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		95	842348			4039	9962	12171	223095
89.36	16	52 0	COAST NO 5							4335	9772	12147	220488
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		85	822300			3586	9728	12141	220133
89.41	18	0 2	ORIENT PAYLOAD	APS				3	32071	1181	9543	12118	217759
89.41	19	0 0	DEPLOY PAYLOAD 2 (779 KG)								9528	12116	217567
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				9	88093	873	8749	10082	155274
95.36	21	5 50	COAST NO 6							1270	8709	10077	154836
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		95	735735			2815	8704	10076	154777
147.52	23	52 9	COAST NO 7							4705	8538	10055	152942
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		85	717965			2774	8494	10049	152458
147.58	25	0 2	ORIENT PAYLOAD	APS				3	29002	827	8332	10029	150660
147.58	26	0 0	DEPLOY PAYLOAD 3 (779 KG)								8319	10027	150517
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				9	75924	469	7541	7993	77337
150.18	28	2 31	COAST NO 8							682	7506	7989	76985
153.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	1783					1456	7504	7988	76962
155.44	30	5 10	COAST NO 9							678	5064	7679	51937
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		15	76944			979	5059	7678	51890
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	1693					996	5038	7675	51665
157.63	33	2 5	COAST NO 10							480	3467	7476	35559
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	762					699	3465	7476	35539
163.56	35	5 53	COAST NO 11							973	2927	7408	30021
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	24526	202	2922	7407	29968
163.59	37	0 0	SHUTTLE DEORBIT								2911	7407	29969
164.22	38	0 42	TOUCHDOWN										
TOTALS					8571	370	3440159	47	481078	80893			

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM								65000		
1.75	3	0 7	DEPLOY TUG	APS				10	21384	590	62397	12120	360204
6.39	4	4 38	COAST NO 1							628	62300	12116	359852
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	536					1977	62290	12115	359811
7.74	6	1 19	COAST NO 2							583	60073	12021	351698
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	7820					1961	60069	12021	351681
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)	RJM		50	55116			1416	35523	10980	257570
13.13	9	5 6	COAST NO 3							474	35386	10974	257008
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	5859					1427	35375	[illegible]	256962
13.33	11	0 2	ORIENT PAYLOAD	APS				10	8177	339	23861	10463	205729
13.33	12	0 0	DEPLOY PAYLOAD 1 (1716 LB)								23824	10483	205544
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				30	22698	277	22108	8983	165142
37.14	14	23 45	COAST NO 4							591	22007	8979	164734
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		280	189300			908	21962	8977	164549
89.36	16	52 0	COAST NO 5							974	21543	8959	162773
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		280	184861			896	21447	8955	162364
89.41	18	0 2	ORIENT PAYLOAD	APS				10	7210	266	21038	8939	160613
89.41	19	0 0	DEPLOY PAYLOAD 2 (1716 LB)								21005	8936	160471
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				30	19804	196	19289	7436	114524
95.36	21	5 50	COAST NO 6							286	19201	7432	114207
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		280	165400			633	19189	7432	114159
147.52	23	52 9	COAST NO 7							1058	18829	7416	112806
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		280	161405			624	18727	7412	112449
147.58	25	0 2	ORIENT PAYLOAD	APS				10	6295	186	18369	7397	111123
147.58	26	0 0	DEPLOY PAYLOAD 3 (1716 LB)								18341	7396	111017
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				30	17068	105	16624	5895	57041
150.18	28	2 31	COAST NO 8							153	16549	5892	56782
153.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	5851					327	16544	5892	56765
155.44	30	5 10	COAST NO 9							197	11164	5664	39307
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		50	17298			720	11154	5663	39273
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	5555					224	11106	5661	39107
157.63	33	2 5	COAST NO 10							108	7644	5514	26227
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2500					157	7640	5514	26213
163.56	35	5 53	COAST NO 11							219	6453	5464	22143
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	5514	45	6442	5463	22134
163.59	37	0 0	SHUTTLE DEORBIT								6418	5463	22104
164.22	38	0 42	TOUCHDOWN										
TOTALS					28121	1220	773379	155	108151	18185			

Table 7-4. Storable Monopropellant System Reliability Summary

Failure Mode	Component	Generic Failure Rate	Redundancy	Failure Contrib. 10^6 Missions	
Thrust loss	Thruster (16)	0.27	14 of 16		
Propellant loss	Thruster valve (32)	4.8	Standby	86	
	Isolation valve (4)	6.5	None		
	Tank & bladder (1)	1.5		301	835
Pressure Loss	Vent & relief valve (2)	9.0	Standby	2	
	Vent & relief valve (1)	9.0	None		$\bar{R}$ = .9992
All others	-	-	-	446	

7.2 BIPROPELLANT SYSTEM DESCRIPTION

The N_2O_4/MMH bipropellant APS utilizes a single bladder tank for each of the propellants. A constant pressure is maintained on the bladders by use of a pressure regulator which has two redundant standby regulators in parallel. The use of these backup regulators is controlled by pressure switches and solenoid valves.

Two 98-N (22-lb) and two 400-N (90-lb) thrusters are used in each of four quads to provide the desired thrust levels for translation, pitch, yaw, and roll maneuvers. Figure 7-3 is the mechanical flow diagram for this system and Figure 7-4 is the process flow diagram and shows the pertinent tank and system operating characteristics.

A detailed system weight and cost statement was prepared from Figure 7-3, as shown in Table 7-5. The data sources were the cryogenic Tug system studies, other programs (72-2, CTS, etc.), and direct contact with suppliers. The data from the weight breakdown were then analyzed on a mission basis to determine propellant and pressurant quantities. The vehicle weight summary is given in Table 7-6, while the timeline is described in Table 7-7.

As for the monopropellant design, the bipropellant system capability was assessed assuming that a start basket is used. The payload capability is 2259 kg (4981 lb). Use of the APS for propellant settling reduces the payload by 48 kg (105 lb) and use of THIM settling increases the payload by 50 kg (110 lb).

The important reliability elements of the system are described in Table 7-8, which may be compared with the cryogenic system reliability elements given in Table 5-13.

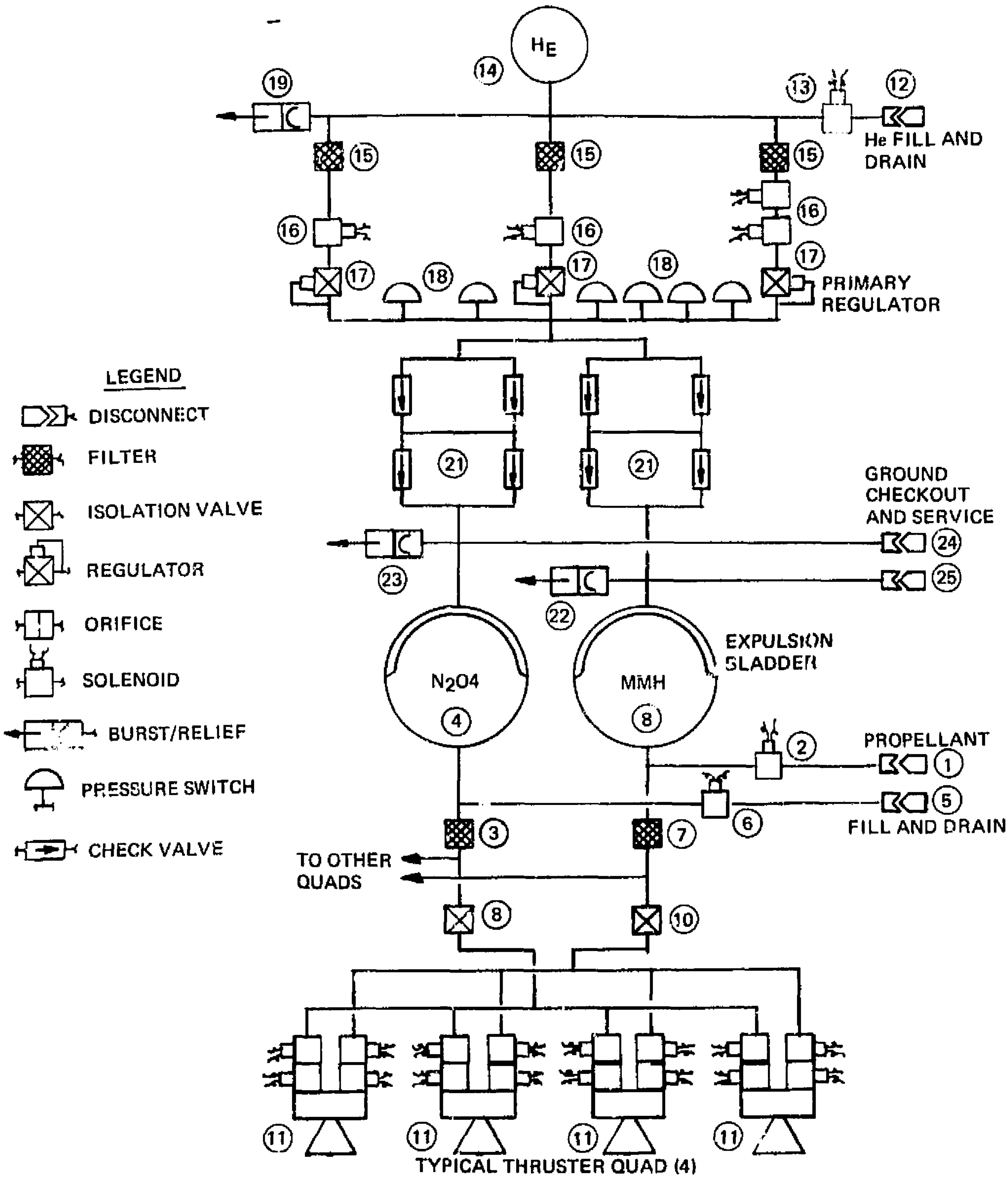

Figure 7-3. Bipropellant APS Mechanical Flow Diagram

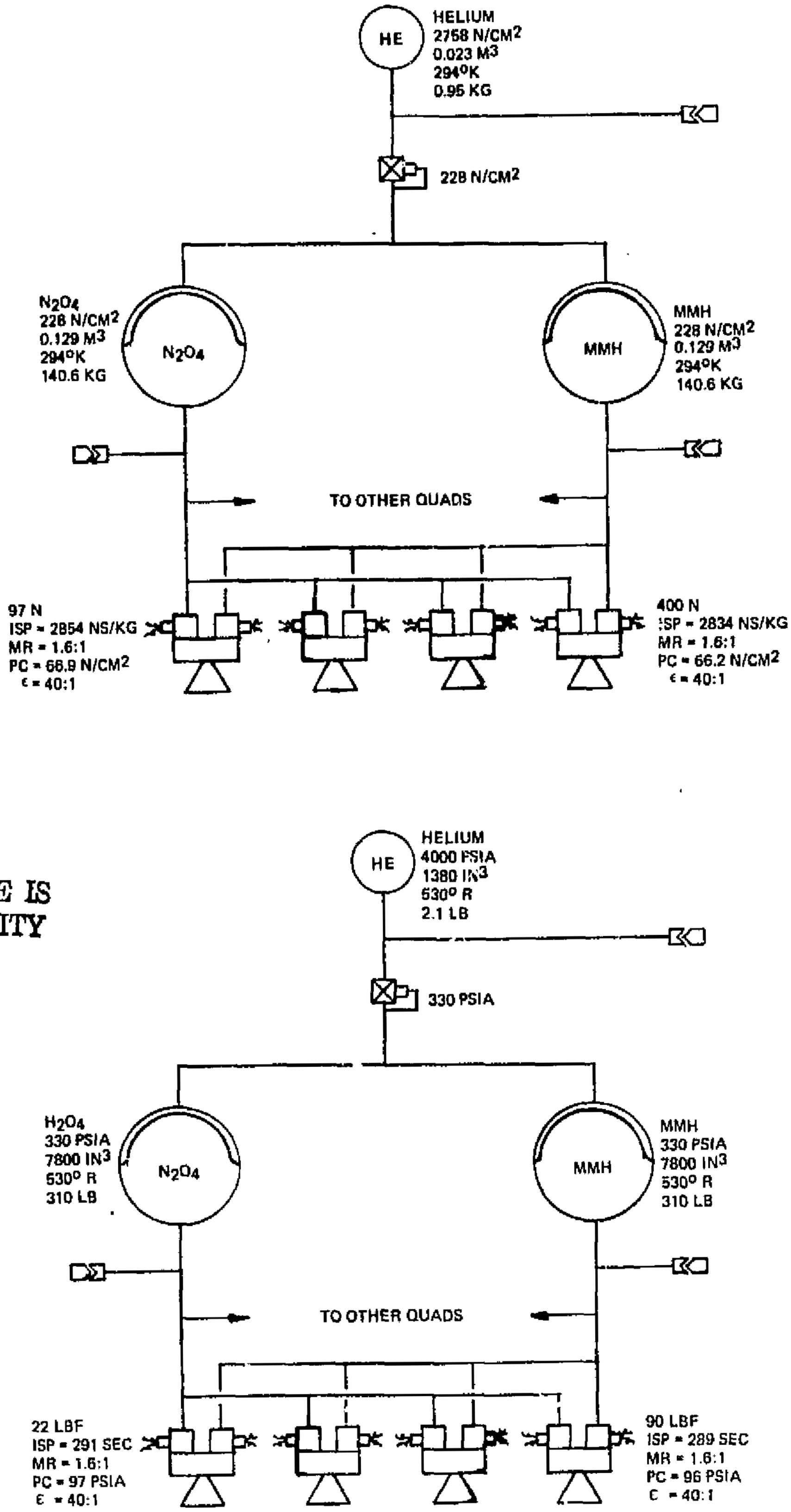

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Figure 7-4. Bipropellant APS Process Diagram

Table 7-5. Weight Statement and Cost Summary for Storable Bipropellant System

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG	DDT&E $1000.	1ST UNIT $1000.	REFURB $1000.
FILL AND DRAIN SYSTEM		(4)			(5.0)	(2.3)	(96.7)	(16.7)	(12.6)
N2O4 FILL DISCONNECT	1	1	1.0	0.5	1.0	0.5	17.1	2.1	0.0
N2O4 FILL SHUTOFF	2	1	1.5	0.7	1.5	0.7	27.3	6.3	6.3
MMH FILL DISCONNECT	5	1	1.0	0.5	1.0	0.5	25.1	2.1	0.0
MMH FILL SHUTOFF VALVE	6	1	1.5	0.7	1.5	0.7	27.3	6.3	6.3
PRESSURIZATION SYSTEM		(28)			(45.4)	(20.6)	(391.9)	(137.0)	(432.6)
HE FILL DISCONNECT	12	1	1.5	0.7	1.5	0.7	33.1	2.1	0.0
HE FILL SHUTOFF VALVE	13	1	1.5	0.7	1.5	0.7	43.3	6.3	0.0
HE TANK	14	1	19.5	8.8	19.5	8.8	36.1	7.9	0.0
HE PRESSURE FILTER	15	3	0.5	0.2	1.5	0.7	0.0	1.9	9.5
HE PRESSURE SHUTOFF VLV	16	4	2.0	0.9	8.0	3.6	27.3	25.3	6.3
HE PRESSURE REGULATOR	17	3	1.5	0.7	4.5	2.0	116.1	23.7	244.9
HE PRESSURE SWITCH	18	6	1.0	0.5	6.0	2.7	19.3	37.9	12.6
HE BURST/RELIEF VALVE	19	1	0.5	0.2	0.5	0.2	32.8	1.6	0.0
MMH HE PRESSURE CHECK V	20	4	0.3	0.1	1.2	0.5	42.0	15.2	79.6
N2O4 HE PRESSURE CHECK V	21	4	0.3	0.1	1.2	0.5	42.0	15.2	79.6
PROPELLANT CONTROL SYSTEM		(2)			(0.0)	(0.0)	(545.7)	(126.4)	(126.4)
N2O4 TANK BLADDER	0	1	0.0	0.0	0.0	0.0	272.9	63.2	63.2
MMH TANK BLADDER	0	1	0.0	0.0	0.0	0.0	272.9	63.2	63.2
PROPELLANT FEED SYSTEM		(28)			(59.0)	(26.8)	(474.8)	(152.3)	(15.5)
N2O4 FILL FILTER	3	1	0.5	0.2	0.5	0.2	0.0	0.6	2.5
N2O4 TANK	4	1	13.9	6.3	13.9	6.3	136.4	31.6	0.0
MMH FILL FILTER	7	1	0.5	0.2	0.5	0.2	0.0	0.6	2.5
MMH TANK	8	1	13.9	6.3	13.9	6.3	136.4	31.6	0.0
N2O4 ISOLATION VALVE	9	4	3.0	1.4	12.0	5.4	59.3	25.3	6.3
MMH ISOLATION VALVE	10	4	3.0	1.4	12.0	5.4	59.3	25.3	0.0
MMH CHECKOUT DISCONNECT	24	1	1.0	0.5	1.0	0.5	17.1	2.1	2.1
N2O4 CHECKOUT DISCONNECT	25	1	1.0	0.5	1.0	0.5	17.1	2.1	2.1
LINE & TANK HEATER	0	14	0.3	0.1	4.2	1.9	49.2	33.2	0.0
OVERBOARD VENT		(2)			(1.0)	(0.5)	(49.6)	(3.2)	(1.6)
MMH BURST/RELIEF VALVE	22	1	0.5	0.2	0.5	0.2	24.8	1.6	1.6
N2O4 BURST/RELIEF VALVE	23	1	0.5	0.2	0.5	0.2	24.8	1.6	0.0
THRUSTER QUAD		(64)			(85.6)	(38.8)	(1100.0)	(1049.1)	(126.4)
90-LB THRUST CHAMB & NOZ	11	8	4.0	1.8	32.0	14.5	533.4	328.6	0.0
THRUSTER HEATER	0	16	0.3	0.1	4.8	2.2	33.2	37.9	37.9
90-LB THRUSTER VALVE	0	16	0.6	0.3	9.6	4.4	0.0	177.0	44.2
22-LB THRUST CHAMB & NOZ	11	8	3.7	1.7	29.6	13.4	533.4	328.6	0.0
22-LB THRUSTER VALVE	0	16	0.6	0.3	9.6	4.4	0.0	177.0	44.2
INSTRUMENTATION	0	(27)	0.4	0.2	(10.8)	(4.9)	(16.7)	(34.1)	(5.1)
COMPONENT TOTAL		155			206.8	93.8	2675.4	1518.9	720.2
LINES					26.5	12.0			
					0.0	0.0			
COMPONENT MOUNTINGS					10.3	4.7			
DRY SYSTEM					243.6	110.5			

Table 7-6. Vehicle Weight Summary for Storable Bipropellant System

DESCRIPTION	WEIGHT LB	WEIGHT KG
STRUCTURE	(2127.)	(965.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1367.)	(620.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	245.	111.
DRY WEIGHT	4900.	2223.
CONTINGENCY (13%)	637.	289.
TOTAL DRY SYSTEM	5537.	2512.
NONUSABLE FLUIDS	(537.)	(244.)
APS TRAPPED PROPELLANT	11.	5.
APS PRESSURANT	2.	1.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	370.	168.
FLIGHT RESERVES	(380.)	(172.)
APS RESERVE (10%)	60.	27.
MPS RESERVE	[illegible]	145.
BURNOUT WEIGHT	6454.	2927.
EXPENDED FLUIDS	(50962.)	(23116.)
APS USABLE PROPELLANT	595.	270.
MPS USABLE PROPELLANT	50083.	22717.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	134.	61.
PAYLOAD	(4981.)	(2259.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

Table 7-7. Mission Timeline for Propulsion Events - Bipropellant System

MISSION TIME HR	EVENT NO	DURATION HR	DURATION MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0	0	LIFTOFF										
	1			SHUTTLE BURNOUT										
1.63	2	1	38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0	7	DEPLOY TUG	APS				3	95136	2841	28303	16243	492435
6.39	4	4	38	COAST NO 1							5139	28268	16239	492139
6.43	5	0	3	PHASING ORBIT INSERTION	MAIN	163					8663	28242	16237	492044
7.74	6	1	19	COAST NO 2							3218	27257	16113	471994
7.99	7	0	15	TRANSFER ORBIT INSERTION	MAIN	2384					8629	27255	16110	471373
9.02	8	0	2	MIDCOURSE CORRECT (DV 1)	PIM		15	245290			6199	16119	14697	344315
13.13	9	5	6	COAST NO 3							5046	16055	14693	343569
13.26	10	0	8	MISSION ORBIT INSERTION	MAIN	1786					6265	16051	14683	343431
13.30	11	0	2	ORIENT PAYLOAD	APS				3	36395	1590	10827	14026	275044
13.30	12	0	0	DEPLOY PAYLOAD 1 (753 KG)								10814	14025	274852
13.41	13	0	7	PAYLOAD SURVEILLANCE	APS				9	101345	1587	10061	12357	221125
37.16	14	23	45	COAST NO 4							19886	10025	12057	220672
37.19	15	0	2	PHASING ORBIT INSERT (DV 2)	MAIN		85	944329			3988	9996	12049	220313
89.34	16	52	9	COAST NO 5							42389	9805	12029	217934
89.37	17	0	2	MISSION ORBIT INSERT (DV 3)	MAIN		85	823514			3932	9742	12017	217143
89.41	18	0	2	ORIENT PAYLOAD	APS				3	32124	1279	9557	11993	214902
89.41	19	0	0	DEPLOY PAYLOAD 2 (753 KG)								9545	11992	214653
89.51	20	0	6	PAYLOAD SURVEILLANCE	APS				9	88562	1258	8792	10024	153815
95.34	21	5	50	COAST NO 6							6459	8760	10020	153467
95.37	22	0	2	PHASING ORBIT INSERT (DV 4)	MAIN		35	739811			2792	8752	10019	153381
147.52	23	52	9	COAST NO 7							51364	8586	9993	151549
147.55	24	0	2	MISSION ORBIT INSERT (DV 5)	MAIN		85	719982			2747	8518	9949	150835
147.58	25	0	2	ORIENT PAYLOAD	APS				3	28307	942	8356	9969	149315
147.59	26	0	0	DEPLOY PAYLOAD 3 (753 KG)								8345	9967	148901
147.67	27	0	5	PAYLOAD SURVEILLANCE	APS				9	766[illegible]	301	7592	7999	77867
150.19	28	2	31	COAST NO 8							3684	7565	7996	77596
150.27	29	0	5	TRANSFER ORBIT INSERTION	MAIN	1783					1480	7561	7996	77547
155.44	30	5	10	COAST NO 9							7755	5103	7684	52334
155.47	31	0	2	MIDCOURSE CORRECT (DV 6)	THIM		15	77486			990	5095	7681	52251
155.54	32	0	4	PHASING ORBIT INSERTION	MAIN	1693					1013	5013	7680	52025
157.63	33	2	5	COAST NO 10							3611	3491	7479	35904
157.68	34	0	3	CIRCULARIZE FOR RENDEZVOUS	MAIN	762					711	3488	7479	35772
163.56	35	5	53	COAST NO 11							10317	2946	7410	30219
163.59	36	0	2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	24656	352	2936	7409	30117
163.59	37	0	0	SHUTTLE DEORBIT								2927	7409	30116
164.29	38	0	42	TOUCHDOWN										
TOTALS						8571	370	3451311	47	482784	282401			

MISSION TIME HR	EVENT NO	DURATION HR	DURATION MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0	0	LIFTOFF										
	1			SHUTTLE BURNOUT										
1.63	2	1	38	CIRCULARIZE AT 160 NM								65000		
1.75	3	0	7	DEPLOY TUG	APS				10	21387	639	62397	11980	355867
6.39	4	4	38	COAST NO 1							1155	62321	11977	355590
6.43	5	0	3	PHASING ORBIT INSERTION	MAIN	536					1947	62308	11976	355542
7.74	6	1	19	COAST NO 2							724	60393	11882	347465
7.99	7	0	15	TRANSFER ORBIT INSERTION	MAIN	7820					1940	60388	11882	347443
9.02	8	0	2	MIDCOURSE CORRECT (DV 1)	PIM		50	55143			1394	35536	10840	253957
13.13	9	5	6	COAST NO 3							1134	35401	10835	253404
13.26	10	0	8	MISSION ORBIT INSERTION	MAIN	5859					1408	35397	10834	253349
13.30	11	0	2	ORIENT PAYLOAD	APS				10	8182	357	23871	10346	202867
13.30	12	0	0	DEPLOY PAYLOAD 1 (1660 LB)								23841	10344	202721
13.41	13	0	7	PAYLOAD SURVEILLANCE	APS				30	22783	357	22181	8893	163096
37.16	14	23	45	COAST NO 4							4470	22100	8890	162762
37.19	15	0	2	PHASING ORBIT INSERT (DV 2)	MAIN		280	169948			597	22037	8887	162496
89.34	16	52	9	COAST NO 5							9529	21617	8869	160742
89.37	17	0	2	MISSION ORBIT INSERT (DV 3)	MAIN		290	185133			884	21478	8863	160159
89.41	18	0	2	ORIENT PAYLOAD	APS				10	7222	287	21069	8846	158432
89.41	19	0	0	DEPLOY PAYLOAD 2 (1660 LB)								21043	8845	158322
89.51	20	0	6	PAYLOAD SURVEILLANCE	APS				30	19910	203	19383	7393	113450
95.34	21	5	50	COAST NO 6							1452	19313	7390	113193
95.37	22	0	2	PHASING ORBIT INSERT (DV 4)	MAIN		280	166316			628	19296	7390	113130
147.52	23	52	9	COAST NO 7							17547	18928	7374	111779
147.55	24	0	2	MISSION ORBIT INSERT (DV 5)	MAIN		280	161858			617	18779	7369	111233
147.58	25	0	2	ORIENT PAYLOAD	APS				10	6314	212	18421	7353	109709
147.59	26	0	0	DEPLOY PAYLOAD 3 (1660 LB)								18399	7352	109926
147.67	27	0	5	PAYLOAD SURVEILLANCE	APS				30	17193	203	16738	5903	57433
150.19	28	2	31	COAST NO 8							828	16678	5898	57224
150.27	29	0	5	TRANSFER ORBIT INSERTION	MAIN	5851					333	16670	5897	57197
155.44	30	5	10	COAST NO 9							1744	11250	5667	38639
155.47	31	0	2	MIDCOURSE CORRECT (DV 6)	THIM		50	17420			223	11232	5667	38539
155.54	32	0	4	PHASING ORBIT INSERTION	MAIN	5555					228	11183	5665	38372
157.63	33	2	5	COAST NO 10							812	7697	5517	26411
157.68	34	0	3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2500					160	7690	5516	26335
163.56	35	5	53	COAST NO 11							2319	6496	5466	22280
163.59	36	0	2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	5543	79	6474	5465	22213
163.59	37	0	0	SHUTTLE DEORBIT								6454	5465	22211
164.29	38	0	42	TOUCHDOWN										
TOTALS						29121	1720	776919	155	109534	63486			

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Table 7-8. Storable Bipropellant System Reliability Summary

Failure Mode	Compnent	Generic Failure Rate	Redundancy	Failure Contrib. 10^6 Missions
Thrust Loss	Thruster (16)	0.76	14 of 16	263
Propellant Loss	Thruster valve (64)	4.8	Standby Qual	
	Isolation valve (8)	6.5	Standby Isol	1716
	Tank & bladder (2)	3.6	None	1184
Propellant Feed Loss	Feed check valves (8)	9.0	Qual connected	198 $\overline{R}$ = .9983
All Others	-	-	-	71

8. CRYOGENIC CONCEPT EVALUATION

The potential of the selected liquid-liquid O/H APS design was measured by comparison with the more conventional storable propellant APS designs described in the previous section. A common basis for the comparison was established by providing compatible data in the areas of cost, performance, and reliability. In addition, important information of a partly quantitative nature was generated to define the mission flexibility and vehicle versatility attributes of the APS designs.

8.1 COMPARATIVE EVALUATION

Details of the comparison criteria are given in Section 6. Briefly, the most important criterion is cost, including that of the first unit and incremental life cycle (LCC). The LCC is composed of production and refurbishment, as well as DDT&E costs. All flight equipment costs were generated at the component level, in vehicle sets, as shown in Sections 5 and 7. The total system costs, including other elements besides components, are itemized in Table 8-1, where it is seen that storable system costs are lower.

Performance, the second most important criterion, is simply the payload capability of the Tug using the various APS designs in conducting the triple-payload deployment mission. All of the APS designs meet the basic requirements for functional performance and safety.

Lastly, mission flexibility refers to the beneficial characteristics of the APS which simplify the Tug and its other subsystems, or which increase its repertory of mission types. Many of these characteristics are expressed as real or potential cost savings.

Data for each of the areas of comparison are listed in Table 8-2. The first item, reliability, is not a deciding factor; it is viewed a constraint, and all systems are adequate.

Performance is shown in the table in terms of payload weight for Mission A and B. Maximized performance with respect to main engine start is by using APS settling for the integrated designs and start baskets for the storable designs.

As previously discussed, the performance of dedicated systems is invariably reduced for Mission B. Although low, the payload weight entries for the storable systems in Mission B do not include the additional weight of accommodations for the greatly increased tank volumes.

Although the life cycle costs pertaining strictly to the APS are higher for the integrated designs (Table 8-1), the gains in mission flexibility and vehicle versatility (described in Sections 5.3 and 6.3) from the use of the

Table 8-1. Storable/Cryogenic Cost Comparison

	INTEGRATED CRYOGENIC APS ($M)		DEDICATED STORABLE APS ($M)	
	PRIMARY BATTERY POWER 15-1	FUEL CELL POWER 15-2	MONO-PROPELLANT	BI-PROPELLANT
FIRST UNIT				
VEHICLE SYSTEMS	.01	.02	.02	.02
THRUSTERS	1.24	1.25	.94	1.05
FEED SYSTEM	.71	.71	.29	.47
TEST, ENGR, BUSINESS MANAGEMENT, ETC.	.24	.24	.21	.22
TOTAL	2.20	2.22	1.46	1.76
DDT&E				
VEHICLE SYSTEM	.04	.04	.32	.32
THRUSTERS	5.81*	5.81*	1.33	1.10
FEED SYSTEM COMPONENTS	1.99	1.99	1.13	1.58
SYSTEM TEST HARDWARE	1.12	1.12	.58	.80
SYSTEM TEST	.76	.76	.51	.56
ENGINEERING, BUSINESS MANAGEMENT, ETC.	2.84	2.84	2.24	2.33
TOTAL	12.56	12.56	6.11	6.69
PRODUCTION & REFURBISHMENT				
SHIP SETS (17)	27.93	28.23	18.49	22.32
REPLACEMENT ITEMS	5.68	4.93	4.08	5.69
REPLACEMENT OPERATIONS	2.84	2.47	2.04	2.85
TOTAL	36.45	35.63	24.61	30.86
APS INCREMENTAL LIFE CYCLE (17 SHIPS)	49.01	48.19	30.72	37.55

*GFP

Table 8-2. Storable/Cryogenic System Comparison

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	INTEGRATED CRYOGENIC APS		DEDICATED STORABLE APS	
	PRIMARY BATTERY 15-1	FUEL CELL POWER 15-2	MONO-PROPELLANT	BI-PROPELLANT
RELIABILITY	ADEQUATE	ADEQUATE	SUPERIOR	SUPERIOR
BASIC PERFORMANCE - PAYLOAD KG (LB)				
MISSION A	2437 (5373)	2389 (5267)	2336 (5149)	2259 (4981)
MISSION B	2146 (4731)	2195 (4839)	360 (794)	904 (1994)
COST - ($M)				
DDT&E	12.6	12.6	6.1	6.7
GROSS LIFE CYCLE (FLEET OF 17)	**49.0**	**48.2**	**30.7**	**37.6**
PROGRAM SAVINGS:				
ELIMINATE MAIN ENGINE PUMPED IDLE (DDT&E)	-2.8	-2.8	-	-
REDUCE MAIN ENGINE REL. DEMO. (DDT&E)	-3.6	-3.6	-	-
REDUCED MAIN ENGINE OVERHAUL (OPS)	-	-.9	-	-
VEHICLE RECOVERY (OPS)	-6.3	-6.3	-	-
PAYLOAD RECOVERY (OPS)	-12.4	-12.4	-	-
NET LIFE CYCLE	**23.9**	**22.2**	**30.7**	**37.6**
UTILITY IMPROVEMENTS				
MAIN ENGINE BACKUP (BURNTIME FRACTION)	.6	.6	-	-
MPS/APS INTERCHANGE (IMPULSE FRACTION)	0	.86	-	-
POWER AVAILABLE (KW/MISSION FRACTION)		1.0/.98		
VEHICLE SYSTEM IMPACT - MAIN ENGINE	**LOWER CRITICALITY**		START TECHNOLOGY DEVELOPMENT	
		LOWER DUTY CYCLE		
DEVELOPMENT RISK POTENTIAL				
ISSUE	CONCEPT TECHNOLOGY		RESIZING	
COST IMPACT ($M)	1.3		.5	.9
SCHEDULE IMPACT (MONTHS)	8		6	6

integrated design appreciably lower that cost. These "net" life cycle costs greatly improve the competitive position of the integrated designs.

Elimination of the main engine pumped idle mode (PIM) is estimated by the manufacturer to reduce the engine development by $2.83 M and two months. The test program is reduced by two engine sets and 50 tests. PIM may be eliminated because the function is inherent in the integrated APS.

The main engine reliability demonstration tests may be reduced, under DDT&E, in view of the added APS backup capability. Approximately 60 percent of the main engine duty cycle, in any round-trip mission, can be backed up by the APS. To assign a value to this potential cost saving, it is estimated that the engine testing can be reduced by 150 tests at a saving of $3.6 M.

By performing additional low-velocity changes, in the case of Mission B, the integrated APS relieves the main engine of these operations and thereby saves a portion of the overhaul costs. It is estimated that, since 6 out of 11 main engine rotating starts are supplanted by the APS in the reference mission, the overhaul cost is reduced by $9000 per flight. The saving could be realized on 100 of the total of 243 flights composing the baseline mission model, for a total savings of $0.9 M.

Operational cost benefits also can be realized through the capability of the integrated APS to support MPS functions. The MPS reliability goal implies that vehicle and payload losses are to be expected; however, many of these losses can be avoided by retrieval with the APS. Assigning $10.5 M and $20 M unit costs to the Tug and a typical payload yields potential cost savings of $ .3 M and 12.4 M, respectively, under these conditions.

The mission flexibility and vehicle versatility contributions to the net life cycle costs represent a total vehicle system impact which is favorable to the integrated system by several million dollars.

In conclusion, analysis of the data reveals that the best dedicated APS, the storable monopropellant concept, has the lowest gross life cycle cost and an adequate payload capability for Mission A. The storable bipropellant APS is slightly higher in gross life cycle cost, but has a lower payload capability for Mission A. On the same basis, the integrated cryogenic systems cost more initially but provide a higher Mission A payload. Also, the integrated systems provide an adequate payload for Mission B.

However, by translating the appropriate programmatic benefits of mission flexibility and vehicle versatility into cost impacts, the resulting net life cycle costs are lower for the integrated cryogenic APS. Furthermore, the integrated cryogenic APS designs have a greater mission growth potential, which would allow the accomplishment of missions by the Tug that were not previously feasible.

8.2 ADVANTAGE ASSESSMENT

The integrated liquid-liquid O/H APS represents a departure from the conventional propulsion stage APS in its capabilities and constraints; in short, it is different in kind, not degree. It should therefore be expected to have distinct advantages as well as relative ones.

Recapitulating, some of the more important advantages of the integrated APS design are:

1. Mission Versatility - Use of the integrated APS allows the Tug to perform missions which require greater auxiliary propulsion activity than do the standard payload delivery and recovery missions.

2. Operational Flexibility - A virtue of the comingled MPS and APS propellant characteristic associated with the integrated APS is that changes to the mission after it has started are not constrained by the individual system propellant allocations. This simplifies permission contingency planning and allows the Tug a capability to surmount unforeseen difficulties.

3. Total Program Cost Savings - Use of the integrated APS would permit cost savings in several areas:

   a. Development of the main engine pumped idle mode may be eliminated.

   b. The main engine DDT&E reliability demonstration may be reduced.

   c. The period between main engine overhauls may be lengthened.

   d. The probability of Tug recovery after a main engine failure is improved.

   e. The probability of a payload recovery, after either a payload or a Tug failure, is increased.

   f. Additional electrical power is available to payloads during periods when the integrated APS pumps are not in operation.

9. INTEGRATED SYSTEM PRELIMINARY DESIGN SPECIFICATION

9.1 SCOPE

9.1.1 Objective and Applicability. This specification establishes the design and performance requirements for an integrated auxiliary propulsion system (IAPS) for the Space Tug vehicle. It is intended that the data herein serve as the baseline for future design evaluation and system technology development (supporting research and technology) activity.

9.1.2 System Type. The IAPS defined herein is a cryogenic (oxygen/hydrogen) bipropellant auxiliary propulsion system which uses the vehicle main propulsion tanks as a propellant source. The cryogenic system type is restricted to one in which the propellant is supplied to the thrusters in the liquid phase.

9.2 APPLICABLE DOCUMENTS

Johnson Space Center (JSC)

JSC 07700	Space Shuttle Level II Program Definition and requirements:
Vol I	Space Shuttle Program Description and Requirements Baseline
Vol X	Space Shuttle Flight and Ground System Specification
Vol IV	Space Shuttle System Payload Accommodations

Marshall Space Flight Center (MSFC)

MSFC 68M00039	Space Tug Baseline Definition Studies
-1	Baseline Space Tug Requirements and Guidelines
-2	Baseline Space Tug Configuration Definition
-3	Baseline Space Tug Flight Operations
-4	Baseline Space Tug Ground Operations

9.3 REQUIREMENTS

9.3.1 Program Definition.

9.3.1.1 General Description. The IAPS is a candidate auxiliary propulsion subsystem for the Space Tug, a vehicle carried in the Space Shuttle System Orbiter.

The Space Tug is in a preliminary stage of design with a current development schedule (Figure 9-1) calling for an initial operating capability (IOC) date of January 1984. The definition of the Space Tug program and requirements is contained in the MSFC Document series 68M00039, current issue. Additional descriptive information is contained in Section 3.0 of this report.

Excerpts from the MSFC document 68M00039-1 Baseline Space Tug System Requirements and Guidelines are used in the following. The Space Shuttle System definition is contained in the JSC document series 07700 "Space Shuttle Level II Program Definition and Requirements."

The baseline Tug is a cryogenic (oxygen/hydrogen) propulsive stage to be deployed from the Shuttle Orbiter at an initializing orbit, then to deliver and deploy spacecraft into their required orbits, and/or retrieve a spacecraft and to return to a waiting orbit for rendezvous and retrieval by the Orbiter for return to earth for refurbishment and mission recycle.

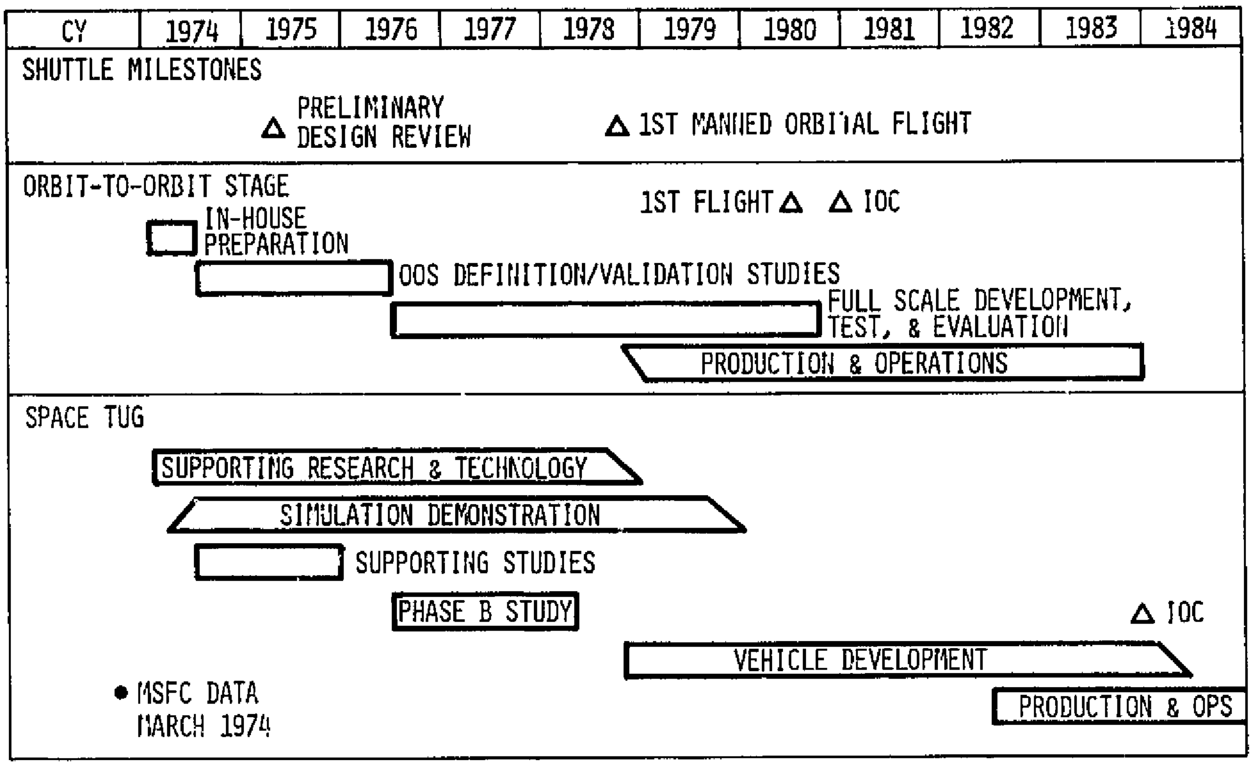

Figure 9-1. Space Tug Preliminary Development Plan

The Tug program is defined to consist of the elements and sub-elements defined in the Baseline Space Tug Work Breakdown Structure, Numbered TBD, dated TBD. This structure shall be utilized in requirements allocations, weights, power requirements, interface identification, and end-item identification.

The detailed description of the baseline Tug configuration except as modified herein, is contained in the MSFC 68M00039-2 Baseline Space Tug Configuration Definition, dated July 15, 1974, or more current issue.

The detailed descriptions of the baseline ground and flight operations plan are contained in:

a. MSFC 68M00039-4 Baseline Space Tug Ground Operations; Verification, Analysis, and Processing, dated July 15, 1974.

b. MSFC 68M00039-3 Baseline Space Tug Flight Operations, dated July 15, 1974.

9.3.1.2 Tug Missions. The Space Tug missions consist of delivery and/or retrieval of DOD, NASA, and other spacecraft identified preliminarily in:

a. DOD Space Mission Model (Secret)

b. The 1973 NASA Payload Model, October 1973 (Being Revised).

The NASA and other missions consist of delivery to and/or retrieval of spacecraft from orbits outside the performance range of the Shuttle Orbiter. Earth escape missions may require expenditure of the Tug. The following, excerpted from Item b above, are typical missions for various spacecraft to be deployed by Tug:

a. Astronomy - Depart from 296 km (160 n mi), 28.5° orbit and transfer to 72227 km (39,000 n mi), 28.5° circular orbit.

b. Atmospheric Physics - Depart from 296 km (160 n mi), 28.5° orbit and transfer to an escape trajectory.

c. Earth Observation - Depart from 185 km (100 n mi), 90° orbit and transfer to 1,667 km (900 n mi), 90° circular orbit.

d. Earth Observation - Depart from 296 km (160 n mi), 28.5° orbit and transfer to 35,786 km (19,323 n mi), 0° circular orbit.

e. Planetary - Depart from 296 km (160 n mi), 28.5° orbit and transfer to an escape trajectory.

f. Communication and Navigation - Depart from 296 km (160 n mi), 28.5° orbit and transfer to 35,786 km (19,323 n mi), 0° circular orbit. Depart from 380 km (205 n mi), 103° orbit and transfer to 1,704 km (920 n mi), 103° orbit.

9.3.1.3 Tug Operational Concept.

9.3.1.3.1 Ground Processing (from Tug acceptance). The present baseline Tug ground processing flow is included here in the most general form.

Tug/Spacecraft Mate and Checkout:

This activity includes the preparations for mating, actual mating of the Tug and spacecraft, and verification of all interfaces.

Tug/Spacecraft Orbiter Mate and Checkout:

This activity includes the physical mating and installation of the Tug/spacecraft in the horizontal position in the Orbiter. Specific activities are as follows:

Prepare Tug/Orbiter interface

Install Tug/spacecraft in Orbiter payload bay

Verify mechanical and electrical interfaces

Perform integrated systems test

Erect in vertical position

Roll out to pad

Launch Operations:

Launch operations are the activities accomplished during the Shuttle launch operations phase and are restricted to those activities that cannot be accomplished earlier. These include:

Installation of flight systems and facilities interface verification

Umbilical and test equipment hookup

Systems verification and operational tests

Fluids and materials servicing

Installation of sensitive items

Countdown and monitoring of systems

Post-Landing Operations:

Post-flight operations include those activities necessary to safe and demate the Tug and the Orbiter. Specific activities are as follows:

Perform safing operations

Remove Tug from Orbiter

Install protective cover and transport Tug to refurbishment area

Refurbishment and Checkout:

Refurbishment encompasses the activities required to service the Tug between each mission, such as:

Performing inspection and checkout to the line-replaceable unit level.

Performing minor structural rework.

Performing optical check for structural alignment.

Performing line-replaceable unit removal and replacement.

Performing cleaning operations.

Storage of the Tug until mission assignment.

Removal from storage and preparation for a mission.

The refurbished Tug systems will be subjected to operational tests to verify their functional operability.

9.3.1.3.2 Tug Flight Operations. The baseline Tug flight operations description is included here in the most general form.

The two Shuttle solid rocket boosters and the Orbiter main engines fire in parallel, providing thrust for liftoff. Following solid rocket booster jettison, the Orbiter main engines continue firing until the vehicle reaches the desired suborbital conditions where the external tank is jettisoned. The Orbiter orbital maneuvering subsystem is then fired to place the Orbiter in the desired Tug/spacecraft initializing orbit. The Tug/spacecraft are deployed from the Orbiter payload bay and the Orbiter moves out to a safe separation distance and relative attitude.

Within orbit phasing requirements, and with all systems enabled and prepared, the Tug acquires the proper vector and the main engine is fired from the ground as necessary to achieve the desired spacecraft orbit or trajectory insertion conditions (or retrieval conditions for a retrieve-only mission). Once the spacecraft is inserted the Tug may be required to verify spacecraft conditions by visual inspection. The Tug, through a series of orbital maneuvers, changes orbit to:

a. Deploy other spacecraft (multi-deployment) and, if required,

b. Retrieve a spacecraft, and then

c. Return to the Orbiter waiting orbit for rendezvous and retrieval.

During the Orbiter rendezvous and retrieval operations, the Tug is configured and safed prior to final docking and stowage into the Orbiter payload bay by the Orbiter. The Orbiter then deorbits and lands for Tug mission recycle.

During flight operations, command and control of the Tug and its spacecraft is maintained from the Tug/Spacecraft operations centers.

9.3.2 System Elements. The IAPS is comprised of the following major subassemblies:

Propellant reservoirs (oxidizer and fuel)

Pumps (oxidizer and fuel)

Accumulators (oxidizer and fuel)

Thruster quad assemblies

Lines and manifolds (oxidizer and fuel)

Insulation and purge system

Controls

Instrumentation

9.3.3 Performance.

9.3.3.1 Functional Performance.

9.3.3.1.1 Functional Description. The IAPS provides the rotational and translational impulses necessary to perform the following maneuvers during Tug flight operations:

Normal Mode

Tipoff disturbance damping

Attitude orientation to align vehicle axes

Steering during APS or MPS ΔV

Attitude stabilization for coast, payload docking, and orbiter retrieval

ΔV for payload docking or vernier adjustment of orbits

Abort Mode

ΔV for vehicle return after any main engine failure not involving excessive loss of MPS propellant

9.3.3.1.2 System Functional Sequence. The IAPS compatibility with the Tug and Orbiter requires the following functional sequence for the system over a mission cycle (reference paragraph 9.3.1.3, Tug Operational Concept):

IAPS Ground Checkout

IAPS Fluid Servicing

Liftoff

Shuttle ascent - insulation venting

Predeployment checkout

Tug erection - Tug/Orbiter disconnects separated

IAPS Activation

Tug deployment (release)

IAPS control of Tug-attitude stabilization only

Tug flight operations - IAPS functional - all modes

Pre-Orbiter docking - IAPS safing and checkout, attitude stabilization only

Manipulator arm contact and IAPS deactivation

Tug-erector mating - Tug/Orbiter disconnects mated

MPS/IAPS purge/inerting

Orbiter descent - MPS/IAPS insulation repressurization

Orbiter landing

IAPS post-landing safing

IAPS refurbishment

9.3.3.1.3 Mission Profiles - Normal Mode. The IAPS functional capability is to meet the flight operations profile of all the planned Tug missions. For preliminary design purposes, the synchronous equatorial triple-payload placement mission is assumed to contain the controlling flight profile for IAPS function and performance levels. The event sequence and mass properties/impulse history of the reference mission flight profile is listed in Table 9-1 and is designated as Mission Profile A. An alternate profile, designated as Mission Profile B is shown in Table 9-2. Profile B is a mission planning option to be exercised when the mission is not payload-limited and the lower payload capability of the B profile is sufficient. Exercise of the B option relieves the main engine of 6 small-velocity maneuvers and results in Tug operation economies and improved mission reliability. The IAPS normal mode functional capability is also to permit substantial variations in the controlling reference mission profiles involving greater or less total impulse and IAPS cycles. This capability is to apply to in-flight profile modifications by Tug control centers in response to mission event opportunities or contingencies as well as to planned profiles needed to accommodate either new, as yet unforeseen, missions or modifications of currently planned missions. This capability is to make maximum use of the integrated aspect of the IAPS--the comingling and interchangeability of MPS and IAPS propellant allocations. It is therefore to be limited only by the total impulse and single-mission operation life requirements specified herein under Paragraph 9.3.3.1.4, Mission Profiles--Abort Mode.

9.3.3.1.4 Mission Profiles - Abort Mode. The IAPS functional capability is to provide for Tug recovery in the event of main engine failure unless: (1) insufficient propellant remains at abort initiation; or (2) main engine failure involves loss of MPS propellant system integrity. The fraction of main engine duty cycle (mission burn time) being covered by IAPS backup is therefore maximized, and is estimated at 0.6 for this preliminary design. A representative profile to be met by the IAPS is shown in Table 9-3. Unscheduled refurbishment is to be accomplished after any abort profiles involving significantly greater than normal-mode impulse.

9.3.3.2 Operability.

9.3.3.2.1 Reliability. A reliability goal of 0.97 has been established for the Tug vehicle for all mission phases. Using that goal, the apportioned numerical reliability goal of the IAPS is 0.996. As noted in section 9.3.3.2.2, Safety, all IAPS elements except primary structure and pressure vessels shall be designed to fail safe in the vicinity of the Shuttle Orbiter. The Tug recovery-to-the-Orbiter success probability (one minus the attrition rate) goal is 0.99, and represents an additional guideline for IAPS reliability.

Table 9-1. IAPS Mission Timeline for Propulsion Events (Mission Profile A)

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0 7	DEPLOY TUG	APS				3	86236	2681	28303	16784	499170
6.38	4	4 38	COAST NO 1							2851	28280	16781	498923
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	162			2	4105	9018	28276	16781	498873
7.74	6	1 19	COAST NO 2							2653	27273	16653	487810
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	2383			2	4363	9048	27271	16653	487791
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)	PIM		14	220856	5	5444	6484	16129	15240	358459
13.13	9	5 6	COAST NO 3							2150	16068	15233	357691
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	1784			5	5441	6583	16063	15232	357630
13.30	11	0 2	ORIENT PAYLOAD	APS				3	33016	1545	10836	14569	286154
13.30	12	0 0	DEPLOY PAYLOAD 1 (826 KG)								10827	14568	286022
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				9	91349	1263	10001	12410	229418
37.16	14	23 45	COAST NO 4							2632	9978	12408	229118
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		83	817394	7	4865	4161	9957	12405	228850
89.34	16	52 9	COAST NO 5							4304	9768	12381	226395
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		83	798157	7	4826	4107	9723	12375	225806
89.41	18	0 2	ORIENT PAYLOAD	APS				3	29062	1211	9538	12352	223381
89.41	19	0 0	DEPLOY PAYLOAD 2 (826 KG)								9530	12351	223281
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				9	79505	889	8705	10193	158320
95.34	21	5 50	COAST NO 6							1282	8684	10191	158096
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		83	712450	8	4577	2892	8679	10190	158037
147.52	23	52 9	COAST NO 7							4674	8514	10169	156192
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		83	695137	8	4537	2850	8468	10163	155686
147.58	25	0 2	ORIENT PAYLOAD	APS				3	25310	844	8307	10143	153876
147.58	26	0 0	DEPLOY PAYLOAD 3 (826 KG)								8300	10142	153802
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				9	68269	465	7474	7985	76658
150.19	28	2 31	COAST NO 8							690	7457	7982	76482
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	1781			8	4271	1492	7455	7982	76458
155.44	30	5 10	COAST NO 9							878	5031	7675	51598
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		12	39600	11	3680	992	5026	7674	51550
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	1690			11	3673	1026	5005	7671	51335
157.63	33	2 5	COAST NO 10							480	3445	7474	35330
157.68	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	758			13	3135	722	3443	7473	35311
163.56	35	5 53	COAST NO 11							973	2908	7405	29827
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	22100	201	2903	7405	29773
163.59	37	0 0	SHUTTLE DEORBIT								2898	7405	29773
164.29	38	0 42	TOUCHDOWN										
TOTALS					8558	358	3303594	134	670230	85406			

MISSION TIME HR	EVENT NO	EVENT DURATION HR MIN	EVENT DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 36	CIRCULARIZE AT 160 NM								65000		
1.75	3	0 7	DEPLOY TUG	APS				10	19387	603	62397	12379	368174
6.38	4	4 38	COAST NO 1							641	62348	12377	367992
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	533			2	4105	2027	62338	12377	367955
7.74	6	1 19	COAST NO 2							596	60126	12283	359795
7.99	7	0 15	TRANSFER ORBIT INSERTION	MAIN	7817			2	4363	2034	60122	12283	359781
8.02	8	0 2	MIDCOURSE CORRECT (DV 1)	PIM		45	49650	5	5444	1458	35559	11241	264389
13.13	9	5 6	COAST NO 3							483	35424	11235	263823
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	5854			5	5441	1480	35414	11235	263778
13.30	11	0 2	ORIENT PAYLOAD	APS				10	7422	347	23889	10746	211060
13.30	12	0 0	DEPLOY PAYLOAD 1 (1820 LB)								23870	10745	210962
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				30	20536	284	22049	9154	169213
37.16	14	23 45	COAST NO 4							592	21998	9151	168991
37.19	15	0 2	PHASING ORBIT INSERT (DV 2)	MAIN		272	183758	7	4865	936	21952	9150	168794
89.34	16	52 9	COAST NO 5							967	21535	9132	166983
89.37	17	0 2	MISSION ORBIT INSERT (DV 3)	MAIN		272	179433	7	4826	923	21436	9128	166548
89.41	18	0 2	ORIENT PAYLOAD	APS				10	6533	272	21028	9110	164766
89.41	19	0 0	DEPLOY PAYLOAD 2 (1820 LB)								21011	9110	164686
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				30	17873	200	19191	7518	116773
95.34	21	5 50	COAST NO 6							288	19146	7516	[illegible]
95.37	22	0 2	PHASING ORBIT INSERT (DV 4)	MAIN		272	160165	8	4577	650	19135	7516	[illegible]
147.52	23	52 9	COAST NO 7							1051	18770	7507	[illegible]
147.55	24	0 2	MISSION ORBIT INSERT (DV 5)	MAIN		272	156273	8	4537	641	18670	[illegible]	[illegible]
147.58	25	0 2	ORIENT PAYLOAD	APS				10	5690	190	18313	[illegible]	[illegible]
147.58	26	0 0	DEPLOY PAYLOAD 3 (1820 LB)								18299	7483	[illegible]
147.67	27	0 5	PAYLOAD SURVEILLANCE	APS				30	15348	105	16478	5889	56541
150.19	28	2 31	COAST NO 8							153	16440	5888	56411
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	5842			8	4271	335	16435	5887	56393
155.44	30	5 10	COAST NO 9							197	11091	5661	38057
155.47	31	0 2	MIDCOURSE CORRECT (DV 6)	THIM		39	13399	11	3680	223	11081	5660	38022
155.54	32	0 4	PHASING ORBIT INSERTION	MAIN	5544			11	3673	231	11035	5658	37863
157.63	33	2 5	COAST NO 10							108	7595	5512	26059
157.69	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2486			13	3135	162	7590	5512	26044
163.56	35	5 53	COAST NO 11							219	6412	5462	22000
163.59	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	4968	45	6400	5462	21960
163.59	37	0 0	SHUTTLE DEORBIT								6388	5462	21960
164.29	38	0 42	TOUCHDOWN										
TOTALS					28076	1172	742673	242	150674	19200			

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Table 9-2. IAPS Mission Timeline for Propulsion Events (Mission Profile B)

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 KM								29483		
1.75	3	0 7	DEPLOY TUG	APS				3	86236	2641	28303	16542	491760
6.38	4	4 38	COAST NO 1							2814	28280	16539	491517
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	162			2	4222	8885	28276	16539	491467
7.75	6	1 19	COAST NO 2							2614	27272	16412	480462
8.00	7	0 15	TRANSFER ORBIT INSERTION	MAIN	2383			2	4469	8916	27271	16411	480443
8.30	8	0 18	MIDCOURSE CORRECT (DV 1)	APS		15	245342			1947	16129	14999	352270
13.12	9	4 50	COAST NO 3							2108	16067	14991	351502
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	1784			5	5483	6472	16063	14990	351445
13.30	11	0 2	ORIENT PAYLOAD	APS				3	33014	1519	10835	14327	281346
13.30	12	0 0	DEPLOY PAYLOAD 1 (793 KG)								10827	14326	281217
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				9	91640	1244	10033	12253	225840
36.14	14	22 44	COAST NO 4							2571	10010	12250	225545
37.19	15	1 3	PHASING ORBIT INSERT (DV 2)	APS		85	843576			1494	9990	12248	225292
88.35	16	51 10	COAST NO 5							4266	9778	12221	222588
89.37	17	1 1	MISSION ORBIT INSERT (DV 3)	APS		85	821928			1468	9734	12215	222020
89.41	18	0 2	ORIENT PAYLOAD	APS				3	29029	1190	9527	12189	219358
89.41	19	0 0	DEPLOY PAYLOAD 2 (793 KG)								9520	12188	219259
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				9	79703	878	8726	10115	156135
94.45	21	4 57	COAST NO 6							1209	8706	10113	155911
95.37	22	0 55	PHASING ORBIT INSERT (DV 4)	APS		85	734769			1143	8702	10112	155862
146.64	23	51 16	COAST NO 7							4631	8517	10089	153816
147.54	24	0 54	MISSION ORBIT INSERT (DV 5)	APS		85	715413			1125	8472	10083	153321
147.58	25	0 2	ORIENT PAYLOAD	APS				3	25267	831	8293	10060	151321
147.58	26	0 0	DEPLOY PAYLOAD 3 (793 KG)								8286	10060	151248
147.66	27	0 5	PAYLOAD SURVEILLANCE	APS				9	68436	466	7493	7987	76845
150.19	28	2 31	COAST NO 8							681	7475	7985	76668
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	1781			8	4240	1495	7473	7984	76644
155.37	30	5 6	COAST NO 9							870	5043	7676	51724
155.47	31	0 6	MIDCOURSE CORRECT (DV 6)	APS		15	76644			344	5039	7676	51677
155.53	32	0 4	PHASING ORBIT INSERTION	MAIN	1690			11	3650	1028	5019	7673	51479
157.62	33	2 5	COAST NO 10							479	3455	7475	35430
157.67	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	758			13	3117	724	3453	7475	35411
163.56	35	5 53	COAST NO 11							974	2916	7407	29911
163.58	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	22163	201	2911	7406	29857
163.58	37	0 0	SHUTTLE DEORBIT								2906	7405	29857
164.28	38	0 42	TOUCHDOWN										
TOTALS					8558	370	7437672	88	547501	70550			

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM								65000		
1.75	3	0 7	DEPLOY TUG	APS				10	19387	594	62397	12201	362709
6.38	4	4 38	COAST NO 1							633	62348	12199	362529
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	533			2	4222	1997	62338	12199	362492
7.75	6	1 19	COAST NO 2							588	60126	12105	354375
8.00	7	0 15	TRANSFER ORBIT INSERTION	MAIN	7817			2	4469	2004	60122	12105	354362
8.30	8	0 18	MIDCOURSE CORRECT (DV 1)	APS		50	55155			438	35559	11363	259824
13.12	9	4 50	COAST NO 3							474	35422	11057	259258
13.26	10	0 8	MISSION ORBIT INSERTION	MAIN	5854			5	5483	1455	35412	11056	259216
13.30	11	0 2	ORIENT PAYLOAD	APS				10	7422	342	23888	10567	207513
13.30	12	0 0	DEPLOY PAYLOAD 1 (1749 LB)								23869	10567	207418
13.41	13	0 7	PAYLOAD SURVEILLANCE	APS				30	20602	280	22120	9038	166574
36.14	14	22 44	COAST NO 4							578	22068	9036	166355
37.19	15	1 3	PHASING ORBIT INSERT (DV 2)	APS		280	189644			336	22025	9034	166169
88.35	16	51 10	COAST NO 5							959	21557	9014	164175
89.37	17	1 1	MISSION ORBIT INSERT (DV 3)	APS		280	184777			330	21459	9010	163756
89.41	18	0 2	ORIENT PAYLOAD	APS				10	6526	267	21004	8990	161792
89.41	19	0 0	DEPLOY PAYLOAD 2 (1749 LB)								20987	8990	161770
89.51	20	0 6	PAYLOAD SURVEILLANCE	APS				30	17918	157	19238	7461	[illegible]
94.45	21	4 57	COAST NO 6							272	19194	7459	114996
95.37	22	0 55	PHASING ORBIT INSERT (DV 4)	APS		280	165183			257	19184	[illegible]	114959
146.64	23	51 16	COAST NO 7							1041	18777	[illegible]	113450
147.54	24	0 54	MISSION ORBIT INSERT (DV 5)	APS		280	160831			253	18678	7437	113085
147.58	25	0 2	ORIENT PAYLOAD	APS				10	5680	187	18282	7420	111610
147.58	26	0 0	DEPLOY PAYLOAD 3 (1749 LB)								18268	7420	111556
147.66	27	0 5	PAYLOAD SURVEILLANCE	APS				30	15305	105	16519	5891	56679
150.19	28	2 31	COAST NO 8							153	16481	5889	56548
150.27	29	0 5	TRANSFER ORBIT INSERTION	MAIN	5842			8	4240	336	16475	5889	56531
155.37	30	5 6	COAST NO 9							195	11119	5662	38150
155.47	31	0 6	MIDCOURSE CORRECT (DV 6)	APS		50	17230			77	11109	5661	38116
155.53	32	0 4	PHASING ORBIT INSERTION	MAIN	5544			11	365[illegible]	231	11066	5660	37970
157.62	33	2 5	COAST NO 10							108	7616	5513	26132
157.67	34	0 3	CIRCULARIZE FOR RENDEZVOUS	MAIN	2486			13	31[illegible]	163	7612	5513	26118
163.56	35	5 53	COAST NO 11							219	6430	5463	22062
163.58	36	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	4982	45	6418	[illegible]	22022
163.58	37	0 0	SHUTTLE DEORBIT								6406	5462	22022
164.28	38	0 42	TOUCHDOWN										
TOTALS					28076	1220	772820	196	123083	15860			

Table 9-3. Abort Profile

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV M/SEC	ORB MANEUVER DV M/SEC	ORB MANEUVER IT N-SEC	APS TRANSLATE DV M/SEC	APS TRANSLATE IT N-SEC	ATT CONT IT N-SEC	START WEIGHT KG	INERTIAS KG-M SQ ROLL	INERTIAS KG-M SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 296 NM								29483		
1.78	3	0 7	DEPLOY TUG	APS				3	86232	3703	28303	23835	686742
6.38	4	4 38	COAST NO 1							3823	28280	23932	686428
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	162			4	7359	12416	28275	23832	686360
7.74	6	1 19	COAST NO 2							3649	27269	23704	672474
7.95	7	0 13	TRANSFER ORBIT INSERTION	MAIN	2193			4	7416	12342	27267	23704	672445
8.45	8	0 30	COAST - MPS FAILURE							2721	16810	22378	502781
9.20	9	0 45	LONGEST APS DELTA-V	APS		62	1036958			2815	16809	22378	502759
9.70	10	0 30	COAST							2693	16544	22344	497420
22.48	11	12 47	TOTAL OF 42 APS BURNS	APS		2732	32552448			4566	16543	22344	497397
140.26	12	118 47	TOTAL OF 42 COAST PERIODS							5648	8232	21290	245180
140.29	13	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				8	61901	1336	8131	21277	240026
140.29	14	0 0	SHUTTLE DEORBIT								8115	21275	239190
140.99	15	0 42	TOUCHDOWN										
TOTALS					2355	2794	33589392	19	213852	56981			

MISSION TIME HR	EVENT NO	DURATION HR MIN	DESCRIPTION	BURN MODE	MAIN ENG DV FT/SEC	ORB MANEUVER DV FT/SEC	ORB MANEUVER IT LB-SEC	APS TRANSLATE DV FT/SEC	APS TRANSLATE IT LB-SEC	ATT CONT IT LB-SEC	START WEIGHT LB	INERTIAS SLUG-FT SQ ROLL	INERTIAS SLUG-FT SQ PITCH
		0 0	LIFTOFF										
	1		SHUTTLE BURNOUT										
1.63	2	1 38	CIRCULARIZE AT 160 NM								65000		
1.78	3	0 7	DEPLOY TUG	APS				10	19396	833	62397	17580	506522
6.38	4	4 38	COAST NO 1							859	62346	17579	506290
6.43	5	0 3	PHASING ORBIT INSERTION	MAIN	532			4	7359	2791	62335	17578	506240
7.74	6	1 19	COAST NO 2							820	60118	17483	495998
7.95	7	0 13	TRANSFER ORBIT INSERTION	MAIN	7196			4	7416	2775	60113	17483	495977
8.45	8	0 30	COAST - MPS FAILURE							612	37061	16505	370838
9.20	9	0 45	LONGEST APS DELTA-V	APS		204	233118			633	37058	16505	370821
9.70	10	0 30	COAST							605	36473	16480	365983
22.49	11	12 47	TOTAL OF 42 APS BURNS	APS		8963	7318083			1026	36470	16480	366866
140.24	12	119 47	TOTAL OF 42 COAST PERIODS							1270	18149	15703	180839
140.29	13	0 2	SHUTTLE RENDEZVOUS AND DOCK	APS				25	13916	300	17927	15693	177036
140.29	14	0 0	SHUTTLE DEORBIT								17991	15692	176420
140.99	15	0 42	TOUCHDOWN										
TOTALS					7729	9167	7551201	43	48076	12810			

9.3.3.2.2 Safety. IAPS performance during post-deployment and pre-retrieval operations within TBD Orbiter/Tug separation distance is Orbiter crew-safety critical. Within these intervals:

a. Provision shall be made for IAPS control by the Orbiter crew.

b. No single IAPS failure shall result in unprogrammed motion of the Tug (fail safe).

c. As a minimum, the IAPS shall be designed to sustain a failure and retain the capability to hold attitude and position without damaging the Orbiter of injuring flight personnel of the Orbiter. Critical failure indicators/status signals shall be provided to the Orbiter crew.

d. IAPS pressure levels and cryogen heating and/or venting levels shall be held to minimum values compatible with necessary IAPS operations. The capability for system command venting to minimum levels pre- and post-Tug flight operations is to be provided for this purpose.

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IAPS functions while stowed aboard the Orbiter shall comply with the following:

a. All propellant venting will be through Tug/Orbiter umbilicals.

b. Prior to descent, IAPS fluid quantities shall be reduced to a safe value consistent with dumping/venting/helium purge requirements for the MPS. The interconnection of the IAPS/MPS shall permit common propellant purge/venting. Minimum IAPS pressure levels sufficient to prevent pressure vessel implosion shall be established.

9.3.3.2.3 Maintainability. Refurbishment of the IAPS by replacement and overhaul of subassemblies/components is planned to permit attainment of required useful life in the most economical manner while maintaining system reliability above the apportioned goal.

9.3.3.2.4 Useful Life. The IAPS design and maintenance/refurbishment plan is to comply with a useful life corresponding to 20 missions (design) reference Mission Profile B) over a ten-year period.

The operating life required of the system elements is that which is necessary to accomplish the normal missions between scheduled refurbishments. The design life is the total of the normal mission operating life plus one aborted mission involving IAPS maximum recovery capability plus margin.

9.3.3.2.5 Environments. Natural and induced environments are as specified in MSFC 68M00039-1 Baseline Space Tug System Requirements and Guidelines. Cargo bay door-open pressure levels apply to the IAPS (and MPS) insulation venting back pressure.

9.3.3.3 Performance Allocations.

9.3.3.3.1 Performance Trades. The nominal, preliminary design performance allocations for IAPS elements are given in paragraph 9.3.5, Design. Allocation for other functionally interfacing subsystems are given in paragraph 9.3.3.3.2 Functional Interfaces. All of these allocations are tradeable during development with a coefficient of -2.9 kg of payload per kg of burned weight (-2.9 lb of payload per lb of burnout weight). Where useful, unit performance parameter trade coefficients are shown in terms of payload or burnout weight in the section specifying unit performance.

9.3.3.3.2 Functional Interfaces.

9.3.3.3.2.1 Main Propulsion System.

(1) The MPS interfaces with the IAPS by providing a propellant source in the Tug main tanks. During normal IAPS modes propellant is withdrawn at the normal mode mixture ratio and is within the condition limits of Table 5-18 of this report. During abort mode operation, MPS autogenous

pressurization is not available and propellant supplied is saturated (after initial blowdown expulsion). The abort timeline shall permit tank heat leak sufficient to maintain a minimum propellant pressure of 10.3 N/cm^2 (15 psia).

(2) MPS and APS propellant loaded quantities are comingled and are to be treated as such in flight contingency allocations by root sum square combination. Flight performance reserves are 2 percent of the planned mission ΔV for the MPS and 10 percent of the planned mission total impulse for the IAPS except for any ΔV covered by the MPS reserve allocation. The MPS mixture ratio loaded bias is assumed at 29.5 kg (65 lb) of fuel). The IAPS mixture ratio is controllable and loaded bias is TBD.

(3) The IAPS is to provide a two- or four-thruster propellant-settling maneuver equivalent to 4 free falls prior to each main engine rotating start. The acceleration by the main engine THIM operation necessary for engine chill shall be included in the computation.

(4) The reference MPS engine for this IAPS design is the Pratt and Whitney Category IIB RL-10 with a 2/15 (oxidizer/fuel) minimum NPSH capability at full thrust.

(5) The existance of IAPS main engine backup capability requires that main engine design goals include a fail-safe objective of minimizing MPS propellant loss in order to maximize Tug recovery probability.

9.3.3.3.2.2 Electrical Power System. The Tug electrical power system shall provide 0.85 kw of conditioned power to the IAPS pump drives and TBD power to IAPS controls and heater at all times during Tug flight operations. Reactants shall be supplied to the EPS fuel cells from the MPS tanks on demand during any phase of flight operations. For the purposes of this preliminary design, the baseline vehicle's main tank zero-g device feedout at low pressure is incorporated. Tapoff at high pressure from the IAPS is a beneficial alternate that could be implemented. The EPS power conditioning system for the IAPS shall include variable frequency inverters in conformance with the speed variation requirements of the IAPS pump drives. Using the current baseline EPS, each of the redundant fuel cells must be increased in rating from 1.75 kw to 2.60 kw to meet this requirement.

9.3.3.3.2.3 Stabilization and Control System. All control signals to the IAPS shall be provided by the Tug stabilization and control system (SCS). Ground control also shall be through the SCS during flight operations. The SCS shall provide for semi-autonomous IAPS operation by furnishing status data, thruster logic, sequencing logic, and feedback control loop electronics.

9.3.4 Design and Construction Standards. The IAPS shall conform with Tug system specifications on design and construction standards.

9.3.4.1 Factors of Safety. The factors of safety for pressure vessels are 2.0 ultimate and 1.5 proof with respect to limit pressure.

9.3.4.2 Fluid Seals. To minimize leak hazards in the Orbiter and loss of fluid during flight operations, all-welded or brazed connections will be used where practicable. Breakable joints shall be minimized to locations where disassembly requirements preclude application of chipless cutter/weld stub technique and in-situ welding.

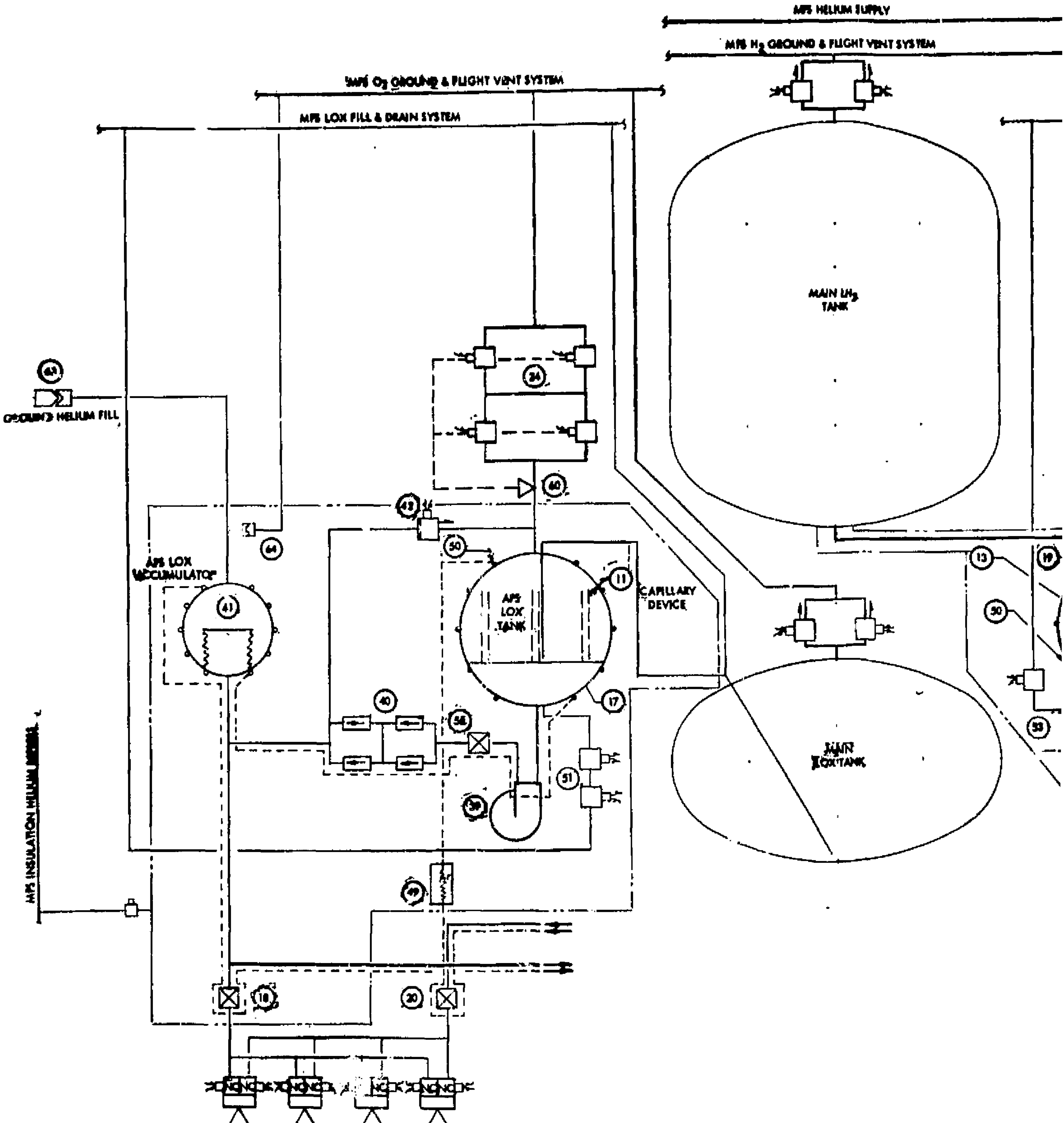
9.3.5 Design.

9.3.5.1 System Characteristics.

9.3.5.1.1 General Arrangement and System Properties. The interconnection of system fluid elements and components is defined on Figure 9-2, IAPS mechanical schematic. The nominal function of system fluid elements is as defined by the fluid state and mass/energy balance data on the IAPS process diagram of Figure 9-3. Preliminary design of the physical/geometric relationships of the Space Tug and IAPS elements is as shown on Figure 9-4. The IAPS detailed weight statement and reference Tug stage weight statements for this preliminary design are in Tables 9-4, 9-5, and 9-6.

9.3.5.1.2 Operation. The IAPS utilizes capillary reservoirs as the source of propellant during all flight phases except sustained velocity maneuvers. The reservoirs are refilled on demand from the MPS tanks by venting their vapor contents to space. Refill is controlled to occur only when main tanks are settled during either IAPS or MPS velocity maneuvers. Reservoir capacity is to be sufficient for the controlling coast period between scheduled velocity maneuvers of the reference mission but may be refilled as necessary by unscheduled velocity maneuvers. During sustained velocity maneuvers propellant flows through the reservoirs and they are not depleted.

Each reservoir (oxidizer and fuel) contains capillary screens and collector tubes to provide vapor-free propellant to their thruster feed pumps. An accumulator is provided downstream of each pump to minimize pump cycles. The entire system is insulated with multilayer insulation (MLI) to provide a radiation barrier and minimize heat leaks to the cold feedlines, pumps, and tanks. The system heat loads are absorbed by hydrogen bleed flow which is tapped off upstream of the pump and expanded through a Joule-Thompson expander to a pressure of 7.9 N/cm^2 (11.4 psia) and a temperature of 35 K (63 R). This cold hydrogen bleed is first routed through cooling coils mounted on the outside of the fuel zero-g reservoir. The bleed then traces the hydrogen feed line manifold, absorbing heat through saddleblock segments brazed between the chill line and feed line. After leaving the hydrogen system, the bleed is electrically heated above the freezing temperature of oxygen and routed along the oxygen system in a manner similar to that for the hydrogen system, after which it is vented overboard through the MPS vent system.



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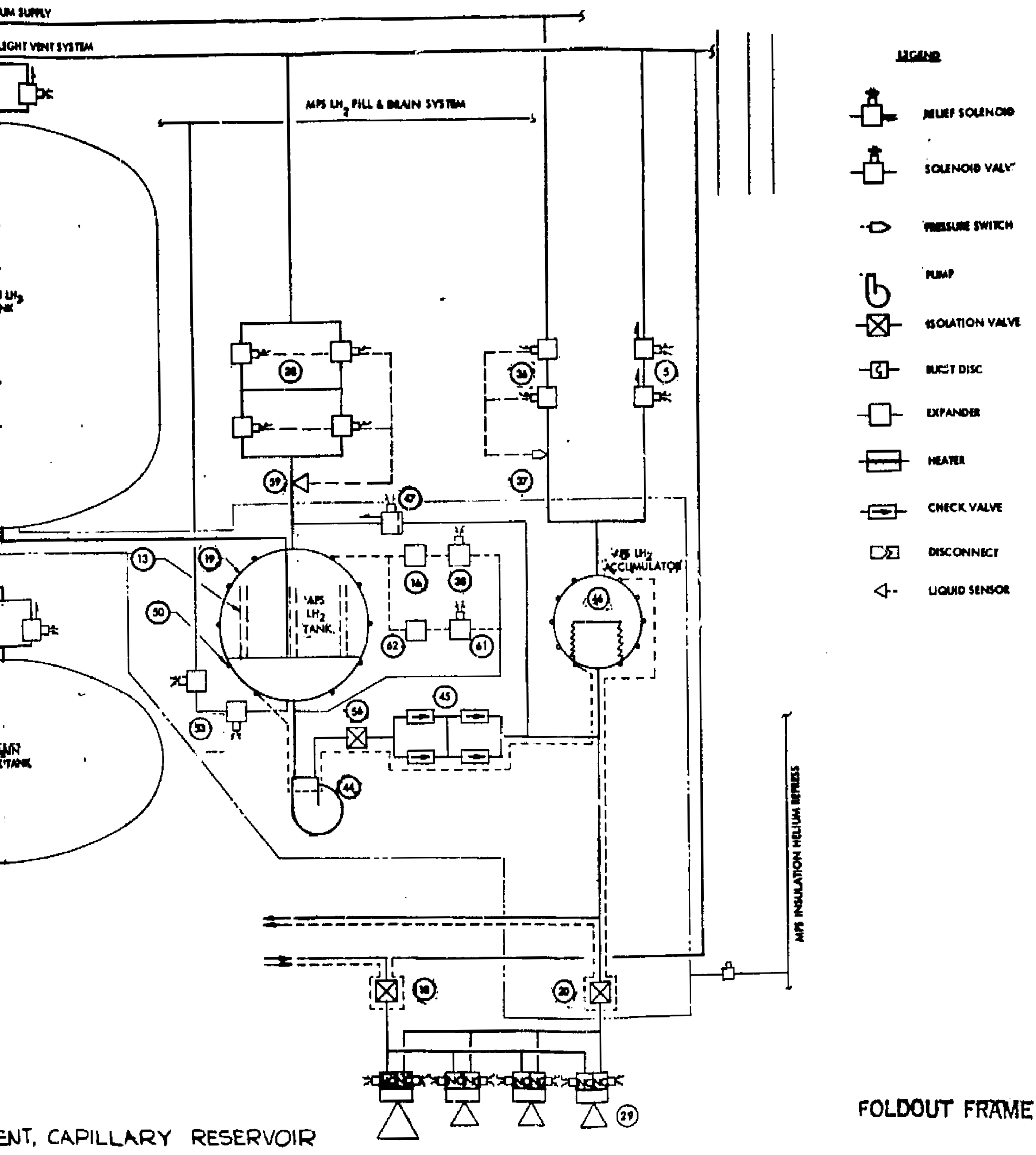

ENT, CAPILLARY RESERVOIR

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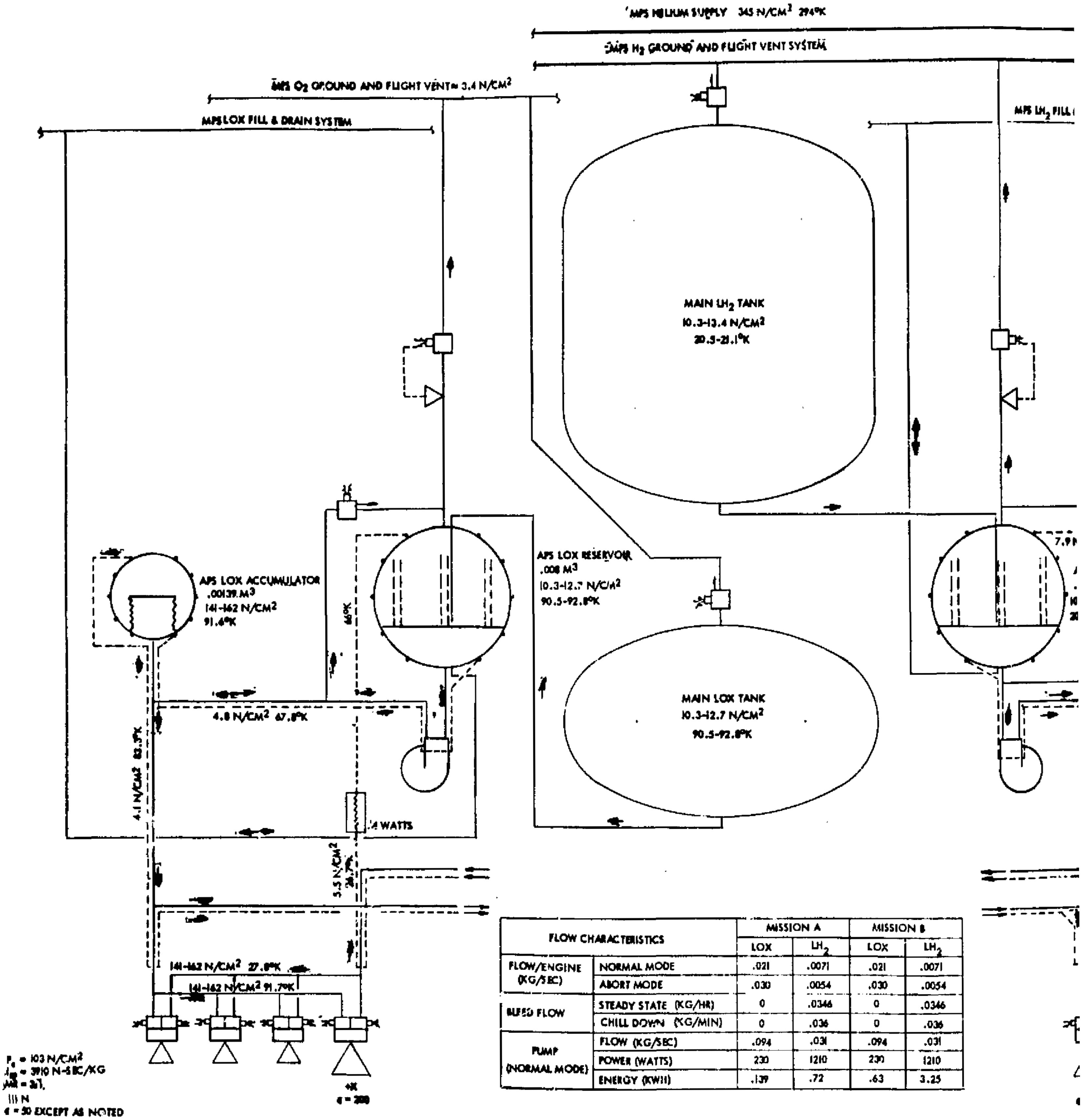
Figure 9-2. IAPS Mechanical Flow Diagram



FLOW CHARACTERISTICS		MISSION A		MISSION B	
		LOX	LH_2	LOX	LH_2
FLOW/ENGINE (KG/SEC)	NORMAL MODE	.021	.0071	.021	.0071
	ABORT MODE	.030	.0054	.030	.0054
BLEED FLOW	STEADY STATE (KG/HR)	0	.0346	0	.0346
	CHILL DOWN (KG/MIN)	0	.036	0	.036
PUMP (NORMAL MODE)	FLOW (KG/SEC)	.094	.031	.094	.031
	POWER (WATTS)	230	1210	230	1210
	ENERGY (KWH)	.139	.72	.63	3.25

Figure 9-3. Process Diagram, I

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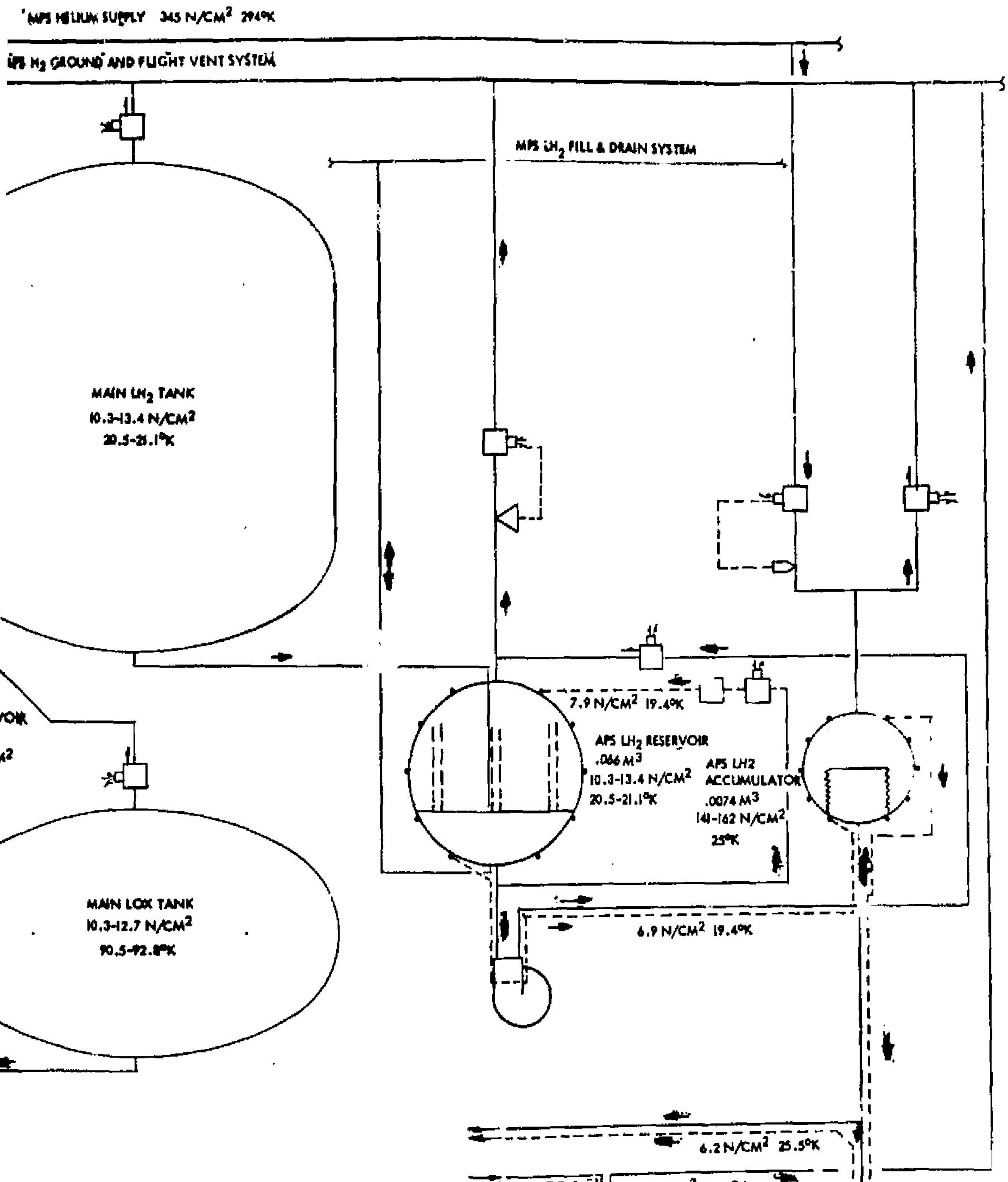

CTERISTICS	MISSION A		MISSION B	
	LOX	LH_2	LOX	LH_2
ORMAL MODE	.021	.0071	.021	.0071
ORT MODE	.030	.0054	.030	.0054
EADY STATE (KG/HR)	0	.0346	0	.0346
ILL DOWN (KG/MIN)	0	.036	0	.036
OW (KG/SEC)	.094	.031	.094	.031
WER (WATTS)	230	1210	230	1210
ERGY (KWH)	.139	.72	.63	3.25

Figure 9-3. Process Diagram, IAPS (Metric Units)

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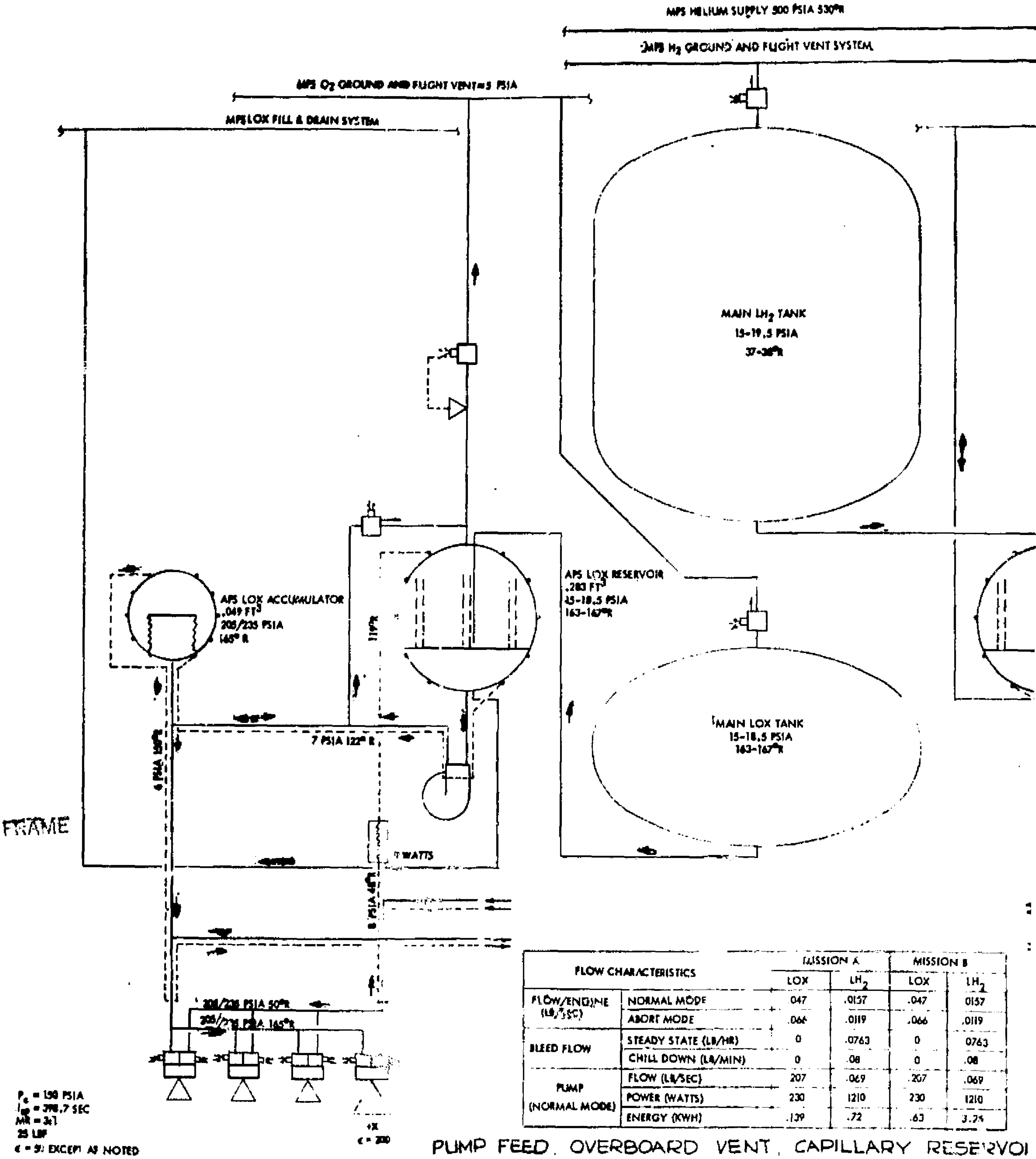
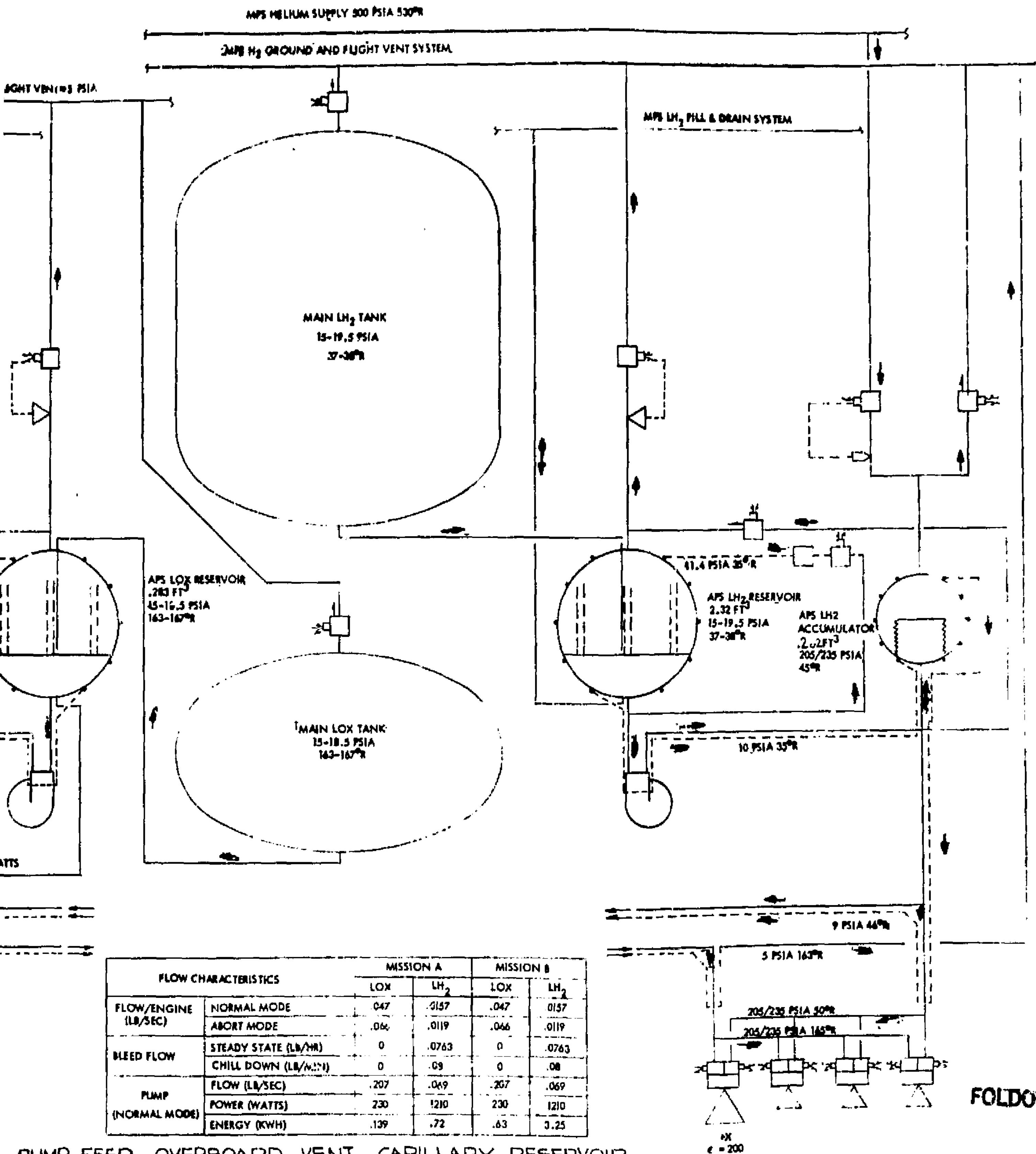


FLOW CHARACTERISTICS		MISSION A LOX	MISSION A LH_2	MISSION B LOX	MISSION B LH_2
FLOW/ENGINE (LB/SEC)	NORMAL MODE	.047	.0157	.047	.0157
	ABORT MODE	.066	.0119	.066	.0119
BLEED FLOW	STEADY STATE (LB/HR)	0	.0763	0	.0763
	CHILL DOWN (LB/MIN)	0	.08	0	.08
PUMP (NORMAL MODE)	FLOW (LB/SEC)	.207	.069	.207	.069
	POWER (WATTS)	230	1210	230	1210
	ENERGY (KWH)	.139	.72	.63	3.75

PUMP FEED, OVERBOARD VENT, CAPILLARY RESERVOI

Figure 9-3. Process Di



FLOW CHARACTERISTICS		MISSION A		MISSION B	
		LOX	LH_2	LOX	LH_2
FLOW/ENGINE (LB/SEC)	NORMAL MODE	.047	.0157	.047	.0157
	ABORT MODE	.066	.0119	.066	.0119
BLEED FLOW	STEADY STATE (LB/HR)	0	.0763	0	.0763
	CHILL DOWN (LB/MIN)	0	.09	0	.08
PUMP (NORMAL MODE)	FLOW (LB/SEC)	.207	.069	.207	.069
	POWER (WATTS)	230	1210	230	1210
	ENERGY (KWH)	.139	.72	.63	3.25

PUMP FEED, OVERBOARD VENT, CAPILLARY RESERVOIR

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Figure 9-3. Process Diagram, IAPS (English Units)

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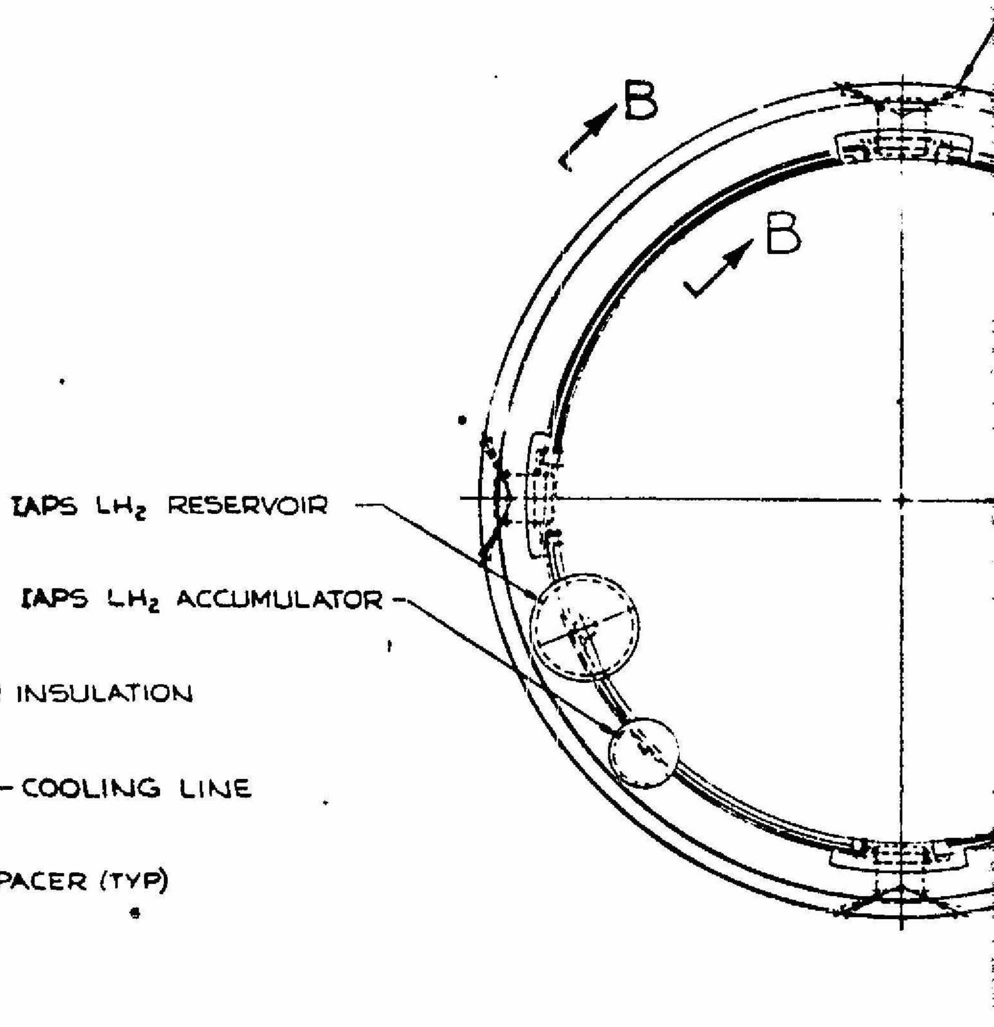

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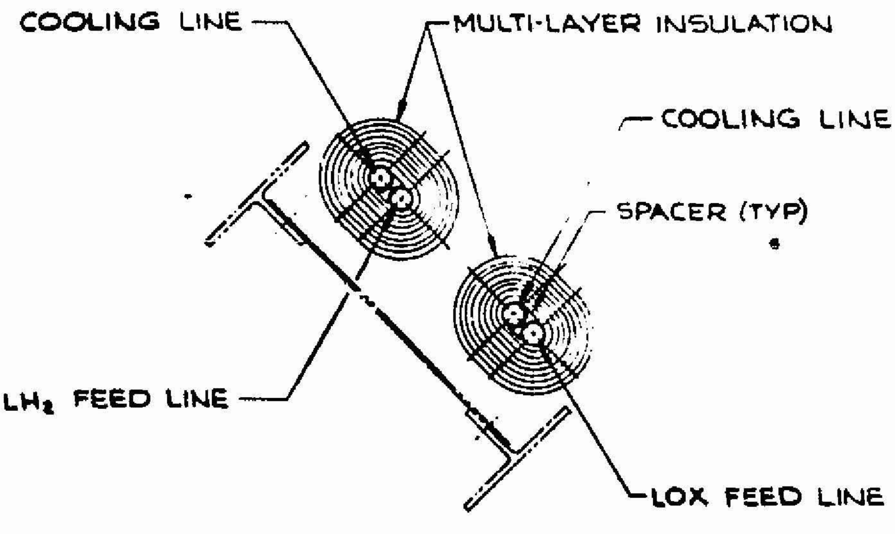

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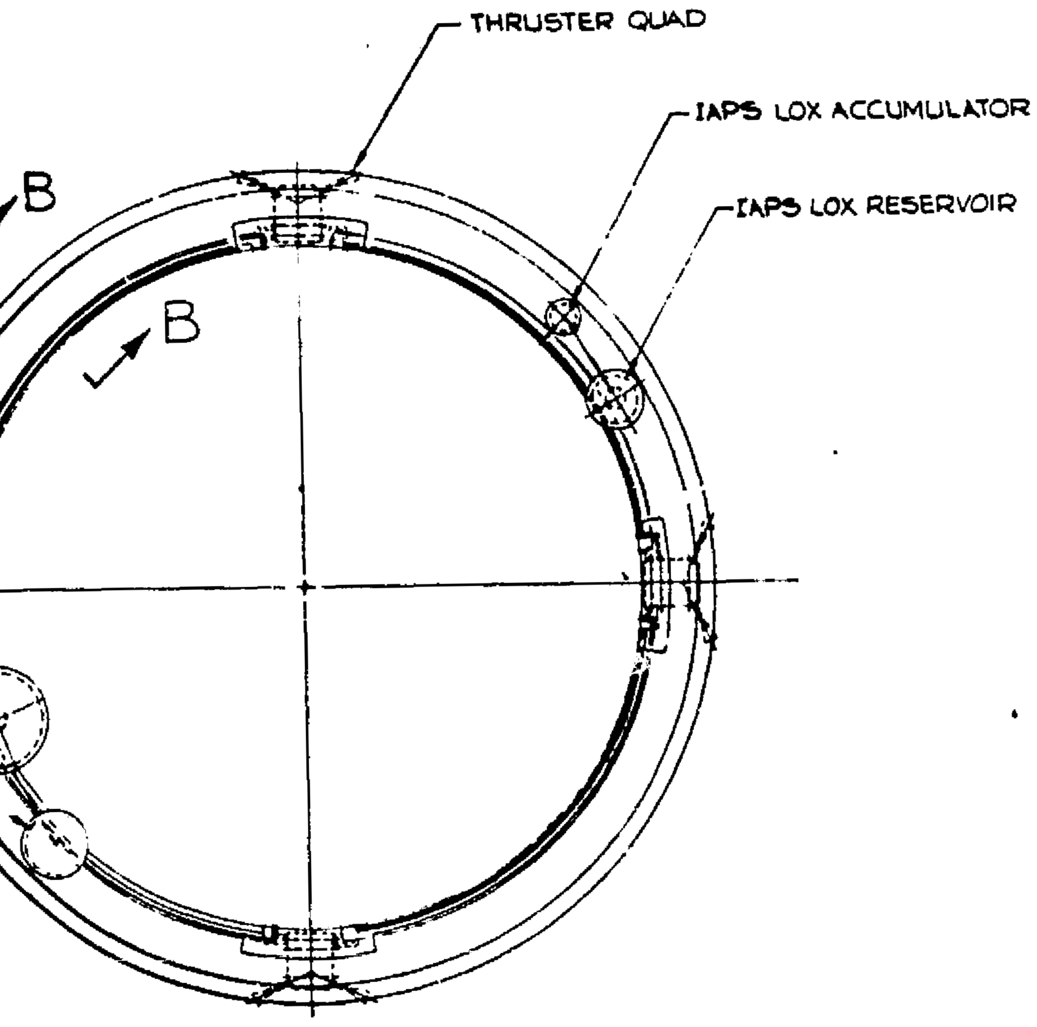

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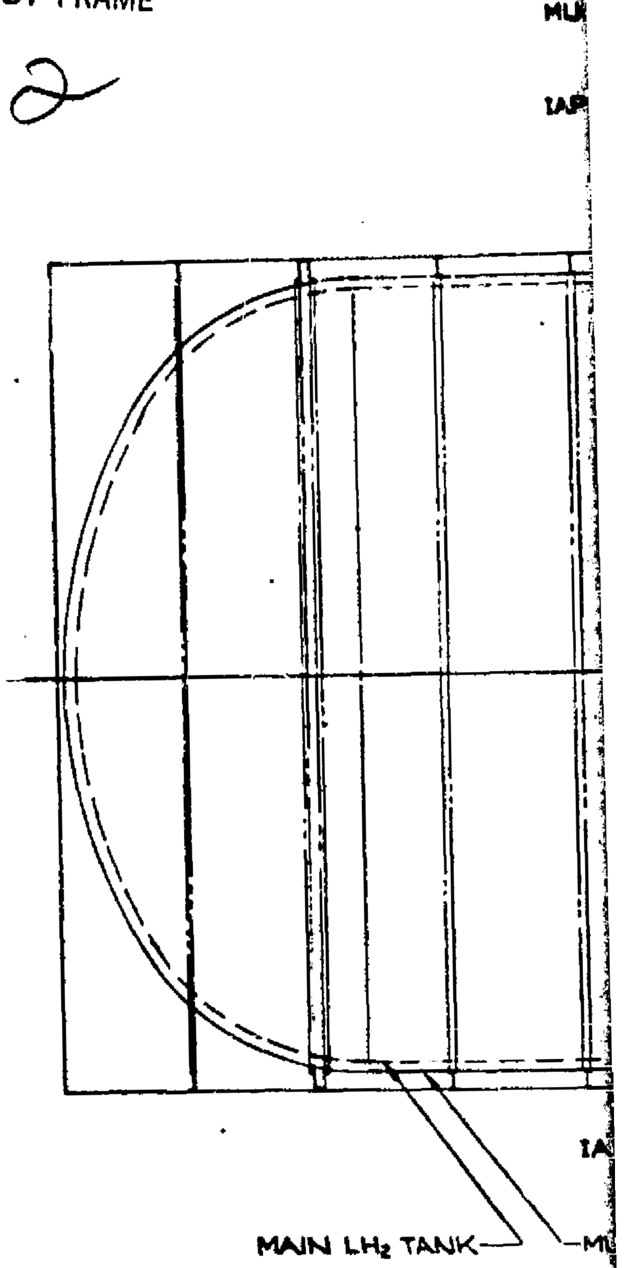

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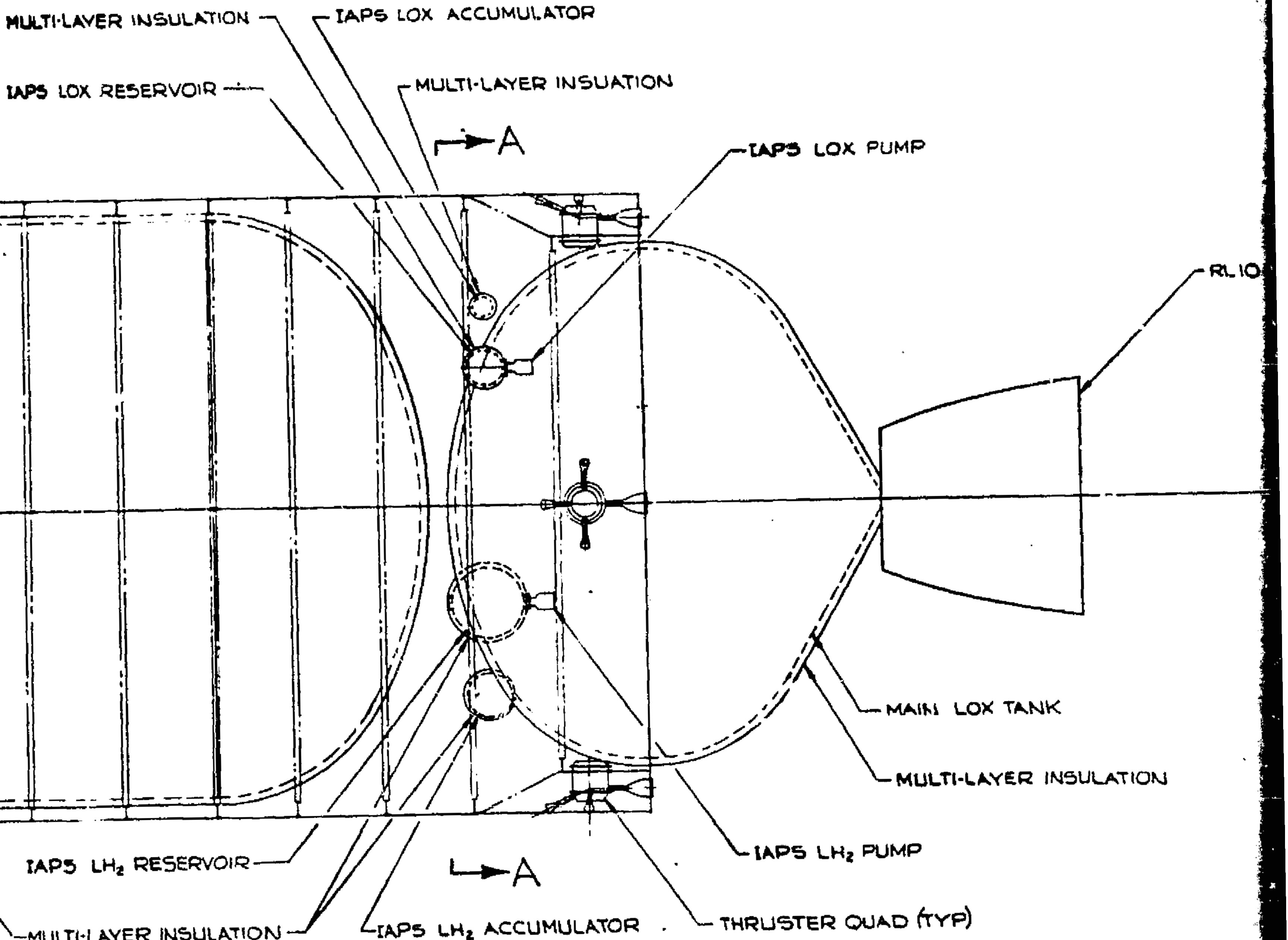

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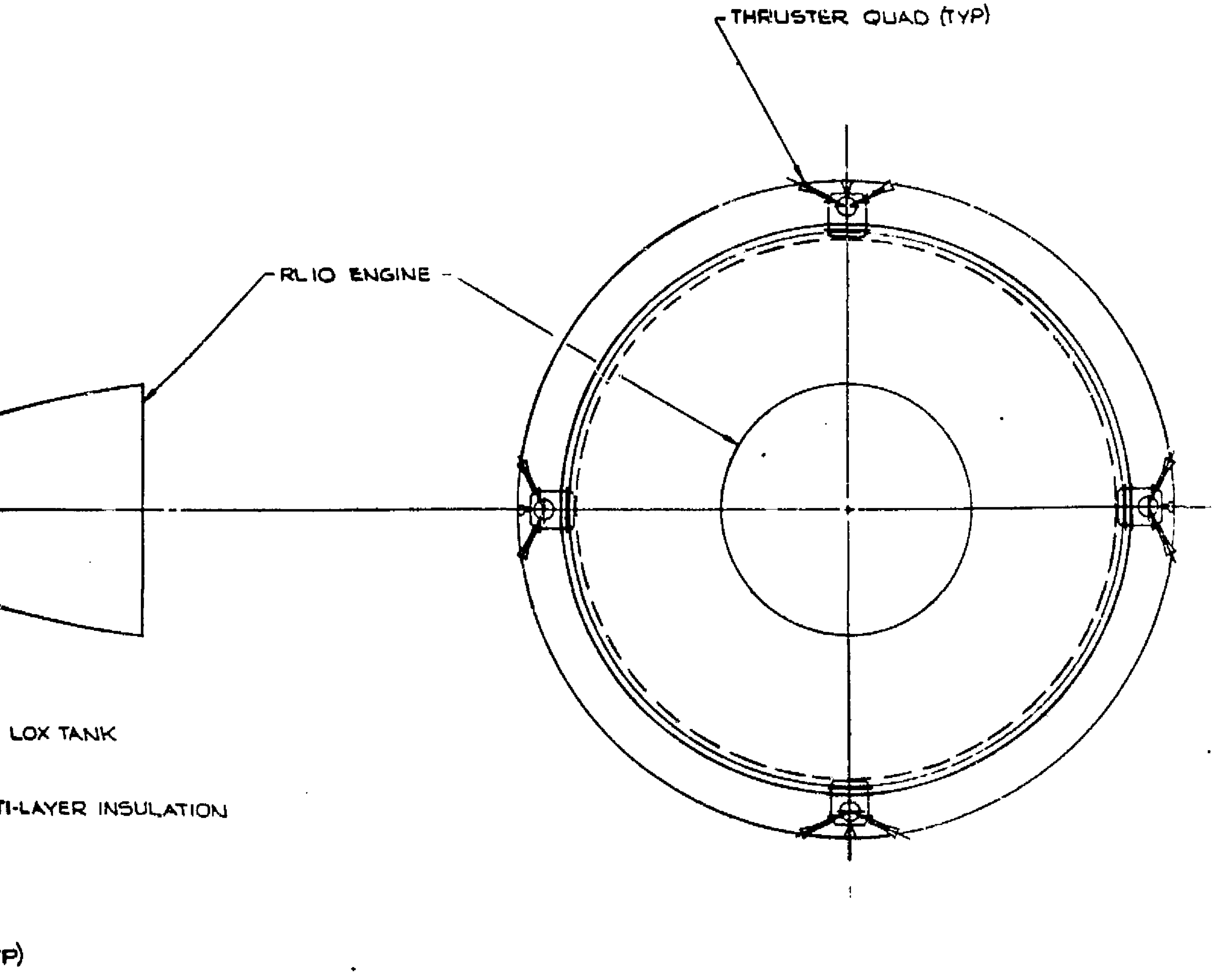

Figure 9-4. IAPS Installation, Space Tug

Table 9-4. IAPS Component Weight Statement

	ID NO	QTY PER VEHICLE	ITEM WEIGHT LB	ITEM WEIGHT KG	SYSTEM WEIGHT LB	SYSTEM WEIGHT KG
FILL AND DRAIN SYSTEM		(6)			(7.[illegible])	([illegible])
LOX DRAIN VALVE	51	2	1.5	0.7	[illegible]	[illegible]
LH2 DRAIN VALVE	52	2	1.5	0.7	[illegible]	[illegible]
He FILL DISC - LOX	63	1	1.0	0.5	1.[illegible]	[illegible]
He [illegible] DISC - LOX	64	1	0.5	0.[illegible]	[illegible]	[illegible]
PRESSURIZATION SYSTEM		(4)			([illegible])	([illegible])
HELIUM ISO VALVE - LH2	36	2	1.2	0.[illegible]	2.[illegible]	1.[illegible]
PRESSURE SWITCH - LH2	37	2	1.0	0.5	2.[illegible]	[illegible]
PROPELLANT CONTROL SYSTEM		(9)			(1[illegible])	([illegible])
LOX TANK CAPILLARY DEV	11	1	1.5	0.7	1.[illegible]	0.7
LH2 TANK CAPILLARY DEV	13	1	2.1	1.0	2.[illegible]	1.0
LH2 LH BLEED SHUTOFF VLV	[illegible]	1	0.8	0.4	[illegible]	[illegible]
LH2 LH BLEED EXPANDER	16	1	0.8	0.4	[illegible]	0.4
LH2 HE BLEED SHUTOFF VLV	61	1	0.8	0.4	[illegible]	0.4
LH2 HE BLEED EXPANDER	62	1	0.8	0.4	0.[illegible]	0.4
LH2 BLEED HEATER	49	1	0.5	0.2	[illegible]	0.[illegible]
LOX EXT COOLING COILS	53	1	0.6	0.3	[illegible]	0.[illegible]
LH2 EXT COOLING COILS	50	1	2.3	1.3	[illegible]	1.[illegible]
PROPELLANT FEED SYSTEM		(28)			([illegible])	([illegible])
LOX RESERVOIR	17	1	2.4	1.1	2.[illegible]	1.[illegible]
LOX RES INSULATION	0	1	0.5	0.2	[illegible]	0.[illegible]
LH2 RESERVOIR	19	1	8.4	3.8	[illegible]	3.[illegible]
LH2 RES INSULATION	0	1	0.5	0.2	0.[illegible]	0.2
LOX SYSTEM ISO VALVE	55	1	0.8	0.4	[illegible]	0.[illegible]
LH2 SYSTEM ISO VALVE	56	1	0.8	0.4	[illegible]	0.[illegible]
LOX QUAD ISOLATION VLV	18	4	1.0	0.5	4.0	1.[illegible]
LH2 QUAD ISOLATION VLV	20	4	1.0	0.[illegible]	4.0	1.[illegible]
LOX PUMP & DRIVE	39	1	7.9	3.6	7.[illegible]	3.[illegible]
LH2 PUMP & DRIVE	44	1	10.5	4.3	10.[illegible]	4.[illegible]
LOX PUMP CHECK VALVE	43	4	0.3	0.1	1.2	0.5
LH2 PUMP CHECK VALVE	45	4	0.3	0.1	1.2	0.5
LOX ACCUMULATOR/BELLOWS	41	1	1.5	0.7	1.[illegible]	0.7
LH2 ACCUMULATOR/BELLOWS	46	1	4.9	2.2	4.[illegible]	2.2
LOX RELIEF SOLENOID	42	1	0.8	0.4	0.[illegible]	0.4
LH2 RELIEF SOLENOID	47	1	0.8	0.4	0.[illegible]	0.4
OVERBOARD VENT		(12)			(8.[illegible])	(3.6)
LH2-HELIUM VENT VALVE	5	2	0.5	0.2	1.[illegible]	0.5
LOX REFILL VENT SOL	24	4	0.8	0.4	3.2	1.5
LH2 REFILL VENT SOL	28	4	0.8	0.4	3.2	1.5
LOX REFILL VENT PT SENS	60	1	0.3	0.1	0.[illegible]	0.1
LH2 REFILL VENT PT SENS	59	1	0.3	0.1	0.[illegible]	0.1
THRUSTER QUAD		(68)			(74.[illegible])	(33.6)
THRUST CHAMBER (AR=50)	29	12	2.6	1.2	31.2	14.[illegible]
THRUST CHAMBER (AR=200)	29	4	3.2	1.5	12.8	5.8
THRUSTER VALVES	0	32	0.6	0.3	19.2	8.7
IGNITER	0	16	0.2	0.1	3.2	1.5
EXCITER	0	4	1.9	0.9	7.[illegible]	3.4
INSTRUMENTATION	0	(43)	0.4	0.2	(17.2)	(7.8)
PUMP PWR SUPP - APS CHARGE		(4)			(54.7)	(24.8)
INVERTER	0	1	11.3	5.1	11.3	5.1
FUEL CELL	0	2	16.2	7.3	32.4	14.7
FUEL CELL RADIATOR	0	1	11.0	5.0	11.0	5.0
COMPONENT TOTAL		174			226.[illegible]	102.6
LINES					27.3	12.4
INSULATION					8.2	3.7
COMPONENT MOUNTINGS					11.[illegible]	5.[illegible]
DRY SYSTEM					273.4	124.0

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Table 9-5. IAPS Stage Weight Statement (Mission A)

DESCRIPTION	WEIGHT	
	LB	KG
STRUCTURE	(2094.)	(950.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1395.)	(633.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	273.	124.
DRY WEIGHT	4895.	2220.
CONTINGENCY (13%)	636.	288.
TOTAL DRY SYSTEM	5531.	2509.
NONUSUABLE FLUIDS	(535.)	(243.)
APS TRAPPED PROPELLANT	10.	5.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	371.	168.
FLIGHT RESERVES	(321.)	(146.)
APS RESERVE	3.	1.
MPS RESERVE	318.	144.
BURNOUT WEIGHT	6387.	2897.
EXPENDED FLUIDS	(50549.)	(22929.)
APS USUABLE PROPELLANT	416.	189.
APS LH2 BLEED OVERBOARD	13.	6.
MPS USABLE PROPELLANT	49833.	22604.
MPS BOILOFF VENTED	150.	68.
FUEL CELL REACTANTS	137.	62.
PAYLOAD	(5461.)	(2477.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

Table 9-6. IAPS Stage Weight Statement (Mission B)

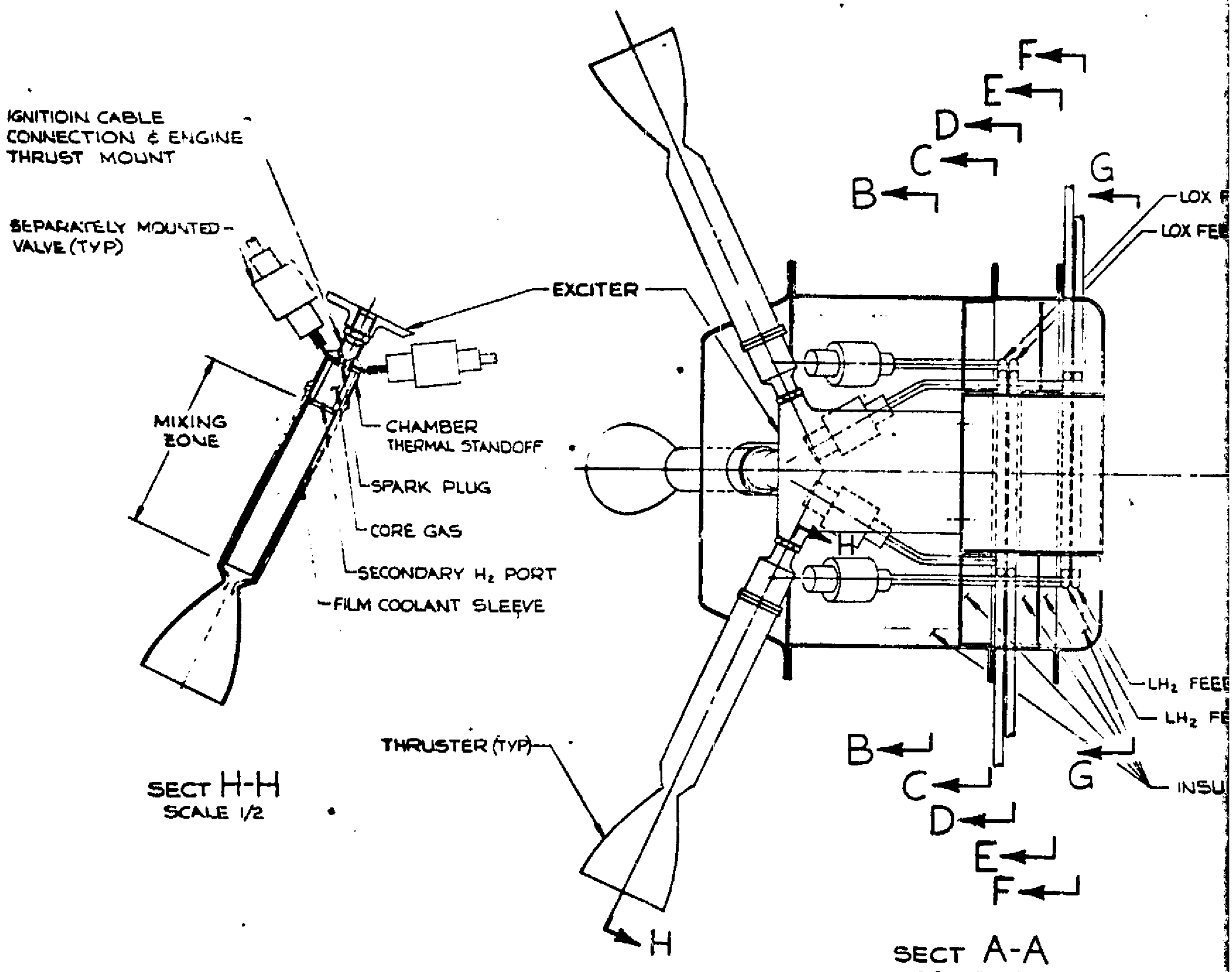
DESCRIPTION	WEIGHT	
	LB	KG
STRUCTURE	(2114.)	(959.)
THERMAL CONTROL	(394.)	(179.)
ASTRIONICS	(1012.)	(459.)
PROPULSION	(1395.)	(633.)
MAIN PROPULSION	1122.	509.
AUXILIARY PROPULSION	273.	124.
DRY WEIGHT	4915.	2229.
CONTINGENCY (13%)	639.	290.
TOTAL DRY SYSTEM	5554.	2519.
NONUSUABLE FLUIDS	(540.)	(245.)
APS TRAPPED PROPELLANT	10.	5.
MPS TRAPPED PROPELLANT	154.	70.
MPS PRESSURANT	376.	171.
FLIGHT RESERVES	(312.)	(142.)
APS RESERVE	4.	2.
MPS RESERVE	308.	140.
BURNOUT WEIGHT	6406.	2906.
EXPENDED FLUIDS	(50744.)	(23017.)
APS USUABLE PROPELLANT	2234.	1013.
APS LH2 BLEED OVERBOARD	13.	6.
MPS USABLE PROPELLANT	48210.	21868.
MPS BOILOFF VENTED	150.	66.
FUEL CELL REACTANTS	137.	62.
PAYLOAD	(5247.)	(2380.)
GROSS WEIGHT AT ORBITER SEP	62397.	28303.
TUG CHARGEABLE INTERFACES	(2603.)	(1181.)
GROSS LIFTOFF WEIGHT	65000.	29483.

The reservoirs are loaded prior to launch with the remainder of the downstream system isolated and filled with ambient temperature propellant vapor. Initial chilldown of the isolated section requires the use of a higher flow rate bypass expander. The oxygen accumulator is pressurized with helium on the ground at ambient temperature and a pressure high enough to result in operating pressure after chilldown to LOX operating temperature. The hydrogen accumulator is pressurized from the MPS helium system during on-orbit IAPS activation. An MLI purge bag surrounds that portion of the system which is loaded with liquid propellant.

9.3.5.2 Thruster.

9.3.5.2.1 General Requirements and Characteristics. The IAPS thruster is a liquid-liquid oxygen/hydrogen type with nominal preliminary design characteristics as described in Table 9-7.

The preliminary design of the thruster is shown in Figure 9-5. The thruster consists of the thrust chamber assembly, igniter and two propellant valves. The thruster installation in quad arrays, also is shown in Figure 9-5. The thruster quad consists of three 50 area ratio thrusters, one 200 area ratio thruster, an exciter-control unit, manifolding, insulation, and housing.



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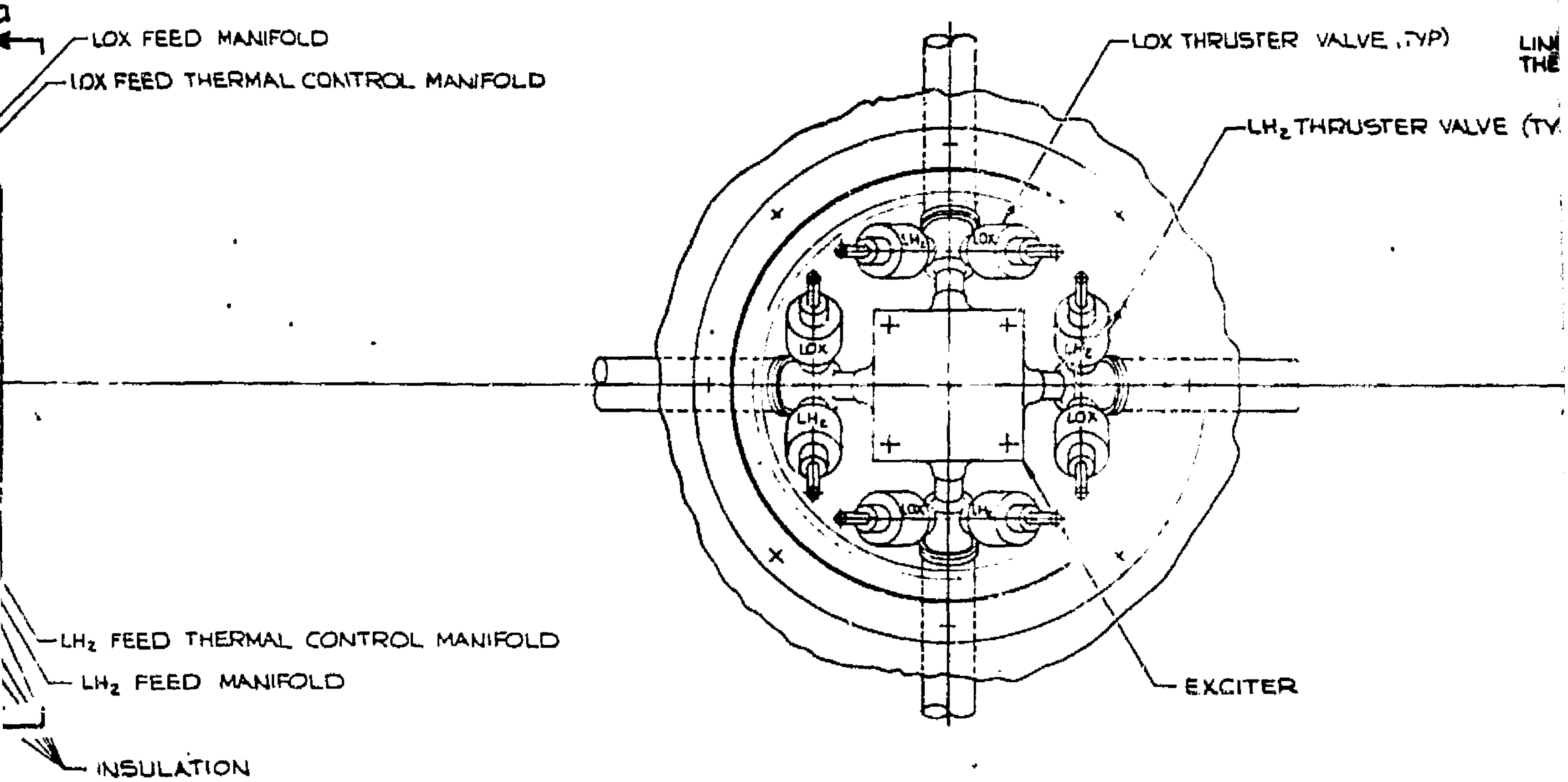

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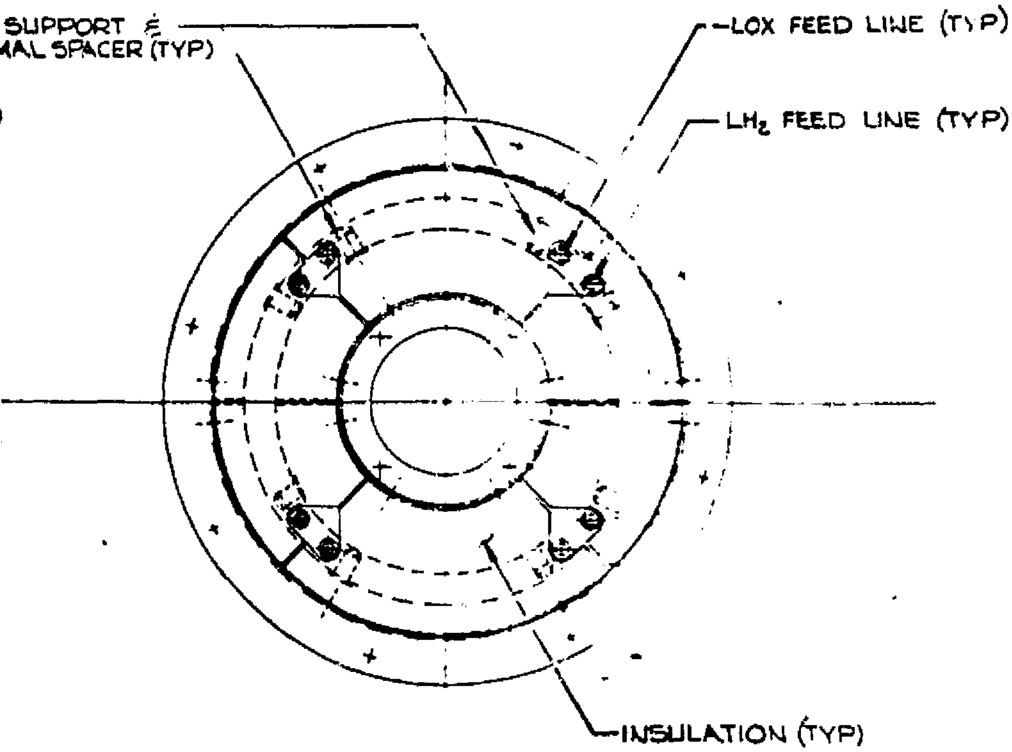

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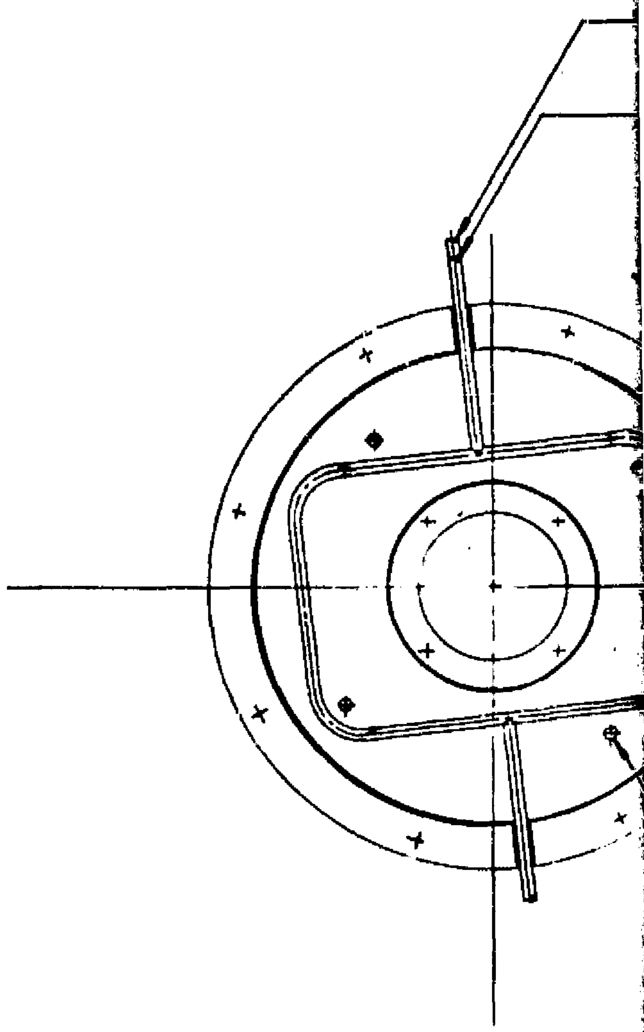

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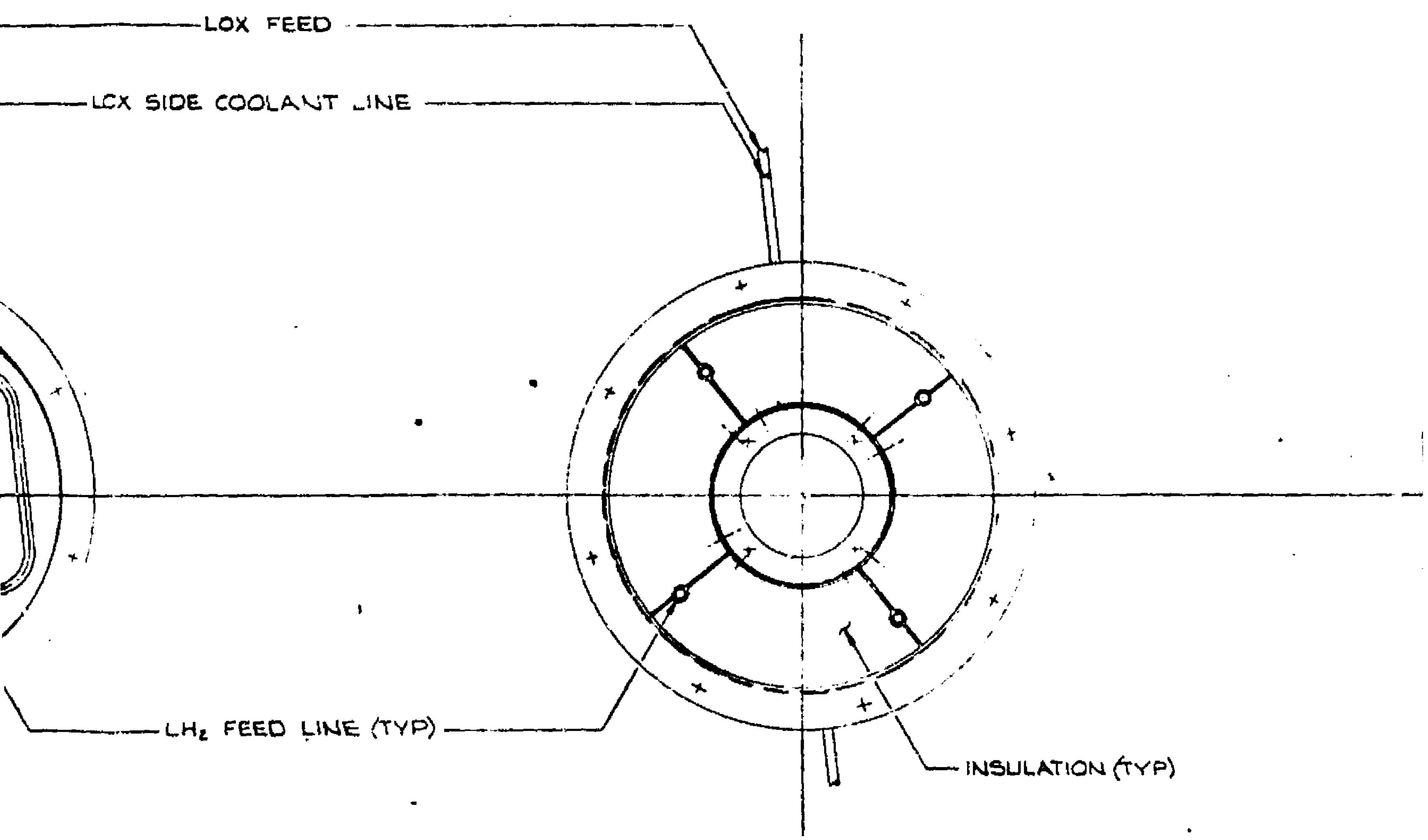

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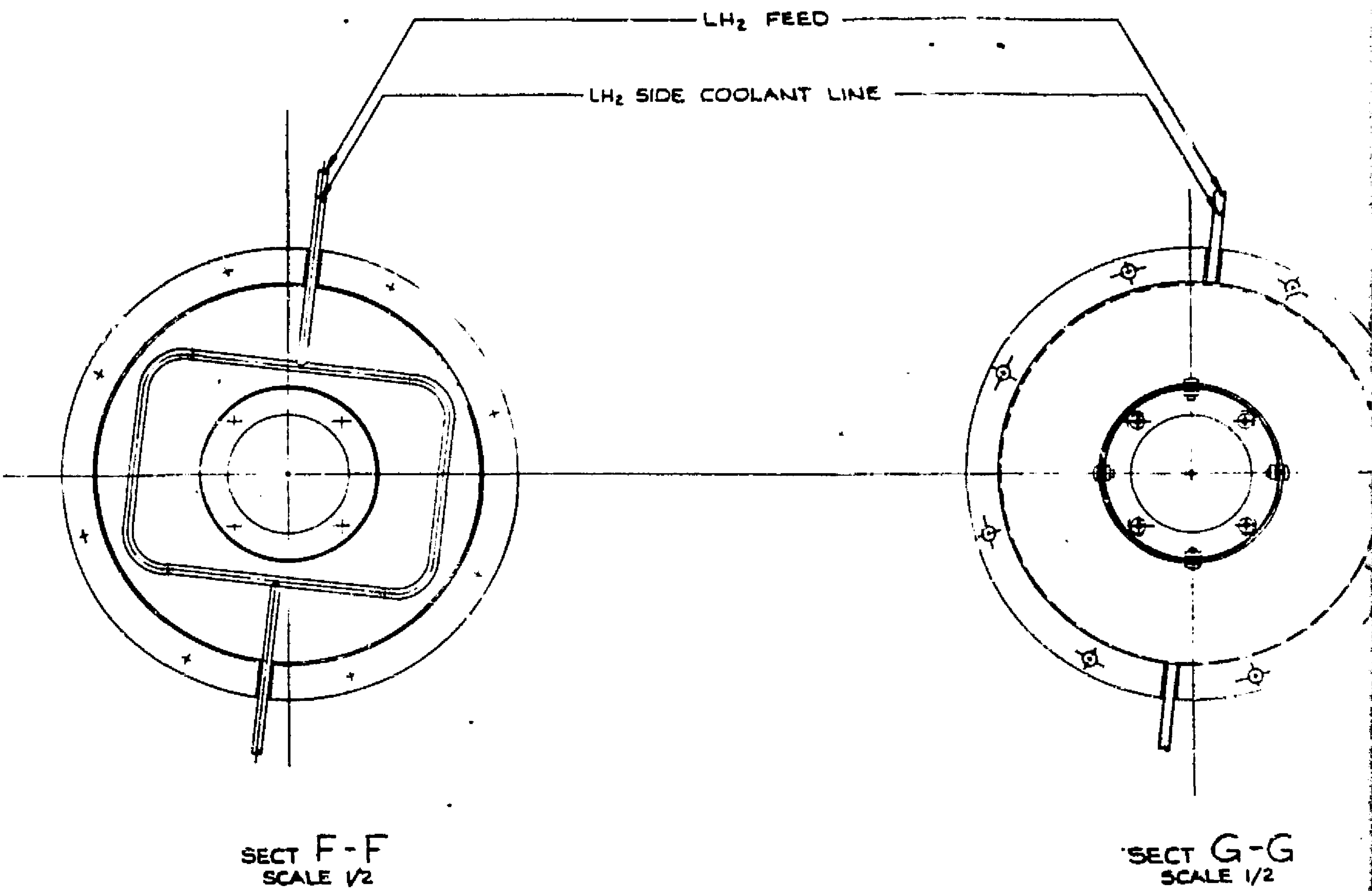

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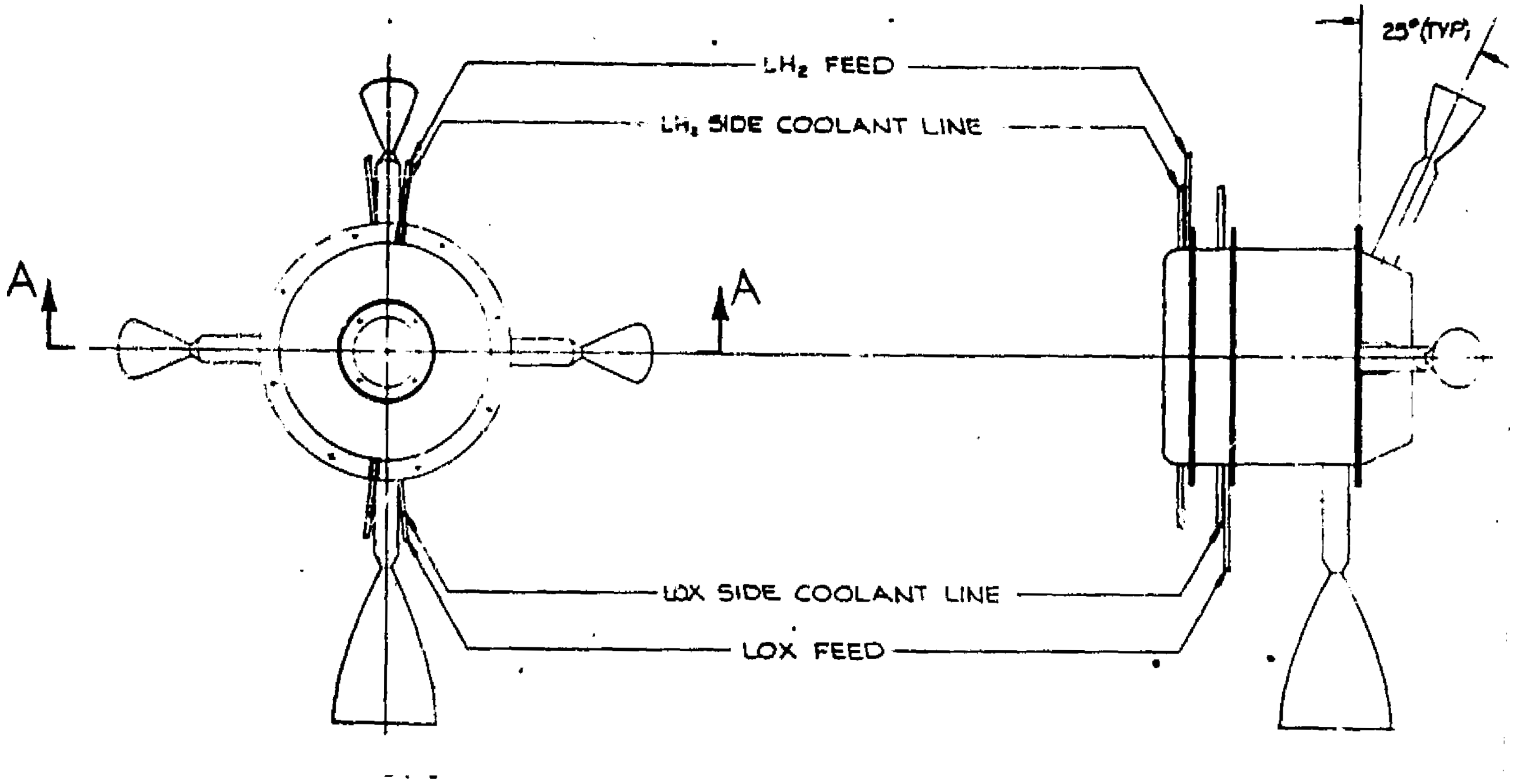

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Figure 9-5. Thruster Preliminary Design

Table 9-7. Thruster Requirements/Characteristics

Item	Normal Mode		Abort Mode
	Roll and -X Thrusters	+X Thrusters	+X Thruster Differences
Thrust, N (lb)	111 (25)		133 (30)
Chamber pressure, N/cm^2 (psia)	103 (150)		125 (180)
Mixture ratio	3		5.6
Nozzle area ratio	50	200	
Specific impulse, N-sec/kg (sec)			
Steady state	3910 (398.7)	4002 (408.1)	3923 (400)
Pulse train (cold)	3069 (319)	3202 (326)	TBD
Minimum bit, N-sec (lb-sec)	2.2 (0.5)		TBD
Steady state flow rate, kg/sec (lb/sec)	0.0284 (0.0627)		0.034 (0.075)
Throat diameter, cm (in.)	0.909 (0.358)		
Chamber diameter, cm (in.)	1.90 (0.748)		
Nozzle length, cm (in.)	8.15 (3.21)	18.7 (7.35)	
Nozzle exit diameter, cm (in.)	6.60 (2.6)	12.9 (5.1)	
Propellant inlet temp, K (R)			
Fuel min/nominal/max	25/28/31 (37/50/55)		
Oxidizer min/nominal/max	86/92/98 (163/165/200)		
Propellant inlet pressure, N/cm^2 (psia)	157 $\pm$8 (220 $\pm$ 15)		
Quad weight, kg (lb)			
Thrust chamber assemblies (3/1)	3.5 (7.8) 1.4 (3.2)		
Valves (8)	2.2 (4.8)		
Redundant power supply (1)	1.2 (2.7)		
Total	8.4 (18.5)		

9.3.5.2.2 Design Point Trades. The thruster nominal performance characteristics are, effectively, minimum requirements. Design and technology development improvements in thruster performance affect system performance (payload) through engine weight, specific impulse, mixture ratio, and chamber pressure. The effect of engine weight is defined in paragraph 9.3.3.3. The estimated effects of the other three design parameters are shown in Figure 9-6.

9.3.5.2.3 Sensitivity to Inlet Conditions. The thruster nominal performance requirements and characteristics, shown in Table 9-7, specify injector manifold and propellant inlet conditions for steady-state and rapid pulse train operation. Steady-state conditions are reached in burns of at least 10 sec. Rapid pulse train operation is defined as having less than a 1-sec off-time between pulses and extending for greater than 1112 N-sec (250 lb-sec).

For initial and intermittent pulse operation, the injector manifold and the inlet propellant adjacent to the thruster valve are at higher than nominal temperature. The temperatures for 2.5 cc (0.15 cu. in.) of either adjacent propellant are influenced by thruster heat soakback which is as yet indeterminant. The design goal is to maintain the bulk temperature of the oxidizer and fuel adjacent volumes to 111 and 33 K (200 and 60 R), respectively. Beyond that point, the propellant thermal control system is

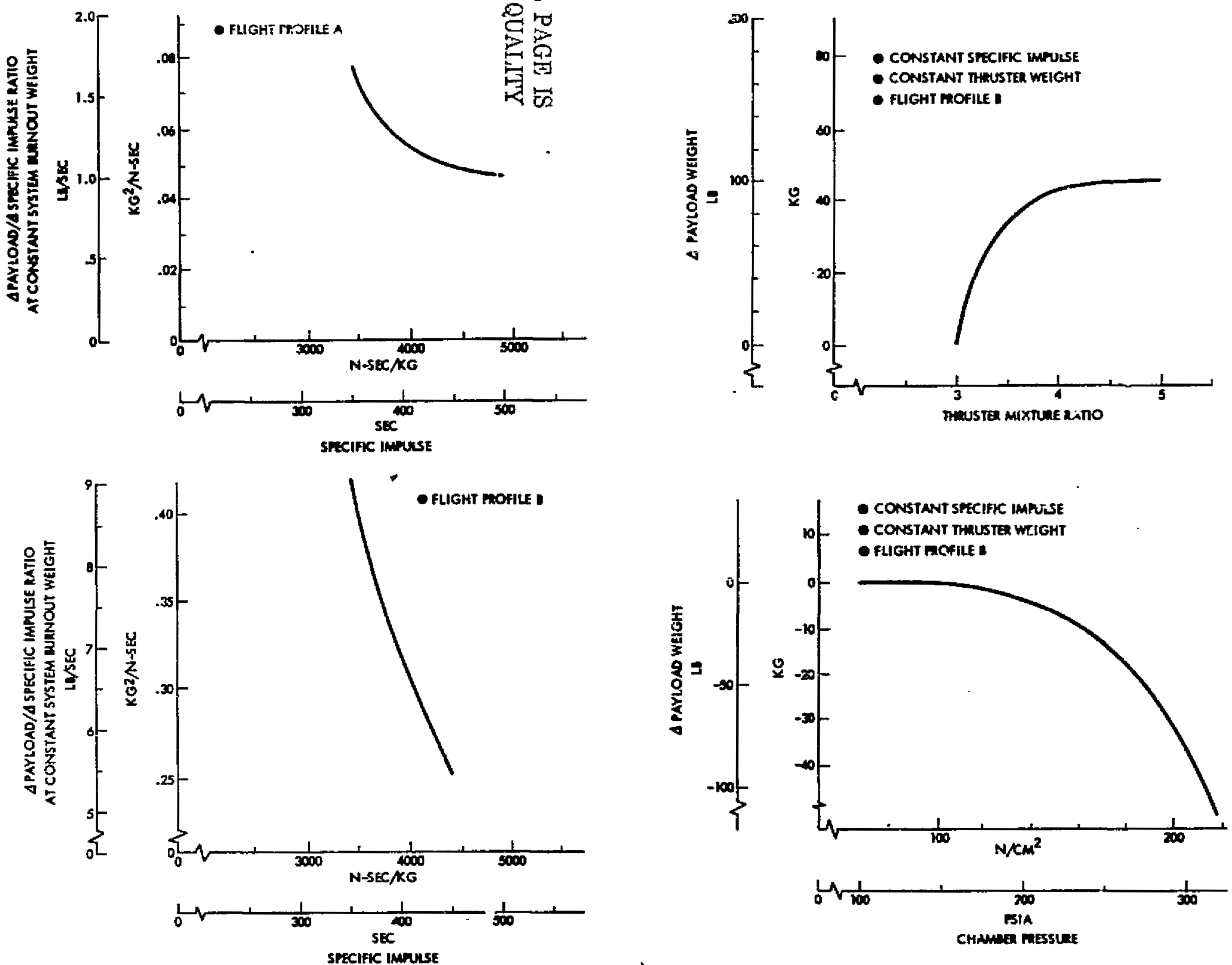


Figure 9-6. Thruster Parametric Design Analysis

expected to maintain nominal temperature conditions. Under these higher temperature conditions, high mixture ratio excursions are not to result in loss of ignition or excessive thruster life/reliability degradation. Specific impulse for pulsing performance is acceptable at values as low as TBD percent of nominal, but minimum bit size shall be not greater than twice nominal. Thruster performance after initiation of a steady-state firing shall reach the nominal value within 10 sec.

9.3.5.2.4 Abort Mode Operation. Steady state operation of the +X thrusters is required for the IAPS abort mode at the MPS (offloaded mission) peak-specific-impulse mixture ratio of 5.6. Thruster performance in this mode shall be not less than 3727 N-s/kg (380 sec). The thruster design is to accommodate this mode of operation without requiring propellant fluid power input in excess of that required for nominal steady-state conditions. Higher than nominal thrust is beneficial. Thruster design firing duration and cycle life requirements are not exceeded in this mode for one abort profile at the end of a life cycle.

9.3.5.2.5 Useful Life. The operating and design life requirements of the thruster are as follows:

Requirement	Thrust Chamber	Igniter	Valves	Exciter
Operating Life				
Cycles per mission	2300	2300	2300	2300
Operating hours permission	2.8	2.8	2.8	168
Missions between refurbishment	20	20	20	20
Design Life				
Cycle	10,000	10,000	10,000	10,000
Operating hours	80	80	80	13,440

9.3.5.2.6 Refurbishment Frequency. No schedule refurbishment of thruster quad elements is anticiapted. Unit replacements will be in response to checkout/flight performance data. Failure rate assessment implies the replacement of 22 of the 32 thruster valves in a ship set over 20 missions at random intervals, and no replacement of other elements.

9.3.5.2.7 Reliability. The apportioned reliability goal for the thruster quad with associated isolation valves is 0.9993.

9.3.5.3 Reservoirs.

9.3.5.3.1 Performance. The reservoirs supply vapor-free propellant to the pumps within the required suction conditions (paragraph 9.3.5.4) during all flight phases. In conjunction with other system elements, they are to be refillable from the MPS tanks during MPS or IAPS ΔV maneuvers once main tank settling is achieved.

The usable propellant capacity of the reservoirs between refills shall be sufficient to meet the IAPS duty cycle requirements of the reference mission Profiles A or B with a margin of 10230 N-sec (2300 lb-sec).

Refill during or soon after an MPS ΔV results in reservoir fluid being subcooled and amount corresponding to MPS autogenous pressurization suppression head. After long coast periods, the MPS tank propellants reach a saturated condition and an IAPS ΔV (only) refill does not change that condition. The function of the reservoir chill circuit heat exchanger is to provide the subsequent subcooling necessary for capillary device stability in 0.5 hr or less. The heat exchanger also functions to intercept and remove environmental heating in order to maintain the subcooled state.

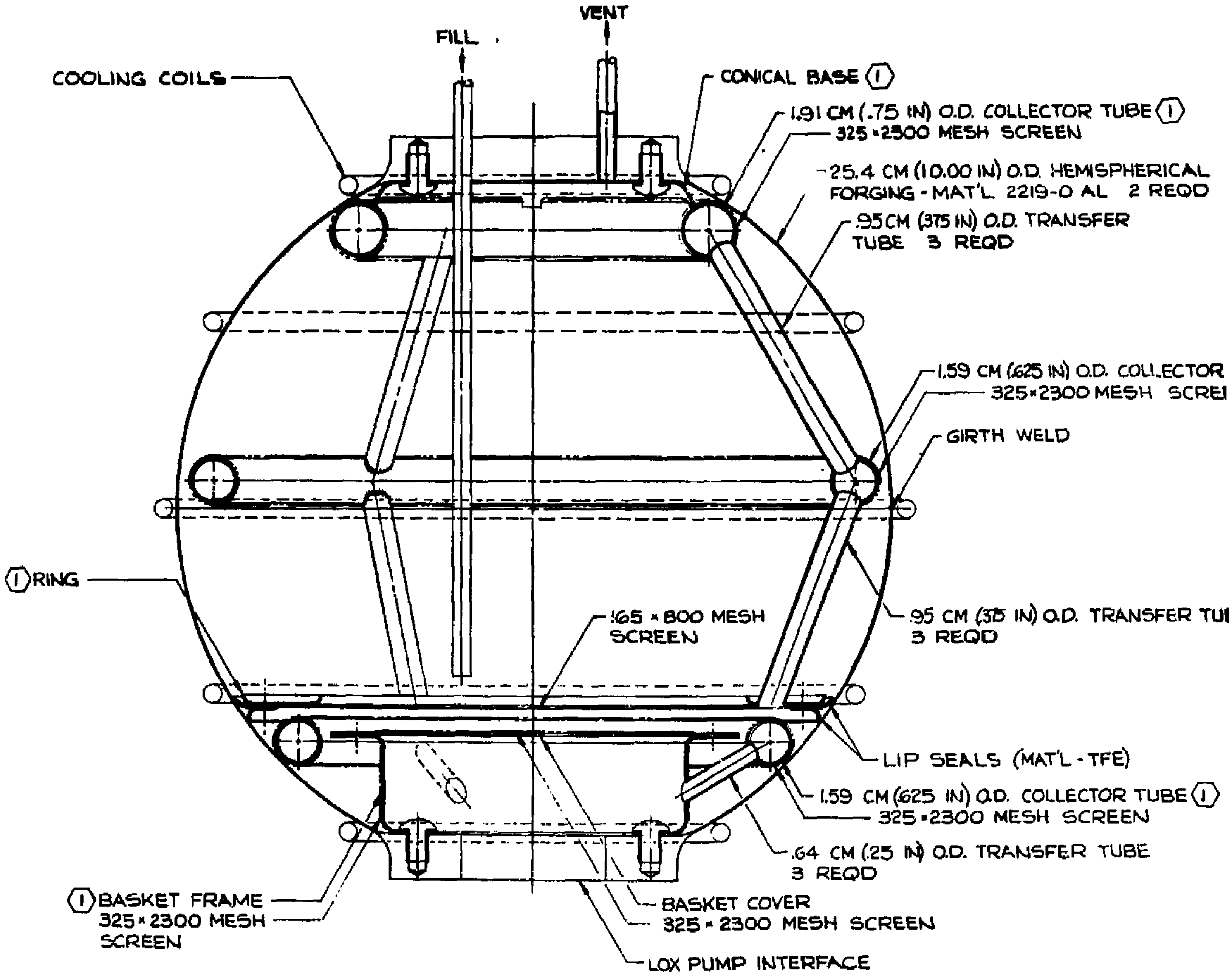
The preliminary design characteristics of the reservoirs are shown in Table 9-8.

Table 9-8. Reservoir Characteristics

	Oxygen	Hydrogen
Volume, m^3 (ft^3)	0.008 (.28)	0.065 (2.3)
Propellant Capacity, kg (lb)	9.5 (20.9)	4.8 (10.5)
Equivalent Impulse, N-sec (lb-sec)	47150 (10600)	47150 (10600)
Attitude Hold Capability, hr	100	100
Trapped Residuals, kg (lb)	0.4 (0.9)	0.3 (0.6)

9.3.5.3.2 Design and Operation. The reservoirs are comprised of a propellant tank, internal capillary device, and external, fuel-vapor-cooled heat exchanger as shown in Figures 9-7 and 9-8. The capillary device divides the reservoir into two compartments separated by a capillary barrier. The upper compartment has the capability of being refilled (purged of vapor) during each of the APS translational maneuvers and MPS burns. The lower compartment offers a degree of redundancy. The design is such that under normal operating conditions, no vapor will enter the lower compartment until depletion of the upper compartment. If an off-design condition occurs, the lower compartment can accumulate some vapor before any vapor is drawn into the acquisition tubes and to the APS pump. As the thermal control system condenses this vapor, the capillary acquisition tubes will replenish the lower compartment with more liquid.

The acquisition tubes provide a communication path for liquid from the upper to the lower compartment. They are arranged to be in contact with liquid under any adverse acceleration that might be imposed. The tube's self-wicking screen covers are designed to be wetted during the entire mission in order to pass liquid and block vapor flow.



LOX RESERVOIR CROSS-SECTION VIEW

FOLDOUT FRAME

Figure 9

5 IN) O.D. COLLECTOR TUBE (1)
300 MESH SCREEN

CM (10.00 IN) O.D. HEMISPHERICAL
NG - MAT'L 2219-0 AL 2 REQD

CM (.375 IN) O.D. TRANSFER
BE 3 REQD

1.59 CM (.625 IN) O.D. COLLECTOR TUBE (1)
325×2300 MESH SCREEN

GIRTH WELD

.95 CM (.375 IN) O.D. TRANSFER TUBE
3 REQD

ESTIMATED WEIGHT	KG	LB
DRY	1.76	3.87
CAPACITY	9.47	20.89
TOTAL LOADED WEIGHT	11.23	24.76

LIP SEALS (MAT'L - TFE)

(.625 IN) O.D. COLLECTOR TUBE (1)
325×2300 MESH SCREEN

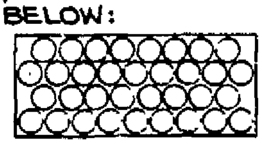
5 IN) O.D. TRANSFER TUBE

REEN

FOLDOUT FRAME

2

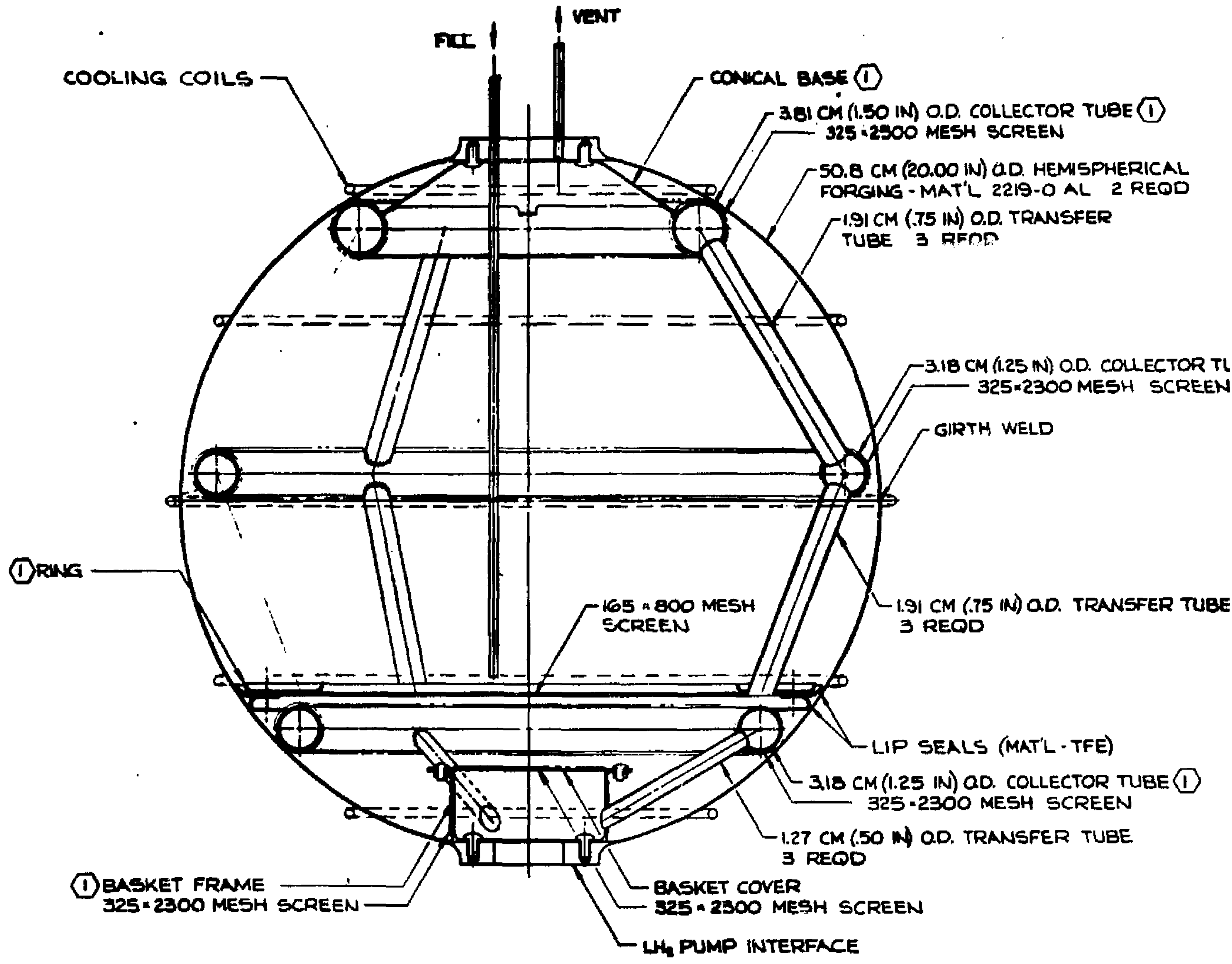
(1) 2. THESE COMPONENTS SHALL HAVE .64 CM (.25 IN) DIA HOLES (60% OPEN AREA) AS SHOWN BELOW:



1. ALL MAT'L ARE CRES EXCEPT AS NOTED

NOTES:

Figure 9-7. IAPS Liquid Oxygen Reservoir



LH_2 RESERVOIR CROSS-SECTION VIEW

FOLDOUT FRAME

Figu

N) O.D. COLLECTOR TUBE (1)
O MESH SCREEN

(20.00 IN) O.D. HEMISPHERICAL
-MAT'L 2219-O AL 2 REQD

(.75 IN) O.D. TRANSFER
3 REQD

3.18 CM (1.25 IN) O.D. COLLECTOR TUBE (1)
325×2300 MESH SCREEN

GIRTH WELD

ESTIMATED WEIGHT	KG	LB
DRY	4.76	10.50
CAPACITY	4.78	10.54
TOTAL LOADED WEIGHT	9.54	21.04

91 CM (.75 IN) O.D. TRANSFER TUBE
3 REQD

FOLDOUT FRAME

2

SEALS (MAT'L - TFE)

25 IN) O.D. COLLECTOR TUBE (1)
×2300 MESH SCREEN

O.D. TRANSFER TUBE

N

NOTES:

1. ALL MAT'L ARE CRES EXCEPT AS NOTED

(1) 2. THESE COMPONENTS SHALL HAVE 1.27 CM (.50 IN) DIA HOLES (60% OPEN AREA) AS SHOWN BELOW:

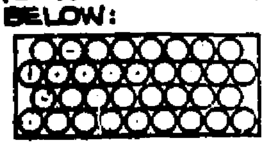


Figure 9-8. IAPS Liquid Hydrogen Reservoir

After the propellants are settled during either an MPS or APS translation maneuver, the IAPS refill vent valves are opened and the reservoir is replenished to 100-percent liquid. Baffles in the tank vent exit are to minimize losses due to liquid entrainment in the vented fluid. Liquid point sensors are installed to activate closure of the vent valves. APS propellant demand for the remainder of any ΔV maneuver is supplied by direct flow of liquid from the MPS tank through the reservoir and to the APS pump. The reservoir size thus is unaffected by extended APS ΔV.

9.3.5.3.3 Useful Life. The operating life of the reservoirs is 20 missions without refurbishment. The design life is 80 missions.

9.3.5.3.4 Reliability Apportionment. The apportioned reliability of each reservoir is .99995.

9.3.5.4 Pump and Drive Assemblies.

9.3.5.4.1 Performance. The function of the oxidizer and fuel pumps is to provide the head rise from MPS tank to IAPS thruster inlet pressure at a maximum combined flow rate equal to the steady-state demand of four thrusters. During normal mode operation, the oxidizer and fuel mass flow rates correspond to the normal mode mixture ratio. During abort mode operation, the rates correspond to the MPS offloaded mission mixture ratio.

The pumps are required to provide nominal performance at zero NPSH and up to 10-percent vapor (by volume). Pump requirements and characteristics resulting from preliminary design are listed in Table 9-9.

9.3.5.4.2 Design and Operation. The pump and drive assemblies are comprised of an axial flow inducer stage, a reciprocating pump stage and a 3-phase 115/200v, 210/240 Hz motor (oxidizer/fuel). The preliminary design of these units, provided by Sundstrand Corporation, is shown in Figure 9-9. In operation, the axial flow inducers boost the saturated inlet propellant to the pressure level acceptable for the piston stage design suction specific speed. Motor speed is controlled by inverter power supply frequency. For safety reasons, the oxidizer motor uses a canned stator. The discharge check valve is required to seal only to the degree necessary for acceptable volumetric efficiency. During pump inactive periods, fuel flow from the accumulator is prevented by line-mounted quad redundant check valves.

9.3.5.4.3 Useful Life. The pump drive units have a required operating life of 10 Profile B missions plus one worst-case abort mission between refurbishments. This involves a total normal mission operating time of 26 hours, plus 17 hours of abort mode operation and 1500 operating cycles. The design life for the units is 125 hours and 6000 cycles.

9.3.5.4.4 Reliability Apportionment. The reliability apportionment for the oxidizer and fuel pump and drive units is 0.998 and 0.997, respectively.

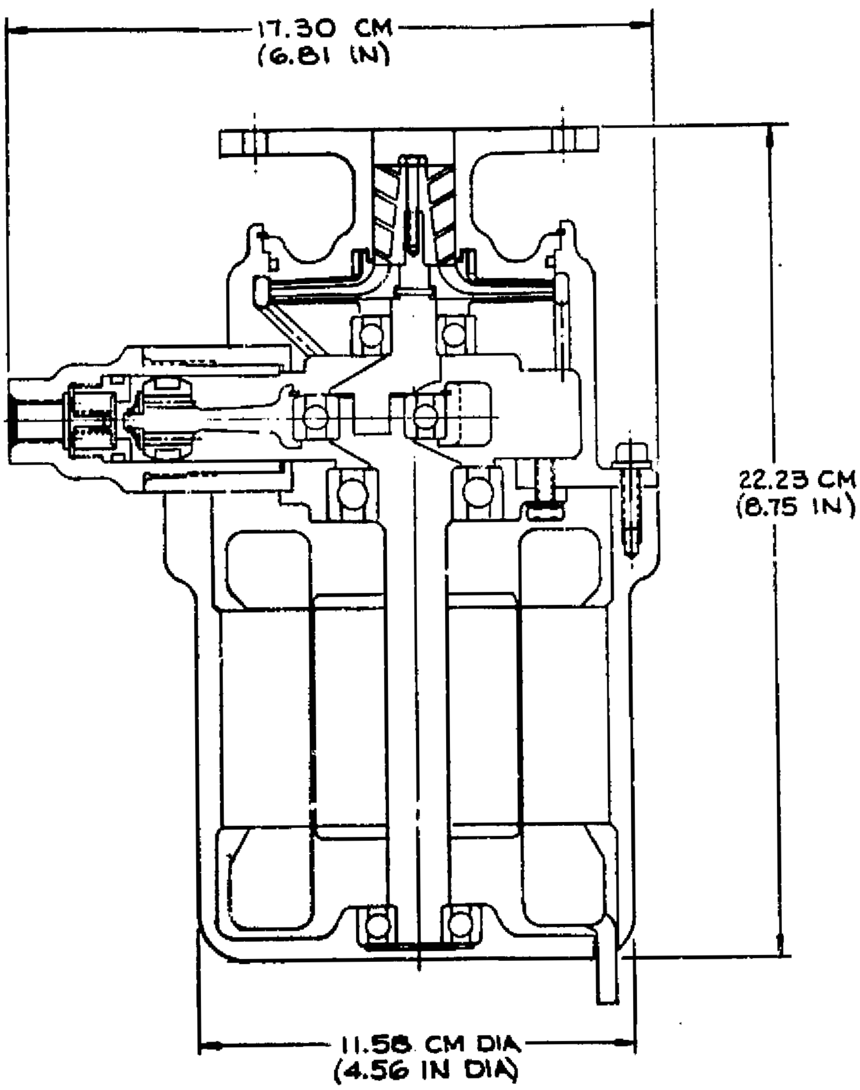
Table 9-9. Pump Requirements and Characteristics

	Normal Mode		Abort Mode (Constant Power)	
	Oxidizer	Fuel	Oxidizer	Fuel
Requirements				
Flow rate, m^3/sec (gpm)	.00454 (1.2)	.0245 (6.47)	.00640 (1.69)	.0186 (4.92)
Head rise, m (ft)	129 (425)	2110 (6920)	199 (654)	2170 (7128)
Fluid output power, kw (hp)	.109 (.147)	.590 (.792)	.238 (.320)	.461 (.620)
NPSP	0	0	0	0
Vapor capacity, %	10	10	10	10
Characteristics				
Boost stage				
Head rise, N/cm^2 (psi)	1.1 (1.6)	.52 (0.75)	1.5 (2.2)	.31 (.45)
Efficiency, %	32	35	36	28
Pump Stage				
RPM	3000	3000	4230	2280
Design specific speed	35.7	10.1	43.2	36.5
Displacement per rev, cc (cu. in.)	1.61 (.0985)	8.90 (.594)	Same	
Overall effiency, %	75	78	70	80
Weight, kg (lb)	.23 (.50)	.27 (.60	Same	
Motor				
Efficiency, %	70	92	82	85
Output power, kw (hp)	.147 (.198)	.750 (1.01)	.334 (.45)	.575 (.775)
Weight, kg (lb)	2.04 (4.5)	3.09 (6.8)	Same	
Assembly Total				
Weight, kg (lb)	3.6 (7.9)	4.8 (10.5)	Same	
Power demand, kw (hp)	.246 (.330)	1.01 (1.35)	.559 (.75)	.77 (1.03)

9.3.5.5 Accumulators.

9.3.5.5.1 Performance. The function of the accumulators is (1) to damp reciprocating pump flow pulsations, and (2) to provide feed pressure storage sufficient to limit short-cycling of the pump/drive units and supply enough propellant for Shuttle docking attitude hold in the event of a pump failure during that operation. The usable propellant capacity of the units corresponds to 1081 N-sec (243 lb-sec) of total impulse which provides for approximately 1.5 hours of attitude hold capability.

9.3.5.5.2 Characteristics. The preliminary designs of the accumulators are shown in Figures 9-10 and 9-11. Each accumulator is a trapped ullage device. Helium external to the bellows is compressed or expanded as propellant is pumped into or withdrawn from within the bellows. The helium volumes correspond to the pressure limits of the system with adiabatic compression and isothermal expansion. The units are all-welded with provision for bellows replacement by cutting and rewelding. Externally mounted bellows



LIQUID HYDROGEN PUMP

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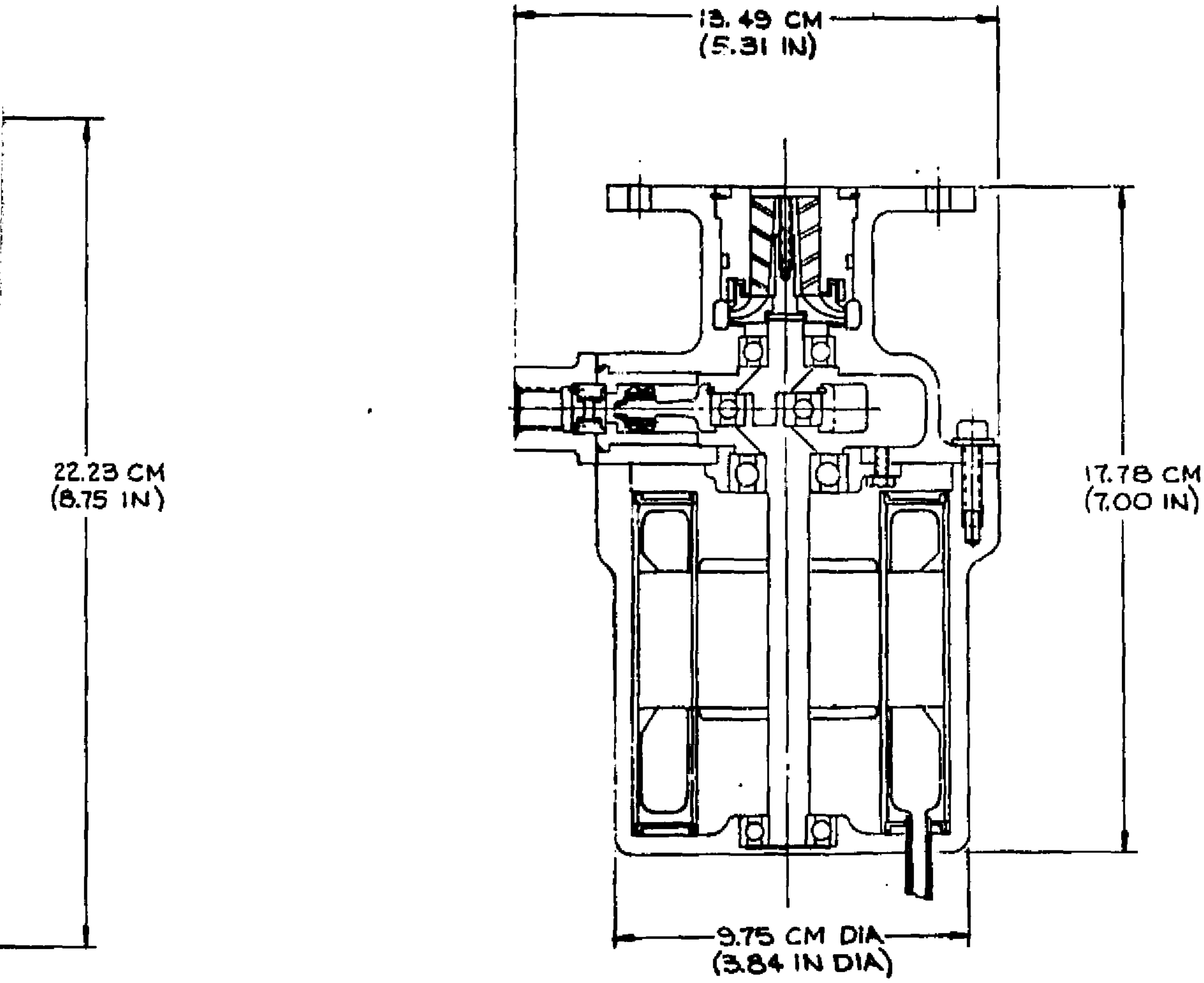
Figure 9-9. IAPS Motor



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LIQUID OXYGEN PUMP

NOTE: DESIGN PROVIDED BY
SUNSTRAND CORP

Figure 9-9. IAPS Motor-Driven Pump Piston

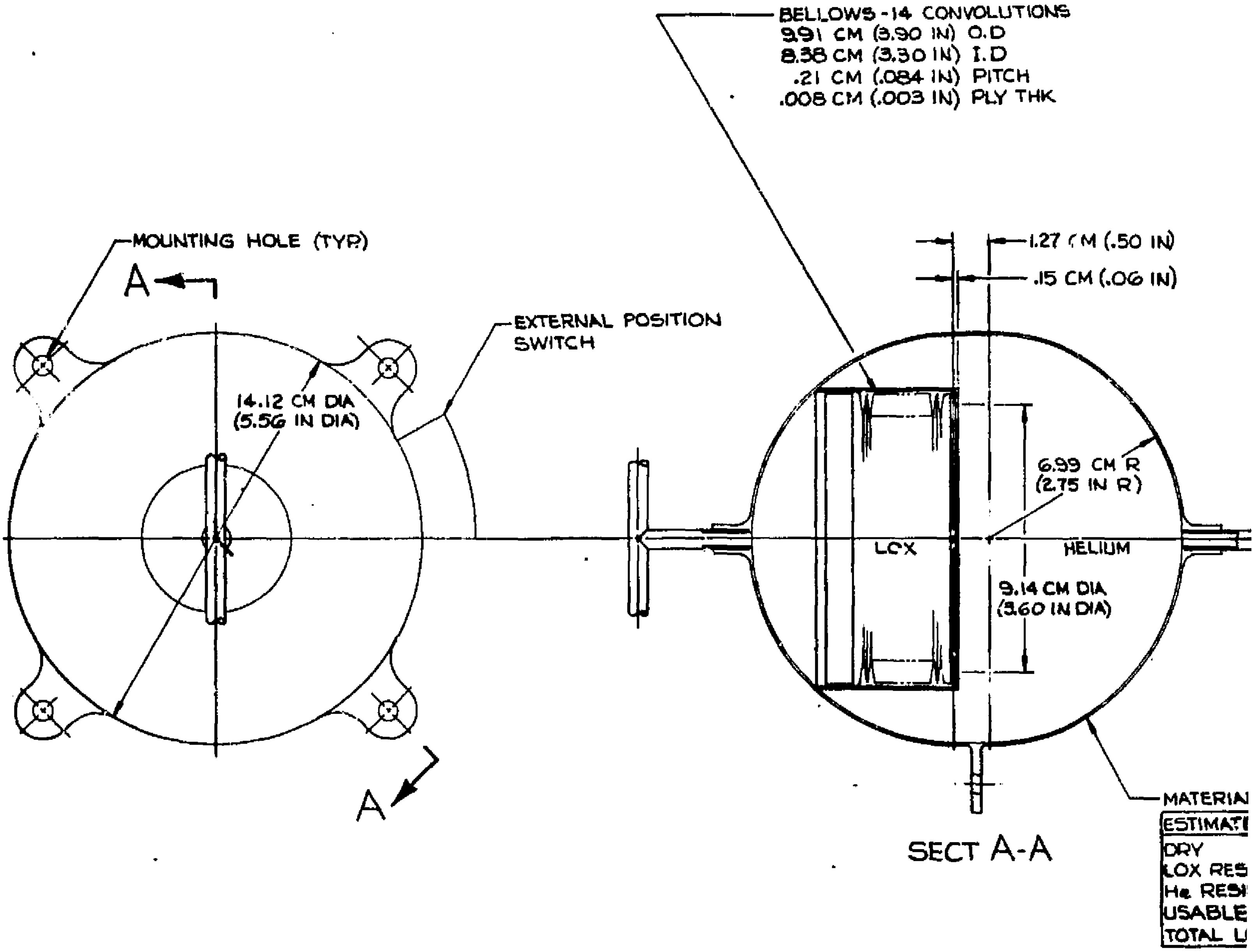

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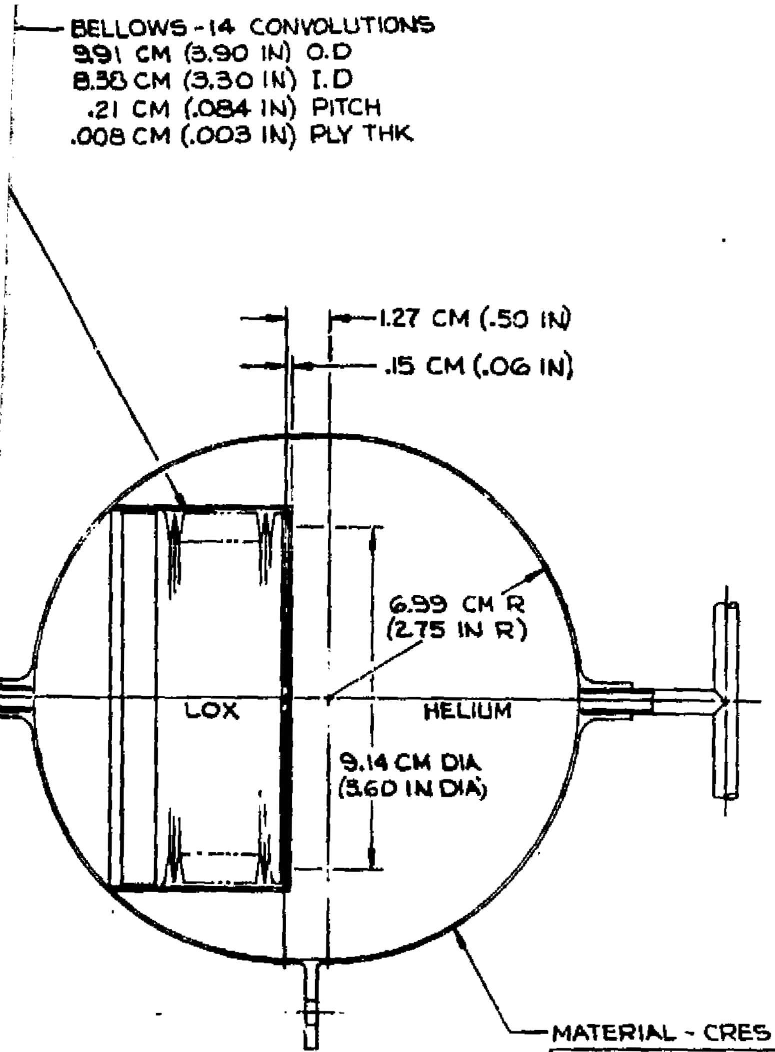
Figure 9-10. IAPS Liquid Oxygen Accum



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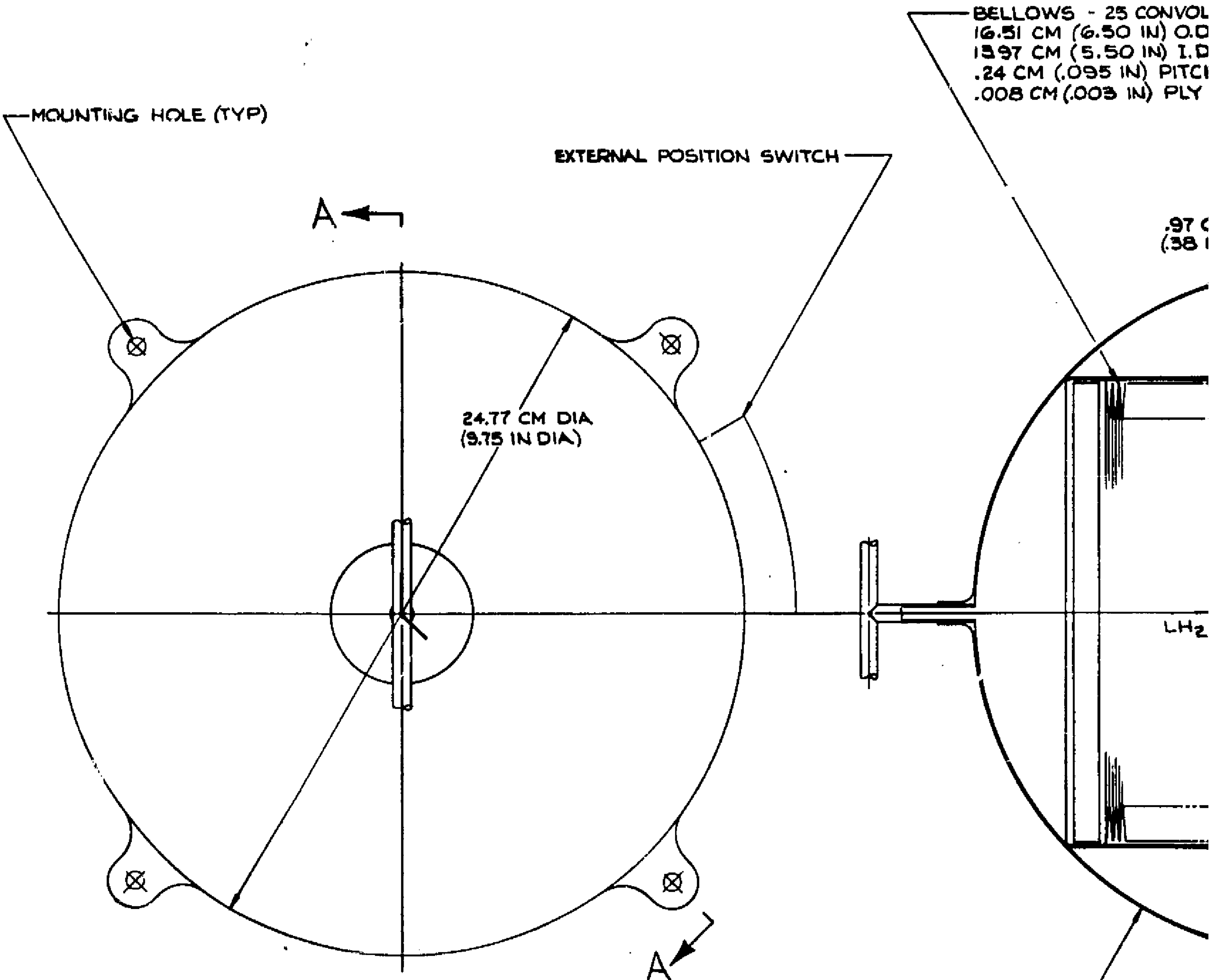
ESTIMATED WEIGHT	KG	LB
DRY	.665	1.466
LOX RESIDUALS	.258	.563
He RESIDUALS	.010	.023
USABLE LOX	.207	.457
TOTAL LOADED WEIGHT	1.140	2.515

Figure 9-10. IAPS Liquid Oxygen Accumulator



MATERIAL - CRES

ESTIMATED WEIGHT	KG	LB
DRY	2.18	4.80
LH_2 RESIDUALS	.07	.16
He RESIDUALS	.20	.45
USABLE LH_2	.07	.15
TOTAL LOADED WEIGHT	2.52	5.56

FOLDOUT FRAME

Figure

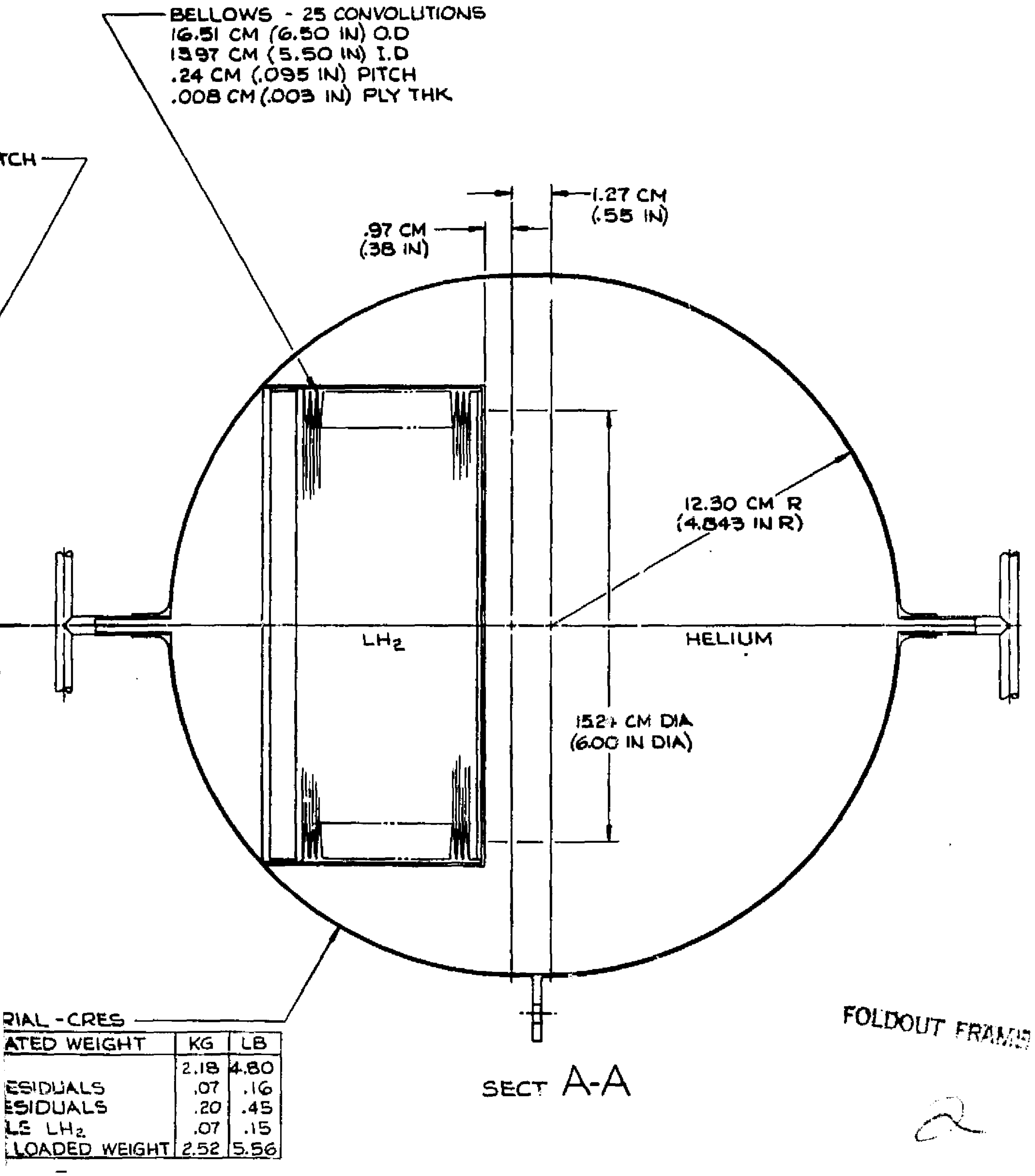


Figure 9-11. IAPS Liquid Hydrogen Accumulator

position switches act to provide pump control loop information (see paragraph 9.3.5.7). Loading of the helium is a ground service operation for the oxygen accumulator and a flight activation operation for the fuel accumulator. Both operations result in external pressure on a collapsed, flat bellows. Subsequent activation by filling with propellant chills the oxygen unit helium to working pressures and eliminates differential pressure across each of the bellows to permit cycling at low stress levels.

9.3.5.5.3 Useful Life. The accumulator operating life requirement is for 10 normal mode missions plus one abort mode mission between bellows replacement. This operating life corresponds to 1500 bellows expulsion cycles plus 20 external pressurized collapsing cycles. The design life is 6000 cycles at equalized internal and external pressure plus 80 collapsing cycles with external pressure maintained for 30 and 2 hours between cycles for oxidizer and fuel units, respectively.

9.3.5.5.4 Reliability. The apportioned reliability of the oxidizer and fuel accumulators is .998 and .997, respectively. A failure of the accumulators, which are single-point failure components, is to be fail-safe during Orbiter-Tug docking. The failure mode to be considered is bellows rupture. This type of failure is to be detected by bellows position indicator signals showing lack of bellows cycling. The downstream capability of the propellant system is to be sufficient to permit safe continuation or termination of Orbiter-Tug docking after helium pressurant/propellant mixing upon bellows rupture. Downstream volume provisions shall include a capillary tube channel or similar device to retard helium gas migration through the downstream propellant sufficiently to permit this docking operation or termination.

9.3.5.6 Insulation and Purge System

9.3.5.6.1 Performance. The insulation system has the function of minimizing heat flow from the environment to all propellant-containing elements of the IAPS. The insulation purge system has the function of providing an enclosure (bag) around the insulation to exclude all contaminants. During ground servicing with propellants, the purge bag contains helium to preclude cryopumping of moisture and oxygen from the air. After servicing and disconnect, insulation is to be effective, as limited by helium conductivity, in minimizing launch through orbit insertion boiloff losses to the same value per unit of surface area as the MPS insulation. Insulation venting during ascent shall limit internal gauge pressure to 0.14 N/cm^2 (0.2 psia) and less than 10^{-4} torr shall be achieved within 0.5 hour after orbit insertion. After venting to that level, insulation effectiveness through IAPS deactivation shall be equivalent to an effective emittance of 0.002 including the contribution of penetrations and supports. The insulation shall be repressurized during descent with Orbiter cargo bay helium provided to the MPS for this purpose.

9.3.5.6.2 Design and Operation. The elements of the IAPS enclosed by the insulation and purge system are indicated in Figures 9-2 and 9-4. The system consists of 1.27 to 2.54 cm (0.5 to 1.0 in.) thick blankets of multi-layer insulation of the same material and installation method to be selected for the MPS. (A Mylar or Kapton purge bag, MLI of single-aluminized Mylar or single goldized Kapton, thermal isolation supports using titanium struts, or axially oriented S-glass rods are representative choices.) Vent valves for self-actuation during ascent and actuated closed for descent also are to be similar to the type selected for the MPS.

9.3.5.6.3 Useful Life. The insulation and purge system is to be designed for a 20-mission operating life with no scheduled refurbishment.

9.3.5.7 Controls.

The system is controlled through the Tug umbilical disconnect prior to launch and at any time while attached to the Tug erection ring of the Shuttle. In free flight, the system is automatically controlled by the Tug onboard computer, with specific measurements and commands to/from the ground or Shuttle via the Tug communication system. Control logic and interfaces are described in Figure 9-12. The IAPS sequence of events is listed in Table 9-10, where the numbers in parentheses identify components.

9.3.5.7.1 Major Control Functions. Operation of the system involves the following major control functions.

9.3.5.7.1.1 Reservoir Fill and Drain. APS reservoirs are filled and drained for prelaunch test, checkout, and purge through the MPS fill and drain system using the tank isolation valves. The valves are opened during Tug flight activation and closed at deactivation.

9.3.5.7.1.2 Reservoir Refill. During flight operations the reservoirs are refilled during delta-V maneuvers by opening the vent valves. These valves close by liquid-level sensor signals when the reservoirs are full.

9.3.5.7.1.3 Propellant Pump Speed. The pumps provide a low flow rate for low thruster duty cycle conditions such as attitude stabilization. Low constant-speed pump operation is activated by the minimum bellows deflection switch and is deactivated when the maximum bellows deflection switch is reached. High flow rates are provided for delta-V thruster operation, in which the pump speed is proportionally controlled between specified accumulator pressure set points.

9.3.5.7.1.4 Hydrogen Bleed Control. The bleed system has two control modes: low and high circulation flow, with controlled variable heat input. High circulation flow is obtained for APS activation and deactivation, while low circulation flow is used during free flight. High flow also may be used whenever the upper hydrogen temperature bound is reached at the reservoir

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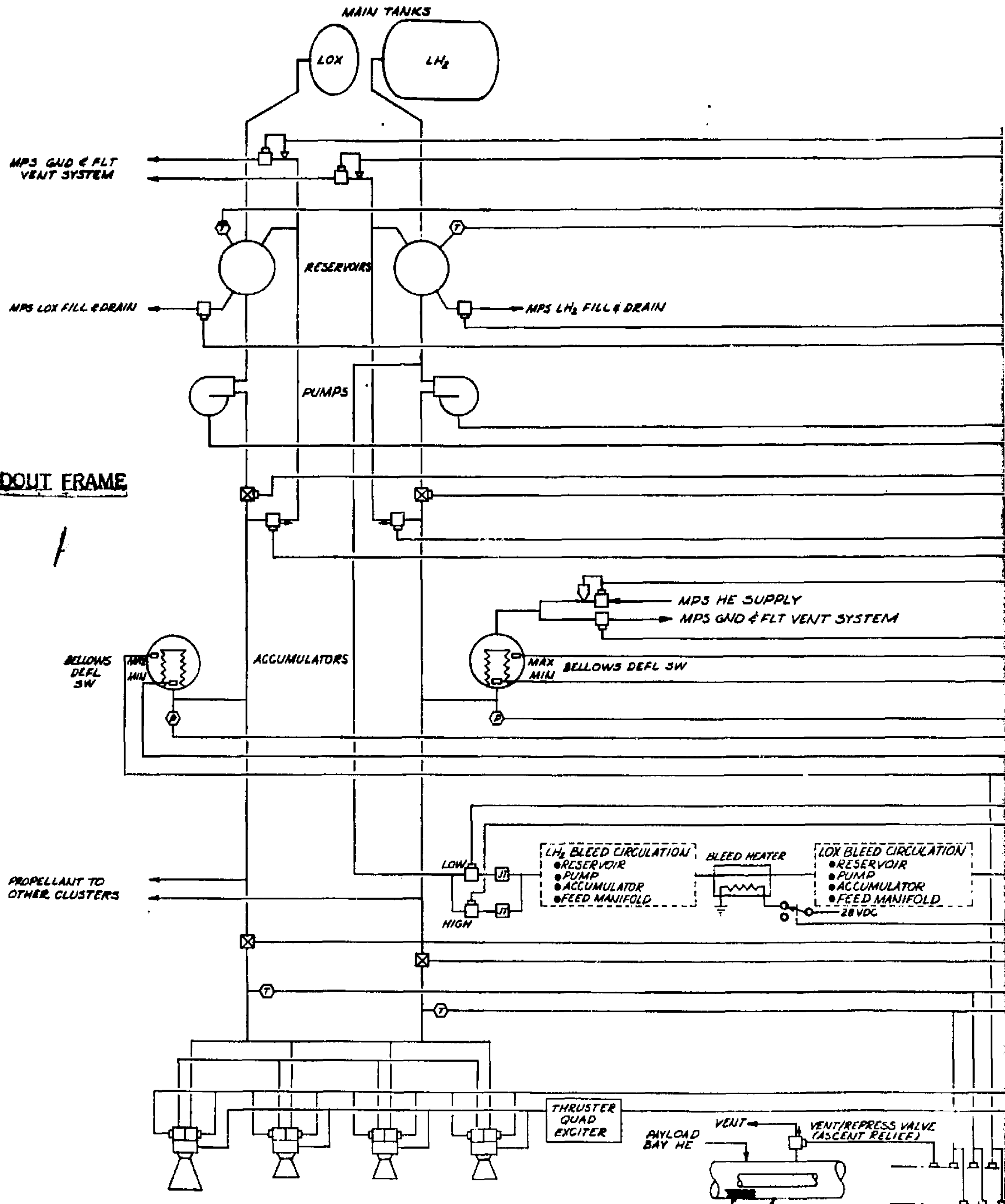

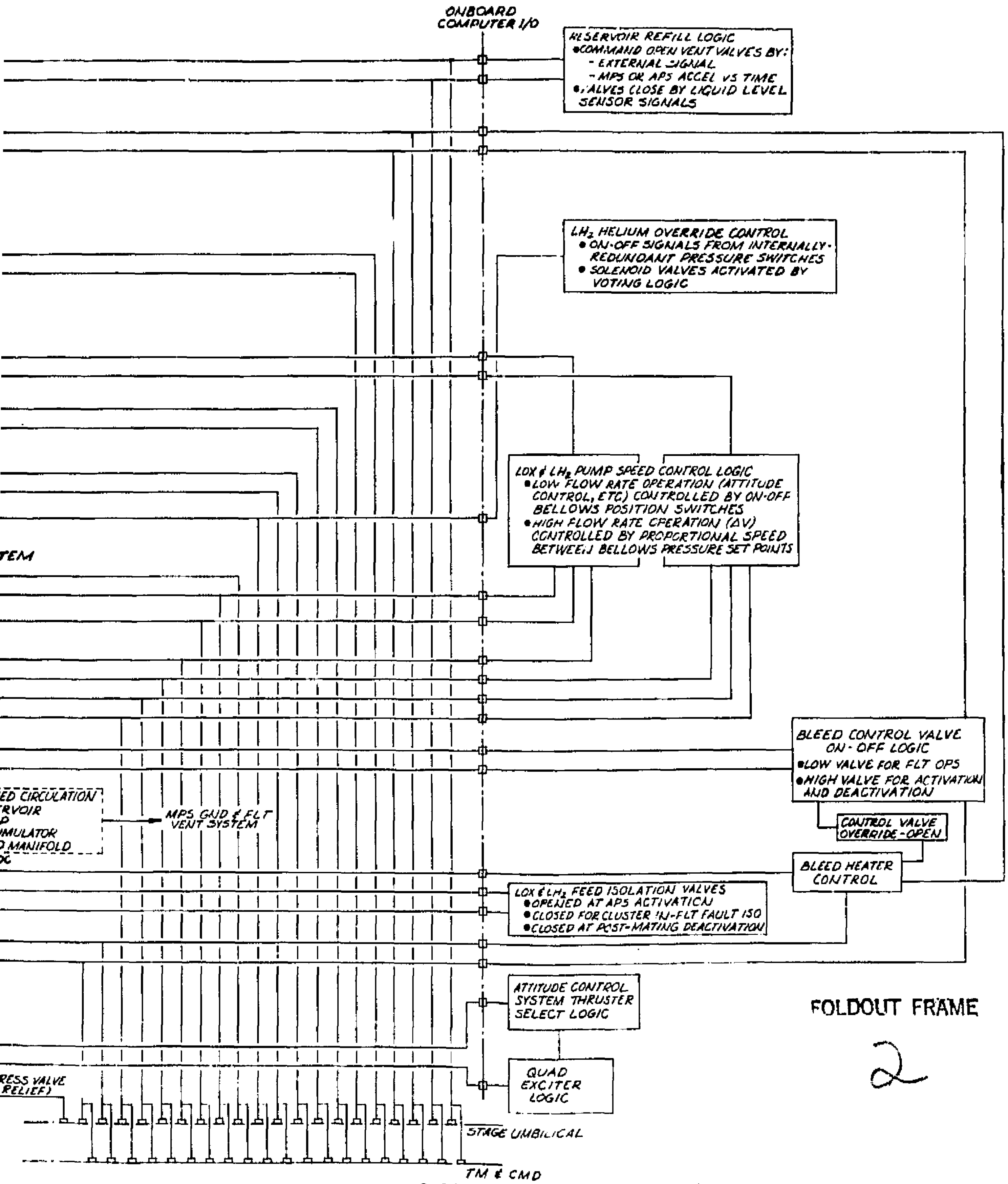

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Figure 9-12. Tug IAPS Controls Diagram

Table 9-10. IAPS Mission Sequence of Events

- Ground Checkout
- Fluid Servicing (Including Reservoirs)
- Liftoff
- Shuttle Ascent
  - Insulation venting - purge bag vent valve self-opening and latched at set differential pressure
- Predeployment Checkout
- Tug Erection
  - Orbiter-Tug MPS fluid disconnects separated
- APS Activation
  - Open fuel accumulator pressurization helium valves (36)
  - Open tank and feed isolation valves (55, 56, 18, 20)
  - Open high and low H_2 bleed valves (62, 38) and vent through Tug MPS nonpropulsive vent
  - Arm pump controls (pumps operate)
  - Verify IAPS ready (control temps and press normal)
  - Close high hydrogen bleed valve (39)
- Tug Deployment (Release)
  - Attitude control initiated TBD sec after manipulator arm separation
- Tug Flight Phase
  - Bleed control continuously operational [low valve (38) Open]
  - Pump control
    - Low flow rate operation (attitude control, etc.) is controlled from ON-OFF bellows position switches
    - High flow rate operation (normal mode ΔV MR = 3:1, abort mode ΔV MR = 5.6) controlled by proportional speed between bellows pressure set points
  - Reservoir refill
    - Open fill vent valves (26, 28) based on APS or MPS ΔV acceleration vs. time logic or ground control command
    - Close fill vent valves by liquid point sensor (59, 60) signals
    - Thruster quad isolation activated by failure mode leak identification (onboard computer or ground)
- Tug Retrieval by Orbiter
  - Predocking
    - Command accumulator fill cycle
    - Safety checkout, system safing to attitude stabilization mode
  - Manipulator arm contact and IAPS deactivation
    - Disarm thruster firing command logic
    - Shut bleed control valve (38)
    - Disarm pump control logic
  - Tug-erector mating
  - Orbiter-Tug MPS fluid disconnects mated
  - IAPS purge/inerting
    - MPS-Orbiter vent system activated
    - Open IAPS bleed and vent valves (38, 62, 24, 28) to purge
    - MPS helium purge cycle accomplished
    - Close bleed cutoff valves
    - Close tank and feed isolation valves
    - Close insulation vent valves and pressurize bag
- Orbiter Descent and Landing
  - Insulation purge bag pressure program - controlled differential pressures using MPS repress system

or at the thruster quad manifold. Power to the bleed heater is controlled using oxygen reservoir and thruster quad manifold temperature measurements. If the oxygen temperature reaches its upper bound the bleed heater logic can override the valve logic to open the valves.

9.3.5.7.1.5 Feed Isolation Valves. The oxygen and hydrogen thruster quad isolation valves are opened at Tug activation and closed during deactivation. A thruster quad set may be closed during flight to isolate a thruster valve failure.

Table 9-11. Component Design Requirements

COMPONENT NAME	ID NO.	FUNCTION	OPERATION/DESCRIPTION	PERFORMANCE/PROCESS DATA PER PROCESS DIAGRAM AND AS NOTED	REFURBISH INTERVAL	OPERATING LIFE PER MISSION		DESIGN LIFE		POWER
					MISSIONS	CYCLES	HOURS	CYCLES	HOURS	WATTS
FILL AND DRAIN SYSTEM										
LOX DRAIN VALVE	51	GROUND SERVICING	NC SOLENOID		20	2		160		20
LH_2 DRAIN VALVE	53	GROUND SERVICING	NC SOLENOID		20	2		160		20
PRESSURIZATION SYSTEM										
He RUPTURE DISCONNECT - LOX	64	SAFETY - PROTECTION		330 TO 345 N/cm^2 (480 TO 500 psia) SET PRESS	2	0	336		1,350	
He DISCONNECT - LOX	63	GROUND SERVICING		345 N/cm^2 (500 psia) LIMIT PRESSURE	20	3		240		
He ISOLATION VALVE - LH_2	36	FLIGHT SERVICING	NC SOLENOID		10	5		200		20
He PRESSURE SWITCH - LH_2	37	INITIAL CHARGE AND MAKEUP		135/141 N/cm^2 (195/205 psia) ON/OFF	20	5		400		
PROPELLANT CONTROL SYSTEM										
LH_2 BLEED SHUTOFF VALVE	38	NORMAL BLEED SHUTOFF	NO SOLENOID		10	3		120		20
LH_2 BLEED SHUTOFF VALVE	61	CHILLDOWN BLEED	NO SOLENOID		10	2		80		20
LH_2 BLEED EXPANDER	16	NORMAL BLEED JT EXPANSION	THROTTLING ORIFICE		20		168		13,500	
LH_2 BLEED EXPANDER	62	CHILLDOWN BLEED JT EXPANSION	THROTTLING ORIFICE		20		2		160	
LH_2 BLEED HEATER	49	REHEAT TO LOX TEMPERATURE	ELECTRIC ROD AND TUBE		20		168		13,500	4
PROPELLANT FEED SYSTEM										
LOX SYSTEM ISOLATION VALVE	55	IN-ORBITER LEAK SUPPRESSION	NO SOLENOID		20	3		120		20
LH_2 SYSTEM ISOLATION VALVE	56	IN-ORBITER LEAK SUPPRESSION	NO SOLENOID		20	3		120		20
LOX PUMP CHECK VALVE	40	BACK FLOW CHECK	SPRING CHECK	SLOW RESPONSE - LESS THAN 3 cps	3	150		1,800		
LH_2 PUMP CHECK VALVE	45	BACK FLOW CHECK	SPRING CHECK	SLOW RESPONSE - LESS THAN 3 cps	3	150		1,800		
LOX QUAD ISOLATION VALVE	18	SHUTOFF FAILED - OPEN THRUSTER	NO LATCHING SOLENOID		20	2		140		20
LH_2 QUAD ISOLATION VALVE	20	SHUTOFF FAILED - OPEN THRUSTER	NO LATCHING SOLENOID		20	2		140		20
LOX RELIEF VALVE	42	RELIEF BYPASS	ORIFICED, SPRING, CLOSED	LINE LOCKUP PROTECTION:	10	2		80		
LH_2 RELIEF VALVE	47	RELIEF BYPASS	ORIFICED, SPRING, CLOSED	CRACK - 180 N/cm^2 (260 psia)	10	2		80		
				RESEAT - 172 N/cm^2 (250 psia)						
				FLOW - .02 kg/sec (.05 lb/sec)						
OVERBOARD VENT										
LH_2 HELIUM VENT VALVE	5	SAFETY - PROTECTION	SPRING LOADED, PLUS	CRACK - 180 N/cm^2 (260 psia)	20	2		160		20
		VENT THERMAL PRESSURE RISE	NC SOLENOID	RESEAT - 172 N/cm^2 (250 psia)						
		AT DEACTIVATION	OVERRIDE							
LOX REFILL VENT VALVE	24	VENT VAPOR	NC SOLENOID		20	20		1,600		20
LH_2 REFILL VENT VALVE	28	VENT VAPOR	NC SOLENOID		20	20		1,600		20
LOX REFILL VENT VALVE	60	DETECT LIQUID	DISCRETE SENSOR		20	20		1,600		.1
LH_2 REFILL POINT SENSOR	59	DETECT LIQUID	DISCRETE SENSOR		20	20		1,600		.1

9.3.5.7.1.6 Thruster Valves. All IAPS propulsion is controlled by the thruster select logic of the attitude control system, which generates open and close signals to each of the 32 thruster valves.

9.3.5.7.1.7 Exciters. Each thruster quad uses a single, internally redundant, electrical exciter which provides energy to the spark igniter. The exciters are controlled by the thruster select logic and fire all four igniters of a quad simultaneously.

9.3.5.7.1.8 Vent and Repressurization Relief Valves. Purge bag pressure is controlled by relief valves which are self-opening during Shuttle ascent. The valves latch at a pressure differential set point. During APS deactivation, the valves are closed by an electrical command from the Shuttle to allow controlled purge bag pressure through descent.

9.3.5.7.2 Remaining Control Components. The functional requirements and descriptive data for the components not previously covered are presented in Table 9-11.

10. INTEGRATED SYSTEM TECHNOLOGY DEVELOPMENT REQUIREMENTS

10.1 TECHNOLOGY DEVELOPMENT PLAN

The development plan, including supporting research and technology (SR&T), for the integrated auxiliary propulsion system is driven by the overall development plan and schedule for the Space Tug vehicle. The Tug vehicle development plan is as shown in Figure 9-1. Initial operating capability (IOC) is scheduled for early in 1984, requiring a five-year Phase C/D design and development period beginning mid-1978. Selection of the Tug APS concept to be implemented would have to be made during the Phase B study period.

SR&T effort would be necessary before any promising but unproven APS concept could be chosen for application to the Tug. This SR&T effort initially would be intended to confirm concept design estimates so that the concept would qualify for inclusion in Tug Phase B design activity commencing early in 1976. Subsequent SR&T activity would have goals similar to those for Tug elements which already have been selected in concept but still require technology development. The purpose of this later SR&T effort is to explore fully the functional/performance potential of the design concept and to provide a broader technology base that will pay off in reduced total vehicle DDT&E phase costs.

In this section, SR&T technical requirements are identified which are associated with the initial, concept-confirmation type of SR&T goals. The balance of the SR&T activity is considered to be of a follow-on nature and needs no specific identification at this time. However, estimated SR&T costs shown are based on completion of all SR&T goals.

The SR&T costs developed are extremely judgmental because the level of effort required to satisfy the goals as well as the precise nature of the technology which needs to be explored are subject to engineering appraisal. Cost estimates are nevertheless presented since they help define the extent of SR&T effort considered appropriate by Rockwell and participating subcontractors and, as such, are useful in concept selection evaluations. These costs are mean values obtained from estimates of the extreme (high/low) limits of the SR&T cost. To qualify each estimate, a measure of its uncertainty is obtained by computing an approximate standard deviation from the high and low values. These costs and the estimated duration of each SR&T item identified are shown in Table 10-1. It is considered that none of the technology development required is more elusive or formidable than the Tug improvements resulting from system application would warrant. All of the technology needed is based on firm prior art--at least in building-block form, if not as a unit.

Table 10-1. Integrated System Technology Development Requirements

IAPS Element	SR&T Objectives	Program Duration (mo.)	Program ROM Costs $\$10^6$ Low Est.	High Est.	Mean	One-Sigma Uncertainty
Thruster	Performance, life, and inlet requirements verification	16	1.0	2.6	1.8	0.27
Pumps	Functional and performance verification	9	0.3	0.5	0.4	0.04
Zero-g Reservoirs	Concept validation, functional and performance verification	12	0.4	1.2	0.8	0.13
Thermodynamic Control	Functional and performance verification	12	0.8	1.6	1.2	0.13
Total		·			4.0	.55

The following sections describe the background basis, SR&T objectives, key technology issues, test simulation method, and acceptability criteria for each APS element determined to require SR&T effort. These elements are:

- Thruster
- Pump
- Zero-g reservoir
- Thermodynamic control

Figure 10-1 presents an overall schedule for the completion of this SR&T program. The major test phases for each element have been sequenced to provide an orderly, building-block progression from basic materials and subcomponent investigations to ultimate concept conformation. The thermodynamic control tests, which require integration of a thermally representative segment of the IAPS, have been scheduled to maximize use of residual hardware from the three previous SR&T programs.

10.2 THRUSTER TECHNOLOGY

It is reasonable to expect that a liquid-liquid O/H APS thruster can be developed to meet the performance, weight, and reliability goals identified by this study. This confidence results from extensive industry oxygen-hydrogen thruster experience and, most particularly, ALRC experience with a liquid-liquid thruster.

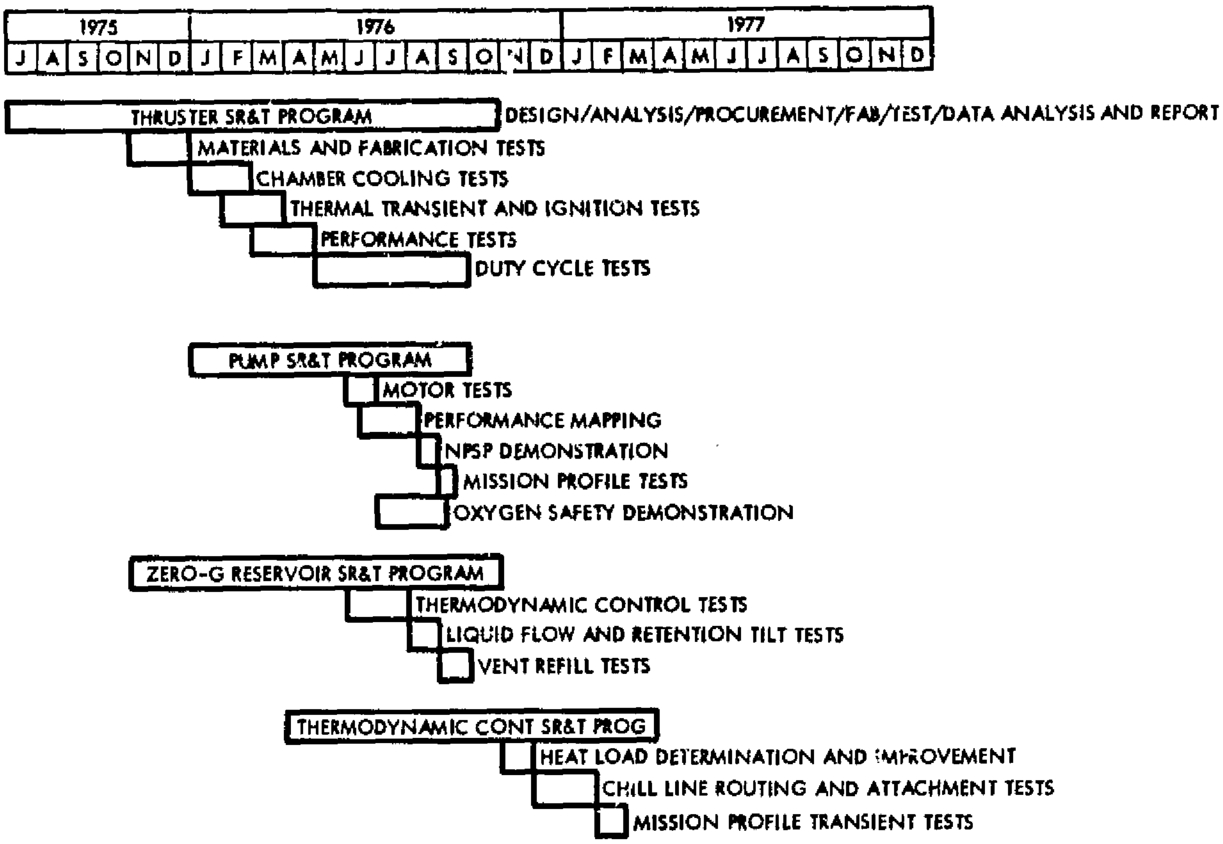

Figure 10-1. Integrated System Technology Development Schedule

BACKGROUND

ALRC recently completed a NASA LeRC contract on a liquid/liquid thruster, Extended Temperature Range (ETR) ACPS Thruster Investigation (Contract NAS3-16775). The objective of this contract was to establish the operational feasibility of an oxygen-hydrogen ACPS-type thruster for the Shuttle Orbiter in which cryogenic propellants are supplied to the thruster valves. Two temperature design points were investigated, resulting in LH_2/LO_2 injection for Design Point 1 and GH_2/LO_2 injection for Design Point 2:

Design Point	H_2 Temperature K (R)	O_2 Temperature K (R)
1	25 (45)	83 (150)
2	83 (150)	23 (150)

Two-flight-type thrusters were built and tested at a nominal thrust level of 5560 N (1250 lb) and a chamber pressure of 345 N/cm^2 (500 psia). This thrust level is far greater than required for the Tug IAPS. Moreover, no test experience exists at the 111-N (25-lb) thrust scale needed for Tug.

The torch igniter successfully used on this and other programs produced approximately 111 N (25 lb) by itself and formed the basis for this Tug APS study. Applying the igniter technology directly to a 111-N (25-lb) thrust engine involves two possible areas of concern: performance and durability. The performance of the igniter by itself is very low (approximately 60 percent of theoretical I_s) and the igniter has not been operated for long steady-state firings. Nevertheless, the igniter program tests and the analyses of this study form the basis for confidence in the thruster concept.

In adapting an igniter to an APS thruster, a change in design criteria is required. This may lead to significant design differences; in fact a design from scratch may be equally successful. Any design, of course, requires concept validation and performance verification, which is the purpose of this SR&T.

SR&T OBJECTIVES

The objectives of the thruster SR&T program are (1) to design, analyze, and obtain experimental data on a liquid-liquid O/H thruster designed to meet the Tug IAPS requirements, and (2) to resolve potential problem areas. A program of analysis, hardware design, and experimental evaluation is required. This overall thruster demonstration program should include rigorous analysis of the thermal management problems associated with the very cold temperatures encountered at the propellant valves, injector, and thruster manifolds; evaluation of the interaction of chamber cooling, performance, and injector design requirements; design and fabrication of injector chamber, igniter assembly, and associated thruster hardware; and experimental testing to evaluate the thruster over ranges of operating conditions and pulse mode duty cycles. During the design phase, attention shall be directed to thruster ignition requirements, combustion chamber cooling schemes, combustion stability, pulse mode limitations, propellant valve requirements, component material selections, and attainable specific impulse performance under both normal mode steady-state and pulse mode operating conditions, as well as for abort mode steady-state.

Another very important SR&T objective is to evolve a design that will minimize thruster DDT&E costs while still satisfying all performance requirements. The results of this study reveal that the thruster DDT&E costs are a significant portion (approximately 45 percent) of the DDT&E for the total integrated APS. Simplifications in manufacturing, assembly, and test operations that would not unduly compromise performance or reliability should be investigated in concept during thruster SR&T.

KEY TECHNOLOGY ISSUES

The key technology issues pertaining to an SR&T program for the IAPS thruster stem from the fact that no previous test experience exists for a liquid-liquid O/H thruster in the 111-N (25-lb) thrust range.

High specific impulse is an SR&T goal which will involve the design, analysis, and experimental evaluation of possible alternatives to the baseline sleeved chamber cooling concept such as partially regenerative or internally regenerative cooling of the outer wall, or slot cooling along the axial length. Performance and film cooling requirements for the current design depends heavily on the rate at which the coolant hydrogen mixes with the core gases. For performance to be reasonably high, mixing must be complete before the core gases reach the throat. This can result in a relatively long and heavy thrust chamber and high temperatures in the outermost coolant channel, thus reducing the throat film cooling effectiveness.

Thrust chamber length and performance may be improved by one of the alternate cooling concepts mentioned above or by providing a splash plate or similar mixing device at the outlet of the innermost coolant sleeve. Testing will require a means of varying the cooling design and mixing length until optimum performance is achieved.

The selection of suitable materials and manufacturing processes compatible with the high temperatures and small size of the IAPS thruster is another key technology issue. Selection of the basic thrust chamber alloy, close tolerance machining processes, and protective coating application are all interrelated problems that will affect thruster durability and production cost. Axial and radial thermal gradents are severe. Estimates of external temperature range from approximately 28 K (-410 F) at the igniter, or chamber inlet, to near 164 K (2500 F) at the throat. Temperature gradients may be as high as 110 K/cm (500 F/in.). Thermal stresses in the concentric sleeves making up the coolant flow channels could result in cracking of any coating or base metal at points of high stress concentrations.

Thermal transient effects also are a key issue for the technology development of this thruster. Pulse mode thrust profiles after both hot and cold starts must be defined and their acceptability established. Proper ignition after extended cold soak periods with liquids at the valve inlets also must be verified. It is expected that these issues can be resolved by a series of variable duty cycle tests followed by correction of differences, if any, as they occur.

TESTING APPROACH

The thruster SR&T objectives can be satisfied by first conducting precursor experiments in support of materials selection analysis and manufacturing process evaluations. These tests will support the selection of the basic thrust chamber alloy and protective coating materials, and will verify acceptable machining techniques and reproducible dimensional tolerances.

Chamber cooling tests should then be conducted utilizing typical, but not flight-type valves, manifolds, and ignition system. A bolt-on thrust chamber assembly could be used so that alternate cooling designs may be easily installed and tested. This would permit a parametric approach to the problem of optimizing thrust chamber length, cooling configuration, and performance.

Thermal transient and cold ignition tests should be accomplished using the optimized thrust chamber from the previous test series. Various manifold lengths and internal platelet or other insulator designs should be installed and tested with different injector configurations, ignitor locations, and support methods. Propellant thermodynamic state at the valve inlets should be varied to simulate the range of heat soakback expected over an actual mission duty cycle. The produceability, potential DDT&E costs, and compatibility with future development tests of each configuration must be thoroughly evaluated.

Performance tests of the most favorable arrangement of valves, manifolds, injector, and igniter should be run to determine steady-state and transient thrust, specific impulse, and mixture ratio as a function of valve inlet pressure and temperature. The limits of performance should be explored and the capability for inlet pressure controlled mixture ratio shift as required for Tug abort recovery must be verified.

Finally, mission profile duty cycle tests should be conducted. Satisfactory start transient and minimum impulse bit characteristics should be verified. Sufficient extended duration and repetitive cyclic firing tests should be conducted to demonstrate the durability of this thruster design concept.

ACCEPTABILITY CRITERIA

Satisfactory thruster performance and durability are major criteria that must be demonstrated prior to selection of a liquid-liquid O/H APS for use on the Space Tug. To do this, the SR&T test program must provide reasonable confidence that the weight, length, thrust, specific impulse, minimum impulse bit, inlet condition sensitivity, and abort mode requirements specified in Section 9 can be achieved. Prior to initiation of performance and duty cycle tests, the precursor materials and fabrication, thrust chamber cooling, and thermal transient and ignition tests must have led to satisfactory solutions of the problems previously identified. Following IAPS thermodynamic control SR&T tests, thruster qualification development testing can be initiated as a part of the Tug Phase C/D vehicle development program.

10.3 PUMP TECHNOLOGY

Neither oxygen nor hydrogen pumps currently exist for the IAPS since the pressure, flow, and suction requirements are unique. In particular, the flow rates are orders of magnitude smaller than that of flight-weight pumps previously developed for O/H rocket engines. The pressure requirement is moderate but high enough to demand special design provisions with positive displacement machines. The low suppression head suction requirement has been satisfied previously but only for high-flow centrifugal/axial rocket engine pumps. However, it is considered that the IAPS pump SR&T involves relatively straightforward development of custom designs using well-established concepts scaled from other applications.

BACKGROUND

The pumping of cryogens can, in many respects, be considered a mature technology. Large turbo-machinery goes back as early as the 1940's for the German-built V-2 liquid oxygen and alcohol propelled rocket. Smaller, electric motor-driven, cryogenic pumps were developed later. In the period between 1956 and 1959, the Pesco Products Division of Borg-Warner Corporation designed, fabricated, and tested an electric motor driven liquid hydrogen centrifugal pump for the U.S. Government under Contract AF 18(660)1658. During this program, the feasibility of wet running ac motors submerged in LH_2 was proven, and rolling contact bearings for use in cryogenic pumps were shown to be practical. Since that time, many electric motor-driven pumps have been built for a variety of applications including a two-stage liquid hydrogen boost pump for destratification studies under Contract AF 33(1616)5810, and liquid hydrogen and oxygen recirculation pumps for the Saturn S-II and S-IVB stages under Contracts NAS7-200 and NAS7-101. Although small compared to the turbo-pumps used for rocket engines, these electric motor-driven pumps are still large when compared to the IAPS requirements: .113 m^3/sec (30 gpm) for the S-IVB LOX recirculation pump as compared to 0.00454 m^3/sec (1.2 gpm) for the LOX IAPS feed pump. This low flow coupled with the moderate head rise required results in specific speed values that fall in the regime of positive displacement pumps.

No flight-qualified positive displacement pumps have yet been built. AIRCO and others have estensive experience with both hydrogen and oxygen stationary piston pumps for high-pressure transfer service and other cryogenic applications. These pumps have demonstrated high efficiency, long life, and good reliability but tend to be heavy due to their low operating speeds.

Sundstrand Corporation has some applicable vane pump experience developed as a part of a General Electric Company contract for NASA LeRC, Final Pumping System Liquid Hydrogen/Liquid Methane, J85 Control System. This pump operated at 0.098 m^3/sec (26 gpm) and 186 N/cm^2 (270 psig). Demonstrated volumetric efficiency was low, however, and problems remain to be solved to reduce internal leakage. A pressure-loaded balanced vane concept is currently being studied by Sundstrand to improve efficiency.

SR&T OBJECTIVE

The objective of the pump SR&T program is to verify by empirical data that the flow, pressure, efficiency, and weight requirements specified in Section 9 can be satisfied by practical and inexpensive LOX and LH_2 IAPS feed pumps. This objective involves substantiating zero NPSH and momentary two-phase flow capability, supporting material selection studies, verifying cryogenic clearances, and demonstrating oxygen safety. It is considered that satisfaction of these objectives will require LOX and LH_2 flow tests of full-scale prototype pumps.

Although a piston pump with a centrifugal boost stage was selected as baseline for the preliminary design of this study, other pump concepts such as a vane with boost stage, a low-speed piston without a boost stage, or one with a higher speed boost stage cannot, at this point, be discounted. As discussed in Section 11, both vane and piston-type pumps are considered competitive alternative concepts based on the limited available test data and experience. Thus, the selection of the basic pump type becomes an objective of this SR&T program.

The funding level estimated for pump SR&T refers to single-source procurement. However, it is considered that two competitive programs, one for a vane type and one for a reciprocating type, may be warranted. If two programs are not possible, pump SR&T procurement action could be aimed at obtaining a firmer basis for pump type selection by making the pump type an option in a competitive bid which calls for substantiation of the type selected by the bidder.

KEY TECHNOLOGY ISSUES

The basic technology issues requiring SR&T effort for the IPAS feed pumps stem from the fact that experience at the low flow rates and suction conditions required for this system are practically nonexistant. The development of a flight-type positive displacement cryogenic pump leads to problems not supported by the previously well-developed axial or centrifugal dynamic pump designs. The requirement for high efficiency requires special attention to internal leakage paths, close tolerance clearances at cryogenic temperatures, and material thermal contraction and dynamic sealing properties.

The requirements for zero NPSH and momentary operation with vapor ingestion at reduced flow rates also introduces design and technology problems not previously addressed for small pumps. Vane pumps would probably require a centrifugal boost stage while piston pumps could possibly operate at low enough speeds to avoid excessive cavitation. In either case, detailed thermodynamic/fluid-dynamic analysis and subsequent test verification will be required.

An important SR&T issue is the problem of oxygen safety. A thorough pump failure mode analysis must be made and preventative measures incorporated in the basic pump design where possible. Some of the failure modes previously considered that could potentially lead to an oxygen fire or explosion include mechanical rubbing, mechanical impact, abrasion due to fluid-born particles, or electric arcing due to a short circuit. A comprehensive treatment of oxygen pumping safety is discussed in Reference 28. It is concluded that an entirely safe oxygen pump can be designed and demonstrated by induced failure mode testing. Safety design features that should be considered include a hermetically sealed stator cavity, triple redundant winding and connector insulation, underspeed or overload sensing safe shutdown interlocks, and the use of high-conductivity materials for dissipating localized frictional heating caused by particle ingestion or bearing failures. Material flammability and LOX compatibility limits must, of course, be carefully evaluated.

Another unique feature of the IAPS pumps is the requirement for speed control to modulate flow during extended delta velocity maneuvers and to change the flow rate for a shift in thruster mixture ratio in the event of MPS abort recovery by the IAPS. SR&T testing is required to verify proper response of the pump/motor performance to variations in electrical input frequency.

Other SR&T issues include establishment or verification of the motor/pump assembly performance, power drain, weight, structural integrity, external leakage, and start and stop transient characteristics.

TESTING APPROACH

Full-scale prototype LO and LH_2 pumps will be designed, fabricated, and tested in liquid oxygen and hydrogen at design flow rates and pressures. Power supply at variable frequency will be from a nonflight-type ground test support unit. Separate electrical motor dynamometer testing will be completed first. Cryogenic fit and clearance checkout would then be accomplished, followed by design nominal pump performance tests and performance mapping. These tests might best be accomplished using special test devices simulating the variable fluid and pressure capacitance characteristics of the downstream IAPS accumulator and the upstream inertia and flow loss. Propellant inlet pressure, temperature, and quality limits would then be explored to develop performance loss characteristics and recovery times. Prototype pump tests would be completed by following a typical worst-case pump timeline for a Tug mission profile.

A separate oxygen unit is anticipated for induced failure mode testing and oxygen safety analysis. This pump will incorporate all the anticipated basic design safety features, but also will include the capability to simulate physically various failure modes such as a bearing failure, locked rotor, or particle ingestion. The unit will be operated in oxygen until failure occurs, after which it will be disassembled and the damage evaluated.

ACCEPTABILITY CRITERIA

This SR&T program must demonstrate that the design and performance criteria specified in Section 9 can be achieved for both the LOX and LH_2 IAPS feed pumps. Any deviation greater than approximately 10 percent will require re-evaluation of the design on a system basis. Of particular interest is the ability of the pump to operate with a two-phase fluid at its inlet. A momentary reduction in flow and efficiency is not as critical as the pump's ability to expel the gas and recover nominal performance.

The induced failure mode tests must verify satisfactory performance of the pump fail-safe design features and demonstrate that pump operation in the IAPS oxygen system does not constitute a hazard to the safety of the Tug vehicle, its payload, or the Space Shuttle and its crew.

10.4 ZERO-G RESERVOIR TECHNOLOGY

SR&T requirements for the zero-g reservoir are dictated primarily by the internal capillary devices. Although similar devices are operational with storable propellant, no flight experience exists for cryogenic propellants. However, theoretical analyses, fluid properties research, negative one-g tests, and scale-model drop and pushover flight tests (short duration zero-g tests), accomplished in a series of research programs carried out for more than a decade, have placed cryogenic capillary device technology on a sound basis. The approach to SR&T for the zero-g reservoir is to apply these well-developed simulation and experimental analysis techniques to a full-scale IAPS prototype design. No technology advance is needed, but concept confirmation is required. This section presents the technological background, SR&T objectives, key technology issues, test simulation method, and acceptability criteria for the zero-g reservoir SR&T.

BACKGROUND

References 29 through 34 present the results of some of the more recent study and test programs dealing with design and operation of the zero-g reservoir. The Reference 29 test program demonstrated in 1972 the feasibility of long-term cryogenic storage through the use of multilayer insulation, low-conductivity supports, and heat load interception utilizing a shield cooled by the vented cryogen vapor. This program showed that by close attention to design detail and proper selection of materials, an effective thermal isolation and heat rejection system could be fabricated, assembled, and operated for long periods (48 days).

Early in 1973 tests were conducted by Rockwell (Reference 30) to explore the problems associated with start and shutdown transient flow in a typical capillary device used for feedout from a propellant tank in a zero- or low-gravity environment. The effect of significant variables such as pressure, flow rate, line length, and valve actuating time was determined. It was found that the shutdown surge had no adverse effect on the capillary device tested. However, under some conditions, start flow transients can cause gas ingestion into the capillary device. Several methods for reducing or eliminating start transient gas ingestion were successfully tested.

Later in 1973, an experimental program was performed by McDonnell Douglas Astronautics Company under the direction of NASA LeRC to determine the feasibility of integrating an internal thermodynamic vent system and a full wall-screen liner for the orbital storage and transfer of liquid hydrogen. The results of this program are reported in Reference 31. The annulus formed by the screen and tank wall was used to provide a flow path for pumped LH_2 which absorbed tank incident heating and then rejected this heat to the thermodynamic vent system. Ten screens were selected for test after a comprehensive screen survey. The experimental results measured screen bubble point, flow-through pressure loss, and pressure loss along rectangular channels lined with screens on one side using LH_2 saturated at 34.5 N/cm^2 (50 psia). The study demonstrated this concept to be fluid-dynamically feasible and, although somewhat different from the design proposed for the IAPS reservoir, resulted in basic screen and channel data supportive of the design of an SR&T test article.

Additional experimental testing was accomplished by McDonnell Douglas Astronautics Company to develop capillary system design characteristics in (1) basic surface tension screen performance, (2) screen acquisition device fabrication methods, and (3) screen surface tension device operational failure modes. These data are presented in Reference 32. Various screen materials, screen joints, screen support structures, and sealing devices were fabricated and tested. Correlation techniques were developed for predicting screen LH_2 bubble points using isopropyl alcohol as a test media. The criticality of screen vibration and direct heating of a screen retaining a cryogenic fluid also were demonstrated.

Reference 33 presents the results of a 1974 program involving the design, fabrication, and test of a multipurpose full-scale liquid hydrogen acquisition and thermal control system for inclusion in a NASA/MSFC auxiliary propulsion system breadboard. This design differs from the proposed Tug IAPS zero-g reservoir in that feedout is accomplished with the reservoir pressure isolated from the main propellant tank. However, two features common to the Tug IAPS design performed satisfactorily: (1) use of a tank wall-mounted heat exchanger system for heat load interception, and (2) overboard venting for reservoir refill from acceleration-settled propellant in the main tank.

The tank wall-mounted heat exchanger system also has been experimentally evaluated by Lockheed Missiles and Space Company as described in Reference 34, dated January 1975. During this study analytical models were developed describing the heat and mass transfer and energy distribution in the contents of a cryogenic propellant tank under varying gravity fields. Pressure and temperature histories were computed for tanks ranging in size from 1.2 to 6.8 m (4 to 22.5 ft) in diameter and gravity levels from 0 to 1.0. Results of subscale testing utilizing both cryogenic and noncryogenic fluids compared well with the analytical models, and demonstrated that a tank wall-mounted heat exchanger can effectively control tank pressure and propellant temperature.

Previous technology studies have shown the potential criticality of vibration, warm gas exposure, and feed system startup and shutdown fluid dynamics on the performance of capillary screen acquisition systems. For this reason, the Reference 35 experimental study has been sponsored by NASA/MSFC. Some cryogenic testing has been completed and data evaluation is underway. Assessment of the progress to date indicates that the results of this study will be beneficial to an SR&T program for the IAPS zero-g reservoir.

SR&T OBJECTIVES

The objective of SR&T for the zero-g reservoir is to apply previously well-developed simulation and experimental analysis techniques to a full-scale prototype design. No technology advance is needed, but concept confirmation is required. The experimental program will take the form of prototype hardware subjected to operational thermal conditions and negative one-g demonstration of retention and feedout capability. The test program is considered to involve validation of thermal control aspects as much as capillary phenomena influences. This is because the two factors are intimately related. Capillary device performance is tied to the design's ability to avoid vapor formation and retain subcooling at critical points within the device and in the operational cycle.

KEY TECHNOLOGY ISSUES

One of the major technology issues requiring SR&T experimental evaluation is propellant refill of the zero-g reservoir by the venting of vapor to space. The vent line will probably require a flow restrictor so that during steady flow, reservoir pressure does not drop enough to cause propellant flashing. Considering the pressure ratio involved, this is not an easy problem to evaluate. During flow start transients, the pressure in the reservoir can drop sharply, causing some propellant flashing. As long as the flashing does not occur within the capillary collectors, the loss of propellant is small and effects inconsequential. A motorized vent valve to control rate and magnitude of pressure changes within the reservoir may be required. To assure that liquid is not vented to space, a liquid sensor at the top of the reservoir, which activates the vent valve, is provided. Propellant sloshing due to the low acceleration from APS thrust or splashing due to liquid boiling may wet the sensor and could cause premature valve closure if appropriate design provisions are not explored in SR&T. Majority vote sensors could be considered and evaluated.

Another problem which must be faced during refill is thermal and thermodynamic state control. The contents of the reservoir are subcooled with respect to the propellant in the main tank due to the cooling of the thermodynamic vent system. The introduction of large quantities of warmer liquid during refill temporarily overloads the thermodynamic vent system. Several hours are required to cool the reservoir's contents to steady-state level. The thermal isolation and thermodynamic cooling systems must be designed to prevent boiling of propellant within the collector tubes during this period. SR&T testing analysis will be necessary to define the map of acceptable pressure transients and operating conditions, including the required degree of subcooling, ratio of incoming to resident propellant, and vent valve and liquid sensor characteristics.

Additional technology issues that must be addressed during a zero-g reservoir SR&T program are:

1. Liquid retention during adverse acceleration maneuvers.

2. Screen drying due to localized heat transfer and screen vibration.

3. Fluid flow startup and shutdown transient effects on screen bubble pressure stability.

4. Fluid temperature stratification.

5. Liquid subcooling to satisfy pump inlet requirements and to avoid vapor entrapment in capillary compartments.

6. Cooling coil attachment design and assembly techniques.

7. Screen installation, support, and joint sealing methods.

TESTING APPROACH

Since the IAPS zero-g reservoirs are relatively small--51 cm (20 in.) and 25.5 cm (10 in.) in diameter for the LH_2 and LOX reservoirs, respectively--full-scale prototype SR&T tests are recommended for each. Although test results using storable fluids could be used to predict the fluid-dynamic behavior of the IAPS propellants, cryogenic hydrogen and oxygen tests are recommended to provide an adequate understanding of the combined fluid-dynamics and thermodynamic effects in a thermal environment representative of on-orbit operation.

The test articles should be designed for easy disassembly to facilitate testing of various capillary screen materials and internal acquisition system geometries. Testing will be conducted in a vacuum chamber with provisions for varying the reservoir heat load, coolant vent flow rate, entering and exiting propellant flow rates, internal liquid level, propellant inlet temperature, refill vent pressure, and refill vent flow rate. The test support fixture should be designed to rotate the reservoir through 180 degrees between tests to allow tilt tests ranging from plus one g to minus one g. The cooling coil attachment and thermodynamic vent flow path should be designed for ease of modification to allow for temperature and subcooling optimization and proper cooling of heat shorts. A thorough analytical math model and a comprehensive network of propellant temperature measurements would be required to extrapolate one-g temperature profile data to a predicted zero-g temperature distribution. Preliminary math model results should be used to establish the proper combination of test control conditions and range of parameter variations in order to gather the maximum amount of useful data with the minimum number of test runs. It is expected that variation of heat loads, coolant flow rates, and coolant flow routing will require the most extensive testing because of the need to extrapolate propellant temperature profile data to a zero-g environment.

ACCEPTABILITY CRITERIA

Correlation of the zero-g reservoir SR&T results must show that vapor-free propellant can be supplied to the feed pump during zero-g coast and during all APS or MPS maneuvers. Short transient periods of vapor ingestion may be acceptable if they are within the performance limitations of the downstream pump and accumulator combination. Such conditions would require evaluation as a part of the pump SR&T program. The reservoir SR&T goal is to preclude any such vapor ingestion transients, however, and thus allocate any two-phase pump capability to system performance margin.

Demonstration of acceptable component dry weight and coolant flow expenditure also are important SR&T goals. An increase of 10 percent or more in either of these parameters above that specified in Section 9 will require a re-evaluation of the IAPS design on a systems level.

10.5 THERMODYNAMIC CONTROL

The design of the thermodynamic control system for the Tug IAPS will play a key role in the success or failure of the proposed concept. It is principly the need for thermodynamic control that sets the cryogenic APS apart from competing storable propellant systems. Thermodynamic control of the oxidizer and fuel inlet temperatures must be provided to supply propellant at the thermodynamic states required for satisfactory thruster operation. Liquid subcooling must be achieved within screened compartments to preclude vapor disruption of the propellant acquisition capillary devices. Excessive formation of vapor at the feed pump inlet could cause inadequate pump flow and efficiency.

To verify that these problems have been adequately reflected in the specifications for primary IAPS elements (thruster, reservoir, pump) and are properly resolved in the design of the thermodynamic control system, the following SR&T program has been identified. It should be noted that the APS will use the same insulation and thermal isolation support concepts developed for the Tug main propulsion system. While thermal isolation performance will be evaluated, further development of this technology will not be a primary objective of this SR&T program.

BACKGROUND

Cryogenic thermal control technology is as old as the study of cryogenic fluids. Not until cryogens were considered for extended space application, however, did the problem of zero-g venting appear. Studies in this area have been conducted as early as 1966 when General Dynamics under Contract NAS8-20146 evaluated four different vent system liquid/vapor separation methods: (1) heat exchange, where the vent fluid is throttled to a low pressure and temperature and allowed to exchange heat with the tank fluid to vaporize any liquid initially present in the vent; (2) mechanical, employing a rotating element for centrifugal separation; (3) dielectrophoresis, utilizing an electric field to separate liquid from vapor (both total liquid control and local separator devices were considered); and (4) surface tension, utilizing fluid surface forces to orient the liquid in a tank with baffles or screens, or to effect local separation at the vent. The results of this study are documented in Reference 36. Selection criteria included system weight, vent losses, power required, and reliability. It was concluded that the heat exchange system was the most promising, with the mechanical separator a close second.

In 1967, a prototype zero-g hydrogen vent system using the heat exchanger principle was designed, fabricated, and tested by General Dynamics under Contract NAS8-20146. The results of this test program are presented in Reference 37. During the design phase, tradeoffs were made to determine the type of heat exchanger (bulk versus wall), type of pump drive (electric versus turbine), optimum vent flow rate, vent cycle, and fluid mixing criteria.

An internal heat exchanger and electric bulk mixing pump were selected for test in a 1.0-m (40-in.) diameter by 2.3-in. (89-in.) long hydrogen tank. Intermittent operation was used to minimize total pump power input to the tank. The pump provides forced convection heat exchanger flow as well as liquid temperature destratification. Deactuation and actuation of vent and pump were controlled by a pressure switch. Various liquid and ullage mixing modes and mixing flow directions were tested. Liquid mising flow directed at the liquid-vapor interface was found best for tank pressure control and minimum vent losses. Similar tests were conducted in 1968 by the Lockheed Missiles and Space Company using an internal mixer and heat exchanger in a larger hydrogen tank--2.8-m (110-in.) diameter oblate spheroid--with similar results. These tests are reported in Reference 38. A zero-g thermodynamic vent system (TVS) using a tank wall-mounted heat exchanger was tested as a part of a cryogenic fluid storage and expulsion subsystem built by McDonnell Douglas Astronautics Company in 1974. This test program is documented in Reference 33. The specific objectives were to demonstrate (1) both steady-state and transient operation of the TVS, (2) control and predictability of the vented flow rate, (3) that the TVS could provide coolant flow for additional hardware such as feedlines and turbopumps, and (4) the fabricability of the wall-mounted heat exchanger subsystem.

To meet these objectives, the TVS was configured with a number of flow-control orifices in parallel, which gave a vent flow rate variable from 0.09 to 2.3 kg/hr (0.2 to 5 lb/hr). Bypass flow also was provided for turbopump or feedline cooling. The TVS heat exchanger coils on the tank were configured to provide a number of alternate flow paths. It was found that successful heat interception operation of the TVS required that the wall-mounted heat exchanger b insulated from the external main tank fluid (to prevent condensatior and loss of cooling capacity). This condition is not applicable to the Tug I/ S design since the cooling lines are all external to the main tanks.

Vent system fabricability was fully verified. Both dip-brazing and epoxy bonding techniques were used successfully. Vent mass flow rate versus pressure drop data were measured for various orifice and flow path combinations. Complete heat flux interception was demonstrated during steady-state thermal control by maintaining constant vent flow rate at the proper conditions.

During the same period, the tank wall-mounted heat exchanger vent concept was evaluated by Lockheed Missiles and Space Company. Results of this experimental program are documented in Reference 34. Testing of LH_2, LH_2, oil, Freon, and water were conducted using a 0.56-m (22-in.) diameter spherical tank for the noncryogenic fluids. Test data were obtained for various ullage volumes, heat loads, fluid temperatures, and vent flow rates, and compared with analytical models. Both the model and test results indicate that a passive tank wall-mounted heat exchanger can effectively control tank pressure and propellant temperature.

SR&T OBJECTIVES

The principal objective of this SR&T program is to verify the satisfactory operation of the proposed thermodynamic control concept. Satisfaction of this objective will require thermal-vacuum test of a prototype system including thermally representative segments of the entire APS. Thermal interactions between the zero-g reservoir, pump, accumulator, thrusters, and interconnecting manifolds will have a strong influence on individual component and system performance and must be verified experimentally. The test setup will use prototype hardware available from the previous component SR&T programs. Satisfactory temperature, pressure, and flow rate control must be demonstrated for both transient and steady-state thruster operation. System activation and deactivation using representative on-orbit sequences will be simulated. Techniques for improving system performance and reducing coolant consumption will be identified and experimentally evaluated. Satisfactory control of the thruster propellant inlet conditions within the required limits as specific in Section 9 will be demonstrated.

KEY TECHNOLOGY ISSUES

Because of the criticality of weight and reliability, thermodynamic control of the proposed IAPS relies heavily on passive design techniques. Multilayer insulation and low-conductivity supports are used to minimize tankage and feedline heat leaks. Series flow coolant coils with only two on-off flow control valves and one variable power heater are used both to reject the heat load and to control all 16 thruster propellant inlet temperatures for both the LOX and LH_2 storage and feed systems. Passive tank wall-mounted cooling coils are employed for thermally conditioning the propellant tanks rather than active bulk mixers and internal heat exchangers. Thus, system temperatures and propellant thermodynamic state throughout the IAPS will depend on the balance of heat leak distributions and conduction paths between the propellant and cooling fluid.

Satisfactory thruster, pump, and capillary system temperature requirements can be assured by controlling to the warmest allowable measured temperature. Efficient use of the expendable coolant, however, will depend on minimizing the special variation in propellant temperatures by proper design and allocation of the system thermal resistances. Analytical predictions will require experimental verification. It is expected that design modifications such as increasing or decreasing the spacing of cooling tube saddleblocks or rerouting tank wall-mounted coils will be required as the result of this testing. It is these adjustments and subsequent verification of efficient and proper system temperature control that is the key issue of this SR&T program.

TESTING APPROACH

SR&T requirements for the thermodynamic control system will be satisfied by thermal-vacuum testing of a prototype system including thermally representative semgnets of both the LOX and LH_2 portions of the entire IAPS. The test specimen will include the zero-g reservoir, pump, accumulator, a single thruster, and a segment of the interconnecting manifold and cooling lines. Prototype hardware from the previous SR&T programs will be used where available. The system will be installed in a vacuum chamber and tested at approximately 0.133 N/cm^2 (10^{-5} torr) to assure insulation performance representative of a space environment. Thruster exhaust will be to sea level pressure

The test apparatus will be designed for ease in varying cooling coil routing, attachment location, and saddleblock spacing. The test sequence will follow a representative Tug operational profile including initial IAPS activation on-orbit, system chilldown, reservoir vent refill, attitude control pulsing, APS delta velocity maneuver, quiescent coast, and system deactivation and safing. System pressures, temperatures, and flow rates will be measured and the thermal control system modified as required to meet temperature requirements and minimize hydrogen coolant consumption. Although not a prime objective of this SR&T program, insulation performance will be monitored and potential improvements such as additional coverage or better design of penetrations or joints will be considered.

ACCEPTABILITY CRITERIA

SR&T testing and analysis of the IAPS thermodynamic control system must demonstrate proper propellant temperature control as required by the thrusters, feed pump, and zero-g reservoir capillary devices. These requirements are specified in Section 9. Hydrogen coolant consumption in excess of 50 percent above that predicted will require re-evaluation of the IAPS at a system level.

11. INTEGRATED SYSTEM PRELIMINARY DESIGN ANALYSES

The IAPS preliminary design is defined in terms of requirements and primary characteristics in Section 9 (Integrated System Preliminary Design Specification). Additional descriptive information, supplementary to the preliminary design definition, is contained in Sections 5.2 and 5.3 in discussions of Concept I-5. Most of those discussions are applicable, except where changes were introduced during preliminary design. To provide design change traceability, this section summarizes those changes. Subsequent sections present design analyses for the principal issues resolved in preliminary design.

11.1 PRELIMINARY DESIGN CHANGE SUMMARY

The changes resulting from preliminary design caused a substantial payload performance increase: 40 kg (88 lb) and 185 kg (408 lb) for Mission Profiles A and B, respectively. In Mission Profile B, most of this was due to the effective specific impulse gain occasioned by the change to a zero cant angle. For Mission Profile A, this effect was less important, the system dry weight reduction of 18 kg (41 lb) being the principal contributor. With respect to earlier (conceptual phase) comparison with storables, the relative position of the IAPS was enhanced; however, the cant angle change and possibly most of the valve weight reductions apply equally to the storables. For this reason, the conceptual phase comparisons are to be considered the only valid ones made on a common basis. It can be noted, however, that the cant angle change has the greatest favorable impact on the IAPS in Mission Profile B, which cannot be performed by the storable systems.

Process/mechanical diagram changes were as follows:

1. The hydrogen bleed tapoff was moved from the high-pressure feed line to the low-pressure reservoir outlet. This improves thermal performance, reduces the pressure ratio of the Joule Thompson expander, and minimizes the size of the hydrogen accumulator. This change also requires the bleed circuit shutoff valves to act as ascent isolation valves.

2. The oxygen accumulator pressure vessel wall was determined to be at minimum gauge at the pressure corresponding to its gas at ambient temperature. To eliminate components and simplify activation, the unit is provided with a ground servicing (ambient) helium disconnect with a closure manually sealed after loading.

3. For preliminary design purposes, the fuel cell-supplied power option was selected. The battery-powered option is still a competitive alternate to be considered in Tug Phase B.

4. An additional high flow rate JT expander and shutoff valve was added in parallel to the normal flow expander. This parallel path is required to accommodate the transient chilldown heat load prior to IAPS activation.

The conceptual phase solenoid valve weights were conservatively estimated. In preliminary design, these were redefined and reduced according to the following criteria:

Reservoir pressure valves	0.36 kg (0.8 lb)
Feed pressure valves	0.45 kg (1.0 lb)
MPS helium pressure valve	0.54 kg (1.2 lb)

The additional weight changes resulting from preliminary design refinements are listed in Table 11-1. The total of all refinements results in a weight decrease of 18.5 kg (40.9 lb).

In accordance with conceptual phase conclusions and recommendations, the IAPS thruster quad was moved to the Tug aft skirt from the baseline midship location. This change involves vehicle integration aspects too extensive to be resolved within the scope of this subsystem study. LOX tank diameter and shape, primary structure (particularly the aft docking ring), and the Orbiter attachment are all involved. Since the necessity for this change applies to both storable and integrated APS, the resulting vehicle impact is of no consequence to the APS concept comparisons of this study. A design solution applicable to this change has been worked out with acceptable results for an earlier vehicle version with even larger main tanks (Reference 39). The solution involves a tapered aft outer shell and is shown in Figure 11-1.

11.2 PUMP TYPE SELECTION

To further the design analysis of the conceptual phase, (Section 5.3) preliminary design phase activity centered on making a choice between vane and reciprocating pumps. Data received from Sundstrand Corporation covered both types. Part of these data are presented in Figures 9-10, 11-2, and 11-3, and in Table 11-2, where preliminary designs and characteristics of both types of oxidizer and fuel pumps are presented. Although considerable engineering, mostly by Sundstrand, has been accomplished substantially more design development and test data are necessary to make a firm evaluation. Based on the limited information developed, a pump type selection for the preliminary design purposes of this study was made and is described in this section.

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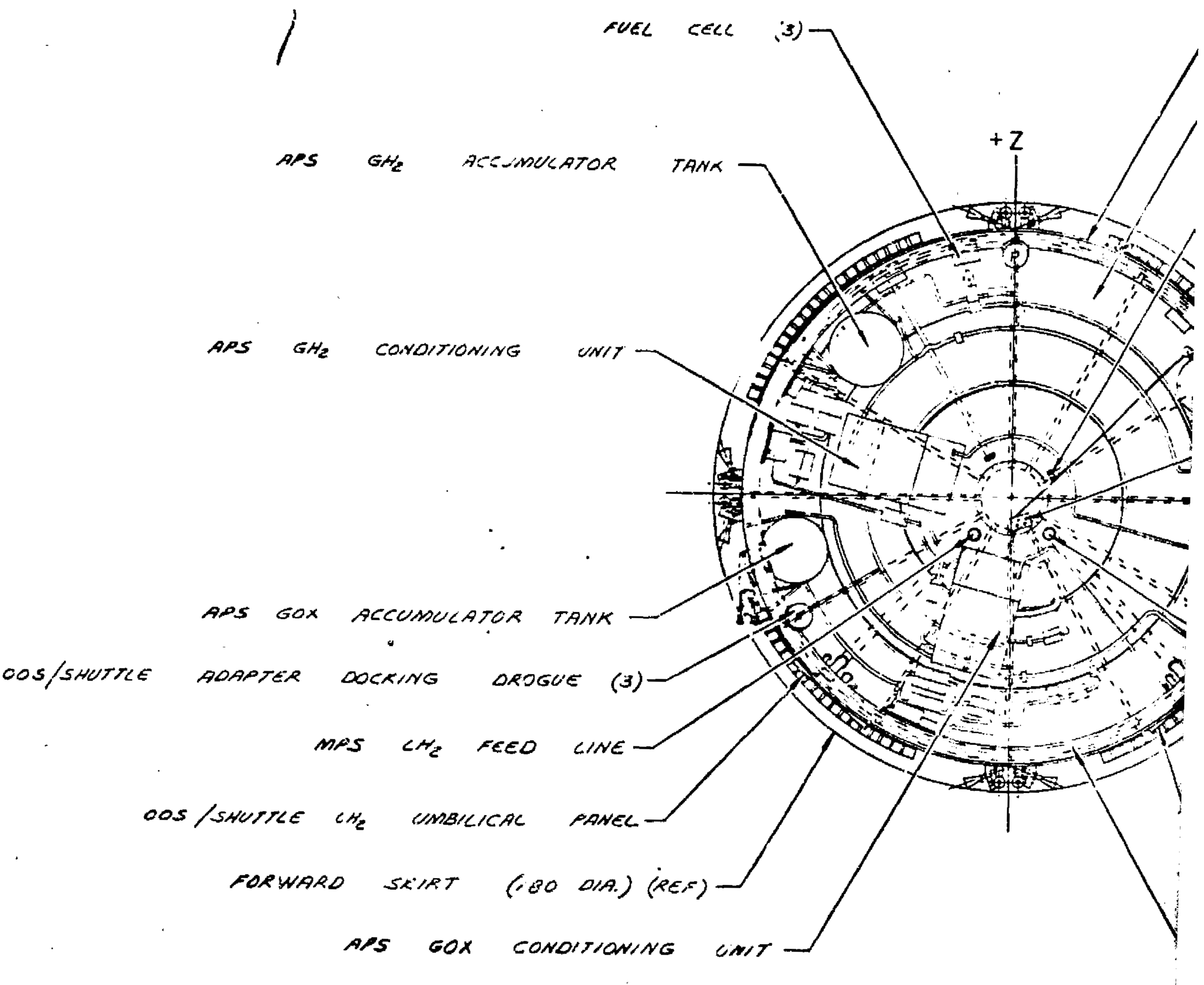

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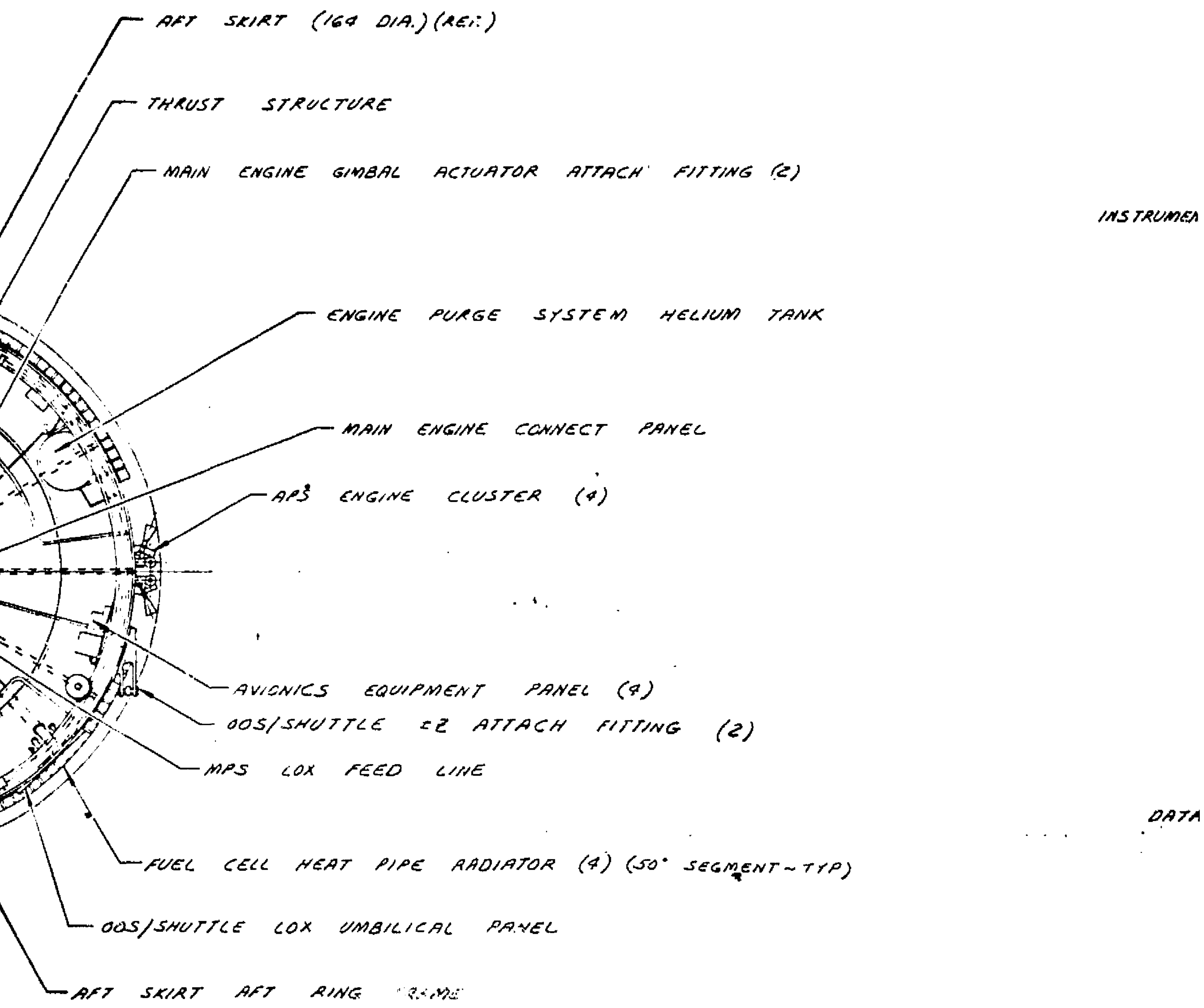
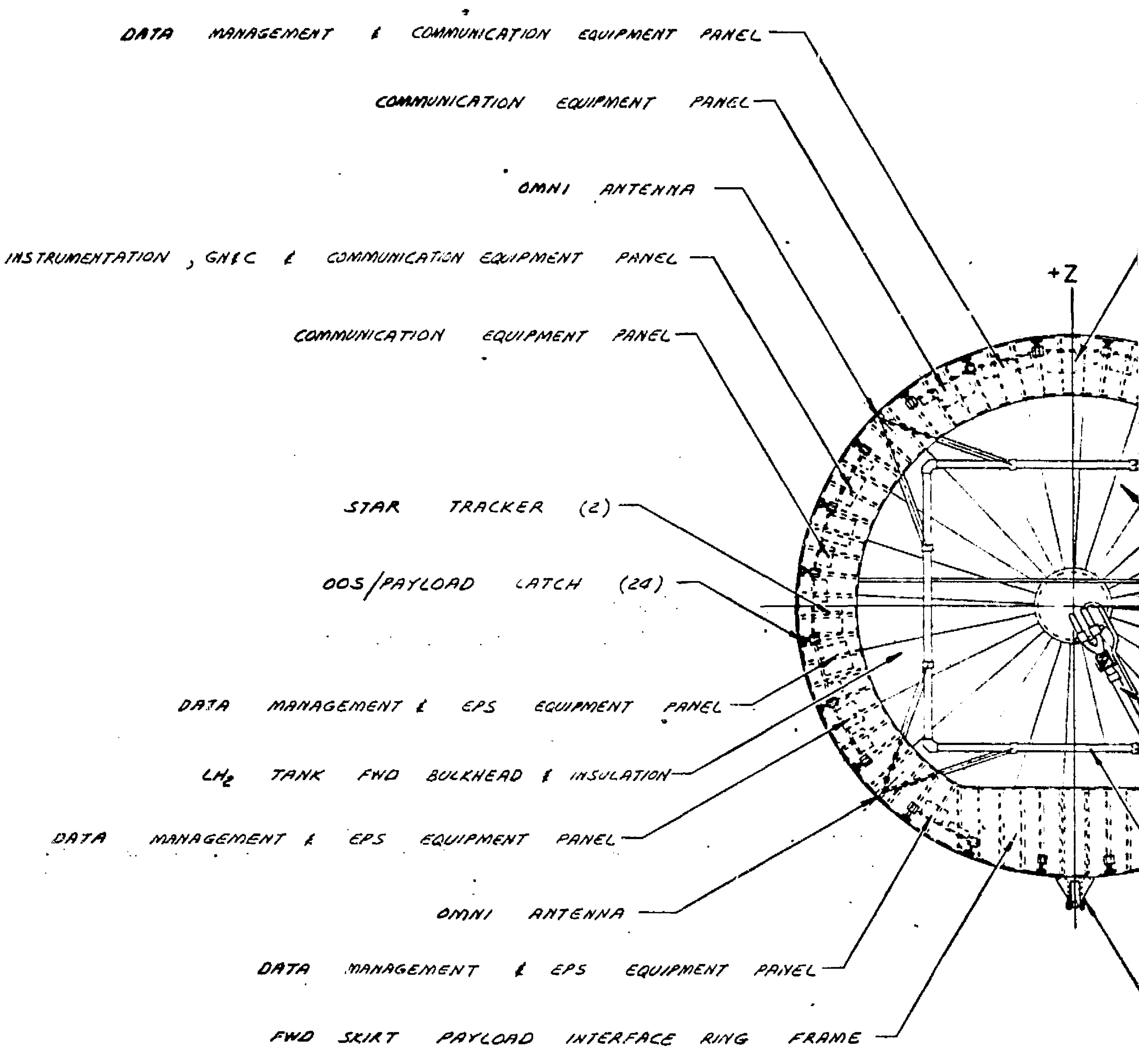


AFT SKIRT (164 DIA.)(REF.)
THRUST STRUCTURE
MAIN ENGINE GIMBAL ACTUATOR ATTACH FITTING (2)
ENGINE PURGE SYSTEM HELIUM TANK
MAIN ENGINE CONNECT PANEL
APS ENGINE CLUSTER (4)
AVIONICS EQUIPMENT PANEL (4)
OOS/SHUTTLE ±Z ATTACH FITTING (2)
MPS LOX FEED LINE
FUEL CELL HEAT PIPE RADIATOR (4) (50° SEGMENT ~ TYP)
OOS/SHUTTLE LOX UMBILICAL PANEL
AFT SKIRT AFT RING



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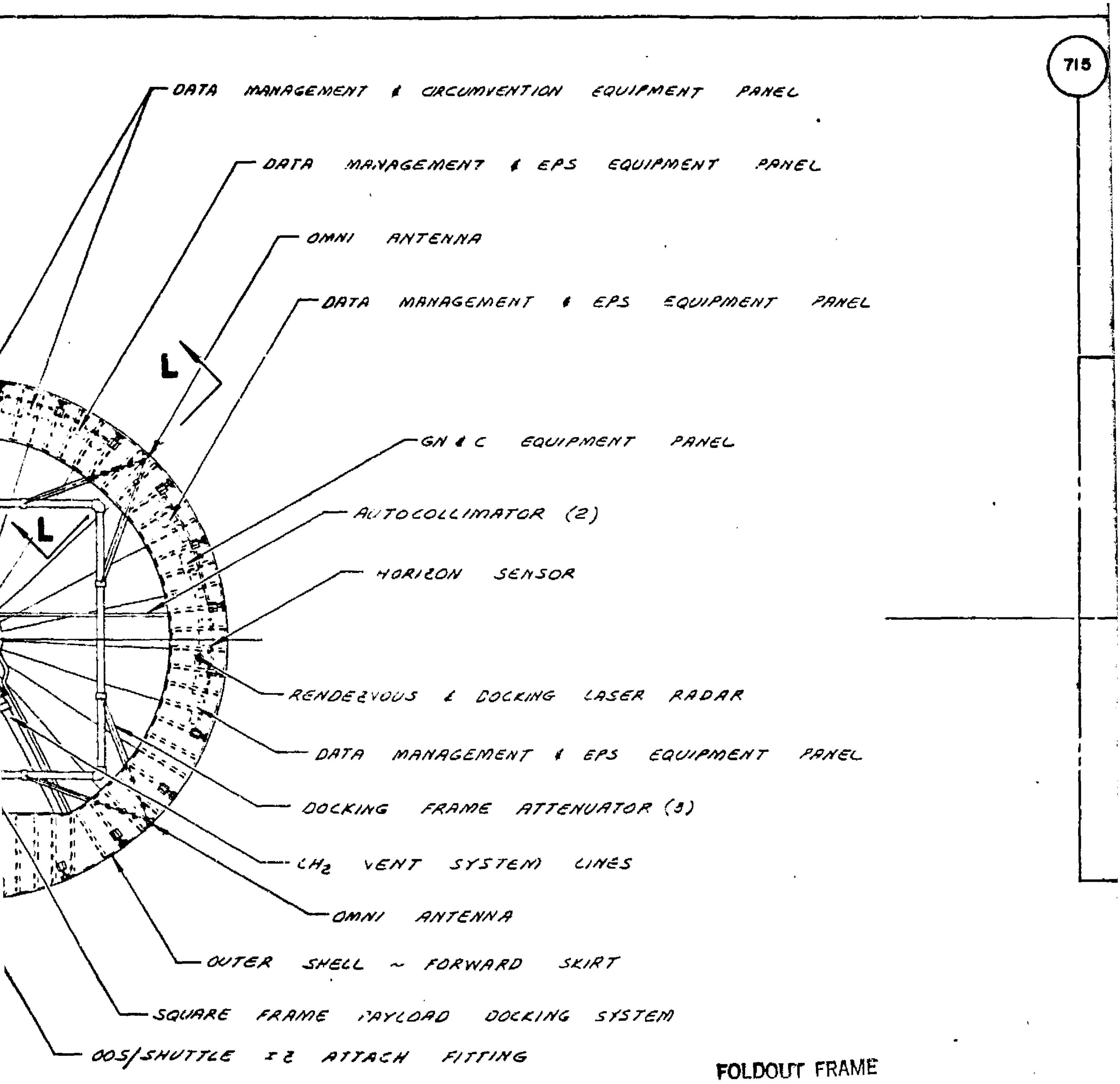
PURGE BAG/METECRCID SHIELD M



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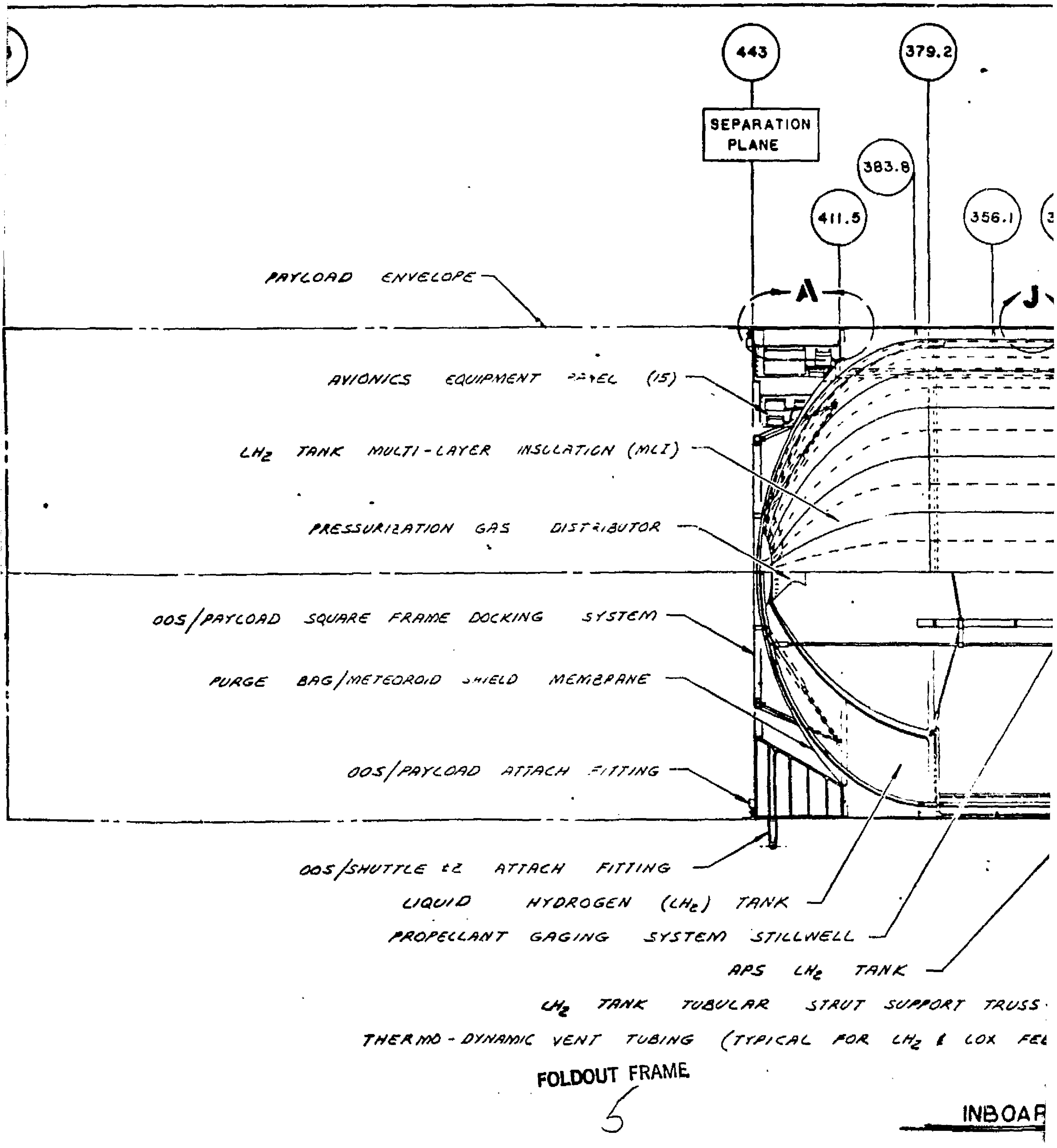

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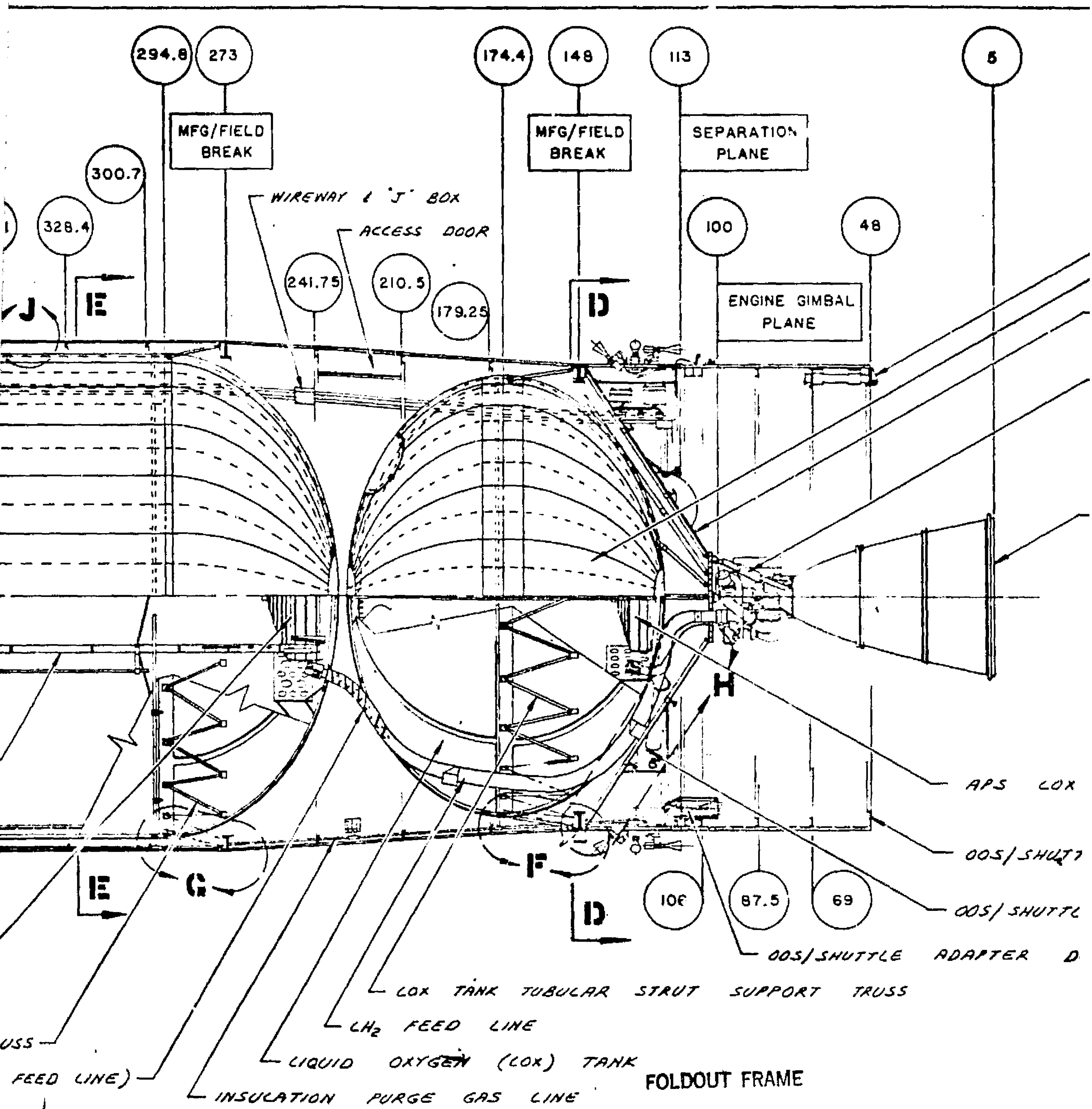

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OARD PROFILE

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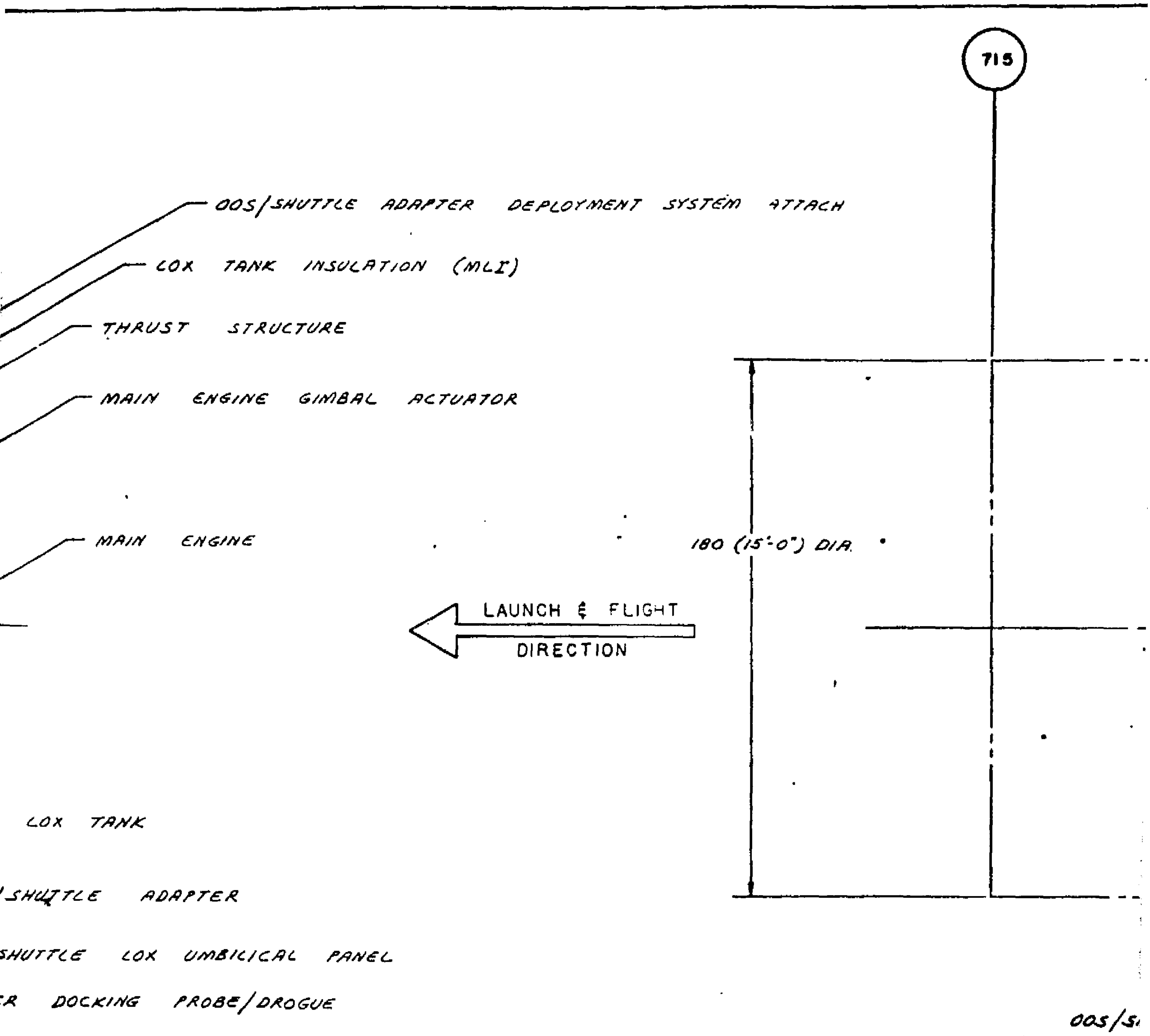

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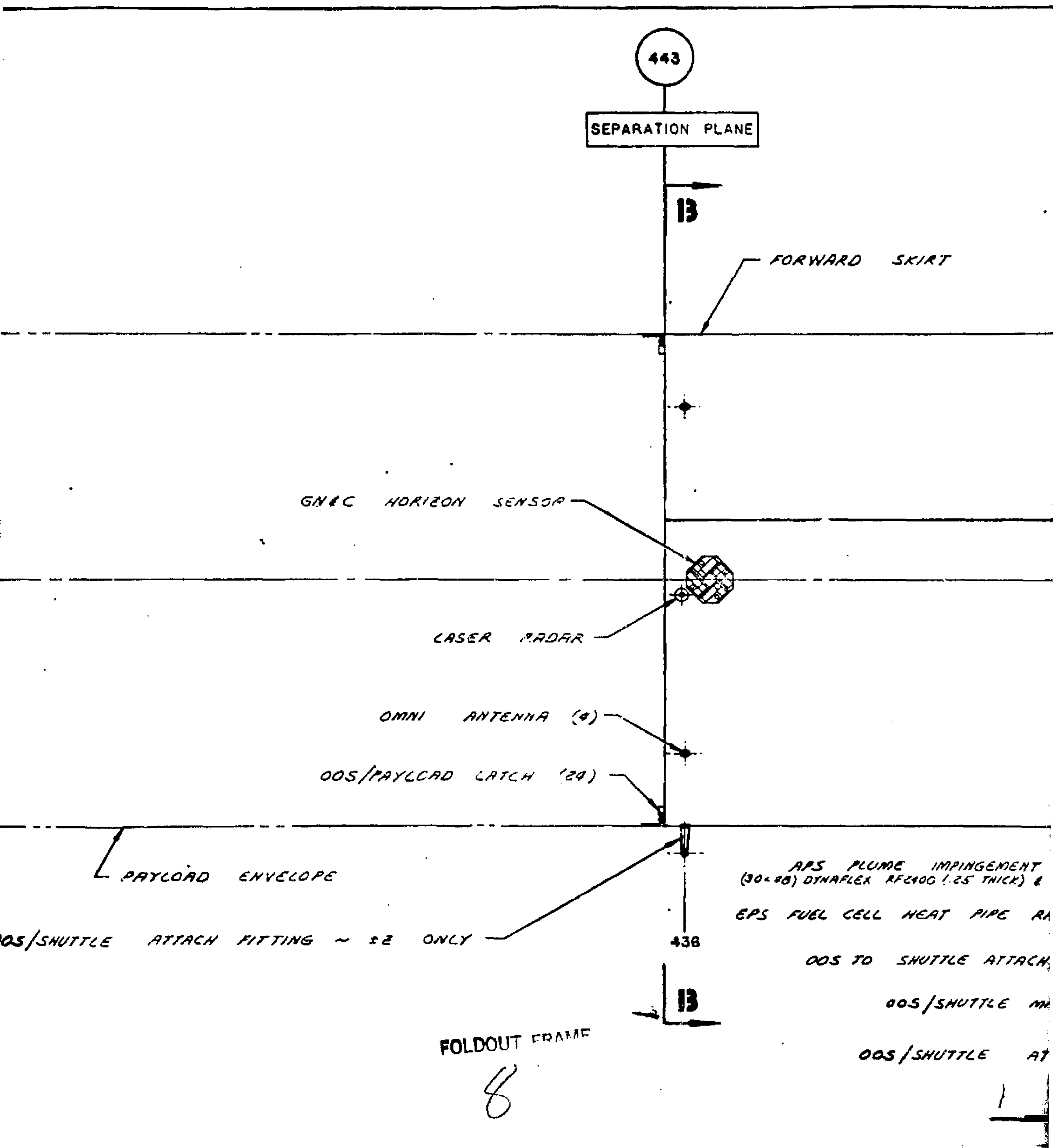

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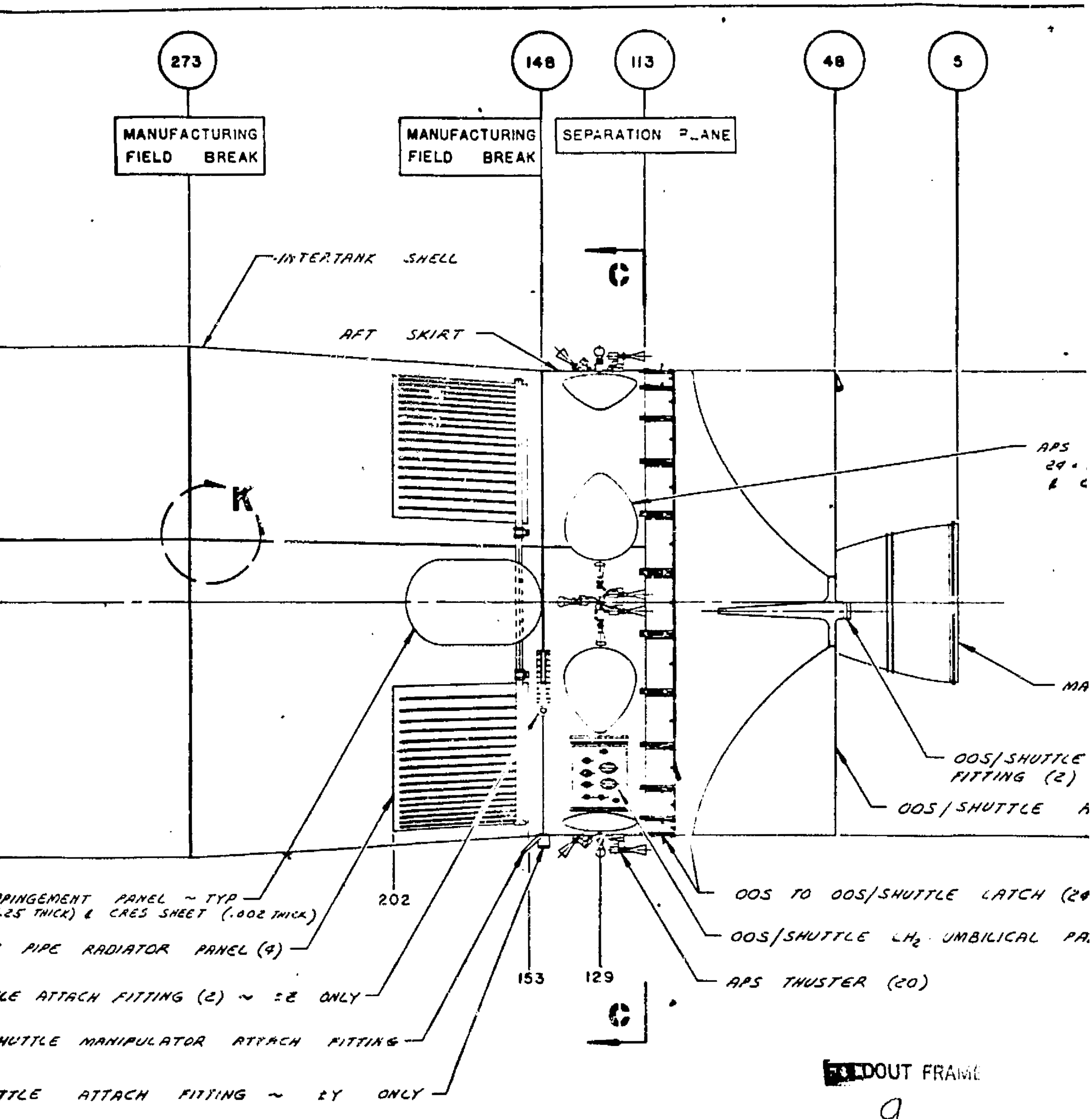

1 OUTBOARD PROFILE

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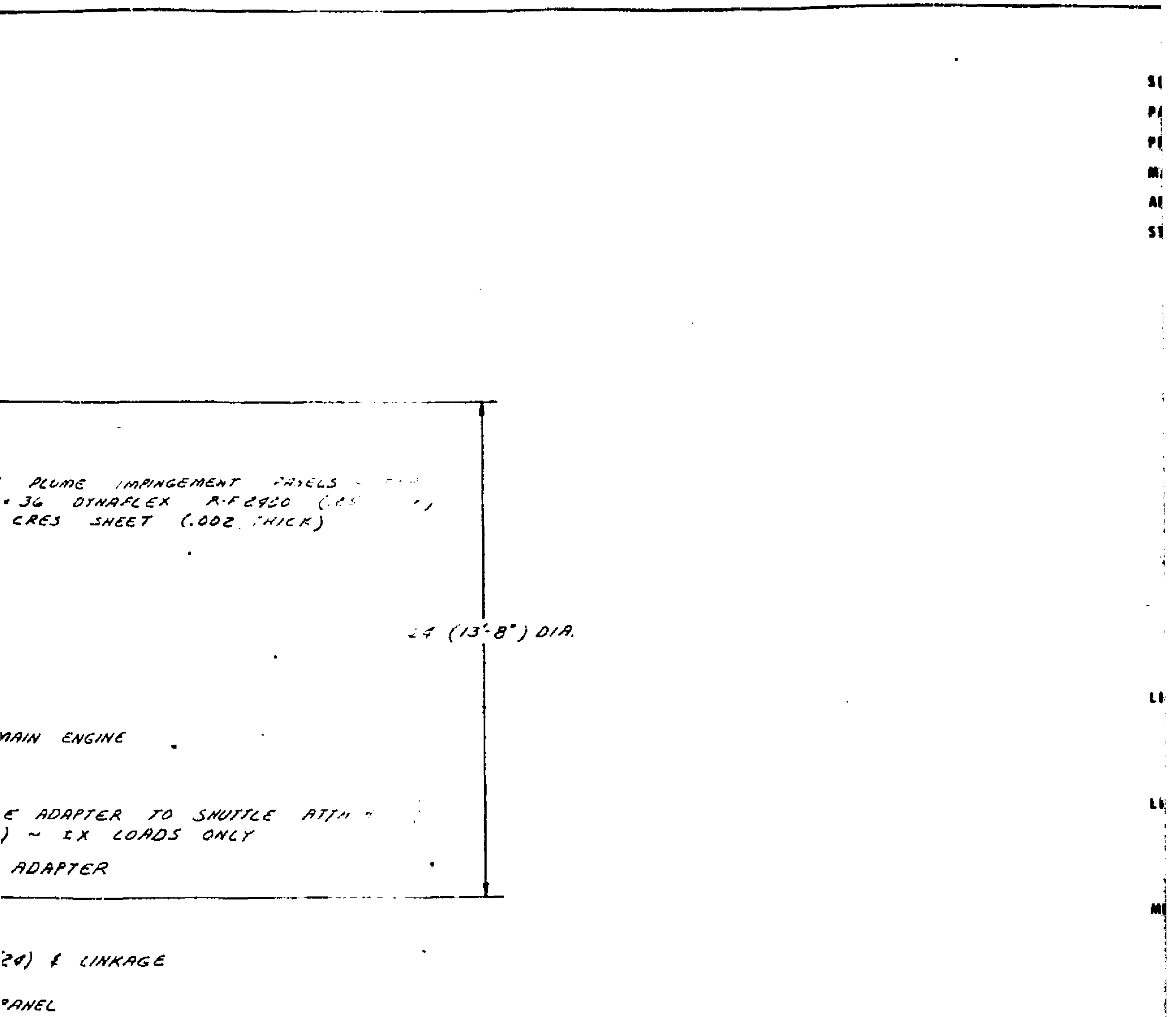

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SIZE: 164 IN. DIA. TO 180 IN. DIA X 36 FEET 6 IN LONG

PAYLOAD ENVELOPE 15 FEET DIA. X 22 FEET 8 IN. LONG

PROPELLANT: LIQUID OXYGEN / LIQUID HYDROGEN

MAIN ENGINE ΔV PROPELLANT LOAD 59,692 LBS.

APS ΔV PROPELLANT LOAD 194 LBS

STRUCTURE:
- **FORWARD SKIRT** GRAPHITE COMPOSITE SKIN (0.008 IN THICK) / ALUMINUM HONEYCOMB CORE (0.720 IN. THICK)
- **INTERTANK SHELL** GRAPHITE COMPOSITE SKIN (0.008 IN. THICK) ALUMINUM HONEYCOMB CORE (0.94 IN. THICK)
- **AFT SKIRT:** GRAPHITE COMPOSITE SKIN (0.010 IN THICK) / ALUMINUM HONEYCOMB CORE (0.51 IN THICK)
- **OOS/ORBITER** ADAPTER GRAPHITE COMPOSITE SKIN (VARYING THICKNESS) / ALUMINUM HONEYCOMB CORE (1.12 IN THICK)
- **LOX TANK** SUPPORT LONGITUDINAL BORON FILAMENTS WITH GRAPHITE HOOP FILAMENTS
  - **24 TUBULAR** STRUTS 2.0 DIA X 0.040 IN THICK WALL
- **LH_2 TANK** SUPPORT S-GLASS LONGITUDINAL AND HOOP FILAMENTS
  - **24 TUBULAR** STRUTS 2.0 IN DIA X 0.024 IN. THICK WALL
- **THRUST STRUCTURE** LONGITUDINAL BORON FILAMENTS WITH GRAPHITE HOOP FILAMENTS
  - **12 TUBULAR** STRUTS 1.5 IN. DIA X 0.02 IN THICK WALL
  - **STABILITY** CONE GLASS CLOTH 0.010 IN THICK
  - **STABILITY** RING FRAMES GRAPHITE COMPOSITE
    - 0.40 IN. X 2.25 IN. DEEP X 0.03 IN THICK
    - 0.45 IN. X 2.50 IN. DEEP X 0.09 IN THICK
- **RING FRAMES** GRAPHITE COMPOSITE
  - **PAYLOAD** INTERFACE CHANNEL SECTION
    - 2.5 IN. X 20-30 IN DEEP X 0.080 IN THICK
  - **LOX TANK** SUPPORT "I" SECTION
    - 3.0 IN. X 6.0 IN. DEEP X 0.050 IN. THICK
  - **LH_2 TANK** SUPPORT: "I" SECTION
    - 1.8 IN. X 3.7 IN. DEEP X 0.034 IN. THICK
  - **FORWARD** SKIRT STABILITY FRAMES (4) CHANNEL SECTION
    - 1.0 IN. X 1.5 IN. DEEP X 0.049 IN THICK
  - **FORWARD** SKIRT AVIONICS ATTACH STABILITY FRAME CHANNEL SECTION
    - 1.0 IN. X 12.0 IN. DEEP X 0.050 IN. THICK
  - **INTERTANK** SHELL STABILITY FRAMES (3) CHANNEL SECTION
    - 1.0 IN. X 1.5 IN. DEEP X 0.050 IN. THICK
  - **AFT SKIRT** STABILITY FRAME CHANNEL
    - 1.0 IN. X 1.5 IN. DEEP X 0.050 IN. THICK

LIQUID HYDROGEN TANK
- **INTERNAL** VOLUME 2109.8 CUBIC FEET
- **PROPELLANT** LOAD 3936 LBS
- **SIZE:** 168 IN. DIA. X 204.4 IN LONG
- **SHAPE:** 1.4 RATIO ELLIPTICAL BULKHEADS WITH CYLINDRICAL SECTION BETWEEN
- **MATERIAL:** 2014-T651 ALUMINUM

LIQUID OXYGEN TANK
- **INTERNAL** VOLUME 766.1 CUBIC FEET
- **PROPELLANT** LOAD 52,460 LBS
- **SIZE:** 152.4 IN. DIA. X 108.9 IN LONG
- **SHAPE:** 1.4 RATIO ELLIPSOID
- **MATERIAL:** 2014-T651 ALUMINUM

FOLDOUT FRAME

11

METEOROID PROTECTION
- **OUTER** SHELL STRUCTURE
  - **FORWARD** SKIRT
  - **INTERTANK** SHELL
  - **AFT** SKIRT
- **RUBBER** IMPREGNATED GLASS CLOTH ON FORWARD END (PURGE BAG)
- **THRUST** STRUCTURE CONICAL CLOSEOUT ON AFT END (PURGE BAG)

THERMAL CONTROL SYSTEM
- **INSULATION** MULTI-LAYER INSULATION (MLI) ON LH_2 AND LOX TANKS
- **ZINC OXIDE** COATING
  - **OUTER** SHELL STRUCTURES
  - **FORWARD** END METEOROID SHIELD (PURGE BAG)
  - **AFT END** THRUST STRUCTURE CONICAL CLOSEOUT (PURGE BAG/METEOROID SHIELD)
  - **INNER** SURFACE OF AFT SKIRT
- **RADIATORS:**
  - **INTERTANK** SHELL ELECTRICAL POWER SYSTEM FUEL CELLS

LE CHARACTERISTICS SUMMARY

THICK

PURGE BAG
- EPOXY RESIN COATING
  - INNER SURFACE OF OUTER SHELL STRUCTURE
    - FORWARD SKIRT
    - INTERTANK STRUCTURE SHELL
  - INNER SURFACE OF THRUST STRUCTURE CONICAL CLOSEOUT
- RUBBER IMPREGNATED GLASS CLOTH
  - FORWARD END METEOROID SHIELD

MAIN ENGINE HI P_c STAGED COMBUSTION LOX/LH_2

THRUST:	20,000 LBS.
I_{sp}:	471.0 SEC.
P_c:	1740 PSIA
MR:	6 : 1
ϵ	400
WEIGHT:	370 LBS.
LENGTH:	97.0 IN.
EXIT DIAMETER	36.0 IN.

ACS THRUSTERS GOX/GH_2
FOUR FIVE ENGINE CLUSTERS

THRUST:	100 LBS.
I_{sp}:	420 SEC. (THRUSTERS) 386 SEC. (SYSTEM)
P_c:	250 PSIA
ϵ	40
MR:	4.2 : 1

PROPELLANT MANAGEMENT SYSTEM
- REFILLABLE ACS TANKS IN MAIN TANKS
- GAGING SYSTEM
  - POINT SENSOR STILLWELL (UTILIZATION)
  - CAPACITANCE PROBE (LOADING)
- SETTLING BAFFLES
- ANTI-VORTEX BAFFLES

OOS / ORBITER UMBILICAL PANELS
- LOCATED ON AFT SKIRT
- RADIAL ENGAGEMENT
- LOCATED IN LOWER PORTION OF ORBITER CARGO BAY

AVIONICS:
- FORWARD SKIRT
  - GN & C
  - COMMUNICATIONS
  - DATA MANAGEMENT
  - RENDEZVOUS AND DOCKING
  - INSTRUMENTATION
  - ELECTRICAL POWER DISTRIBUTION
  - CIRCUMVENTION
  - PROPELLANT MONITOR
- AFT SKIRT:
  - ELECTRICAL POWER DISTRIBUTION
  - THRUST VECTOR CONTROL SYSTEM
  - APS EQUIPMENT
  - INSTRUMENTATION
  - DATA MANAGEMENT

DOCKING SYSTEM:
- OOS / PAYLOAD: SQUARE FRAME TYPE
  - OOS ACTIVE (ATTENUATOR SYSTEM)
  - PAYLOAD PASSIVE (GUIDE SYSTEM) OR ACTIVE (ATTENUATOR SYSTEM)
- OOS / ORBITER ADAPTER (PROBE / DROGUE)
  - OOS PASSIVE (DROGUE)
  - ADAPTER ACTIVE (PROBE)

LATCHING SYSTEM
- OOS / PAYLOAD
  - ACTIVE LATCHES ON
  - PAYLOAD PASSIVE
- OOS / ORBITER ADAPTER
  - ACTIVE LATCHES ON
  - OOS PASSIVE (FITT

OOS / ORBITER ATTACH
- X-LOAD @ ORBITER THRU
  - SHOULDER FITTINGS
- Z-LOAD @ ORBITER MID-
  - SHOULDER FITTINGS
  - LOWER CENTER FITT
- Y-LOAD @ ORBITER MID-
  - LOWER CENTER FITT

FOLDOUT FRAME

Figure 11-1. Inboard/Outboard Profile

12

LATCHING SYSTEM
- OOS / PAYLOAD
  - ACTIVE LATCHES ON OOS (4)
  - PAYLOAD PASSIVE
- OOS / ORBITER ADAPTER
  - ACTIVE LATCHES ON ADAPTER (4)
  - OOS PASSIVE (FITTINGS ONLY)

OOS / ORBITER ATTACH
- X-LOAD @ ORBITER THRUST BEAM
  - SHOULDER FITTINGS ON ADAPTER (2)
- Z-LOAD @ ORBITER MID-BODY FRAMES
  - SHOULDER FITTINGS ON OOS LOX TANK SUPPORT FRAME (2)
  - LOWER CENTER FITTING AT [illegible] INTERFACE FRAME (1)
- Y-LOAD @ ORBITER MID-BODY FRAMES
  - LOWER CENTER FITTING ON OOS LOX TANK SUPPORT FRAME (1)

FOLDOUT FRAME

13

SYSTEM)

Figure 11-1. Inboard/Outboard Profile, Orbit-to-Orbit Shuttle

Table 11-1. I5-2/IAPS Weight Comparison

Component	ID	Qty	I5-2 Weight kg	(lb)	IAPS Weight kg	(lb)	Delta Weight kg	(lb)
Fill and Drain								
LOX and LH_2 Drain Valves	51,53	2/4	0.91	(2.0)	0.68	(1.5)	+0.91	(+2.0)
He Fill Disconnect - LOX	63	1	-	-	0.45	(1.0)	+0.45	(+1.0)
He Rupture Disc - LOX	64	1	-	-	0.23	(0.5)	+0.23	(+0.5)
Pressurization								
He Isolation Valve - LOX	36	2	0.91	(2.0)	-	-	-1.81	(-4.0)
He Isolation Valve - LH_2	36	2	0.91	(2.0)	0.54	(1.2)	-0.73	(-1.6)
Pressurization Switch - LOX	37	2	0.45	(1.0)	-	-	-0.91	(-2.0)
Propellant Control								
LOX Tank Capillary Device	11	1	0.32	(0.7)	0.68	(1.5)	+0.36	(+0.8)
LH_2 Tank Capillary Device	13	1	1.91	(4.2)	0.95	(2.1)	-0.95	(-2.1)
LH_2 Bleed Return Solenoid	15	1	0.68	(1.5)	-	-	-0.68	(-1.5)
LH_2 Low Bleed Shutoff Valve	38	1	0.68	(1.5)	0.36	(0.8)	-0.32	(-0.7)
LH_2 Low Bleed Expander	16	1	0.68	(1.5)	0.36	(0.8)	-0.32	(-0.7)
LH_2 High Bleed Shutoff Valve	61	1	-	-	0.36	(0.8)	+0.36	(+0.8)
LH_2 High Bleed Expander	62	1	-	-	0.36	(0.8)	+0.36	(+0.8)
Propellant Feed								
LOX Reservoir	17	1	0.54	(1.2)	1.09	(2.4)	+0.54	(+1.2)
LH_2 Reservoir	19	1	2.31	(5.1)	3.81	(8.4)	+1.50	(+3.3)
LOX/LH_2 System ISO Valves	55,56	2	1.36	(3.0)	0.36	(0.8)	-2.20	(-4.4)
LOX/LH_2 Quad ISO Valves	18,20	8	1.36	(3.0)	0.45	(1.0)	-7.26	(-16.0)
LOX Pump and Drive	39	1	6.12	(13.5)	3.58	(7.9)	-2.54	(-5.6)
LH_2 Pump and Drive	44	1	7.71	(17.0)	4.76	(10.5)	-2.95	(-6.5)
LOX Accumulator/Bellows	41	1	0.32	(0.7)	0.68	(1.5)	+0.36	(+0.8)
LH_2 Accumulator/Bellows	6	1	1.00	(2.2)	2.18	(4.8)	+1.18	(+2.6)
LOX/LH_2 Relief Solenoids	42,47	2	0.91	(2.0)	0.36	(0.8)	-1.09	(-2.4)
LOX/LH_2 Relief Orifices	57,58	2	0.14	(0.3)	-	-	-0.27	(-0.6)
Overboard Vent								
LOX H_e Vent Valve	5	2	0.23	(0.5)	-	-	-0.45	(-1.0)
LOX/LH_2 Vent Solenoids	24,28	8	0.68	(1.5)	0.36	(0.8)	-2.54	(-5.6)
Total							-18.55	-40.9

Table 11-3 summarizes evaluation information on four pump types—one vane and three reciprocating. The Type 4 pump, a nonboosted reciprocating pump, is slightly heavier but has all the advantages of proven service since many similar units are in successful field use (AIRCO units) However, these all require 1.5 N/cm^2 (2 psi) NPSP suction conditions and the zero NPSP requirement for this application probably cannot be met at any reasonable speed and weight.

Types 2 and 3 are both boosted reciprocating pumps. Type 2 is a high-speed (3000 rpm) Sundstrand pump concept running at the speed chosen for the inducer which provides 1.6/0.75 NPSP (oxidizer/fuel). Experience with AIRCO ground service units is that above 800 rpm more than 1.4 N/cm^2 (2 psi) NPSP is required for hydrogen and the peak speed attained at high NPSP has been 1200 rpm. At that high speed, the AIRCO pumps lose volumetric efficiency but have not experienced cavitation damage. It is possible, however, that a design effort to increase suction specific speed could result in a successful high-speed unit at the boost pressures chosen. If unsuccessful, Type 3

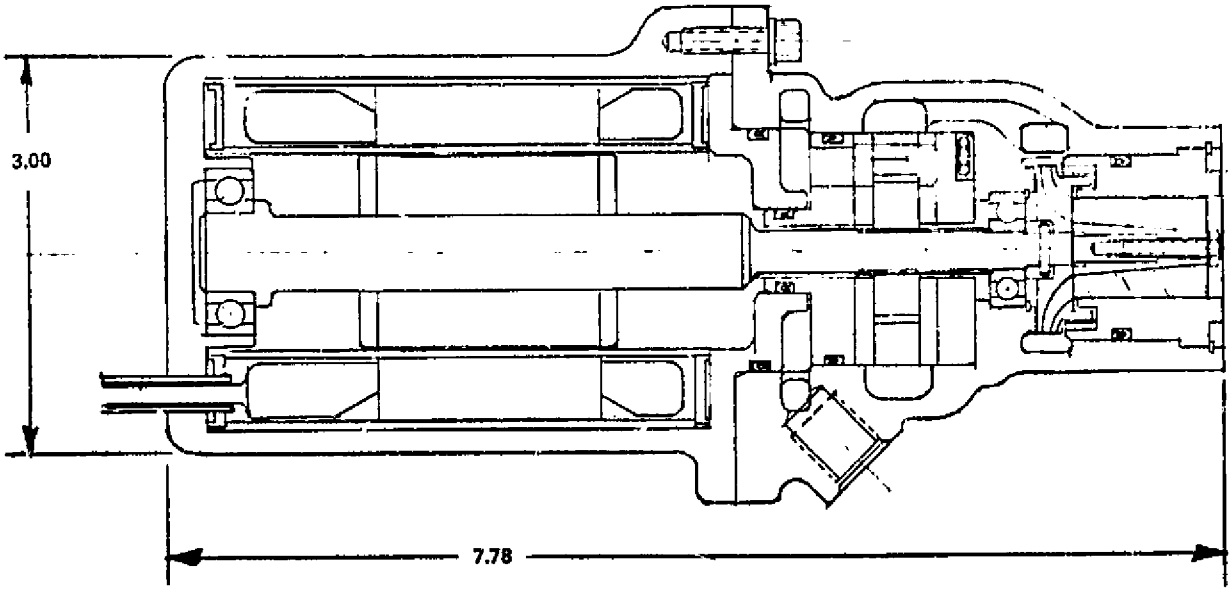

Figure 11-2. Liquid Oxygen Motor-Driven Vane Pump

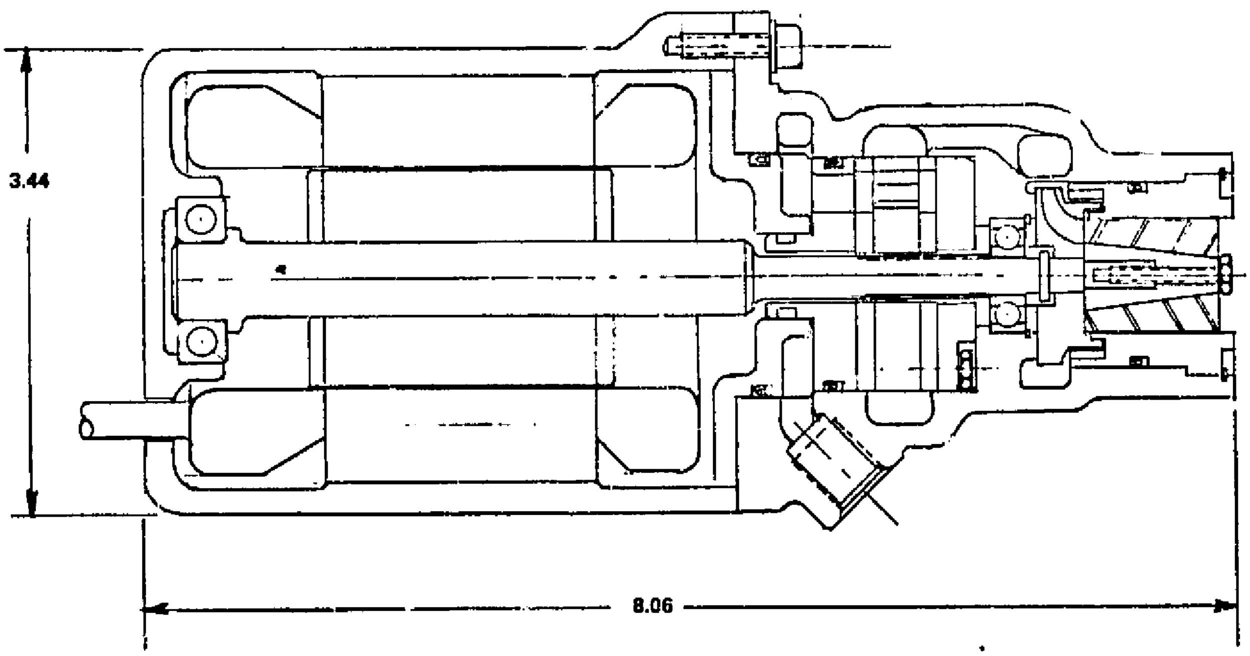

Figure 11-3. Liquid Hydrogen Motor-Driven Vane Pump

Table 11-2. Candidate Pump Characteristics

	Vane		Reciprocating	
	Oxygen	Hydrogen	Oxygen	Hydrogen
Requirements				
Flow rate, m^3/sec (gpm)	0.00454 (1.20)	0.0245 (6.47)	Same	
Head rise, m (ft)	129 (425)	2110 (2920)		
Fluid output power, kw (hp)	0.109 (0.147)	0.590 (0.792)		
NPSP	0	0		
Vapor capacity, %	101	10		
Characteristics				
Boost stage				
Head rise, N/cm^2 (psi)	2.0 (2.9)	1.3 (1.9)	1.1 (1.6)	0.52 (0.75)
Efficiency, %	40	45	32	35
Pump stage				
RPM	12 [illegible]	12,000	3,000	3,000
Design specific speed	4[illegible]	39.7	35.7	10.1
Displacement per rev, cc (in^3.)	1 [illegible].074)	2.23 (0.136)	1.61 (0.0985)	8.90 (0.544)
Overall Efficiency, %	75	64	75	78
Weight, kg (lb)	0.23 (0.50)	0.23 (0.50)	0.23 (0.50)	0.27 (.60)
Motor				
Efficiency, %	78	92	78	92
Output power, kw (hp)	0.147 (0.198)	0.920 (1.24)	0.147 (0.198)	0.750 (1.01)
Weight, kg (lb)	1.86 (4.1)	1.95 (4.3)	2.04 (4.5)	3.09 (6.8)
Assembly total				
Weight, kg (lb)	3.0 (6.6)	3.2 (7.1)	3.6 (7.9)	4.8 (10.5)
Power demand, kw (hp)	0.189 (0.253)	0.994 (1.33)	0.246 (0.330)	1.01 (1.35)

could be considered an alternate proposal. The Type 3 piston stage is geared down from the inducer shaft to the speed range of AIRCO experience. The lightweight motor and reasonable size inducer features of Type 2, which follows from their high shaft speed, is retained in the proposed Type 3 concept.

Type 1 is a boosted vane pump. It is still considered competitive but is not selected for two reasons. First, its ability to pump without excessive leakage is unknown. In fact, the only, limited experimental program on a much larger hydrogen unit yielded seriously low volumetric efficiency due to high leakage. For this application, the unit is smaller and that introduces

Table 11-3. Pump Selection Basis

		PUMP TYPE (○ = ID)			
		BOOSTED VANE ①	RECIPROCATING: ② HIGH SPEED BOOSTED	RECIPROCATING: ③ LOW SPEED BOOSTED	RECIPROCATING: ④ LOW SPEED UNBOOSTED
		← SUNSTRAND → (①–②)			
PUMP CHARACTERISTICS					
BOOST STAGE	TYPE	← AXIAL FLOW INDUCER → (①–③)			NONE
	RPM (O/F)	4,000/12,000	← 3,000/3,000 → (②–③)		-
	DRIVE	← DIRECT → (①–③)			-
PUMP STAGE	RPM (O/F)	4,000/12,000	3,000/3,000	300	300
	DRIVE	← DIRECT → (①–②)		← GEARED → (③–④)	
SELECTION CRITERIA					
INLET FLOW COMPATIBILITY (ZERO NPSH, 10% VAPOR)		← PROVEN → (①–③)			UNKNOWN
OUTLET FLUID COMPATIBILITY		NONPULSATING	← PULSATING, ACCEPTABLE → (②–④)		
PUMP STAGE VOLUMETRIC EFFICIENCY					
LEAKAGE		UNKNOWN - ONLY PRIOR TESTS UNACCEPTABLE. SCALE EFFECT MORE ADVERSE FOR THIS APPLICATION	← PROVEN → (LONG-LIFE TEFLON/BRASS RINGS) (②–④)		
CAVITATION			HIGHER RPM THAN CURRENT PRACTICE		
RELIABILITY					
FAILURE MODE		SUDDEN	← DETERIORATION → (②–④)		
PRIOR ART DATA		NONE APPLICABLE	← GOOD DATA ON ALL ELEMENTS → (②–④)		
OXYGEN SAFETY COMPATIBILITY		UNKNOWN - VANE MATERIAL AN ISSUE	← PROVEN → (②–④)		
MECHANICAL COMPLEXITY					
ARRANGEMENT		BEST	ACCEPTABLE	MORE DIFFICULT	ACCEPTABLE
SHAFT SEAL REQUIRED		← NO → (①–②)		← POSSIBLY → (③–④)	
ESTIMATED WEIGHT					
OXYGEN		6.6	7.9	11.0	11.0
HYDROGEN		7.1	10.5	13.0	13.0
TOTAL		13.7	18.4	24.0	24.0
DELTA FROM BOOSTED VANE		0	+4.7	+10.3	+10.3

introduces an aggravating scale effect--the several leak paths of a vane pump are an even larger proportion of the displacement than on the unit tested. Second, the vane materials for high-velocity rubbing seal surfaces which are compatible with oxygen safety hazard standards are evidently not among those known to be best for the functional requirement.

On the basis of this evaluation, the Type 2 high-speed, boosted reciprocating pump is selected for preliminary design.

11.3 RELIABILITY ANALYSES

SHUTTLE DOCKING

The safety criterion is that no single-point failure shall affect safety of Orbiter-Tug docking (fail safe). In the case of the APS, it is assumed that this requires capability to hold attitude long enough to complete the operation or enable the Orbiter to be maneuvered to a safe distance. This criterion also implies that there will be no uncontrolled APS thrust.

In the APS, only two elements, the pump and the accumulator, represent single-point failures in this category. If a pump failure occurs, the accumulator provides an operation completion backup since a refill is commanded just prior to docking and it contains sufficient propellant. Failure of an accumulator is considered in this section and also is found to be fail-safe in this context.

Helium leakage from an accumulator bellows rupture can alter the APS thrust characteristics sufficiently to make Tug recovery by the Orbiter potentially unsafe. The degree of the hazard caused by the helium contamination of propellant is not yet defined and should be investigated further. Even though a bellows rupture is highly unlikely, the bellows are moving components and single-point failure elements. Fail-safe accommodations should therefore be provided.

Recovery involves the half hour of retrieval operations preceeding APS deactivation and manipulator attachment. A fail-safe design must provide enough propellant to stabilize the Tug attitude to $\pm$ 0.5 deg in all axes during this half-hour period.

Any of four Tug configurations can exist during recovery: full or empty of propellants, and with or without a payload. A comparison of the configurations shows that the worst case of propellant usage exists when the Tug is empty of propellants and has no payload. Using moment-of-inertia values of 7400 kg-m^2 (5462 slug-ft^2) and 27400 kg-m^2 (22021 slug-ft^2) for roll and pitch or yaw, respectively, and a minimum bit size of 2.22 N-sec (0.5 lb-sec), the propellant needed for stabilization is approximately 0.113 kg (0.250 lb).

Either the LOX or the LH_2 accumulator bellows may fail; however, since the volume is the same in each line and the hydrogen flow velocity is greater, the hydrogen line will pass the leaking helium to the thrusters first. The line segments that are common to all thrusters hold a total of approximately 0.022 kg (0.048 lb) of propellant.

The most straightforward solution, and the one that is selected, is to increase the line capacity past the accumulators so that they always hold the required amount of propellant in ready. This is accomplished by adding a section of line of larger diameter with capillary provisions as necessary to retard helium imigration to the thrusters during the short operational period involved.

To provide the half-hour of attitude control propellant for the most demanding Tug configuration, 0.085 kg (0.188 lb) of LOX is needed in the liners. The hydrogen needed for attitude control is .029 kg (.063 lb).

ACCUMULATOR FAILURE RATE PREDICTION

During the conceptual phase, the accumulators were identified as items to be subjected to SR&T to improve their reliability. Subsequent preliminary design activity revealed that the SR&T approach is unnecessary and reliability goals are attainable as a result of design improvements.

The principal design improvement was to eliminate all static seals in the assembly and utilize an all-welded design (Figures 9-10 and 9-11). This eliminated the predominate failure mode of the conceptual phase--leakage over the 168-hour mission.

The only other failure mode to be assessed for the accumulators is bellows rupture as a fatigue failure. The required operating life of the bellows is 150 cycles per mission with replacement after 10 missions. Investigation of industry practice relative to assessing and demonstrating reliability for applications similar to this one resulted in the conclusion that a failure rate goal of one per million cycles was reasonably attainable without SR&T improvements or demonstration. Demonstration of this reliability can be deferred until the component qualification program of Tug Phase C and then be accomplished by an engineering evaluation test at four times design operating life.

This appraisal is briefly supported by the following. Small, 7.6 cm (3-in.) diameter bellows in commercial service for vacuum pumps frequently demonstrate 3 billion cycles without a failure. These units are manufactured by Metal Bellows Co. and are a welded, as opposed to formed, type of bellows. In this application, the bellows is subjected to a moderate differential pressure and is therefore similar to the accumulator bellows in that the stress level is predominately due to extension/contraction deflection as in a spring. Metal Bellows Co. is currently developing bellows of this welded type for the Space Shuttle Orbiter ECS.

At cryogenic temperatures, as for application here, both metallurgical and component test data indicate that the fatigue limits are higher with the CRES materials of interest. The Rockwell Rocketdyne Division practice in testing engine cryogenic feed line bellows flex joints (applications involving internal pressure with limited deflection) is to conduct them at the more severe condition of room temperature. Data reported by Battelle (Reference 40) show the endurance life (flexing to failure) of test coupons is four times better at LN_2 temperatures than at ambient and is 15 times better at LH_2 temperature.

The approach recommended involves an engineering evaluation of the bellows instead of a statistical reliability demonstration. The design is to account for the fatigue properties of the material by holding operating stress levels to an appropriate low value. This is the approach taken by Rocketdyne on the Shuttle main engine line bellows where the cycle life requirement is higher than for previous expendible engine programs. They have tested units designed to 4000 maximum deflection cycles to 16,000 cycles without a failure, thus meeting program demonstration test requirements. It is considered that Tug requirements are similar and therefore the same approach is warranted. Pulsing flow influence on bellows stress levels and life is a consideration that is yet to be investigated.

11.4 HEAT LOAD VARIATION SENSITIVITY

The design and performance of the IAPS thermodynamic control system depends on the effectiveness of the thermal isolation system. An increase in heat load, for example, will require an increase in LH_2 bleed consumption and a corresponding increase in the capacity of the LH_2 zero-g reservoir.

The combined performance of the insulation and low conductivity supports can be expressed in terms of effective emittance, ϵ. A review of the literature as well as thermal-vacuum testing of a Rockwell-built cryogenic radiator (Reference 41) indicate that an effective emittance value of approximately 0.006 is representative of current technology. It is anticipated, however, that future improvements in ploymeric metalizing processes and multilayer insulation layup and installation techniques can reduce effective emittance levels to approximately 0.002 by 1978. This level has already been demonstrated in the laboratory on calometric test tanks.

In the event that the predicted insulation improvements are not realized in time to support the Tug vehicle program, a weight sensitivity analysis is presented in Figure 11-4. The additional weight incurred by an increase in LH_2 bleed expenditure and zero-g reservoir capacity is presented, in terms of Tug payload weight penalty, as a function of insulation system effective emittance. As can be seen, use of current insulation would result in approximately a 35-kg (73-lb) loss in Tug payload capability as compared to the use of insulation providing performance at the level projected for 1978.

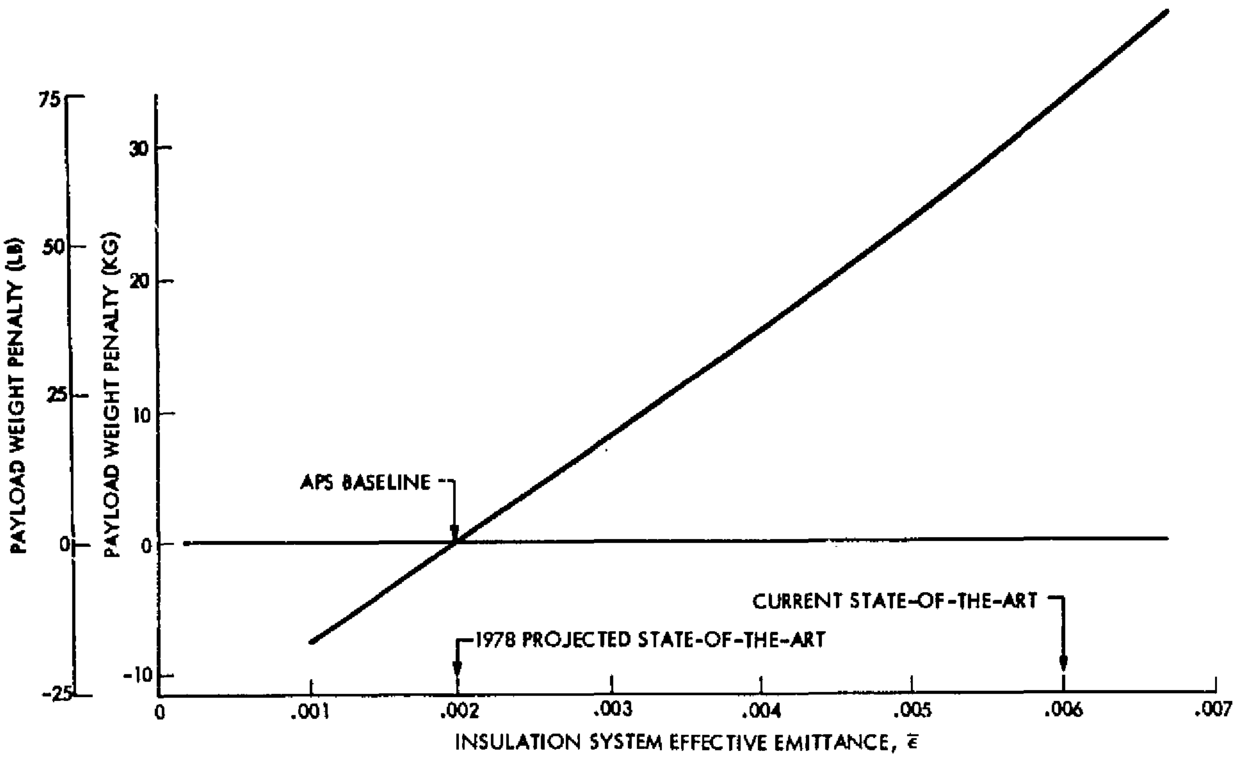

Figure 11-4. Tug Payload Sensitivity to APS Insulation Performance

12. CONCLUSIONS AND RECOMMENDATIONS

The study objective--to evolve a promising Tug auxiliary propulsion system using the liquid-liquid O/H concept--was achieved. The selected concept, an integrated design which uses the main tanks as the propellant source, has several superior attributes which arise only because it is integrated. Since no other competitive integrated concepts for Tug now exist and since these attributes are extremely beneficial to the Tug program as well as to Tug functional capability, the continued development of the concept is recommended. The next phase of development requires experimental activity. To that end, a technology development plan with specific goals and approaches has been prepared.

<u>Tug Performance</u>

Significant conclusions regarding the performance of a Tug vehicle incorporating the cryogenic integrated auxiliary propulsion system are summarized as follows:

1. Basic payload performance on the study reference mission (triple payload deployment) is approximately 5-percent superior to dedicated systems when the mission profile involves the main engine for all small velocity maneuvers within its capability (Mission A profile).

2. For the mission profile which replaces six of the main engine small velocity maneuvers with APS maneuvers (Mission B profile) only the integrated system retains adequate payload capability. Its payload decrement from the A to the B profile is only 8 percent. This makes the B profile a viable one for flights that are not payload-limited.

3. For most main engine failures which do not result in propellant loss, the integrated APS can be used for abort return to the Shuttle. This capability exists for recovery of the Tug in its liftoff configuration (with payload, if present) for up to 60 percent of the main engine duty cycle.

4. Since the APS and MPS propellant is stored in common tanks, the propellant allocations are interchangeable. This permits, without vehicle change, the accommodation of missions requiring high APS impulse--up to a value limited only by MPS tank capacity and the lower (86 percent) specific impulse of the APS thrusters.

5. The reliability goal of 0.996 apportioned to the APS is achievable by either an integrated or dedicated cryogenic system. Inherent reliability is greatest for the integrated system because of component commonality with the MPS for functions such as overboard venting and ground loading.

6. The use of an APS ullage (settling) maneuver prior to main engine start does not penalize Tug payload performance since the cryogenic APS specific impulse is higher than the main engine tank head idle mode. This is most advantageous since other main engine start options are not yet proven.

Tug Impacts

Tug vehicle and program impacts resulting from incorporation of the selected integrated APS are, in sum, beneficial. To summarize, they are:

1. A relatively high DDT&E cost - \$12.6 M̄

2. A higher load (increased by 0.85 kw) and duty cycle (increased by 0.85 kw hr - Mission A and 3.9 kw hr - Mission B) for the Tug power system. An APS dedicated battery or higher power level fuel cells are required.

3. The opportunity exists to reduce the main engine development test plan by 150 tests since main engine criticality is reduced by the presence of APS backup capability. This would result in a program DDT&E cost saving of \$3.6 M̄.

4. APS settling for main engine start eliminates the need to develop either a main tank start basket or a main engine tank head idle mode self-settling capability. The start basket is estimated to be the most inexpensive of the two alternatives at a DDT&E cost of \$0.32 M̄. For either APS settling or the start basket choice, the main engine development effort is substantially reduced, but the cost saving has not been estimated.

5. The capability of the selected APS permits elimination of the pumped idle mode in the main engine with negligible vehicle performance penalty. An engine DDT&E cost saving of \$2.83 M̄ would result.

6. The opportunity to use the Mission B profile at 92 percent of maximum payload is estimated to apply to 100 of the 243 planned flights. The resulting elimination of 5 out of 11 main engine rotating starts per mission would save \$0.9 M̄ in main engine overhaul costs at negligible increase in APS maintenance cost.

7. APS backup of the main engine over 243 planned flights results in a statistically predicted savings for otherwise lost Tug vehicles and payloads equivalent to \$18.7 M̄.

Auxiliary Propulsion Design

Principal conclusions regarding the design of the integrated liquid-liquid O/H concept are summarized as follows:

1. An APS propellant reservoir to provide for propellant acquisition during zero gravity is necessary. The best design is one which is refilled on demand during MPS or APS velocity maneuvers. For zero-g control of propellant in the reservoir, a capillary device is essential for adequate performance and functional flexibility.

2. In order to make maximum use of the attributes of an integrated APS, pump feed rather that pressure feed is required. Unlike the pressure feed system, the pumped system design and weight is unaffected by total impulse rating.

3. Redundant pumps are not required. Since the pump duty cycle is low, probable failure rates do not significantly affect the system mission success goal. Although the pumps are single-point failure elements, the use of downstream accumulators allows for safe continuation of Shuttle docking after a pump failure. Similarly, an accumulator failure permits pump-only continuation of docking.

4. The very low flow rate required leads to the preference for positive displacement pumps in order to hold machine speeds to practical levels and yet maximize efficiency. No existing fully developed flight-weight pumps meet the requirements even with modification. However, the development of either piston or balanced vane type of pumps is considered readily achievable for the application.

5. Thermal control for all cryogenic elements of the system is best accomplished by an active system using hydrogen bleed as the coolant.

6. The thruster design point is sensitive to the predicted thruster specific impulse variation with mixture ratio and chamber pressure. Based on the data developed in this study, the selected thruster design point is:

   Thrust - 111 N (25 lb)

   Chamber pressure - 103 N/cm^2 (150 psia)

   Nominal mixture ratio - 3

   Abort mode mixture ratio - 5.6 (aft engines)

   Area ratio - 200 (aft engines), 50 (all others)

The capability for the thruster to shift mixture ratio from 3 to 5.6 is essential if a principal attribute of the integrated system-- the capability to back up the main engine for abort recovery of the Tug and its payload--is to be retained. The thrust level of 111 N (25 lb) is very close to the best value for Tug normal operations. The attitude hold duty cycle and propellant consumption is excessive at a thrust level of 445 N (100 lb), for example. In the abort mode, the +X thrusters could beneficially use higher thrust to reduce extended burn gravity losses.

7. For two important reasons, the location of APS thrusters in the Tug vehicle should be moved from the baseline vehicle mid-ship position to an aft skirt location. For any APS design, the aft location provides greater thruster redundancy in pitch and yaw by enabling roll thrusters to perform this function. At the mid-ship location, the thrusters are, at times, nearly at the same station as the vehicle c.g. and the roll thrusters have insufficient authority in pitch or yaw. For the integrated APS, the aft thruster location enhances the velocity mode (either normal or abort backup) effective specific impulse nearly 10 percent since the thruster cant angle can be reduced from the 25 degrees required at mid-ship to zero.

8. Two electrical power system options are viable for supplying auxilary propulsion pump power. These are: (1) the addition of silver-zinc primary batteries, and (2) an increase in the size and capability of the Tug fuel cell. Although heavier for Mission A by approximately 11.3 kg (25 lb), fuel cell modification provides the greatest mission flexibility by allowing extended APS operation without affecting other power demands. The fuel cell system is approximately 12.7 kg (28 lb) lighter for Mission B. Abort recovery capability could still be provided with the battery system with a provision for switching to the fuel cell and reducing the non-essential vehicle power loads during abort. Since either approach is acceptable, the final selection of the pump power supply system is deferred to Tug Phase B.

Technology Development

Four parts of the IAPS have SR&T requirements which are recommended for completion prior to Tug Phase B subsystem design selections in mid CY 1976. The SR&T program is estimated to cost approximately $4 $\bar{M}$. None of these developments are considered excessively difficult. Beginning with the most critical programs, they are:

1. Thruster performance, life, and inlet requirements verification (16 months, $1.8 $\bar{M}$).

2. Pump functional and performance verification (9 months, $0.4 $\bar{M}$).

3. Zero-g reservoir concept validation, and functional and performance verification (12 months, \$0.8 M̄).

4. Thermodynamic control functional and performance verification (12 months, \$1.2 M̄).

Reliability improvement testing of the accumulator bellows was identified as a fifth SR&T program during the conceptual phase of study, but was later eliminated by redesigning the component to eliminate the principal failure mode.

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