CR-171848

(NASA-CR-171848-Vol-1) SPACE TRANSPORTATION SYSTEM (STS) PROPELLANT SCAVENGING SYSTEM STUDY. VOLUME 1: TECHNICAL REPORT Final Report (Rockwell International Corp.) 293 p HC A13/MF A01 CSCL 22B G3/16

N85-20000

Unclas 18781

STS 84-0570

SPACE TRANSPORTATION SYSTEM (STS) PROPELLANT SCAVENGING SYSTEM STUDY FINAL REPORT

VOLUME I TECHNICAL REPORT

JANUARY 1985

Contract NAS9-16994
DRL T-1811

Rockwell International

Space Transportation Systems Division

STS 84-0570

SPACE TRANSPORTATION SYSTEM (STS) PROPELLANT SCAVENGING SYSTEM STUDY FINAL REPORT

VOLUME I
TECHNICAL REPORT

JANUARY 1985

Contract NAS9-16994
DRL T-1811

Rockwell International
Space Transportation
Systems Division

CONTENTS

PRECEDING PAGE BLANK NOT FILMED

ILLUSTRATIONS

PRECEDING PAGE BLANK NOT FILMED

Figure		Page

TABLES

PRECEDING PAGE BLANK NOT FILMED

ACRONYMS AND ABBREVIATIONS

AOTV	advanced orbital transfer vehicle
ASE	airborne support equipment
BRM-1	Baseline Reference Mission 1
CAS	capillary acquisition system
CFES	continuous flow electrophoresis system
c.g.	center of gravity
CO_2	carbon dioxide
CRT	cathode ray tube
CVM	controlling valve module
DI	direct insertion
ET	external tank
EVA	extravehicular activity
E_{mps}	excess MPS propellant
FY	fiscal year
GFJ	gas-filled flex joint
GH_2	gaseous hydrogen
GO_2	gaseous oxygen
GPC	general-purpose computer
GSE	ground support equipment
IOC	initial operational capability
IR&D	independent research and development
JSC	Johnson Space Center
LEO	low earth orbit
LH_2	liquid hydrogen (cryogenic propellant)
LO_2	liquid oxygen (cryogenic propellant)
LPS	launch processing system
MDM	multiplexer/demultiplexer
MECO	main engine cutoff (SSME's)
MLR	multilatex reactor
MMH	monomethylhydrazine (storable propellant)
MPS	main propulsion subsystem
MPTA	main propulsion test article
MSFC	Marshall Space Flight Center
NTO	nitrogen tetroxide, N_2O_4, (storable propellant)

PRECEDING PAGE BLANK NOT FILMED

OMS	orbital maneuvering subsystem
OMV	orbital maneuvering vehicle
OP	OMS on-orbit propellant requirement
OPF	Orbiter Processing Facility
ORB	STS orbiter
OTV	orbital transfer vehicle
OV	orbiter vehicle
PAM	payload assist module
PBTS	payload bay tank system
PRSD	power reactant storage and distribution
QD	quick disconnect
RCS	reaction control system
RHC	rotation hand controller
RMS	remote manipulator system
ROM	read only memory
RPC	remote power controller
RSS	root sum square
RSV	relief shutoff valve
RTLS	return to launch site
SBH5R	MPS propellant inventory designation
scar	fixed on-orbiter hardware and weight
SI	standard insertion ascent (2 OMS burns)
SINDA	system improved numerical differencing analyzer
SRB	solid rocket booster
SSME	Space Shuttle main engine
STS	Space Transportation System
TAL	transatlantic landing (abort mission)
TRANSDAP	transition digital auto pilot
TRASYS	Thermal Radiation Analysis System
TSS	tethered satellite system
TVS	thermodynamic vent system
W_{oms}	OMS propellant load
XSI	orbiter X-axis sun inertial
ZLV	orbiter Z-axis local vertical
ZSI	orbiter Z-axis sun inertial

1. EXECUTIVE SUMMARY

1.1 INTRODUCTION

The ability to place a low-cost supply of cryogenic and/or storable propellants in low earth orbit (LEO) is of primary concern to the goals of extended space-based operations, such as the manned Space Station, the orbital transfer vehicle (OTV), the orbital maneuvering vehicle (OMV), and operational satellites. The objectives of this study are first to define the most efficient and cost-effective methods for scavenging cryogenic and storable propellants and then define the requirements of these scavenging systems. The following scavenging system concepts were defined for this study. For cryogenic propellants, scavenging is the transfer of propellants from the Shuttle orbiter external tank (ET) and/or main propulsion subsystem (MPS) propellant lines into storage tanks located in the orbiter payload bay for delivery to the user station by a space-based transfer stage or the Space Transportation System (STS) by direct insertion. For storable propellants, scavenging is the direct transfer from the orbital maneuvering subsystem (OMS) and/or tankage in the payload bay to users in LEO as well as users in the vicinity of the Space Station. As required by the contract, the majority of the study effort was placed on cryogenic propellant scavenging with each concept treated individually.

1.2 SCAVENGING STUDY RESULTS

This study has determined that the best method of delivering cryogenic propellant to orbit is to load propellant into payload bay tankage before lift-off and then scavenge reserve MPS propellant after main engine cutoff (MECO). For storable propellant, the best method is to load the STS OMS tanks full on each mission even if the mission requirements call for a smaller load. In addition, propellant could also be loaded into payload bay tankage. Both of these cryogenic and storable propellant operational scenarios can only be performed on STS missions where excess lift capability exists so the additional weight of propellant and hardware can be loaded on the vehicle. Also, for scenarios using payload bay tankage, this can only be done on missions where sufficient payload bay space exists to install the tankage.

The propellant will be delivered to the space-based propellant depot by either the orbiter if the STS mission goes to the Space Station (250-nautical mile altitude) where the depot is located or the OMV, which will rendezvous with the orbiter at a lower altitude (160 nautical miles for this study). The OMV will carry empty propellant tanks that will be large enough to receive all of the propellants to be transferred from the orbiter. After the propellant transfer has been completed, the orbiter is free to complete the mission it was manifested to perform.

The study has determined that significant quantities of propellant, either 1.3 or 1.4 million pounds of cryogenic or storable propellant, can be made available for the decade from 1991 through 2000 using a flight manifest developed for the least number of STS flights. In addition, the propellant can be delivered at a cost-effective rate of approximately $100 per pound. For comparison, the rate if a dedicated STS tanker mission were used would be approximately $2,000 per pound plus the development costs of the propellant tankage. The evaluations of each concept assume that on any given mission, only one of the cryogenic or storable propellants is scavenged.

A secondary objective of this study was to identify a method to deliver 2.5 million pounds of cryogenic propellant to orbit. The results of this evaluation show that this quantity can be delivered using the operating scenario described above. The only change required would be to create additional flights and, thereby, more scavenging opportunities. This concept is referred to as a relaxed manifest. The costs associated with the additional flights must be borne by the scavenging system. The results for this concept are the delivery of 2.5 million pounds of cryogenic propellant for approximately $700 per pound, still much lower than the dedicated tanker rate.

The scenarios presented above differ from the concept of propellant scavenging typically applied in the past. The concept had been that the propellants in the ET and orbiter MPS after MECO were transferred under a low-g environment to tankage located in the payload bay. The thought behind this concept was that the payload bay tankage could be of a lightweight design because it would not have to include boost structural loads when full of propellant. This study has indicated that this is of only minor importance because the tankage and supporting structure would essentially be designed to minimum thicknesses.

The concept selected in this study is to load the tankage before lift-off with the propellant required to cause the usable MPS impulse propellant to be driven to zero. Therefore, only reserve propellant is transferred after MECO (5,690 pounds of LO_2 and LH_2). This concept yields more propellant to orbit for three reasons. First, large-diameter transfer lines and pumps for the post-MECO transfer are not required, and, therefore, the weight of associated hardware is reduced. Second, the impacts to the ET trajectory are minimal because the transfer times required just after MECO are reduced. Third, and most important, since the change in impulse propellant weight with initial orbiter weight is less than 1-to-1 (0.91 for the study ground rules), more propellant can be made available on orbit if the maximum possible payload weight is launched on each mission.

The first task of the study was to use the Marshall Space Flight Center (MSFC) nominal mission model (Revision 6) to determine the flight manifest for the years 1991 through 2000. Payload information was obtained from previous Rockwell independent research and development (IR&D) studies. The Baseline Reference Mission 1 (BRM-1) reference flight mission was used to obtain vehicle lift capability and was modified only to update the MPS propellant inventory and calculate the OMS propellant requirements for each mission. With this information, a flight manifest of 202 missions during the decade was obtained. Because some missions utilize the entire STS lift capability or missions. For cryogenic propellant, 96 flights were identified as having scavenging possibilities. For storable propellants, 165 missions were identified as possible scavenging flights. More storable propellant scavenging flights were identified than for cryogenic propellants because it is possible to scavenge the OMS propellant even though the payload bay is full. More detailed information on these results is contained in Section 4.1 of this report.

The second study task was to perform trade evaluations to determine which tankage concepts and sizes would yield the largest quantities of propellant to orbit. Trade data was also obtained for capillary acquisition system (CAS) size and flow rate requirements and heat load and insulation thickness requirements for cryogenic propellant scavenging concepts. The results of the trade studies indicated that an arrangement that has a central cylindrical LO_2 tank surrounded by a ring-shaped LH_2 tank was the best concept for cryogenic propellants. This concept is also favorable if more than one size of tankage is selected because the end caps would be the same and only the cylindrical portions would require changes in length. This tank design is necessary because it best minimizes the length of payload bay tankage for LH_2 propellant, which has a low density and, therefore, requires a significant storage volume. For storable propellants, which have a relatively high density, an arrangement of existing tank designs was selected. Detailed information on the trade evaluations is contained in Section 4.2 of this report.

The study next proceeded into study tasks that essentially performed a point design on the selected concepts, including detailed schematics, preliminary layouts and component requirements, and preliminary operational requirements. Detailed information on the point design for both cryogenic and storable propellant systems is contained in Section 4.4. Detailed schematics are presented in Figures 1 and 2 for the cryogenic system and Figures 3 and 4 for the storable system. The orbiter scar weight associated with the cryogenic scavenging system is minimal (~80 pounds), assuming that the helium tank located under the payload bay liner will be removed on weight-critical missions that are not identified as scavenging missions. For storable propellant scavenging, however, the scar weight is considerably higher (470 pounds) because of all the OMS pod plumbing and valving modifications required.

After the point design system weights were evaluated, the propellant availability analysis was updated, and propellant losses and trapped residuals were included to determine the propellant quantities deliverable to the user station. The values obtained are presented in Table 1 for both cryogenic and storable propellants along with the costs associated with each concept.

1.3 STUDY GROUND RULES

The following list contains the study ground rules that were listed in the study contract or agreed to during the course of the study.

1. STS as specified in JSC 09095-71 Shuttle Systems Weights and Performance (latest publication) shall be used as the baseline for

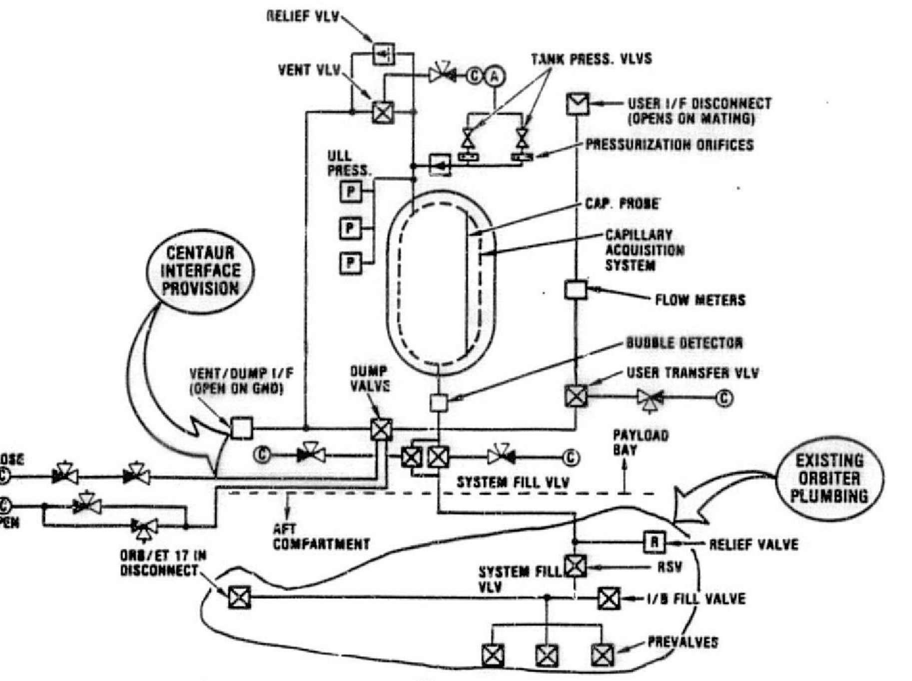


Figure 1. Cryogenic Scavenging System LO_2 System Schematic

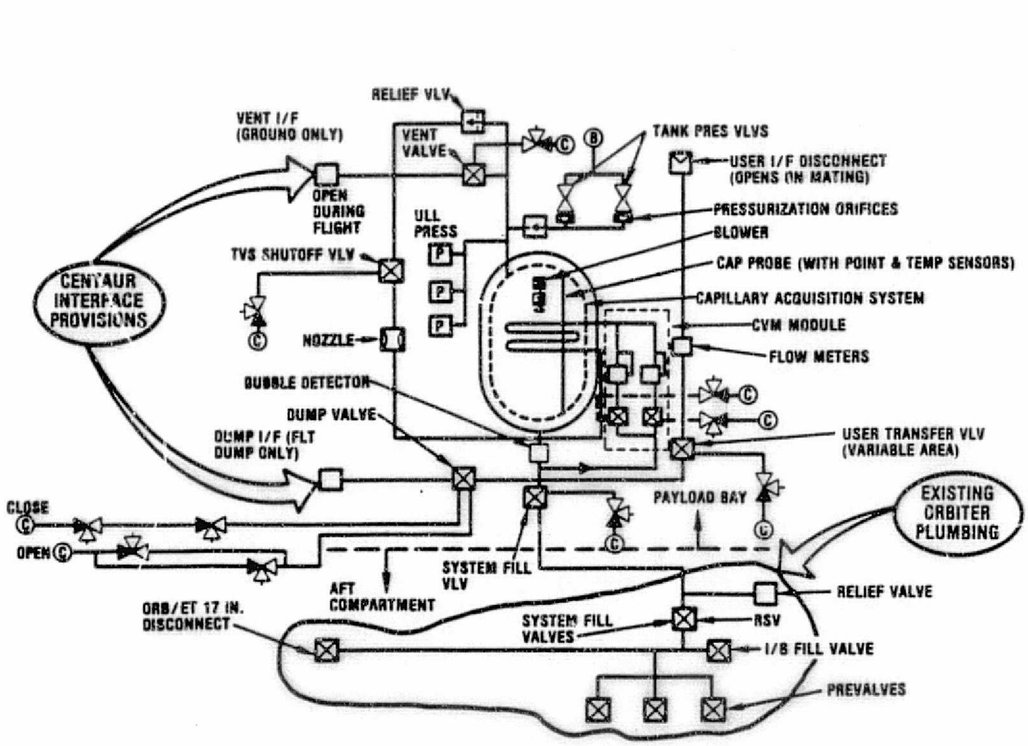


Figure 2. Cryogenic Scavenging System LH_2 System Schematic

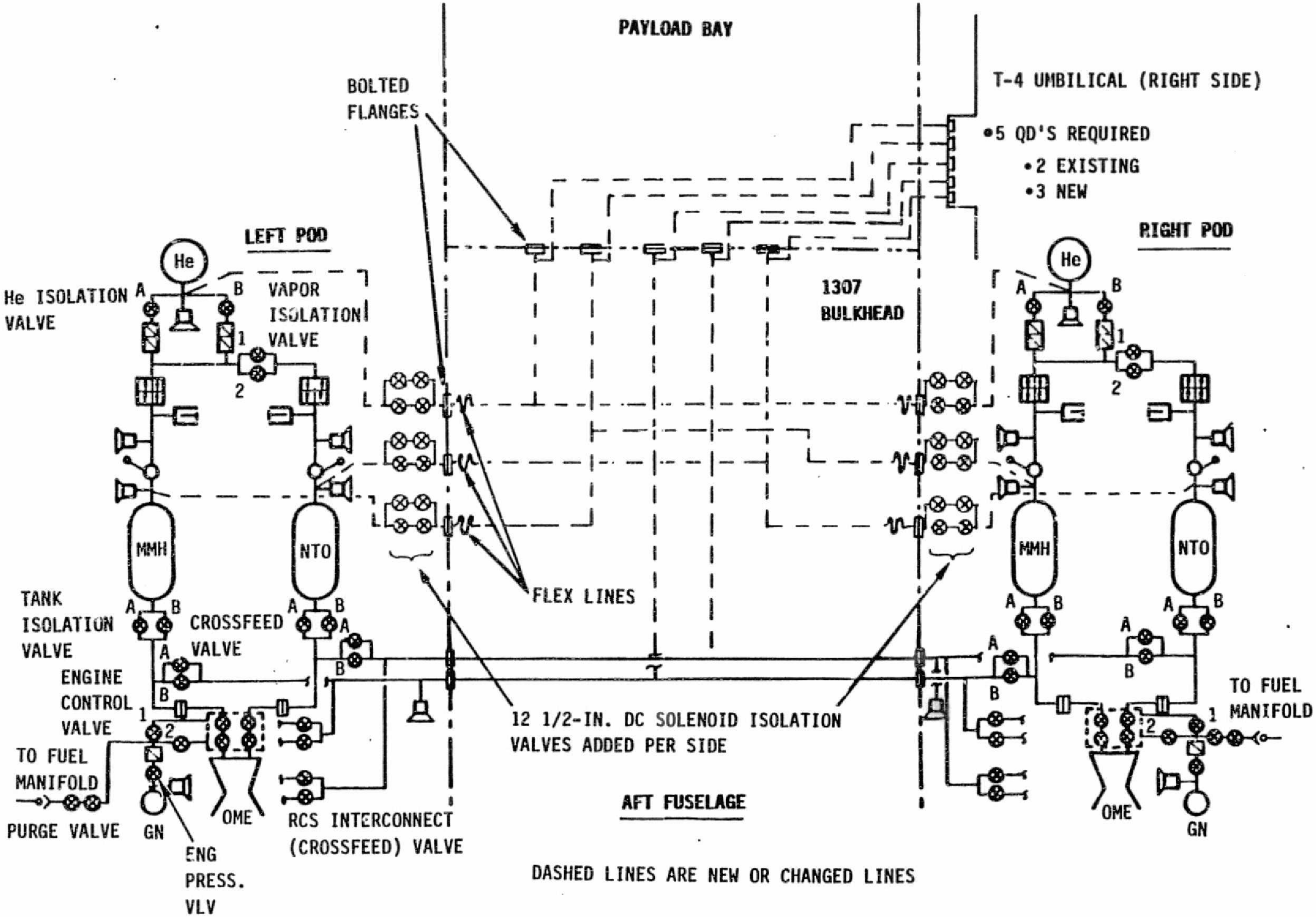


Figure 3. OMS Scar Plumbing Schematic

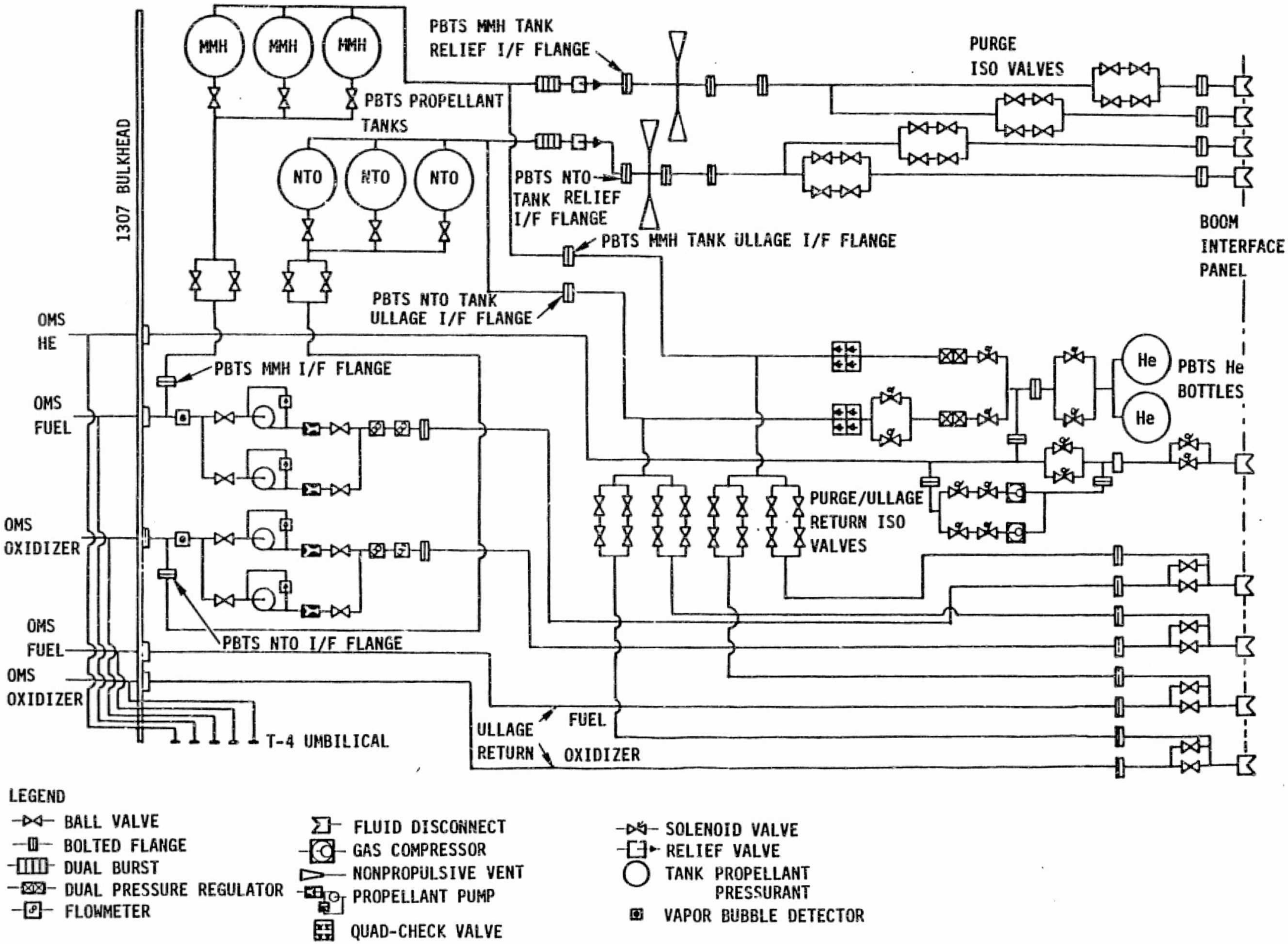


Figure 4. PBTS Schematic

Table 1. Scavenged Propellant Quantities

System	Number of Tank Sizes	Number of Scavenging Flights	Deliverable Propellant (1,000 lb)	Total Cost ($M)	Propellant Cost ($/lb)
Cryogenic propellants	1	69	926	75.6	82
	2	89	1,042	91.3	88
	3	89	1,182	106.5	90
	4	89	1,220	119.1	98
	5	89	1,247	128.4	103
	15 (max.)	93	1,321	223.4	169
Storable propellants	1	165	1,403	159.1	113
	2	165	1,491	175.9	118

determining performance except with modification required to account for the latest operational MPS propellant inventory and mission-required OMS propellant.

2. State-of-the-art technology shall be used to the highest degree possible.

3. Current STS specification for cryogenic and earth storable propellants shall be used.

4. On-orbit storage, if used, shall be located on or in the vicinity of the Space Station (28.5-degree inclination, 250-nautical mile altitude).

5. Assume 1991 initial operational capability (IOC) of a scavenging system that shall be compatible with the requirements of a Space Station, with OTV capability, or with other requirements for on-orbit propellant. Include the 1991 through 2000 decade in the study evaluation.

6. Payload manifest is based on MSFC nominal mission model, Revision 6.

7. The latest available STS cost per flight data shall be used as the basis for developing operational cost per flight estimates.

8. All costs shall be based on FY 1984 dollars.

9. Concepts requiring capabilities beyond the current capabilities of the STS main and on-orbit maneuvering engines shall not be considered.

10. The OMV shall be the earth-storable bipropellant, space-based concept initially located at the Space Station (28.5-degree inclination, 250-nautical mile altitude).

11. The OTV shall be space-based beginning in 1994 with two dry vehicles launched to orbit in that year.

12. Only non-Department of Defense (DOD) payloads with 28.5-degree inclinations shall be considered except that four DOD payload assist module (PAM) DII payloads per year shall be included.

13. The maximum payload weight is 65,000 pounds, even though BRM-1 lift capability is greater.

14. The maximum manifested payload length is 60 feet, which must include 6 inches for the dynamic envelope between payloads and a 4-foot clearance for extravehicular activity (EVA) or a 7-foot docking module (5,000 pounds) on Space Station flights.

15. An STS operator-chargeable weight of 3,600 pounds must be included on each flight whenever possible.

16. A maximum of five payloads or four PAM payloads can be manifested per launch.

17. Reflights are excluded as scavenging opportunities.

18. OMV propellant requirement costs must be included.

19. STS launch cost is $90 million (FY 1982 dollars), equivalent to $101.4 million in FY 1984 dollars.

1.4 RECOMMENDATIONS FOR FUTURE ACTIVITIES

During the course of this study, certain items that require further evaluation beyond the scope of the present study were discovered. The items include hardware, software, and basic STS capabilities. The following list identifies these items.

1. Development of CAS's for cryogenic propellant application.
2. Bipropellant pump development for storable propellants
3. Quick disconnect (QD) development for storable propellants
4. Helium compressor development
5. Flight software changes to support vehicle control requirements during on-orbit operations
6. Flight operations requirements for abort missions
7. Trajectory optimization to minimize performance penalties associated with delaying OMS 1 burn

8. Center of gravity (c.g.) limitations on the development of flight manifests

9. Accommodation of STS lift capability changes as program maturity is obtained

10. Entry and landing weight restrictions on normal and abort missions to be reassessed by 6.0 loads cycle

2. RESULTS

The data presented in this report shows that propellant scavenging for both cryogenic and storable propellants is feasible and cost-effective. The amounts of propellants that can be made available and the cost expressed as dollars per pound of propellant to orbit are presented in Table 1. The data is presented for different propellant tank sizes because the on-orbit requirements for propellant have not been defined. The data presented should be compared to the costs for an STS tanker, which would be $2,033 per pound to deliver propellant plus the development and production costs of the tankage set.

After the study was initiated, the goal of delivering 2.5 million pounds of cryogenic propellant to orbit was added to the study. The results of this evaluation are presented in Table 2. These data show that increased propellant to-orbit capability can be attained but that this can only be accomplished by creating more scavenging opportunities by adding STS flights to the flight manifest. The cost sensitivity to the quantity of cryogenic propellant achieved is presented in Figure 5. The data indicates a sharp increase in cost as the quantity of propellant increases beyond that available with the baseline manifest because the cost for each flight above 202 in the manifest is $101.4 million.

Table 2. Increased Cryogenic Propellant Quantities

Tankage Concept	Flights		Delivered Propellants (1,000 lb)	Total Cost ($M)	Propellant Cost ($/lb)
	Total	Scavenging			
Baseline manifest					
One tank size	202	69	926	75.6	82
Five tank sizes	202	89	1,247	128.4	103
Relaxed manifest					
Five tank sizes	212	103	2,222	1,174.7	529
	215	106	2,452	1,481.6	604
	216	107	2,507	1,583.7	632

PRECEDING PAGE BLANK NOT FILMED

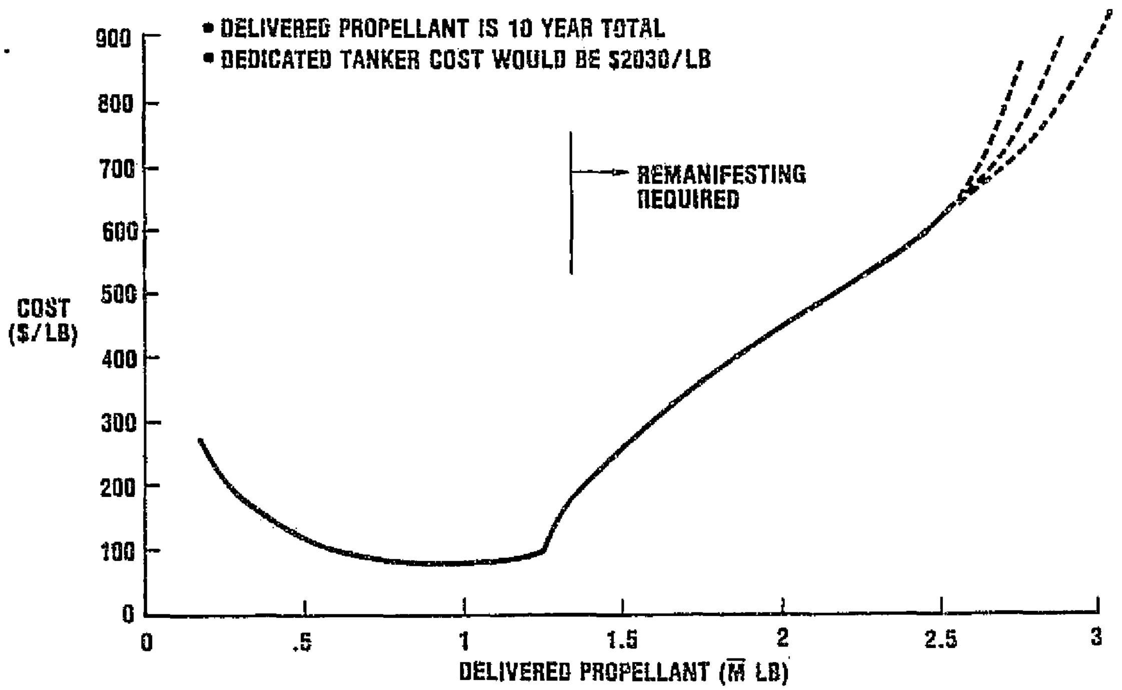


Figure 5. Cryogenic Propellant Cost Sensitivity

A specific result requested from this study was the mixture ratio (MR) at which propellants could be made available on orbit. The data presented in this report for cryogenic propellants is at an MR of 6. This value can be changed, however, once the desired on-orbit MR is selected simply by adjusting the design volumes of the payload bay tankage. This volume is selected based on design MR, propellant losses from heating, and trapped propellant residuals. For storable propellants, the data presented is for equal storage volumes, which is an MR of 1.6. This value could also be changed to any desired value by adjusting the payload bay tankage volume, but the maximum propellant available from the OMS tanks could only be available at the 1.6 MR. MR's above 1.6 can be obtained from the OMS tanks but only with a reduction of total propellant available because the maximum quantity of fuel that is available would not be required.

3. CONCLUSIONS AND RECOMMENDATIONS

The data presented in this report presents the quantity of propellant that can be scavenged; a point design for the optimum method of scavenging, including hardware and procedures; the costs required for the scavenging; and recommendations for tests necessary to develop and verify the scavenging concepts. The information is presented in greater detail for cryogenic propellant scavenging than for storable propellant scavenging because this was specifically requested in the contract.

The hardware systems designed are all considered to be the state of the art except for some specific storable propellant components that require development. Operationally, the procedures required do not present high risk to the STS program although certain test programs will be required to establish the best methods for system operations.

A number of items associated with hardware development, software changes, and basic STS performance capabilities have been identified during the study as requiring further study efforts. They are listed below:

1. CAS development for cryogenic propellants application
2. Bipropellant pump development for storable propellants
3. QD development for storable propellants
4. Helium compressor development
5. Flight software changes to support vehicle control requirements during on-orbit operations
6. Flight operations requirements for abort missions
7. Trajectory optimization to minimize performance penalties associated with delaying OMS 1 burn
8. Center of gravity limitations on development of flight manifest
9. Accommodation of STS lift capability changes as program maturity is obtained

4. STUDY REPORT

This section presents the results of the study. It begins with an evaluation of the quantity of propellants that potentially is available for scavenging. To do this, a flight manifest, based on NASA data, was developed, and then the STS lift capability was defined. In addition, an evaluation of the potential on-orbit users of storable propellants is included. This was done at the special request of NASA.

System trades that were performed will be described for propellant tank design, thermodynamic vent system (TVS) design, propellant heat loads, and capillary vent system sizing. These trades were necessary either to define the best design or to determine whether the assumed approach to a problem was feasible.

The rationale for the selection of cryogenic and storable propellant systems is presented next. This is followed by a point design of each of the system concepts selected. The point design includes operations; time lines; purging procedures; hardware design, including components; a system performance evaluation; and instrumentation and control system requirements. Test plans are delineated for those areas or components of the selected systems that may require long-term development.

A final evaluation was made during the study to determine the actual propellants that would be delivered to the storage depot at the Space Station. A discussion of this evaluation includes the factors of system hardware weight, heat losses, and trapped residuals. A technique for increasing the deliverable propellants is also presented.

4.1 PROPELLANT AVAILABILITY

The concept of propellant scavenging began with the thought of saving and using cryogenic propellants that would otherwise be discarded with the ET or dumped from the orbiter. The propellants to be saved could include unused reserves and residuals as well as excess usable propellant resulting from less-than-maximum-performance missions. The point of use for scavenged propellants would be the Space Station. For this study the concept of simply saving what would be lost has been broadened to include carrying cryogenic propellants as payload in the orbiter payload bay and carrying storable (OMS) propellant that exceeds basic mission requirements. This can be done by using the full OMS tank capacity and by providing payload bay tanks. The availability of propellants to be scavenged in any of these ways is the subject of this section.

It is important to recognize that excess lift capability is vital to scavenging even if only the cryogenic reserves and residuals are to be saved. Lift capability must be allocated to the weight of the scavenging system and to the extra OMS propellant needed for orbital delivery maneuvers. Only after those requirements are met can excess lift capability be converted into more scavenged propellant.

The propellant availability task relied on the NASA mission model for the years 1991-2000 to yield a set of STS missions for which excess performance, if any, was determined. Excess performance was turned into available cryogenic or storable propellant. Since this task preceded the identification of scavenging system weights, such weights were not deducted from lift capability at this point in the study. They are accounted for, however, in the tank sizing task and in the final evaluation of scavenged propellant quantities.

A special task included here identified potential users of scavenged storable propellant. The scavenging operation itself will use storable propellant in those cases where an OMV is needed to complete the delivery of propellants to the Space Station. This OMV propellant requirement was evaluated later in the study for selected scavenging systems.

4.1.1 STS Mission Descriptions

For the calculation of excess performance capability, hence propellant availability, it is sufficient to describe missions simply by their payload manifests (total weights and lengths) and their destination orbits. The MSFC mission model (FY 1983-2000), Revision 6, PS-01, provided the necessary mission data. Only the years 1991-2000 are included in this study.

The overall ground rules for defining missions, in addition to the MSFC model, were the existence of a Space Station in a 250-nautical mile, 28.5-degree orbit, the existence of an OMV to retrieve payloads from lower orbits, and the requirement to manifest payloads to obtain the minimum number of STS flights. Since scavenged propellants are destined for delivery to the Space Station, only payloads to 28.5-degree inclinations (or unrestricted inclinations) were considered. Manifesting for the minimum number of flights means that the scavenging concept is accorded no special help, and scavenging systems must fly on an "opportunity" basis.

Several additional ground rules and assumptions were made to ensure reasonable manifests. They are listed below:

1. Composite STS payloads will not exceed 65,000 pounds or 60 feet in length (including major payloads, STS operator-chargeable weight, ancillary payloads, STS docking module, payload bay egress length, and dynamic envelope).

2. No more than one major altitude change (>30 nautical miles) will be made per mission to avoid excessive OMS propellant requirements.

3. Whenever possible, an STS operator-chargeable weight of 3,600 pounds and an STS ancillary payload weight of 2,000 pounds were included. These represent getaway specials; student experiments; mid deck experiments, such as continuous flow electrophoresis system (CFES), multilatex reactor (MLR), etc.; crew experiments; payload attachment hardware; and extra crew seats, crew members, and power reactant

storage and distribution (PRSD) tanks. These payloads or experiments have no length in the payload bay (usually they are placed in the mid deck or under the payload bay liner), but their weight must be included.

4. No identical payloads would be manifested on the same flight. This allows the recognition of realities in allowing for periodic flights (such as four per year Space Station resupply missions) and that insurance for commercial payloads usually will not allow an entire commercial system to be flown on one flight (such as two Telstars or two Palapas).

5. People's Republic of China communication payloads should not be comanifested with DOD PAM-D's.

6. LEO station and LEO platform payloads may be comanifested.

7. MPS payloads may fly at any altitude but should not fly with tethered satellite system (TSS) payloads.

8. An upper stage mission may be launched from the orbiter at any altitude. This allows the deployment of upper stage payloads from Space Station altitudes.

9. A space-based advanced orbital transfer vehicle (AOTV) was assumed. To accommodate this, two empty AOTV's were launched to the Space Station in 1994 and used for all subsequent payloads requiring the AOTV. The weight and length of the AOTV, plus airborne support equipment (ASE), were deducted from the MSFC mission model data to obtain the payload's weight and length, and only these values (+10 percent for ASE) were included in the manifest. The AOTV weight was taken to be 5,930 pounds dry (48,840 pounds wet). It was 29 feet long, and the ASE weighed 6,000 pounds.

10. DOD PAM-D's were baselined to be 15,867 pounds and 11.5 feet long (3,600 pounds down weight). All other PAM-D's would weigh 10,100 pounds and be 8 feet long (3,600 pounds down weight).

11. All STS flights must include at least 4 feet of clearance in the payload bay to allow contingency EVA egress or 7 feet for a docking module on any Space Station or space platform flight. Three inches on each side of the payload must be kept clear as a dynamic envelope (1/2 foot between payloads).

The last three ground rules were imposed on this analysis after an initial review of the manifest.

Toward the end of the study, it was agreed that the 2,000 pounds for ancillary payloads be deleted. This change is reflected in the final manifests and performance data.

The ten flight manifests prepared for the years 1991-2000 are presented in Section A.1 of the appendix. Each table lists the flight number, major payload, length (individual payloads and total), weight up and down, and destination altitude for each payload. Under the adopted ground rules, 202 flights were required to carry all 28.5-degree inclination payloads, with an overall load factor of 94 percent. For many flights the manifests show a payload altitude of 160 nautical miles and direct insertion is indicated. These cases signify an upper stage, which may optionally launch from 160 nautical miles in accordance with the ground rules. Unless a flight carries a payload that must go to 250 nautical miles, the flight has been assumed in using the flight manifests to be a standard insertion to 160 nautical miles to gain better STS performance.

Table 3 summarizes the resulting ten-year schedule by orbital destination. The category of "other" orbits includes flights to higher altitudes not compatible with scavenging. Flights to 160 and 124 nautical miles refer to the deployment of a TSS at 124 nautical miles after other payloads are delivered at 160 nautical miles. These flights are the only ones requiring a major altitude change.

Three payload considerations have been ignored here consistent with the preliminary nature of the payload manifesting exercise. Orbiter c.g. restrictions are a concern in today's STS flight plans and may disallow some of the flight manifests developed for this study. Also no consideration was given to payload specific requirements (such as lighting) when payloads were combined. Greater realism in these areas could result in lower STS load factors and more excess performance to devote to scavenging.

The third payload consideration ignored in manifesting is the orbiter landing weight limit, currently set at 211,000 pounds for a normal mission.

Table 3. Flight Manifest Summary

Altitude (nmi)	Number of Flights per Year										
	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	Total
250	8	8	9	12	11	10	9	10	10	11	98
160	6	10	4	11	8	10	8	5	3	8	73
160 and 124	1	1	2	1	2	2	1	2	2	1	15
Other*	2	1	2	2	1	0	3	2	1	2	16
Total	17	20	17	26	22	22	21	19	16	22	202

*Flights to "other" orbits do not offer scavenging because of lift capability and/or length constraints.

This figure is based on nominal end-of-mission OMS and reaction control system (RCS) loads and a 32,000-pound return payload. In an intact abort situation, landing with a 65,000-pound payload (240,000 pounds total weight) is a requirement that will be met by dumping propellants, flying a shallow glide slope, and restricting the landing speed and sink rate. The ongoing 6.0 loads analysis, to be completed in two years, will include landing weights as large as 256,000 pounds in order to probe the limits of orbiter capability. The results of that analysis will apply to Orbiter Vehicle (OV) 103 and subs and may indicate safe landing weights larger than the current limit.

Of the 202 flights manifested for this study, 37 have return payloads that weigh more than 32,000 pounds, the largest being 45,700 pounds. These were allowed on the basis that the return payload limit is somewhat soft, considering the discussion in the previous paragraph. In addition, the return payload weights are less certain than those of up payloads due to uncertainty in the consumables included. In all cases a reduction of return payload weight would logically require a corresponding reduction in up payload weight, creating additional flights and more lift capability and space on the original flights. Thus, eliminating all 37 "overweight" cases would greatly increase the amount of propellant potentially scavengeable.

It is appropriate to mention here that many of the potential scavenging flights identified in Section 4 carry a combined weight of up payload plus scavengeable propellant exceeding the nominal 65,000-pound limit. This poses no problem for ascent (all weights are within the lift capability), and by dumping scavenged propellant the abort landing weight could be brought within the limit in all but a few cases. In these cases the weight of scavenging system hardware plus payloads would exceed 65,000 pounds, but the amounts are considered to be within the tolerance of a preliminary mission analysis.

4.1.2 STS Performance Capability

With the flight manifests developed above, only three mission profiles need to be considered in the propellant availability task and the scavenging tank sizing tasks. In the simplest terms these are:

1. Standard insertion to 160 nautical miles, propellant delivery to OMV, other payload operations, and deorbit from 160 nautical miles

2. Same as No. 1 but transfer to 124 nautical miles prior to deorbit

3. Direct insertion to 250 nautical miles, propellant delivery to Space Station, other payload operations, and deorbit from 250 nautical miles

Detailed time lines for the propellant delivery part of these missions are covered in a later section, but these basic descriptions plus appropriate ground rules are sufficient to define the STS performance capabilities in each case. The 16 flights shown going to other altitudes in Table 3 have also been evaluated and found to lack significant scavenging potential. Their high altitudes tend to require full lift capability and nearly full OMS loads. They have not been considered further.

Table 4. BRM-1 Description From JSC-09095-74

Ground Rules		Weight Summary (lb)	
Missions	BRM-1	Current manager's reserve	5,287
Launch date	Winter	Payload	65,000
Launch site	ETR	Lift capability	70,287
Orbiter	OV-103		
External tank	LWT-7	STS operator	---
EPS tank sets	3	Orbiter inert	146,843
Software release		SSME X 3 inert	20,655
Crew/days	4/7	Personnel	3,505
SSME throttle settings (%)	109/109	Nonpropellant consumables	4,749
SSME Isp	Spec nom	OMS propellant	19,700
SRB	MWC/HPM	RCS propellant	7,416
Ascent aerodynamics	IVBC-3	MPS propellant unexpended	13,554
Trajectory shaping	Nominal	MPS pressurants	423
Altitude (nmi)	150	ET inert	67,020
Inclination (deg)	28.45	Shuttle system at MECO	354,152
Design season	Winter		
Q-M design constraint	TPS limit	Nonpropulsive weight loss (4)	442
Max Q	720	SRB X 2 inert	375,382
Load relief Q-alpha	-2,000	SRB X 2 propellant	2,216,964
Entry target line	Steep	MPS propellant expended	1,575,528
		Shuttle system at SRB ignition	4,522,468

The primary study ground rule for STS performance is the assumption of BRM-1 lift capability per the August 24, 1984, Green Book adjusted for the Revision 5B propellant inventory. The information in Table 4, taken from the Green Book, summarizes BRM-1 conditions and capability. Tables 5 and 6 present the original and revised inventories, respectively, and Table 7 shows the steps required to adjust the propellant inventory. The change in propellant load at lift-off affects propellant consumption at MECO by a factor that must be evaluated from trajectory simulations. The value of $W\,W_p/W\,W_o = 0.91$ used here came from a recent Rockwell evaluation of BRM-1.

The adjusted manager's reserve of 6,646 pounds is really excess usable MPS propellant at MECO and applies only to the BRM-1 MECO conditions and weights for payloads (65,000 pounds) and OMS load (19,700 pounds). The set of potential scavenging flights, however, involves a variety of payloads and requires a variety of OMS loads. Also, those flights going by direct insertion (DI) to 250-nautical mile altitudes reach a different MECO velocity and flight path angle. Thus, further adjustments to the basic BRM-1 lift capability are required before scavenging flight performance can be evaluated.

Table 5. BRM-1 MPS Propellant Inventory

	LH_2	LO_2	Total
Loaded	231,480	1,384,200	1,615,680
Orbiter lines	249	3,304	3,553
SSME X 3	58	1,325	1,383
ET (HXT = 1046.3, HUP = 0.3, LXT = 420.0, LUP = 0.1)	231,173	1,379,571	1,610,744
Loss before engine start command (DBT--8M:54S)	104	10,000	10,104
Boiloff, bleed, seal leak, pogo flush	104	10,000	10,104
Load at engine start command	231,376	1,374,200	1,605,576
Orbiter lines	249	3,304	3,553
SSME X 3	58	1,325	1,383
ET	231,069	1,369,571	1,600,640
Transferred from ET	58	172	230
Used for THR buildup and SRB ignition delay	1,733	9,434	11,167
Load at SRB ignition command	229,643	1,364,766	1,594,409
Orbiter lines	249	3,304	3,553
SSME X 3	116	1,497	1,613
ET	229,278	1,359,965	1,589,243
Unusable	2,113	4,798	6,911
Orbiter lines (LO_2 ECO T = 3.398, NPSP = 7.2)	249	550	799
SSME X 3	58	1,325	1,383
ET wet walls, bellows	0	175	175
ET: LO_2 lines; LH_2 lines and tank	670	0	670
Flight press	1,136	2,748	3,884
Usable reserves	1,785	4,628	6,413
Orbiter lines (FPR)	0	2,754	2,754
SSME X 3	0	0	0
ET (FPR)	769	1,874	2,643
Bias	1,016	0	1,016

Table 5. BRM-1 MPS Propellant Inventory (Cont)

	LH_2	LO_2	Total
Usable impulse	225,745	1,355,340	1,581,085
Used at OBMR	224,970	1,353,875	1,578,845
Shutdown consumption	717	1,293	2,010
0 SSME from nominal MTS percent	0	0	0
3 SSME from 65 MTS percent	717	1,293	2,010
Vented after SSME valve closure	58	172	230

Notes:
Nominal T = T1 + 0.00
Fuel bias = 1,016, MTS nominal/abort = 109,109, MR = 6.0000
Total FPR = 5,397, Ullage O_2, = 2.25, Sigma = 3, OMBR = 6.0180
Delta FPR = -300, Ullage H_2 = 1.63

The adjustment for variable ascent payload (PL) and OMS load (W_{oms}) is easily made using the W W_p/W W_o factor since a change in either of these items is simply a change in launch weight, W_o. Thus for excess MPS propellant, we have (E_{mps}, pounds)

$$\begin{aligned} E_{mps}(SI) &= 6,646 - 0.91\ [(PL - 65,000) + (W_{oms} - 19,700)] \\ &= 83,723 - 0.91\ (PL + W_{oms}) \end{aligned} \tag{1}$$

This equation applies to standard insertion (SI) flights. Adjustment for direct insertion conditions was estimated in two ways using the JSC Red Book data base for generic standard and direct insertion ascent trajectories. Table 8 shows the steps involved.

In the second column, the BRM-1 MECO weight from Table 4 is noted. The third column shows the standard insertion data corrected for 109-percent Space Shuttle main engine (SSME) thrust, and the fourth column shows direct insertion data corrected for a 250-nautical mile apogee. The difference between the standard and direct insertion cases is a decrease in payload weight of 9,149 pounds for the direct insertion case. This translates (by the 0.91 factor) into a delta E_{mps} of 8,326 pounds. Finally, a correction for the large difference in MPS load between BRM-1 and the Red Book propellant inventory is shown in the fourth column. The final MECO weight for direct insertion is 8,598 pounds less than the BRM-1 MECO weight, but there is significant uncertainty in this value due to the large correction for propellant load and some doubt about the consistency between the BRM-1 and Red Book ascent performance calculations. In addition direct insertion target conditions have been revised since the October 1982 Red Book issue and the effect of this is

Table 6. Revised MPS Propellant Inventory

	LH_2	LO_2	Total
Loaded	231,650	1,388,379	1,620,029
Orbiter lines	249	3,304	3,553
SSME X 3	58	1,325	1,383
ET (HXT = 1044.6, HUP = 0.3, LXT = 412.6, LUP = 0.4)	231,343	1,383,750	1,615,093
Loss before engine start command (DBT--5M:0S)	104	5,700	5,804
Boiloff, bleed, seal leak, pogo flush	104	5,700	5,804
Load at engine start command	231,546	1,382,679	1,614,225
Orbiter lines	249	3,304	3,553
SSME X 3	58	1,325	1,383
ET	231,239	1,378,050	1,609,289
Transferred from ET	58	172	230
Loss for THR buildup and SRB ignition delay	1,733	9,434	11,167
Load at SRB ignition command	229,813	1,373,245	1,603,058
Orbiter lines	249	3,304	3,553
SSME X 3	116	1,497	1,613
ET	229,448	1,368,444	1,597,892
Unusable	2,223	4,770	6,993
Orbiter lines (LO_2 ECO T = 0.478, NPSP = 6.4)	249	522	771
SSME X 3	58	1,325	1,383
ET wet walls, bellows	0	175	175
ET: LO_2 lines; LH_2 lines and tank	780	0	780
Flight press	1,136	2,748	3,884
Usable reserves	1,594	4,195	5,789
Orbiter lines (FPR)	0	2,782	2,782
SSME X 3	0	0	0
ET (FPR)	694	1,413	2,107
Bias	900	0	900

Table 6. Revised MPS Propellant Inventory (Cont)

	LH_2	LO_2	Total
Usable impulse	225,996	1,364,279	1,590,275
Used at OBMR	225,329	1,362,838	1,588,167
Shutdown consumption	609	1,269	1,878
0 SSME from nominal MTS percent	0	0	0
3 SSME from 65 MTS percent	609	1,269	1,878
Vented after SSME valve closure	58	172	230

Notes:
Nominal T = T1 + 0.00
Fuel bias = 980, Throttle setting (nominal/abort) = 104/104, MR = 6.0301
Total FPR = 4,888, Ullage O_2, = 1.57, Sigma = 3, OMBR = 6.0482
Delta FPR = 0, Ullage H_2 = 1.55

Table 7. BRM-1 Inventory Adjustment

	Propellant (lb)		
	Revision S1 Inventory	Revision 5B Inventory	Delta
Load at SRB ignition command	1,594,409	1,603,058	+8,649
Unusable	6,911	6,993	
Usable reserve	6,413	5,789	
Usable impulse	1,581,085	1,590,275	+9,190
MPS expended	1,575,528*	1,583,399	+7,871**
Vented	230	230	
Manager's reserve	5,327*	6,646	+1,319***

*From Table 3.3 of Green Book. Note 40-lb discrepancy in Green Book manager reserve.
**W Expended/W loaded = 0.91 for this trajectory.
***Excess MPS varies as -0.91 x W (payload + OMS load).

unknown. In the end it was decided conservatively to use 8,598 pounds as the delta E_{mps} figure, yielding

$$E_{mps}\ (DI) = 75{,}125 - 0.91\ (PL + W_{oms}) \qquad (2)$$

Equations 1 and 2 express ascent performance capability to MECO, but additional data and assumptions are needed to define the OMS propellant requirements for the remainder of the flight maneuvers.

Table 8. Performance Adjustments for Direct Insertion to 250-Nautical Mile Apogee

	Mission		
	BRM-1	Standard Insertion to 160 nmi	Direct Insertion to 250 nmi
Data source	Green Book JSC-09095-74 (8-24-83)	Red Book JSC-17332 Rev. A (9-82)	Red Book JSC-18628 (10-82)
MECO weight (lb)	354,152	345,659	344,439
Conditions	BRM-1 Rev. S1 inventory	• q_{max} 719 psf • 102% SSME	• q_{max} 720 psf • 240 nmi apogee
For 109% SSME		+7,505	
For 250 nmi apogee			-424
Adjusted MECO weight		353,164	344,015
For 17,103-lb difference in MPS loading			+1,539
Adjusted MECO weight			345,554

Orbiter weight for OMS maneuvers is obtained from the weight summary in Table 4 by adding the weight of the inert orbiter, SSME's, personnel, nonpropulsive consumables, and RCS propellant, giving 183,168 pounds. In performance calculations, the RCS load has not been depleted, resulting in a conservative estimate of OMS consumption. The orbiter's weight is increased by payloads (up and down) and scavenged propellants as appropriate to each flight. In tank sizing and final performance calculations, the scavenging system weight is also included.

OMS performance ground rules are given in Tables 9 and 10. The OMS propellant inventory comes from the JSC 104-percent Generic Performance Study ground rules and the delta-V's from Rockwell trajectory data. The 35-fps rendezvous allowance was derived from a Space Station user reference mission analysis and applies to all flights on which the orbiter delivers propellant to the Space Station (i.e., all scavenging flights to 250 nautical miles). The apogee delta-V for direct insertion is based on the MECO condition given in the direct insertion Red Book. Additional OMS delta-V data are shown as a function of altitude in Figure 6.

Table 9. OMS Propulsion Data

	Weight (lb)
Tank capacity	25,064
Unusable propellant	
Residuals	799
Gaging error	518
Engine failure	125
FPR (1.5% full load)	349
Total	1,791
Maximum usable propellant	23,273
Specific impulse	313.2 seconds

Table 10. OMS Delta-V's for Scavenging Missions

	Insertion/Deorbit Altitude (nmi)		
	160/160	160/124	250/250
Delta-V with up payload and scavenging propellants (fps)			
• Ascent	433	433	380
• Rendezvous	-	-	35
Delta-V with down payload (fps)			
• Payload deployment/retrieval	20	20	-
• Hohmann transfer	-	127	-
• Deorbit	297	273	413

The analysis of STS performance for each flight moves in reverse from deorbit back to MECO. The steps, in principle, are as follows:

1. Compute OMS propellant (OP_2) for deorbit and other postscavenging maneuvers with the down payload.

2. Compute OP_1 for ascent and rendezvous maneuvers carrying the up payload, scavenged propellants, and OP_2.

3. Compute E_{mps} from Equation 1 or 2 with $W_{oms} = OP_1 + OP_2$ and PL = manifested up payloads plus any extra storable propellant carried for scavenging.

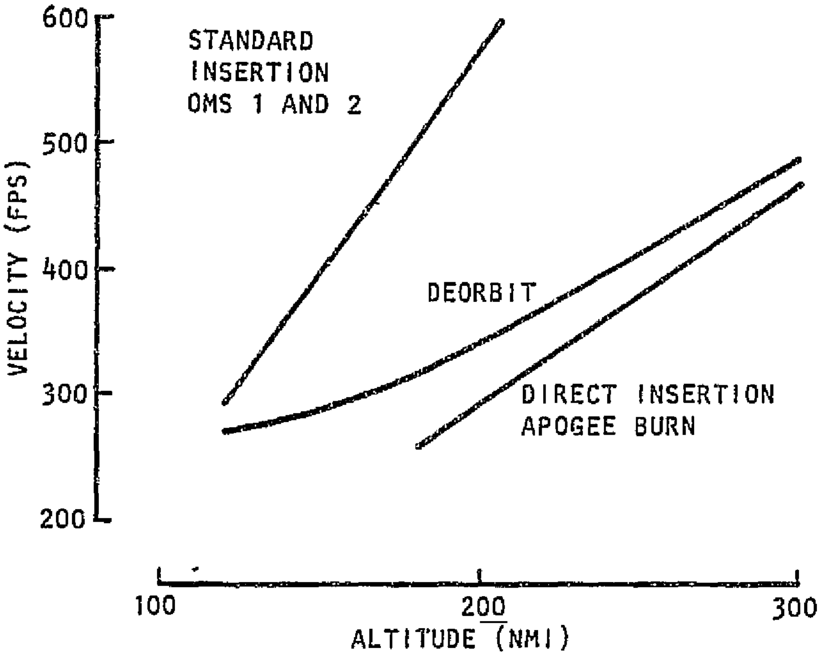


Figure 6. OMS Delta-V Requirements

When scavenging system weights are considered, they are included in both the up and down payloads. In actual practice these steps are combined in two relationships among up payload, down payload, scavenged cryogenic propellant, and scavenged storable propellant. One of these relationships follows from the ascent performance limit ($E_{mps} = 0$ in Equation 1 or 2) and the other from the OMS tank limit, which requires $OP_1 + OP_2 +$ unusable $\leq$ tank capacity. Simple modifications are made to treat the scavenging of cryogenic reserves and residuals and the use of payload bay tanks for additional storable propellant. Computer programs were developed for the automatic analysis of the 202 flights.

When this analysis began, the emphasis was on post-MECO scavenging of excess and residual cryogenic propellants. Later, emphasis shifted to the prelaunch loading of cryogenic propellants in payload bay tanks to use up any predicted MPS propellant margin. However, the calculations for cryogenic propellant scavenging were not changed to reflect prelaunch loading. As a result, the cryogenic propellant weights shown for scavenging in this report are conservative by an estimated 5 to 6 percent because of the 0.91 factor relating E_{mps} to launch weight. This situation does not apply to storable propellant scavenging results.

An additional product of this analysis is the general payload-to-altitude curve in Figure 7. This may be used to compare the performance ground rules and assumptions used here with those of other studies. The mission profile for this plot consists of payload delivery, 20 fps for deployment, and deorbit with no down payload. Other delta-V's are taken from Figure 6.

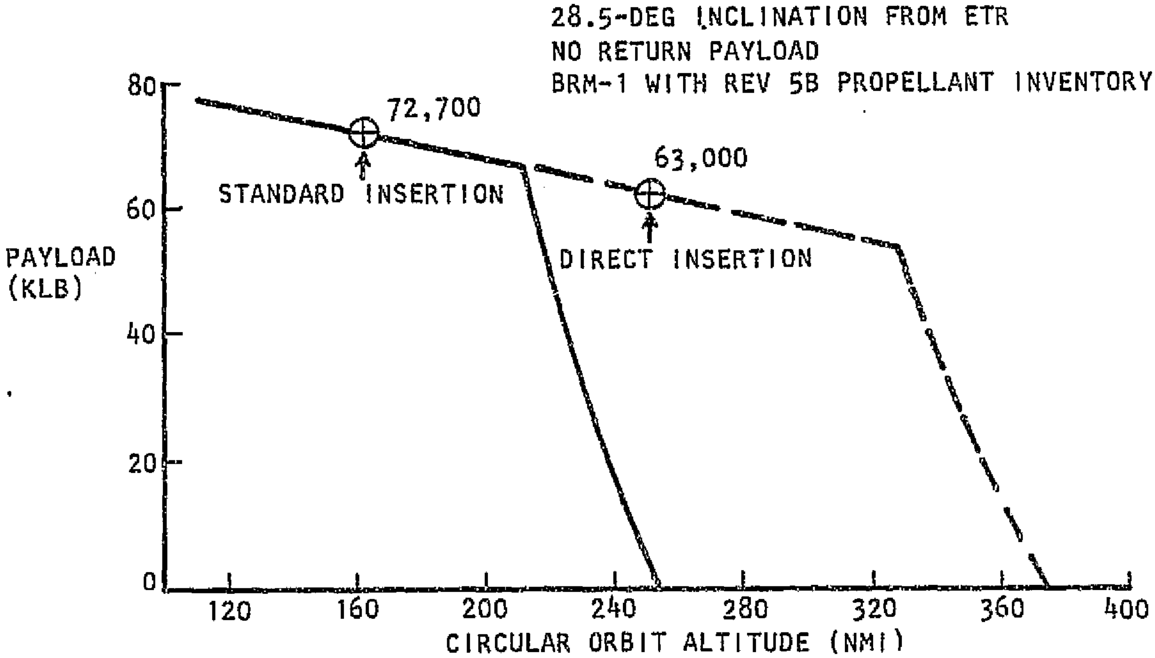


Figure 7. Payload Delivered Vs. Circular Orbit Altitude

4.1.3 Availability Analysis

Combining the ten-year flight manifest with the STS performance capability just described yields the amounts of cryogenic and storable propellants potentially available for scavenging. This analysis was first performed early in the study to provide an indicator of the overall attractiveness of scavenging and as a guide to the development of scavenging concepts. Alternative concepts and designs can be rated partly on how much of the potentially available propellant they are actually able to deliver. During the study, modifying the ground rules had some effect on propellant availability. The information presented in this section reflects updates of the final study ground rules discussed in the previous sections, in particular the deletion of ancillary payloads.

Cryogenic propellants are potentially available from two sources: excess MPS propellant on less-than-maximum payload flights and the reserve and residual propellants normally remaining at MECO even on maximum payload flights. The MPS propellant is loaded to fixed sensors in the ET for each flight. The excess then varies because impulse propellant on each flight varies due to payload weight, MECO targets, and OMS loading for post-MECO maneuvers. The availability of reserve propellant depends on the system and environmental dispersions experienced on each flight, which may be good or bad for propellant consumption. This study assumed that the baseline propellant inventory reserve and residuals would be available on each flight since on the average that is true.

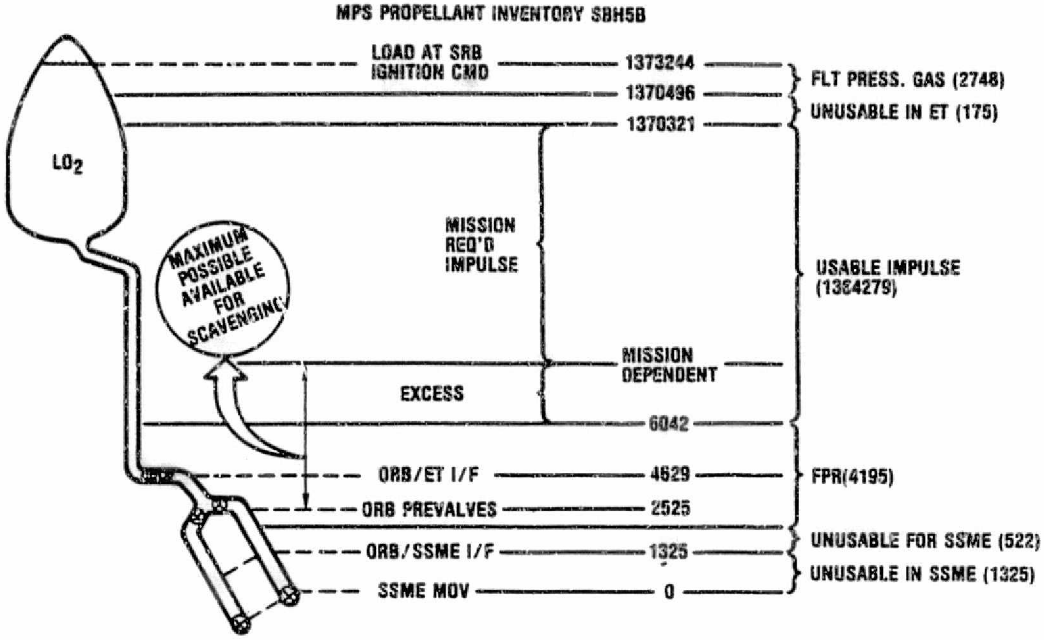


Figure 8. LO_2 Propellant Allocation

Figures 8 and 9 illustrate the complete allocation of MPS propellants, LO_2 and LH_2, respectively, according to the baseline SBHB inventory. Propellants are considered to be available potentially all the way to the orbiter/SSME interface. Thus, the LO_2 reserve and residual propellant contributes

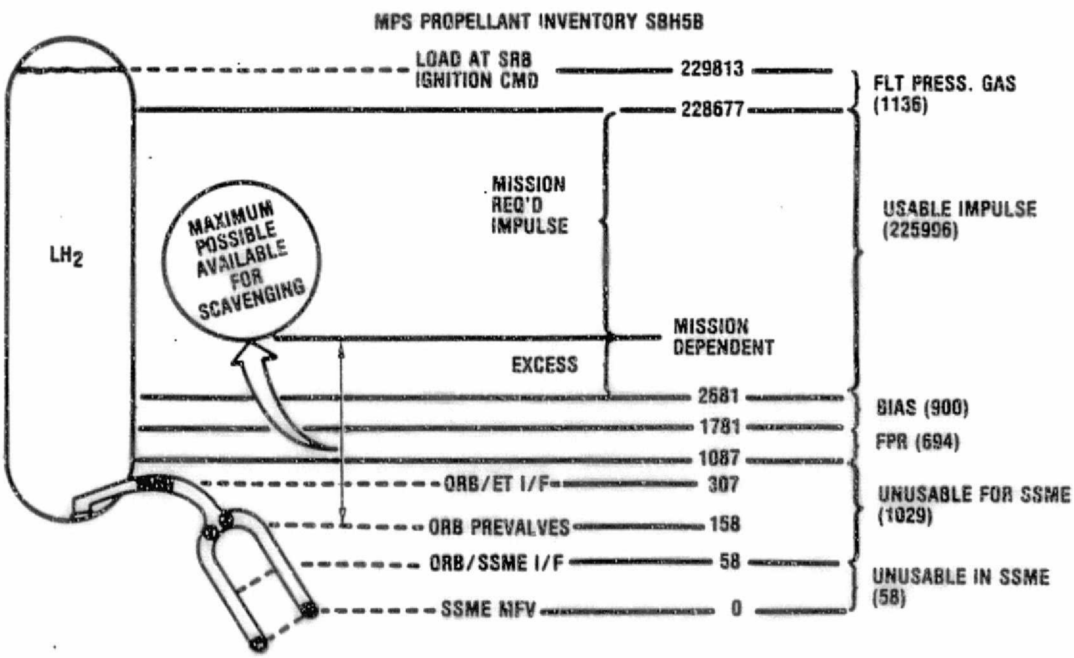


Figure 9. LH_2 Propellant Allocation

4,717 pounds and there are 2,623 pounds of LH_2, for a total of 7,340 pounds of potentially available reserves and residuals. This value has been used here in the cryogenic propellant availability calculations. The selected concept for post-MECO scavenging, however, will obtain propellants only down to the orbiter prevalves. In this case the corresponding numbers for LO_2 and LH_2 are 3,517 pounds and 2,523 pounds. In addition, 350 pounds of LH_2 will be left in the ET due to feed line dropout, leaving a total of 5,690 pounds of scavengeable reserves and residuals. This is the value used later in determining the cryogenic propellant deliverable by the selected tank configuration.

The evaluation of the potential cryogenic propellant available made one concession to the realities of scavenging system design: a length of 2 feet in the cargo bay was assumed as a minimum for even the smallest system. This constraint and the available STS performance capability eliminated scavenging on 106 out of the 202 manifested flights because there is no room in the payload bay and/or there is no excess performance. Table 11 shows the resulting total number of scavenging flights and propellant quantities by year. The

Table 11. Potential Excess Cryogenic Propellant Availability for 1991-2000

	Year										
	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	Total
Total flights	17	20	17	26	22	22	21	19	16	22	202
Excess performance but excess length <2 ft	2	5	4	13	9	6	4	10	8	8	69 (34%)
Excess length ≥2 ft but no excess performance	1	0	2	3	2	2	2	4	3	2	21 (10%)
No excess performance or length	2	1	1	3	1	0	3	2	1	2	16 (8%)
Potential scavenging flights	12	14	10	7	10	14	12	3	4	10	96 (48%)
Excess MPS (1,000 lb)	191	293	102	120	243	331	254	70	61	168	1,833
Propellant reserves (1000 lb)	85	99	71	50	71	99	85	21	28	71	680
Total (1,000 lb)*	276	392	173	170	314	430	339	91	89	239	2,513

*Total propellant available from potential scavenging flights only.

remaining 96 flights provide 1.83 million pounds of cryogenic propellant from excess performance and another 0.68 million from MPS reserves and residuals, for a total of 2.51 million pounds of potentially available propellant. This quantity does not consider any reduction that is required for the scavenging system weight or propellant losses from boiloff, propellant transfer, or trapped propellant. It does make allowances in ascent performance, however, for the OMS propellant needed to push the weight of scavenged propellant through ascent and on-orbit OMS maneuvers.

Storable propellant scavenging can be accomplished by loading OMS propellant in the orbiter OMS tanks beyond the amount required by the mission or by loading tanks containing storable propellant in the payload bay if space is available. Either approach requires excess ascent performance, but since all 202 manifested flights offer extra room in the OMS tanks, no flight was eliminated because it lacked payload bay space. Table 12 shows that additional propellant can be loaded in the OMS tanks on 165 flights, for a total of 0.80 million pounds. Additional performance capability exists to carry 2.48 million pounds in payload bay tanks (for a grand total of 3.28 million pounds). Again this quantity does not consider the weight of the scavenging system or payload bay tanks but does consider the increased OMS propellant required by OMS maneuvers.

This study recognized the possibility of scavenging some OMS reserves before the deorbit burn, but this option is operationally unattractive. However, some of the reserve items in Table 9 are believed to be quite conservative and may be reduced in the future, freeing more OMS tank capacity for scavenging.

Table 12. Potential Excess Storable Propellant Availability for 1991-2000

	Year										
	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	Total
Potential scavenging flights--includes excess length < 2 ft	14	19	14	20	19	20	16	13	12	18	165
Excess OMS from orbiter tanks (1,000 lb)	68	96	60	104	91	106	83	65	54	88	815
Excess OMS from orbiter and payload bay tanks (1,000 lb)	229	388	180	390	371	503	358	307	216	335	3,277

The scope of the propellant availability analysis was limited to scavenging cryogenic propellant or storable propellant but not both on the same flight. It is instructive, nonetheless, to consider the fundamental tradeoff between cryogenic and storable scavenging imposed by the ascent performance limit and the OMS tank limit. Figure 10 displays the possibilities presented by three typical flights chosen from the ten-year set. In the first case (Flight 15 in 1991), A represents potentially available cryogenic propellant with no available storable propellant. This point is determined by ascent performance capability, as is the entire line from A to C. Point B represents potentially available storable propellant with no available cryogenic propellant, determined by the OMS tank capacity limit line. With the option of payload bay tanks for storable propellant, the available amount jumps to C on the ascent performance limit line. Points A, B, and C are the cases investigated in this study.

From the viewpoint of overall transportation efficiency, the small slope of the OMS tank limit line is significant. Beginning at B, it is possible to provide cryogenic propellant for scavenging with very little loss in available storable propellant, using only the OMS tank capacity. The intersection of the constraint lines at D yields the maximum sum of available propellants of both types. The near 1-to-1 slope of the ascent performance limit line means that the sum of available propellants is nearly constant anywhere along this line, but storable propellant payload bay tanks are required in the region from D to C.

The other cases shown in Figure 10 illustrate similar tradeoffs for other payload weights and orbit destinations. The second case exhibits a very large

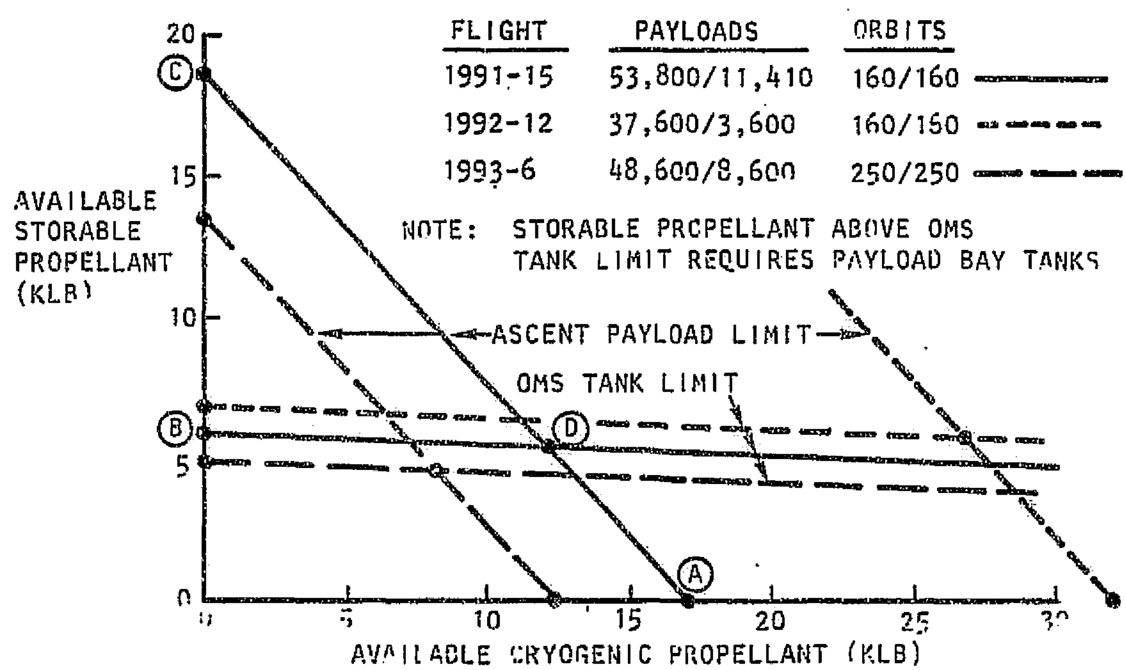


Figure 10. Typical Tradeoffs for Scavenging Cryogenic and Storable Propellants

cryogenic propellant capability due to the relatively small up payload. The third case exhibits reduced availability of both propellants due to the 250-nautical mile orbit destination.

Lack of payload bay space would generally preclude carrying both types of tanks on one flight, but scavenging cryogenic propellant, plus storable propellant from the OMS tanks, is a possibility. Future interest in combined scavenging schemes would depend on the perceived demand for each type of propellant and the desired size and frequency of individual deliveries.

4.1.4 Relaxed Flight Manifest

After the study had started and the flight manifest was defined, NASA defined a propellant delivery requirement of 2.5 million pounds of cryogenic propellant to the space-based depot. Within the constraints on scavenging defined by this study, the only method available to attain this goal is to create more scavenging opportunities. This is accomplished by creating more STS flights ("relaxing" the baseline flight manifest). Specifically, flights that are full with multiple payloads and that are not scavenging flights in the baseline manifest are split into two flights with scavenging performed on both flights. The cost associated with each additional flight must be borne by the scavenging system.

Table 13 presents one baseline flight for each year of the 1991-2000 decade that was split into two flights. The number of additional flights required depends on the amount of propellant required on orbit. This concept is limited because only a given number of flights fit the requirements.

4.1.5 Storable Propellant Users

A Rockwell IR&D study (Project 84286) has identified a number of potential bipropellant users in the four orbital inclinations of 28.5 to 33 degrees, 56 to 57 degrees, 63.4 to 65 degrees, and 90 to 99.5 degrees. These on-orbit bipropellant quantity requirements were determined for the years 1990 through 2000.

From the mission scenarios that had worst-case usage of bipropellants, the OMV required approximately 90 percent of the on-orbit resupply quantities to perform satellite deployment, retrieval, refurbishment, and resupply operations. The mission traffic models also indicated that resupplying the OMV by scavenging the Shuttle orbiter OMS pod propellants is highly desirable. It was also assumed that orbiter resupply of the OMV occurred in the traffic model.

Other ground rules for the propellant resupply traffic model included the following:

- The OMV IOC was 1990, with ground-basing in 1990 and 1991, and space-basing for 1992 through 2000.

- All satellites are deployed from the orbiter payload bay.

Table 13. Relaxed Manifest Propellant Availability

Flight No.	Baseline Propellant Available (lb)	New Flights	Propellant Available for Each New Flight (lb)	Delta Propellant Available (lb)
91-17	14,700	A B	51,237 49,511	86,048
92-12	25,200	A B	50,095 53,827	78,722
93-11	14,700	A B	33,389 42,719	61,408
94-24	0	A B	56,402 49,870	106,272
95-16	16,800	A B	56,290 48,732	88,222
96-20	6,300	A B	5[illegible],555 62,518	107,773
97-14	6,300	A B	56,270 43,329	93,299
98-15	0	A B	56,108 48,877	104,985
99-13	0	A B	53,891 48,713	102,604
00-17	0	A B	54,028 45,444	99,472

Note: MPS reserves are included.

- All retrieved satellites are returned to the orbiter payload bay.
- All maintenance missions were considered candidates for resupply.
- Extended mission kits were assumed to be available.
- Inclination angle changes were limited to 5 degrees.
- Multiple manifest missions were considered.

- OMV resupply for 28.5 degrees was assumed at the Space Station and at the orbiter for all other inclinations.

- Three space-based OMV's were assumed, one each at 28.5 and 97.5 degrees and one for 56 and 63.4 degrees in a hybrid ground- and space-based mode.

Figure 11 shows the user propellant requirements by orbital inclination and year. In one year (1996), the Space Station-based OMV will require as much as 18,500 pounds of propellant, supplied from the Space Station, for OMV operations. At the higher inclination of 97.5 degrees, extended-mission kit and OMV usage in one year requires as much as 66,300 pounds of bipropellant, to be supplied by the Shuttle orbiter from scavenging or a boosted tank farm.

Table 14 lists storable propellant users by year and orbital inclination. Also broken out are the satellite and extended-capability OMV tank kit resupply requirements.

4.2 PRELIMINARY SYSTEM TRADES

This section defines the necessary system trades that were made to obtain data so that a system concept selection could be made. Determining the best concept for delivering cryogenic propellants to orbit required an evaluation of when the propellant transfer should be performed during the prelaunch or ascent phases. This evaluation is necessary because the ET is jettisoned with usable residuals shortly after MECO. For storable propellants, the concept

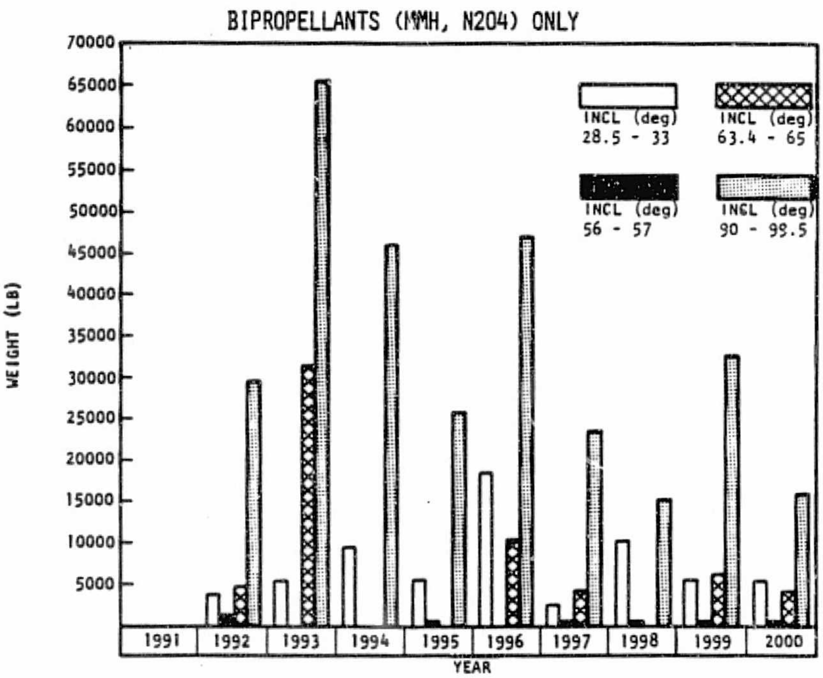


Figure 11. Storable Propellant Users' On-Orbit OMV and Satellite Propellant Resupply

Table 14. Storable Propellant Users' On-Orbit
OMV and Satellite Propellant Resupply

Altitude (nmi)	Inclination (deg)	Use	Propellant Resupply Requirements (MMH, NTO Only) by Year									
			1991	1992	1993	1994	1995	1996	1997	1998	1999	2000
250	28.5	OMV Satellite		3,800	5,300	9,500	5,400	18,500 (450)	2,600	10,200 (750)	5,400	5,200 (750)
160	56.0	OMV Satellite		1,200			400		400	400	400	400
160	63.4	OMV Satellite		4,800	31,500			10,300 (180)	4,100		6,200 (180)	4,000
160	97.5	OMV Kit Satellite		29,700 (540)	51,800 14,500 (6,540)	41,700 4,350 (4,540)	19,200 6,500 (8,040)	39,700 7,400 (2,040)	17,100 6,500 (8,000)	15,200	30,500 2,200	16,000

must be to transfer propellants from the OMS tanks on orbit and load tank sets of propellant in the payload bay on missions when space is available. The types and arrangement of payload bay tanks were also evaluated because payload bay space is limited and an optimization of the tank configuration is required. The heat load to the cryogenic propellant tanks in the payload bay was evaluated so insulation requirements could be defined and TVS concepts evaluated. The flow rate capability that various sizes of capillary acquisition systems (CAS's) can support was defined so that a system size can be selected once the transfer system sizes are defined. A zero-g CAS is required because transfer capability must exist when the orbiter is docked with the Space Station.

4.2.1 Optimization of Propellant Scavenging Time Frame

4.2.1.1 Cryogenic Propellant Scavenging. Cryogenic propellant scavenging could be performed at various times during the ascent phases of a given mission. For example, propellant for scavenging could be loaded in tanks located in the payload bay before lift-off, thereby increasing the payload weight until all the MPS impulse propellant loaded on the STS is required for the ascent phase of the mission. The ET is always loaded to fixed level sensors and cannot be off-loaded. Scavenging could also be performed during SSME-powered flight by monitoring vehicle performance and transferring propellants that are identified by this evaluation as not being required for the mission. Scavenging also could be performed after MECO by transferring as much as possible of the usable and unusable propellants that remain in the MPS, some of which are in the ET and some in the orbiter. Combinations of these schemes are also possible.

Scavenging concepts that have been evaluated in the past typically focused on scavenging after MECO because the propellant tankage located in the payload bay would be dry-launched and therefore would weigh less. However, the transfer system sizes required to transfer the propellant in a reasonable time, 20 minutes, for example, are very large and could require pumps. The results, which are presented in Reference 1, are shown in Figures 12 and 13. They indicate that for 3,000 pounds of LH_2 and 18,000 pounds of LO_2 the system line sizes are 3.2 and 6 inches, respectively, for pressurized transfer. The LO_2 size could be reduced to 2 inches if a pump were installed. Other important factors that must be considered are propellant interface control and ET trajectory impacts. For post-MECO LO_2 propellant transfer quantities above 8,000 pounds, a pitching maneuver cannot be used for propellant orientation because the propellant would be thrown forward. Therefore, linear acceleration must be applied. The impacts of pitching or linear acceleration are presented in Section 4.2.4.

To transfer propellants during SSME-powered flight requires real-time, accurate knowledge of the vehicle's performance. At present, no such evaluation exists, and the accuracy that could be attained is questionable. Transferring too much propellant during ascent could result in a propellant depletion cutoff and force the orbiter into an abort mission.

The option of transferring (loading) the propellants on the ground before lift-off has one important advantage that the other options don't have. That

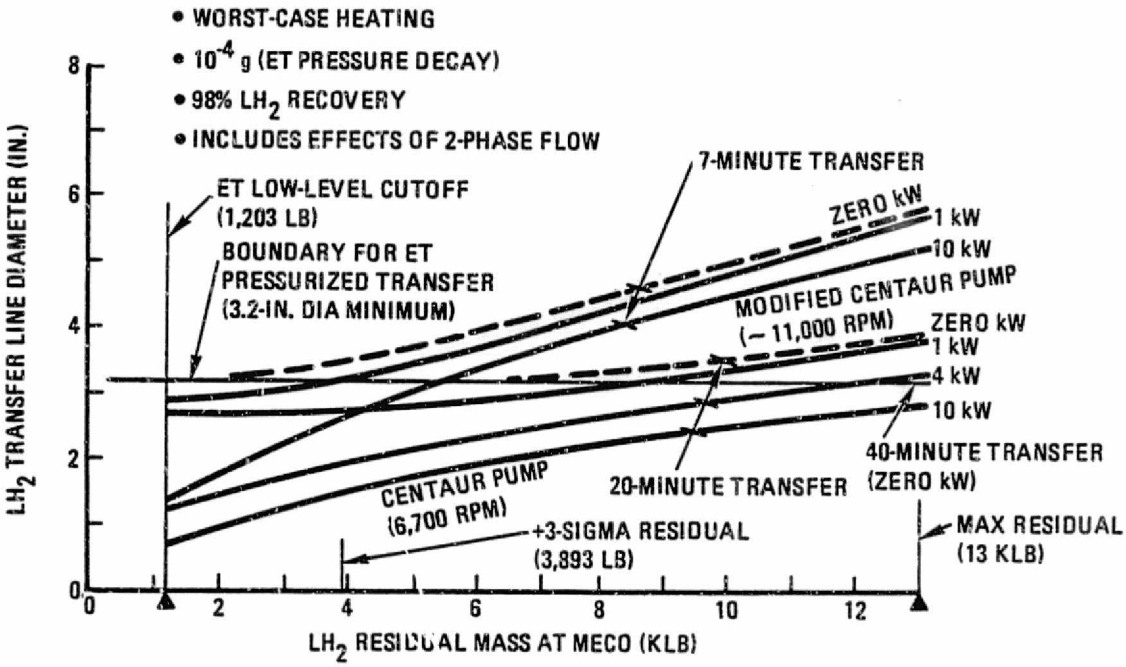


Figure 12. LH_2 Transfer Boost Pump Trade

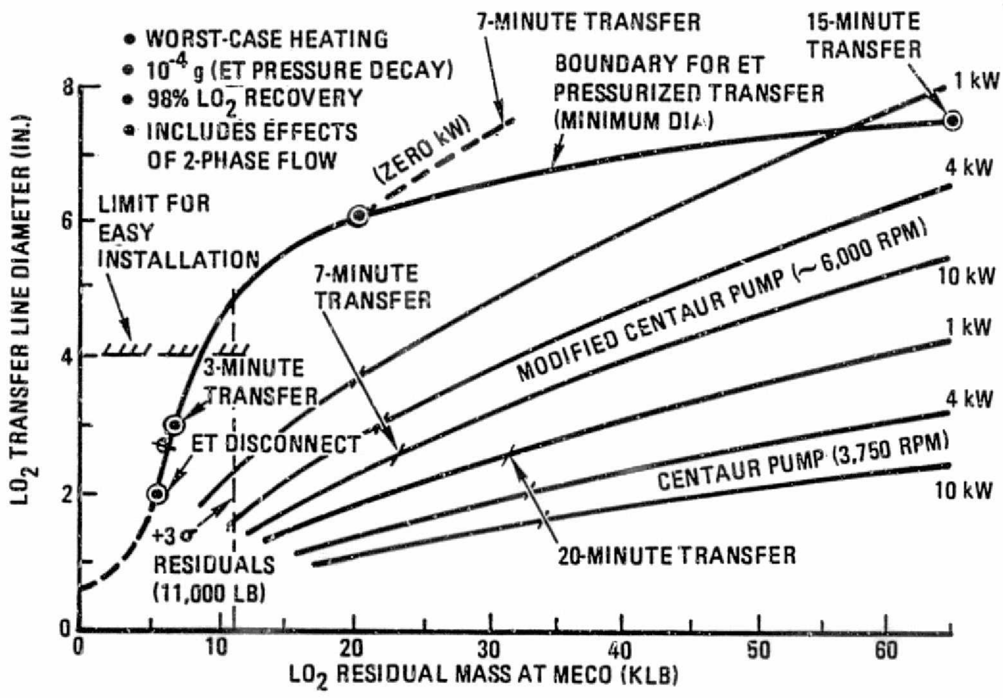


Figure 13. LO_2 Transfer Boost Pump Trade

is that the STS baselined for this study has a performance sensitivity factor of impulse propellant weight-to-initial orbiter weight of 0.91. This factor means that if 11,000 pounds is added to the payload bay (increased payload), then only 10,000 pounds of additional MPS impulse propellant is required to boost that weight to orbit. Therefore, on missions with excess MPS performance, tankage and propellant can be loaded in the payload bay to drive this excess to zero and thereby gain payload capability equal to the excess divided by 0.91. By doing this, each mission that has room in the payload bay for tankage will use up all the impulse propellant. The reserve propellant, however, will still exist at MECO and can be transferred through 2-inch lines without the use of pumps in less than 10 minutes by using an orbiter pitching maneuver at 2 deg/sec. This rate and duration would require only 155 pounds of RCS propellant, which is considered minimal. On roughly a third of the total number of scavenging missions, the payload bay tanks will be loaded full on the ground and the post-MECO transfer is not required. Data presented in Table 15 shows why, from a propellant to orbit viewpoint, the desired option is prelift-off loading and post-MECO transfer of reserves. Three cases for initial excess propellant are included.

4.2.1.2 Storable Propellant Scavenging. An optimization of the best time to perform storable propellant scavenging is not required because all of the propellant is contained within the orbiter. If space and payload weight capability exists on a given mission, a tankage set will be loaded with storable propellant before the launch and the OMS tanks will be filled.

4.2.2 Propellant Tank Length, Weight, and Propellant Capacity

4.2.2.1 Cryogenic Propellants. Three basic payload bay tank conceptual arrangements were developed for cryogenic propellants. The first concept is an arrangement of conventional cylindrical tanks with ellipsoidal ends. The second concept is a conventional spherical LH_2 tank with a toroidal LO_2 tank. The third concept is a conventional cylindrical LO_2 tank surrounded by a ring-shaped LH_2 tank. The concepts are shown in Figure 14, and the preliminary weights and lengths of these concepts are presented in Figures 15 and 16. The weight curves for each of the concepts do not account for components, fluid lines, helium system, and docking provisions. A weight of 1,750 pounds was included to account for these factors.

4.2.2.2 Storable Propellants. The system weight and length were estimated for a storable bipropellant payload bay tank system (PBTS), including the orbiter vehicle scar and required airborne support equipment (ASE) to provide scavenging from the OMS pod tankage and PBTS. The scar weight required for the orbiter to perform scavenging from the OMS pods was estimated to be 470 pounds, which includes the valves, lines, flanges, electrical equipment and the modifications required to the T-4 disconnect panel. The ASE required for scavenging includes the plumbing to the umbilical arm. the umbilical arm, and the electrical equipment and is estimated to weigh 930 pounds. These weights were determined in a Rockwell IR&D study (Project 84244). The total additional weight required to perform scavenging from the OMS pods only is 1,400 pounds. The weight required for scavenging from the OMS pods and a PBTS is 1,400 pounds plus the weight of the PBTS.

Table 15. Propellant Scavenging Optimization Data

	Scavenging Timeframe								
	All Post-MECO (Baseline)			Ascent + Post-MECO			Prelift-off + Post-MECO		
Excess propellant (lb)	7,000	14,000	24,500	7,000	14,000	24,500	7,000	14,000	24,500
Δpayload weight (lb)	0	0	0	0	0	0	7,690	15,380	26,920
Δtank weight (lb)	0	0	0	-40	-40	-40	-100	-100	-100
Δcomponent weight (lb)	0	0	0	+50	+50	+50	+40	+40	+40
Reserves (lb)	5,690	5,690	5,690	5,690	5,690	5,690	5,690	5,690	5,690
Propellant to orbit (lb)	12,690	19,690	30,190	12,700	19,700	30,200	13,320	21,010*	32,550

Note: Δ = (baseline-concept)

*Maximum propellant to orbit for 14,000-pound excess propellant cases

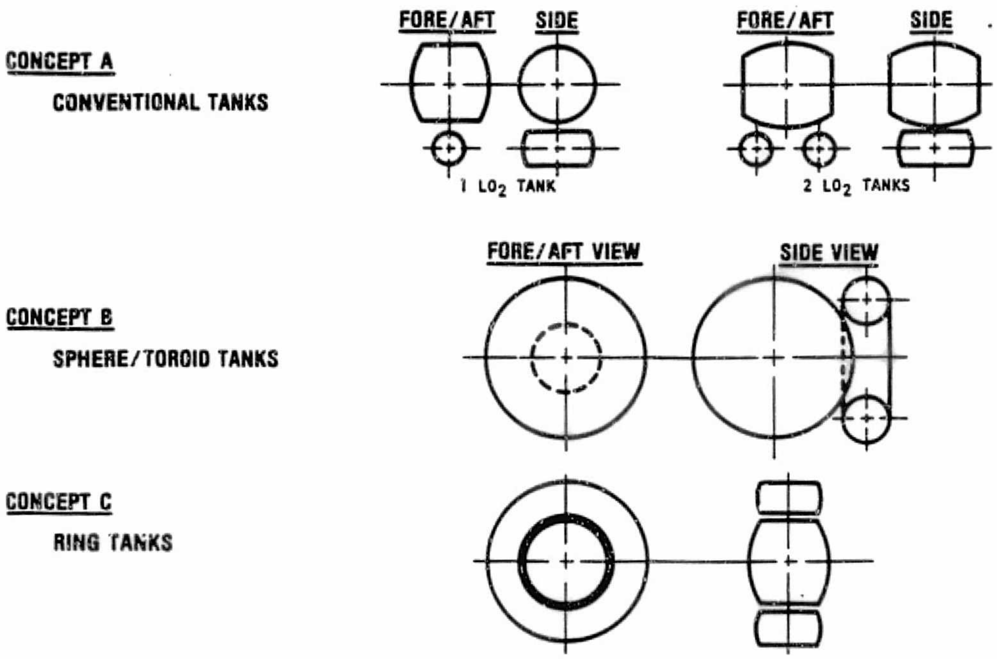


Figure 14. Cryogenic Propellant Payload Bay Tankage Concepts

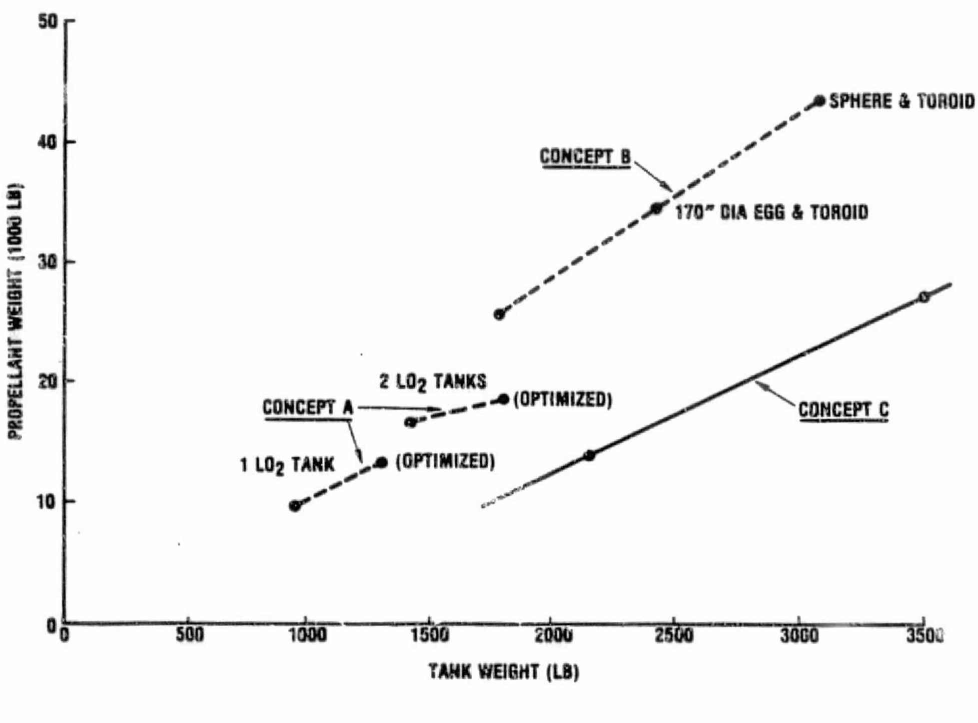


Figure 15. Cryogenic Propellant Trades Payload Bay Tankage Weight

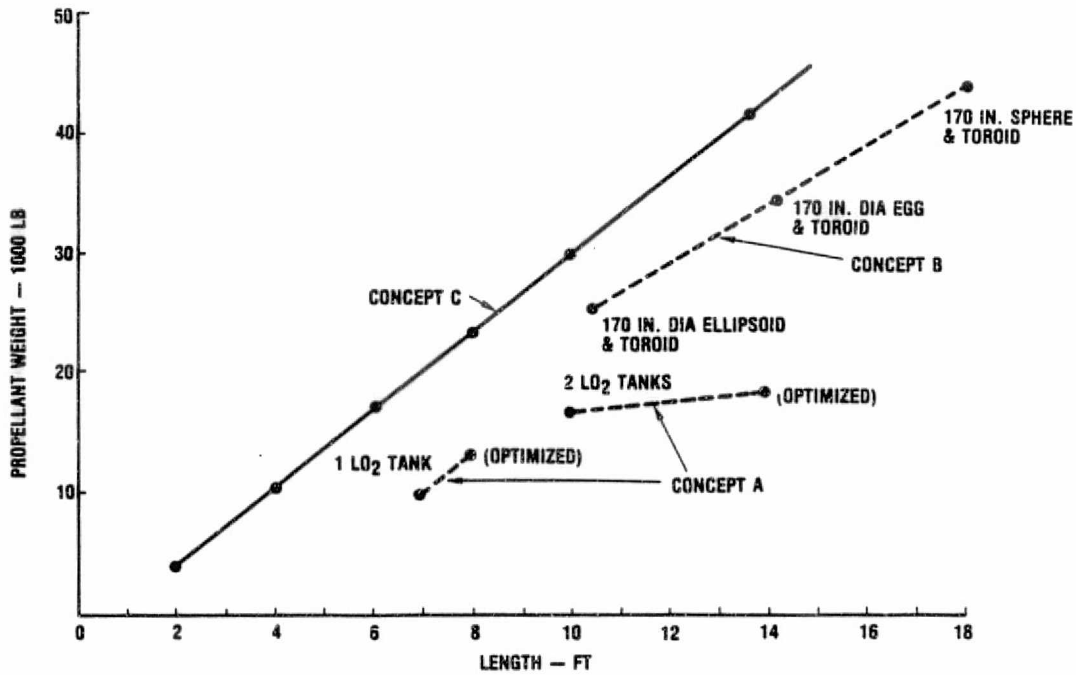


Figure 16. Cryogenic Propellant Payload Bay Length Requirements

The PBTS weights were estimated using the results of a Rockwell IR&D study (Project 83226) that determined system weights for various propellant weight requirements utilizing previously built flight-qualified tanks. Included in the PBTS weights are estimates for the forward and aft frame and interconnecting structure, sill and keel trunnions, valve panels, pressure panels, test coupling panels, payload bay fluid and electrical interface panels, thermal control (active and passive) systems, fluid and pressurant lines, and propellant and helium tanks, valves, and pressurant components. The weights for various PBTS designs are shown in Figure 17.

The length of the PBTS was taken as the propellant system tankage dimension plus 18 inches to allow for mounts, frame structure, and thermal control blankets. Estimates for the PBTS length are shown in Figure 18. The dashed lines connect the four PBTS concepts utilized in the storable propellant availability study.

4.2.3 Propellant to Orbit Vs. Tank Size and Configuration

As noted in the previous section a tank configuration that requires the least payload bay length for a given capacity may weigh more than competing configurations. Thus, it is not at once obvious which configuration will best utilize the excess performance and payload bay space available on a scavenging flight or set of flights. Within one configuration, moreover, a tank of large size will not fit on many potential scavenging flights and, so, loses its effectiveness, while a tank that is too small fails to accommodate much of the available propellant. These tradeoffs have been evaluated in this study to

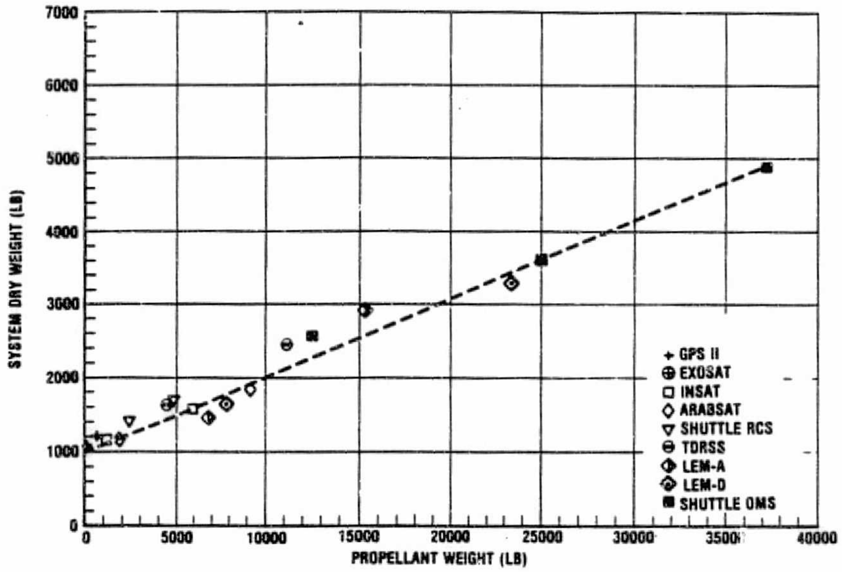


Figure 17. Payload Bay Storage Tank System Dry Weight Vs. Propellant Weight

maximize the total propellant scavenged over the ten-year flight set. In the case of cryogenic propellant scavenging, the three tank configurations were compared at discrete size increments up to 40,000 pounds capacity. In the case of storable propellant, the four tank configurations were compared as point designs. In both cases the use of more than one size to increase the amount of propellant delivered was considered.

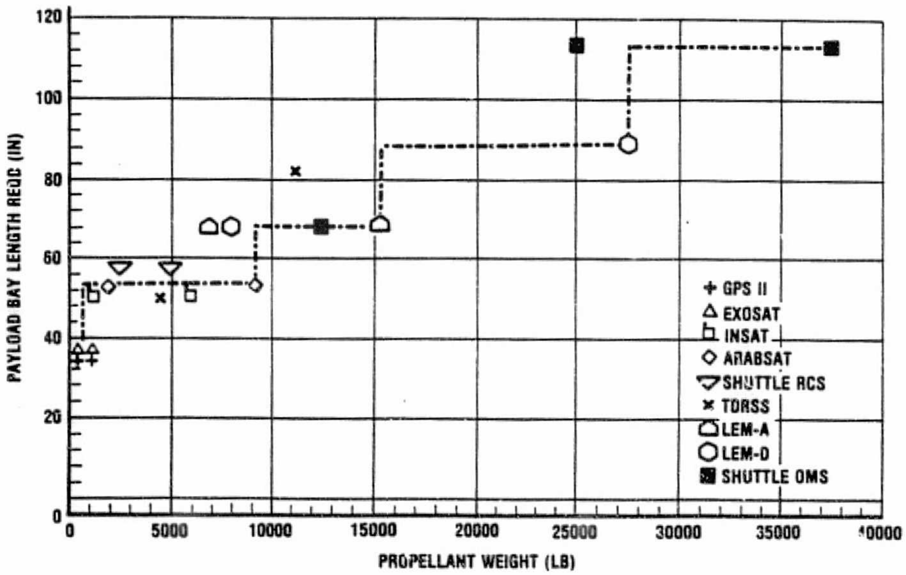


Figure 18. Payload Bay Tankage System Length Vs. Total Propellant Weight

Unlike the propellant availability results, all data in this section fully account for scavenging system weight and length. They do not, however, account for propellant trapped or lost to boiloff, effects which are evaluated in a later section. The cyrogenic tank configuration comparison and multiple size selection used the interim mission manifests, which include the 2,000 pounds of ancillary payloads and exclude reserve and residual scavenging. All other results shown are based on the final manifests and, in the case of cryogenic propellant, are presented with and without reserves and residuals.

The majority of both cryogenic and storable propellant scavenging flights go to 160 nautical miles and require an OMV sortie to complete the delivery of propellant to the Space Station. OMV propellant for these flights is identified at the end of this section.

4.2.3.1 Cryogenic Propellant Tank Selection. For each tank configuration the dependence of the total propellant scavenged on tank capacity exhibits similar characteristics, which are dictated by the constraints of ascent performance capability, payload bay length, and tank capacity. Figure 19 presents the situation for two flights with the ring tank design. The picture for total propellant is the aggregate of many such individual situations.

Following the solid lines in Figure 19 for Flight 1 in 1972, we see a large excess propellant limit (35,000 pounds) that must be reduced to account for scavenging system weight. The net scavengeable propellant, thus, decreases slowly with tank capacity along the downward sloping performance limit line. The payload manifest in this case leaves only 3.5 feet for tank

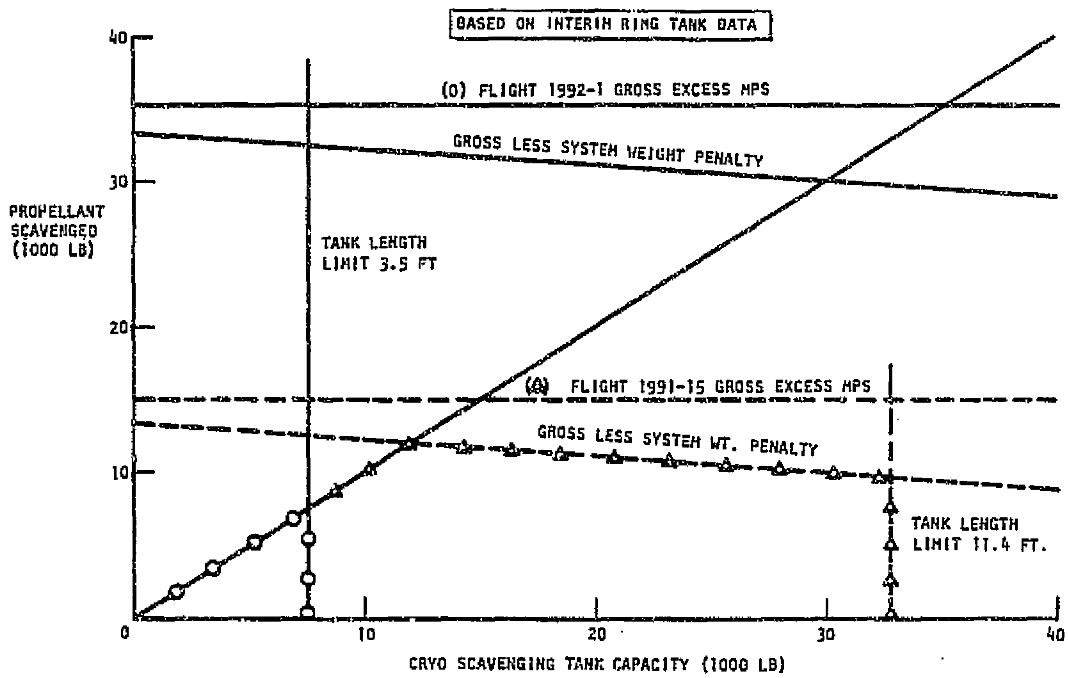


Figure 19. Typical Variation of Scavenged MPS With Tank Size

length, corresponding to the approximately 8,000-pound capacity shown by the vertical limit line. Finally, scavenged propellant cannot exceed tank capacity, as shown by the upward sloping capacity limit line. For this flight, scavenged propellant increases with tank capacity only up to the tank length limit and drops to zero for larger tanks.

The second flight, 1991 Flight 15, provides much less excess propellant but allows much more tank length. Scavenged propellant increases up to the performance limit, slowly decreases to the length limit, then drops to zero. In every case the optimum tank capacity (for that flight) occurs at the intersection of the capacity line with either the performance or length limit line.

In accord with the above discussion, the total propellant scavenged over the ten-year set of flights should increase strongly with tank size when the size is small enough to fit in most flights. As size increases, some flights will drop out, leaving a discontinuous drop in the curve of propellant versus size. Other flights will hit the performance limit, causing a gradual decrease with size. At some point a further increase in tank size will reduce scavenged propellant as more and more flights drop out or hit the performance limit. The curves in Figure 20 illustrate these very effects. However, since only discrete points have been calculated (at capacity increments of 2,100 pounds), the discontinuous drops are not seen here.

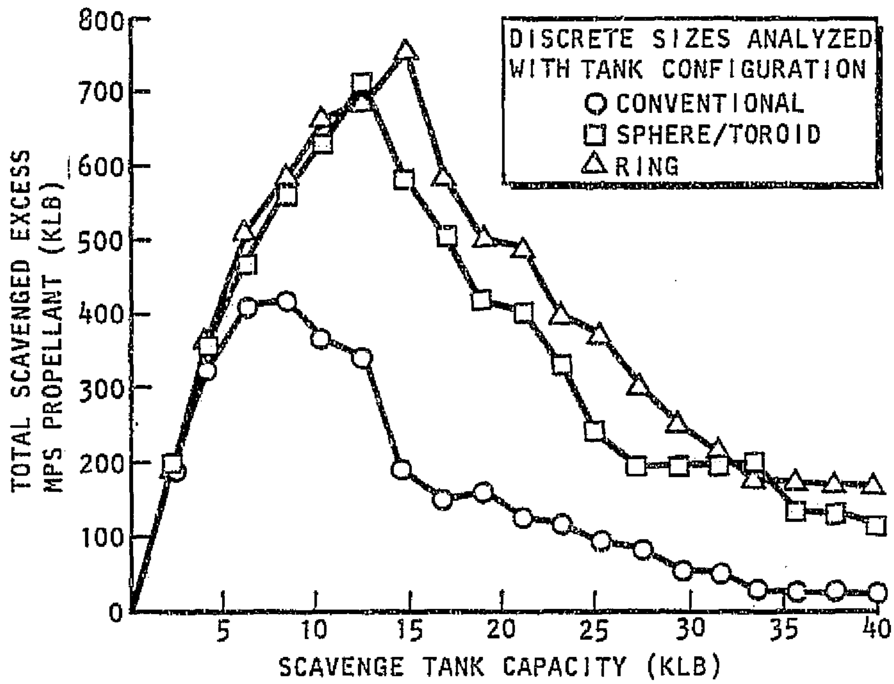


Figure 20. Scavenged Excess MPS Propellant Vs. Tank Configuration and Size (Interim Data)

The comparison of configurations in Figure 20 shows the general superiority of the ring tanks despite their heavier weight. The ring tank configuration was selected over the sphere/toroid for further analysis. The conventional configuration is a distant third choice due to its poor weight in small sizes and excessive length in larger sizes.

For the selected ring tank configuration, the best of the discrete sizes analyzed is 14,700 pounds. This result is reinforced by the curves in Figure 21, which show final ring tank data with and without scavenging of reserves and residuals. In both cases, 14,700 pounds is the best single size; but when reserves and residuals are scavenged, total propellant falls more slowly with increasing size, as would be expected. The peak values of total propellant scavenged with the 14,700-pound capacity tank are 834,000 pounds without reserves and residuals and 972,000 pounds with.

Although the 14,700 pounds capacity appears best over the ten-year regime, it is of further interest to see which tank sizes performed best for the variety of yearly flight manifests. Table 16 shows the yearly breakdown of scavenged propellant, with and without reserves and residuals, for capacities from 10,700 to 18,900 pounds. This range encompasses all the yearly optimum sizes, and 14,700 pounds is seen to be the best size in 14 out of 20 instances. This result indicates relative insensitivity to variations in manifesting.

Obviously the total propellant scavenged would increase if more than one tank size were available so the best size could be selected for each flight.

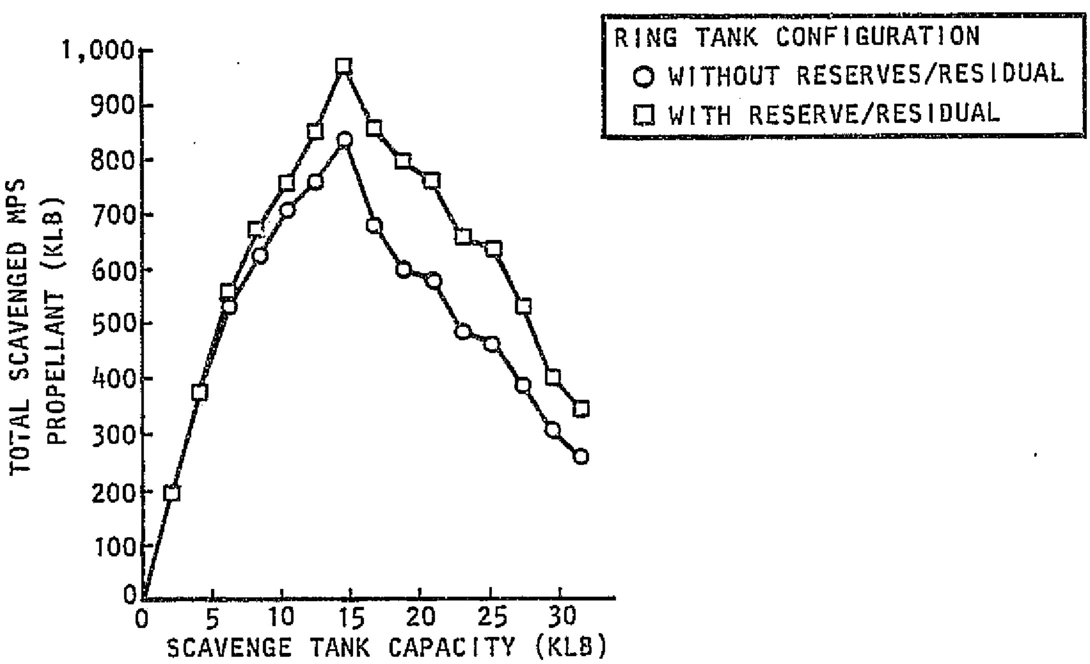


Figure 21. Scavenged MPS Propellant Vs. Ring Tank Size (Final Data)

Table 16. Scavenged MPS Propellant With Ring Tanks by Year (Thousands of Pounds)

Year	10,500-lb, 4.5-ft Tank		12,600-lb, 5.1-ft Tank		14,700-lb, 5.8-ft Tank		16,800-lb, 6.4-ft Tank		18,900-lb, 7.1-ft Tank	
	A*	B*	A	B	A	B	A	B	A	B
1991	113	116	123	139	128**	161**	97	142	96	145
1992	137	137	142	151	158**	176**	119	147	123	157
1993	60**	74	52	72	53	78**	38	65	37	65
1994	61	63	70	76	78	87	84	97	89**	105**
1995	69	74	69	74	79	85	90**	95**	62	68
1996	96	107	102	113	111**	130**	88	113	74	102
1997	66	74	76	86	86**	97**	63	74	31	42
1998	28	32	32	38	37**	42**	7	12	7	12
1999	28	32	32	37	35**	41**	20	28	19	30
2000	53	53	63	63	69**	74	75	84**	62	73
Total	710	758	761	848	834	972	680	858	601	799

*A = Without reserves and residuals; B = With reserves and residuals
**Best annual totals

The optimum pair of sizes was established by a computer analysis of all possible discrete pairs with the results shown in Figure 22. This is a contour map of the total scavenged propellant as a function of first (larger) and second (smaller) size. The 45-degree line represents equal sizes (really just one size), and point A is the 14,700-pound optimum. The narrow ridge to the right of A represents high performance from 14,700 pounds paired with larger sizes. The optimum pair, however, is 6,300 and 14,700 pounds, which lies at point B. This calculation was made with interim payload manifests and without scavenging reserves and residuals. Because of the good agreement of single-tank sizing with the interim and final ground rules (Figures 20 and 21), the two-tank result was assumed also to hold for the generation of final study data.

Further analysis of multiple cryogenic tank sizes assumed that 6,300- and 14,700-pound capacity tanks would be the best pair. Then other sizes were added one at a time to obtain the largest increments of scavenged propellant. This policy is not optimal but is believed to yield close to the maximum curve of propellant versus number of tanks. The difficulty in finding the optimal three tanks, for example, arises because the optimal two may not be included

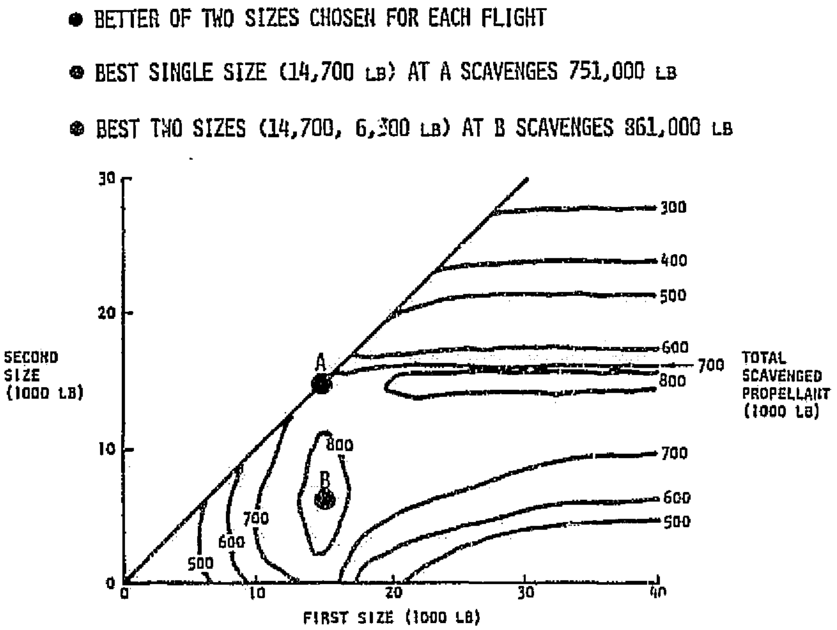


Figure 22. Total Cryo Propellant Scavenged With Two Tank Sizes

in the optimal set of three. However, some numerical experimentation shows that the total propellant with three or more sizes is not very sensitive to exactly which three or more are chosen. In any case, Figure 23 shows the trend obtained with up to five sizes and also the limiting case in which the

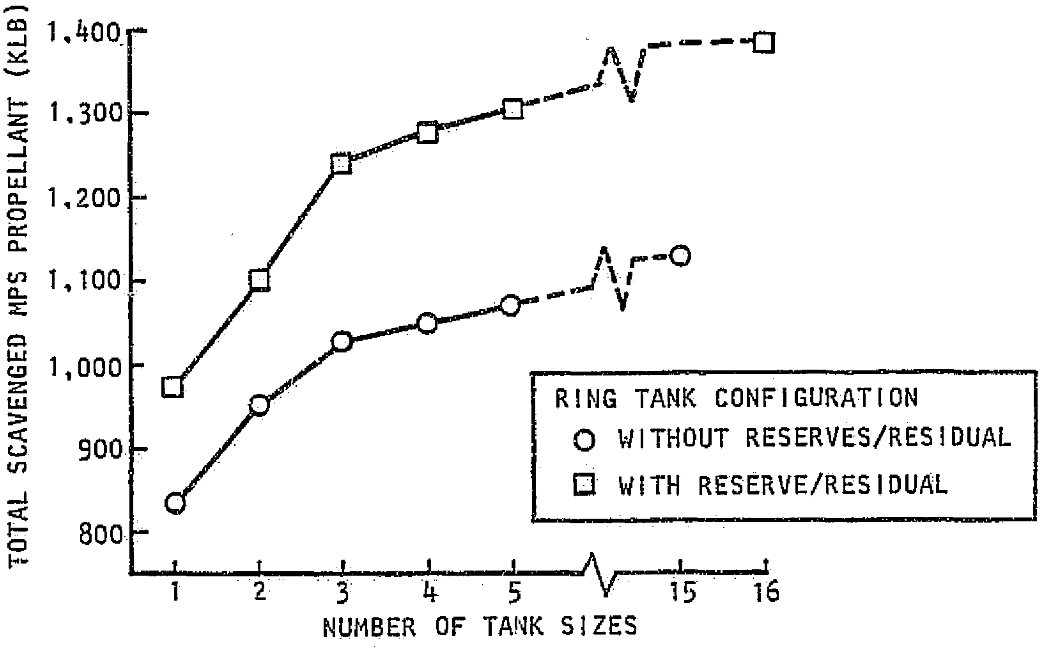


Figure 23. Scavenged MPS Propellant Vs. Number of Available Tank Sizes

optimal size is chosen for each flight. The latter approach requires 15 different sizes when reserves and residuals are excluded and 16 sizes when they are included. All these calculations used the discrete-size ring tank data only.

In the curve with reserves and residuals, the incremental propellant going from two to three sizes slightly exceeds the 1-to-2 increment. This indicates that 6,300 pounds is not the best second size; something larger than 14,700 pounds would have been better. This situation has no significant impact on study conclusions.

Table 17 presents a detailed breakdown of multiple tank size usage without scavenging reserves and residuals and Table 18 with reserves and residuals. With only one tank size there are 69 scavenging flights. This jumps to 89 with two to five sizes. At this point only four potential scavenging flights are omitted. Those that can fit only the 2,100-pound capacity tank. The absolute total number of scavenging flights is 93, compared to the 96 indicated as potential scavenging flights in the propellant availability discussion (Figure 10) because three of the 96 have insufficient excess performance to carry the weight of even the smallest (2,100-pound) tank.

4.2.3.2 Storable Propellant Tank Selection. The analysis of storable propellant tank sizing considered five cases: use of excess OMS tank capacity alone and with one of the four payload bay tank designs discussed in Section 4.2.2. In each case there is an allowance of 1,400 pounds for a propellant transfer system in addition to the payload bay tank weight, if any. The total amount of propellant scavenged over the ten-year flight set was the evaluation criterion as in the cryogenic propellant analysis.

In finding the amount of propellant that can be scavenged on each flight, one of four things will occur, three of which are illustrated in Figure 24. First, the ascent performance limit may preclude filling the OMS tank, so a payload bay tank would be useless. Second, the payload bay tank under consideration may be partially filled as determined by the intersection of the performance and OMS tank capacity limit lines. Third, the OMS and payload bay tanks may both be filled. Finally, the tank under consideration may not fit in the payload bay, so only the OMS capacity is available. The resulting calculation logic is more complicated than with cryogenic propellant tank sizing but leads again to an optimum size based on the best combination of capacity, length, and weight.

The results of this analysis are presented in Table 19 with the scavenged storable propellant detailed by year. Utilizing only the excess OMS tank capacity results in 784,780 pounds of propellant scavenged. The best of the four payload bay tanks, with 15,000-pound capacity, boosts the total by 81 percent to 1.438 million pounds. This tank design is the best in seven of the ten years. Altogether 165 flights permit storable scavenging, 98 with the OMS capacity only and 67 more with the 15,000-pound payload bay tank.

The incremental propellant gained by providing two storable propellant tank sizes (15,000 and 28,000 pounds) is 94,000 pounds, for a further 6.5 percent increase. This improvement does not justify the development of two tank sizes.

Table 17. Summary of Multiple Cryo Tank Sizing Without Reserve/Residual Scavenging

Tank Size (lb)	Number of Scavenging Flights					
	1 Tank	2 Tanks	3 Tanks	4 Tanks	5 Tanks	15 Tanks
2,100						4
4,200						2
6,300		26	26	26	22	18
8,400						6
10,500					20	16
12,600						9
14,700	69	63	49	49	33	18
16,800						6
18,900						2
21,000			14	8	8	5
23,100						1
25,200				6	6	2
27,300						1
29,400						1
≥31,500					2	2
Total flights	69	89	89	89	89	93
Total propellant	834	951	1,028	1,051	1,072	1,129

Table 18. Summary of Multiple Cryo Tank Sizing With Reserve/Residual Scavenging

Tank Size (lb)	Number of Scavenging Flights					
	1 Tank	2 Tanks	3 Tanks	4 Tanks	5 Tanks	15 Tanks
2,100						2
4,200						2
6,300		22	22	22	9	9
8,400					13	9
10,500						4
12,600						5
14,700	69	67	29	29	29	18
16,800						18
18,900						9
21,000			38	29	29	7
23,100						2
25,200				9	9	3
27,300						0
29,400						0
≥31,500						5
Total flights	69	89	89	89	89	93
Total propellant	972	1,098	1,240	1,279	1,306	1,383

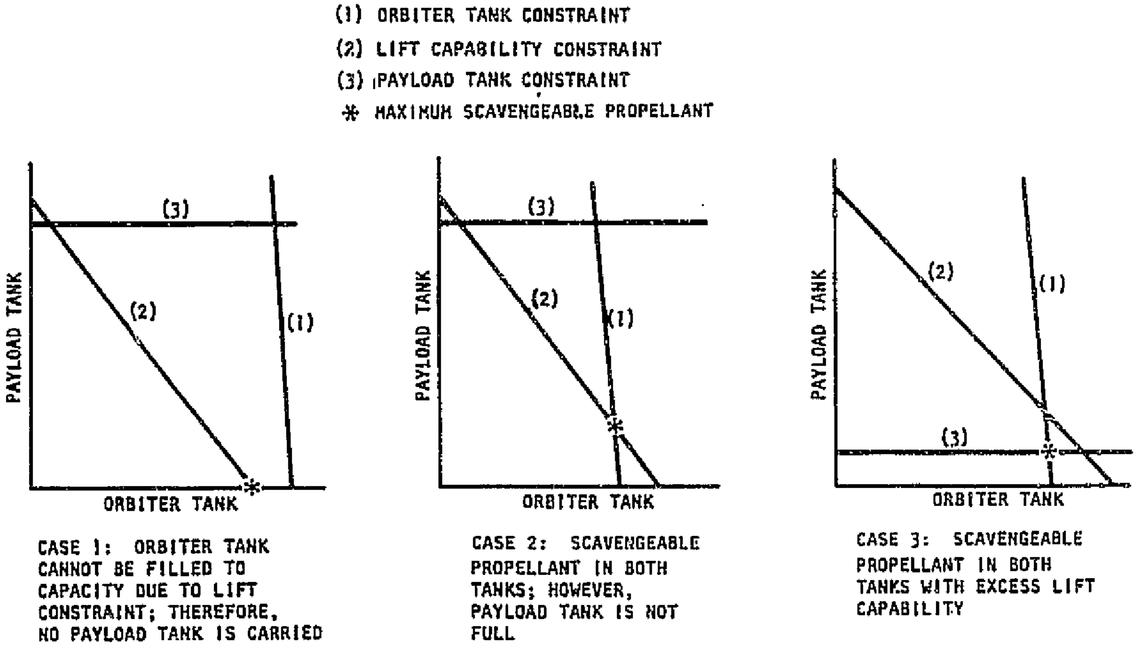


Figure 24. Limiting Cases for OMS Scavenging

Table 19. Scavenged Storable Propellant by Year (in Pounds)

Year	No Tanks	Tank 1 (8,935 lb/ 4.42 ft)	Tank 2 (15,156 lb/ 5.67 ft)	Tank 3 (27,656 lb/ 7.63 ft)	Tank 4 (37,141 lb/ 9.42 ft)
1991	66,391	146,407	151,421*	120,509	102,946
1992	93,961	199,128	220,790*	191,679	174,901
1993	55,473	92,379*	84,136	68,902	58,842
1994	102,238	148,046	162,784	164,650*	130,109
1995	82,760	134,317	154,231	158,401*	146,533
1996	101,279	176,460	196,049*	174,983	118,165
1997	80,871	128,691	155,204*	107,886	107,327
1998	63,531	82,822	93,526*	64,673	63,531
1999	52,286	70,601	77,438*	62,641	63,882
2000	85,990	126,717	142,261*	140,799	114,897
Total	784,780	1,305,567	1,437,839	1,255,124	1,081,132

*Best annual total

4.2.3.3 OMV Propellant Requirements. The OMV will be based at the Space Station. In orbiter flights to 160 nautical miles, the OMV will transport empty tanks from the Space Station down to a waiting orbit for rendezvous with the orbiter. Propellant transfer to these tanks will take place at the 160-nautical miles altitude. The OMV will then transport the propellant back to the Space Station where it will be transferred again or held in the tanks transported by the OMV.

By study ground rules the OMV propellant (in pounds) required on each sortie is given by 450 + 0.0564 x payload (pounds), where payload includes the scavenged propellant and the tanks to hold it. These tanks have been estimated conservatively to weigh 20 percent of their propellant capacity. Table 20 presents the amount of OMV propellant required to deliver both cryogenic and storable propellants over the ten-year mission model. The number of sorties and amount of propellant transported lead to OMV requirements of 66,000 to 92,000 pounds of propellant for the cryogenic scavenging flights to 160 nautical miles, depending on the number of tank sizes used. The figures shown apply to scavenging of reserves and residuals. For the 88 flights to 160 nautical miles with scavenged storable propellant the OMV requirement is 105,000 pounds of propellant.

Table 20. OMV Propellant Requirements for Delivering Scavenged Propellants

No. of Tank Sizes Cryogenic[3]	Number of OMV Round Trips[1]	Propellant Delivered (klb)	OMV Propellant (klb)[2]
1	47	666	66
2	59	735	76
3	59	843	84
4	59	870	85
5	59	888	87
All	61	958	92
Storable scavenging with 1 tank size	88	961	105

[1]From Space Station to orbiter at 160 nautical miles and back to Space Station.
[2]Includes 20-percent factor for tanks to carry scavenged propellant.
[3]Assumes scavenging of reserves/residuals.

4.2.4 Propellant Orientation Control

In order to allow post-MECO scavenging from the MPS (ET and orbiter lines), the remaining liquids must be oriented favorably. The LO_2 must be impelled toward the aft end of the feed line and the LH_2 toward the tank bottom. As a system sizing goal, ten minutes has been selected as a reasonable time to allow for the post-MECO propellant transfer, and the favorable liquid orientation is assumed to be maintained for this time. The methods of accomplishing propellant orientation investigated are centrifugal acceleration, by pitching the mated orbiter/ET, and linear acceleration, by continuous thrust through the mated vehicle c.g. The requirements for each method differ in hardware development, software development, and operational complexity.

Previous studies of propellant settling indicate that 10^{-4} g's acceleration in the desired direction is adequate in the absence of other disturbing accelerations. For the case of centrifugal acceleration, a pitch rate of 2 deg/sec was selected as adequate and well within the current Shuttle software limit of 5 deg/sec. This rate produces a 10^{-4} g's negative x-component in the LO_2 feed line at a position about 3 feet aft of the vehicle c.g., a point corresponding to 8,000 pounds of LO_2 remaining in the system. Of course any LO_2 forward of the c.g. would be driven in the wrong direction by the pitch maneuver, but that propellant would represent a large excess performance margin that normally would be eliminated by prelift-off tanking on a scavenging flight. At the bottom of the LH_2 tank, the 2 deg/sec rate produces about 1.5×10^{-3} g's of x-acceleration. The centrifugal acceleration field is illustrated in the top of Figure 25.

A large pitch maneuver with the ET attached is not part of the normal Shuttle post-MECO mission profile, but such a maneuver can be accomplished readily with the available control modes. To establish RCS requirements a +Y rotation was simulated under the assumptions of transition digital auto pilot (TRANSDAP) flight control, 10-degree and 0.5-deg/sec error limits, and no slosh model. Acceleration to 2 deg/sec required approximately 5-second firings of the primary fore and aft Z-thrusters. Attitude control in X and Z axes was negligible during the ten-minute Y axis drift. Five-second firings again arrested the rotation. Propellant consumption was 62 pounds by the forward jets and 93 pounds by the aft jets, for a total of 155 pounds of RCS propellant. This value is proportional to the chosen 2-deg/sec pitch rate and could probably be reduced.

Because of asymmetry in the fore and aft jet effectiveness, there is a net Z force acting during the rotational accelerations. It is small, however, and will produce no more than 0.2 to 0.3 ft/sec translational velocity.

For settling propellants by linear acceleration, the present orbiter RCS is not suitable. Only primary +X jets are provided. Their thrust (870 pounds) is much more than required, and even one jet firing for 10 minutes would consume 1,900 pounds of propellant. What is required for 10^{-4} g's is 35 to 40 pounds of thrust, a range that matches well with the use of two orbiter vernier jets at 24 pounds of thrust each. These would be located in the left and right aft pods and should be canted through the nominal c.g. to avoid the firing of other attitude control jets, insofar as possible, during

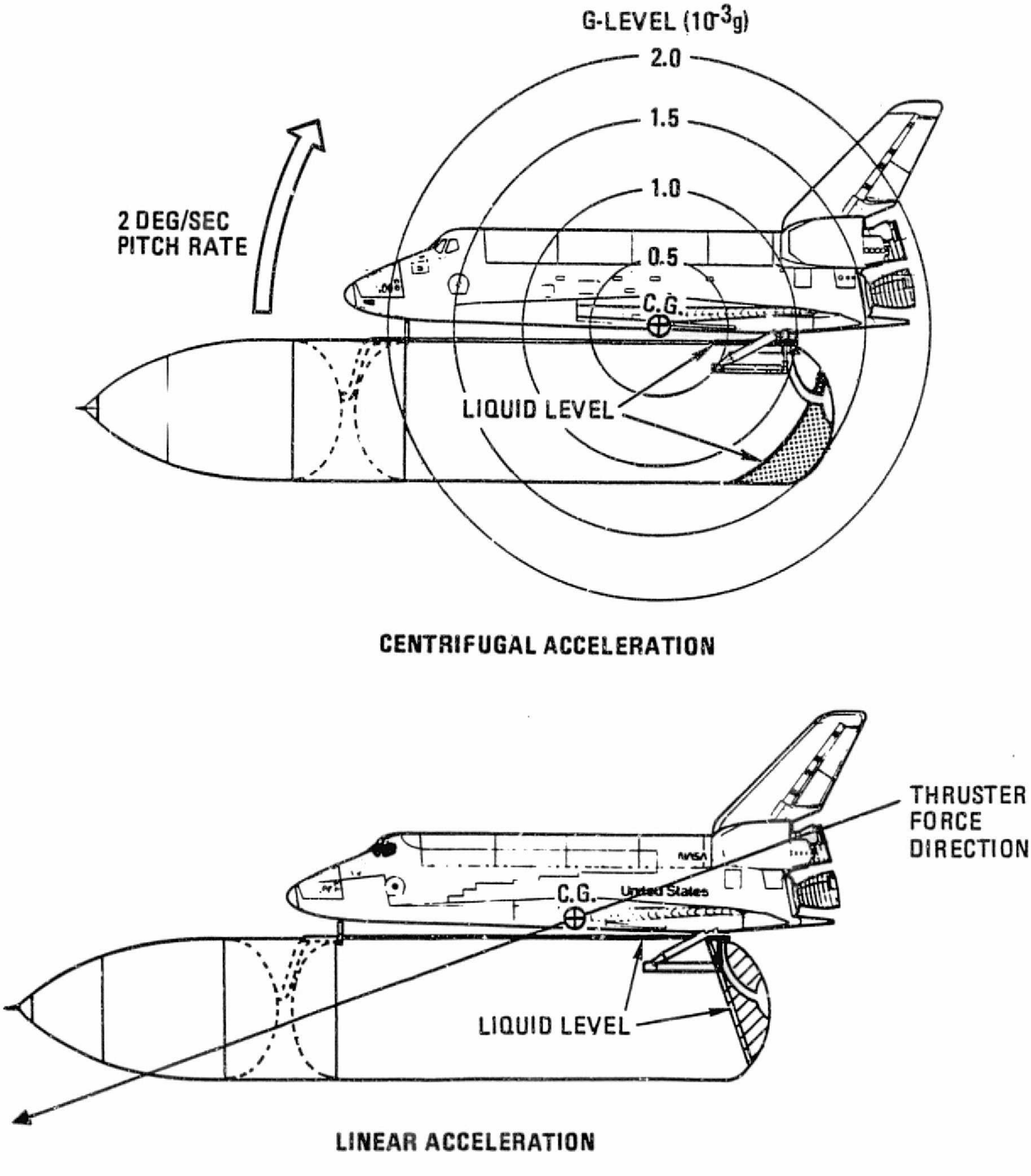


Figure 25. Post-MECO Propellant Orientation Methods

propellant settling. The resulting acceleration direction is illustrated in the lower portion of Figure 25. A delta-V of 2.5 ft/sec would be added in a ten-minute firing.

Using two added +X vernier thrusters during the transfer process would require approximately 110 pounds of aft ARCS propellant. The certified steady-state burn time limit of the present vernier thrusters is 125 seconds;

however, flights and ground qualification testing have shown that steady-state burn times of over 200 seconds in flight and 1,500 seconds in ground qualification tests are acceptable. The thruster and structure reach thermal equilibrium in approximately 100 seconds of firing.

The current life limitation of the vernier thruster is due to thermal cycling, short pulsing, and burn time. White Sands test data indicates a firing life in excess of 70,000 seconds. A 600-second burn time per scavenging mission translates into well over 100 scavenging missions per +X vernier set.

It should be noted that the LH_2 will be oriented quite differently by the two settling methods (Figure 25). The thrust direction from aft pods to c.g. is preferable to the centrifugal acceleration in locating the LH_2 near the tank outlet. In either method the effects of post-MECO attitude disturbances and LH_2 sloshing must be investigated if post-MECO transfer is to be considered further because these effects may be quite significant to the settling requirements.

Another consideration with either settling method is the need to modify or work around current Shuttle software sequences in order to insert the settling maneuver (and of course the propellant transfer itself) between MECO and ET separation. The pitch manuever, being somewhat more complex than the linear acceleration, will be used to illustrate what is required.

The upper diagram in Figure 26 indicates the primary events that initiate the ET separation sequence and the TRANSDAP flight control mode. These sequences normally execute automatically up to the -Z translation maneuver, which is performed manually. However, several options exist for manual override of the automatic sequences. One possible procedure described in the lower diagram requires the crew to make four manual selections (A-D). Selecting manual enable (C) for ET SEP prior to MECO inhibits the separation sequence until auto separation is again selected (D). The use of Y axis rotational pulse mode (A) allows the pilot to achieve the 2-deg/sec pitch rate with the rotation hand controller (RHC) while the auto maneuver mode continues attitude hold in the X and Z axes. Selecting the Y axis auto maneuver (B) causes the Y rate to be driven to zero after propellant transfer is complete. A procedure such as this might be acceptable on an experimental basis, but for routine post-MECO scavenging operations a new automatic software sequence would certainly be needed.

In summary, the evaluation of two propellant orientation methods favors linear acceleration for better LH_2 location and simpler operations. However, it requires hardware development to provide suitable thrust capability. The centrifugal acceleration method is theoretically feasible with current Shuttle hardware and software, but in reality both methods require development of new software sequences. RCS propellant consumption is acceptably small with either method. The cost of implementing either method will help determine the attractiveness of the third option, which is to forgo post-MECO scavenging.

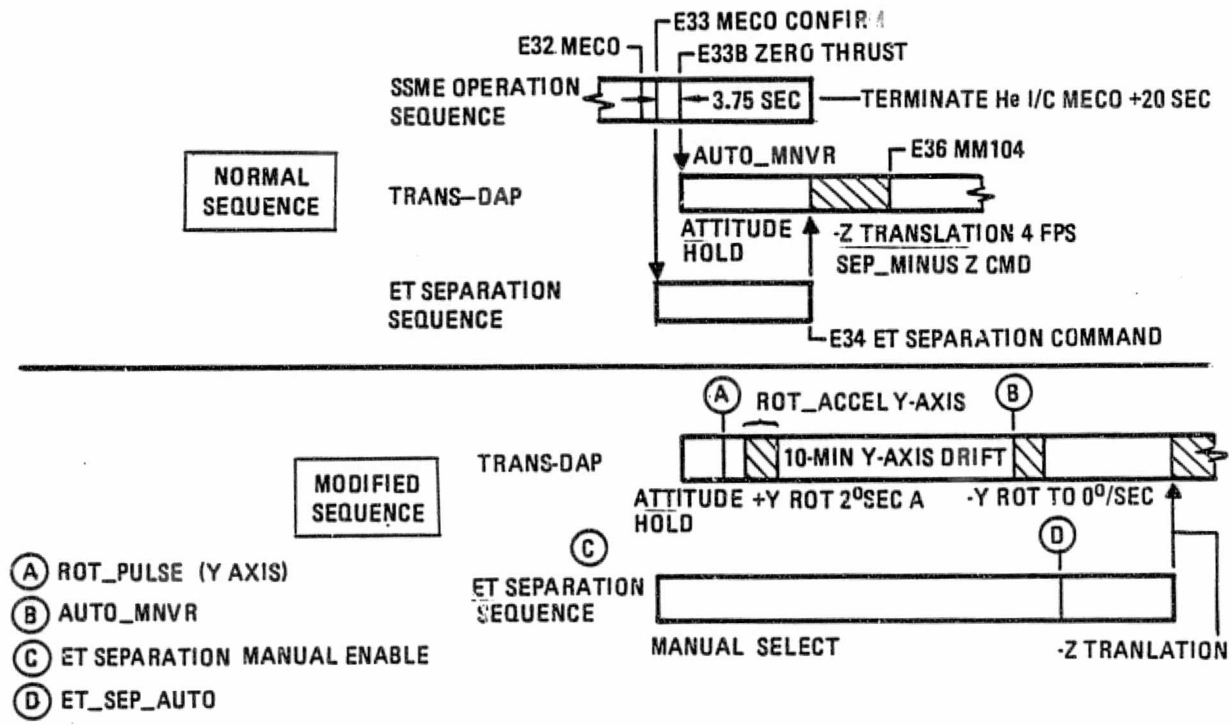


Figure 26. Orbiter/ET Post-MECO Sequences

4.2.5 Propellant Tank Insulation Thickness and Heating Rate

The thermal analysis effort took the form of a determination of heating rate to the LH_2 and LO_2 receiver tanks for three orbital orientations, two tank configurations, and a variety of tank insulation thicknesses. The first tank configuration thermally evaluated consisted of cylindrical LH_2 and LO_2 tanks with the hydrogen tank placed above. Figure 27 shows an isometric view of the tanks and the cylinder which represents a full payload bay. The second, and favored, configuration was a ring LH_2 tank with the LO_2 receiver tank located in the LH_2 tank's annulus. The thermal representation of the ring tanks is shown in Figure 28. In both tank configurations, the tanks are assumed to be placed within the aft 10 feet of the payload bay.

Modeling of the orbiter was provided by the 390-node model of the orbiter mid fuselage. A cylinder was added to the payload bay to represent a full payload bay.

Form factors and incident heating for the various orbital orientations used in the evaluation were obtained using the Thermal Radiation Analysis System (TRASYS) program. In all analyses, it was assumed that the payload bay doors were open.

In order to bound the heating, it was decided to evaluate a hot case (full sun in payload bay), a cold case (no sun, but planetary heating in payload bay), and a case between the two.

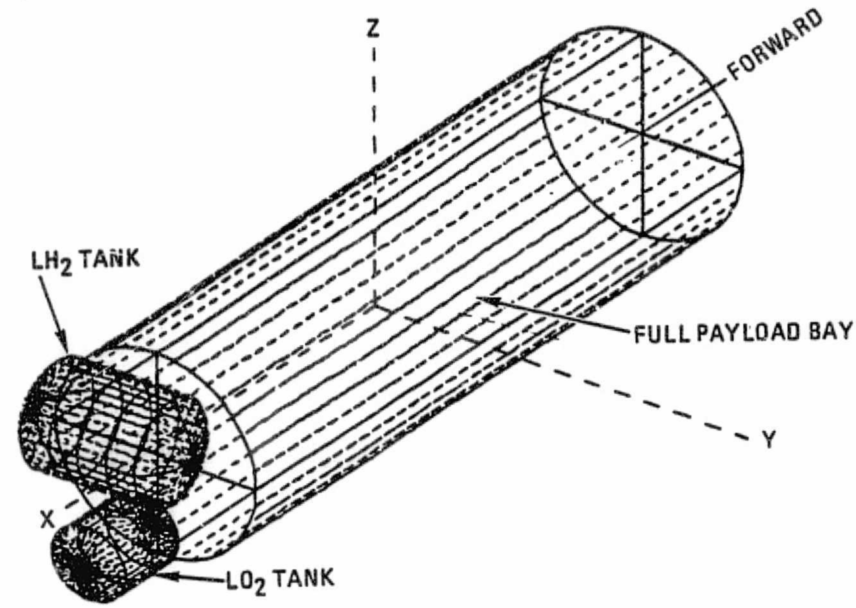


Figure 27. Conventional Receiver Tanks

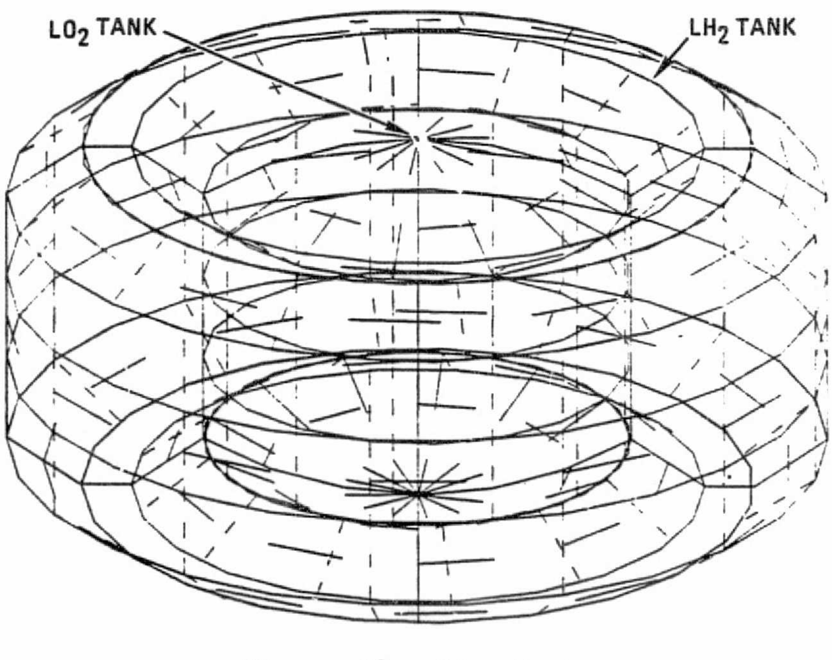


Figure 28. Ring Tanks

Figure 29 shows the hot orientation at an angle between the orbit plane and the sun equal to 90 degrees. This represents a hot condition in which the sun is constantly shining in the payload bay. The cold case is depicted in Figure 30. In this orientation (XSI), the sun is always parallel to the

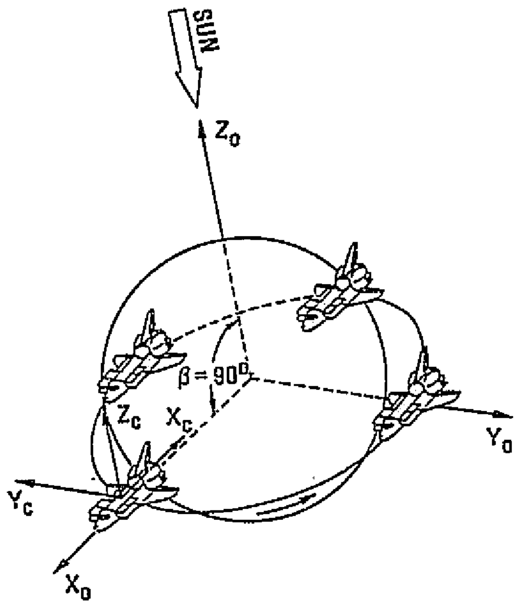


Figure 29. Orbiter in Top Solar Inertial (+ZSI), 150-nmi, and Beta = 90 deg Orientation

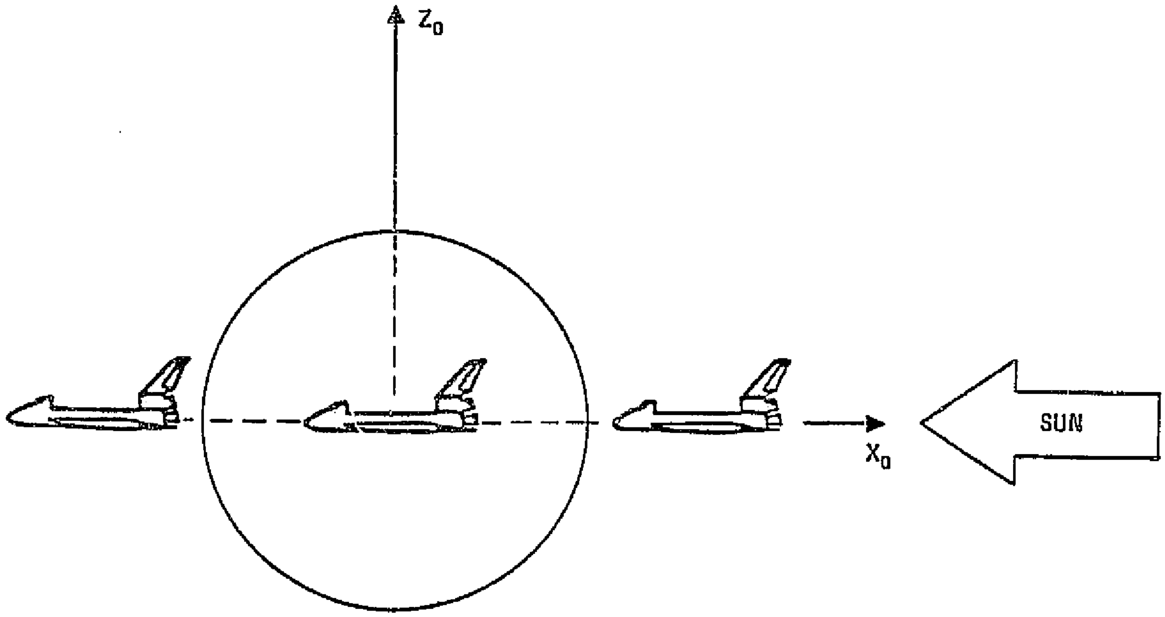


Figure 30. Orbiter in Tail Solar Inertial (+XSI), 150-nmi, and Beta = 0 deg Orienation

orbiter X axis (tail toward sun), resulting in a minimum heat input into the payload bay and a day/night cycle because of the orientation. Figure 31 shows the constant orientation of the payload bay toward the earth (ZLV). This ZLV orientation provides a thermal severity between the hot and cold cases.

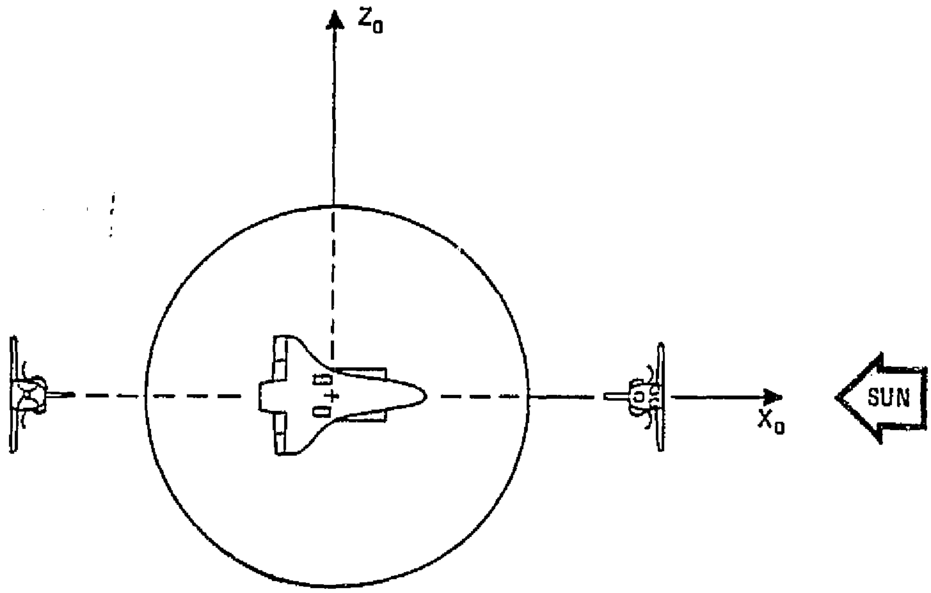


Figure 31. Orbiter in Top Local Vertical (+ZLV), 150-nmi, and Beta = 0 deg Orientation

The Systems Improved Numerical Differencing Analyzer (SINDA) program was used to determine the effect of the heating and form factors on the tank configurations. It was assumed that the receiver tanks were covered with a sprayable external insulation. In order to get a sensitivity of insulation thickness on heating rate to liquid propellant, the external insulation thickness was varied.

Figure 32 shows the integrated heating rate for the cylindrical LH_2 and LO_2 tank as a function of tank insulation thickness. As seen, the total heating rate for the LH_2 tank varies from 12,200 Btu/hr for 0.25 inch of insulation to 5,300 Btu/hr for 2.0 inches of insulation. The LO_2 tank's total heating rate is 11,550 and 4,400 Btu/hr for 0.25 and 2.0 inches of insulation, respectively. In this case and all subsequent results, the heat rate noted is the total heating rate to the propellant.

The effect of the ring tank configuration for the above (ZSI) orientation is shown in Figure 33. For the ring configuration, the LH_2 tank's total heat input is 28,000 Btu/hr (more than twice that of the cylindrical LH_2 tank for the same insulation thickness), and the LO_2 heat is reduced by a factor of more than 3 to approximately 3,200 Btu/hr.

This relatively large heat input has prompted the addition of a shade to close out the region between the tanks and the payload bay liner (and between the tanks and the bulkhead/simulated payload), thereby eliminating the solar trapping phenomenon (increased heating due to the inability of surfaces to radiate to space). This occurs when large-diameter payloads are close to one another, near bulkheads, or a wall. Adding the shade reduces the total heating rate to the LH_2 from 28,000 Btu/hr to approximately 18,500 Btu/hr (see

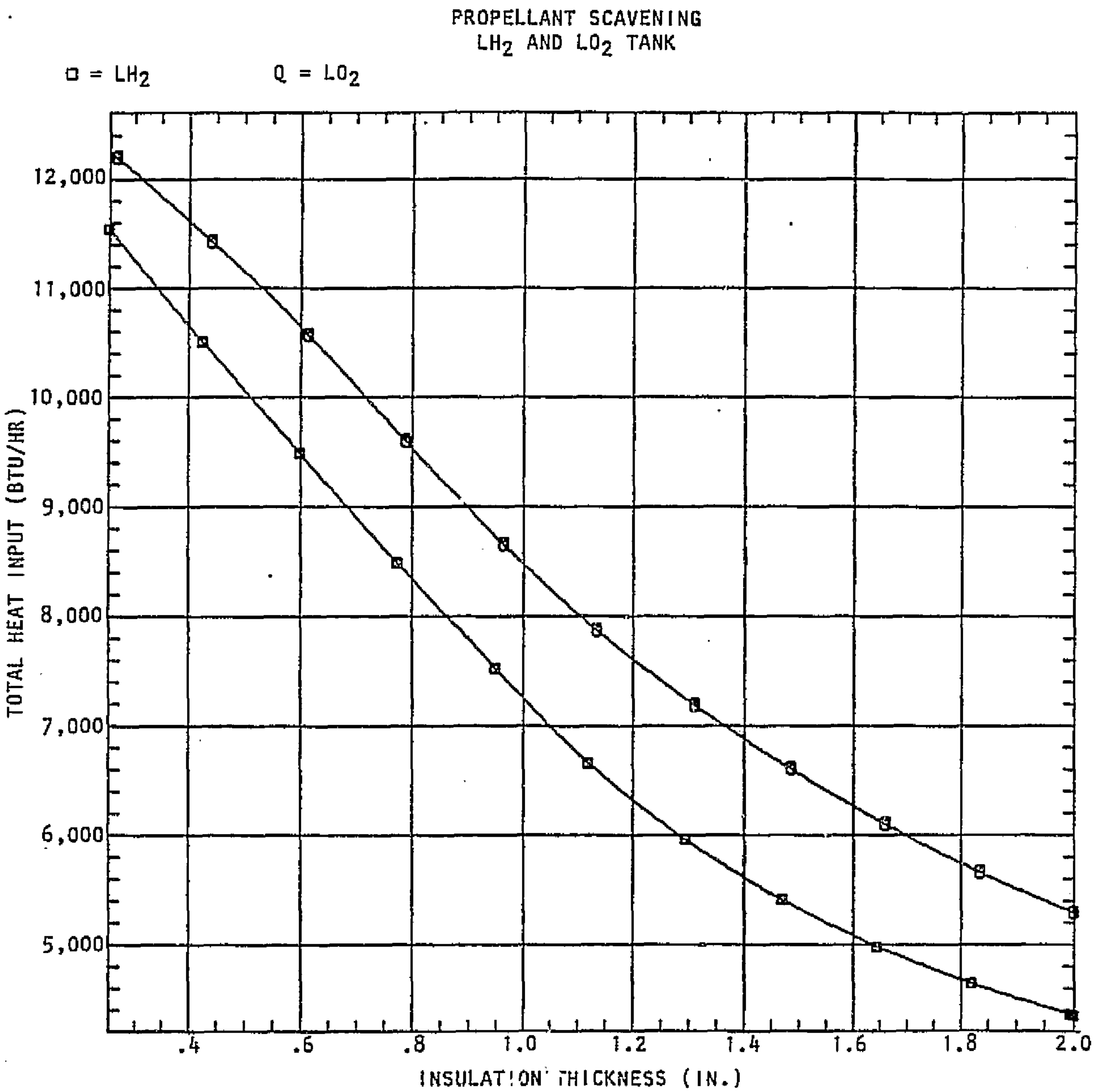


Figure 32. Conventional Tank Heating ZSI Orientation

Figure 33). A much smaller total heating rate to the LO_2 propellant is observed because the LH_2 tank almost completely encloses the LO_2 tank. It should be noted that in all the ring tank configurations there was assumed to be 4 inches of insulation between the LH_2 and LO_2 ring tankage.

Because the ZSI orientation represents such an extreme condition (hot) and this orientation would present a high-temperature condition to other payloads in the bay, it probably would not be flown. Therefore, investigations of other orientations were deemed prudent.

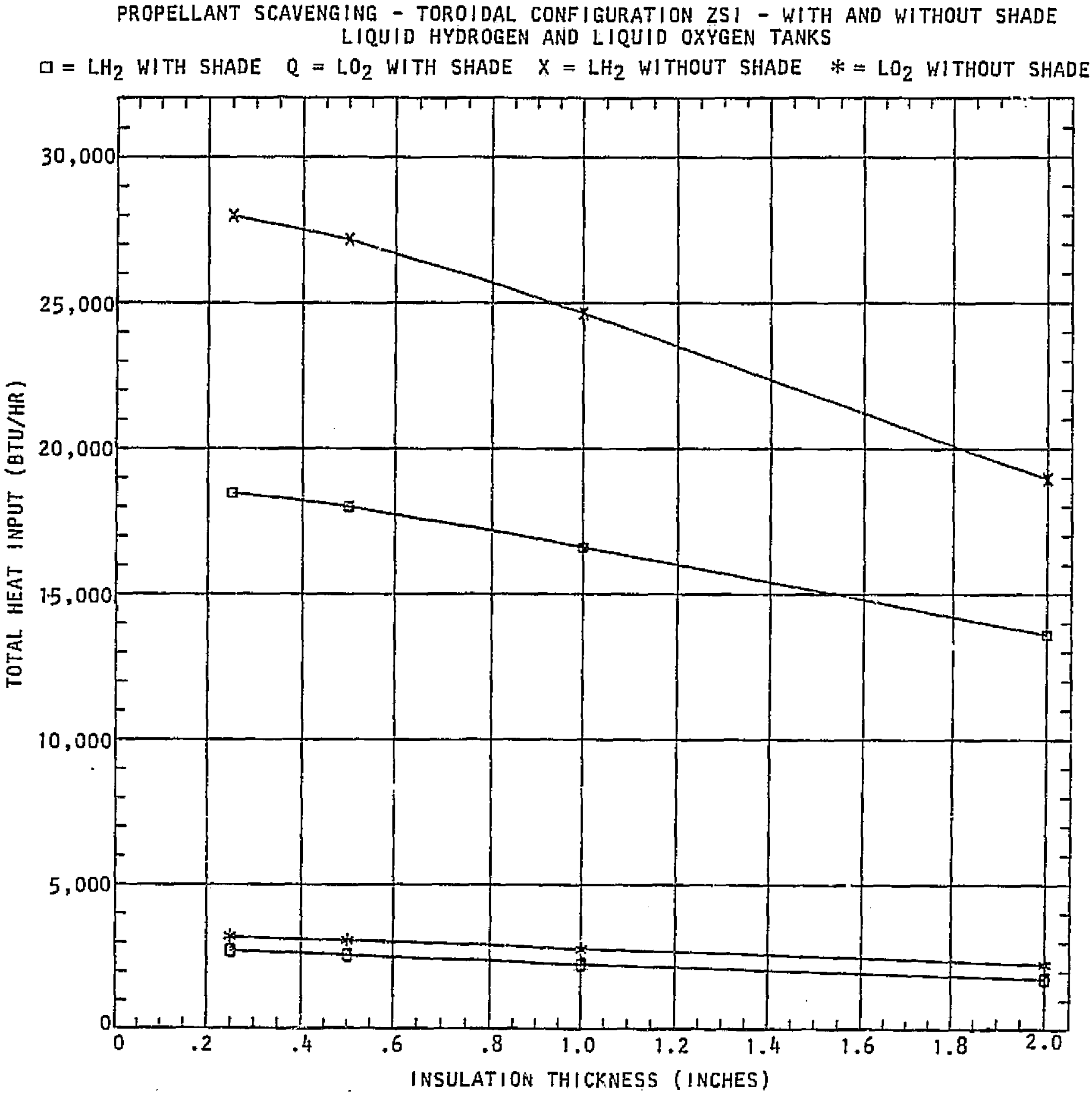


Figure 33. Ring Tank Heating ZSI Orientation

The orientation of ZLV (payload bay toward earth) was determined to be meaningful since it represents somewhat of an analysis standard for orbiter work. The result of using the environment produced by a ZLV orientation is shown in Figure 34. As expected, this orientation results in tank total heating rate inputs that are significantly less than for ZSI. The total heat input for 0.25 inch of tank insulation is on the order of 9,700 Btu/hr, but adding the shade reduces that value to 8,750 Btu/hr. The LO_2 tank total heating rate is near 900 Btu/hr.

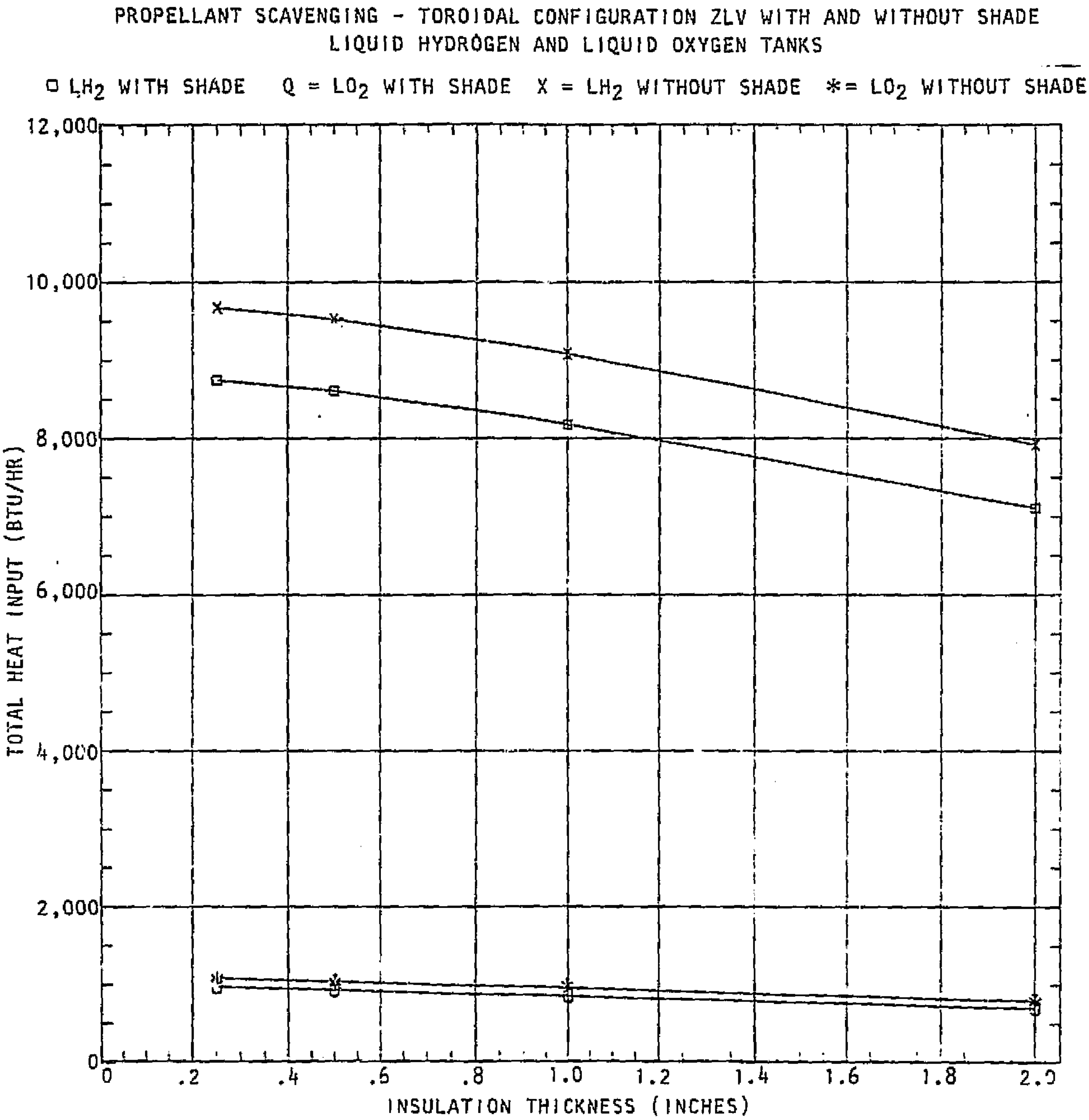


Figure 34. Ring Tank Heating ZLV Orientation

In order to evaluate a minimum heat input to the tank, an additional orientation was considered. The result of the +XSI (tail towards the sun) environmental case is shown in Figure 35. From the curves, the minimum heating rates are seen to be 5,150 and 500 Btu/hr for LH_2 and LO_2 tanks, respectively.

In summary, the maximum and minimum heating rates for the ring LH_2 tank are 28,000 and 5,150 Btu/hr, respectively, with a realistic operational value

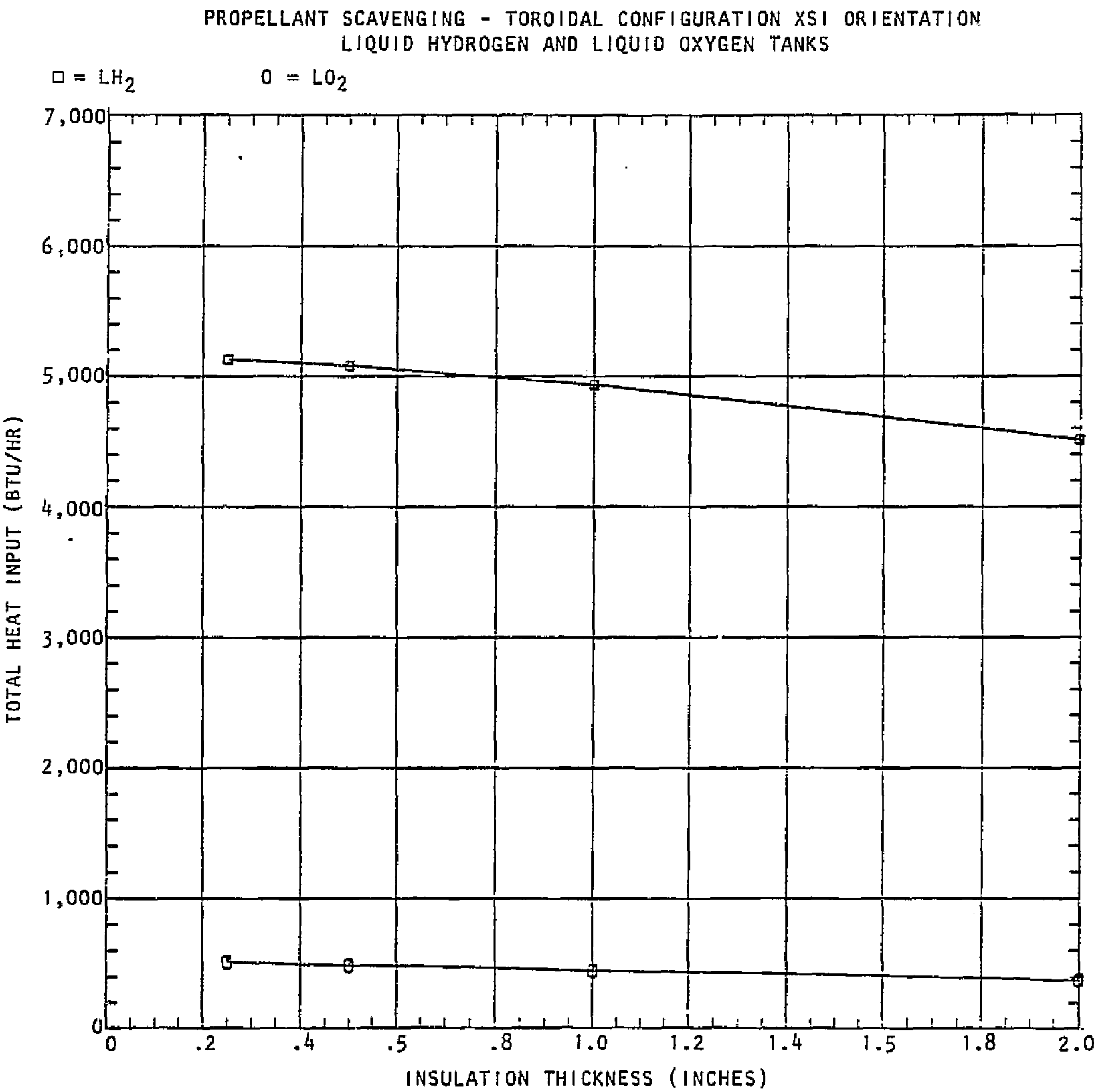


Figure 35. Ring Tank Heating XSI Orientation

of 9,700 Btu/hr. Adding a shade reduces this value to 8,750 Btu/hr. Maximum LO_2 heating rate is on the order of 3,200 Btu/hr with a minimum value of 500 Btu/hr. Table [illegible] lists pertinent results from the analyses for the ring tank geometry.

4.2.6 Thermodynamic Vent Systems (TVS)

In a low-g environment, long-term subcritical storage of cryogens is complicated by the lack of positive control of the location of the liquid and

Table 21. Ring Tank Heating Rate

Orientation	Heating Rate Without Shade (Btu/hr)				Heating Rate With Shade (Btu/hr)			
	LH_2 Insulation (in.)		LO_2 Insulation (in.)		LH_2 Insulation (in.)		LO_2 Insulation (in.)	
	0.25	2.0	0.25	2.0	0.25	2.0	0.25	2.0
ZSI	28,000	19,000	3,200	2,200	18,500	13,800	2,600	1,800
ZLV	9,700	7,900	1,100	800	8,750	7,200	900	700
XSI	5,150	4,500	500	390	-	-	-	-

vapor phases in the tank. Vapor generated by tank external heat leakage must be vented to effectively control tank pressure; however, this is not easily assured. An alternate method of regulating tank pressure is to use a TVS where a limited quantity of liquid propellant is throttled, vaporized and vented continuously (or on a required basis) in a pressure-controlled environment while passing through a heat exchanger to extract heat and reduce the pressure from the bulk propellant.

TVS's differ primarily in the heat exchanger configuration used to absorb excess energy from the tank contents. TVS concepts include:

1. An internally mounted TVS heat exchanger (employing conduction/free convection heat transfer with no forced mixing)

2. An externally mounted tank wall heat exchanger

3. An internally mounted compact heat exchanger with a pump mixer (employing forced convection heat transfer)

The prime objective of this task is to identify the optimum method of thermodynamically conditioning the LH_2 ring tank for a storage period of eight hours in a low-g (10^{-5} g's) environment. The TVS must be capable of maintaining a tank pressure of 18 psia while the tank is exposed to an external heat load of approximately 7,200 Btu/hr (see Figure 34). This is the heat load resulting with an insulation thickness of 2 inches.

4.2.6.1 Pressure Rise in a Stratified Propellant Tank. The need for thermal conditioning can best be realized by examining the pressure rise in a completely stratified propellant tank. As a heating rate, Q, is entering the tank, newly created boiloff vapor occupies a volume approximately 40 times greater than its original liquid state. Considering the liquid to be incompressible relative to the vapor, the boiloff vapor creates a change in pressure by compressing the initial ullage and restricting the growth of the new vapor layer. Assuming perfect gas behavior, the most rapid rate of pressure rise that is possible in the propellant tank can be given by

$$\frac{\Delta P}{\Delta T} = \frac{R}{(Vg/Vt)} \frac{1}{Cp} \left(\frac{Q}{Vt}\right) \tag{1}$$

where Vg/Vt is the minimum percent ullage volume (~5 percent) and Q/Vt is the heating rate per unit tank volume (see Reference 2). For example, assuming an initial tank pressure of 18 psia and a heat rate of 7,200 Btu/hr, a 1,070-ft^3 ring LH_2 tank will experience a significant pressure rise of 4.0 psia/min. The need for thermal conditioning is apparent. The nomenclature for the TVS-related analyses is presented in Section 4.2.6.4.5.

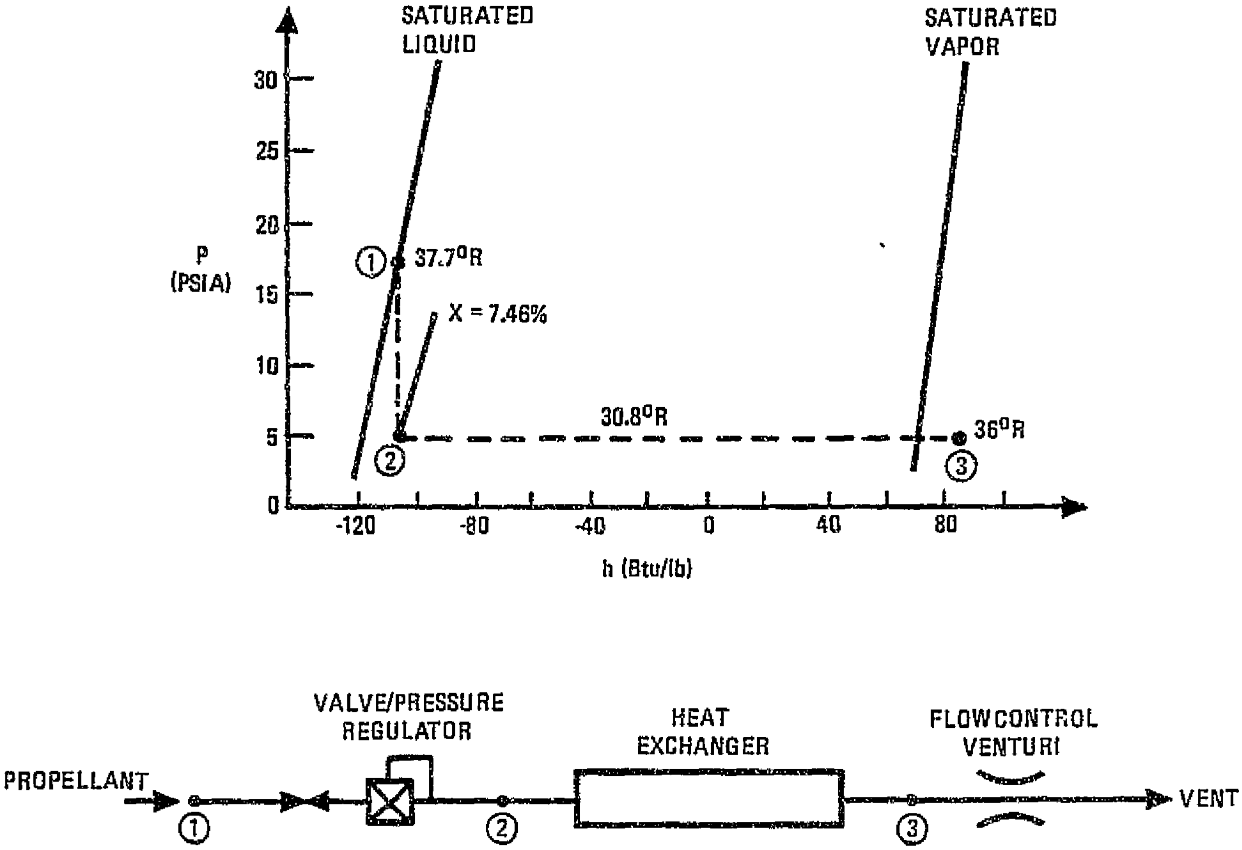
4.2.6.2 Thermodynamic Venting Process. The basic function of thermodynamic venting is to control tank pressure and ensure that liquid is not dumped overboard during the vent process. The thermodynamic process required to accomplish this task is shown in the pressure-enthalpy diagram in Figure 36. During TVS operation, liquid hydrogen (at 37.7°R, 18 psia) fed from a capillary acquisition device is throttled and expanded adiabatically (Joule-Thompson expansion) through a shutoff valve and a pressure regulator down to 30.8°R and 5 psia. The vent fluid is then heated isobarically up to 36°R in a heat exchanger, absorbing excess heat energy from the stored bulk fluids in the tank. The valve and pressure regulator must be sized according to the flow rate and pressure drop (ΔP = 13 psia) requirements so that all of the vented liquid is vaporized before it exits the heat exchanger. A flow-limiting nozzle is incorporated downstream of the heat exchanger to maintain a nearly constant flow rate and back pressure through the TVS line. The throttled-down pressure of 5 psia allows the vented hydrogen to stay above its triple-point pressure (~1 psia), eliminating the possibility of solid hydrogen formation in the flow passages. Hydrogen vapor passing through the nozzle is then vented overboard.

The TVS flow rate required to intercept the 7,200-Btu/hr heat leak can be calculated from the energy balance equation:

$$Q = \dot{m}\,(h_2 - h_1) \tag{2}$$

where Q is the heat leak, H_1 and H_2 are the TVS heat exchanger inlet and outlet enthalpies, and $\dot{m}$ is the flow rate. Taking the enthalpy values for States 1 and 2 from Figure 36, the necessary flow rate for proper heat removal is at least 36.6 lbm/hr (1.03 gpm) of LH_2.

where Q is the heat leak, H_1 and H_2 are the TVS heat exchanger inlet and outlet enthalpies, and $\dot{m}$ is the flow rate. Taking the enthalpy values for States 1 and 2 from Figure 36, the necessary flow rate for proper heat removal is at least 36.6 lbm/hr (1.03 gpm) of LH_2.



STATE/PROCESS

① P = 18.0 PSIA, T = 37.7°R, h = 107.9 Btu/lb

① → ② ISENTHALPIC EXPANSION
P = 5 PSIA, T = 30.8°R, h = 107.9 Btu/lb
QUALITY, X = 7.46%

② → ③ APPROXIMATELY ISOBARIC IN HEAT EXCHANGER
P = 5 PSIA, T = 36°R, h = 86.01 Btu/lb

Figure 36. Thermodynamic Process

4.2.6.3 Pressure/Flow Control Selection. It is necessary to control both the flow rate and the pressure in the cold side of the heat exchanger so the heat loads and the available temperature differences are known. A method of pressure and flow control was presented by Lockheed in Reference 2. Basically, pressure and flow control can be achieved by an intermittent or continuous flow expansion unit. The unit has a dual function, which is performed by independent but physically joined valve sections. The shutoff valve portion of the unit has a snap-acting type of operation that utilizes a Belleville spring and functions to regulate the mass flow rate. The regulator valve portion of the unit functions to maintain an outlet pressure at a predetermined value (4 to 6 psia). Controlling the discharge pressure in this way ensures control over the downstream portion of the thermal conditioning system (e.g., the heat exchanger) and results in improved system performance. A discharge pressure of about 5 psia was selected to keep the TVS fluid sufficiently above the triple-point pressure (~1 psia) to prevent solid hydrogen from forming in the system line.

An expansion unit similar to that discussed above has been developed by Consolidated Controls Corporation (Reference 3). This expansion unit is referred to as a controlling valve module (CVM) by Consolidated Controls and features a redundant shutoff valve and pressure regulator configuration (see Figure 37). The estimated weight of the CVM is approximately 6.6 pounds.

A flow-limiting nozzle will be positioned downstream of the heat exchanger. Since the fluid state at the heat exchanger exit will always be a vapor within a few degrees of the tank temperature and at a pressure of 4 to 5 psia, the nozzle will ensure that the flow through the system essentially will be constant. This then allows the heat exchanger cold-side flow rate to be closely matched to the system requirements.

Assuming that the vapor exiting the TVS heat exchanger may be treated as an ideal gas, the effective area for choked flow in the TVS nozzle can be calculated from

$$A = \frac{W\sqrt{T}}{P} \sqrt{\frac{R}{\gamma gc} \left(\frac{\gamma + 1}{2}\right)^{\frac{\gamma + 1}{\gamma - 1}} \frac{1}{3,600}} \; (\text{in.}^2) \tag{3}$$

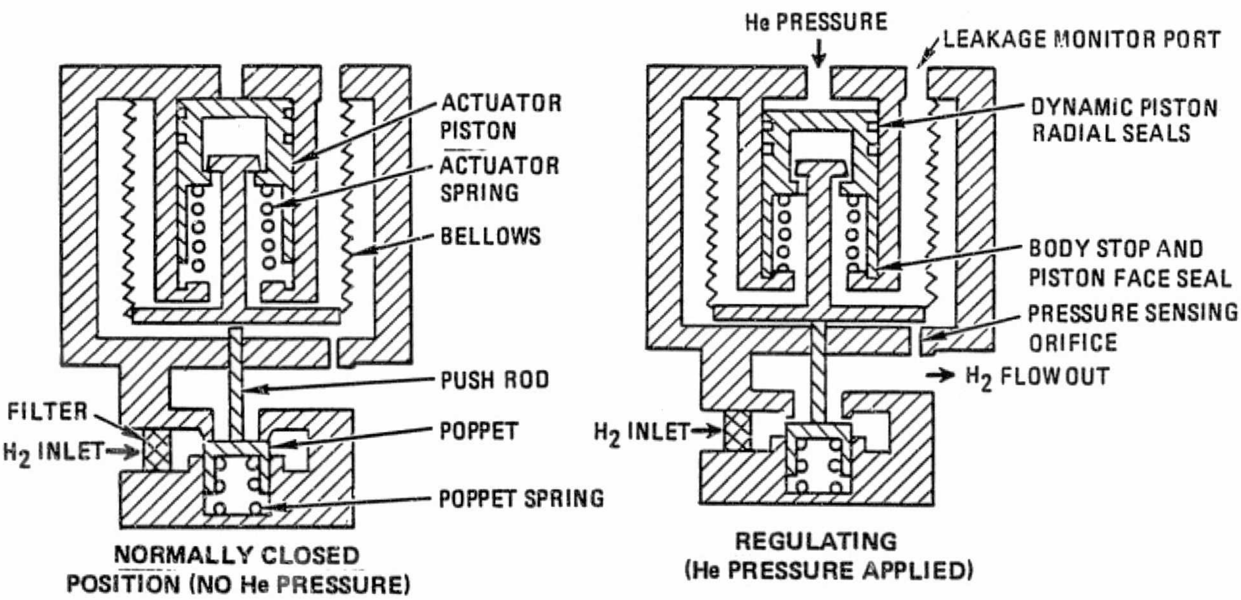


Figure 37. Integrated CVM Design Concept

where

Gc = gravitational constant, 32.2 lbf/lbm sec^2

P = pressure, 4 psig

R = gas constant, 766 ft lbf/lbm °R

T = fluid temperature, 36°R

w = flow rate, 36.6 lb/hr

γ= ratio of specific heats, 1.4

The area required for choked flow at the nozzle throat is A = 0.1086 in.^{2}.

4.2.6.4 TVS Heat Exchanger Feasibility Trade Study

4.2.6.4.1 Internally Mounted, TVS Heat Exchanger. An internally mounted TVS heat exchanger, which basically is a tube that passes through the tank interior, is one method of transferring tank energy to a TVS. The mathematical model that describes the heat transfer process assumes that free convection and conduction are the driving modes of heat transfer to the heat exchanger. Such a system can be called "passive" as opposed to an "active" system, where forced convection (as supplied by a pump mixer) is employed. Assuming that the LH_2 tank uses a capillary channel propellant acquisition system, the internal heat exchanger may be attached to the capillary device as shown in Figure 38. This configuration offers the advantage of providing localized cooling to the screened acquisition channel, precluding the possibility of screen dryout and vapor intrusion into the channels. Although the analysis makes no allowance for thermal stratification, it does provide a basis for determining the length of heat exchanger tubing required for thermal control.

The heat transfer rate into the TVS fluid can be expressed as

$$\frac{dQ}{dt} = \frac{T_B - T_V}{\Sigma_R} \tag{4}$$

where

$$\Sigma R = \frac{1}{h_B A_B} + \frac{\ln r_o/r_i}{2\pi KL} + \frac{1}{h_V A_V}$$

Making an assumption that the tube/capillary channel arrangement may be approximated as a vertical fin, the outside free convection coefficient, h_B, can be given as

$$h_B = \frac{K}{L} Nu \tag{5}$$

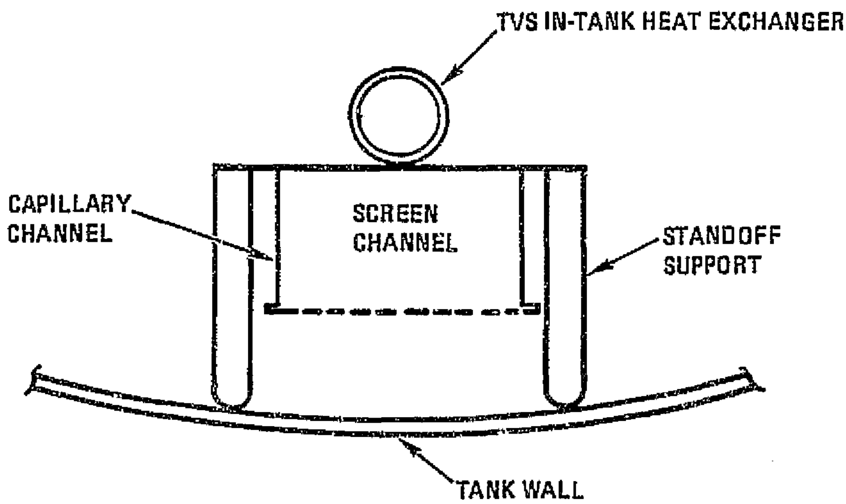


Figure 38. Thermodynamic Vent In-Tank Heat Exchanger

where the Nusselt number, Nu, is given by

$$Nu = \left[\frac{2P_r}{5\left(1+2\,P_r^{0.5}+2\,Pr\right)}\right]^{0.25}(Gr\,Pr)^{0.25}$$

and the Grashof number is

$$Gr = \frac{g\beta\left(T_b-T_w\right)}{\nu^2}\,L^3$$

The effective area for heat transfer is

$$A_B = WL + \pi\,D_oL = L\left(W + \pi\,D_o\right) \qquad (6)$$

where W is the width of the capillary channel, L the length of tubing, and D_o the tubing outside diameter. All equations were evaluated with properties at the bulk fluid film temperature

$$T_F = \frac{T_b + T_w}{2} \qquad (7)$$

Assuming fully developed turbulent flow inside the heat exchanger tube, the inside heat transfer coefficient is given by

$$h_v = Nu\,\frac{K}{D_i} = \frac{K}{D_i}\,(0.023)\,Re^{0.8}\,Pr^{0.4} \qquad (8)$$

Since the heat transfer coefficient, h_v, will vary as the TVS fluid is heated from a two-phase fluid to a single-vapor phase, a simplifying assumption can be made that the first portion of the heat exchanger will contain a two-phase fluid of a known average quality which will remain in this state until the heat required to vaporize this fluid has been absorbed. After the heat of vaporization has been absorbed, the remainder of the heat exchanger will contain a single-phase vapor. The inside heat exchanger area for heat transfer is

$$A_v = \pi D_i L \tag{9}$$

The results of this analysis are presented in Figure 39 for heat transferred versus length of tubing for 3/8-inch nominal 304-A stainless steel. The results indicate that for TVS operation ($\dot{m}$ = 36.6 lbm/hr LH_2) in a 10^{-3} g environment, a heat exchanger length of 2,820 feet is needed to transfer 7,200 Btu/hr. For TVS operation in 10^{-5} g's, this same length of tubing will only transfer 2,214 Btu/hr out of the 7,200-Btu/hr total heat leak. The effect of the gravity level on the heat transfer rate is well illustrated. Although the tubing may provide heat removal capability for temperature control, thermal stratification effects may cause excessive pressure rises, and only that area of vent tube exposed to the vapor fan be relied on to effectively control the pressure. The length of tubing to compensate for anticipated stratification effects will be several times greater than that required to maintain the temperature of the liquid. This preliminary feasibility analysis indicates that an internal passive heat exchanger is not practical

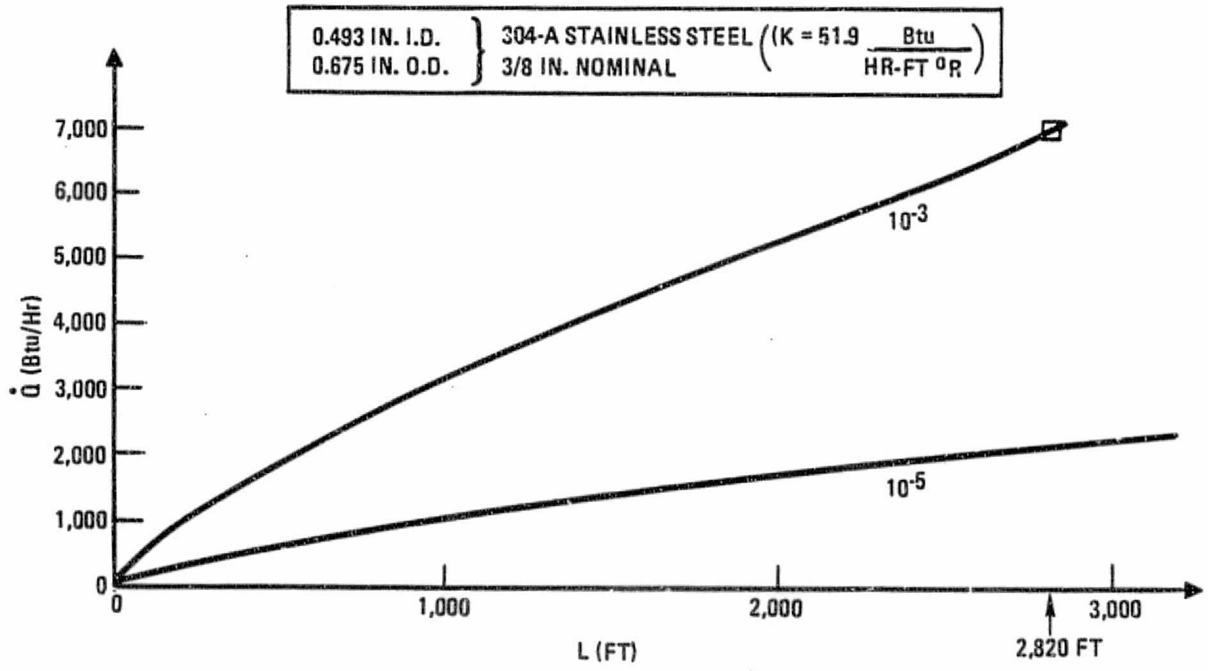


Figure 39. Heat Transfer Vs. Internal Heat Exchanger Length

for TVS applications where high leakage rates are encountered under low-g conditions due to the excessively long heat exchanger tubing required to transfer propellant energy adequately for pressure control. Hence, further design analysis is not warranted.

4.2.6.4.2 Externally Mounted Tank Wall Heat Exchanger. An externally mounted heat exchanger consists of a tube or channel attached to the tank exterior. The wall surface serves as an extension of the tube as a fin to enhance the effective heat transfer area and provide a path to intercept the incident heat flux to the tank. Some primary advantages of this concept are as follows:

1. Heat leakage to the bulk liquid and vapor can be controlled separately. This decouples the control of the ullage pressure from that of the bulk liquid temperature.

2. Vapor formation along the walls can be controlled. Heat is intercepted before tank absorption.

A few disadvantages of this system are as follows:

1. Any tube leakage would impact the performance of the tank insulation.

2. The concept requires that the tank wall be capable of structurally supporting the tube attachment and be sufficiently thick to provide adequate paths for heat conduction.

3. Hardware weight may be greater than for other TVS concepts.

4. Vapor bubbles within the propellant will not be collapsed easily.

The following analysis, given by Reference 4, provides a method of determining the external heat exchanger tube spacing for proper TVS operation. The model assumes that the wall temperature at the midpoint between tubes is kept at the bulk fluid temperature. The differential model for the analysis is shown in Figure 40. Assuming convection within the propellant tank and conduction in the tank wall, the heat transfer differential equation is given by

$$K_W \delta C_i \frac{\partial^2 T_x}{\partial x^2} - hC_i\left(T_x - T_F\right) = -qC_i \tag{10}$$

which reduces to

$$\frac{\partial^2 T_x}{\partial x^2} - \frac{h}{K_W \delta} \cdot T_x = \frac{-\left(q + hT_F\right)}{K_W \delta} \tag{11}$$

The solution to the above equation is

$$T_x = ae^{-\lambda x} + be^{\lambda x} + q/h + T_F \tag{12}$$

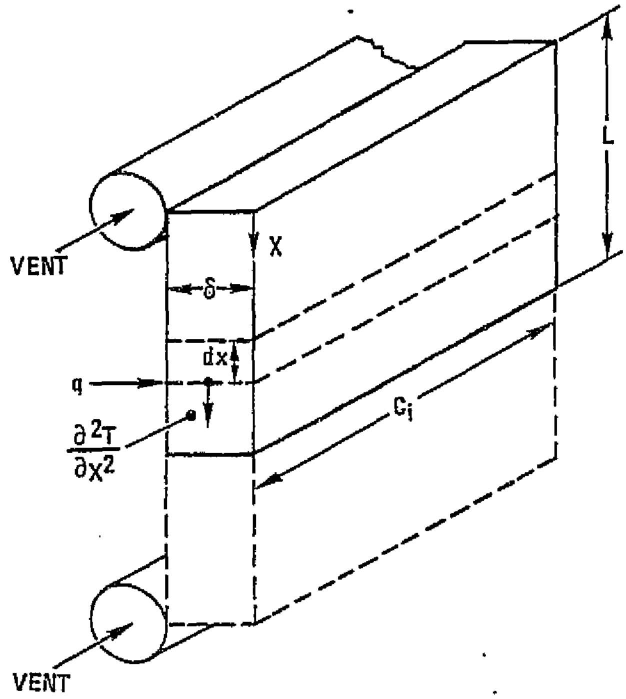


Figure 40. External Heat Exchanger Model

where

$$\lambda + \left(\frac{h}{K_w \delta}\right)^{1/2}$$

At the boundary conditions

$$X = 0,\ T_x = T_v \qquad (13)$$

$$X = L,\ T_x = T_s,\ \frac{\partial T}{\partial X} = 0 \qquad (14)$$

Therefore, by differentiating the general solution and applying the boundary conditions

$$\frac{\partial T}{\partial X} = a\lambda\ e^{-\lambda x} + b\lambda\ e^{\lambda x}$$

$$a = b\ e^{2\lambda L}$$

$$T_L = a\ e^{-\lambda L} + b\ e^{\lambda L} + q/h + T_F + T_s$$

$$2b\ e^{\lambda L} + q/h + T_F = T_s$$

We obtain the expressions for the coefficients a and b

$$a = \left[\frac{(T_s - T_F) - q/h}{2}\right] e^{\lambda L}; \; b = \left[\frac{(T_s - T_F) - q/h}{2}\right] e^{-\lambda L} \tag{15}$$

If we let $T_F = T_s$

$$T_x = -q/h \cosh\left(\lambda(L-X)\right) + q/h + T_s \tag{16}$$

Therefore

$$T_o = -q/h \cosh(\lambda L) + q/h + T_s = T_v \tag{17}$$

and

$$T_s - T_v = -q/h \left[1 - \cosh(\lambda L)\right] \tag{18}$$

From which the final solution can be written as

$$\cosh(\lambda L) = \frac{(T_s - T_v)h}{q} + 1 \tag{19}$$

In the above equation, T_s and T_v are the bulk saturation and venting temperatures, respectively; q is the heat flux; h is the heat transfer coefficients; and L is the half spacing between the external heat exchanger tubing.

The average inside heat transfer coefficient, h, is given

$$h = \frac{K}{L} Nu \tag{20}$$

where

$$Nu = 0.13\, Ra^{1/3}; \; Ra = G_r P_r$$

and

$$G_r = \frac{\beta g \Delta T L^3 \rho^2}{\mu^2}$$

The heat flux, q, is the heat rate, Q (7,100 Btu/hr), divided by the tank surface area. All equations were evaluated with properties at the tank saturation temperature and pressure (37.7°R, 18 psia). The vent temperature, T_v, was assumed to be 30.8°R and the tank wall thickness, δ, was taken to be

0.06 inch. Results indicate that the tube spacing required to intercept 7,100 Btu/hr is 7.93 inches. If the LH_2 ring tank is wrapped with tubing, approximately 1,625 feet of 3/8-inch stainless steel tubing weighing 185 pounds is needed for the external heat exchanger.

One of the initial assumptions in the analysis was that the incident heat flux was uniformly distributed throughout the tank surface. This assumption may not be entirely realistic, and the heat exchanger length may actually be longer than the analysis suggests. Also, considering that highly efficient heat transfer is desired, welding the heat exchanger tubing to the wall is required. However, there is no acceptable way to attach the vent tubes structurally without greatly adding weight to the tank. Thin tank walls may also provide a poor conduction path for heat transfer and reduce the effectiveness of the method.

4.2.6.4.3 Internally Mounted Compact Heat Exchanger With Pump Mixer. Since various missions involve proplonged orbital coast at low gravity, where natural or free convective heat transfer is low, heat can be transferred more effectively from the liquid propellant to the heat exchanger by forced convection. This concept is shown schematically in Figure 41. The pump mixer, in addition to providing forced convection across the heat exchanger, will also limit thermal stratification and ensure pressure control by providing circulation and uniform heat transfer through the liquid.

The cold-side TVS heat exchanger flow rate has been previously determined to be 36.6 lbm/hr of LH_2. The hot-side forced convection flow rate can be calculated from

$$Q = \dot{m}\ C_p \Delta T \tag{21}$$

assuming that the bulk fluid maintains a nearly constant C_p over the heat exchanger temperature range. Assuming a ΔT of 2°R and a C_p of 2.3 Btu/hr°R, the hot-side flow rate required to transfer 7,200 Btu/hr is approximately

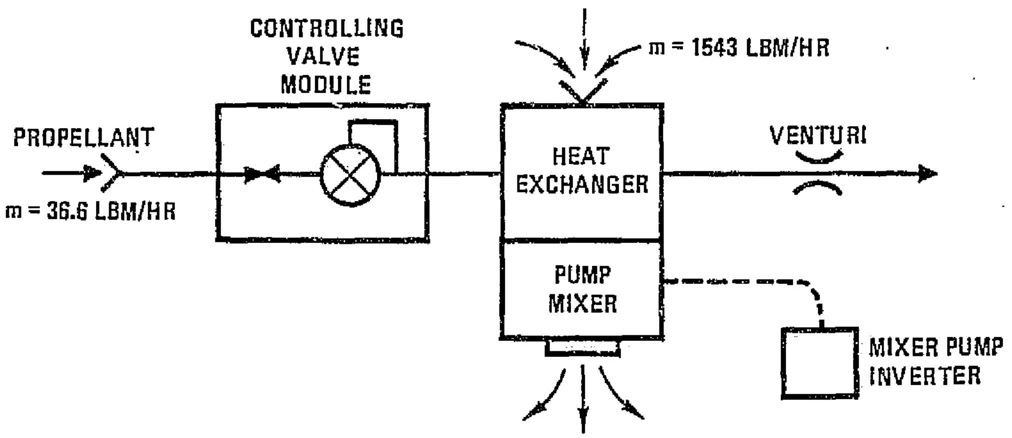


Figure 41. Internal Compact Heat Exchanger TVS Concept

Table 22. Compact Heat Exchanger Requirements

	Mass Flow Rate (lbm/hr)	Pressure (psia)	Temperature (R)	Quality
Hot-side inlet	1,543	18.1	38.0	0.0
Hot -side outlet	1,543	18.0	36.0	0.0
Cold-side inlet	36.6	5.1	30.8	0.075
Cold-side outlet	36.6	4.0	36.0	1.0
Note: Effective heat transfer coefficient ~575 Btu/hr ft^2°R				

1,543 lbm/hr (43.7 gpm) of LH_2. The TVS compact heat exchanger flow rate, temperature, and pressure drop requirements are summarized in Table 22.

Regardless of the flow conditions, delivery of dry superheated vapor out of the TVS heat exchanger is required. One compact heat exchanger design that could potentially satisfy the stringent TVS performance requirements is a helical tube coil (Reference 5). Swirl flow in a helical coil would induce a relatively high artificial g field that would force a liquid layer onto the tube walls for the majority of the coil's length. This would provide for high heat transfer coefficients with relatively small heat exchanger sizes.

The heat transfer capabilities of a helical coil can best be illustrated by Figure 42, which presents the helical flow heat transfer coefficients for high-quality steam. Heat transfer coefficients as high as 6,000 Btu/hr ft^2°F were attainable with a coiled design. Test runs indicated that no moisture was detectable at the heat exchanger outlet even with only several degrees of superheat (References 5 and 6). The helical coil heat exchanger has been demonstrated to be an effective design for energy transfer and appears to be the one most capable of meeting the TVS requirements.

Sundstrand Corp. has investigated using helical coiled compact heat exchangers similar to that shown in Figure 43 for the Centaur LH_2 zero-g TVS. The heat exchanger compresses two helical flow paths machined from aluminum in which the cold-side flow is constrained to flow through the inside passage and the hot-side flow passes through the outside passage in a counter-flow fashion (Reference 5). Heat transfer coefficients of 500 to 887 Btu/hr ft^2°F will be 14 inches in diameter and 24 inches long and will weigh approximately 24 pounds, including the pump mixer. A heat exchanger design similar to that described above appears capable of transferring 7,200 Btu/hr.

A pump mixer similar to that shown in Figure 44 is electrically powered; thus, dissipation of this power introduces a source of heat. For a mixer to

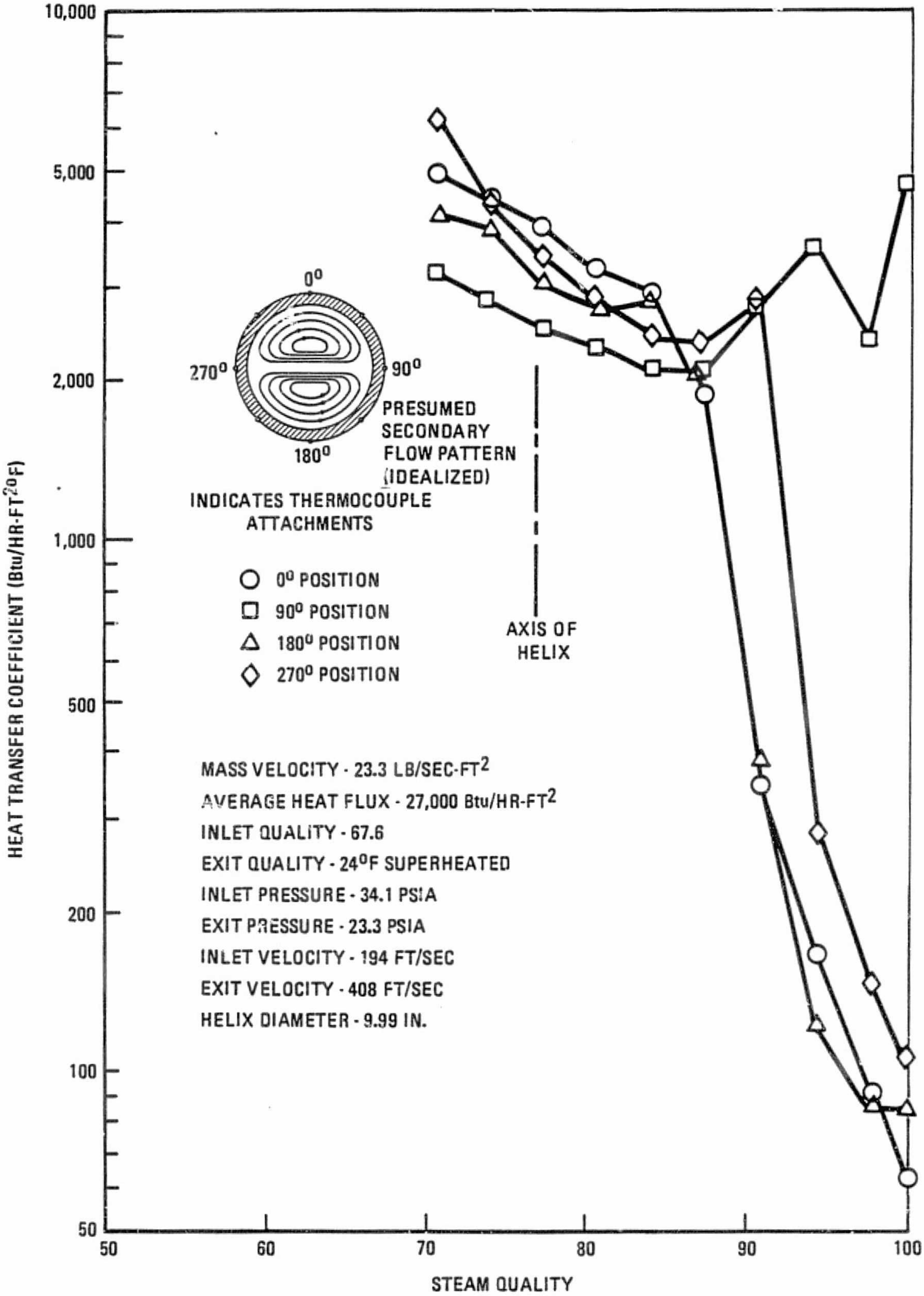


Figure 42. Helical Flow Evaporation Heat Transfer Coefficients for High-Quality Steam

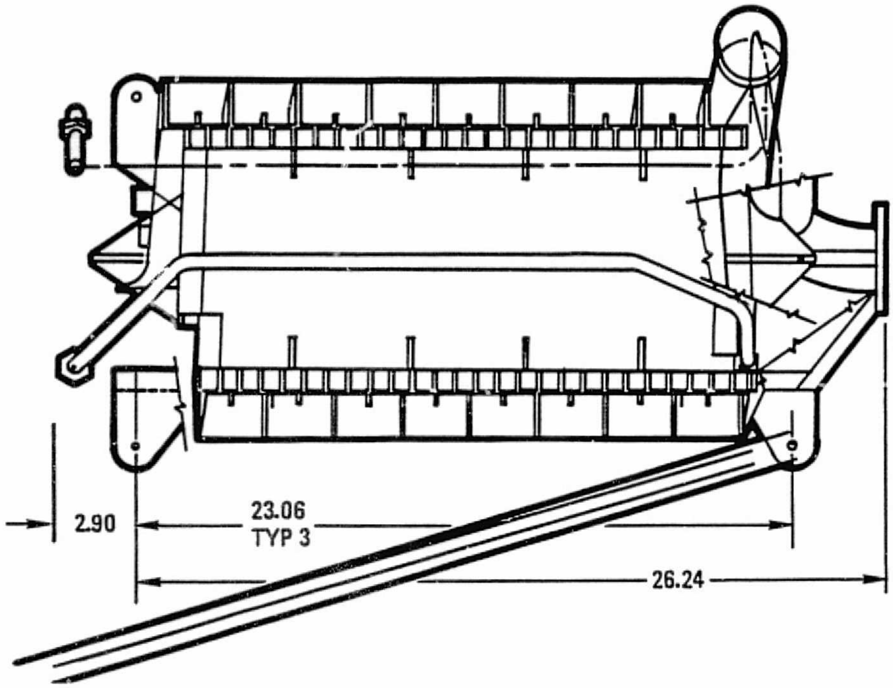


Figure 43. Sundstrand Heat Exchanger Configuration

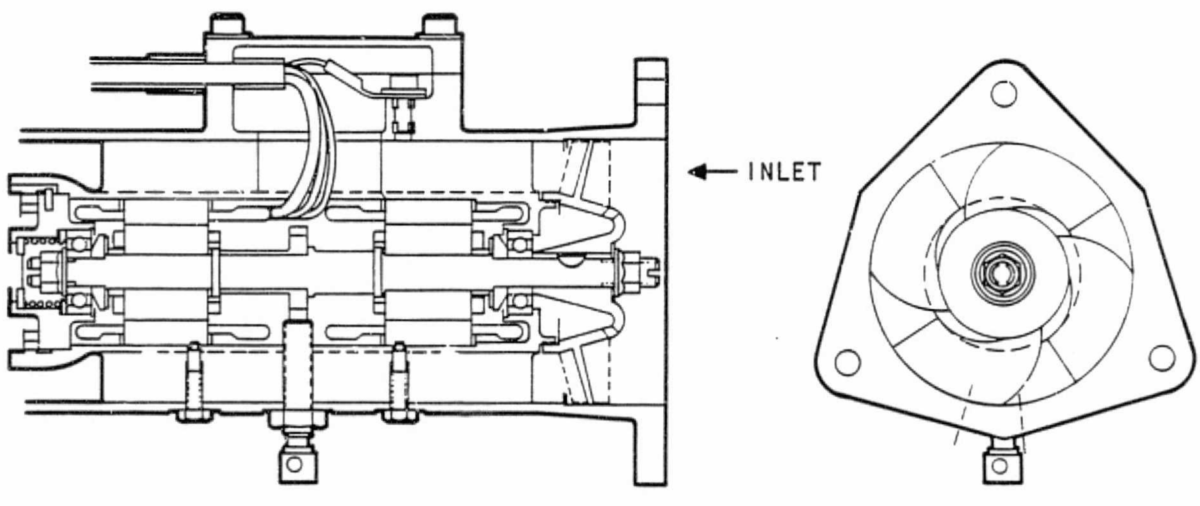


Figure 44. Pump Mixer

pump 60 gpm of LH_2, a nominal mixer power usage of 6 watts (20.5 Btu/hr) has been reported (Reference 6). For a flow rate of 43.7 gpm, the estimated power usage is 4.4 watts (15 Btu/hr). This additional heat input essentially is negligible in comparison to the 7,200 Btu/hr heat leak and poses no severe disadvantage to the compact heat exchanger concept. The estimated weight for the pump mixer is approximately 2.3 pounds.

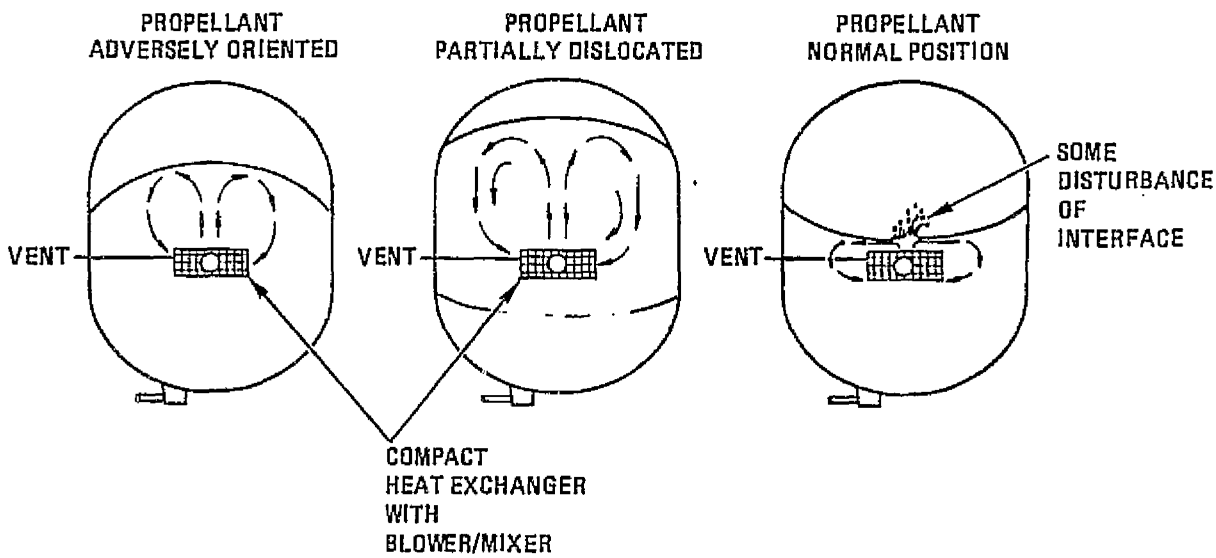


Figure 45. Thermodynamic Vent System With Compact Heat Exchanger Concept

Figure 45 shows three potential propellant-heat exchanger spatial relationships. The compact heat exchanger concept is feasible regardless of whether the heat exchanger is immersed in liquid or vapor. If it is immersed in vapor, the forced flow of vapor through the warm side of the heat exchanger cools and/or condenses the vapor thus limiting the tank pressure. In this case, it is not necessary to mix the liquid for destratification since ullage condensation will keep the ullage pressure within limits. If the heat exchanger is immersed in liquid, bulk cooling and mixing will limit the tank pressure.

4.2.6.4.4 System Selection. Three TVS's have been presented and evaluated. The passive internal heat exchanger concept is favorable because of its simplicity; however, the high heat loads and low-gravity free convective heat transfer have rendered this concept impractical due to the extreme lengths of heat exchanging tubing required. The external tank wall heat exchanger concept can be eliminated for similar reasons. In each case, free or natural convection cannot guarantee successful heat transfer of 7,200 Btu/hr to the vent system.

The most favorable TVS concept for low-g, high-heat load operation is the internally mounted compact heat exchanger with forced convection (Figure 41). This system is ideal because of its small size and low weight. The combined weight of the TVS is approximately 34 to 44 pounds, which includes a pump mixer and inverter, coiled heat exchanger, controlling valve module, and nozzle (Table 23). Another asset of this concept is the availability of existing hardware. Components for the TVS may be supplied by a vendor with little or no modification of existing hardware.

Table 23. Estimated Thermodynamic Vent System Weight

Component		Weight (lb)
Mixer pump		2.3
Coiled heat exchanger		20-30
Controlling valve module (integrated redundant shutoff valves and pressure regulators)		6.6
Nozzle		2.0
Inverter (for mixer pump)		3.1
	Total	34-44

4.2.6.4.5. TVS Analytical Nomenclature

A effective nozzle throat area

A_B internal heat exchanger outer surface area

A_v internal heat exchanger inner surface area

C_p specific heat ratio

D_i inside diameter

D_o outside diameter

g gravity field

g_c gravitational constant

Gr Grashof number

h enthalpy/heat transfer coefficient

h_b external heat transfer coefficient

h_v internal heat transfer coefficient

K thermal conductivity

L length

m mass flow rate

Nu Nusselt number

P pressure

Pr Prandtl number

Q heat rate

q heat flux

R gas constant

Re Reynolds number

r_i inside radius

r_o outside radius

T_b bulk fluid temperature

T_v vent fluid temperature

T_w tank wall temperature

V_g ullage volume

V_t tank volume

w flow rate

β volume coefficient of expansion

δ tank wall thickness

μ dynamic viscosity

ν kinematic viscosity

ρ density

4.2.7 Capillary Acquisition System (CAS)

The LH_2 and LO_2 propellant tank systems must be designed to have the capability to feed out vapor-free liquid cryogens with maximum expulsion efficiency under the reduced gravity of the LEO space environment. To meet this requirement, it is necessary to configure a system so that a part of the capillary flow acquisition device is always in contact with the bulk liquid regardless of where the liquid is positioned in the tank. Capillary devices derive their effectiveness from the phenomenon of bubble pressure; i.e., the pressure required to drive a bubble through a hole in a screen. Under the worst-case conditions of flow rate and on-orbit acceleration, the total hydrostatic and hydrodynamic pressure losses in the liquid flow passages must be less than the screen bubble point pressure so that the ullage vapor will not be prematurely ingested into the channel system and reduce expulsion efficiency.

An attractive propellant acquisition system consists of multiple screened channels mounted along the tank wall circumference. The screened capillary channels are basically rectangular ducts of width, D, and depth, δ, screened on one side with 325 by 2,300 mesh Dutch twill stainless steel. The 325 by 2,300 screen was selected because of its high bubble point and the wealth of experience in its fabrication and use, including the orbiter program. The screened channels are separated from the tank wall a suitable distance by nylon supports to decrease the heat transfer effects of the thermal boundary layer from the tank wall on the channels (Figure 46). These channels are then connected to a common manifold to provide a fluid sump for feedout.

The CAS for the toroidal LH_2 tank consists of four equidistant radial channel loops with two interconnecting channels on the tank top and bottom as shown in Figure 47. The worst-case LH_2 propellant orientation is assumed to be located on the tank side as shown in the figure. The tank length is initially assumed to be 70 inches.

The CAS for the cylindrical LO_2 tank consists of four equidistant radial channels feeding to a common manifold as shown in Figure 48. The worst-case propellant orientation is assumed to be at the tank top, opposite the feedout line.

4.2.7.1 CAS Analysis. A schematic of the capillary channel network is shown in Figure 49. The analytical model for fluid flow through the channel can be divided into tw major sections: (1) flow through the manifold (submerged) section of the channel, where the velocity varies with length and (2) flow through the channel exposed to the tank ullage, where the velocity remains relatively constant.

4.2.7.1.1 Analytical Model for Flow in the Manifold. The analysis for fluid flow in the CAS manifold section, given in References 7, 8, and 9, is presented below. A nomenclature for the CAS analysis is presented in Section 4.2.7.2. The analytical model is shown in Figure 50, where the

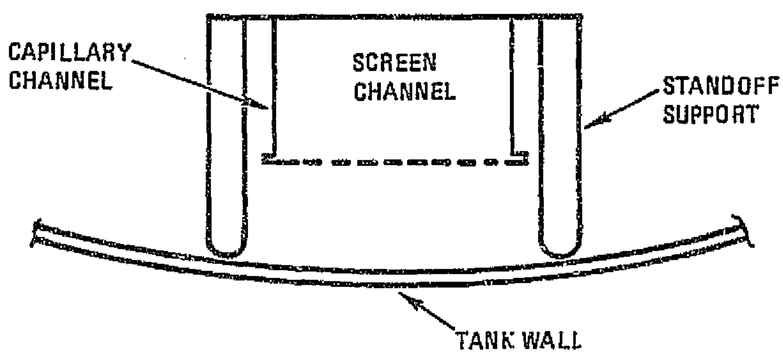


Figure 46. Capillary Channel

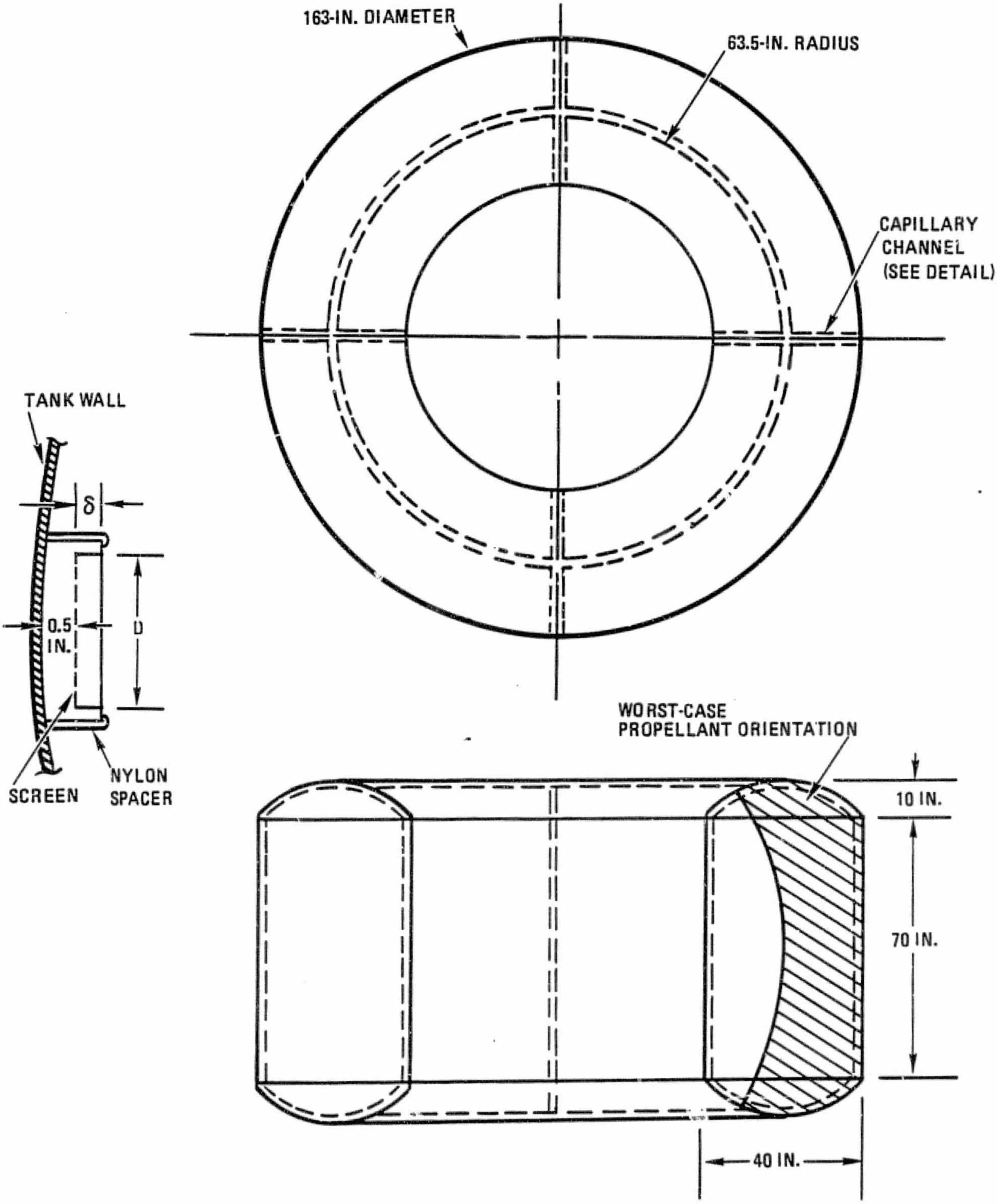


Figure 47. LH_2 Tank Configuration

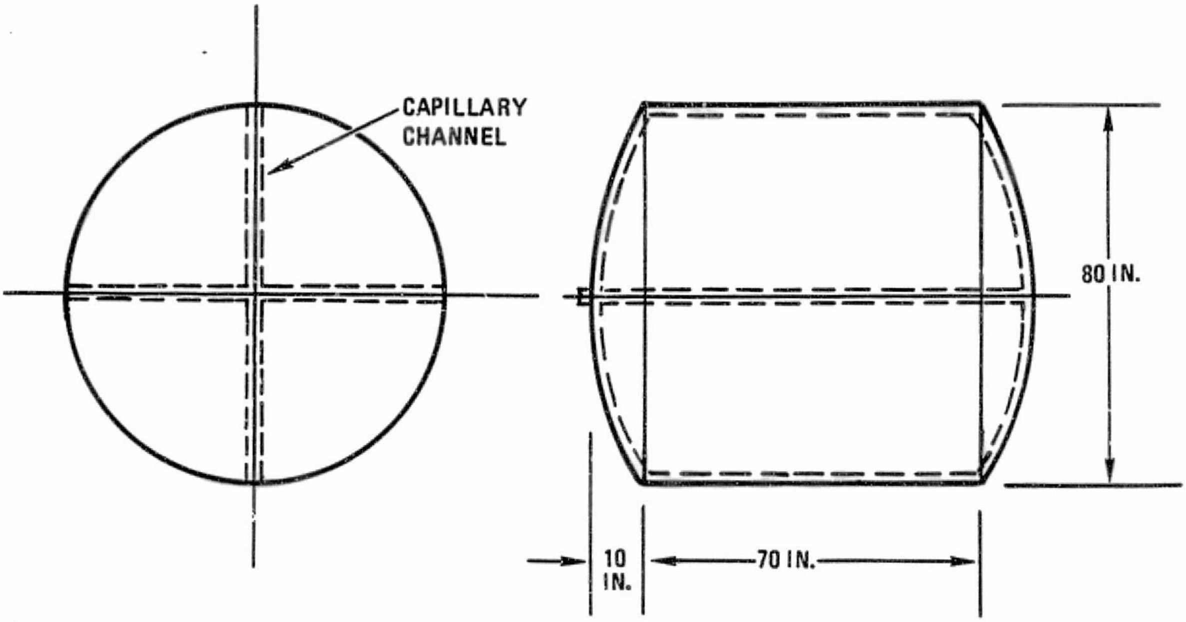


Figure 48. LO_2 Tank Configuration

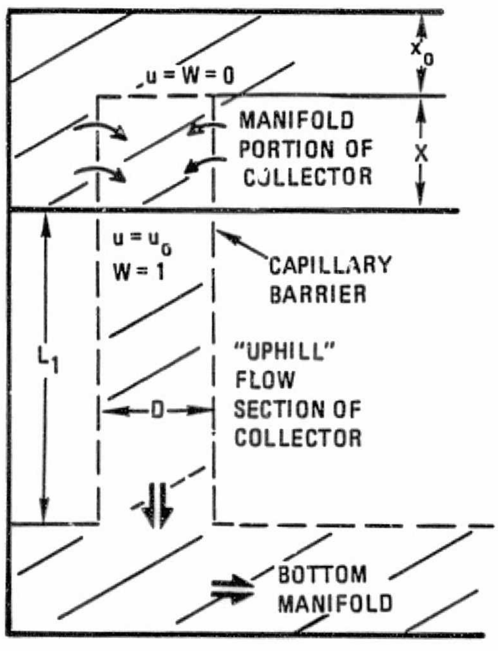


Figure 49. Tank and Collector Schematic

differential element of the capillary collector is shown with the positive vertical direction upward. The positive direction for the gravitational field is vertically downward, and X and u are positive in the upward direction. Liquid flows through the capillary barrier walls into the collector along its length and then downward.

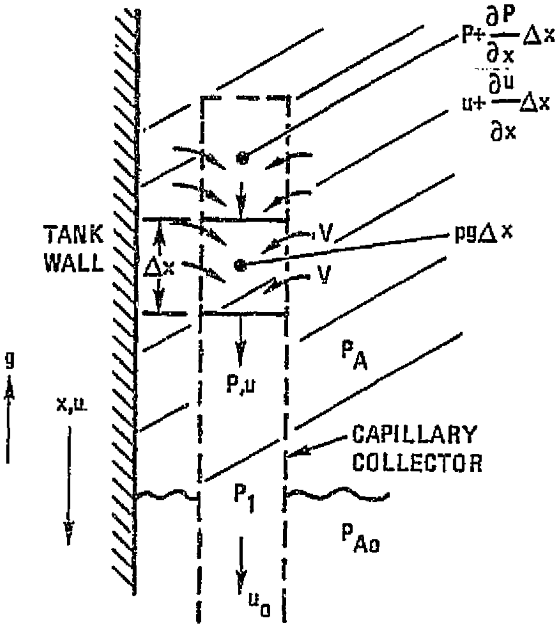


Figure 50. Manifold Portion of Capillary Collector

Assuming a rectangular capillary collector channel screened on one side only with screen width, D, and with channel depth, δ, the conservation of mass (continuity) equation is given by

$$\left[u - \left(u + \frac{du}{dx}\Delta x\right)\right] \delta D + V D\Delta x = 0$$

which simplifies to

$$V = \delta \frac{du}{dx} \tag{22}$$

The pressure drop of the liquid flowing across the capillary barrier into the collector is given by

$$P_A - P = \frac{E_u \rho V^2}{2g_c} \tag{23}$$

The pressure, P_A, of the liquid within the tank is given by

$$P_A = P_{AO} - \rho g x \tag{24}$$

where P_{AO} is the tank ullage pressure. By rearranging Equations 22 through 24, there results

$$du = \frac{1}{\delta E_u^{1/2}} \left[\frac{2g_c\left(P_{AO} - \rho g x - P\right)}{\rho} \right]^{1/2} dx \tag{25}$$

A vertical momentum balance gives

$$\frac{dP}{dx} = -\frac{2k\rho}{g_c} u \frac{du}{dx} - \rho g + \frac{\rho f u^2}{g_c D\delta}(D+\delta) \tag{26}$$

In the above equations, k is an empirical manifold constant. In the numerical computations that follow, a value of k of 0.79 was used.

To normalize the equations and generalize the results, the following variables are defined

$$R = \frac{P_{AO} - P}{P_B} \tag{27}$$

$$W = \frac{u}{u_o} \tag{28}$$

$$\eta = \frac{x - x_o}{X} \tag{29}$$

where P_B is the bubble pressure of the capillary material, u_o is the propellant velocity at the end of the manifold, x_o is the distance from the top of the tank to the top of the collector, and X is the length of the manifold portion of the capillary collector.

The differentials are given by

$$dR = dP/P_B \tag{30}$$

$$d\eta = dx/X \tag{31}$$

Substituting Equations 27 through 31 into 25 and 26, placing in finite difference form, and changing to a coordinate system that is positive in the downward direction, we obtain Equations 32 and 33

$$\Delta W = \frac{X}{\delta u_o} \left[\frac{2g_c\rho_B}{\rho E u} \left(R + \frac{\rho g\left[X\left(\eta + \frac{\Delta\eta}{2}\right) + x_o\right]}{g_c P_B} \right) \right]^{1/2} \Delta\eta \tag{32}$$

$$\Delta R = \frac{2k\rho u_o^2}{g_c P_B}\left(W + \frac{\Delta W}{2}\right)\Delta W - \frac{\rho g X}{P_B}\Delta\eta + \frac{\rho f uo^2\left(W + \frac{\Delta W}{2}\right)^2 X\ (D+\delta)}{g_c D P_B \delta}\Delta\eta \qquad (33)$$

These equations are solved numerically with the manifold divided into 100 elements with the boundary conditions $W_1 = 0$, $W_{100} = 1$ and $\eta_1 = 0$. The results are used to calculate the manifold exit pressure, P_1.

The equations for the Euler number and pressure drop across a capillary screen barrier given by Reference 7 are

$$E_u = \frac{2t}{D_a}\left[\frac{A_1}{(R_e)}\right] \text{ and } \Delta P = \frac{Eu\rho v^2}{2g_c}$$

Combining the above equations results in:

$$\frac{A_1}{Re_1} = \frac{\Delta P g_c D_a}{t\rho v^2} \qquad (34)$$

where $Re_1 = D_a V\ \rho/\mu$

The pressure drop across a screen can also be described in terms of a friction factor, f, and the Reynolds number, Re_2, as given by References 10 and 11

$$f = \frac{\alpha}{Re_2} = \frac{\Delta P g_c \varepsilon^2 D_a}{Qt\rho v^2} \qquad (35)$$

$$Re_2 = \frac{\rho v}{\mu a^2 D_a} \qquad (36)$$

Dividing Equation 15 by 13 and rearranging terms produces the final form of the constant, A_1

$$A_1 = \frac{\alpha Q Re_1}{\varepsilon^2 Re_2} = \frac{\alpha Q D_a{}^2 a^2}{\varepsilon^2} \qquad (37)$$

The values of each of the constants in the above equations for 325 by 2,300 Dutch twill screen material are $\alpha = 3.2$, $a = 3.3598 \times 10^4$, $D_a = 1.64 \times 10^{-5}$, $\varepsilon = 0.245$, and $Q = 1.3$. Therefore, the constant A_1 has value $A_1 = 21.04$.

4.2.7.1.2 Constant Velocity Flow in the Channel Exposed to the Ullage. In the exposed, or uphill, portion of the channel, the velocity, U_o, no longer varies with length; and the pressure drop is the sum of the viscous and gravitational terms. The pressure, P_2, at the end of the channel is given by

$$P_2 = P_1 - \frac{\rho g L_1}{g_c} - \frac{\rho f L_1 U_o^{\,2} (D+\delta)}{D\delta g_c} - \frac{1.3\ \rho U_o^{\,2}}{2g_c\ P_B} \qquad (38)$$

where the last term is the pressure loss due to a right angle turn in the collector channel. The pressure differential across the screen interface at the end of the channel must satisfy the ullage and bubble pressure relationship

$$P_{AO} - P_2 \leq \phi P_B \qquad (39)$$

to prevent ullage gas intrusion. The constant, ϕ, is a safety factor less than one.

A computer program has been developed to solve the system equations for the maximum channel velocity allowable in the following fashion. An initial value for the terminal fluid velocity, Uo, is assumed and is used in the first iteration. The manifold and exposed channel equations then generate pressures, P_1 and P_2. If the value of P_2 does not satisfy the bubble pressure constraint, a new value of Uo is obtained by bisection, and the process is repeated until the bubble pressure relationship is satisfied. Convergence is rapid, taking about nine iterations around Uo. The maximum flow rate, $\dot{m}$, is then calculated from the resulting velocity.

Ten channel geometries were considered for each tank capillary system in the initial trade study. Channel dimensions varied from 7.0 to 3.0 inches in width and 2.0 to 0.75 inch in depth. The effects of gravity were assumed negligible, and the tank pressures were taken to be 27.6 psia and 28.81 psia for the LO_2 and LH_2 tanks, respectively. Results are summarized in Figures 51 through 53 for three of the ten channel geometries analyzed. Briefly, the results indicate that 7- by 1.5-inch channel is required for an LH_2 feedout capability of 8 lbm/sec, assuming a worst-case propellant orientation. 5- by 1-inch channel is required for an LO_2 feedout rate of 34 lbm/sec. The above flow rates are the design capabilities for each of the propellant tank systems (see Section 4.4.1.3). The above channel geometries will be reevaluated at a later time as the scavenging tank system operating parameters are better defined.

4.2.7.2 <u>Nomenclature for Analytic Model.</u>

- a1 calculated surface area to unit volume ratio
- A_1 empirical constant for pressure loss in flow through capillary barrier
- D capillary channel width
- D_a characteristic pore size for capillary barriers

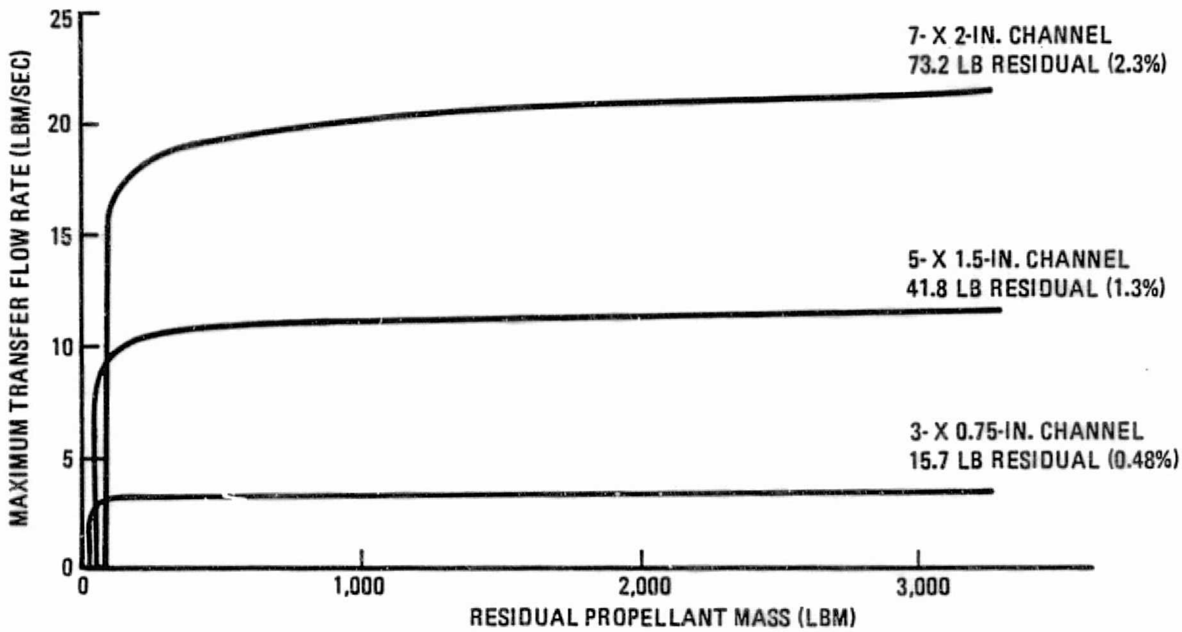


Figure 51. CAS Outflow Capability--LH_2 Tank (P = 28.81 psia)--LH_2 Oriented at Tank Top

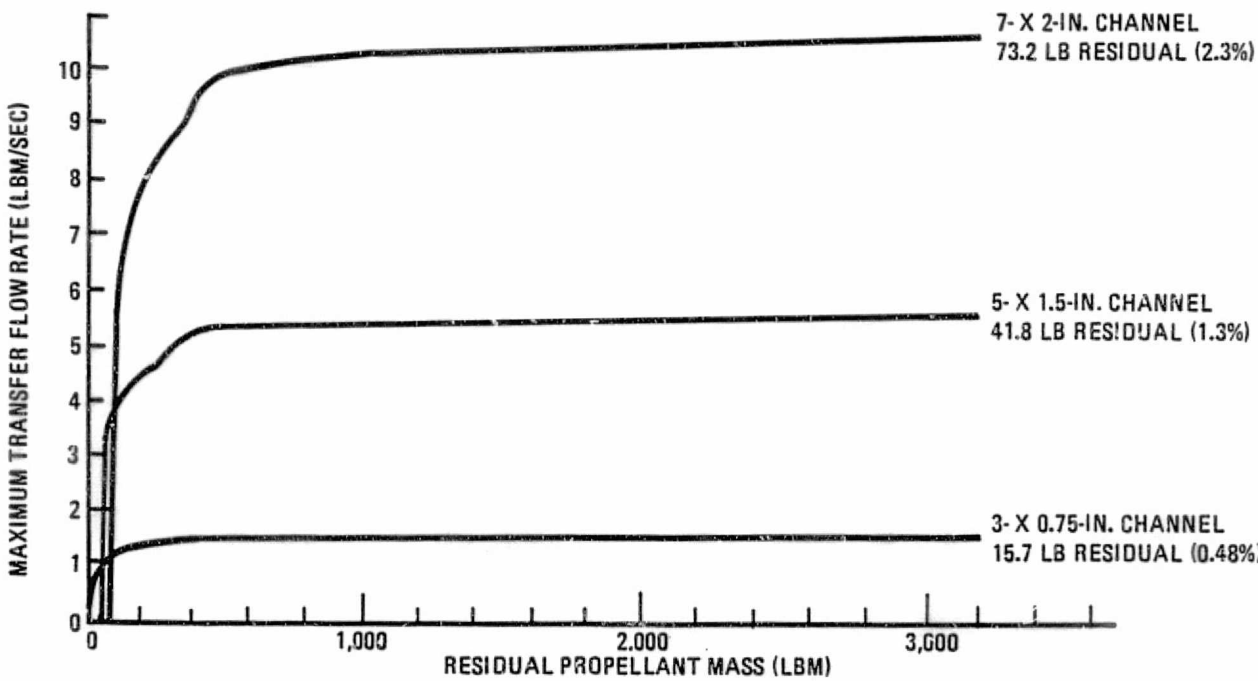


Figure 52. CAS Outflow Capability--LH_2 Tank (P = 28.81 psia)--LH_2 Oriented at Tank Side

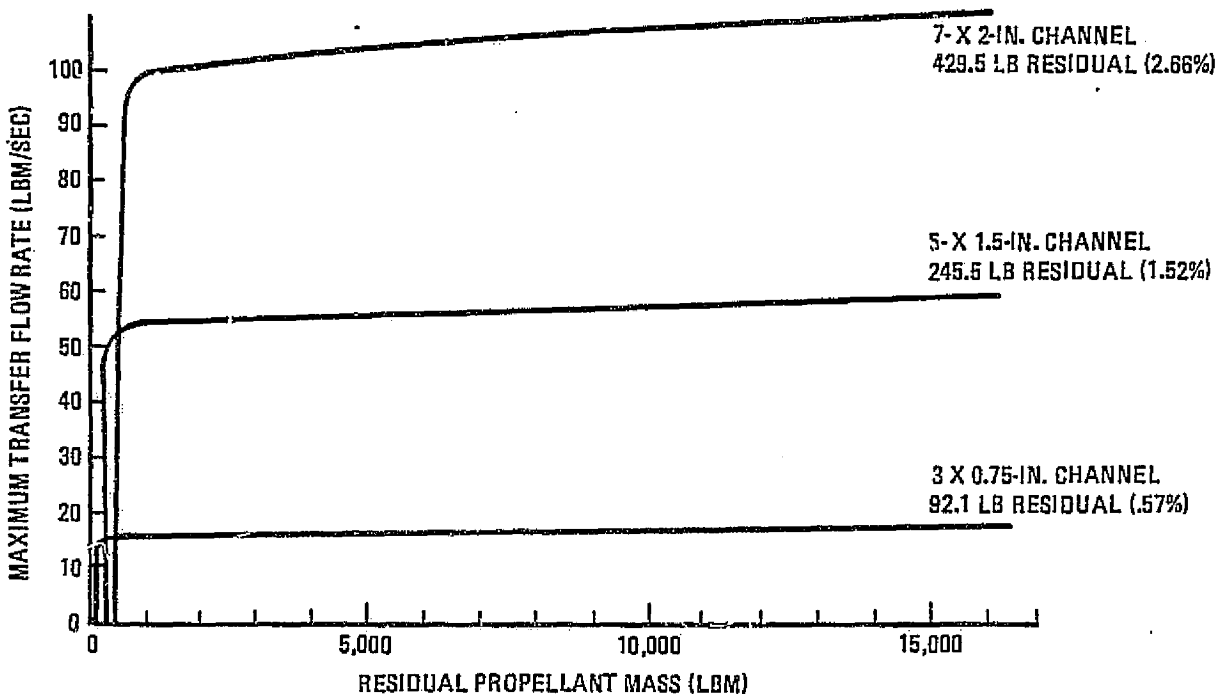


Figure 53. CAS Outflow Capability--LO_2 Tank (P = 27.6 psia)--LO_2 Oriented at Tank Top

Eu	Euler number
f	friction factor
g	gravitational field
g_c	earth's gravity field
K	manifold screen constant
L_1	length of exposed channel
P	pressure
P_A	pressure outside of collector
P_{AO}	ullage pressure
P_B	bubble pressure
Q	tortuosity factor
R	dimensionless pressure ratio
Re	Reynolds number
t	screen thickness

U velocity parallel to capillary barrier

U_o velocity in exposed portion of channel

V velocity perpendicular to capillary barrier

W dimensionless velocity

X submerged channel length

X_o distance from top of tank to collector channel

α viscous resistance coefficient

δ capillary channel depth

Δ incremental difference

η dimensionless distance

ε screen volume void fraction

μ fluid viscosity

ϕ safety factor, 0.65

ρ fluid density

4.2.7.3 Removing Vapor in Dry Capillary Channels. It is expected that the LO_2 and LH_2 propellant scavenging tanks will not be filled to capacity for launch; therefore, the CAS channels will not be completely submerged in liquid during the ground hold and ascent mission phases. Any portion of the screened channel that is not submerged in liquid will contain ullage vapor. Provisions must be made to eliminate this vapor prior to propellant transfer on orbit.

There are several possible methods of eliminating vapor bubbles trapped in the capillary channels. One method is to prevent vapor from entering the system by truncating the channels so no portion of the CAS is exposed to the ullage vapor during the ground hold. This would assure that the CAS remains immersed in liquid during the ascent and subsequent mission phases. A disadvantage of this method is that the propellant volume may vary significantly with each mission, thus requiring a different capillary channel length for each flight. Shortening the channels would also reduce expulsion efficiency and increase tank residual quantities.

On-orbit roll and pitch maneuvers may be used to bathe and fill dry capillary channels with liquid. This operation is best suited for spherical or cylindrical tanks, where the entire tank wall-mounted CAS can be swept with fluid. It may be more difficult to wet the channels of a tank with an unusual geometry, such as the toroidal LH_2 scavenging tank, because of the inner and outer tank wall surfaces. Propellant may tend to orient toward the surface of the outer wall of the toroidal tank, leaving the inner surfaces dry.

One potential passive method of removing entrapped vapor is to pressurize the tank, which would instantaneously provide a liquid subcooled relative to the vapor. The vapor bubbles would collapse in time depending on the size of the bubbles, gravity level, and the thermal properties of the propellant. Further thermodynamic analysis beyond the scope of this study will be required to determine the vapor bubble collapse time for various conditions.

One other potential alternative is to use a bubble trap device similar to that used on the orbiter RCS propellant tanks. The capillary channels in the RCS tanks feed to a common manifold that consists of screened compartments that preferentially permit fluid to pass while retaining vapor due to the bubble pressure phenomenon. Although this device has been used only for storables and noncondensible pressurant, the technology is applicable to cryogens as well. Future analysis would be required to determine the feasibility of such a device for cryogenic transfer.

Based on present technology, the most practical method of eliminating trapped vapor in the CAS appears to be a combination of on-orbit roll and pitch maneuvers and pressurizing the tanks to allow vapor bubbles to collapse. Truncating the capillary channels so that they remain immersed in fluid during launch is not an acceptable solution due to the varying propellant requirements. State-of-the-art cryogenic bubble traps have not been developed at this time but may have an application in zero-g cryogenic transfer in the future.

4.3 SYSTEM SELECTION

This section presents the rationale utilized to make the system concept selections for both cryogenic and storable propellant systems based on the data generated during system trades and ground rules data and information obtained during the early stages of the study.

4.3.1 Cryogenic Propellant System Selection

The data presented in Section 4.2.3 shows that the ring tank concept provides the largest quantity of cryogenic propellants to orbit because the concept best utilizes available payload bay length for propellant storage even though it does have a higher dry weight than the other concepts. Therefore, the ring tank concept was selected for payload bay cryogenic storage.

In Section 4.2.1, transferring the propellants during ascent or after MECO was discussed. The most efficient method is to load propellant and tank set weight in the payload bay before lift-off to drive the excess MPS propellant to zero. This method eliminates the requirement for propellant transfer during ascent, which would necessitate making a practically impossible guarantee that sufficient flight performance reserve propellant remains in the ET before MECO. Loading the propellants on the ground also provides a better environment for accurate control of the loaded quantity. The reserve propellants that remain after MECO can be transferred through small lines to the payload bay tankage without the use of pumps by using the pressure source available in the ET ullage.

Propellant orientation for this transfer in both the MPS and payload bay tanks would be obtained by placing the ET/orbiter combination in a 2-deg/sec pitching rate. The only restriction on this transfer is that the transfer systems be large enough to make the transfer in a reasonable time (less than ten minutes) under the prevailing pressure differential (ET ullage pressure minus tank pressure). This line size was selected as part of the system point design, which will be documented in Section 4.4.

A lightweight spray foam insulation was selected for the cryogenic storage tanks. This type of insulation sufficiently limits the heat load to the propellant during propellant loading, ascent, and on orbit. The insulation thickness evaluation (Section 4.2.5) essentially resulted in the determination that thicker insulation yields lower heat loads. To keep the application problems minimal and the dimensions within reason, a 2-inch thickness was selected. A sun shade was also included in the baseline. The TVS analysis presented in Section 4.2.6 indicated that an internal compact heat exchanger and mixer, similar to the Centaur concept, will thermodynamically vent the heat load that results from the insulation thickness.

A CAS is also included in the baseline concept because of the requirement to transfer propellants to the Space Station/space depot, which requires feedout capability under zero-gravity forces. Including a system with this capability also gives the scavenging system greater operational flexibility.

4.3.2 Storable Propellant System Selection

The storable propellant payload bay tankage size evaluation presented in Section 4.2.3 indicated that an arrangement of existing tank designs could permit significant quantities of propellants to be delivered to orbit. An intricate design concept, such as the cryogenic ring tank, is not required for the storable propellants because the high density of both of the storable propellants makes existing tank designs and sizes usable.

The excess OMS propellant not required for any given mission remains in the orbiter OMS tanks and, therefore, can be transferred from the OMS tanks to the user directly. This concept eliminates any special propellant transfer or vehicle control requirements (such as those for cryogenic propellant transfer just after MECO) other than what will exist for the transfer to the user. Transfer from the payload bay tankage will be made to the user sequentially with the OMS transfer.

The requirement to transfer propellants to the user under zero-gravity forces when docked with the Space Station/space depot dictates the inclusion of a CAS in the payload bay tanks. The OMS tanks already contain screened compartments that control propellant orientation. The existing tank designs evaluated earlier do not have acquisition systems and must be modified.

4.4 SYSTEM DEFINITION

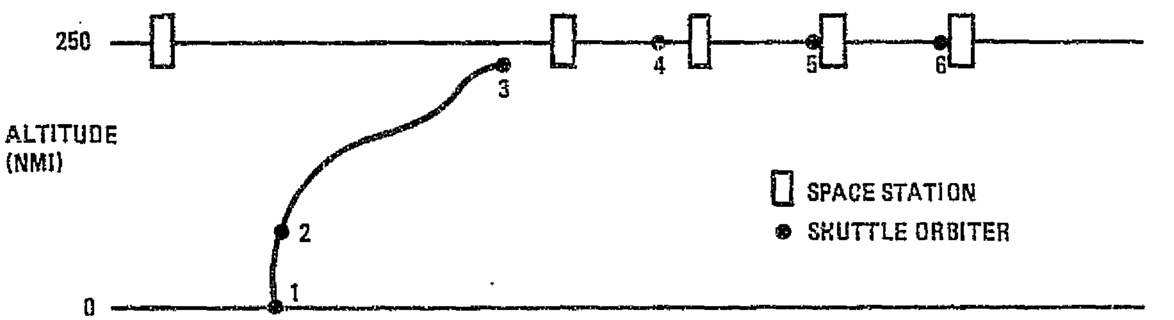
The information presented in this section details the point design of the selected cryogenic and storable propellant concepts. The first part details system operations and mission time lines. Next, the detailed hardware design and operating characteristics are presented. And finally, proposed test requirements and schedules are presented.

4.4.1 Mission Operations

4.4.1.1 Cryogenic Propellant Operations Mission Time Line. The ascent trajectory and on-orbit maneuvers of a scavenging mission establish the major events that set the framework for more detailed scavenging events and sequences. The purpose here is to define realistic and representative major event time lines in a way that will be adequate for the subsequent discussion of both cryogenic and storable propellant scavenging operations. It is most important that in doing so to demonstrate that the launch and orbital geometry of the combined Space Transportation System (STS) Space Station/OMV system will be compatible with the efficient delivery of propellants to the Space Station. The mission profiles should not entail excessive plane changes (performance penalty) or long phasing times during cryo propellant delivery (cryo losses).

The rapid delivery of cryo propellant to the Space Station has been assumed to be a requirement, as noted under the mission ground rules in Section 4.1.1. There appear to be no special time considerations for delivering storable propellant. Thus, for consistency, all propellant deliveries are assumed to precede other payload deployments.

The manifesting of payloads and the analysis of propellant availability in Section 4 leads to the use of just two alternative scavenging mission profiles. In the first profile (Figure 54), the Shuttle launches by direct insertion to the Space Station orbit (250 nautical miles, 28.5 degrees) and



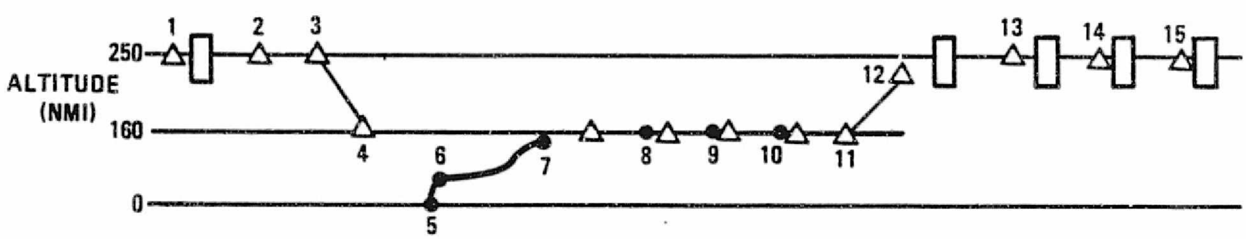
TIME (HR)	EVENT
0	1. SHUTTLE LAUNCH
0.14	2. MECO
0.78	3. OMS-2 COMPLETE, BEGIN ORBITER ACTIVE RENDEZVOUS
4.48	4. BEGIN MANUAL PHASE PROX OPS
7.08	5. DOCKING COMPLETE, BEGIN PROPELLANT TRANSFER
8.08	6. PROPELLANT TRANSFER COMPLETE, CONTINUE ORBITER MISSION

Figure 54. Time Line for Propellant Delivery by Direct Insertion to Space Station Orbit (First Profile)

immediately begins a rendezvous sequence. After docking and propellant transfer the orbiter continues its mission at the Space Station or elsewhere. The rendezvous time of 3.7 hours reflects the preferred orbiter active mode requiring 2-1/2 revolutions. The manual proximity operations time includes 45 minutes of station keeping to guarantee favorable lighting. Both rendezvous and proximity operation times are based on Rockwell Space Station support studies. The one hour allowed for propellant transfer is a maximum requirement established by this study.

The cryo propellant delivery time in the first profile is 8.1 hours from launch to completion of propellant transfer at the Space Station.

The second mission profile begins (Figure 55) with an OMV leaving its base at the Space Station and descending to a 160-nautical mile circular orbit, where it waits for the Shuttle. A standard insertion (two OMS burns) brings the Shuttle to the 160-nautical mile vicinity, and rendezvous begins at once with the Shuttle active again. An hour is allowed for propellant transfer to tanks on the " Another hour is provided for undocking and safe separation, after which the orbiter is free to continue its mission. The OMV ascends to the Space Station vicinity where it performs an OMV-active rendezvous requiring 1-1/2 revolutions (2.1 hours). The proximity operations and propellant transfer times at the Space Station are the same as in the first profile.



TIME (HR)	EVENT
−34.13	1. DEPLOY OMV FROM SPACE STATION
−33.13	2. OMV PLANE CHANGE
−32.70	3. OMV DEPART 250 NMI
−31.93	4. OMV ARRIVE 160 NMI
0.	5. SHUTTLE LAUNCH
0.14	6. MECO
0.75	7. OMS-2 COMPLETE, BEGIN ORBITER ACTIVE RENDEZVOUS
4.32	8. BEGIN MANUAL PHASE PROX OPS
6.92	9. DOCKING COMPLETE, BEGIN PROPELLANT TRANSFER
7.92	10. PROPELLANT TRANSFER COMPLETE, UNDOCK,AND CONTINUE ORBITER MISSION
8.92	11. OMV DEPART 160 NMI
9.69	12. OMV ARRIVE 250 NMI, BEGIN OMV ACTIVE RENDEZVOUS
11.82	13. BEGIN AUTOMATIC PROX OPS
14.42	14. DOCKING COMPLETE, BEGIN PROPELLANT TRANSFER
15.42	15. PROPELLANT TRANSFER COMPLETE

□ SPACE STATION
● SHUTTLE ORBITER
△ OMV

Figure 55. Time Line for Propellant Delivery by Standard Insertion to 160-nmi Orbit With OMV Transfer to 250 nmi (Second Profile)

The cryo propellant delivery time in the second profile includes 7.9 hours in the orbiter tanks and 7.5 hours in the OMV tanks.

The analysis of propellant availability in this study has assumed that any flight in the ten-year mission model is potentially able to deliver scavenged propellant to the Space Station without significant performance penalties. It remains to be determined whether the two scavenging mission profiles just described are compatible with this assumption. The key factor here is the adoption of a rendezvous-compatible (once-per-day) orbit for the Space Station. The advantages overall to the Space Station make this a reasonable assumption at this time. The altitude required (257 nautical miles) at a 28.5-degree inclination is close enough to the scavenging study guidelines to avoid any significant impact to the study conclusions regarding propellant availability and scavenging system sizing. Throughout this report and particularly in this section, the Space Station altitude continues to be shown as 250 nautical miles with the understanding that the orbit will be rendezvous compatible. The penalties associated with a nonrendezvous-compatible Space Station orbit, however, are briefly described below.

The first mission profile is readily accomplished on a once-a-day basis when the Space Station is in a rendezvous-compatible orbit. At the instant of launch, the launch site leads the Space Station orbital position by about 154 degrees and is ideally located with respect to the desired plane. No phasing maneuvers are required other than those inherent in the basic rendezvous profile.

In the second profile, two effects must be accommodated so the OMV can return quickly and efficiently to the Space Station. These are the differences in nodal regression rate and orbital angular rate between the Space Station orbit and the OMV orbit profile. The OMV (without payload) must make a plane change of 0.52 degree after leaving the Space Station to compensate for differential nodel regression. The maneuver is made at 90 degrees from the line of nodes and actually shifts the node eastward by about 1 degree without changing inclination. At Event 12, the OMV will be in the Space Station plane again. After transferring to the 160-nautical mile orbit, the OMV must wait 32 hours before the Shuttle launch. This delay achieves correct phasing for rapid return to the Space Station at Event 11. There is no performance penalty for waiting, but the plane change will require a delta-V of 230 fps, adding about 20 percent (12,000 pounds) to the ten-year OMV propellant requirements shown in Table 20.

A third concern in the second profile is the out-of-plane launch requirement for the Shuttle at Event 5. At this time in the mission the OMV trails the Space Station by 80 degrees. At 4 degrees per minute, the OMV will require 20 minutes to move into the correct orbital position for the Shuttle launch and rendezvous to begin. Launch window data for 28.5-degree orbits indicate less then 100 pounds yaw steering penalty for a 20-minute launch delay past the ideal coplanar opportunity, assuming that launch azimuth is properly chosen.

At this point it is appropriate to put the possibility of a nonrendezvous-compatible orbit for the Space Station into perspective. With orbit inclination so close to launch latitude, very long launch windows exist for moderate yaw steering penalties. This is the case particularly when the launch azimuth is freely optimized, as it would be for a planned launch time as opposed to an unplanned hold. The data cited above yield an 84-minute window for a 550-pound penalty on a direct insertion to 250 nautical miles. This window would cover 90 percent of all possible Space Station-to-launch site phasings. Furthermore, the yaw steering penalty can be reduced by trading against on-orbit phasing, where delivery time is not critical, as it is for storable propellant scavenging. Thus, while a rendezvous-compatible orbit assures day-by-day consistent performance, it is not a critical assumption to the validity of delivering scavenged propellants.

An unplanned launch hold of up to 30 minutes will likewise entail a penalty of 500 pounds or less for yaw steering into the correct 28.5-degree plane. However, the subsequent phasing time for a rendezvous with the Space Station or OMV will increase considerably. A typical phasing profile for ascent to a 257-nautical mile target adds approximately 15 hours for a 30-minute delay. Allowance for launch holds, thus, can have some influence on the design of insulation for cryogenic scavenging tanks.

Post-MECO scavenging operations are discussed elsewhere, but it should be noted here that they require an extended mated coast that will impact the OMS-1 burn in the second (standard insertion) profile. Available targeting and simulation data support the feasibility of delaying OMS-1 as much as 20 minutes with some small performance penalty. Constraints on targetting may account for most of the penalty, and these constraints may not apply to Shuttle operations in the 1990 era. A realistic treatment of this trajectory phase must await further studies.

The detailed sequences of propellant scavenging will be controlled by the on-board propellant scavenging microprocessor. Each of these sequences will be initiated either by the orbiter general-purpose computer (GPC) or the ground launch processing system (LPS). Table 24 lists the input commands.

Before it is installed in the orbiter payload bay, the scavenging pallet will be checked out (leak and functional) and purged by a special checkout console. This console will be capable of checking out the microprocessor and also providing individual component control through a special breakout connector located between the microprocessor and functional components. Also regulated helium and closeout plates will be provided for leak checking and purging. At the completion of checkout, both tanks will be purged with helium and maintained at a standby pressure of 20 psia until after installation in the payload bay.

After installation, all the final LO_2 and LH_2 connections will be leak-checked using closeout plates and drag-in helium in conjunction with existing orbiter and facility valves. At the completion of this leak check and with the orbiter powered up, the power to the scavenging pallet will be turned on, which will bring the scavenging microprocessor on line. At this time, the LPS can issue a standby command to the microprocessor, provided the MPS helium bottles are pressurized to 1,500 psia.

Table 24. Scavenging Microprocessor Inputs

Input Command	Origin
1. Standby	LPS through GPC, crew through GPC
2. Pretanking line purge	LPS through GPC
3. LO_2 ground fill	LPS through GPC
4. LH_2 ground fill	LPS through GPC
5. Stop fill	LPS through GPC
6. Ground LO_2 drain	LPS through GPC
7. Ground LH_2 drain	LPS through GPC
8. Launch preparation	LPS through GPC
9. LH_2 TVS on	Crew through GPC
10. On-orbit fill	Crew through GPC
11. LO_2 user transfer	Crew through GPC
12. LH_2 user transfer	Crew through GPC
13. Vacuum inerting	Crew through GPC
14. Abort dump	Crew through GPC
15. Terminate	Crew through GPC, LPS through GPC

In response to the standby command, the microprocessor will:

1. Open the primary side of the helium supply and initiate failure monitoring to determine that the helium supply is correct. If not, it will switch to the backup control leg.

2. Control the pressure in each scavenging tank to 20 ± 2 psia through the pressurization control valves.

3. Apply the closing commands to both solenoid valves for both the LO_2 and LH_2 dump valves and to the LH_2 TVS shutoff valve.

4. Remove commands from all the solenoid valves.

The above logic will remain in effect unless specifically overridden in the following sequences.

The pretanking line purge command will be issued at the completion of the MPS purge operation and will purge each of the scavenging lines by blowing helium from the tanks overboard by performing the following:

1. Open the LH_2 fill valve for two minutes. (The orbiter LH_2 relief shutoff valve [RSV] and LH_2 inboard and outboard fill valves must be opened to the facility vent at this time.)

2. Open the LH_2 dump valve for two minutes.

3. Open the LH_2 vent valve for two minutes.

4. Open the LH_2 TVS valves for two minutes.

5. Open the LO_2 fill valves for two minutes. (The Orbiter LO_2 RSV and inboard and outboard fill valves must be opened to the facility vent at this time.)

6. Open the LO_2 dump valve for two minutes.

7. Open the LO_2 vent valve for two minutes.

The LO_2 ground fill command will be issued by the LPS when the ET is in stable LO_2 replenish and will accomplish the following:

1. Terminate LO_2 standby pressure control.

2. Open the LO_2 vent valve.

3. Open the LO_2 high-resistance fill valve.

4. Monitor the LO_2 ullage pressure. If the pressure is $>$ 28 psia, fill valves will be closed; if $<$ 15 psia, the vent valve will be closed.

5. Monitor the LO_2 liquid level. When greater than 5 percent, it will open the low-resistance fill valve. When greater than 95 percent, it will close the low-resistance fill valve. When greater than 100 percent, it will close the high-resistance fill valve. The 100 percent indication will be mission dependent (mission initial-load) since each mission will have a different payload margin. When the liquid level drops below 98 percent, the high-resistance fill valve will be reopened and then closed again when the level is greater than 100%.

The LPS must also open the MPS LO_2 RSV at this time.

The LH_2 ground fill command will be issued by the LPS when the ET is in stable LH_2 replenish and will accomplish the following:

1. Terminate LH_2 standby pressure control.

2. Open the LH_2 vent valve.

3. Open the LH_2 fill valve.

4. Monitor the LH_2 ullage pressure. If > 28 psia, it will close the tank fill valve. If < 15 psia, the LPS will close the tank vent valve.

5. Monitor the LH_2 liquid level. If greater than the mission-dependent 100-percent level (mission initial load), the LPS will close the fill valve. If the level is less than 98 percent, it will open the fill valve.

The LPS must also open the MPS LH_2 RSV at this time.

The stop fill command will be issued by the LPS normally at T(0)-9 minutes and will accomplish the following:

1. Close the LO_2 fill valve.

2. Close the LH_2 fill valve.

At T(0)-9 minutes, the LPS will also issue the close commands to the MPS LO_2 and LH_2 RSV's. This command will also be issued by the LPS in conjunction with any ET loading stop flow or revert condition. To reestablish flow to either tank, the appropriate ground fill command must be reissued.

The ground LO_2 drain command will be issued by the LPS if the LO_2 tank must be drained on the ground. It will not be issued until after the ET LO_2 drain flow has been terminated. This command will accomplish the following:

1. Close the vent valve.

2. Open the fill valve.

3. Pressurize the LO_2 tank to 20 ± 2 psia.

The ground LH_2 drain command will be issued by the LPS if the LH_2 must be drained on the ground. It will not be issued until after the ET LH_2 drain flow has been terminated. This command will accomplish the following:

1. Close the vent valve.

2. Open the fill valve.

3. Pressurize the LH_2 tank to 20 ± 2 psia.

The LPS must also open the MPS LH_2 RSV, inboard and outboard fill valves, and appropriate facility valves.

The LPS must also open the MPS LO_2 RSV, inboard and outboard fill valves, and appropriate facility valves.

The launch prep command will be issued by the LPS at T(0)-2 minutes and will accomplish the following:

1. Close the LO_2 vent valve.

2. Close the LH_2 vent valve.

3. Terminate all ullage pressure control (close all pressurization solenoid valves).

The LH_2 TVS on command will be initiated by the GPC after SRB separation and will activate the TVS as follows:

1. Open the primary supply side shutoff valve.

2. Open the TVS shutoff valve.

3. Start both TVS motors.

4. Initiate TVS supply redundancy monitor logic. If the primary supply fails, it will activate the secondary side and terminate the primary side.

The on-orbit fill command will be initiated by the GPC after MECO and after the orbiter has initiated and attained a 1/3 rpm tumbling mode by pitching the nose up. The ET separation will be delayed until after this on-orbit fill operation is accomplished. The on-orbit fill command will accomplish the following:

1. Open the LO_2 fill valve.

2. Open the LH_2 fill valve.

3. Activate the LO_2 vent valve control. If the ullage pressure is > 18.7 psia, it will open the vent valve. If it is > 18.3 psia, the vent valve will be closed.

4. Activate the LH_2 vent valve control. If the ullage pressure is > 22.3 psia, it will open the vent valve. If it is < 21.9 psia, the vent valve will be closed.

5. Monitor the LH_2 fill bubble detector and LO_2 overfill sensor. If bubbles are detected or the tank is overfilled, the LO_2 fill operations will be terminated as follows:

   A. Close the LO_2 fill valve.

   B. Close the LO_2 vent valve.

6. Monitor the LH_2 fill bubble detector and LH_2 overfill sensor. If bubbles are detected or the tank is overfilled, the LH_2 fill operations will be terminated as follows:

   A. Close the LH_2 fill valve.

   B. Close the LH_2 vent valve.

7. When both the LO_2 and LH_2 fill are terminated, the microprocessor will inform the GPC.

After the on-orbit fill is complete, the orbiter will be stabilized and the ET separation performed. The orbiter will then perform the required maneuvers to mate with the propellant user. After mating with the propellant user, the transfer to the user will be initiated through the GPC.

Upon receipt of the LO_2 user transfer command from the GPC, the microprocessor will accomplish the following:

1. Open the user transfer valve.

2. Monitor the tank pressure. If the pressure is < 21.3 psia, the pressurization valves will be opened. If > 21.7 psia, it will close the pressurization valves.

3. Monitor the LO_2 bubble detector. Transfer is terminated as follows when bubbles are detected.

   A. Close the user transfer valve.

   B. Close the pressurization valves.

   C. Inform the orbiter that LO_2 transfer is complete.

Upon receipt of the LH_2 user transfer command from the GPC, the microprocessor will accomplish the following:

1. Open the user transfer valve.

2. Terminate TVS operation. Turn off both motors and close both TVS shutoff valves.

3. Monitor the tank pressure. The pressure is < 21.3 psia, the pressurization valves will be opened. If > 21.7 psia, it will close the pressurization valves.

4. Monitor the LH_2 bubble detector. Transfer is terminated as follows when bubbles are detected:

   A. Close the user transfer valve.

B. Close the pressurization valves.

C. Inform the orbiter that the LH_2 transfer is complete.

After the transfer of both propellants has been completed and the orbiter has been demated from the user, the GPS will issue the vacuum inerting command, which will accomplish the following:

1. Open the LO_2 user transfer valve.

2. Open the LO_2 vent valve.

3. Open the LO_2 dump valve.

4. Open the LH_2 user transfer valve.

5. Open the LH_2 vent valve.

6. Open the LH_2 dump valve.

7. Start a ten-minute timer. When this timer expires, the LO_2 and LH_2 user transfer, vent, and dump valves will be closed.

8. Inform the orbiter that vacuum inerting is complete.

The terminate sequence command can be issued through the GPC at any time during the mission to handle any off-nominal condition. When issued, this command will result in the microprocessor removing electrical power from all the pneumatic solenoids and motors, allowing all components to return to their normal positions.

The abort dump command will be issued by the GPC to dump all the propellants from the scavenging tanks prior to reentry. The initiation time will be consistent with orbiter abort requirements. Upon receipt of this command, the following will be accomplished:

1. Open the LO_2 dump valve.

2. Open the LH_2 dump valve.

3. Close the LO_2 vent valve.

4. Monitor LO_2 tank pressure. If the pressure is < 26.0 psia, the pressurization valve will be opened. If > 26.5 psia, the pressurization valves will be closed.

5. Turn off the TVS.

6. Close the LH_2 vent valve.

7. Monitor the LH_2 tank pressure. If the pressure is < 26.0 psia, the pressurization valves will be opened. If > 26.5 psia, the pressurization valves will be closed.

8. Start a 250-second timer. When this timer expires, the LH_2 and LH_2 pressurization valves will be closed.

9. Allow 30 more seconds for tank pressure to blow down and then close the LO_2 and LH_2 dump valves.

10. Return to configuration defined by standby.

Prior to any nonabort reentry, the GPC will issue the standby command, which has been defined earlier.

Since the scavenging tanks are in the standby mode, no additional activity is required after landing. The pallet can be removed and stored if it is not required for the next flight. If needed for the next flight, no activity is required unless a review of the flight data reveals an anomaly that requires corrective action.

4.4.1.2 Storable Propellant Operations Mission Time Line. Prior to launch, the payload bay tank system (PBTS) will be mounted at the aft end of the payload bay by keel and still trunnions while the orbiter is in the Orbiter Processing Facility (OPF). Pressurant and propellant loading of the PBTS could be performed either at the OPF or at the launch pad. Nominally, pressurant and propellant transfer from the PBTS and the OMS pods is performed as part of the on-orbit operations. Once the transfer is completed, the PBTS would be safed with no further action required prior to orbiter reentry.

Due to a constraint on orbiter landing weight, a means of dumping the PBTS propellants will be required during a return-to-launch-site (RTLS) abort. One method of dumping the PBTS propellants is to provide the orbiter with overboard dump ports similar to the dump ports proposed for use with an earlier OMS payload bay kit design. The second option is to burn the PBTS propellants through the OMS and RCS engines. Due to a software change, starting with STS-61B(30) there will be approximately 300 seconds available to burn the PBTS propellants during an RTLS. Some failure modes could leave as little as 100 seconds in which to burn the PBTS propellants. Burning the PBTS propellants instead of dumping them overboard would require increasing the feedline diameters from the OMS crossfeed lines to the PBTS from 1/2-inch to approximately 2-1/2 inches.

Overboard dump ports are the recommended means of dumping propellants from the PBTS during an RTLS. Future tanker concepts may also require the dumping of propellants during an RTLS, when it would be unfeasible or impossible to burn them through the OMS and RCS engines, requiring the overboard dump ports.

Following propellant transfer, it may be essential to purge the propellant transfer QD's prior to disengaging. This can be performed by sequentially opening each set of purge isolation valves and flowing gaseous helium through each of the propellant transfer QD's. The waste material from the purge can be expelled through the nonpropulsive vents or, if required, stored in the waste disposal tanks. If the propellant scavenging is from the OMS tank only and the PBTS is not present, a separate helium bottle purge source may be required.

Propellant loading of the PBTS could be performed at the OPF or at the launch pad. The disadvantage of loading the hypergolics and helium in the OPF before orbiter roll-out is that there is no capability of monitoring pressures and temperatures during the roll-out to the pad. This problem could be minimized by leaving the PBTS at pad pressure during the roll-out following loading at the OPF and then loading the helium bottles and pressurizing the propellant tanks of the PBTS at the launch pad.

Propellant loading of the PBTS tanks at the launch pad could be accomplished through the OMS crossfeed lines and the OMS pod scar plumbing to the 1307 bulkhead flange. Propellant would enter from the bottom of the tank (with the orbiter in the vertical position) and may require internal bleed and vent lines to accommodate the vertical orbiter load and drain if surface tension vane or capillary screen propellant acquisition device tanks are used. Ullage gas, which is replaced during propellant loading, is transferred back to the OMS propellant tanks, where it is then vented overboard. Propellants could be drained from the PBTS through the T-4 umbilical panel QD or back through the OMS crossfeed lines.

If propellant loading of the PBTS is performed in conjunction with OMS propellant loading, a fill method similar to that for the RCS propellant tanks could be used with the PBTS. This method uses the ullage vent tube inside the propellant tank as a fill/spill line during propellant loading, leaving a known ullage and propellant volume. Propellant flowing through the fill/spill line would be carried back to the OMS propellant tanks. Propellant flow through the fill/spill line would continue for a few minutes after the tank is full to assure complete loading of the PBTS propellant tanks. An accurate OMS propellant load would be determined by subtracting the propellant weight loaded in the PBTS from the total weight of propellant loaded, as determined from the ground support equipment (GSE) totalizer.

Helium loading and venting of the PBTS would be accomplished through the T-4 umbilical panel QD. The PBTS would be isolated from the OMS helium bottles by the OMS pod scar helium isolation valves during loading or venting.

Propellant and pressurant loading of the PBTS in conjunction with the OMS and RCS is recommended. This option is considered to be less hazardous than loading the PBTS at the OPF and has minimal impact on OMS and RCS loading.

The PBTS is capable of carrying approximately 15,500 pounds of transferable bipropellants. However, there may be many instances when a payload weight constraint would require the PBTS to be off-loaded. Off-loading could be performed in two ways. The first is to partially fill all six propellant tanks. The advantage of this is that any range of propellant loads may be loaded. On-orbit wetting of the capillary acquisition system (CAS) is the main disadvantage. Wetting of the CAS is required to assure bubble-free liquid transfer to the receiver. On-orbit, in a zero-gravity environment, propellant tank ullages settle to the center of the tanks; therefore, with proper CAS design, wetting of the CAS would not be a problem. A bubble trap in the CAS would also be required because on-orbit wetting of the CAS would trap gas bubbles in the CAS channels.

The second method of off-loading is to fill only two or four of the six PBTS propellant tanks full. The advantage of this is the assurance of a wetted CAS without a large amount of gas being trapped in the CAS. The disadvantage is that off-loading could only be performed to discrete load values corresponding to two or four fully loaded propellant tanks.

When the PBTS is not installed in the payload bay and no OMS scavenging is required, leakage protection will be required on the propellant lines leading from the crossfeed lines to the 1307 bulkhead flanges. When the aft propulsion system is operating in crossfeed or interconnect mode, the only inhibit to the payload bay is a cap on the 1307 bulkhead flange. The helium and ullage return lines have scar isolation valves for leak protection. A similar condition existed in the design of the OMS payload bay kit, which proposed a capped QD, giving two inhibits at the 1307 bulkhead when the payload bay kit was not installed. Two inhibits could be employed on the PBTS flanges by installing a dual seal flange cap. It may be desired to use a third inhibit in this location by adding an isolation valve between the crossfeed line and the 1307 bulkhead or by using a triple-seal flange cap.

Prior to launch, the OMS pod scar valves and the PBTS ASE propellant and pressurant valves will be closed. While on orbit these valves will remain closed except during the transfer operations. After the receiver docks to the Shuttle, the transfer QD's and electrical connectors are mated, and the propellant and pressurant transfers operations from the OMS or PBTS tanks can be initiated.

Pressurant will be transferred using a gas compressor after the pressurant tank has been equalized. Pressurant transfer should be performed before the propellant has been transferred if the propellant transfer is by receiver ullage vent or ullage recompression. In this way the pressurant load for the supply-to-receiver pressurant tank equalization transfer operation will be utilized best. This operation can be performed by opening a flow path from the OMS or PBTS pressurant supply to the receiver pressurant tank (only one leg of redundant flow paths). Following the pressurant tank equalization transfer the direct flow path will be closed and one leg of the redundant gas compressors will be opened. The gas compressor will be used to bring the receiver pressurant tank to its final operating pressure. Receiver pressurant tank pressure and temperature visibility are required to determine the pressurant quantity transferred and to protect against pressurant tank compression overheating.

The pressurant system in the PBTS will be used to pressurize the PBTS propellant transfer (except for receiver ullage return transfers), resupplying pressurant to the receiver, and for purging the propellant QD's following the propellant transfer operations. If gas compressors are used in the pressurant system, the helium bottle volume would need to be a 20 percent of the PBTS propellant volume to accommodate these three functions. If no gas compressors are used and pressurant resupply is accomplished only by tank equilization, the helium bottle volume, as a minimum, would need to be 50 percent of the PBTS propellant volume. The recommended structure contains a pressurant volume equal to 20.5 percent of the propellant volume.

Propellant transfer from the PBTS with receiver ullage return can be initiated by opening a flow path from the PBTS tank outlet to the receiver and back to the PBTS tank ullage and activating the transfer pump. This requires the opening of the PBTS propellant tank isolation valves, pump isolation valves, QD isolation valves, receiver isolation valves, and the purge/ullage return isolation valves (only one leg of redundant flow paths would be opened). Propellant transfer will be from one fuel and one oxidizer tank set at a time. When one tank set transfer has been completed, the respective individual tank isolation valves will be closed and the next tank set isolation valves would be opened. A similar procedure would be used for the third tank set. When propellant transfer from the PBTS has been completed, the PBTS tank isolation valves and purge/ullage return isolation valves will be closed in preparation for OMS propellant transfer.

Propellant transfer from the OMS tanks will be performed from one pod at a time. Initiating the OMS propellant transfer, with receiver ullage return, requires opening the OMS propellant tank isolation valves, crossfeed valves, and ullage return isolation valves (only one leg of redundant flow paths would be opened).

Following transfer, the OMS pod valves that were opened will be closed and the corresponding valves in the other OMS pod will be opened to initiate the propellant transfer. When the OMS propellant transfer has been completed, all OMS, PBTS, and receiver valves will be closed.

Propellant transfer may be performed by receiver ullage vent or ullage recompression rather than receiver ullage return. This would require a regulated pressurant supply to the OMS and PBTS propellant tanks during the propellant transfer operations. This regulated pressurant will be supplied to the propellant tanks by opening the helium isolation valves and vapor isolation valves of the OMS pod or PBTS during their respective propellant transfers. The OMS and PBTS ullage return isolation valves will remain closed during propellant transfer.

One concern following oxidizer transfer from the PBTS was whether the NO content of the oxidizer tank residuals was sufficient to prevent stress-corrosion damage of the propellant tanks. It has been found that stress-corrosion damage would not occur with nitrogen tetroxide containing at least 0.4 weight percent NO. Oxidizer loaded in the OMS and RCS propellant tanks must have a minimum NO content of 1.5 weight percent and a maximum of 3.0 weight percent. A computer program that tracks NO was used to predict the NO content in the PBTS propellant tanks following propellant transfer. The worst-case conditions assumed in the simulated propellant transfer were minimum NO content of oxidizer loaded into the PBTS, a tank residual of 1 pound, propellant temperature of 100°F, and uninterrupted propellant transfer from a full tank down to the tank residual. The results indicate that following propellant transfer the NO content would be above 0.4 percent and that the tanks would be in a safe condition. However, if the oxidizer tanks were subjected to any vent/repressurization cycles following propellant transfer, the NO content could drop below 0.4 percent. The NO content could be predicted for vent/repressurization cycles if tank residuals, vent duration, vent pressures, and other specifics were known.

During propellant transfer from the OMS tanks to the receiver, precautions must be taken to preclude the transfer of too much propellant from the OMS tanks. Each transfer line contains a set of two series-redundant flowmeters. The flowmeter concept has been proven and their inaccuracy dispersions will be accounted for in the propellant quantity available for transfer.

For propellant and pressurant transfer operations to occur, there are some requirements that the user/receiver must meet. First, the user must be designed with a QD interface panel common to that of the boom umbilical interface QD panel. The user must be designed with two grapple fixtures. The orbiter's remote manipulator system (RMS) must attach to one grapple fixture so that it can locate the user in the position required for docking. The second grapple fixture is used for docking to the umbilical boom and allowing the mating of the QD interface panels.

The user must provide liquid-free ullage for return to the scavenging system or for vent. To accomplish this, the user must have a propellant management device capable of positioning the ullage during propellant transfer. The user must be designed to allow the purging of the propellant transfer QD's prior to demating the QD panels.

Propellant scavenging with the PBTS lends itself to the possibility of feeding OMS or RCS engines from the PBTS during on-orbit operations. The proposed 1/2-inch lines from the PBTS to the OMS crossfeeds would be capable of supplying propellant to fire the four aft RCS vernier thrusters simultaneously. With a four-vernier thruster flow rate through this line section, a pressure drop of 3 to 5 psid could be expected. This 1/2-inch line section would not be capable of supplying propellants to the RCS primary thruster. The flow rate from one primary thruster would cause a pressure drop of 30 to 35 psid in this line section.

Six RCS primary thrusters could be fired simultaneously for on-orbit maneuvers. The propellant flow rate of six RCS primary thrusters is nearly equivalent to that of one OMS engine. A line diameter of 2 inches would be required for these flow rates with pressure drops of 7 to 10 psid to be expected. A line diameter of 2-1/2 inches would be required to support a two-OMS engine flow rate with a pressure drop of 10 to 13 psid expected.

4.4.2 Hardware Description

4.4.2.1 Cryogenic Propellant Hardware Description.

4.4.2.1.1 Detailed Schematic. The LO_2 and LH_2 cryogenic scavenging systems are shown schematically in Figures 1 and 2. The systems are similar for

LO_2 or LH_2 with the exception of the TVS, which is only used on the LH_2 side. Because of this similarity, the subsequent discussion will be common to both sides. The following philosophy was used in defining the system requirements.

1. The system must be safe after any single-point mechanical failure. Delivery of the propellant to the user need not be accomplished. However, the system must be capable of withstanding two mechanical failures and still not dump or vent propellant into the orbiter payload bay.

2. If an existing component with more capability than required has been found, it will be used rather than a simpler new component.

The 2-inch fill system tees into the existing orbiter MPS between the normally open MPS feed line RSV and the feed line relief valve. The LH_2 fill system incorporates a single normally closed, pneumatic-actuated shutoff valve near the scavenging tank outlet. The LO_2 fill system incorporates two normally closed, pneumatic-actuated shutoff valves in parallel near the scavenging tank outlet. These valves, one a low flow resistance valve and the other a high flow resistance valve, are required to allow ground loading of the LH_2 tank during ET replenishment, when the MPS feed manifold pressure is 70 psig. The high-resistance valve (5 lb/sec at $\Delta P = 70$ psi) will be used during the initial chill and fill of the tank and also for the final fill and replenishment. The ground fast fill and on-orbit fill of the tank will be accomplished through the low-resistance valve. This combination will have minimum impact on the existing ET LO_2 replenishment control system. Since the LH_2 manifold pressure during replenishment is only 4 psig, the use of a single fill valve is considered adequate.

Each of the fill valves is controlled by a single normally closed solenoid valve. The valves, in conjunction with the MPS RSV, which is commanded closed from lift-off to MECO, provide redundancy in isolating the feed system from the scavenging tank. There is also a bubble detector located in the fill line near the scavenging tank that will be used to terminate the on-orbit fill of the scavenging tank when bubbles are present in the fill line.

To ensure that there is no propellant remaining in the scavenging system during reentry on any mission, including any abort cases, a 3.5-inch dump system has been provided. To be compatible with the worst-case dump requirements, the system has been sized to dump the entire system within 250 seconds (Centaur requirement). To prevent inadvertent opening and to ensure opening when required, the dump system utilizes a single valve with parallel pneumatic actuators. The pneumatic control system to the valve will utilize two normally closed closing solenoid valves in series and two normally closed opening solenoids in parallel to ensure proper redundancy (system can be dumped with a single failure). The propellant will be dumped overboard through the Centaur dump locations.

The 2-inch user transfer system contains a single normally closed pneumatic shutoff valve, which is controlled by a single normally closed opening solenoid valve, a flowmeter, and a dual-poppet disconnect valve, where both poppets open independently when mated with the user. The bubble detector located in the tank fill and drain line will be used to terminate flow to the user at the onset of bubble formation.

The vent and relief system incorporates a relief capability to protect the tank structure from overpressure and a vent that is normally closed to permit tank loading, vacuum inerting, and purging. These capabilities can be accomplished by a single integral component or by two separate components. The relief pressure setting will be defined to minimize tank structural weight. The vent will use a 2-inch or greater flow diameter to minimize tank backpressure during loading and will be controlled by a single normally closed solenoid valve. The LO_2 system will be vented overboard to the atmosphere through the existing Centaur LO_2 vent/dump port. The LH_2 system will be vented overboard to the ground vent system through the existing Centaur LH_2 vent disconnect. This disconnect will be open during flight; however, the system will be restricted from venting below 100,000 feet.

The pressurization system will route helium from the 750-psia manifold of the helium supply system to each tank through two parallel normally closed two-way solenoid valves. Each leg will be orificed to reduce the amount of valve cycling since a bang-bang control system will be used to control the ullage pressure in the required band. A check valve will be used to keep the propellant out of the helium system components. The pressurization system will be used to pressurize the tank during purging and standby, user transfer, abort dumps, and reentry.

The ullage pressure measuring system will consist of three 10- to 30-psia pressure transducers that will be used to control the tank pressure. The microprocessor that will be used to control the scavenging system will use a two-of-three voting scheme to maintain the required ullage pressure through either pressurization or venting control.

Since the amount of propellant that can be loaded before lift-off is mission-dependent, a continuous liquid level measurement system will be used. This system will consist of a capacitance probe with a discrete sensor at the low end to allow for cap probe calibration. In addition, another discrete sensor will be located near the top of the scavenging tanks to prevent overfilling the tanks during ground or on-orbit filling.

To allow for transfer to the user under zero-g conditions (Space Station), a passive CAS will be used. This system will prevent any gas from entering the user transfer line until almost all of the liquid is transferred out of the scavenging tank.

Because of the scavenging tank configuration a cylindrical LO_2 tank surrounded by a toroidal LH_2 tank), the TVS will be installed only on the LH_2 tank since the heat leak into the LO_2 tank is insignificant, if not negative. Additionally, since the scavenging tanks are not vacuum-jacketed, the size of an external heat exchanger to intercept the heat coming into the tank is prohibitive. Therefore, an internal heat exchanger with a blower for forced convection will be used to take the incoming heat out of the bulk temperature and discharge it overboard (Centaur G and G-prime concept). The

TVS will consist of a parallel redundant supply side that incorporates a normally closed shutoff valve (pneumatically controlled by a normally closed solenoid valve) and a regulator that maintains an outlet pressure of 5 psia. The supply system redundancy will be managed by the microprocessor, which will switch to the secondary path if the primary path has a malfunction. The heat exchanger portion will be common to both supply paths. The blower will consist of a single compressor powered by parallel redundant motors that will operate at the same time. Downstream of the blower is a flow control nozzle that will maintain the system flow at a nominal 0.6 lb/min and a normally open, pneumatically closed shutoff valve that will provide redundancy in ensuring no H_2 venting during early ascent (100,000 feet) and to prevent contaminants from entering the system during reentry.

The helium supply system will supply and regulate all the helium required for the scavenging system, LO_2 tank pressurization, LH_2 tank pressurization, and pneumatic control for all valves. The system is shown schematically in Figure 56 and will use existing MPS supply system components. The helium system will not be located on the scavenging pallet but will be mounted in the orbiter mid body under the payload liner. The 17.3-cubic-foot bottle will be filled to 4,300 psia before lift-off. It will tee into the existing MPS helium fill line and will be isolated from the MPS by a fill check valve. The helium will be regulated to 750 psia by two parallel redundant paths, each of which contains a normally closed solenoid valve, high-flow regulator, 800-psia relief valve, and check valve. The system redundancy will be managed by the microprocessor to ensure that no single failure will deplete the system or cause supply helium system to fail when it is required.

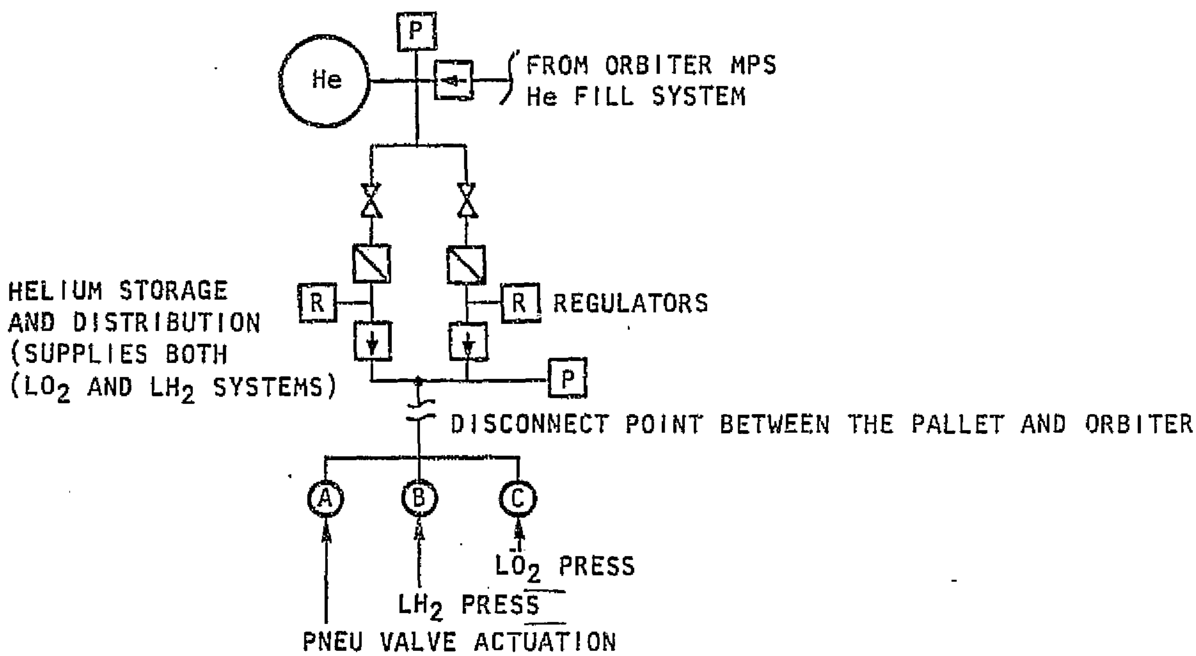


Figure 56. Cryogenic Scavenging System Helium System Schematic

4.4.2.1.2 Layout Drawings and System Weight. The cryogenic tankage and pallet structure is presented in Figures 57, 58, and 59. The figures show the proposed plumbing arrangement for the fill lines, the dump system, the vent system, and the transfer lines to the user interface, including the deployable docking boom mounted on the side of the pallet. The line routing from the MPS interface on the LO_2 and LH_2 fill and drain valves is shown in Figure 60. The lines in this figure, from the fill and drain valves to the 1307 bulkhead, are the only scar weight items that remain on the orbiter for nonscavenging missions. The scar weight is estimated to be less than 80 pounds.

The structural load-transmitting paths for the aft LO_2 cylindrical dome and the aft LH_2 ring tank dome are through struts between the domes and the aft pallet structure. The struts are compressively stiff to resist ascent loads and allow movement for cryogenic contraction.

The forward LO_2 cylindrical dome and the forward LH_2 ring tank dome attach to struts that allow thermal changes between the pallet's forward and aft support structures, which are in fixed planes rigidly attached to side beams.

The pallet structure is designed so that the aft portion is stronger than the forward portion to accommodate ascent loads. The side beams joining the forward and aft portions attach to orbiter longerons and dump the fore and aft loads through the trunnions and deployable latches. The keel trunnion also takes fore and aft loads but mainly accommodates side loads by A frames bolted to the trusses that terminate at a fore and aft beam for mounting the keel trunnion. The side beam trunnions accept the balancing moments because all the side loads cannot be taken out at the keel trunnion due to the moment arm to the payload c.g. The side and keel beams will be capable of disassembly at the bolted joints for individual tank assembly. An internal tank structure will be required for sufficient rigidity to impart loads to the localized support fittings.

In order to reduce weight significantly, it was decided to propose the use of the following state-of-the-art materials that would be available 1989-90:

- Aluminum copper lithium (Al-Cu-Li) for the cryogenic tankage application. Alcoa is developing lithium-bearing aluminum alloys that are expected to reduce weight from 10 to 15 percent and be commercially available in 1985. Alcoa is also planning to conduct extensive cryogenic testing of Al-Cu-Li alloys and is predicting satisfactory weldability.

- Improved graphite polyimide composites for the scavenging system support structure (pallet). By using improved graphite polyimide composites in place of the conventional aluminum alloys, a weight reduction of 30 to 35 percent can be achieved. A material temperature range of -250°F to 600°F is attainable.

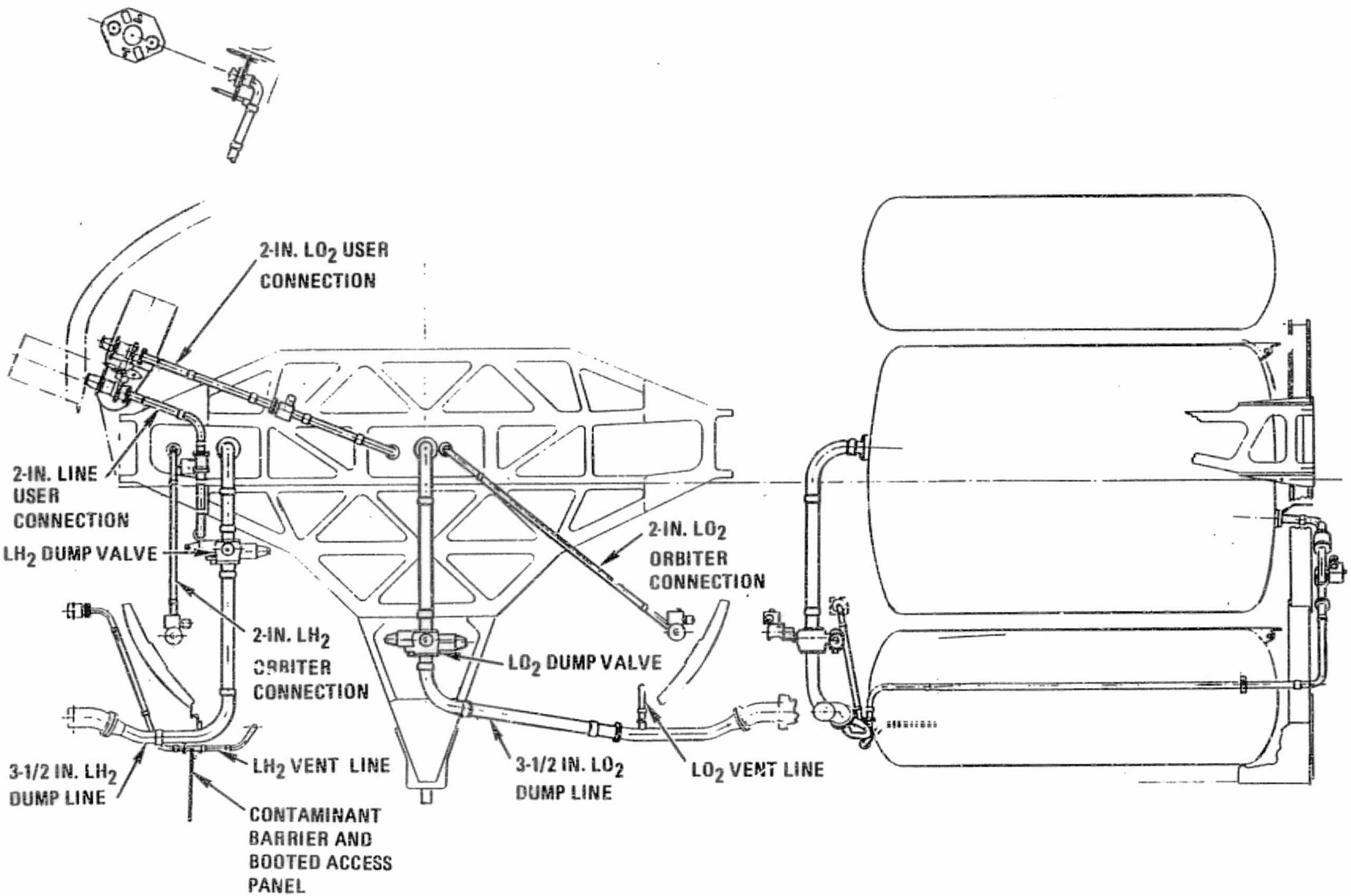


Figure 57. Fill and Dump Lines

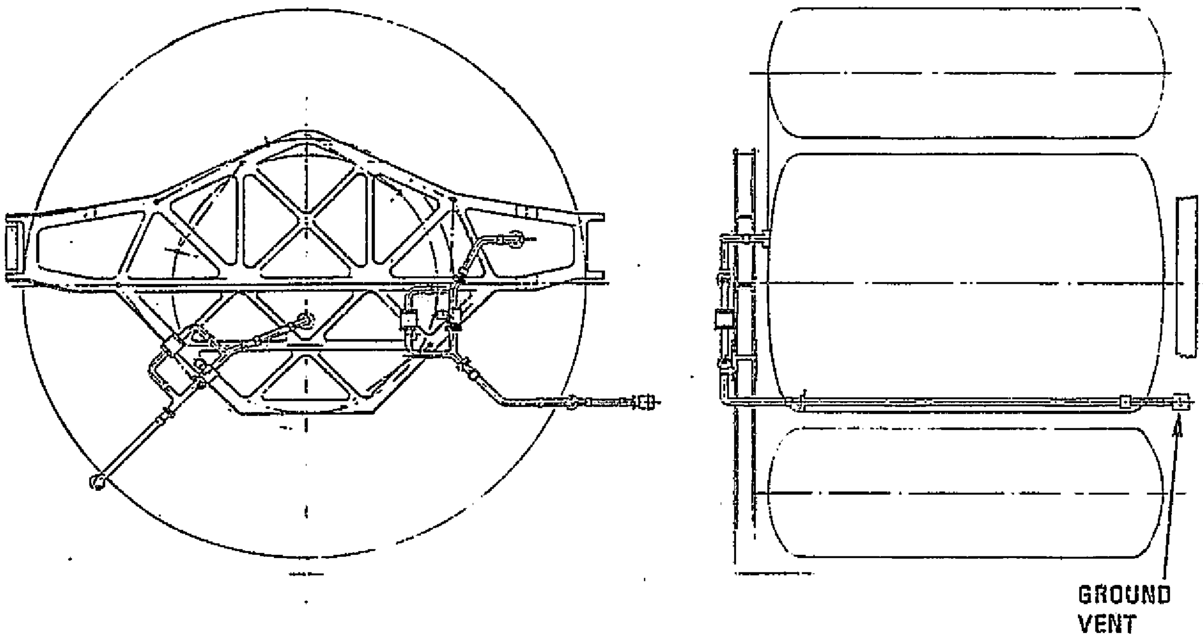


Figure 58. Vent Lines to Interface

The helium supply system consists of one filament-wound 17.3-cubic-foot helium tank loaded to 4,400 ± 50 psia (existing orbiter design) with associated valving and supply lines. The tank will be located in the mid body between X_o 919 and X_o 980 on the right side of the vehicle and below the payload bay liner.

An RMS standard end effector on the left side of the orbiter and an RMS standard grapple fixture on the user vehicle will be used for orbiter to user docking. The orbiter effector is stored in a vertical (up) position to accommodate the payload bay door while in the closed position and then rotated 90 degrees (outboard) after the payload bay door has been opened to accept the user vehicle grapple fixture for docking with the orbiter.

4.4.2.1.3 System Performance Characteristics. The scavenged propellants will be loaded in the orbiter tanks on the ground before lift-off. Propellant conditioning of the LO_2 and LH_2 will be performed by venting the tank pressure down to 15 psia to ensure the presence of cold propellants. After MECO, the mated ET/orbiter are put into a 2 deg/sec pitching rate creating a 10^{-4} g acceleration level, settling the liquid at the feed line inlets and allowing the propellant reserves remaining in the ET/orbiter to be transferred to the orbiter tanks. The time requirement to transfer the maximum available reserves of 5,789 pounds is ten minutes or less. The available reserves used for this analysis differ from the 5,690-pound actual value. In determining line sizes, this difference is insignificant. The propellant transfer maneuver will not be required on flights where the orbiter tanks can be loaded full on the ground. Following ET separation the orbiter circularizes, docks, and transfers the propellant to the user in zero g.

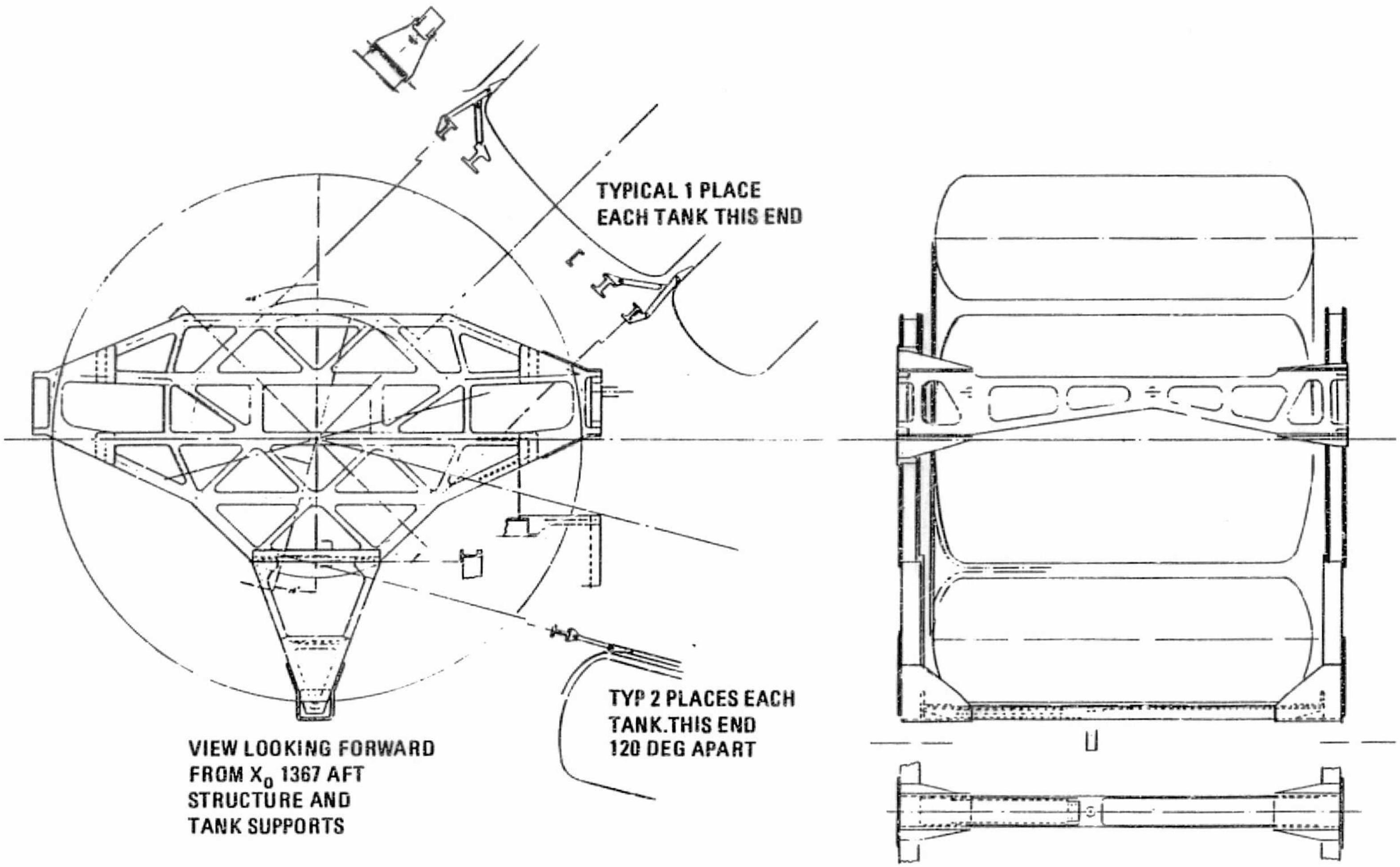


Figure 59. Pallet Structure

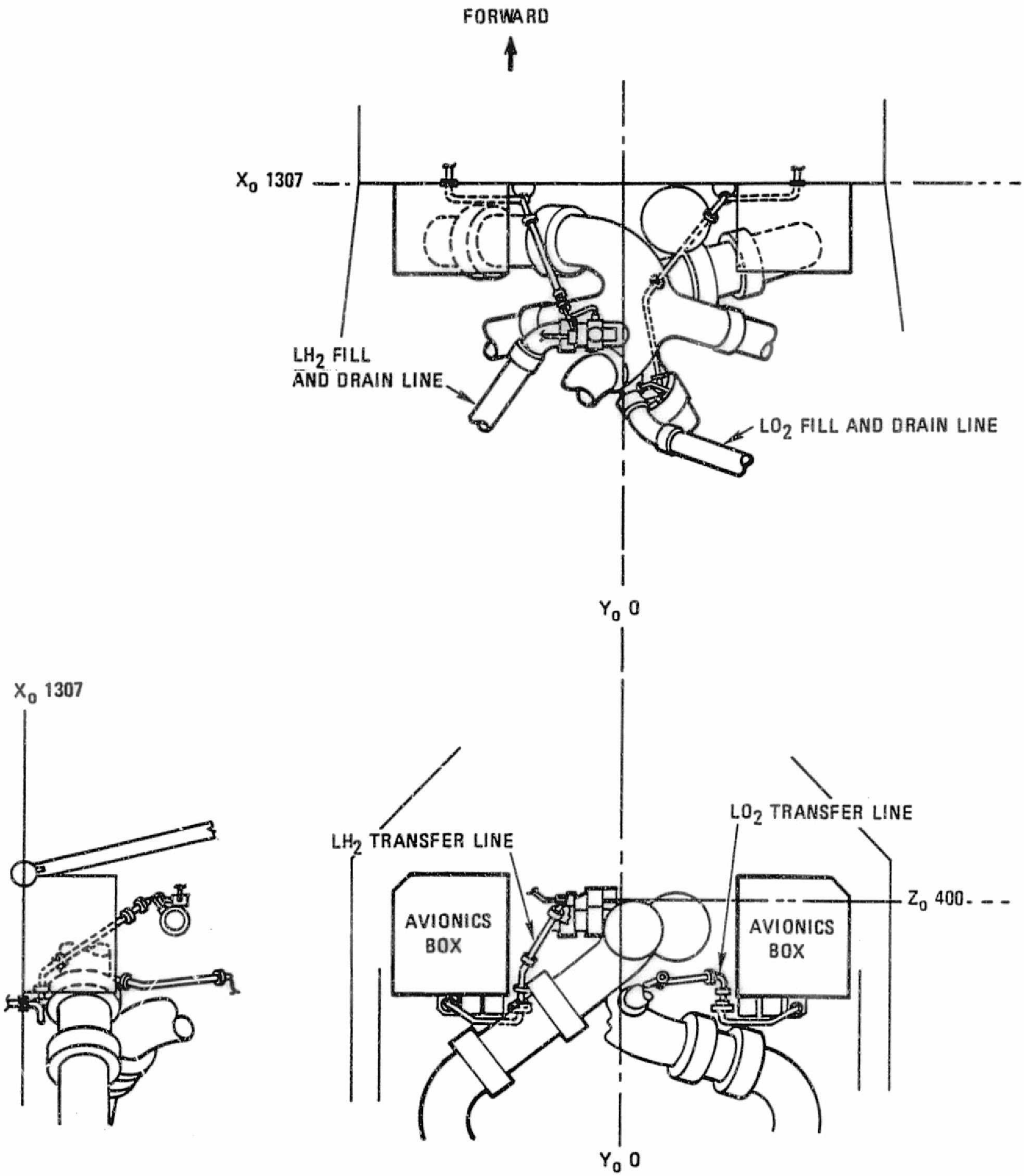


Figure 60. Line Routing From MPS Interface

This section discusses the fluid and thermodynamic analysis completed in support of the performance definition of each phase of the propellant transfer operating mode. The areas evaluated include propellant transfer from the ET to the orbiter tanks, propellant transfer from the orbiter to the user, and propellant dump in the event of a mission abort.

Propellant is transferred from the ET to orbiter tanks with propellant location control provided by the RCS. The transfer is performed using the available ET tank pressure level as the driving force. The ET pressure level at the end of SSME shutdown is approximately 20 psia in the LO_2 tank and 32 psia in the LH_2 tank. The pressure level in the orbiter tanks during transfer will be controlled by venting the ullage gas overboard. A vent pressure level will be selected high enough to keep the propellants subcooled but low enough to permit quick transfer. The vent pressure level is therefore a function of the vapor pressure or temperature of the propellants in the ET.

The propellants in the ET are heated by ground hold and boost heating causing the liquids to warm and stratify within the tanks. To predict the propellant stratification, a thermophysical model was developed at Rockwell using data derived from main propulsion test (MPT) and flight testing. As shown in Figures 61 and 62, the propellant stratification model and flight data from STS-41C (13) agree well. STS-41C data is shown in comparison because it had the lowest propellant residuals to date (3,000 pounds of LO_2 and 3,600 pounds of LH_2) and, therefore, the highest degree of thermal stratification. The propellant stratification model used to predict an increase in propellant temperatures during transfer indicated that the orbiter tank pressure levels must be maintained above 18 psia for the LO_2 and 26.6 psia for LH_2 so that vapor formation is avoided in the feed lines.

The disconnect engagement requires two pneumatically actuated hooks to pull/engage the user vehicle disconnect halves into the orbiter disconnect halves. During the engagement sequence three springs located 120-degrees apart are compressed. For user vehicle/orbiter LH_2 and LH_2 disconnect disengagement/undocking, the two engagement hooks are pneumatically actuated to release the user vehicle disconnect halves, and the spring force moves (undocks) the user vehicle away from the orbiter fluid transfer interface.

A summary of system weights for various lengths and associated propellant capacities of the ring tank pallet structure is presented in Table 25. The scar weight of 80 pounds is included in the component weight.

With the pressure levels known in the ET and the vent pressure levels established in the orbiter tanks, a sensitivity analysis was performed to determine the effect of feed line diameter on flow rate and transfer time. In addition to the transfer time goal, the unique inverted feed line design of the LH_2 ET also imposed a requirement on the transfer flow rate. Because fluid transfer is performed at low g, vapor ingestion can occur when residuals are low and transfer rates are high. An analytical model of vapor ingestion during low-g feedout was developed for the LH_2 tank from subscaled tests in order to analyze vapor ingestion during SSME shutdown. The analytical model defining the incipient vapor ingestion is presented in Figure 63. This model was incorporated into the feed line sensitivity analysis to identify operating regions that may be restrictive.

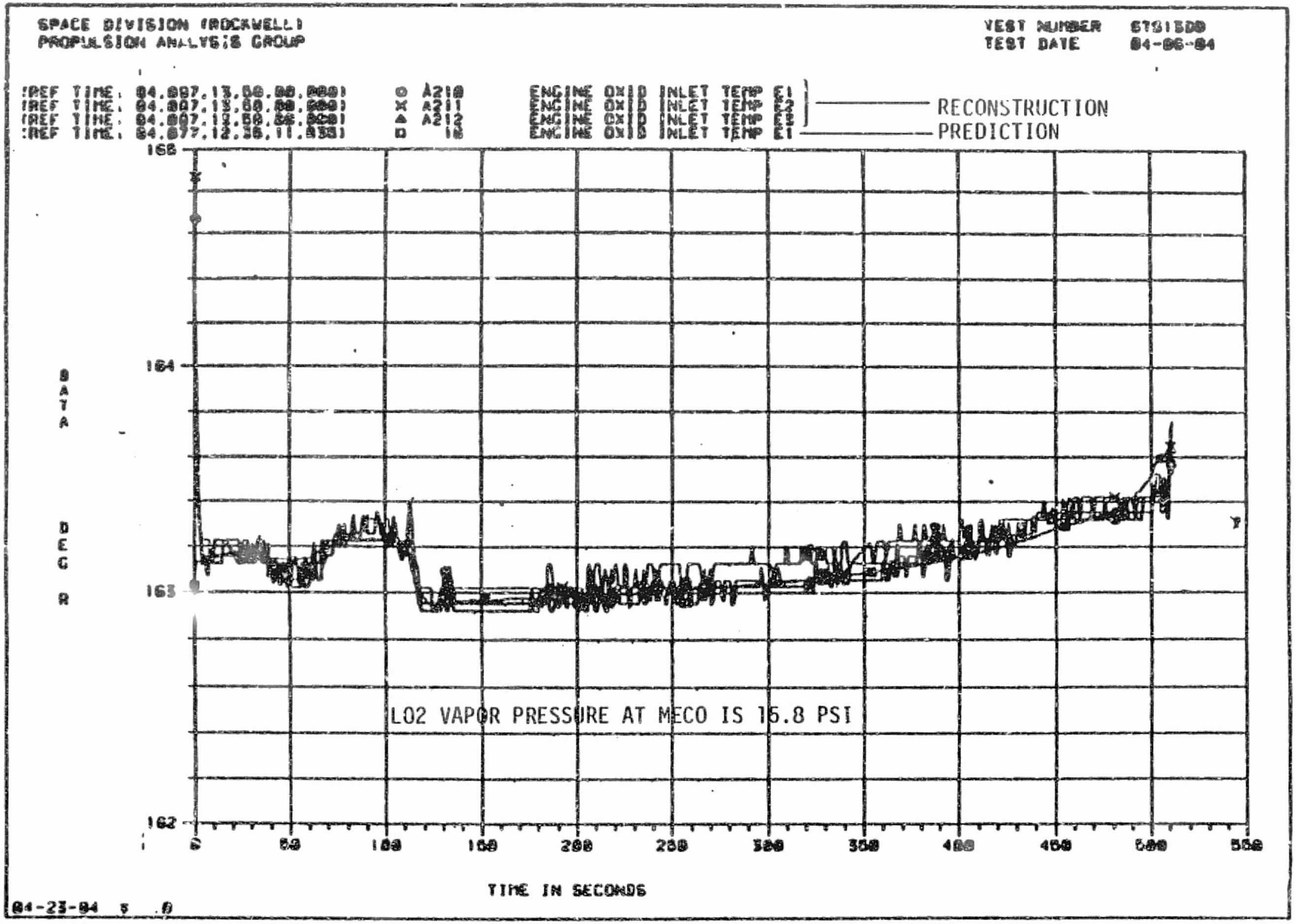


Figure 61. LO_2 Propellant Stratification Comparison Between Predicted and Measured (STS-41C)

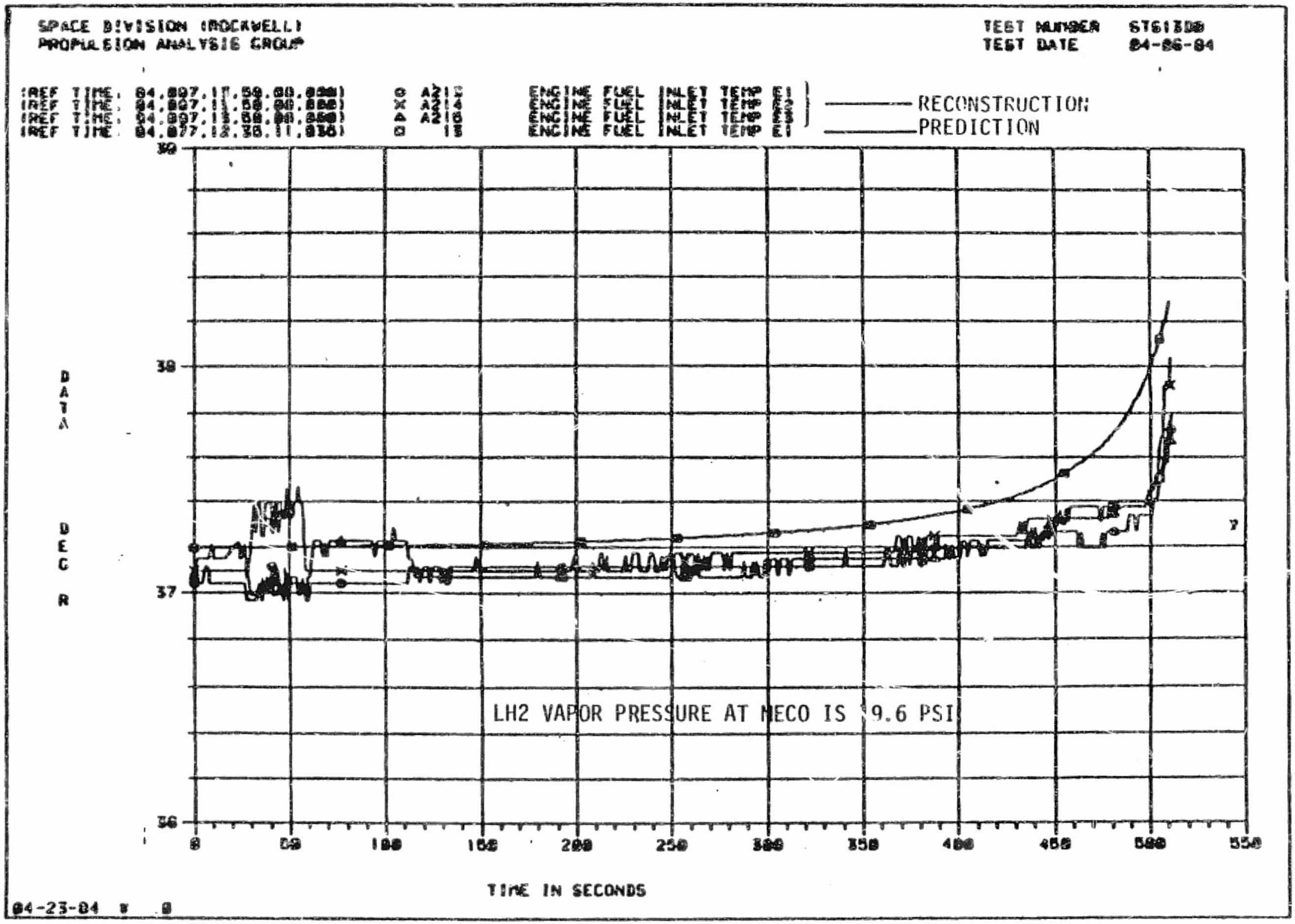


Figure 62. LH_2 Propellant Stratification Comparison Between Predicted and Measured (STS-41C)

Table 25. Cryogenic Propellant Hardware System Weights

Payload Bay Length (in.)	Propellant System	Propellant Capacity (lb)	Tank Weight (lb)	Pallet Weight (lb)	Component Weight (lb)	Helium System Weight (lb)	Insulation Weight (lb)	Total Hardware Weight (lb)
76	LH_2	2,016	342	1,170	560	356	297	2,956
	LO_2	11,990	145				85	
96	LH_2	2,740	408	1,300	604	356	354	3,303
	LO_2	16,483	178				103	
116	LH_2	3,464	474	1,463	626	356	411	3,660
	LO_2	20,976	209				121	
136	LH_2	4,188	539	1,625	648	356	468	4,017
	LO_2	25,518	241				140	

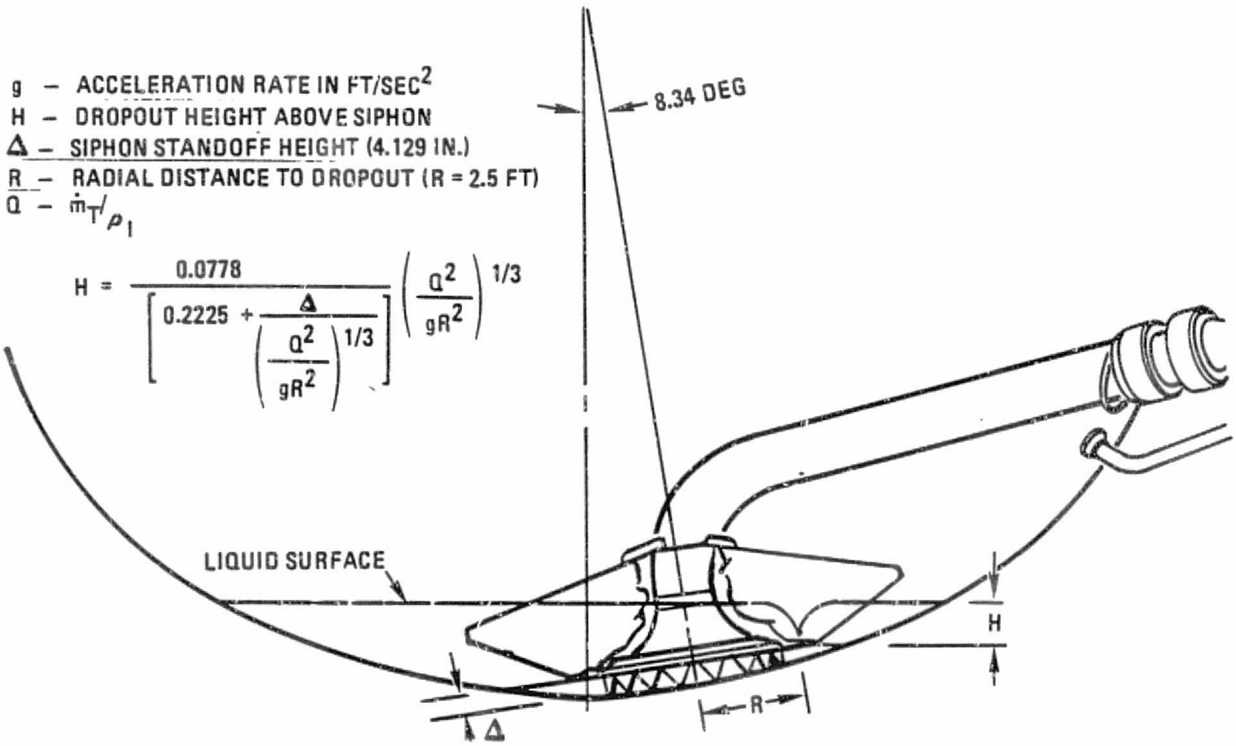


Figure 63. LH_2 Drop-Out Model

The results of the orbiter transfer line sensitivity analysis are presented in Figure 64 which shows the relationship between feed line diameter and transfer time. The example case evaluated corresponds to a mission where the maximum propellant quantity is transferred from the ET (3,517 pounds of LO_2 and 2,272 pounds of LH_2) to a partially loaded orbiter tank (22,197 pounds of LO_2 and 2,014 pounds of LH_2) with an assumed total tank capacity of 30,000 pounds. The analysis assumed that 7,000 pounds of LO_2 and 3,000 pounds of LH_2, which are the nominal reserves, remained in the ET/orbiter at the start of transfer including the +3 s performance allocations.

As can be seen, transfer line diameters greater than 1.5 inches will satisfy the ten-minute transfer requirement. A 2-inch line size was selected, however, in order to shorten the maximum transfer time and because the larger line diameter improves abort dump system performance, which is discussed later in this section. The propellant transfer performance characteristics for the baseline design of 2 inches are presented in Figures A-1 through A-6 in the appendix for the LO_2 system and Figures A-7 through A-13 for the LH_2 system.

The computer simulations show that ullage pressure in the ET decays only slightly as a result of liquid expulsion and low-g heat transfer from the ullage gas to the liquid surface. The orbiter storage tank pressure levels are controlled at 18.5 ± 0.5 psia on the LO_2 side and at 22.1 ± 0.5 psia on the LH_2 side by venting overboard. During the transfer process 9.6 pounds of GO_2 and 31.8 pounds of GH_2 are vented. The time required to transfer 3,517 pounds of LO_2 is 3.7 minutes and 3.6 minutes to transfer 2,272 pounds of LH_2.

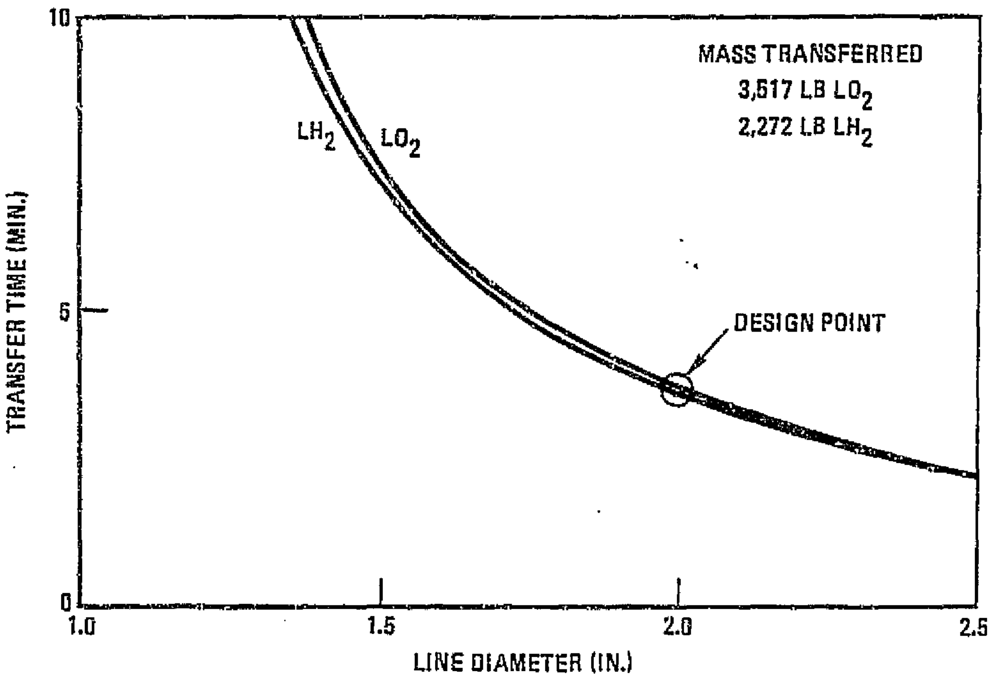


Figure 64. Orbiter Transfer Line Sizing From MPS Feed Line to Orbiter Tank

For the baseline simulation, vapor ingestion into the LH_2 syphon occurred 11 seconds before transfer was completed. Because hydrogen vapor ingestion occurs so late in the transfer and the bubble travels slowly (1.4 fps), the ingested vapor does not reach the orbiter tanks but remains in the 17-inch feed line near the ET/orbiter disconnect. It is very likely that the ingested GH_2 condenses out when it mixes with the surrounding colder LH_2 in the feed line. Since the orbiter tank pressures are maintained above the incoming fluid vapor pressure no other source of vapors are expected in the transfer line. Figures A-6 and A-12 in the appendix show the temperature of the incoming propellant and the maximum allowable temperature above which vapor formation can begin.

Because the orbiter tanks were partially full of much colder propellants at the start of the transfer process, the incoming propellants mix with the colder propellants in the tank, slightly elevating the bulk temperatures. For the oxygen and hydrogen, the increase due to bulk mixing is estimated to be less than 0.2°R and 1.0°R, respectively. The increase in the temperature of the LO_2 propellant due to the mixing of scavenged propellants and heat leakage is considered to be small. The TVS will maintain the temperature of the LH_2 below 37.7°R (18.0 psia vapor pressure) before it is transferred to the user.

Propellant will be transferred from orbiter storage tanks to the space-based user in zero g by helium pressurization of the storage tanks and venting of the user tanks. It is assumed that the user tank has the capability of venting gas in the zero-g environment. The vent pressure level of the user tank must be maintained above the incoming vapor pressure of the LO_2 and LH_2 so the transfer fluid does not vaporize. The LO_2 and LH_2 vapor pressures in the orbiter tanks will be less than 18 psia at the time of transfer as a result of heat absorption in the LO_2 tank and TVS operation in the LH_2 tank.

The propellant transfer process was simulated for a range of transfer line diameters in order to define the propellant transfer time, helium pressurant requirement, and propellants trapped in the orbiter tank CAS. To minimize the helium required, it was assumed that the orbiter tank pressure was controlled at 5 psi above the nominal user tank pressure. The sensitivity analysis also assumed (as an example) that a total of 30,000 pounds of LO_2 and LH_2 are transferred to a set of empty tanks with the same capacity but containing a small quantity of cryogenic liquid so the tanks and lines are at cryogenic temperatures.

The results of the line diameter sensitivity analysis are summarized in Figures 65, 66, and 67, showing the transfer times, helium pressurant required, and average transfer flow rates, respectively. to accomplish the transfer, a 17.4-cubic-foot helium bottle containing approximately 40 pounds of usable helium is required. This helium requirement can be satisfied by a 2-inch transfer line size, with approximately a 5-pound helium margin remaining.

The helium pressurant requirement changes with line size even though the orbiter tanks are pressurized to the same level because of the increased heat loss from the ullage gas to the liquid surface for the longer transfer times or smaller line diameters. For the 2-inch line, the transfer time is estimated to be 24 minutes for LO_2 and 16 minutes for LH_2.

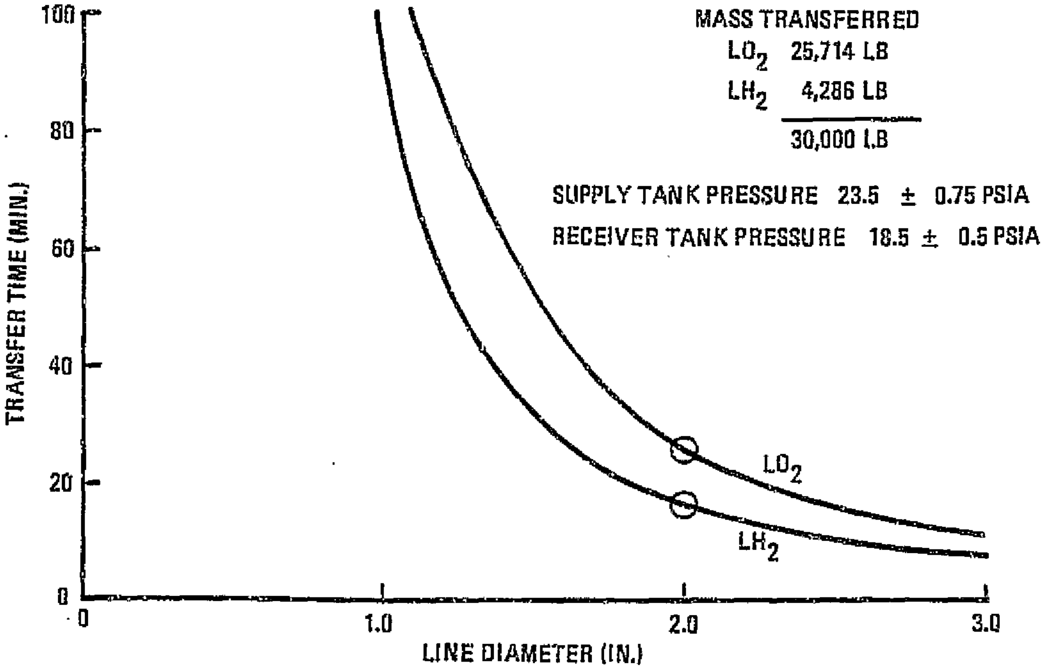


Figure 65. Transfer Line Sizing From Orbiter to User Tank

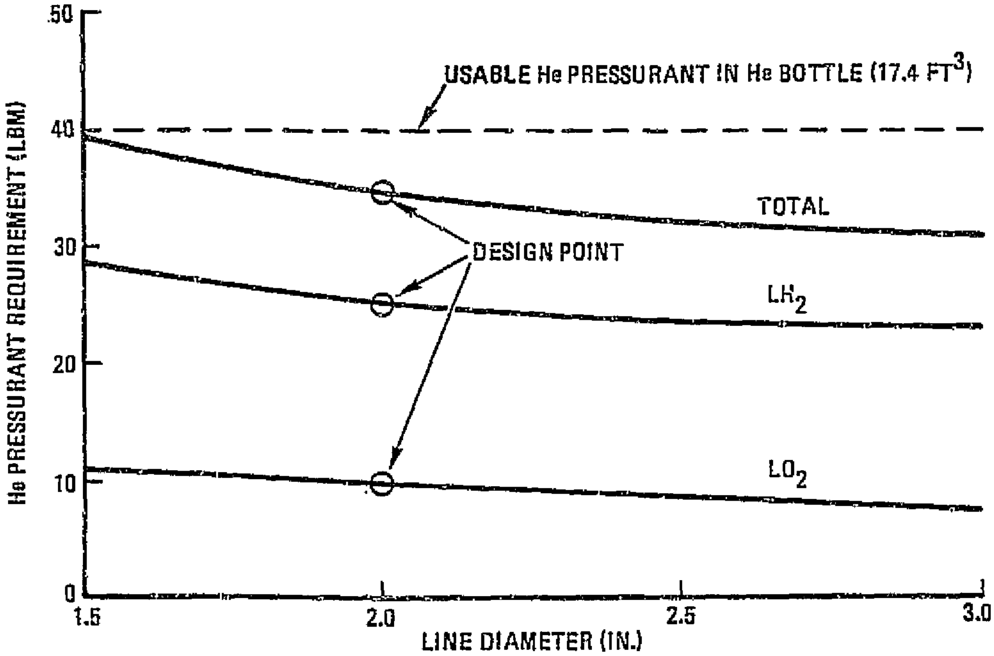


Figure 66. Helium Requirement as Function of Transfer Line Diameter From Orbiter to User Tank

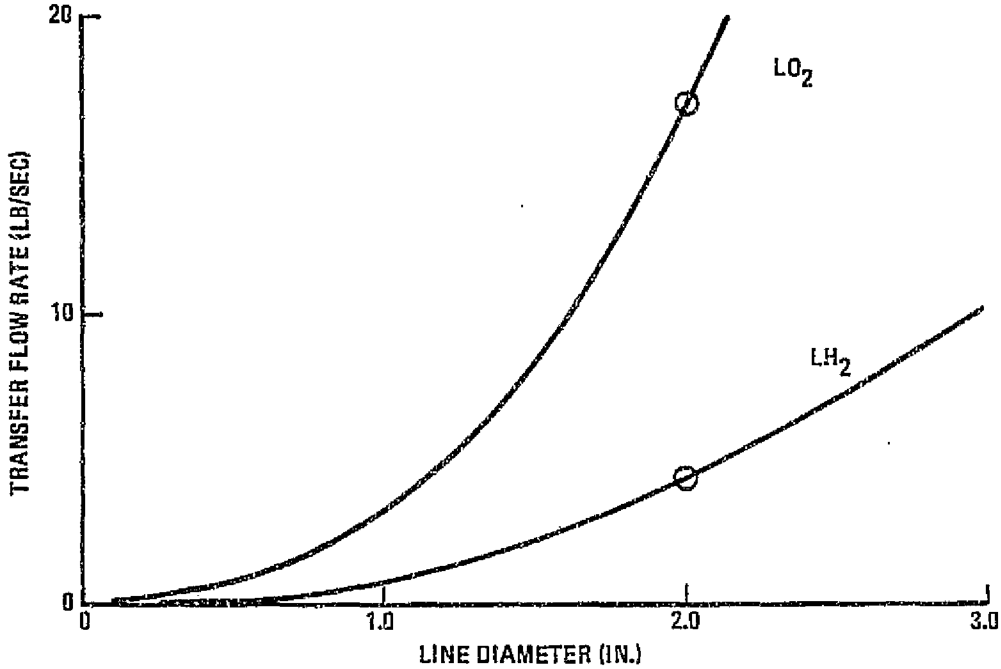


Figure 67. Transfer Line Sizing From Orbiter to User Tank

The propellant transfer performance characteristics for the baseline design of 2 inches are presented in the appendix in Figures A-14 through A-20 for the LO_2 system and Figures A-21 through A-27 for the LH_2 system. The computer simulation shows the ullage pressure, temperature, transfer flow rate, liquid mass quantity, helium pressurant flow rate, and vent flow rate histories in the storage and receiver tanks. The simulation results indicate that approximately 9.8 pounds and 25.2 pounds of helium pressurant are required to transfer 25,714 pounds of LO_2 and 4,286 pounds of LH_2, respectively. During the transfer 80 pounds of GO_2 and 27 pounds of GH_2 were vented. Propellant transfer will be terminated by bubble detectors within the transfer line.

In the event of a mission abort, the LO_2 and LH_2 propellants will be dumped before landing in order to reduce the orbiter landing weight and to minimize the amount of hazardous propellants on board at touchdown. To minimize the dump duration, the propellant dump will be performed with helium pressurization. During an RTLS or transatlantic landing abort, the propellant dump will be performed during SSME operation to take advantage of the available vehicle acceleration rate. The dump thrust disturbances generated at the dump line exit and on the wing will be nulled by the SSME's during these TAL aborts. The minimum dump duration in the event of an RTLS abort for helium sizing is assumed to be the same as that for a Centaur abort dump, which is 250 seconds. In the event of an abort from orbit, the propellant dump will be performed in conjunction with OMS burns and also at low g with the RCS providing propellant settling and attitude control. Since the maximum quantity of propellants to be dumped is less than the Centaur G-prime propellant quantity

(46,000 pounds of LO_2 and LH_2) and because the same dump port locations are used, the RCS requirements are expected to be no greater than those for the Centaur G-prime, which have been determined to be acceptable.

A two-phase flow analysis of the propellant dump has been completed in order to define the dump flow characteristics. The two-phase flow analysis is based on the Lockhart-Martinelli correlation method, which has been used successfully for Centaur and MPS propellant dump predictions. The dump flow rate definition analysis is presented in Figures 68 and 69 for the LO_2 and LH_2 systems. Based on the results of the two-phase flow analysis, a 3.5-inch-diameter dump line was selected. For the 3.5-inch line and assuming 30,000 pounds of total propellant to be dumped, the tanks must be pressurized to 7.5 psi and 4.0 psi above the liquid vapor pressure for the LO_2 and LH_2 tanks, respectively, in order to satisfy the 250-second RTLS dump time requirement. The helium required to perform the pressurized propellant dump is estimated to be approximately 37 pounds, which is within the usable helium quantity of the helium storage bottle. For on-orbit dumps lower tank pressure levels may be used since longer dump times can be accommodated.

The results of the transfer and dump analyses discussed above are summarized in Table 26. They indicate that the fluid transfer is feasible and within the performance capability of the concept presented in this study.

In order to satisfy the operating parameters shown in Table 26, the CAS must be capable of delivering approximately 8 lbm/sec of LH_2 and 34 lbm/sec of LO_2 from a source pressure of 21.5 psia. The analytical equations and computer program that were developed for the trade study described in Section 4.2.7 were used to point design the width and depth of the rectangular

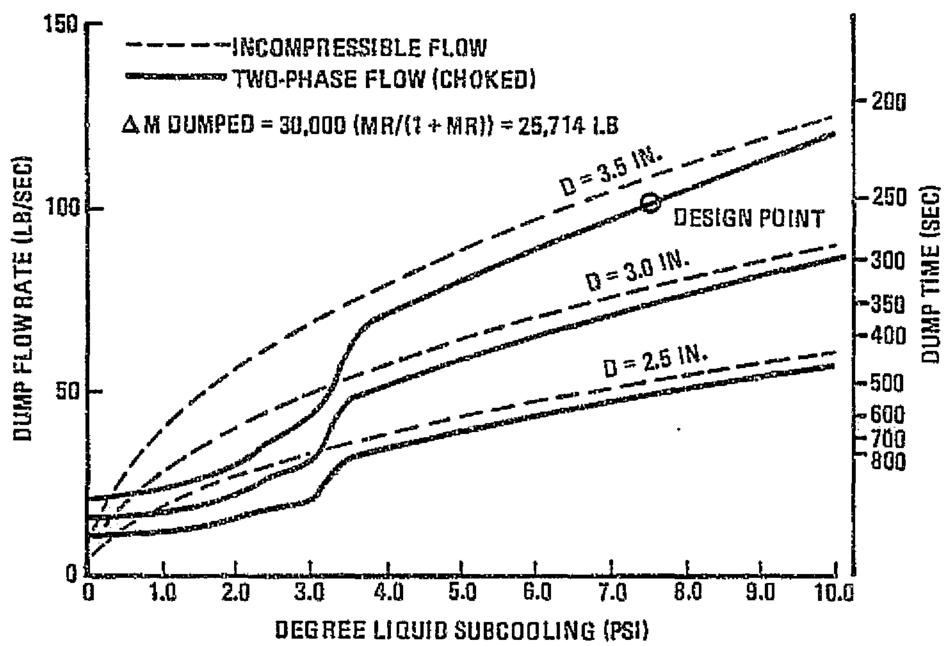


Figure 68. LO_2 Dump Line Sizing for Propellant Transfer System

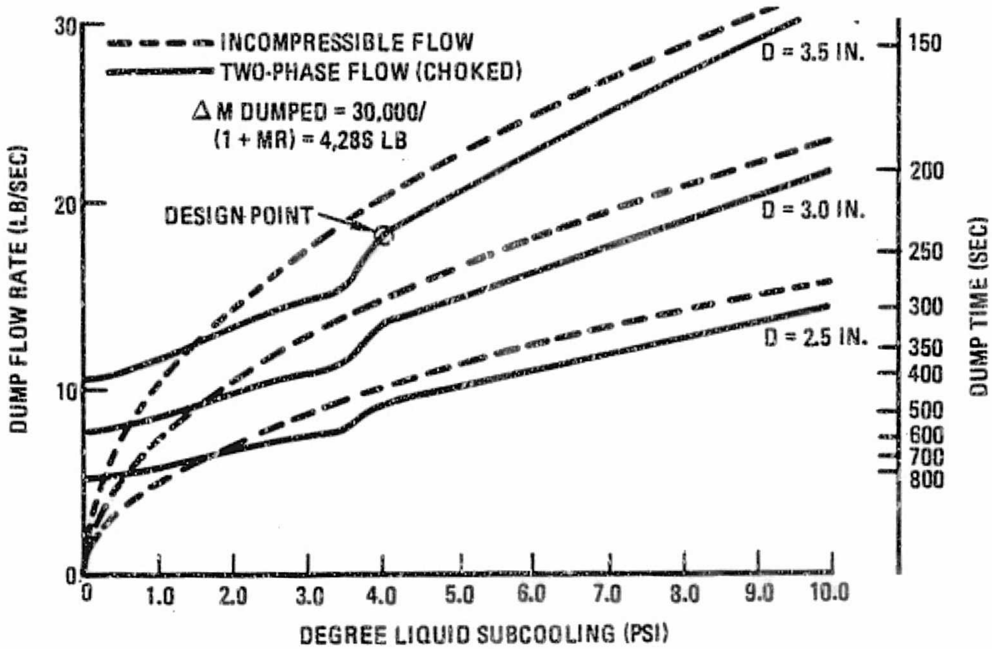


Figure 69. LH_2 Dump Line Sizing for Propellant Transfer System

Table 26. Propellant Transfer System Operating Parameters

	LO_2 System	LH_2 System
Transfer from ET to orbiter tanks		
Line size (ID, in.)	2.0	2.0
Loss coefficient (K)	2.04	1.77
ET Pressure Level (psia)	20 to 19.8	32 to 31.3
Orbiter tank pressure level (psia)	18.5 ± 0.5	22.1 ± 0.5
Propellant mass transferred (lbm)	3,517	2,272
Duration (sec)	220	217
Pressurant requirement (lbm)	None	None
Vapor mass vented (lbm)	9.7	31.8
Transfer from orbiter to user tank		
Line size (ID, in.)	2.0	2.0
Loss coefficient (K)	5.6	5.34
Orbiter tank pressure level (psia)	21.5 ± 0.75	21.5 ± 0.75
User tank pressure level (psia)	16.5 ± 0.50	16.5 ± 0.50
Propellant mass transferred (lbm)	25,714	4,286
Duration (min)	24	16
Pressurant requirement (lbm)	9.8	25.2
Vapor mass vented (lbm)	80	27
Abort dump		
Line size (ID, in.)	3.5	3.5
Loss coefficient (K)	0.88	0.78
Tank pressurization level above vapor pressure (psid)	7.5	4.0
Dump flow rate (lb/sec)	102	18.2
Dump duration (sec)	250	250

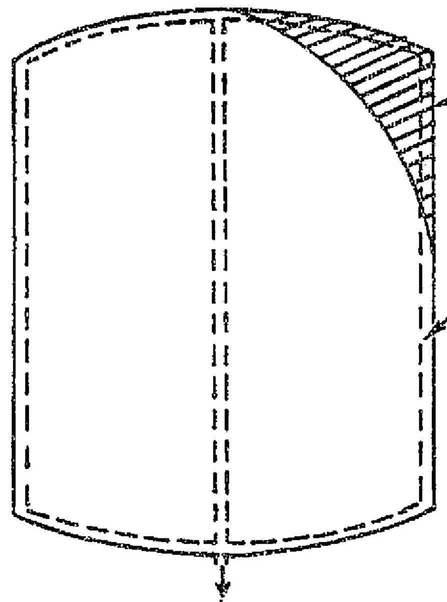


Figure 70. Worst-Case Propellant Orientation--LO_2 Tank

capillary channels for each of the cryogenic storage tanks. The trade study initially had assumed storage tank pressures of 27.6 and 28.81 psia for the LO_2 and LH_2 tanks, respectively. These were updated to 21.5 psia for each tank. The worst-case propellant orientation for the LO_2 tank also had been reevaluated to assume an orientation where only one of the channels was immersed in LO_2 during the final expulsion process (see Figure 70). The flow path analysis was adjusted accordingly in the computer program to accommodate this change. The propellant orientation for the LH_2 tank as assumed to be the same as for the system trade study. The results from the analysis indicated that a 7.0 by 2.0-inch channel is required to satisfy the LH_2 design flow requirement and a 6.0- by 1.5-inch channel to satisfy the LO_2 flow requirement. Figures 71 and 72 show the outflow capability for each of the selected CAS's as a function of the tank fluid volume. Residuals for the LH_2 and LO_2 tanks after maximum fluid expulsion are predicted to be 3 percent and 1 percent, respectively. Tank residuals were determined from the volume of fluid remaining in each tank when the bubble pressure constraints (13.87 lbf/ft^2 for LH_2 and 99.23 lbf/ft^2 for LO_2) were exceeded for the given flow rates.

4.4.2.1.4 Component Definition. Most of the components selected for use in the pneumatic system for valve actuation and tank pressurization are qualified for Space Shuttle environments. These components were certified for a minimum of 100 missions and do not require any type of design changes to meet scavenging system requirements.

Proposed components for the fill and drain, propellant transfer and thermodynamic TVS functions were selected from existing Shuttle or Centaur cryogenic systems. These components are either fully qualified or are undergoing design changes or qualification testing at various subcontractors.

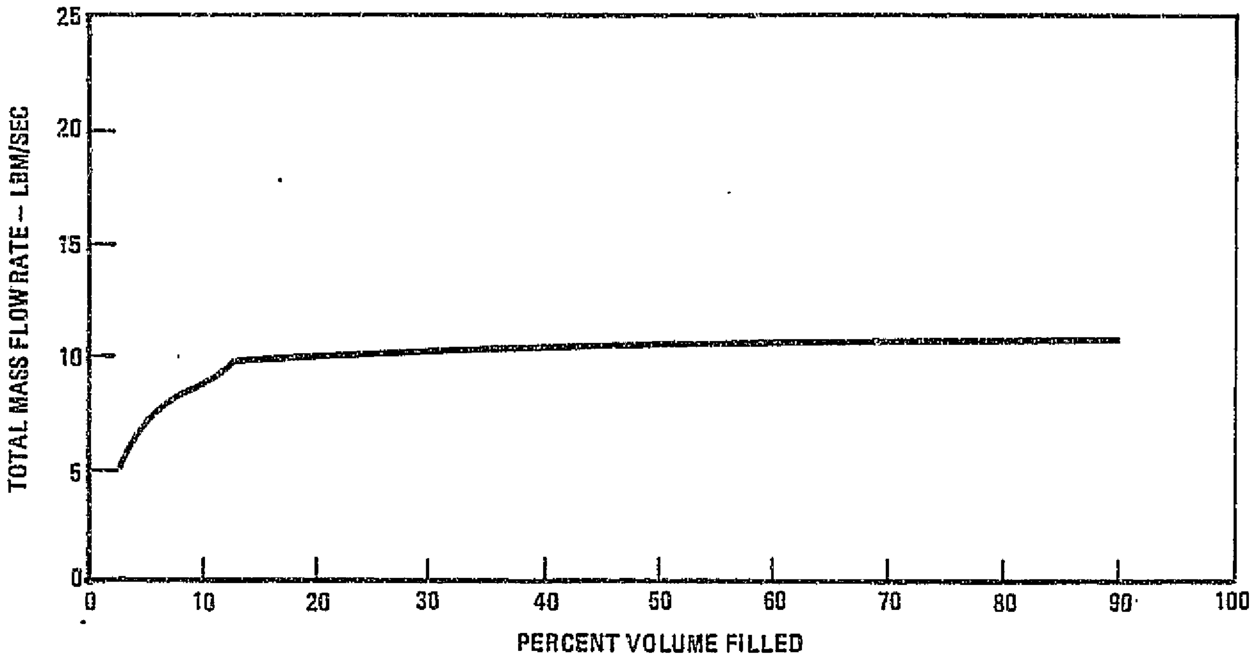


Figure 71. LH_2 CAS Flow Capability--7- by 2-inch Channel

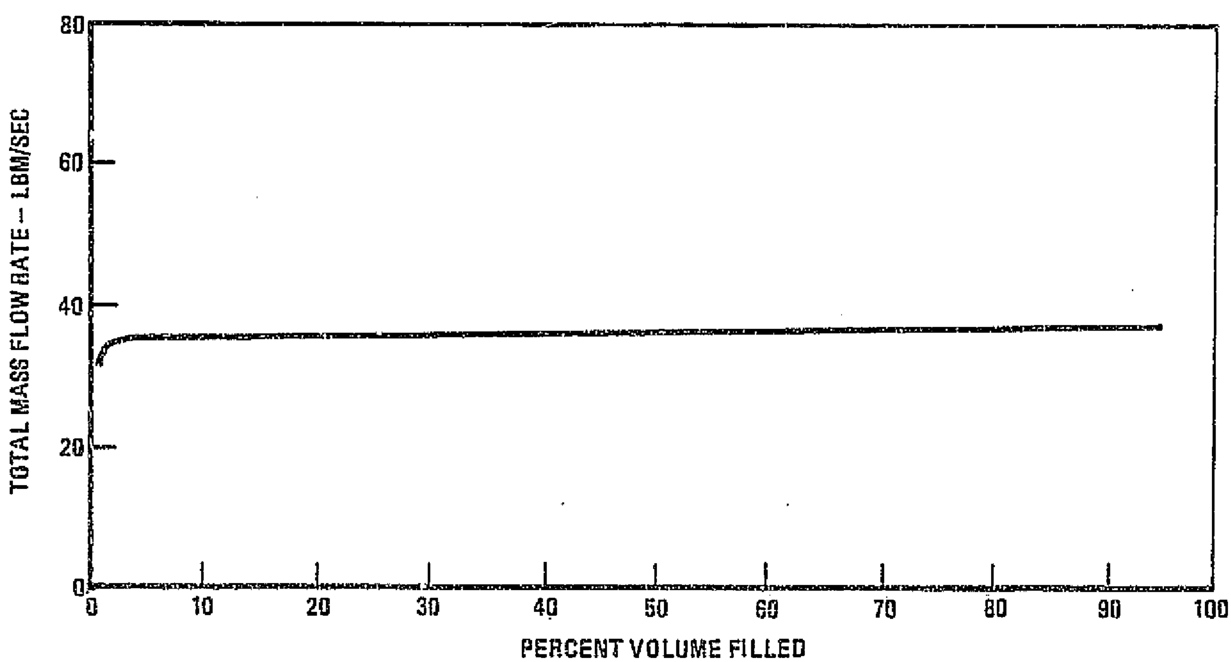


Figure 72. LO_2 CAS Flow Capability--6- by 1.5-inch Channel

It is anticipated that all of these components will be fully certified to meet the Shuttle requirement before the scavenging system need date. In the selection of components, priority was given to the availability of suitable qualified existing hardware. By maximizing the use of off-the-shelf hardware, a cost-effective and reliable system will be realized.

The selected components are listed in Table 27. The system schematics are presented in Figures 1 and 2. A detailed description of each component is presented in the appendix.

Table 27. Preliminary Component List

Name	Part No.	Quantity	Vendor
LO_2 CRYO SCAVENGING SYSTEM			
LO_2 lines	TBD	TBD	Ametek/Straza
System fill valve	MC284-0395 series	1	Consolidated Controls Corp.
Seals	MC261-0045 series	TBD	TBD
Dump valve dual actuator	83053001 (modified)	1	Fairchild Control Systems Co.
User transfer valve	MC284-0395 series	1	Consolidated Controls Corp.
Disconnect--dual poppet	TBD	1	TBD
Solenoid valve--3-way	MC284-0404-0012	7	Wright Components
Solenoid valve--2-way (tank pressure)	MC284-0403-0011	2	Wright Components
Orifice (pneumatic system)	TBD	2	TBD
Check valve	ME284-0472	1	Circle Seal
Relief valve	200600	1	HTL
Vent valve	MC284-0395 series	1	Consolidated Controls Corp.
Orbiter relief shut-off valve	MC284-0395-0052	1	Controlidated Controls Corp.
Orbiter fill and drain valve	MC284-0397-series	1	Fairchild Control Systems Co.
Flowmeter	TBD	1	Quantum Dynamics

Table 27. Preliminary Component List (Cont)

Name	Part No.	Quantity	Vendor
LH_2 CRYO SCAVENGING SYSTEM			
LH_2 lines	TBD	TBD	Ametek/Straza
System fill valve	MC284-0395 series	1	Consolidated Controls Corp.
Seals	ME261-0045-series	TBD	TBD
Dump valve--dual actuator	83053001 (modified)	1	Fairchild Control Systems Co.
User transfer valve	MC284-0395 series	1	Consolidated Controls Corp.
Flowmeter	TBD	1	Quantum Dynamics
Disconnect dual poppet	TBD	1	TBD
Solenoid valve--3-way	MC284-0404-0012	8	Wright Components
Solenoid valve--2-way (tank pressure)	MC284-0403-0011	2	Wright Components
Orifice (pneumatic)	TBD	2	TBD
Check valve	ME284-0472-00XX	1	Circle Seal
Relief valve	200600	1	HTL
Vent valve	MC284-0395 series	1	Consolidated Controls Corp.
Shutoff valve--TVS	MC284-0395-0052	1	Consolidated Controls Corp.
Orbiter fill and drain valve	MC284-0397 series	1	Fairchild Control Systems Co.
Orbiter relief shut-off valve	MC284-0395-0052	1	Consolidated Controls Corp.
Orifice (TVS)	TBD	1	TBD

Table 27. Preliminary Component List

Name	Part No.	Quantity	Vendor
LH_2 CRYO SCAVENGING SYSTEM			
Controlling valve module (CVM)	TBD	1	Consolidated Controls Corp.
Mixer motor/pump/ heat exchanger	TBD	1	Sundstrand
HELIUM PRESSURIZATION SYSTEM			
Check valve	ME284-0472-0002	1	Circle Seal
Check valve	ME284-0472-0034	2	Circle Seal
Solenoid valve--2-way	MC284-0403-0011	2	Wright Components
Regulator--750 psig	MC284-0533-0004	2	Consolidated Controls Corp.
Helium supply tank--17.3 ft^3	MC284-0082-0001	1	Brunswick
Relief valve--850 psi	MC284-0398-0005	2	Consolidated Controls Corp.
INSTRUMENTS/CONTROL SYSTEM			
Control system	TBD	2	TBD
Signal conditioners	TBD	TBD	TBD
Signal conditioners chassis	MC476-0147-3004	1	ELDEC
Temperature sensors	ME444-0010 series	6	RDF
Pressure transducer He tank and line	ME449-0177 series	1	Gould
Pressure transducer GH_2 and GO_2 ullage	TBD	6	TBD
Quantity gaging system	TBD	2	Simmonds Precision
Bubble detectors	TBD	2	TBD

The instrumentation and control system for the LH_2 and LO_2 propellant scavenging system is based on using proven hardware and concepts that will require a minimum of development. The control system is designed to meet the Shuttle high-reliability fail-safe criteria.

Various candidate systems were considered in the development of the control and monitoring block diagram shown in Figure 73. The selected system utilized GPC software to control the vehicle maneuvers for propellant acquisition and valve control for ground and flight use. The actual valve control sequence will be controlled by local microprocessors using sequence initiation or termination commands from the GPC.

An on-board keyboard entry or an uplink command will be utilized for this function. Data parameters required to control the loading, scavenging, and transfer operations will be provided to the GPC by means of multiplexer/demultiplexer (MDM) channels and will be sequentially downlisted and displayed on the crew station CRT display. This control concept will require minimum involvement of the payload specialist.

An alternate system could be used to maintain all valves and instrument controls via GPC commands without any local microprocessor control. This design is only feasible if and when upgraded GPC's with larger memories become available and adequate programming/memory space can be allocated to this system.

Two power and control assemblies will provide the control, logic, switching and driver circuits for the valves, instrumentation, and LH_2 mixer motors. All instrumentation signal conditioners for pressure transducers, temperature transducers, liquid level detectors, and flowmeters will be housed in a one-fourth size Shuttle signal conditioner enclosure. A flexible MDM will be used if payload MDM channels are not available for the data and commands issued to the scavenging system. Microprocessors and associated logic will be housed in separate housings. Plug-in read only memories (ROM's) will contain firmware programs for the microprocessors.

Electrical power for the scavenging system is supplied by the orbiter power distribution system. Two independent 28-Vdc and two independent 115-Vac, 3-phase, 400-Hz fused power sources are utilized. Dc power will tie into orbiter power accommodations provided for the Centaur vehicle, mid body, or payloads. AC power will be provided from payload accommodations for AC 2 and 3 power sources.

A description of the instrumentation and control system hardware follows:

1. Flexible MDM. the flexible MDM will be utilized if payload MDM's are not available. The flexible MDM allows the selection of cards and channels tailored to the requirements of the scavenging system. Use of the flexible MDM's avoids the necessity of utilizing the payload MDM channels.
   P/N: TBD
   Supplier: Sperry Flight Systems

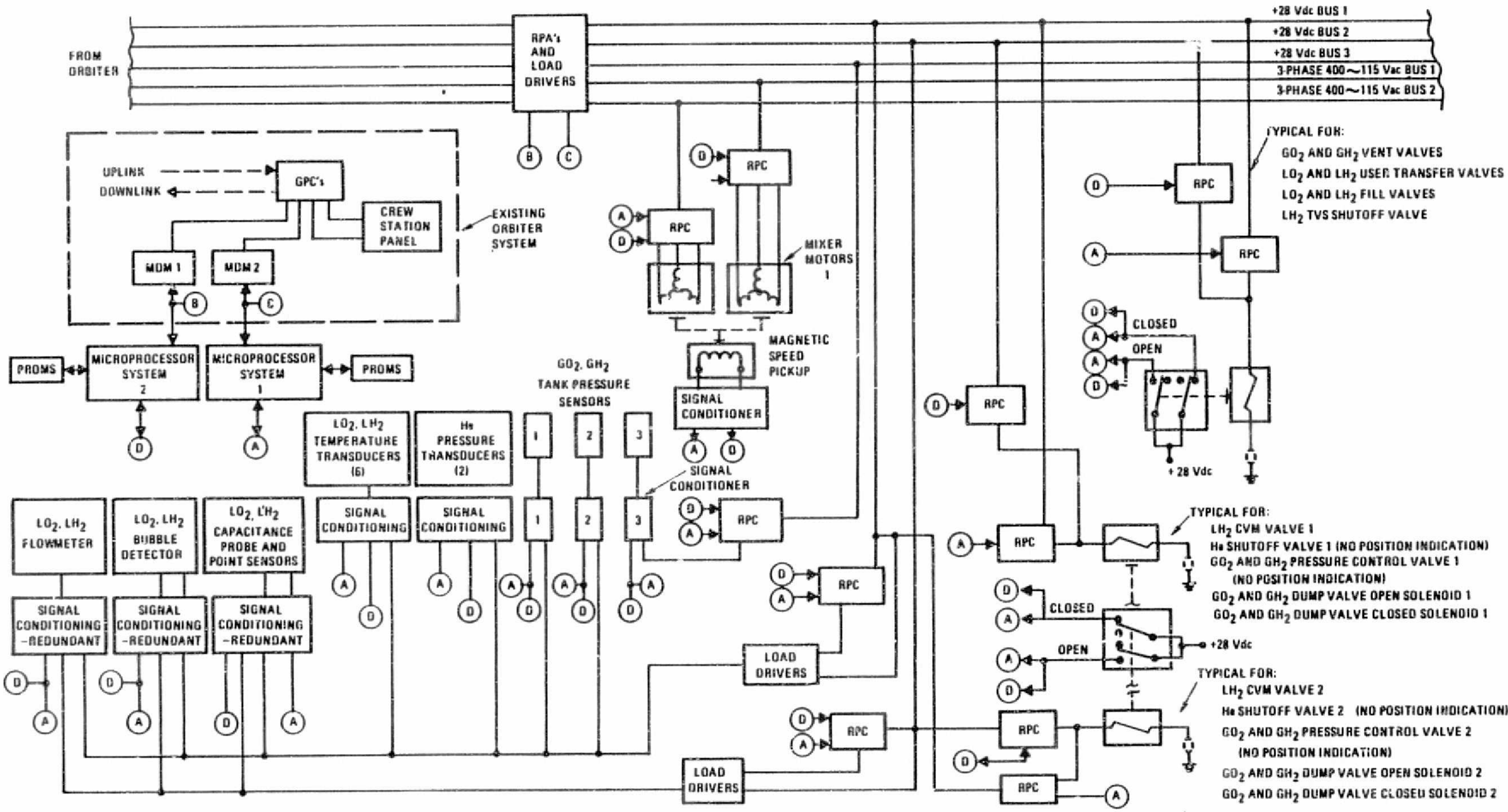


Figure 73. Propellant Scavenging System Power and Control

2. Remote power controllers (RPC's) and load drivers. Existing Shuttle RPC and load drivers would be utilized as required.

3. Dedicated signal conditioner. The existing one-quarter size unit will be utilized.
   Chassis: P/N MC476-0147-3004
   Printed circuit boards: P/N TBD
   Supplier: ELDEC

4. Temperature sensors. The temperature measuring sensors to be used to monitor tank structure and fluids will utilize proven Shuttle platinum sensing elements with TBD ice point. A standard three-wire design is provided.
   P/N: ME444-0010 series
   Supplier: RDF

5. Pressure transducers. Three GO_2 and three GH_2 pressure-sensing transducers will be used to monitor the O_2 and H_2 tank ullage pressure. The requirement to control the pressure in the 20- to 23-psia band with ± 0.3-psi monitoring accuracy during propellant transfer operations creates the need for a high-accuracy unit. A desired range of 0 to 30 psia will satisfy all of the other requirements. To achieve this accuracy a small development program will be required to test various existing and qualified designs that incorporate improved quality and designs. The selected transducer would then be subjected to delta qualification test programs.

   The use of three parallel pressure-sensing systems in each tank will provide the required redundancy to meet the fail-operational, fail-safe requirement. In addition, the use of three units provides the capability to RSS the readings from all three units to improve system accuracy.
   P/N: TBD
   Supplier: TBD

   Two pressure transducers will be provided to monitor the He tank and pressure regulator. Standard Shuttle strain gage transducers will be used.
   P/N: ME449-0177 series
   Supplier: Gould, Inc.

6. Liquid level gaging system. A full-length continuous capacitive gaging system will be provided for ground LO_2 and LH_2 propellant loading. The probe will be between 6 and 10 feet long, depending on the final tank design. It will be located along the X axis of the Shuttle. Concentric lightweight aluminum tubes with small tubular capacitive point sensors at the lower end and other TBD locations will be used. The lower point sensor will also serve as a density compensator and provide automatic probe calibration. The resultant RCS accuracy will be ± 0.41 percent (LO_2) and ± 0.59 percent (LH_2) of full scale. For a 6-foot tank this will be equivalent to ± 0.42 inch, which will meet the scavenging system requirements for propellant loading.

The capacitance probe and point sensors will be excited by a 6-kHz sine wave whose amplitude is proportional to the dielectric constant of the fluid being sensed. The high Z signals from the probes and point sensors will be routed through a preamplifier, a rectifier, and an active filter, which will produce an analog 0- to 5-Vdc signal. The point sensor signals subsequently will be routed through level comparators to produce discrete ON-OFF signals. Quadrature rejection will be used to eliminate noise and cross talk on the probe signals. Redundant cabling and signal conditioners will be utilized through a switching system to meet the reliability requirements.

Capacitance probes have been used extensively on spacecraft, boosters, and aircraft with excellent results. The design to be used here was proven on the Saturn boosters, Apollo, the Shuttle main propulsion test article (MPTA), and various other programs.

P/N: TBD
Supplier: Simmonds Precision

The requirement to gage propellant quantity during flight propellant scavenging operations may not be achievable with the present capacitance probes due to a wetting/meniscus effect from low-g loading (0.0015 g). Therefore, a development program will be performed to determine if the capacitance probes can be optimized to work under low-g conditions. Figures 74 and 75 show a representative capacitance probe and a diagram of the required electronics signal conditioning.

Zero-g gaging concepts have been investigated periodically during the past 15 years; however, none was ever developed sufficiently and qualified for flight use. If the capacitance probes are not usable under low-g conditions, other systems will be utilized. However, a larger development program will be required.

During low-g operations additional liquid level point sensors will be utilized to detect when the tanks are fully loaded. These units will be provided as a backup to the capacitance probes since it is doubtful they can be successfully used under low-g conditions. The sensors will use a heat transfer monitoring principle. Relatively large quantities of heat can be transferred in contact with a liquid compared to gas. As a result the sensor electrical characteristics will vary between gas and liquid. The transducer will have a thin resistive film plated on the tip of an open small prism mounted off the capacitance probe. This concept has been effectively utilized in many gases and liquids and has proven to be very reliable. A small development and qualification program will be required.

P/N: TBD
Supplier: TBD

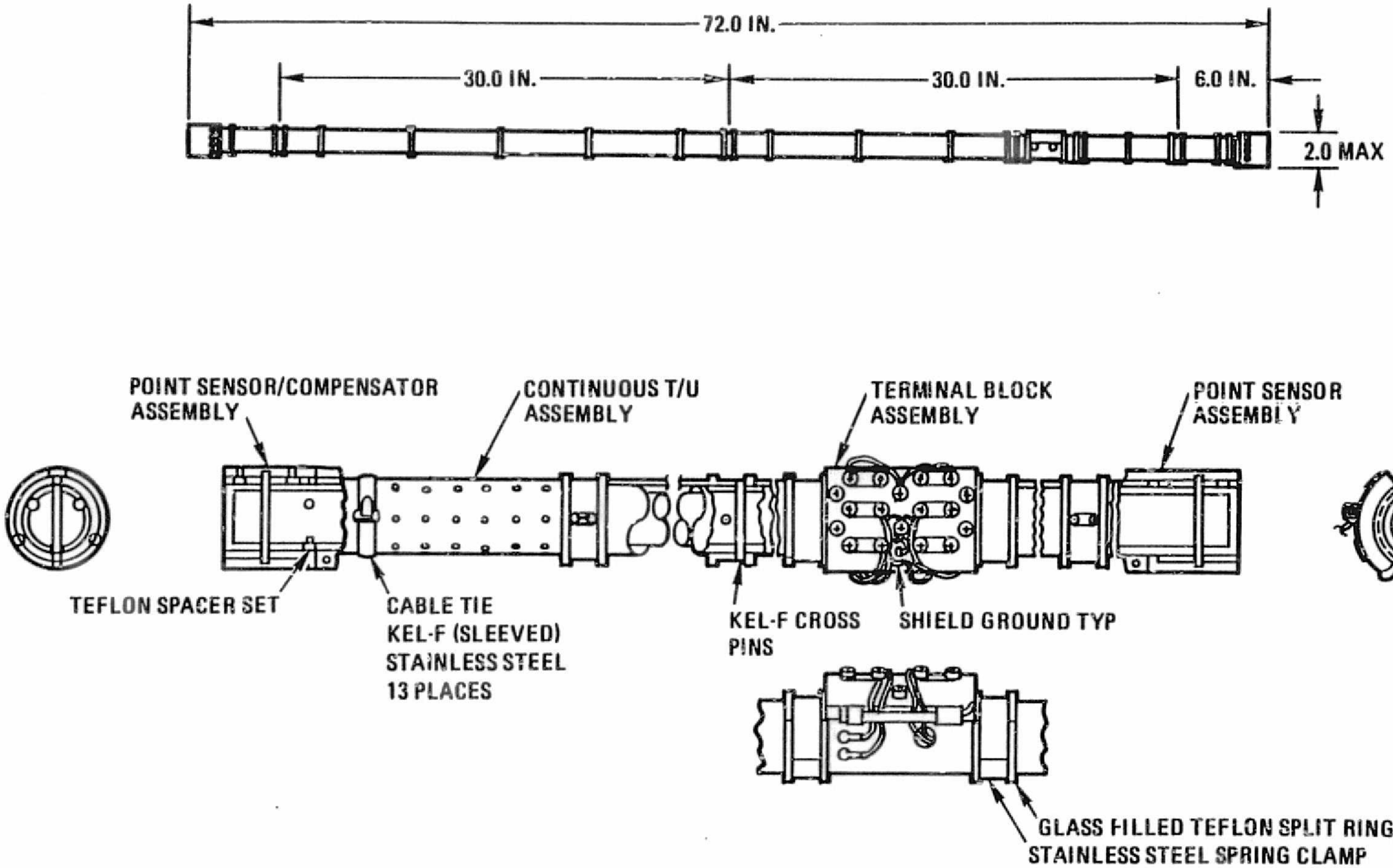


Figure 74. Cap Probe/Point Sensor Assembly

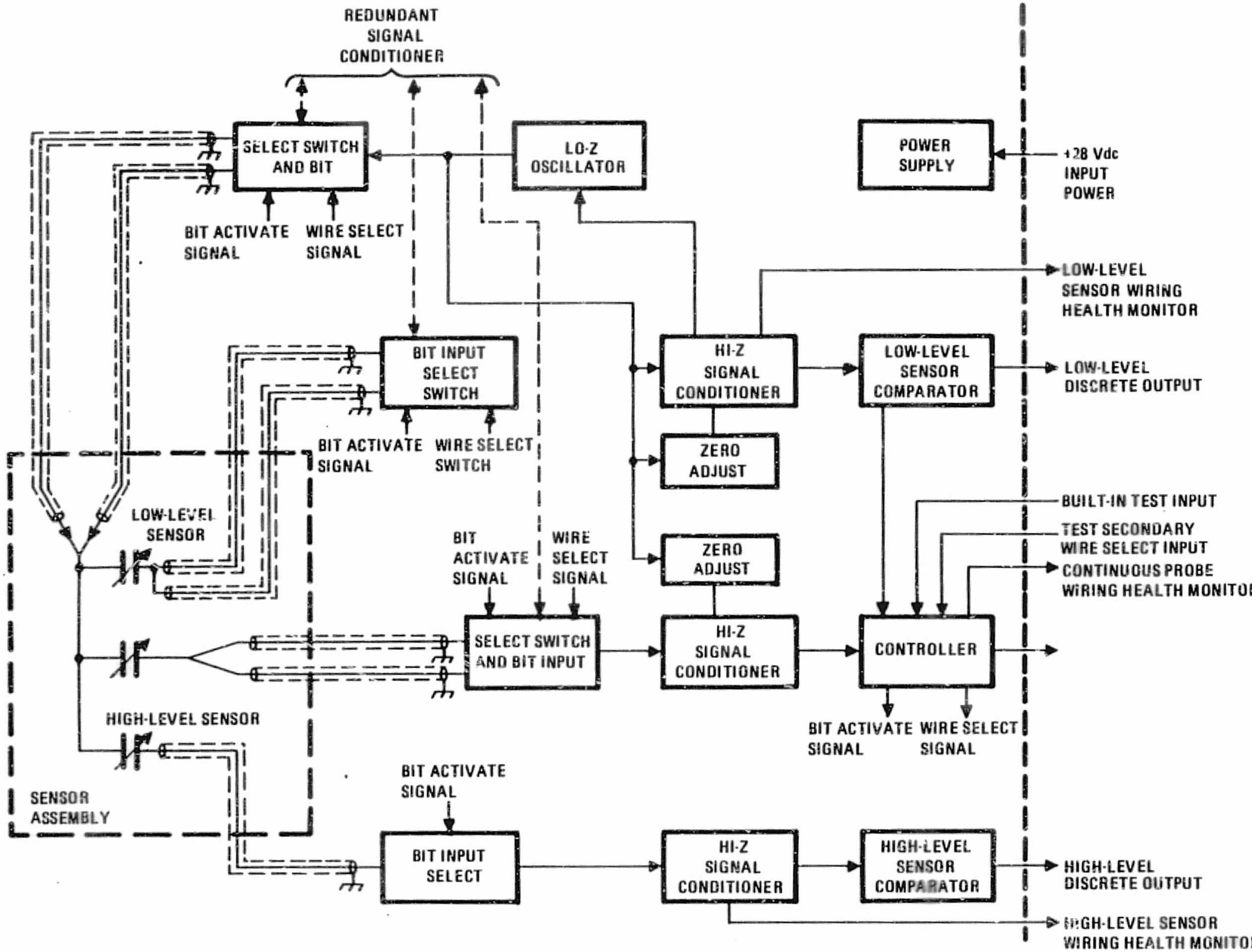


Figure 75. Cap Probe/Point Sensor Block Diagram

7. Flowmeters. A propellant mass sensing flowmeter will be provided in the LH_2 and LO_2 transfer lines. Due to the very flow fluid velocity in the 2-inch line, very few meters can be utilized or are flight worthy. A combination density and velocity meter using a turbine technique will be utilized. The flowmeter is based on a proven design that has been utilized on many flights and in ground programs. It utilizes a coaxial twin turbine in which the flow sensing turbine rides freely on a shaft that is driven by a slave turbine. Both turbines rotate at nearly the same speed. The downstream turbine serves to match flow impedance (rate of pressure change over that of flow) of the upstream turbine. This provides a means of stabilizing pressure and density gradients that would amplify flow perturbations. Combined with this unit is a density-measuring capacitance meter. Redundant signal conditioners and cabling will be utilized to provide a usable propellant mass signal of 0.3 percent (full scale) accuracy. A development and qualification program will be required for the final flight configuration.
   P/N: TBD
   Supplier: Quantum-Dynamics

8. Bubble detectors. A bubble detector will be provided in the transfer lines next to the tank to determine when liquid flow has ceased during filling and transfer operations. An ultrasonic type of detector will be utilized. A small development and qualification program will be required.
   P/N: TBD
   Supplier: TBD

The required line diameters identified in the study (2 and 3.5 inches) are identical to, or derivatives of, the state-of-art sizes of the Shuttle LH_2 recirculation system lines (MC271-0075) and Centaur servicing lines (MC271-0101). All of these line assemblies are certified or are in the process of being certified for their respective mission life (100 or 40 missions).

New technology or extensive development effort will not be required to design and manufacture any scavenging line shown in the layouts. Utilizing Rockwell's suppliers who design and build this size of ducting (such as Ametek/Straza, which builds the MC271-0075 and MC271-0101 lines), the routing, proven construction techniques, existing designs where possible will provide minimal cost, lead time, and risk to provide the prototype, demonstration, or production hardware for the scavenging system.

Rockwell's familiarity with the orbiter aft fuselage and payload bay environment assures a smooth transition to procure these components since all materials, processes, and requirements are defined in the MC271-0075 LH_2 recirculation system lines and MC271-0101 Centaur servicing lines and are directly transferable to the scavenging system.

While heat transfer prior to launch or orbit is not a concern with the scavenging system, the use of insulated lines and insulated flex-joints will

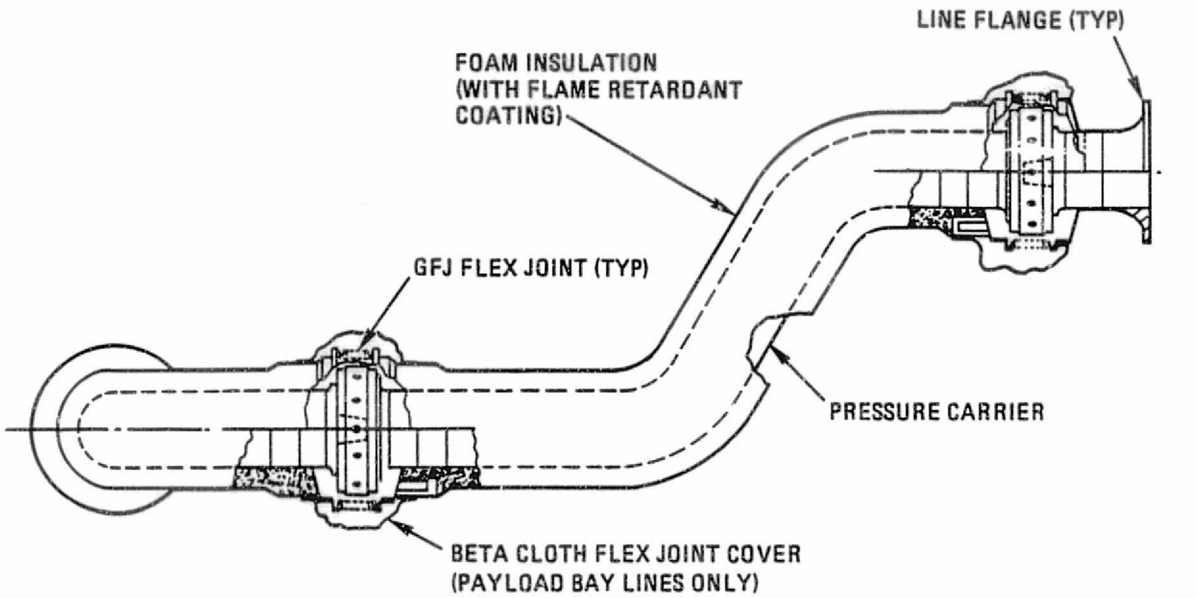


Figure 76. Typical Foam-Insulated LH_2 Line Assembly

be needed to assure proper maintenance of the cryogens during orbital tanking and transfer functions. All lines will have a precast foam/Kevlar-resin insulation system covered with a flame-retardant coating (see Figures 76 and 77). The foam insulation will be flame-retardant polyurethane type (2.1 lb/ft^3, MB0130-133) covered by a flame-retardant sealant (MB0120-065 Type II)

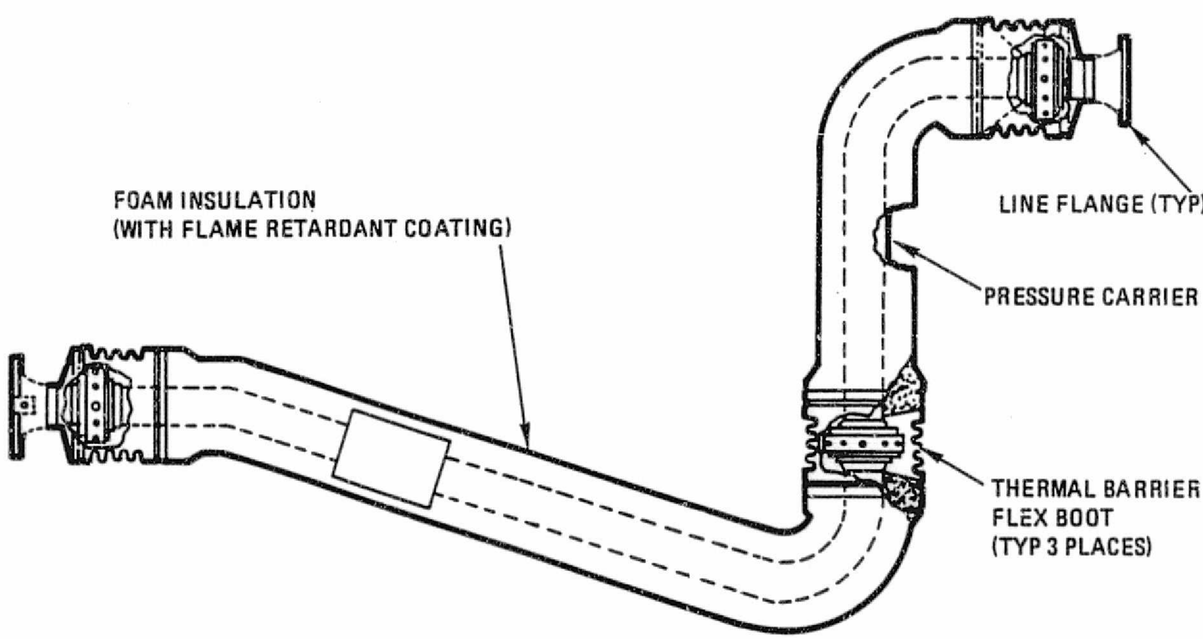


Figure 77. Typical Foam-Insulated LO_2 Line Assembly

to assure no bare metal surfaces are exposed to direct sunlight on orbit (900°F potential temperatures). Beta-cloth boots will be installed over the flex-joints on the LH_2 lines. This is identical to the Centaur servicing line configuration in the payload bay.

Line construction methodology will be that of the existing lines noted previously. The tee junction where the 1.5-inch LO_2 vent line interfaces with the 3.5-inch LO_2 dump line is similar to the qualified MPS line junction of the LH_2 topping line and LH_2 fill and drain line on the orbiter. No design or test concern exists since the method of forming smaller line penetration is a proven technique. Additionally, the transition from 3.5-inch to 5.5-inch diameter at the dump tube junction also is proven. The LH_2 and LO_2 dump exit tubes will be the same 5.5-inch assembly used in the Centaur servicing system (Rockwell P/N's V413-411301 and V413-411401) (see Figure 74).

All LH_2 line flexible elements will be the gas-filled joint (GFJ) type that are used on the Centaur LH_2 servicing lines and the Shuttle SSME cryogenic lines (see Figure 78). These elements are sealed subassemblies that contain a gimbal joint with a lined pressure carrier bellows. An external bellows and two standoff adapters forms an annulus that is back filled with argon gas and sealed off by a pinch tube. During line assembly manufacture, these acceptance-tested GFJ's are welded to the pressure carrier tube sections. In operation, the cryogen fluid cryo-pumps the argon gas volume from nominal atmospheric pressure to less than 1 micron of mercury. These GFJ's have been proven to be totally functional and reliable in the Centaur servicing line certification tests and in the operation of the SSME cryogenic lines in the Shuttle orbiter.

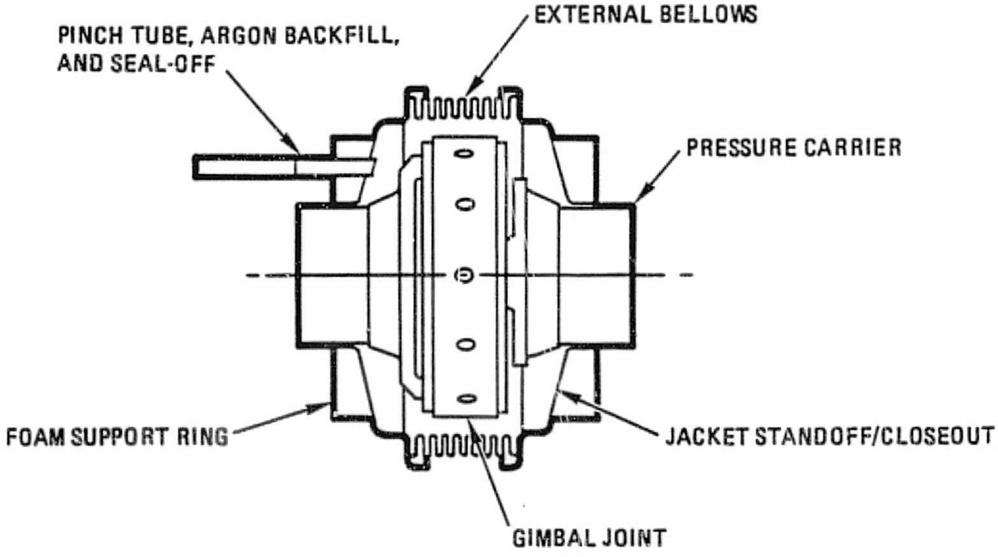


Figure 78. Typical Gas-Filled Flex Joint

It is expected that the thermal barrier flexible boots will be used over the gimbal joints on the LO_2 line assemblies in the manner as on the Centaur LO_2 servicing lines (see Figure 77). these are two-ply silicone-coated (room-temperature vulcanized) fiberglass corrugated boots that are vacuum-baked to preclude outgassing during orbital operations.

The flexible joints needed to accommodate scavenging line installation tolerances, dynamic motions, and thermal effects will be the lined gimbal type currently used on the 1.5-inch and 2-inch MPS lines and Centaur servicing lines (see Figure 79). The scavenging system motion requirements will be less than those imparted by the Centaur payload because the scavenging tank module interface will be closer to the X_o 1307 bulkhead and will have a closer coupled structural attachment in the aft portion of the payload bay than the Centaur/Centaur integrated support structure payload. Consequently, the existing 1.5- and 2-inch gimbals (± 15-degree and ± 13-degree angulation capability, respectively) can be satisfactorily placed in the line routing without any change in the qualified gimbal design. All orbiter gimbal bellows and lines have been assessed and certified to JSC 08123 for flow-induced vibration. The scavenging system bellows will be certified similarly.

A new 3.5-inch gimbal will be required but will not be a design constraint. There are qualified 4- and 5.5-inch gimbals in the orbiter lines that have the same design concept as the aforementioned 1.5-inch and 2-inch units. Consequently, upscaling/downscaling to the 3.5-inch size will not be a problem. No development will be needed.

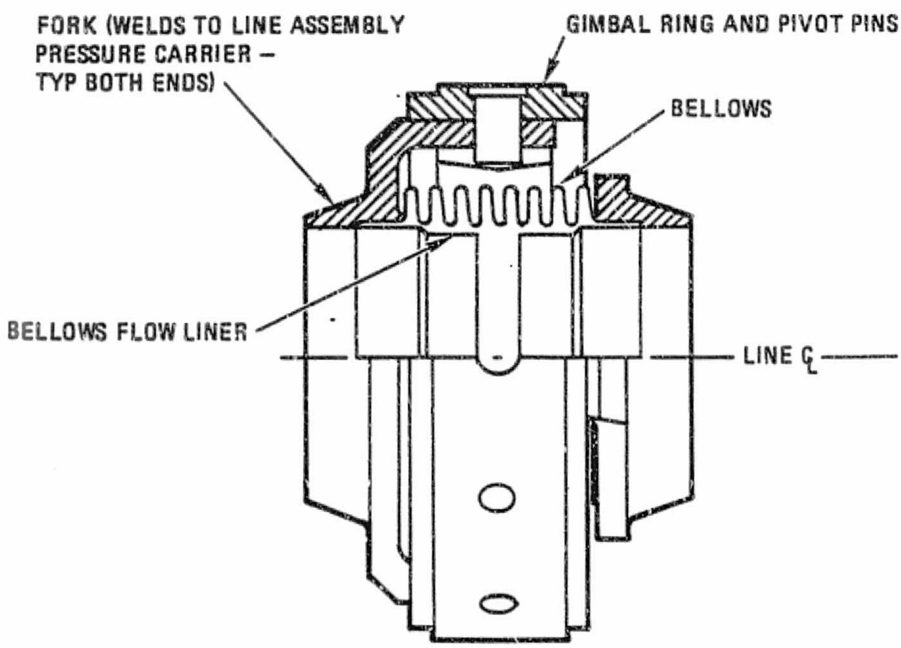


Figure 79. Typical Flexible Line Gimbal

Line assembly flanges will be the same low-profile Inconel 718 bolted design used on all Shuttle MPS cryogenic lines. Since the lines are, for the most part, the same size as existing orbiter and Centaur servicing lines, these same flanges can be used on the scavenging lines with minimum engineering effort. A new 3.5-inch flange will be required for interfaces with the modified relief and dump valves, which are discussed in the appendix.

The low profile flange design is integral with the ME271-0045 cryogenic static-face seal design used throughout the MPS and Centaur servicing line system. The ME261-0045 Type I seal (Figure 80) is a proven cryogenic design with the significant feature of being fully refurbishable, should it be damaged, at less than half the cost of procuring new seals. New seal designs will not be required for the scavenging system. Seals with 1.5-, 2-, 2.5-, 3.5-, and 5.5-inch inside diameters have been certified for 100 missions and can be used without change if required.

4.4.2.2 Storable Propellant Hardware Description.

4.4.2.2.1 Detailed Schematic. The ground rules incorporated into the propellant scavenging schematic assume that venting into the payload bay is limited by three inhibits for any overboard propellant line and two inhibits for any overboard pressurant line. Ball valves, solenoid valves, QD's, burst disks, and relief valves are all considered inhibits. In addition, flow control valves are required so a single-point component failure does not terminate the resupply operation. The failure protection and redundancy rationale are defined below (see Figure 4).

For the propellant tanks of the PBTS, the tank isolation valves and feedline isolation valves are required to protect against a valve "failed-open" condition when OMS propellant scavenging is desired. PBTS propellant isolation valves are also required during propellant loading prior to launch.

The propellant pump isolation valves are used to isolate one pump from the other since only one of the two redundant pumps will be used during propellant transfer. The valves are also required to isolate a pump if propellant leaks because of a pump seal failure.

The propellant isolation valves near the QD's are required to reduce the propellant line volume if a QD seal fails. The ullage return lines also have three inhibits: the QD, QD isolation valves, and OMS pod scar plumbing isolation valves.

The propellant QD purge system also requires three inhibits. The series parallel redundant valves from the helium lines account for two inhibits, and the third is the QD. The PBTS propellant tanks also require overpressurization protection and have three inhibits to the nonpropulsive vents. The three inhibits are made up of dual burst disks and a relief valve.

The pressurant isolation valves near the helium bottles are required to isolate the PBTS helium from the OMS helium. The pressurant valves upstream of the regulators are required to isolate each leg of the propellant tank pressurization system with an additional vapor isolation valve set between the

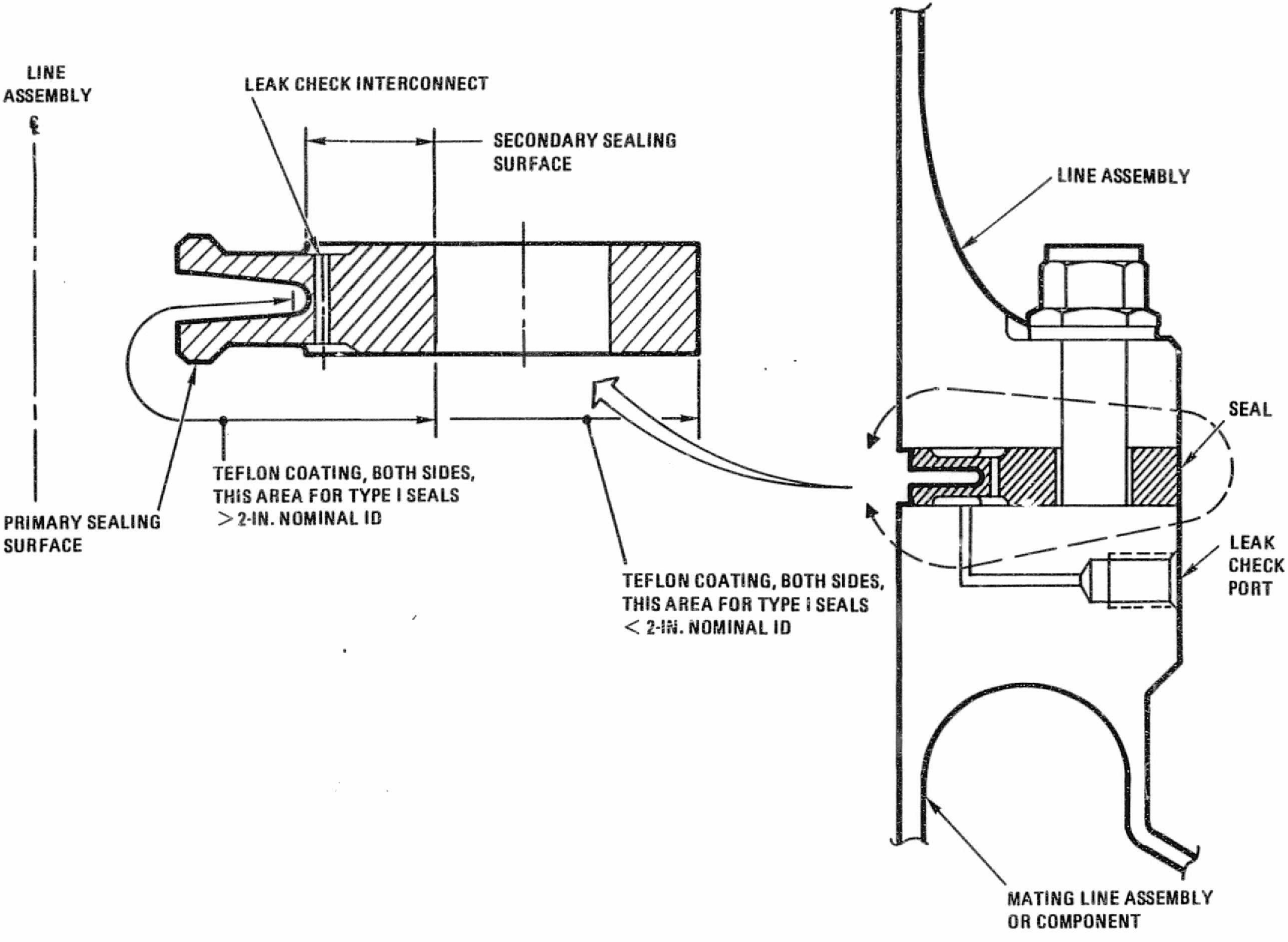


Figure 80. ME261-0045 Cryogenic Seal

quad check valve and the regulators on the oxidizer pressurant leg. Isolation valves are required for gas compressor operation or isolation in the event of a seal failure in the gas compressor. The isolation valves also provide the required two inhibits to the overboard pressurant line.

The orbiter and OMS pod scar plumbing schematic is presented in Figure 3. This scar plumbing brings the OMS propellant, propellant ullage return, and pressurant lines up to the 1307 bulkhead flanges. The diameter of the existing propellant lines from the OMS crossfeed lines would be increased from 1/4 inch to 1/2 inch. The propellant ullage return (1/2 inch diameter) and pressurant (1/2 inch diameter) lines to the 1307 bulkhead would be new.

Bipropellant scavenging may be accomplished by other combinations of components. The schematics associated with these other systems are included in the appendix. Figure A-28 in the appendix presents the basic schematic for OMS propellant and pressurant scavenging without the PBTS. The OMS propellant transfer is assisted by bipropellant pumps mounted on the 1307 bulkhead, and the OMS pressurant transfer operates in a blowdown mode.

Figure A-29 shows the addition of gas compressors to assist in pressurant transfer from the OMS helium bottles and the addition of a QD purge network. The helium bottle in the payload bay will be used as the purge gas source with the QD purge residuals dumped through nonpropulsive vents. Figure A-30 shows waste disposal tanks included for the QD purge residuals.

4.4.2.2.2 Layout Drawing. The PBTS structural layout options for the propellant and pressurant tanks are presented in Figures 81 and 82. Both figures show the six 50-inch-diameter propellant tanks symetrically positioned in the structure. These tanks would have a total capacity of approximately 15,000 pounds of propellant. Both also have twelve 18-inch-diameter pressurant tanks positioned symetrically just outside the radially centered propellant tanks. Figure 81 shows two 40-inch-diameter pressurant tanks in the center of the structure, whereas Figure 82 has fourteen 18-inch-diameter pressurant tanks in the center of the structure. The configuration in Figure 81 requires more length to accommodate the two larger tanks and would, therefore, require more space in the payload bay. The structure in Figure 82 is the recommended layout for the PBTS. The structure would be mounted in the payload bay by four sill trunnions (two on each side) and one keel trunnion.

A system hardware weight tabulation is presented in Table 28. This table indicates the scar weight (470 pounds) required to perform OMS tankage scavenging; the airborne support equipment (ASE) weight, which is the plumbing, pumps, valves, etc., on the 1307 bulkheads; and the deployable docking boom. The payload bay tankage weight accounts for the propellant tanks, plumbing, helium tanks, and pallet structure that together form the entire tankage. Table 29 presents a weight breakdown of the OMS pod and aft fuselage scar plumbing and electrical equipment.

4.4.2.2.3 System Characteristics. Bipropellant scavenging will be accomplished by the transfer of excess propellants from the OMS tanks and from the PBTS. Propellant resupply may be accomplished through an umbilical ring mounted on the PBTS or through an umbilical boom mounted on the payload bay sill.

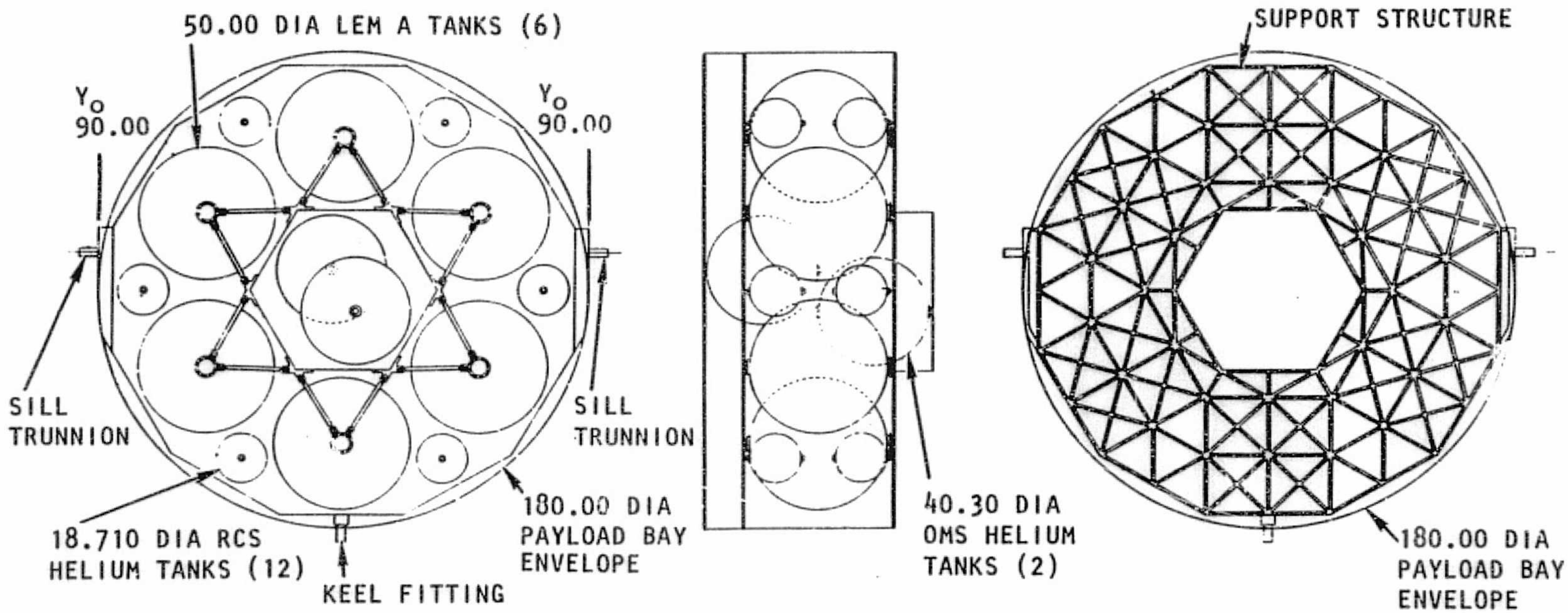
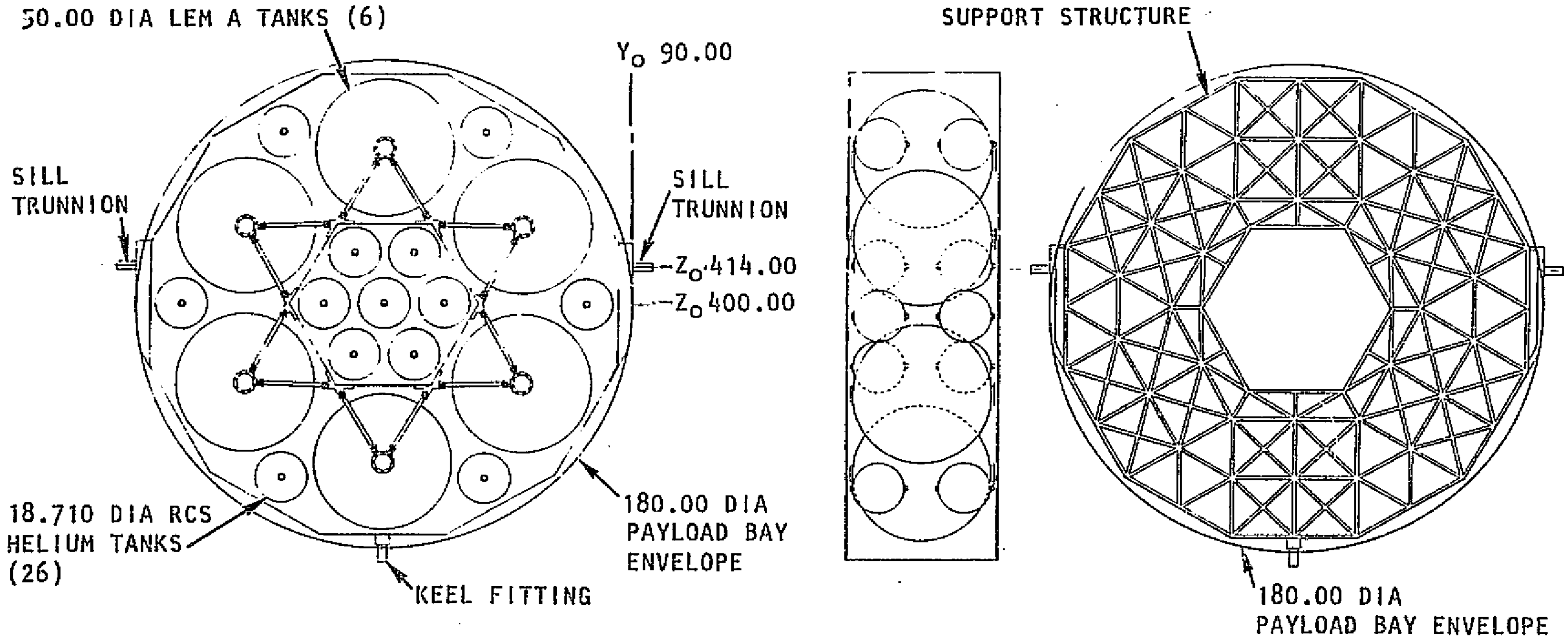


Figure 81. PBTS Layout--Option 1



VIEW LOOKING AFT
FORWARD SUPPORT STRUCTURE
OMITTED FOR CLARITY

VIEW LOOKING FORWARD
TANKS OMITTED FOR CLARITY

Figure 82. PBTS Layout--Option 2

Table 28. Storable Propellant Hardware System Weights

Payload Bay Length (in.)	Propellant Capacity (lb)	OMS Pod and Aft Fuselage Scar Weight (lb)	ASE Weight (lb)	Payload Bay Tankage Weight (lb)	Total Weight (lb)
37	871	470	930	2,497	3,897
53	8,935	470	930	3,147	4,547
68	15,156	470	930	4,168	5,568
92	27,656	470	930	4,415	5,815
113	37,141	470	930	6,209	7,609

Table 29. OMS Scar Weight Breakdown

Components	Weight (lb)
OMS pod additions (per pod)	
• Line from helium bottle to high-pressure quad valve panel	0.61
• Line from monomethylhydrazine (MMH) propellant tank helium inlet line to MMH ullage quad valve panel	1.39
• Line from nitrogen tetroxide (NTO) propellant tank helium inlet line to NTO ullage quad valve panel	3.53
• Three valve panels made up of four high-pressure helium solenoid valves	66.0
• Line to pod/vehicle interface flange from helium panel	0.61
• Line to pod/vehicle interface flange from MMH ullage panel	0.46
• Line to pod/vehicle interface flange from NTO ullage panel	0.48
• Three interface flanges, one high-pressure helium and two low-pressure (MMH ullage and NTO ullage)	0.60
• Line structural supports	3.1
• Valve panel(s) structure (3)	24.0
ROM/pod	100.78
With 10% margin	110.86
Subtotal--ROM/2 pods	221.72
Aft fuselage, payload bay, and T-4 umbilical additions	
• Six interface flanges, two high-pressure helium and four low-pressure (MMH ullage and NTO ullage)	1.2
• Six flex lines for pod installation alignment, two high-pressure helium and 4 low-pressure (MMH ullage and NTO ullage)	7.50
• High-pressure line from left to right helium flex lines	3.2

Table 29. OMS Scar Weight Breakdown (Cont)

Components	Weight (lb)
• High-pressure tee and line from high-pressure crossover line to payload bay interface flange	2.2
• High-pressure interface flange for helium in payload bay	0.20
• High-pressure ground servicing line to T-4 umbilical panel	1.5
• High-pressure ground servicing QD	0.50
• Low-pressure lines from left to right MMH and NTO ullage flex lines	4.20
• Low-pressure tees and lines from low-pressure crossover lines to payload bay interface flanges	2.90
• Low-pressure interface flanges for MMH and NTO ullages in payload bay	0.40
• Low-pressure ground servicing lines to T-4 umbilical panel	1.94
• Low-pressure ground servicing QD's	1.50
• Change crossfeed high-point bleed lines to larger diameters	1.92
• Crossfeed high-point bleed line interface flanges for MMH and NTO in payload bay	0.40
• Reroute ground servicing lines to T-4 right-hand umbilical panel	0
• Line structural supports	9.0
• 1307 bulkhead penetration structural beefup supports	1.2
• QD (T-4) panel structural supports	3.0
Aft fuselage ROM	42.76
With 10% margin	47.04
Electrical wiring, MDM's, control panels, heaters, and thermostats for aft pods, aft fuselage, and payload bay	183.91
With 10% margin	202.30
Plumbing weight impact (2 OMS pods + aft fuselage + payload bay)	268.76
Electrical weight impact (2 OMS pods + aft fuselage + payload bay)	202.30
Total weight impact	471.06

The storable bipropellant scavenging system is made up of several units connected by bolted flanges. A mission's particular resupply requirements will dictate whether some or all of these units will be connected to make up the scavenging system. These units include the PBTS, bipropellant pump network, QD purge system, waste disposal tanks, nonpropulsive vents, and a helium gas compressor network.

The PBTS consists of six 50-inch-diameter spherical propellant tanks (three fuel and three oxidizer) and twenty-six 18-inch-diameter spherical helium bottles. The nominal operating pressure of the propellant tank is approximately 250 psia, and the maximum nominal operating pressure of the helium bottles is 3,600 psia. Propellant transfer flow rates range from 0 to 15 gpm. Pressurant transfer could be accomplished by tank pressure equalization or with the assistance of the gas compressors.

4.4.2.2.4 Component Definition. A component legend for the PBTS schematic is presented in Figure 4. Table 30 lists the components of the PBTS. The table briefly describes each component and its purpose, gives its nominal operating pressure and temperature, and lists a vendor that could supply the component. The quantity of each type of component required in the table is for the complete PBTS shown in Figure 4. Because the pump is a new component that will require development, bipropellant pump operating characteristics are listed as follows:

- Fluids: NTO, MMH
- Flow rate: 0-15 gpm
- Head: 480 feet NTO, 800 feet MMH (ΔP = 300 psid)
- Electrical input: 115 V 3-phase 400 Hz or 28 Vdc
- Inlet pressure: 265 ± 40 psia
- Inlet temperature: +40 to +100°F
- Life: 1,000 hours over 10-year period (20 missions per year, 5 hours per mission for 10 years)
- Ambient pressure: 10^{-10} torr to 15.23 psia
- Ambient temperature: +40 to +100°F operating
+20 to +150°F nonoperating
- Flow control: variable flow rate

4.4.3 Test Plans

The main objective of the test program is to verify the operation and performance of the selected scavenging concepts at both the system and component levels to permit detailed design of the operational system with assurance that it will meet performance requirements.

Table 30. Primary Component List for
Storable Scavenging System

Component/Purpose	Approximate Operating Range		Vendor and MC No.	No. Required
	Pressure (psi)	Temperature (°F)		
Lift-off ball flow control valve (ac motor operated) Isolates and controls propellant flow to and from the tanks through the resupply loading/transfer system	350 max.	30 to 150	Parker-Hannifin MC284-0430	26
Solenoid valve • Isolates high-pressure gaseous helium or nitrogen from the propellant and pressurant resupply systems	200 to 4,000	20 to 150	Consolidated Controls Corp. MC284-0419	14
• Isolates and controls propellant ullage vapors and purge gases (low pressure)	350 max.	30 to 150	Consolidated Controls Corp. MC284-0420	32
Pressurant tanks • Store and provide pressurant via regulators to the propellant tanks • Storage and transfer high-pressure gaseous helium for on-orbit resupply	500 to 4,000	20 to 150	Brunswick MC282-0082	26
Propellant tanks Store and transfer bubble-free propellant for on-orbit resupply	350 max.	40 to 150		6
Fluid transfer disconnects (new component) Provide spacecraft-to-spacecraft interface for the transfer of propellant, pressurant, and resupply-generated waste products	700 max. (propellant)	N/A		4
	5,000 max. (pressurant)	N/A		1
	N/A (waste)	N/A		4

Table 30. Primary Component List for Storable Scavenging System (Cont)

Component/Purpose	Approximate Operating Range		Vendor and MC No.	No. Required
	Pressure (psi)	Temperature (°F)		
Dual pressure regulator Provide regulated source of pressurant to the propellant transfer system and the waste disposal system	4,000 to 500 (inlet)	-65 to 125	Fairchild-Stratos MC284-0418	2
Quad check valve Precludes upstream back-flow of pressurant and propellant vapors or liquid	370 max.	-65 to 125	Rocketdyne Division MC284-0481	2
Relief valve Prevents overpressurization of any portion of the system	315		-65 to 125 (cracking)	4
Dual burst disks Minimize loss of pressurant during normal system operation	330		-65 to 125 (rupture)	4
Filters				
• Pressurant	1,000 to 4,000 (inlet)	20 to 150		
• Propellant	350 max.	150 max.		
Flowmeter (propellant)	350 max.	150 max.		4
Nonpropulsive vent	N/A	N/A		4
Vapor bubble detector	350 max.	40 to 150		2

Table 30. Primary Component List for Storable Scavenging System (Cont)

Component/Purpose	Approximate Operating Range		Vendor and MC No.	No. Required
	Pressure (psi)	Temperature (°F)		
Bipropellant pump (new component) Permits on-orbit transfer of space-storable propellants	265 ± 40 (inlet)	-50 to 150		2
Gas compressor (new component) Permits on-orbit transfer of high-pressure helium	500 to 4,000 (inlet)	20 to 150		2

4.4.3.1 Cryogenic Scavenging Tests.

4.4.3.1.1 Demonstration Requirements. The major test/demonstration items needed to verify satisfactory scavenging system operation and performance are listed below:

- Verify receiver tank ground loading performance and procedures.
- Verify launch safety provisions.
- Verify fluid acquisition from ET and MPS during post-MECO spin.
- Verify post-MECO fluid transfer and fill of receiver tank (in payload bay), preferably without venting.
- Verify zero-g fluid acquisition and transfer from the receiver tank in the payload bay to a receiver tank in the OMV or at the Space Station.

4.4.3.1.2 Test/Demonstration Plan. To meet the predesign confidence levels desired for the scavenging system, it is considered necessary to test at least a subscale prototype of the system in orbit using actual LH_2 and LO_2 propellants. This is needed to verify the acquisition and transfer of cryo fluids in zero g, something which has never before been accomplished. An end-to-end test of the entire scavenging/delivery sequence from ground perchill through post-MECO scavenging and off-loading under zero-g would be included.

Prior to this overall test, post-MECO acquisition of cryogenic propellants from the ET and MPS under a 2-deg/sec pitch rate can readily be demonstrated by tapping flow from a selected drain point in the MPS plumbing and dumping it overboard.

Before the orbital tests, many basic component and system-level tests can be performed on the ground at g levels varying between +1 and -1, depending on hardware orientation. These can help to verify both filling and draining operations, especially the effect of heat leakage and any bubble formation on screen channel performance.

This general approach is considered to be the most cost-effective strategy for developing and testing the cryogenic propellant scavenging system. Another option considered was that of using a small plexiglass subscale system model on the orbiter aft flight deck, using water as a safe test fluid. Cryogenic boiling conditions could be simulated by operating at a low pressure (f 0.5 psia) and using electric heaters as a heat source. The marked difference, however, between the properties of water and LH_2 and LO_2 such as surface tension, vapor density, wicking, and wetting performance, would cast considerable doubt on the validity of test results, both good or bad. Second, the cost of modeling, scaling, designing, and qualifying such a portable system test package is not small. third, a somewhat similar water transfer experiment was already scheduled for an STS flight in late 1984 and will provide some of the data obtainable by this method.

A major question in designing the scavenging/delivery system is whether the capillary acquisition screen channels can function satisfactorily under conditions of heat input, external boiling, and possible internal vapor generation. Figure 83 shows a proposed setup for testing a representative section of screen gallery in a transparent (glass) LH_2 Dewar to verify acquisition under conditions of high heat leakage through the tank wall. As shown, the channel is used to drain the Dewar against a negative g head until the point of bubble breakthrough is reached (at the top of the wetted screen channel). The effects of helium versus hydrogen gas ullage on self-wicking and dryout of the screen and the effects of depressurization and repressurization on bubble behavior also can be evaluated. It is believed that any vapor bubbles left in the screen channel after filling can be condensed and collapsed by moderate pressurization. The above tests will be repeated with LO_2 (or LN_2 as a substitute fluid if safety is a problem).

The data obtained from this type of basic component ground testing is expected to be valuable and cost-effective for later design of the prototype scavenging tanks and associated systems. Bubble detectors and mass flowmeters are long-lead items that can be included in ground component testing.

It is assumed that automated cryogenic disconnects for the final transfer of cryogenic propellants from the payload bay to the user vehicle will be developed under the OTV program before they are needed for handling scavenged propellants. Therefore, such disconnects are not expected to be demonstrated until final full-scale end-to-end scavenging operations are performed.

Ground testing of subscale tankage and associated plumbing can simulate much of the launch pad and flight operations and provide valuable data for use

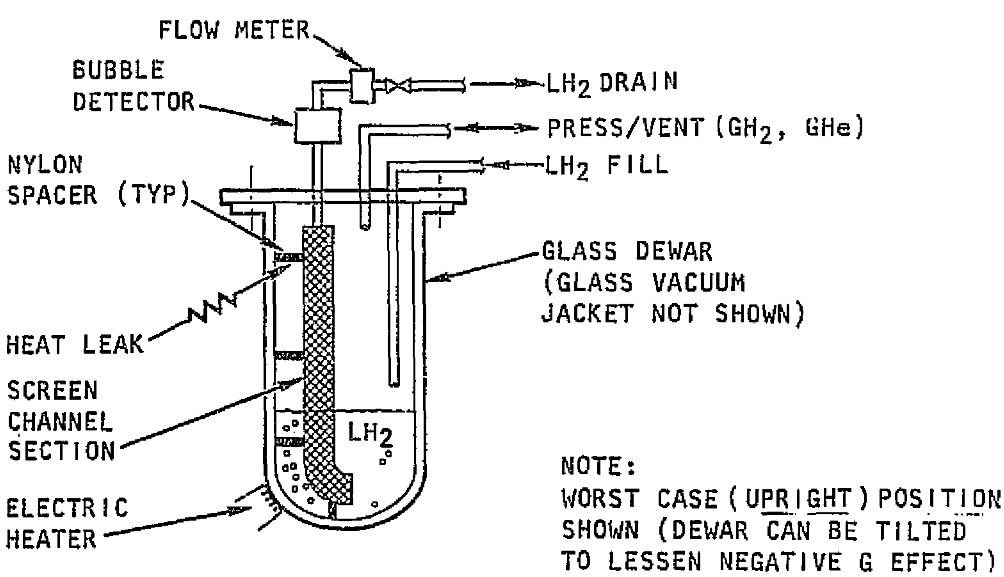


Figure 83. Screen Channel Component Test

in the full-scale system design. Figure 84 shows a schematic of the recommended test setup using a subscale toroidal LH_2 tank, with associated plumbing and an internal mixer pump TVS installed. The central LO_2 tank system is not shown for reasons of clarity but should be tested simultaneously with the LH_2 tank to simulate thermal conditions. LN_2 may be substituted for LO_2 if dictated by safety considerations.

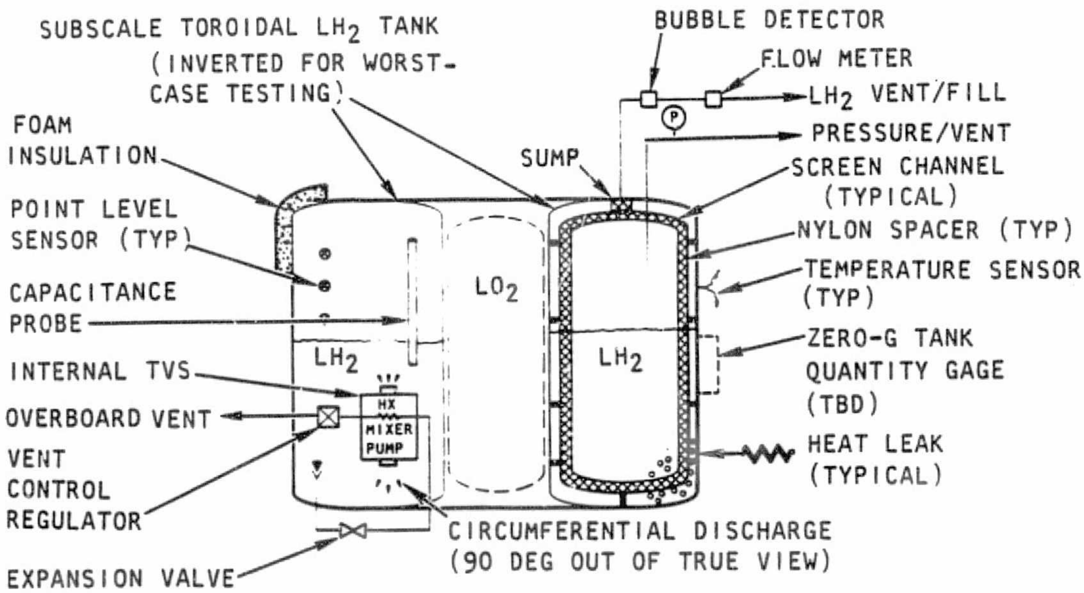
The scope of ground system testing will include:

- Childown, ground fill, and replenish control, as performed on the launch pad

- Fluid acquisition and draining of LH_2 and LO_2 against negative gravity to the limit of screen gallery bubble breakthrough. This will be tested under various conditions of ullage pressure, ullage gas composition (helium and hydrogen), ambient heat leakage, liquid drain rate, and operation of the mixer pump TVS unit. Bubble-free draining at the tank sump against a foot or more of negative g liquid head would be a strong predictive indicator of successful draining in zero g to the desired expulsion efficiency.

- Simulation of emergency dumping from STS

- Effectiveness of the mixer pump TVs in controlling tank temperature and pressure and reducing boiling inside the tank. This type of device has been developed by General Dynamics under the Centaur program, and no need for separate component testing prior to installation into the system test setup is anticipated.



NOTE: WORST-CASE POSITION SHOWN (TANK CAN BE TILTED TO LESSEN NEGATIVE G EFFECT)

Figure 84. Subscale LH_2 Receiver Tank Ground Test

- Propellant gaging. It is doubtful that the capacitance probe shown in Figure 84 will perform satisfactorily in the low-g environment of post-MECO pitch rotation and later zero-g conditions, if accurate monitoring of the receiver tank quantity during orbital fluid transfer operations is desired, it will be necessary to provide a true zero-g gaging system (e.g., radioactive or infrasonics). To gain operating experience with such a gaging system, it would be useful to install it in the ground system and check it against the output of the point sensors and capacitance probe. Final verification would be accomplished during flight testing.

An alternative to the use of a zero-g gaging system would be to closely monitor receiver tank pressure during unvented zero-g fill operations. If a reasonable mixing flow is maintained in the tank, a sharp increase in tank pressure will occur as a full condition is approached (say 97 percent), at which time transfer can be slowed and terminated. A coarse indication of tank quantity at lower levels could be obtained from flowmeter data. A choice between this method and a true zero-g gaging system should be made during follow-on design studies of the scavenging system.

It is recommended that the ET and MPS fluid acquisition method be tested early in the demonstration program to obtain long lead time on any unforeseen problems that might appear. This can be accomplished by simply pitching the orbiter and attached ET (after MECO) at the planned rate of f 2 deg/sec and venting fluid overboard from the MPS fill and drain valve through the existing relief port. Bypass plumbing around the MPS relief valves would be required (Figure 85).

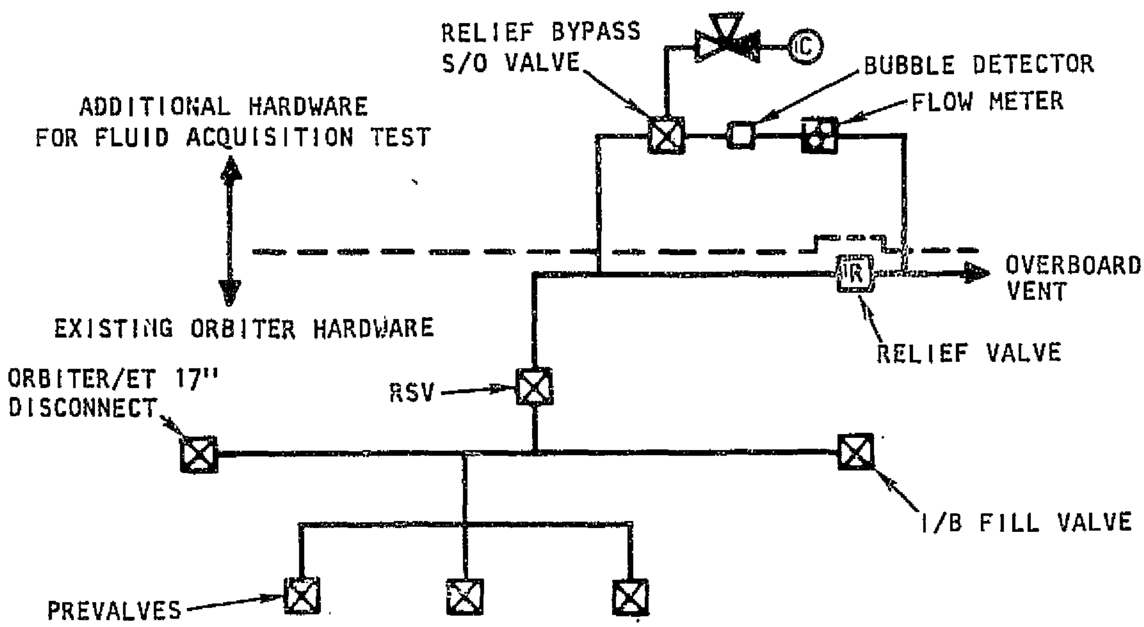


Figure 85. ET/Orbiter Spinup and Fluid Acquisition Test--LO_2 and LH_2 Systems

Flow rates would be measured and integrated to obtain the total quantity of propellants acquired and dumped. Quality meters or bubble detectors should be provided in the relief line to determine the time of vapor ingestion and the time at which the transfer process should be terminated.

A flight test will be conducted to demonstrate end-to-end scavenging operation of the selected system from partial filling of the receiver tanks at the launch pad to transfer, filling, and draining of the user tank on the OMV, OTV or Space Station. Two identical subscale cryogenic receiver tanks could be installed side by side in the payload bay, with plumbing systems equivalent to those shown in the detailed schematics (Figures 1 and 2.) Testing procedures would include:

1. Fill the No. 1 receiver tank on the launch pad and draining to 25 percent prior to lift-off.

2. Spinup the mated ET/orbiter to 2 deg/sec pitch rate after MECO.

3. Scavenge LH_2 and LO_2 from the ET and MPS to fill the No. 1 receiver tank (without venting receiver if possible).

4. Despin orbiter to zero-g condition.

5. Childown and fill No. 2 receiver (from No. 1) at zero g to simulate transfer to OMV or Space Station.

6. Drain No. 2 tank and vent fluids overboard through flowmeters to determine filling efficiency and expulsion efficiency with quick zero-g childown.

An option that should be evaluated is the use of two full-scale receiver tanks (of different operational sizes) instead of two subscale tanks. This would demonstrate full-scale scavenging, provide tanks for later operational use, and save the cost of subscale tanks. One-g system ground testing might also be convenient with the smaller of the two sizes.

If satisfactory zero-g cryogenic propellant transfer data should become available from another test program, it may be possible to eliminate the No. 2 receiver tank and vent directly overboard from Tank 1. In any case, it is still recommended that the flight test of Tank 1 be retained due to the special conditions of low transfer pressure, STS pitch rate, ET/orbiter heating, and the possibility of two-phase transfer flow, which are all unique to the scavenging process.

4.4.3.1.3 Resource Requirements for Cryogenic Scavenging Test/Demonstration. Based on the planned scope of testing, the following resource requirements have been identified:

- Facilities
  - Cryogenic fluids lab with test cell

- Machine shop
- Prototype shop
- Engineering office space
- Modification and checkout areas at the Kennedy Space Center (KSC)

- Man-hours

- Cryofluids analysis/design engineering	3,000
- Cryofluids test engineer	3,000
- Cryofluids test technician	3,000
- STS project engineer	3,000
- STS safety engineer	1,000
- STS fluid system technicians	3,000
- STS fluid systems engineer	2,000
- Machinist	500
- STS software programmer	1,000
- STS payload technician	250
- Flight mission specialist	500
- Ground mission support	200
- STS payload designer	1,000
- STS console design engineer	1,000
Total	22,450

- Material

- Raw stock (tubing, sheet, wire)	$10,000
- Valves and miscellaneous plumbing components	$200,000
- Miscellaneous instrumentation and controls	$200,000
- Mixer pump TVS	$200,000
- Zero-g tank gaging system (based on nucleonic type)	$500,000
Total	$1,110,000

- Auto Comp

Total	5 hours

4.4.3.1.4 Cryo Test Schedule. The preliminary schedule shown in Figure 86 is considered to have the minimum time span consistent with cost-effectiveness (assuming a go-ahead at the start of FY 1986). A reasonable allowance is included for unforeseen technical problems. The schedule can be accelerated, if desired, but at a corresponding increase in total cost.

The test sequence is based on starting system ground test design shortly after critical component verification (i.e., screen channel performance under boiling conditions). STS spinup and vent testing is scheduled for completion prior to the start of final flight system test design.

4.4.3.2 <u>Hypergolic Propellant Scavenging Tests</u>. Three major components of this system can be identified as requiring special developmental testing to prove concept feasibility. These are the propellant pump, the helium compressor, and fluid disconnect. Since the development approach essentially is the same for all three components, they are all covered in this section. The resource requirements and test schedules, however, are identified for each component.

4.4.3.2.1 Demonstration Requirements. The major test/demonstration items required to verify satisfactory component operation and performance are listed below:

- Evaluate component design concept feasibility with a workhorse-type test unit.

- Verify operation and performance of a flight-weight prototype test unit to flight application requirements.

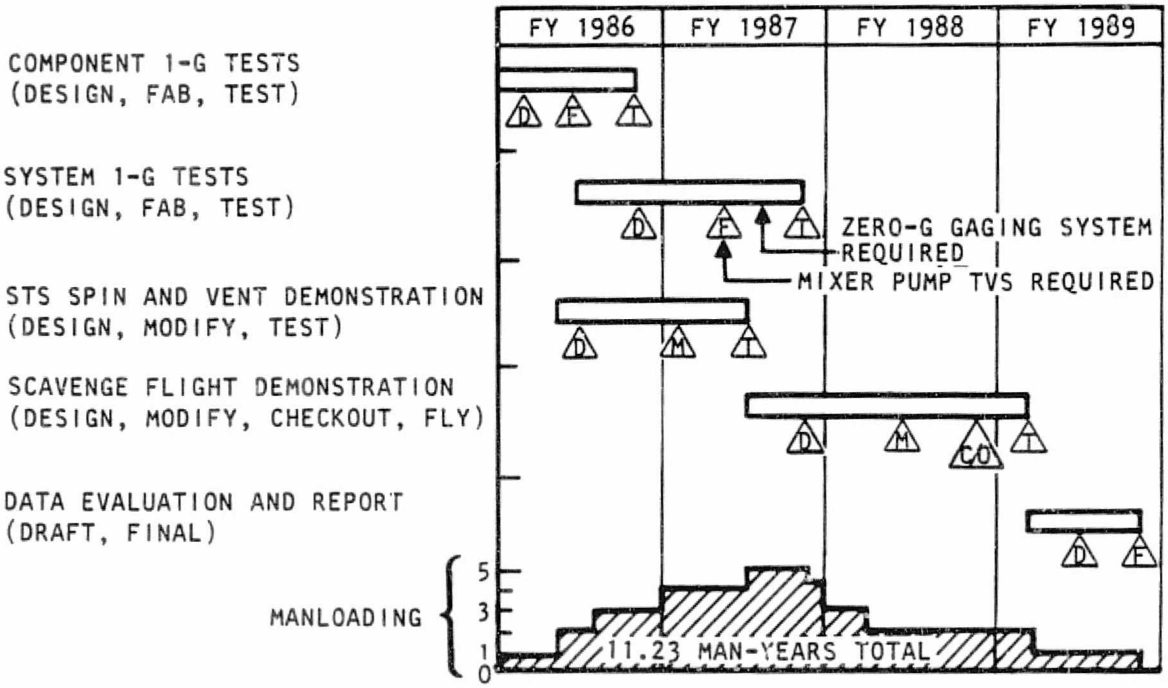


Figure 86. Preliminary Cryo Test Schedule

4.4.3.2.2 Test Plan. The scope of the test program will encompass the following:

- Breadboard-type tests using referee fluids and a development (work-horse) component test unit will provide the basic data base for component functional design evaluation.

- Development tests using referee fluids, propellants, and prototype flight-weight test components will provide the basis for verification of the component design concept feasibility. These tests will assess operation and performance capabilities when subjected to flight environments, ground servicing, and fluid transfer usage. In addition, component life will also be demonstrated.

4.4.3.2.3 Resource Requirements for Scavenging Tests. Based on the planned test program, the resource requirements in Table 31 have been identified.

4.4.3.2.4 Hypergolic Test Schedule. The preliminary schedule for the propellant scavenging system tests is presented in Figure 87 (assuming go-ahead at the start of FY 1986).

Table 31. Scavenging Test Resource Requirements

Resource	Compressor	Disconnect	Pump
Facilities			
• Hypergolic lab and test cell		X	X
• Machine shop	X	X	X
• Referee fluids lab		X	X
• High-pressure gas test area	X		
• Vacuum test cell	X		X
• Engineering office space	X	X	X
Man-hours			
• Analysis engineer	2,000	500	1,000
• Support engineering	1,000	1,000	1,000
• Test engineer	2,000	1,000	1,000
• Technician	1,000	1,000	1,000
• Machinist	200	200	200
• Data engineer	1,000	200	1,000
• Test cell crew	1,000	1,000	1,000
• Computer support	200		
• Design engineering	5,000	3,000	3,000
	13,400	7,900	9,200
Material (Dollars)			
• Raw stock (tubing, sheet, wire)	2,000	4,000	2,00
• Valves and miscellaneous plumbing components	20,000	30,000	20,000
• Miscellaneous instrumentation and controls	30,000	10,000	20,000
• Tankage for transfer tests	20,000	2,000	10,000
• Component test units* (two for breadboard tests, two for development tests)	160,000	160,000	160,000
Auto comp (hr)	3	1	1

*To be provided by outside component supplier.

4.5 DEL VERABLE PROPELLANT

The system weights presented in Section 4.4.1.2 for both cryogenic and storable propellant systems were used to update the propellant availability analysis performed in the initial part of the study. These results were presented in Section 4.1.3. The amount of propellant that can actually be delivered to the user starts with the propellant availability values which must be adjusted to account for the propellant losses associated with trapped propellant and heat loads.

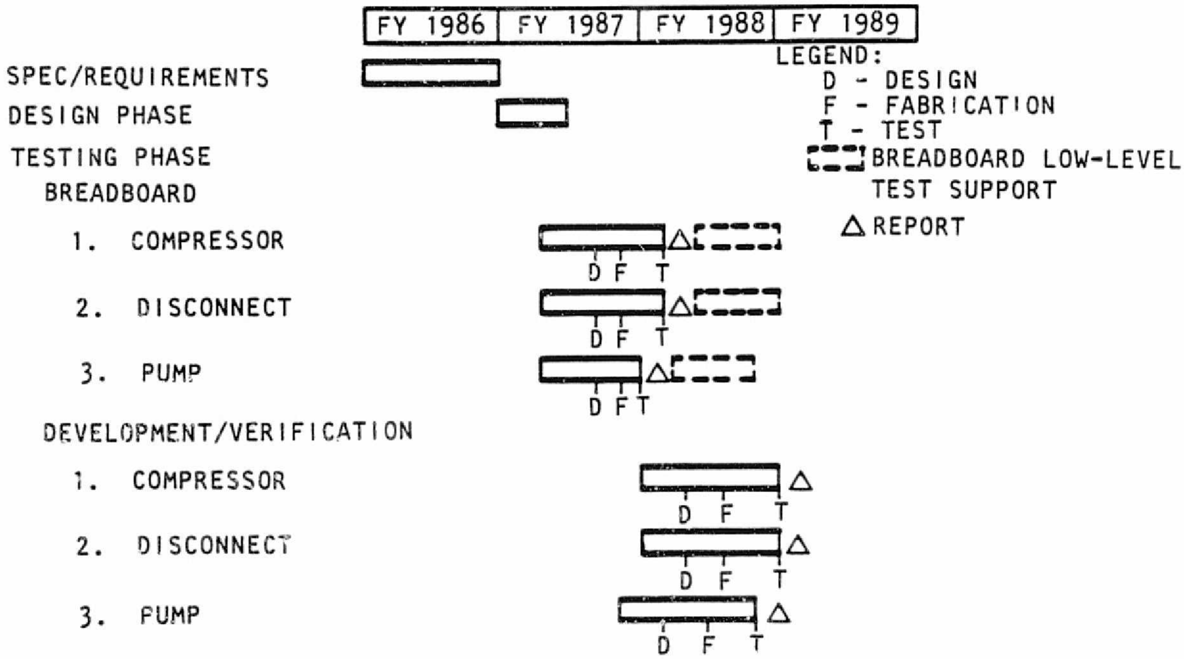


Figure 87. Preliminary Hypergolic Propellant Scavenging Test Schedule

4.5.1 Deliverable Cryogenic Propellant

Propellant is trapped in the MPS and payload bay tankage. There are 7,340 pounds of reserve propellant above the orbiter/SSME interface at MECO. Of this, 2,525 pounds of LO_2 and 158 pounds of LH_2 are trapped below the orbiter prevalves. Additionally, 350 pounds of LH_2 is isolated in the MPS when ullage gas pullthrough into the feed line siphon has occurred and the gas is transported to the MPS transfer port, thus preventing bubble-free propellant from being supplied to the payload bay tankage. The propellant trapped in the payload bay tankage when vapor pullthrough occurs in the CAS has been identified by analysis as 1 percent for LO_2 and 3 percent for LH_2.

The heat loads described in Section 4.2.5 must also be evaluated. The TVS and compact heat exchanger concept selection evaluated in Section 4.2.6 requires 36.6 lbm/hr of vent flow rate. For the two mission altitudes and scavenging portion of the mission completion times, this yields 565 pounds of vent LH_2 for the 160-nautical mile mission and 296 pounds for the 250-nautical mile mission. The 160 nautical mile mission value includes losses while the OMV transports the propellant to the Space Station at 250 nautical miles.

The resulting deliverable quantities when all losses are included are presented in Table 32 for various numbers of tank size options.

4.5.2 Deliverable Storable Propellant

The trapped propellant quantities for the payload bay tankage have been defined from previous experience as 5 percent. The data presented in Table 32

Table 32. Scavenged Propellant Quantities

System	Number of Tank Sizes	Number of Scavenging Flights	Deliverable Propellant (klb)
Cryogenic propellant	1	69	926
	2	89	1,042
	3	89	1,182
	4	89	1,220
	5	89	1,247
	15 (max.)	93	1,321
Storable propellant	1	165	1,403
	2	165	1,491

include the adjustments required to the propellant availability quantities to account for the trapped propellants. The data also indicates the effects of having more than one tank size available when mission performance and payload bay space are available.

5. REFERENCES

1. Brux, A.F. Suborbital Recovery of External Tank Propellants. Rockwell International, SSD 81-0191 (May 1982).

2. Sterbentz, W. H. Liquid Propellant Thermal Conditioning System. Lockheed Missiles and Space Company, NASA CR-72113, LMSC-A839783.

3. Airborne Cryogenic Propellant Tanks Vent Systems Critical Design Review Backup Material, Book 1, Shuttle/Centaur. General Dynamics Convair Division (November 1983).

4. Cook, R. E. Propulsion Systems Optimization Interorbital Transfer Vehicle. North American Rockwell, NAS7-200 (August 8, 1973).

5. Sorenson, G. L., et al. A Zero-Gravity Thermodynamic Vent System for the Shuttle/Centaur Hydrogen Tank. Sundstrand Corporation, General Dynamics Convair Division.

6. Crain, B. J., and K. J. Bell. "Forced Convection Heat Transfer to a Two-Phase Mixture of Water and Steam in a Helical Coil," AICME: Symposium Series, No. 131, Vol. 69.

7. Moses, R. A. LH_2 Acquisition Channel Tests - Summary Report. Rockwell International, IR&D Project 28604 (1982).

8. Schwartz, R. Saturn S-II Advanced Technology Studies: Cryogenic Acquisition and Transfer. North American Rockwell, Contract NAS7-200, Supplemental agreement 2049, (December 21, 1971).

9. Lozano, M.E. "Capillary Channel Propellant Acquisition Performance Studies for LH_2/LOX Tanks in Low-G Environment." Rockwell International, Internal Letter P&PP/IR&D/84-008, (June 13, 1984).

10. Armour, J. C., et al. "Fluid Flow Through Woven Screens," AICHE Journal, Vol. 14, No. 3 (May 1968).

11. Cady, E. C. "Study of Thermodynamic Vent and Screen Baffle Integration for Orbital Storage and Transfer of Liquid Hydrogen." McDonnell Douglas, NASA CR-134482, MDC G-4798 (August 1973).

PRECEDING PAGE BLANK NOT FILMED

APPENDIX

A.1 FLIGHT MANIFEST

Flight manifests for 1991-2000 are presented in this section. The following notes are used in the comments column of the tables:

1. May add ancillary payloads and/or STS operator-chargeable weight.

2. Upper stages may be deployed in the 160- 250-nautical mile range.

3. Mission not dependent on altitude (MPS).

FLIGHT MANIFEST

1991

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
91-1	6005	Large Obs NASA	60	41000	6500	270	(1) DI to 270, Retrieve ESA/MPS @ 160 before return
	1506	S/L ESA/MPS	(30)	0	5000	160	
				41000	11500		
91-2	6006	Large Obs NASA	60	54100	0	378	(1) DI to alt
91-3	7018	Planetary NASA	60	65000	8690	160	(2) DI to alt
91-4	3004	GEO Plat NASA	60	60000	8690	160	(1) (2) DI to alt
91-5	4019	LEO Stat NASA	26	28000	13940	250	(1) DI to alt
	2029	LEO Plat NASA	15	10000	10000		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			49.5	46600	32540		
91-6	4019	LEO Stat NASA	26	28000	13940	250	(1) DI to alt
	2029	LEO Plat NASA	15	10000	10000		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			49.5	46600	32540		
91-7	4019	LEO Stat NASA	26	28000	13940	250	DI to alt
	2708	LEO Plat Com'l/MPS	15	10000	10000		
		Docking Mod	7	5000	5000		
		STS Op.Wt	0	3600	3600		
		Dyn Env	1.5	0	0		
			49.5	46600	32540		

FLIGHT MANIFEST

1991

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
				UP (LB)	DOWN (LB)		
91-8	4019	LEO Stat NASA	26	28000	13940	250	DI to alt
	2708	LEO Plat Com'l/MPS	15	10000	10000		
		Docking Mod	7	5000	5000		
		STS Op Wt	0	3600	3600		
		Dyn Env	1.5	0	0		
			49.5	46600	32540		
91-9	4010	LEO Stat NASA	26	40000	0	250	(2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		STS Op Wt	0	3600	3600		
		Dyn Env	1.5	0	0		
			46	64470	12200		
91-10	4008	LEO Stat NASA	49	40000	0	250	(1) DI to 250
		STS Op Wt	0	3600	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1	0	0		
			57	48600	8600		
91-11	2021	LEO Splat NASA	30	20000	20000	250	(1) (2) DI to 250
	9006	Dod PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			50	44470	32200		
91-12	2507	LEO Splat Foreign	30	20000	20000	250	(1) (2) DI to 250
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			50	44470	32200		

FLIGHT MANIFEST

1991

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
91-13	8725 Intelsat Comm	26.6	30000	2790	160	(2)
	9006 Dod PAM-D	11.5	15870	3600		
	8703 SBS Comm	8	10100	2510		
	STS Op Wt (4+1.5)	0	3600	3600		
	Dyn Env	5.5	0	0		
		51.6	59570	12500		
91-14	8725 Intelsat Comm	26.6	30000	2790	160	(2)
	8523 Columbia Comm	8	10100	2510		
	8529 Foreign Comm	8	10100	2510		
	1005 S/L NASA MPS	0	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env (4+2)	6	0	0		
		48.6	58800	16410		
91-15	8725 Intelsat Comm	26.6	30000	2790	160	(2)
	8523 Columbia Comm	8	10100	2510		
	8702 RCA Comm	8	10100	2510		
	STS Op Wt	0	3600	3600		
	Dyn Env (4+1.5)	5.5	0	0		
		48.1	53800	11410		
91-16	8528 China Comm	30	17250	4590	160	(2)
	8500 Canada Comm	8	13750	2510		
	1504 S/L TSS Italy	10	7640	7640		
	STS Op Wt	0	3600	3600		
	Dyn Env (4+1.5)	5.5	0	0		
		53.5	42240	18340		

FLIGHT MANIFEST

1991/1992

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
91-17	8528 China Comm 8502 Indonesia Comm 1016 S/L NASA MPS STS Op Wt Dyn Env (4+1.5)	30 8 10 0 5.5 53.5	17250 10100 9000 3600 0 39950	4590 2510 9000 3600 0 19700	160	(2) (3)
92-1	1033 S/L NASA MPS STS Op Wt	56 0 56	27500 3600 31100	27500 3600 31100	160	(3) DI to alt
92-2	4021 LEO Stat NASA/E.Obs	60	40000	32000	250	(1) DI to alt
92-3	6001 Large Obs NASA	60	58040	17100	320	(1) DI to alt
92-4	7012 Planetary NASA	60	65000	8690	160	(2) DI to alt
92-5	7027 Planetary NASA	60	65000	8690	160	(2) DI to alt
92-6	4019 LEO Stat NASA 2029 LEO Plat NASA Docking Mod Dyn Env STS Op Wt	26 15 7 1.5 0 49.5	28000 10000 5000 0 3600 46600	13940 10000 5000 0 3600 32540	250	(1) DI to alt
92-7	4019 LEO Stat NASA 2027 LEO Plat NASA/P&A Docking Mod STS Op Wt Dyn Env	26 15 7 0 1.5 49.5	28000 10000 5000 3600 0 46600	13940 10000 5000 3600 0 32540	250	DI to alt

FLIGHT MANIFEST

1992

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
92-8	4019 LEO Stat NASA	26	28000	13940	250	DI to alt
	2029 LEO Plat NASA	15	10000	10000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		49.5	46600	32540		
92-9	4019 LEO Stat NASA	26	28000	13940	250	DI to alt
	2708 LEO Plat Com'l/MPS	15	10000	10000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		49.5	46600	32540		
92-10	4023 LEO Stat NASA/L.Sci	30	20000	20000	250	(1) DI to 250
	2708 LEO Plat Com'l	15	10000	10000		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		53.5	38600	38600		
92-11	4022 LEO Stat NASA/P&A	30	20000	20000	250	(1) DI to 250
	2025 LEO Plat NASA/E.Obs	15	10000	10000		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		53.5	38600	38600		
92-12	8019 F-F NASA/P&A	30	19000	0	160	(1) DI to alt
	8511 W. Germany/Other	15	15000	0		
	Dyn Env (4+1)	5	0	0		
	STS Op Wt	0	3600	3600		
		50	37600	3600		

FLIGHT MANIFEST

1992

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
92-13	8035 F-F NASA Comm	30	17250	4590	160	(1) (2) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	8503 Arab Comm	8	10100	2510		
	Dyn Env (4+1.5)	5.5	0	0		
	STS Op Wt	0	3600	3600		
		55	46820	14300		
92-14	8008 F-F NASA P&A	20	17250	4590	160	(1) (2) (3) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	8702 RCA Comm	8	10100	2510		
	8300 F-F GEO Civil E.Obs	8	10100	2510		
	1005 S/L NASA MPS	5	6000	6000		
	Dyn Env (4+2.5)	6.5	0	0		
	STS Op Wt	0	3600	3600		
		59	62920	22810		
92-15	9006 DoD PAM-D	11.5	15870	3600	160	(1) (2) DI to alt
	8729 US Com'l Comm	8	13750	2510		
	8529 Foreign Comm	8	10100	2510		
	8702 RCA Comm	8	10100	2510		
	Dyn Env (4+2)	6	0	0		
	STS Op Wt	0	3600	3600		
		41.5	53420	14730		
92-16	9006 DoD PAM-D	11.5	15870	3600	160	(1) (2) DI to alt
	8729 US Com'l Comm	8	13750	2510		
	8727 Hughes Comm	8	10100	2510		
	8537 ESA/MPS F-F	10	6900	3600		
	Dyn Env (4+2)	6	0	0		
	STS Op Wt	0	3600	3600		
		43.5	50220	15820		

FLIGHT MANIFEST

1992

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
92-17	8528	China Comm	30	17250	4590	160	(1) (2) (3) DI to alt
	8729	US Com'l Comm	8	13750	2510		
	1016	S/L NASA MPS	10	9000	9000		
		Dyn Env (4+1.5)	5.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	43600	19700		
92-18	2021	LEO Plat NASA	30	20000	20000	250	DI to 250
	8008	F-F NASA/P&A	20	17250	4590		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			58.5	45850	33190		
92-19	1022	TSS/NASA	10	7640	7640	160	(1) (2) DI to alt
	8525	F-F Mexico Comm	8	10100	2510		
	8731	US Com'l Comm	8	10100	2510		
	8502	Indonesia Comm	8	10100	2510		
		Dyn Env (4+2)	6	0	0		
		STS Op Wt	0	3600	3600		
			40	41540	18770		
92-20	1504	S/L Italy TSS	10	7640	7640	160	(1) (2) DI to alt
	8732	US Com'l Comm	8	10100	2510		
	8300	GEO Civil E.Obs	8	10100	2510		
	8727	Hughes Comm	8	10100	2510		
	8539	F-F Foreign Comm	8	10100	2510		
		Dyn Env (4+2.5)	6.5	0	0		
		STS Op Wt	0	3600	3600		
			48.5	51640	21280		

FLIGHT MANIFEST

1993

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
93-1	2023 LEO Plat NASA/LSS	60	40000	32000	250	(1) DI to 250
93-2	2707 LEO Plat Com'l MPS	60	40000	32000	250	(1) DI to 250
93-3	6002 Large Obs NASA	60	57320	32000	320	(1) DI to alt
93-4	7033 Planetary NASA	60	65000	8690	250	(2) Assume DI to 250,
	1506 S/L ESA/MPS	(30)	0	5000		then retrieve ESA/
		60	65000	13690		MPS at 160
93-5	4011 LEO Stat NASA Other	44	40000	0	250	(1) (3) DI to 250
	1005 S/L NASA MPS	5	6000	6000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		57.5	54600	14600		
93-6	4011 LEO Stat NASA Other	44	40000	0	250	(1) DI to 250
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1	0	0		
		52	48600	8600		
93-7	2706 LEO Plat Com'l Other	40	25000	0	250	(1) (2) DI to 250
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		60	49470	24470		

FLIGHT MANIFEST

1993

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
93-8	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 250
	2030 LEO Plat NASA Tech	15	10000	10000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		49.5	58600	32540		
93-9	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 270
	8533 F-F Japan P&A	15	10000	0		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		42.5	58600	22540		
93-10	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 250
	2708 LEO Plat Com'l	15	10000	10000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		49.5	58600	32540		
93-11	2021 LEO Plat NASA	30	20000	20000	250	(1) DI to 250
	2708 LEO Plat Com'l	15	10000	10000		
	Docking Mod	7	5000	5000		
	STS Op Wt	0	3600	3600		
	Dyn Env	1.5	0	0		
		53.5	38600	38600		

FLIGHT MANIFEST

1993

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
93-12	4012 LEO Stat NASA Other	15	40000	0	250	(1) (2) DI to 250
	8528 China Comm	30	17250	4590		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
		53.5	62250	9590		
93-13	8710 F-F Comsat Comsat	8	13750	3600	160	(1) (2) (3) DI to alt
	8729 US Com'l Comm	8	13750	3600		
	8702 RCA Comm	8	10100	3600		
	8706 F-F AT&T Comm	8	10100	3600		
	1016 S/L NASA MPS	10	9000	9000		
	STS Op Wt	0	3600	3600		
	Dyn Env (4+2.5)	6.5	0	0		
		48.5	60300	27000		
93-14	9006 DoD PAM-D	11.5	15870	3600	160	(1) (2) DI to alt
	8020 F-F NASA P&A	30	19000	0		
	8729 US Comm	8	13750	3600		
	STS Op Wt	0	3600	3600		
	Dyn Env (4+1.5)	5.5	0	0		
		55	52220	10800		
93-15	9006 DoD PAM-D	11.5	15870	3600	160	(1) (2) DI to alt
	1504 S/L Italy TSS	10	7640	7640	124	
	8539 F-F Foreign Comm	8	10100	3600		
	8710 F-F Comsat Comsat	8	13750	3600		
	8525 F-F Mexico Comsat	8	10100	3600		
	Dyn Env (4+2.5)	6.5	0	0		
	STS Op Wt	0	3600	3600		
		52	61060	25640		

FLIGHT MANIFEST

1993/1994

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
93-16	9006 DoD PAM-D	11.5	15870	3600	160	(1) (2) DI to alt
	1504 S/L Italy TSS	10	7640	7640		
	8729 US Com'l Comm	8	13750	3600		
	8713 F-F W.Union Comm	8	10100	3600		
	8727 Hughes Comm	8	10100	3600		
	Dyn Env (4+2.5)	6.5	0	0		
	STS Op Wt	0	3600	3600		
		52	61060	25640		
93-17	8538 F-F Foreign E.Obs	8	10100	3600	160	(1) (2) DI to alt
	8526 F-F Canada Comm	20	23000	3600		
	8702 RCA Comm	8	10100	3600		
	8713 F-F W.Union Comm	8	10100	3600		
	Dyn Env (4+2)	6	0	0		
	STS Op Wt	0	3600	3600		
		50	56900	18000		
94-1	2022 LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
94-2	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
94-3	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
94-4	4021 LEO Stat NASA/E.Obs	60	40000	32000	250	(1) DI to alt
94-5	6002 Large Obs NASA	60	57320	32000	320	(1) DI to alt
94-6	6002 Large Obs NASA	60	57320	32000	320	(1) DI to alt

FLIGHT MANIFEST

1994

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
94-7	7028	Planetary NASA	60	65000	5600	160	(2) DI to alt
94-8	7034	Planetary NASA	60	65000	5600	160	(2) DI to alt
94-9	2033	LEO Plat NASA Other	30	20000	20000	250	(1) (2) DI to 250
	9006	DoD PAM-D	11.5	15870	3600		
	8523	Columbia Comm	8	10100	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
		STS Op Wt	0	3600	3600		
			58.5	54570	35800		
94-10	2507	LEO Plat Foreign	30	20000	20000	250	(1) (2) DI to 250
	9006	DoD PAM-D	11.5	15870	3600		
	8702	RCA Comm	8	10100	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
		STS Ob Wt	0	3600	3600		
			58.5	54570	35800		
94-11	4020	LEO Stat NASA Other	26	40000	13940	250	(2) (3) DI to 250
	8713	F-F W.Union Comm	8	10100	3600		
	1016	S/L NASA MPS	10	9000	9000		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
			53	64100	31540		
94-12	4020	LEO Stat NASA Other	26	40000	13940	250	(2) (3) DI to 250
	8713	F-F W.Union Comm	8	10100	3600		
	1029	S/L NASA MPS	15	9750	9750		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
			58	64850	32290		

FLIGHT MANIFEST

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
94-13	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to 250
	8710 F-F Comsat Comsat	8	13750	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		42.5	62350	26140		
94-14	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to 250
	8710 F-F Comsat Comsat	8	13750	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		42.5	62350	26140		
94-15	9006 DoD PAM-D	11.5	15870	3600	250	(1) (2) DI to 250
	AOTV	29	11930	6000		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		49	36400	18200		
94-16	9006 DoD PAM-D	11.5	15870	3600	250	(1) (2) DI to 250
	AOTV	29	11930	6000		
	Docking Mod	7	5000	5000		
	Dyn Mod	1.5	0	0		
	STS Op Wt	0	3600	3600		
		49	36400	18200		

FLIGHT MANIFEST

1994

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
94-17	3003	GEO Plat NASA Other	21.4	15400	1400	160	(1) (2) DI to alt
	8707	F-F GTE Comm	8	10100	3600		
	8539	F-F Foreign Comm	8	10100	3600		
	8706	F-F AT&T Comm	8	10100	3600		
		Dyn Env	6	0	0		
		STS Op Wt	0	3600	3600		
			51.4	49300	15800		
94-18	3700	GEO Plat Com'l Comm	21.4	15400	1400	160	(1) (2) DI to alt
	8731	US Com'l Comm	8	10100	3600		
	8710	F-F Comsat Comm	8	13750	3600		
	8707	F-F GTE Comm	8	10100	3600		
		Dyn Env	6	0	0		
		STS Op Wt	0	3600	3600		
			51.4	52950	15800		
94-19	8528	China Comm	30	17250	4590	160	(1) (2) DI to alt
	8503	Arab Comm	8	10100	3600		
	8529	Foreign Comm	8	10100	3600		
	8702	RCA Comm	8	10100	3600		
		Dyn Env	6	0	0		
		STS Op Wt	0	3600	3600		
			60	51150	18990		
94-20	8528	China Comm	30	17250	4590	160	(1) (2) DI to alt
	8539	F-F Foreign Comm	8	10100	3600		
	8702	RCA Comm	8	10100	3600		
	8703	SBS Comm	8	10100	3600		
		Dyn Env	6	0	0		
		STS Op Wt	0	3600	3600		
			60	51150	18990		

FLIGHT MANIFEST

1994

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
94-21	8002	F-F NASA Other	36.8	46500	7340	160	(1) (2) DI to alt
	8511	W.German/Other	15	15000	0		
		Dyn Env	5	0	0		
			56.8	61500	7340		
94-22	8726	F-F W.Union Comm	36.8	46500	7340	160	(1) (2) DI to alt
	8537	ESA/MPS F-F	10	6900	3600		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			51.8	57000	14540		
94-23	8726	F-F W.Union Comm	36.8	46500	7340	160	(1) (2) DI to alt
	1022	TSS/NASA	10	7640	7640		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			51.8	57740	18580		
94-24	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8510	F-F China Comm	8	10100	3600		
	8042	F-F NASA P&A	15	9000	0		
		Dyn Env	5.5	0	0		
		Op Wt	0	3600	3600		
			59.5	34800	8300		
94-25	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8732	US Com'l Comm	8	10100	3600		
	8728	F-F Hughes Comm	14.3	17250	1790		
		Dyn Env	5.5	0	0		
		STS Op	0	3600	3600		
			58.8	43060	10090		

FLIGHT MANIFEST

1994/1995

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
94-26	8724 F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8710 F-F Comsat Comsat	8	13750	3600		
	8728 F-F Hughes	14.3	17250	1790		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		58.8	46700	10090		
95-1	2022 LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
95-2	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
95-3	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
95-4	6002 Large Obs NASA	60	57320	32000	320	(1) DI to alt
95-5	7030 Planetary NASA	60	65000	5600	160	(2) DI to alt
95-6	2504 LEO Plat Foreign MPS	30	20000	20000	250	(1) (2) DI to alt
	8729 US Com'l Comm	8	13750	3600		
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2	0	0		
	STS Op Wt	0	3600	3600		
		58.5	58220	35800		
95-7	2504 LEO Plat Foreign	30	20000	20000	250	(1) (2) DI to alt
	8729 US Com'l Comm	8	13750	3600		
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2	0	0		
	STS Op Wt	0	3600	3600		
		58.5	58220	35800		

FLIGHT MANIFEST

1995

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
95-8	4022 LEO Stat NASA P&A	30	20000	20000	250	(1) (2) DI to alt
	8729 US Com'l Comm	8	13750	3600		
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	STS Op.Wt	0	3600	3600		
	Dyn Env	2	0	0		
		58.5	58220	35800		
95-9	4023 LEO Stat NASA/L.Sci	30	20000	20000	250	(1) (2) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	8727 Hughes Comm	8	10100	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2	0	0		
	STS Op Wt	0	3600	3600		
		58.5	54570	35800		
95-10	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 250
	3002 GEO Plat NASA Other	21	5500	500		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		55.5	54100	23040		
95-11	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 250
	3700 GEO Plat Com'l Comm	21.4	15400	1400		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		55.9	62000	21940		

FLIGHT MANIFEST

1995

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
95-12	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to 250
	8728 F-F Hughes Comm	14.3	17250	1790		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
		48.8	62250	20730		
95-13	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (3) DI to 250
	1029 S/L NASA MPS	15	9750	9750		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		49.5	58350	32290		
95-14	8002 F-F NASA Other	36.8	46500	7340	160	(1) (2) DI to 160, then retrieve ESA/MPS
	8038 F-F NASA DA	18	11030	0		
	1506 S/L Foreign MPS	(30)	0	5000		
	Dyn Env (4+1)	5	0	0		
	STS Op Wt	0	3600	3600		
		59.8	61130	15940		
95-15	5001 GEO Stat NASA Other	31	14300	1300	160	(1) (2) (3) DI to alt
	8728 F-F Hughes Comm	14.3	17250	1790		
	1005 S/L NASA MPS	5	6000	6000		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		55.8	41150	12690		
95-16	8724 F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8300 F-F GEO Civil E.Obs	8	10100	3600		
	8538 F-F Foreign E.Obs	8	10100	3600		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		52.5	35900	11900		

FLIGHT MANIFEST

1995

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
95-17	8724 F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8300 F-F GEO Civil E.Obs	8	10100	3600		
	8510 F-F China Comm	8	10100	3600		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		52.5	35900	11900		
95-18	8730 F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	8706 F-F AT&T Comm	8	10100	3600		
	8703 SBS Comm	8	10100	3600		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		51.5	41050	15390		
95-19	8730 F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	8526 F-F Canada Comm	20	23000	1790		
	Dyn Env	5	0	0		
	STS Op Wt	0	3600	3600		
		55	43850	9980		
95-20	8730 F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	1022 TSS/NASA	10	7640	7640		
	Dyn Env	5	0	0		
	STS Op Wt	0	3600	3600		
		45	28490	15830		
95-21	8528 China Comm	30	17250	4590	160	(1) (2) DI to alt
	1504 TSS Italy S/L	10	7640	7640		
	Dyn Env	5	0	0		
	STS Op Wt	0	3600	3600		
		45	28490	15830		

FLIGHT MANIFEST
1995/1996

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
95-22	5002	GEO Stat NASA Other	25	52000	16000	160	(1) (3) DI to alt
	1016	S/L NASA MPS	10	9000	9000		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			40	64600	28600		
96-1	2022	LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
96-2	2023	LEO Plat NASA/LSS	60	40000	32000	250	(1) DI to alt
96-3	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
96-4	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
96-5	7020	Planetary NASA	60	65000	5600	160	(2) DI to alt
96-6	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (3) DI to 250
	1016	S/L NASA MPS	10	9000	9000		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
		STS Op Wt	0	3600	3600		
			45	57600	31540		
96-7	8036	F-F NASA Comm	31	11000	1000	160	(1) (2) DI to alt
	8042	F-F NASA P&A	15	9000	0		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			51	23600	13600		

FLIGHT MANIFEST

1996

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
96-8	4020 LEO Stat NASA Other	26	40000	13940	250	(1) DI to 250
	3700 GEO Plat Com'l Comm	21.4	15400	1400		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		55.9	64000	23940		
96-9	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to 250
	8043 F-F NASA P&A	15	9000	0		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		49.5	57600	22540		
96-10	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to 250
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		46	64470	26140		
96-11	4024 LEO Stat NASA LSS	30	20000	20000	250	(1) (2) DI to 250
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		50	44470	32200		
96-12	4025 LEO Stat NASA Tech	30	20000	20000	250	(1) (2) DI to 250
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		50	44470	32200		

FLIGHT MANIFEST

1996

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
96-13	5001	GEO Stat NASA Other	31	14300	1300	160	(1) (2) DI to alt
	8707	F-F GTE Comm	8	10100	3600		
	8731	US Com'l Comm	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			52.5	38100	12100		
96-14	8726	F-F W.Union Comm	36.8	46500	7340	160	(1) (2) DI to alt
	8511	W.German/Other	15	15000	0		
		Dyn Env	5	0	0		
			56.8	61500	7340		
96-15	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
	8500	Canada Comm	8	13750	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			56	45320	11900		
96-16	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8703	SBS Comm	8	10100	3600		
	8537	ESA/MPS F-F	10	6900	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			54.5	32700	11900		
96-17	8527	F-F France Comm	31	10450	950	160	(1) (2) (3) DI to alt
	1029	S/L NASA MPS	15	9750	9750		
	8529	Foreign Comm	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	33900	17900		

C-83

FLIGHT MANIFEST

1996

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
96-18	8528	China Comm	30	17250	4590	160	(1) (2) DI to alt
	8303	F-F Civil/USPS Comm	8	10100	3600		
	1504	S/L Italy TSS	10	7640	7640		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	38590	19430		
96-19	8730	F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	8305	F-F Civil/USPS Comm	8	10100	3600		
	1504	S/L Italy TSS	10	7640	7640		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	38590	19430		
96-20	8730	F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	3701	GEO Plat Com'l Comm	21	5500	2500		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			56	26350	10690		
96-21	8730	F-F Com'l Comm	30	17250	4590	160	(1) (2) DI to alt
	3700	GEO Plat Com'l Comm	21.4	15400	1400		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			51.4	36250	9590		
96-22	5002	GEO Stat NASA Other	25	52000	16000	160	(1) (2) DI to alt
		Dyn Env	4.5	0	0		
		STS Op Wt	0	3600	3600		
			29.5	55600	19600		

FLIGHT MANIFEST

1997

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
97-1	2022	LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
97-2	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
97-3	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
97-4	6002	Large Obs NASA	60	57320	32000	320	DI to alt
97-5	6003	Large Obs NASA	60	28600	45690	320	DI to alt
97-6	7020	Planetary NASA	60	65000	5600	160	(2) DI to alt
97-7	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			46	64470	26140		
97-8	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3003	GEO Plat NASA Other	21.4	15400	1400		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.9	64000	23940		
97-9	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3701	GEO Plat Com'l Comm	21	5500	2500		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.5	54100	25040		

FLIGHT MANIFEST

1997

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
97-10	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3701	GEO Plat Com'l Comm	21	5500	2500		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.5	54100	25040		
97-11	4022	LEO Stat NASA P&A	30	20000	20000	250	(1) DI to alt
	2034	LEO Plat NASA Other	15	10000	10000		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	38600	38600		
97-12	4023	LEO Stat NASA/L.Sci	30	20000	20000	250	(1) DI to alt
	2034	LEO Plat NASA/Other	15	10000	10000		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	.0	0		
		STS Op Wt	0	3600	3600		
			53.5	38600	38600		
97-13	8726	F-F W.Union Comm	36.8	46500	7340	160	(1) (2) DI to 160, then retrieve ESA/MPS
	8529	Foreign Comm	8	10100	3600		
	1506	S/L Foreign MPS	(30)	0	5000		
		Dyn Env	5	0	0		
		STS Op Wt	0	3600	3600		
			49.8	60200	19540		
97-14	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8707	F-F GTE Comm	8	10100	3600		
	9006	Dod PAM-D	11.5	15870	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			56	41670	11900		

FLIGHT MANIFEST

1997

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
97-15	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8727	Hughes Comm	8	10100	3600		
	9006	Dod PAM-D	11.5	15870	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			56	41670	11900		
97-16	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8305	F-F Civil/USPS Comm	8	10100	3600		
	9006	Dod PAM-D	11.5	15870	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			56	41670	11900		
97-17	8527	F-F France Comm	31	10450	950	160	(1) (2) DI to alt
	8703	SBS Comm	8	10100	3600		
	8300	GEO Civil E.Obs	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			52.5	34250	11750		
97-18	8040	F-F NASA E.Obs	31	17600	1600	270	(1) (2) DI to alt
	8523	Columbia Comm	8	10100	3600		
	8533	F-F Japan P&A	15	10000	0		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	41300	8800		
97-19	5001	GEO Stat NASA Other	31	14300	1300	160	
	8538	F-F Foreign E.Obs	8	10100	3600		
	8731	US Com'l Comm	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			52.5	38100	12100		

FLIGHT MANIFEST

1997/1998

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
97-20	5002 GEO Stat NASA Other 1022 TSS/NASA Dyn Env STS Op Wt	25 10 5 0 40	52000 7640 0 3600 63240	16000 7640 0 3600 27240	160	(1) DI to alt
97-21	8706 F-F AT&T Comm 8500 Canada Comm 1029 S/L NASA MPS Dyn Env STS Op Wt	8 8 15 5.5 0 36.5	10100 13750 9750 0 3600 37200	3600 3600 9750 0 3600 20550	160	(1) (2) (3) DI to alt
98-1	2022 LEO Plat NASA/P&A	60	40000	32000	250	(1) DI to alt
98-2	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
98-3	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
98-4	4021 LEO Stat NASA/E.Obs	60	40000	32000	250	(1) DI to alt
98-5	6002 Large Obs NASA	60	57320	32000	320	DI to alt
98-6	7035 Planetary NASA	60	65000	5600	160	(2) DI to alt
98-7	2504 LEO Plat Foreign MPS 3700 GEO Plat Com'l Comm Docking Mod Dyn Env STS Op Wt	30 21.4 7 1.5 0 59.9	20000 15400 5000 0 3600 44000	20000 1400 5000 0 3600 30000	250	(1) (2) DI to alt

FLIGHT MANIFEST

1998

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
				UP (LB)	DOWN (LB)		
98-8	2504	LEO Plat Foreign MPS	30	20000	20000	250	(1) (2) DI to alt
	3700	GEO Plat Com'l Comm	21.4	15400	1400		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			59.9	44000	30000		
98-9	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			46	64470	26140		
98-10	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006	Dod PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			46	64470	26140		
98-11	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			46	64470	26140		
98-12	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) (3) DI to alt
	8732	US Com'l Comm	8	10100	3600		
	1016	S/L NASA MPS	10	9000	9000		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
			53	64100	31540		

FLIGHT MANIFEST

1998

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
				UP (LB)	DOWN (LB)		
98-13	8724	F-F Intelsat	31	12100	1100	270	(1) (2) DI to alt
	8702	RCA Comm	8	10100	3600		
	8044	F-F NASA P&A	15	10000	0		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	35800	8300		
98-14	8724	F-F Intelsat	31	12100	1100	160	(1) (2) DI to alt
	8500	Canada Comm	8	13750	3600		
	8511	W.German/Other	15	15000	0		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	44450	8300		
98-15	8724	F-F Intelsat	31	12100	1100	160	(1) (2) (3) DI to alt
	8703	SBS Comm	8	10100	3600		
	1029	S/L NASA MPS	15	9750	9750		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	35550	18050		
98-16	5001	GEO Stat NASA Other	31	14300	1300	160	(1) (2) DI to alt
	9006	DoD PAM-D	11.5	15870	3600		
	1022	TSS/NASA	10	7640	7640		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			58	41410	16140		
98-17	8528	China Comm	30	17250	4590	160	(1) (2) DI to alt
	8706	F-F AT&T Comm	8	10100	3600		
	8537	ESA/MPS F-F	10	6900	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	37850	15390		

FLIGHT MANIFEST

1998/1999

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT		ALTITUDE (NMI)	COMMENTS
			UP (LB)	DOWN (LB)		
98-18	8528 China Comm 8529 Foreign Comm 1504 TSS Italy S/L Dyn Env STS Op Wt	30 8 10 5.5 0	17250 10100 7640 0 3600	4590 3600 7640 0 3600	160	(1) (2) DI to alt
		53.5	38590	19430		
98-19	5002 GEO Stat NASA Other 3701 GEO Plat Com'l Comm Dyn Env STS Op Wt	25 21 5 0	52000 5500 0 3600	16000 2500 0 3600	160	
		51	61100	22100		
99-1	2022 LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
99-2	2023 LEO Plat NASA/LSS	60	40000	32000	250	(1) DI to alt
99-3	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
99-4	2707 LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
99-5	6001 Large Obs NASA	60	58040	17100	320	DI to alt
99-6	7036 Planetary NASA	60	65000	5600	160	(2) DI to alt
99-7	4020 LEO Stat NASA Other 3700 GEO Plat Com'l Comm Docking Mod Dyn Env STS Op Wt	26 21.4 7 1.5 0	40000 15400 5000 0 3600	13940 1400 5000 0 3600	250	(1) (2) DI to alt
		55.9	64000	23940		

FLIGHT MANIFEST

1999

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
99-8	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3701 GEO Plat Com'l Comm	21	5500	2500		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		55.5	54100	25040		
99-9	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		46	64470	26140		
99-10	4020 LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	1.5	0	0		
	STS Op Wt	0	3600	3600		
		46	64470	26140		
99-11	4022 LEO Stat NASA P&A	30	20000	20000	250	(1) (2) DI to alt
	8702 RCA Comm	8	10100	3600		
	9006 DoD PAM-D	11.5	15870	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2	0	0		
	STS Op Wt	0	3600	3600		
		58.5	54570	35800		

FLIGHT MANIFEST

1999

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
99-12	4023	LEO Stat NASA/L.Sci	30	20000	20000	250	(1) (2) DI to alt
	8727	Hughes Comm	8	10100	3600		
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
		STS Op Wt	0	3600	3600		
			58.5	54570	35800		
99-13	5001	GEO Stat NASA Other	31	14300	1300	160	(1) (2) (3) DI to 160, then retrieve ESA/MPS
	8702	RCA Comm	8	10100	3600		
	1029	S/L NASA MPS	15	9750	9750		
	1506	S/L Foreign MPS	(30)	0	5000		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	37750	23250		
99-14	8528	China Comm	30	17250	4590	160	(1) (2) DI to alt
	8538	F-F Foreign E.Obs	8	10100	3600		
	1504	S/L Italy TSS	10	7640	7640		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			53.5	38590	19430		
99-15	5002	GEO Stat NASA Other	25	52000	16000	160	(1) (2) DI to alt
	8539	F-F Foreign Comm	8	10100	3600		
		Dyn Env	5	0	0		
			38	62100	19600		

FLIGHT MANIFEST

1999/2000

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
99-16	8703	SBS Comm	8	10100	3600	160	(1) (2) DI to alt
	8525	F-F Mexico Comm	8	10100	3600		
	8529	Foreign Comm	8	10100	3600		
	8502	Indonesia Comm	8	10100	3600		
	1504	TSS Italy S/L	10	7640	7640		
		Dyn Env	6.5	0	0		
		STS Op Wt	0	3600	3600		
			48.5	51640	25640		
00-1	2022	LEO Plat NASA/MPS	60	40000	32000	250	(1) DI to alt
00-2	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
00-3	2707	LEO Plat Com'l/MPS	60	40000	32000	250	(1) DI to alt
00-4	6002	Large Obs NASA	60	57320	32000	320	DI to alt
00-5	6002	Large Obs NASA	60	57320	32000	320	DI to alt
00-6	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3700	GEO Plat Com'l Comm	21.4	15400	1400		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.9	64000	23940		
00-7	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3700	GEO Plat Com'l Comm	21.4	15400	1400		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.9	64000	23940		

FLIGHT MANIFEST

2000

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
00-8	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3701	GEO Plat Com'l Comm	21	5500	2500		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.5	54100	25040		
00-9	4020	LEO Stat NASA Other	26	40000	13940	250	(1) (2) DI to alt
	3002	GEO Plat NASA Other	21	5500	500		
		Docking Mod	7	5000	5000		
		Dyn Env	1.5	0	0		
		STS Op Wt	0	3600	3600		
			55.5	54100	25040		
00-10	4024	LEO Stat NASA LSS	30	20000	20000	250	(1) (2) DI to alt
	8731	US Com'l Comm	8	10100	3600		
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	.0		
		STS Op Wt	0	3600	3600		
			58.5	54570	35800		
00-11	4025	LEO Stat NASA Tech	30	20000	20000	250	(1) (2) DI to alt
	8732	US Com'l Comm	8	10100	3600		
	9006	DoD PAM-D	11.5	15870	3600		
		Docking Mod	7	5000	5000		
		Dyn Env	2	0	0		
		STS Op Wt	0	3600	3600		
			58.5	54570	35800		

FLIGHT MANIFEST

2000

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
00-12	2025 LEO Plat NASA E.Obs	15	10000	10000	250	(1) (2) DI to alt
	2034 LEO Plat NASA Other	15	10000	10000		
	8503 Arab Comm	8	10100	3600		
	8702 RCA Comm	8	10100	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2.5	0	0		
	STS Op Wt	0	3600	3600		
		55.5	48800	35800		
00-13	2030 LEO Plat NASA Tech	15	10000	10000	250	(1) (2) DI to alt
	2034 LEO Plat NASA Other	15	10000	10000		
	8525 F-F Mexico Comm	8	10100	3600		
	8539 F-F Foreign Comm	8	10100	3600		
	Docking Mod	7	5000	5000		
	Dyn Env	2.5	0	0		
	STS Op Wt	0	3600	3600		
		55.5	48800	35800		
00-14	5004 GEO Stat NASA Other	31	31280	2840	160	(1) (2) DI to alt
	9006 DoD PAM-D	11.5	15870	3600		
	1022 TSS/NASA	10	7640	7640		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		58	58390	17680		
00-15	5004 GEO Stat NASA Other	31	31280	2840	160	(1) (2) DI to alt
	8703 SBS Comm	8	10100	3600		
	9006 DoD PAM-D	11.5	15870	3600		
	Dyn Env	5.5	0	0		
	STS Op Wt	0	3600	3600		
		56	60850	13640		

FLIGHT MANIFEST

2000

REV. 2

FLIGHT NO.	PAYLOADS		LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
00-16	5001	GEO Stat NASA Other	31	14300	1300	160	(1) (2) DI to alt
	8710	F-F Comsat Comm	8	13750	3600		
	8511	W.German/Other	15	15000	0		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	46650	8500		
00-17	5001	GEO Stat NASA Other	31	14300	1300	160	(1) (2) (3) DI to alt
	8710	F-F Comsat Comm	8	13750	3600		
	1029	S/L NASA MPS	15	9750	9750		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			59.5	41400	18250		
00-18	8036	F-F NASA Comm	31	11000	1000	160	(1) (2) DI to alt
	8706	F-F AT&T Comm	8	10100	3600		
	8300	GEO Civil E.Obs	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			52.5	34800	11800		
00-19	8528	China Comm.	30	17250	4590	160	(1) (2) DI to alt
	8300	F-F GEO Civil E.Obs	8	10100	3600		
	8502	Indonesia Comm	8	10100	3600		
		Dyn Env	5.5	0	0		
		STS Op Wt	0	3600	3600		
			51.5	41050	15390		
00-20	5002	GEO Stat NASA Other	25	52000	16000	160	(1) (2) DI to alt
		Dyn Env	4.5	0	0		
		STS Op Wt	0	3600	3600		
			29.5	55600	19600		

FLIGHT MANIFEST

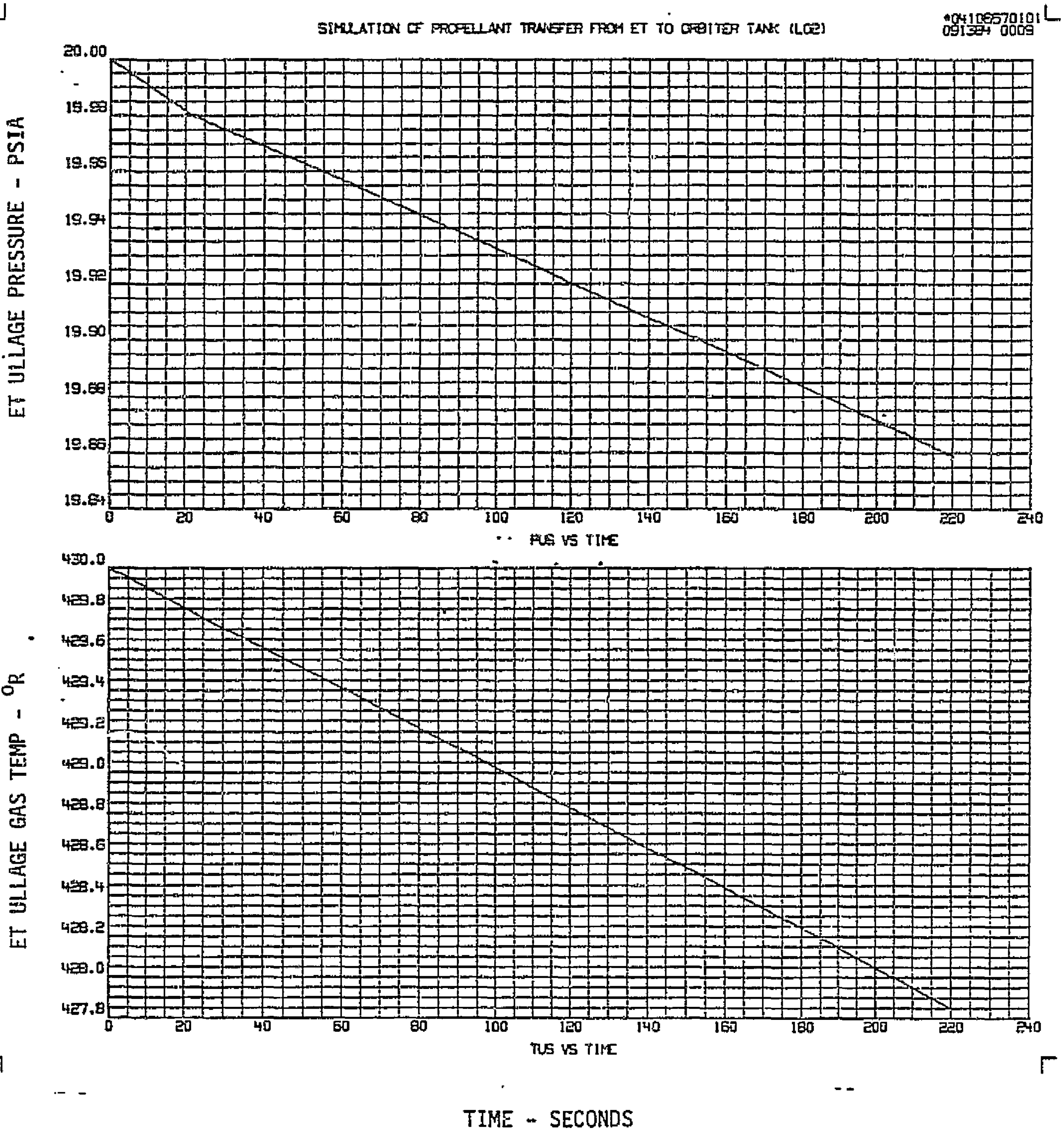
2000

REV. 2

FLIGHT NO.	PAYLOADS	LENGTH (FT)	WEIGHT UP (LB)	WEIGHT DOWN (LB)	ALTITUDE (NMI)	COMMENTS
00-21	5002 GEO Stat NASA Other	25	52000	16000	160	(1) (2) DI to alt
	Dyn Env	4.5	0	0		
	STS Op Wt	0	3600	3600		
		29.5	55600	19600		
00-22	8702 RCA Comm	8	10100	3600	160	(1) (2) (3) DI to alt
	8523 Columbia Comm	8	10100	3600		
	1016 S/L NASA MPS	10	9000	9000		
	8537 ESA/MPS F-F	10	6900	3600		
	Dyn Env	6	0	0		
	STS Op Wt	0	3600	3600		
		42	39700	23400		

A.2 PROPELLANT TRANSFER FIGURES

The performance characteristics of the baseline transfer line design (2 inches) for transferring propellant from the ET to orbiter tanks are shown in Figures A-1 through A-6 for the LO_2 system and Figures A-7 through A-13 for the LH_2 system. The performance characteristics of the baseline design for transferring propellant from orbiter tanks to the user are shown in Figures A-14 through A-20 (LO_2) and Figures A-21 through A-27 (LH_2).



SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04106570101
091384 0009
ET ULLAGE PRESSURE - PSIA
20.00
19.98
19.96
19.94
19.92
19.90
19.88
19.86
19.84
0
20
40
60
80
100
120
140
160
180
200
220
240
PUS VS TIME
ET ULLAGE GAS TEMP - °R
430.0
429.8
429.6
429.4
429.2
429.0
428.8
428.6
428.4
428.2
428.0
427.8
TUS VS TIME
TIME - SECONDS

Figure A-1

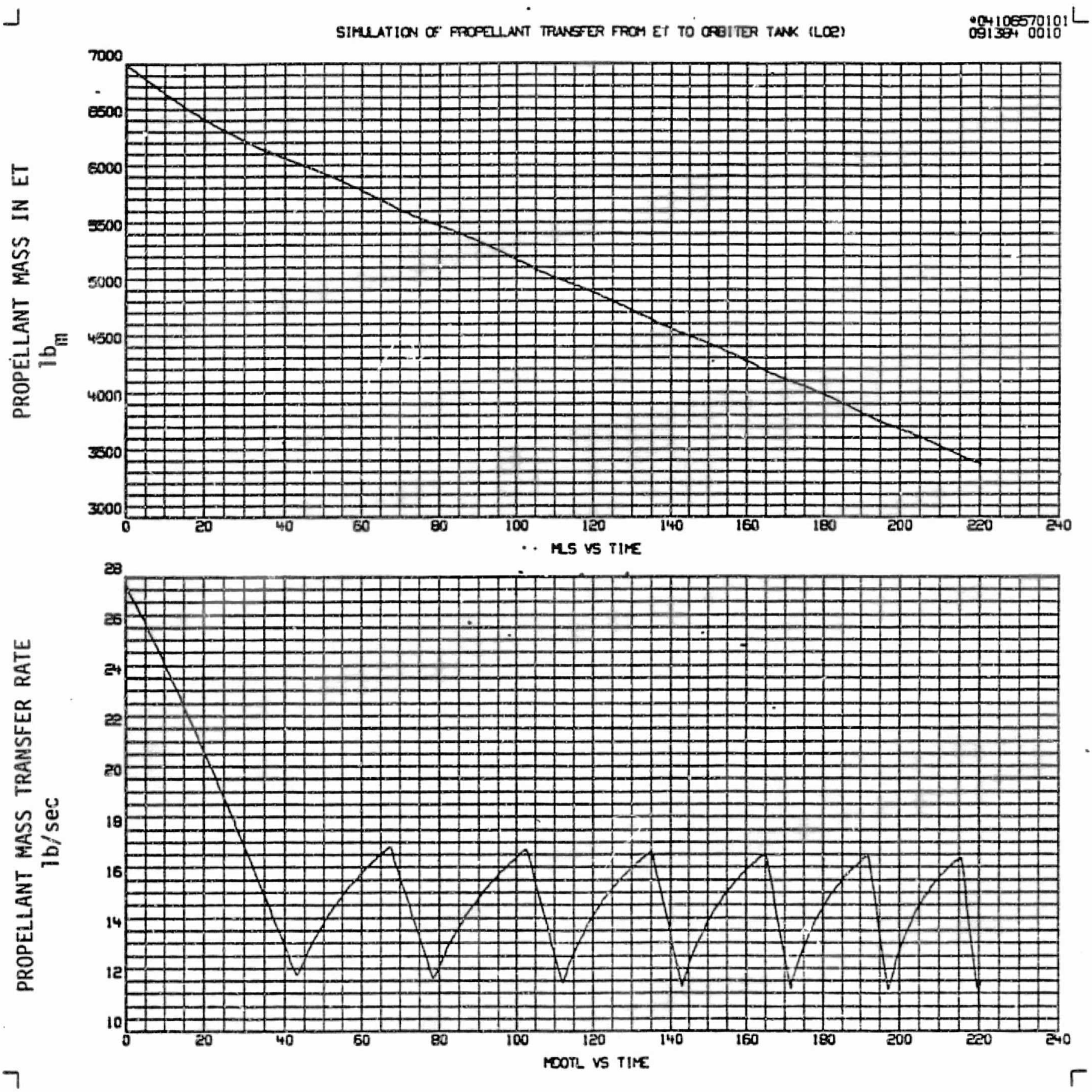

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04106570101
091384 0010
PROPELLANT MASS IN ET
lb_m
7000
6500
6000
5500
5000
4500
4000
3500
3000
0
20
40
60
80
100
120
140
160
180
200
220
240
MLS VS TIME
PROPELLANT MASS TRANSFER RATE
lb/sec
28
26
24
22
20
18
16
14
12
10
MDOTL VS TIME
TIME - SECONDS

Figure A-2

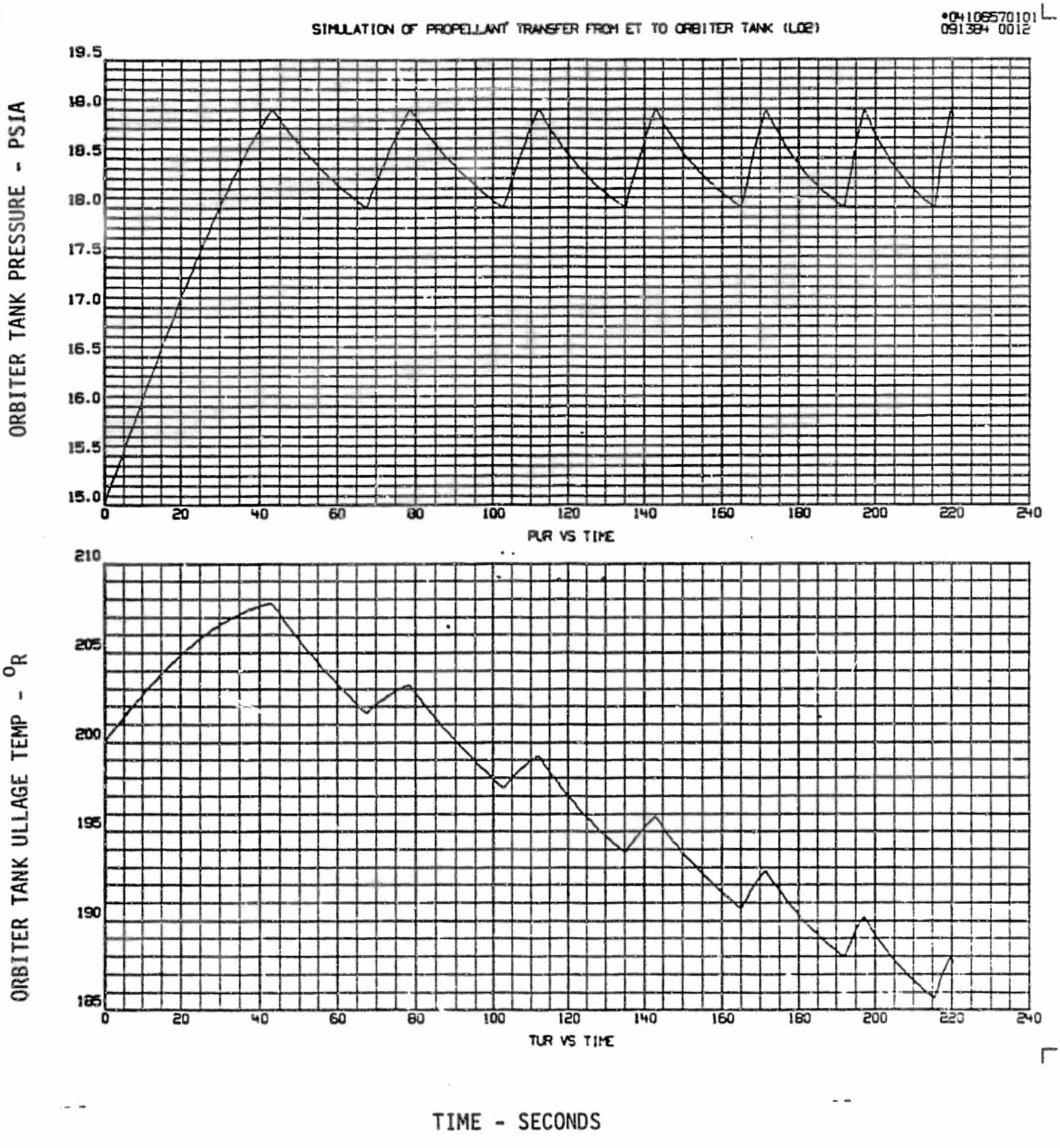

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04106570101
091384 0012
ORBITER TANK PRESSURE - PSIA
19.5
19.0
18.5
18.0
17.5
17.0
16.5
16.0
15.5
15.0
0
20
40
60
80
100
120
140
160
180
200
220
240
PUR VS TIME
ORBITER TANK ULLAGE TEMP - °R
210
205
200
195
190
185
TUR VS TIME
TIME - SECONDS

Figure A-3

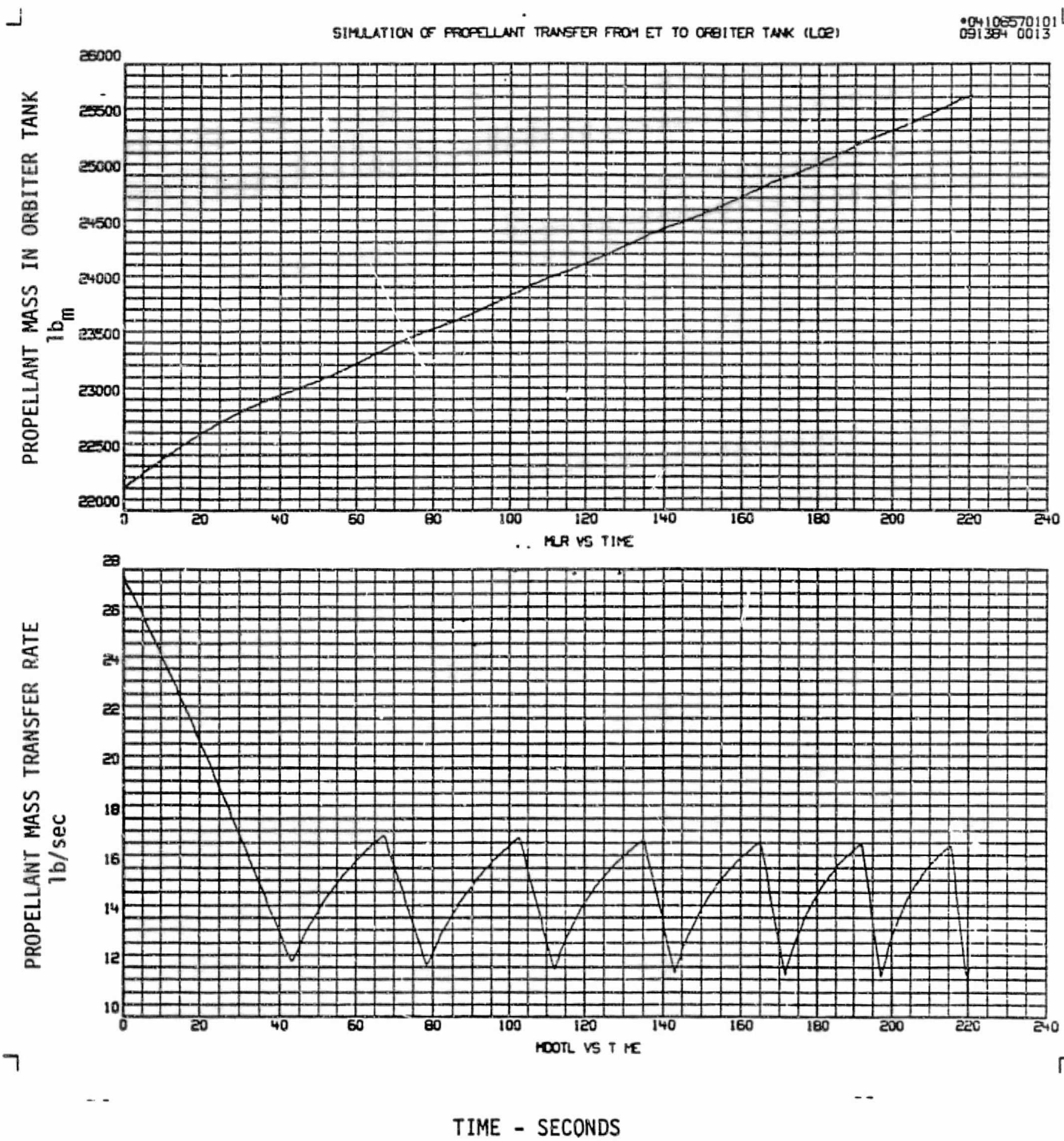

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04108570101
091384 0013
PROPELLANT MASS IN ORBITER TANK
lb_m
26000
25500
25000
24500
24000
23500
23000
22500
22000
0
20
40
60
80
100
120
140
160
180
200
220
240
.. MLR VS TIME
PROPELLANT MASS TRANSFER RATE
lb/sec
28
26
24
22
20
18
16
14
12
10
MDOTL VS T ME
TIME - SECONDS

Figure A-4

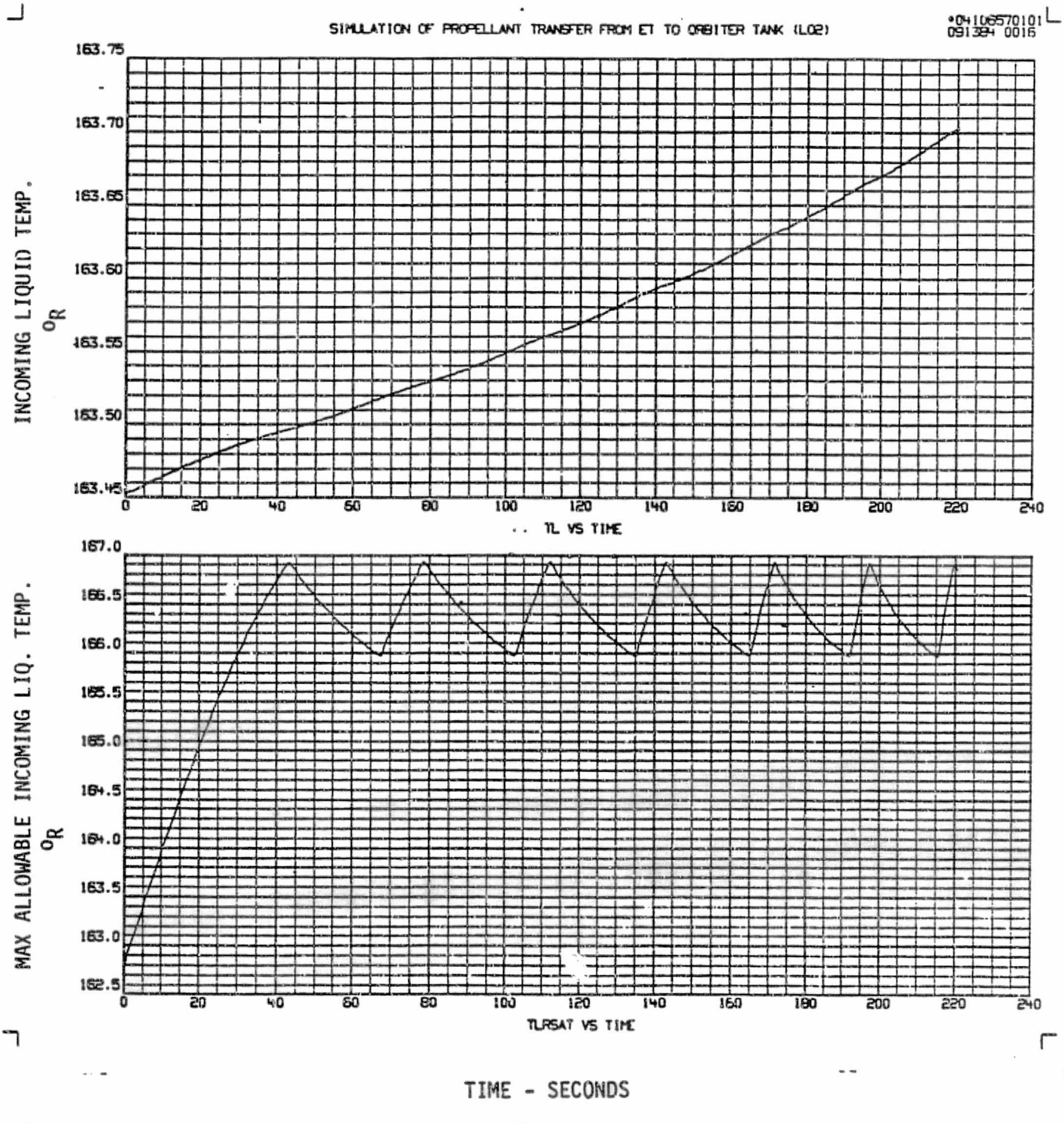
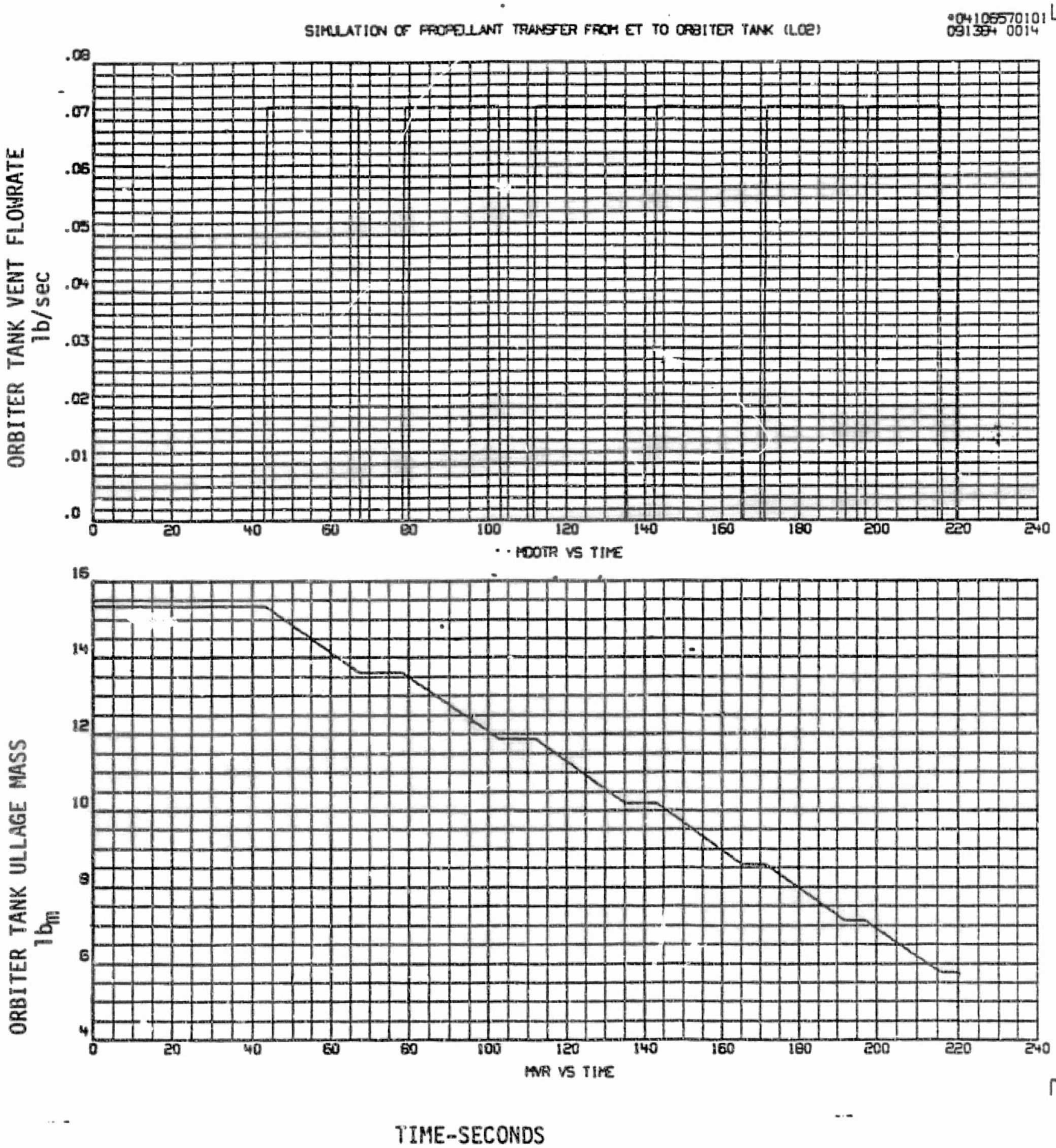

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04106570101
091384 0016
INCOMING LIQUID TEMP. °R
163.75
163.70
163.65
163.60
163.55
163.50
163.45
0
20
40
60
80
100
120
140
160
180
200
220
240
TL VS TIME
MAX ALLOWABLE INCOMING LIQ. TEMP. °R
167.0
166.5
166.0
165.5
165.0
164.5
164.0
163.5
163.0
162.5
TLRSAT VS TIME
TIME - SECONDS

Figure A-5



SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LO2)
*04106570101
091384 0014
ORBITER TANK VENT FLOWRATE
lb/sec
.08
.07
.06
.05
.04
.03
.02
.01
.0
0
20
40
60
80
100
120
140
160
180
200
220
240
MDOTR VS TIME
ORBITER TANK ULLAGE MASS
lbm
16
14
12
10
8
6
4
MVR VS TIME
TIME-SECONDS

Figure A-6

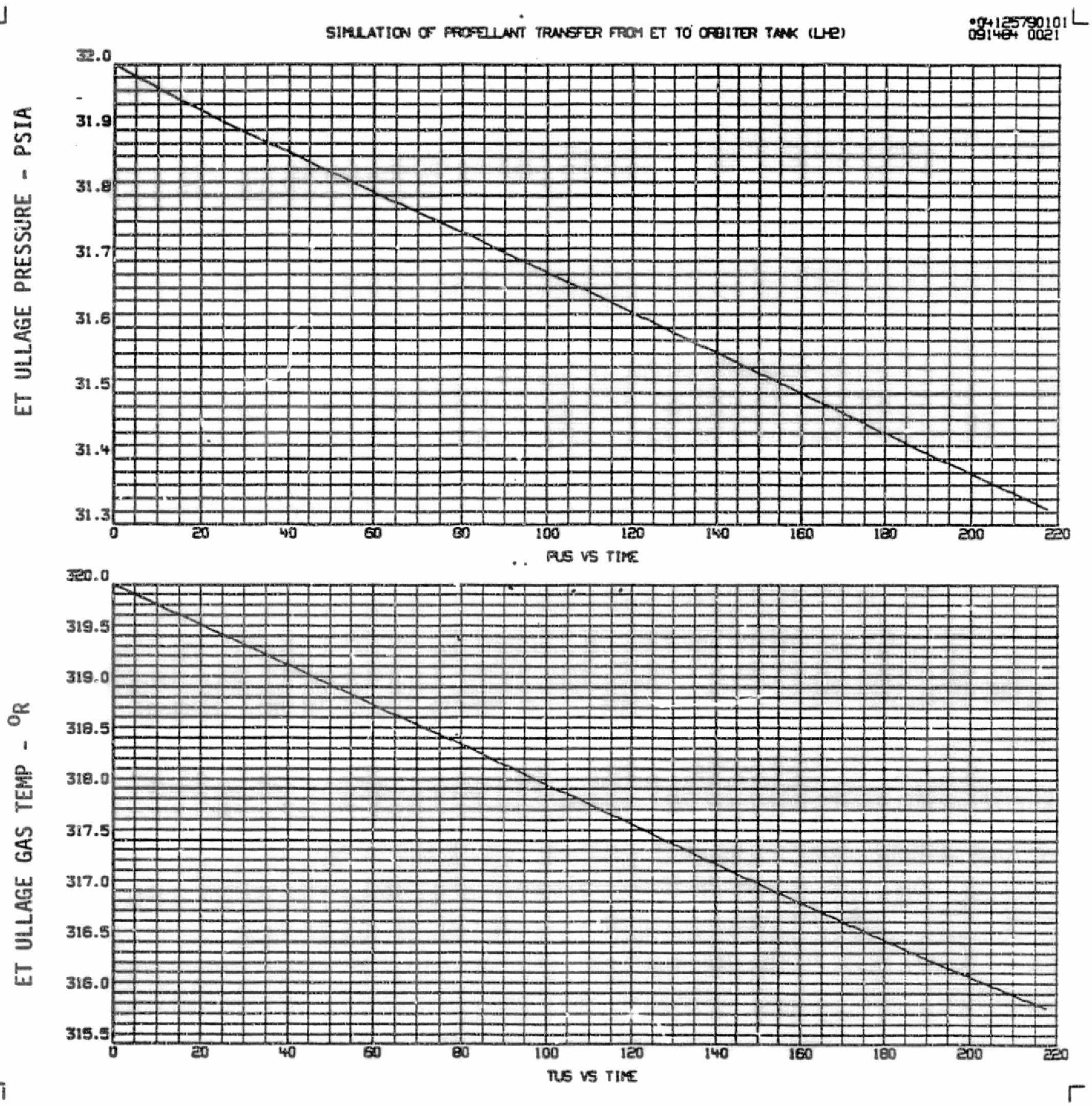

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*0412579010I
091484 0021
ET ULLAGE PRESSURE - PSIA
32.0
31.9
31.8
31.7
31.6
31.5
31.4
31.3
0
20
40
60
80
100
120
140
160
180
200
220
PUS VS TIME
ET ULLAGE GAS TEMP - °R
320.0
319.5
319.0
318.5
318.0
317.5
317.0
316.5
316.0
315.5
TUS VS TIME
TIME - SECONDS

Figure A-7

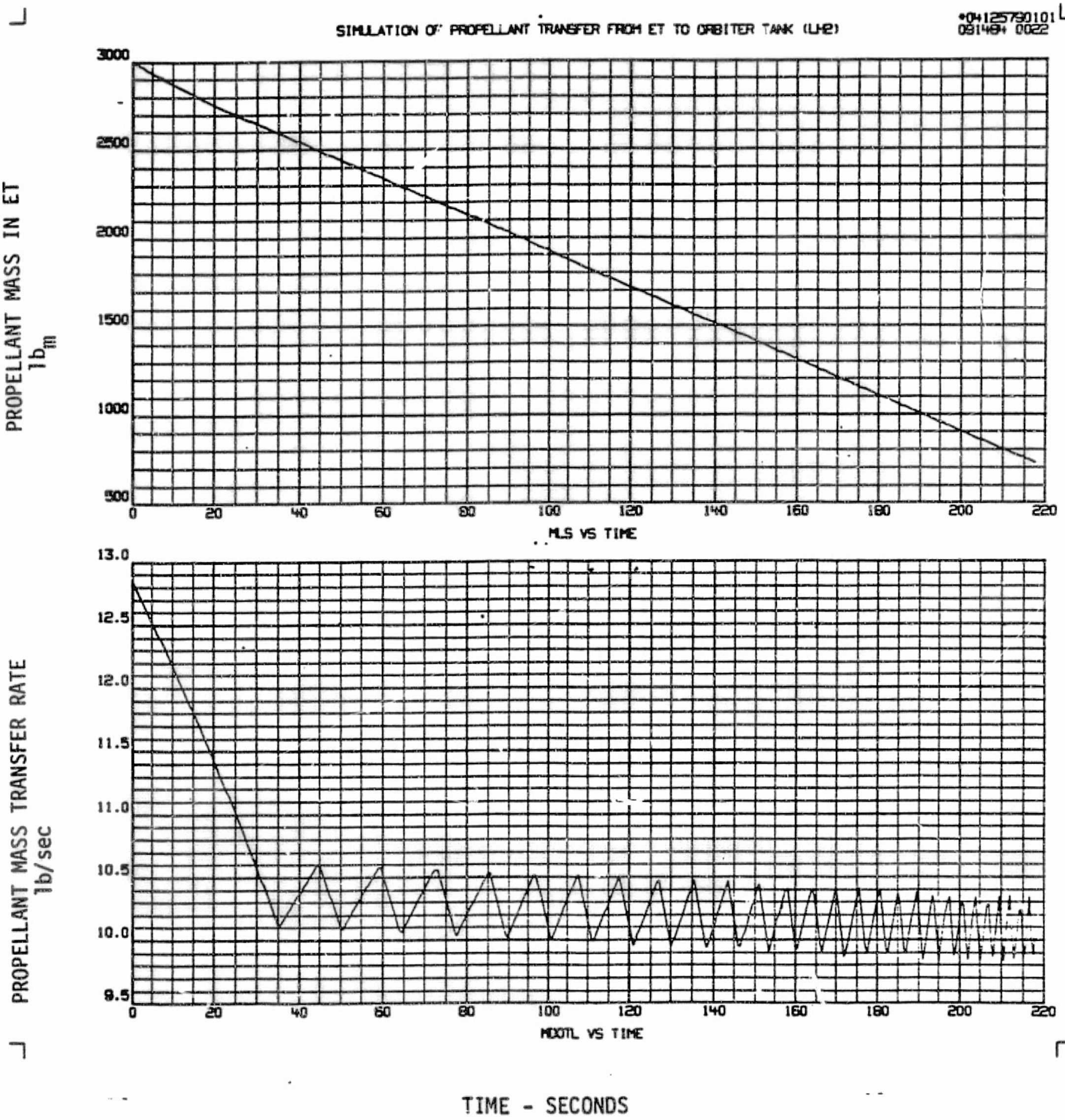

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
081484 0022
PROPELLANT MASS IN ET
lbm
3000
2500
2000
1500
1000
500
0
20
40
60
80
100
120
140
160
180
200
220
MLS VS TIME
PROPELLANT MASS TRANSFER RATE
lb/sec
13.0
12.5
12.0
11.5
11.0
10.5
10.0
9.5
MDOTL VS TIME
TIME - SECONDS

Figure A-8

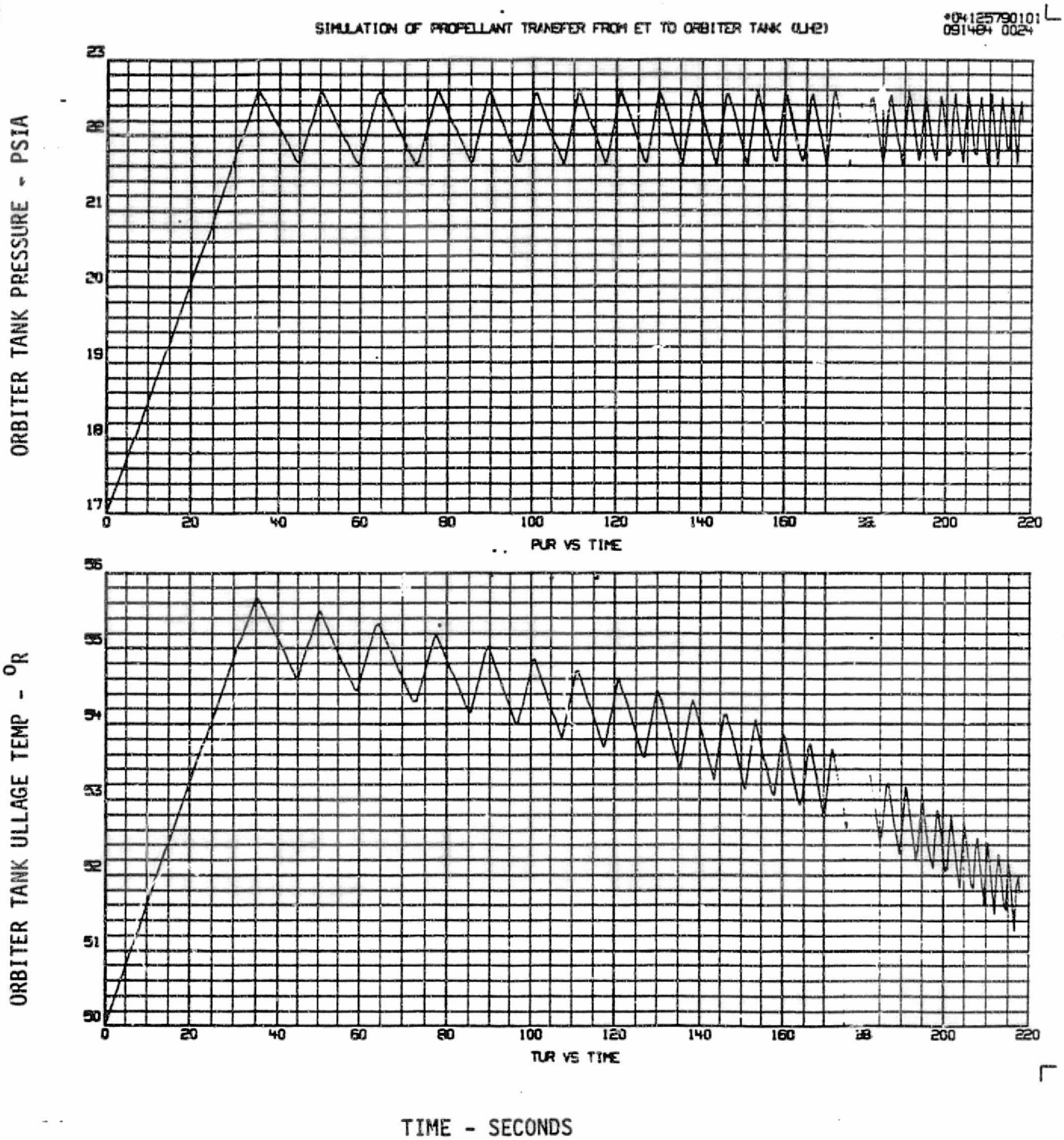
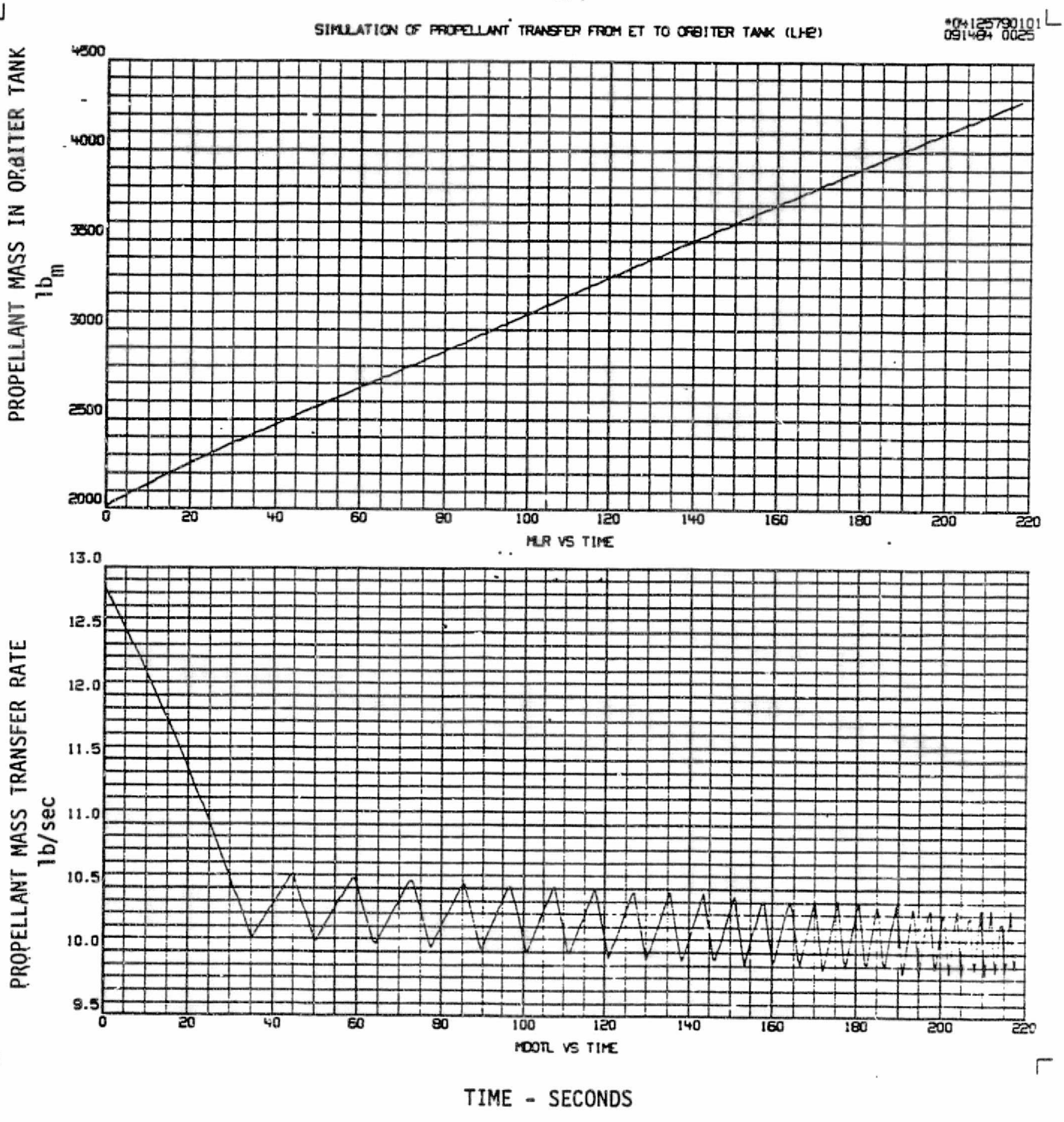

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
091484 0024
ORBITER TANK PRESSURE - PSIA
PUR VS TIME
ORBITER TANK ULLAGE TEMP - °R
TUR VS TIME
TIME - SECONDS

Figure A-9



SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
091484 0025
PROPELLANT MASS IN ORBITER TANK
lb_m
4500
4000
3500
3000
2500
2000
0
20
40
60
80
100
120
140
160
180
200
220
MLR VS TIME
PROPELLANT MASS TRANSFER RATE
lb/sec
13.0
12.5
12.0
11.5
11.0
10.5
10.0
9.5
MDOTL VS TIME
TIME - SECONDS

Figure A-10

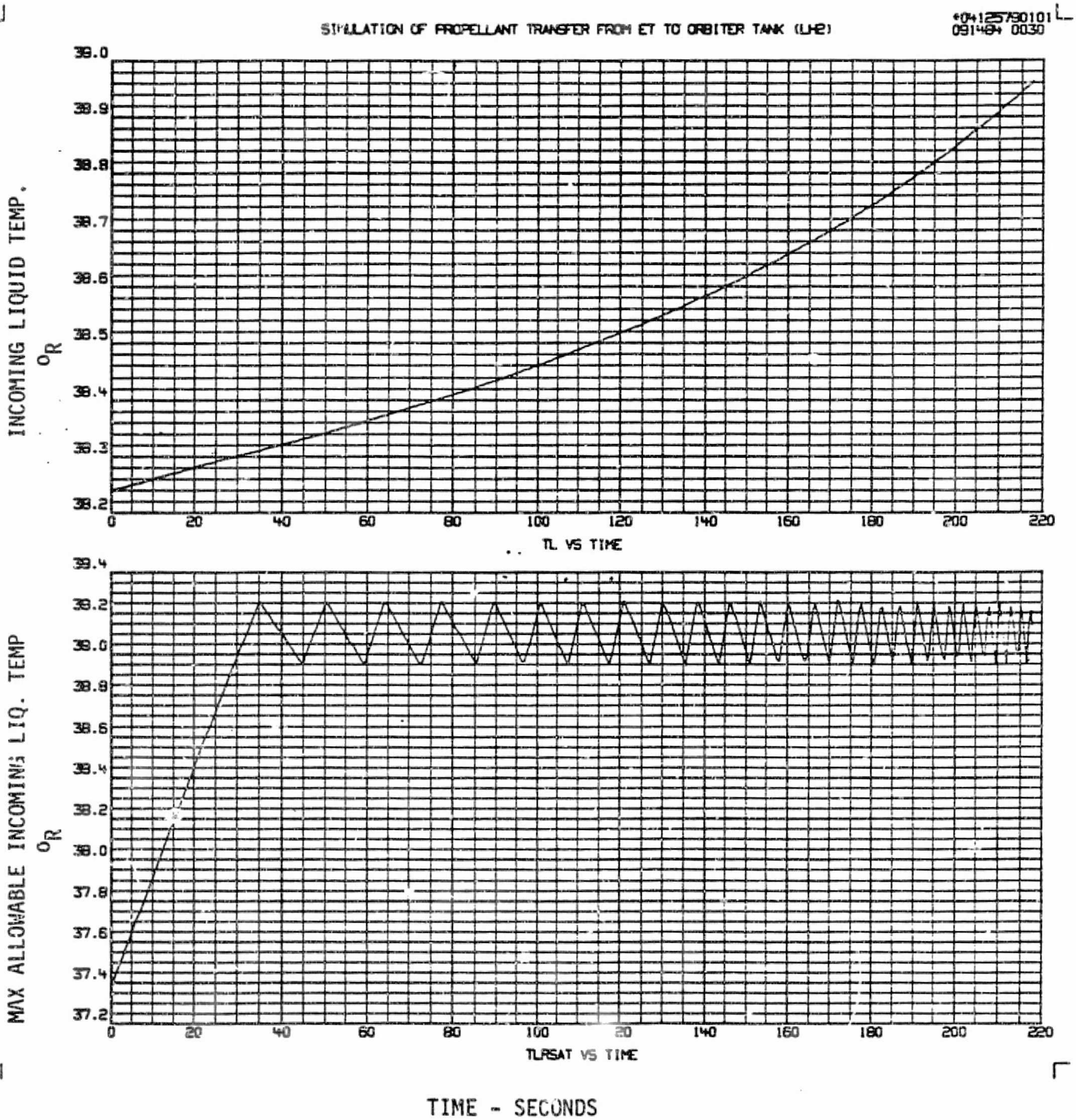

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
091484 0030
INCOMING LIQUID TEMP.
°R
39.0
38.9
38.8
38.7
38.6
38.5
38.4
38.3
38.2
0
20
40
60
80
100
120
140
160
180
200
220
TL VS TIME
MAX ALLOWABLE INCOMING LIQ. TEMP
°R
39.4
39.2
39.0
38.8
38.6
38.4
38.2
38.0
37.8
37.6
37.4
37.2
TLRSAT VS TIME
TIME - SECONDS

Figure A-11

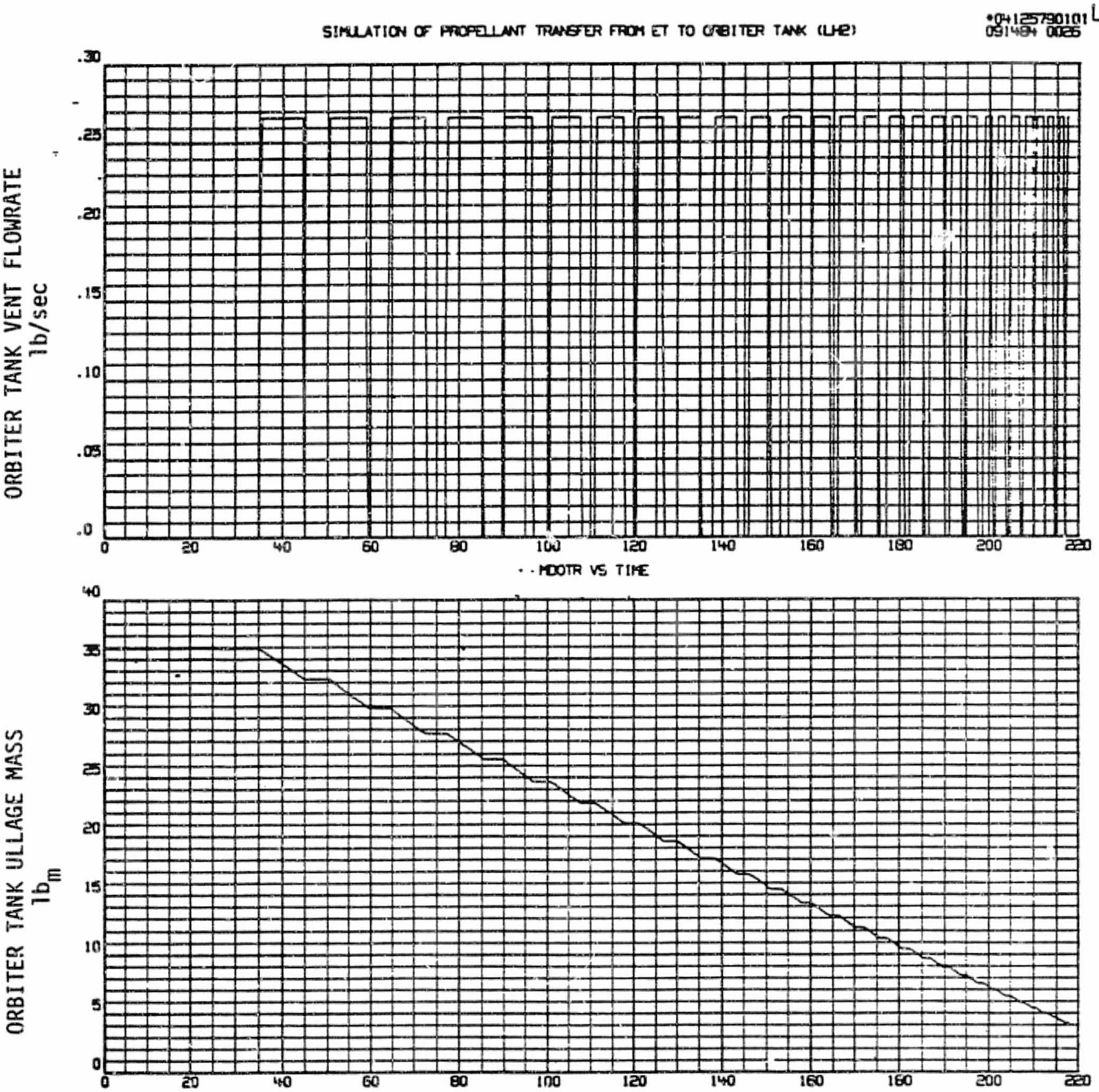

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
091484 0026
ORBITER TANK VENT FLOWRATE
lb/sec
.30
.25
.20
.15
.10
.05
.0
0
20
40
60
80
100
120
140
160
180
200
220
MDOTR VS TIME
ORBITER TANK ULLAGE MASS
lb_m
40
35
30
25
20
15
10
5
0
MVR VS TIME
TIME - SECONDS

Figure A-12

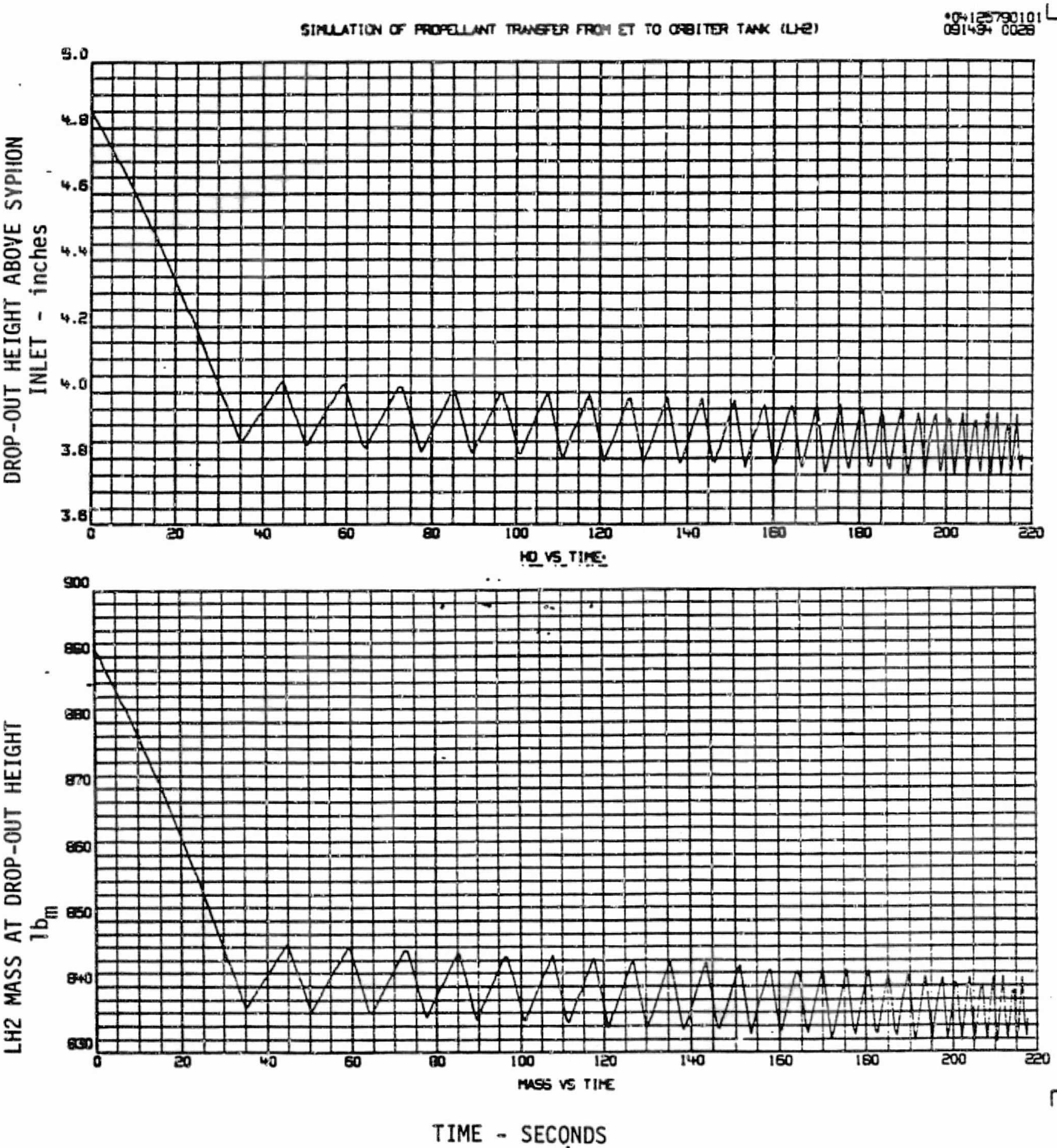

SIMULATION OF PROPELLANT TRANSFER FROM ET TO ORBITER TANK (LH2)
*04125790101
091494 0028
DROP-OUT HEIGHT ABOVE SYPHON INLET - inches
5.0
4.8
4.6
4.4
4.2
4.0
3.8
3.6
0 20 40 60 80 100 120 140 160 180 200 220
HD VS TIME
LH2 MASS AT DROP-OUT HEIGHT lb_m
900
890
880
870
860
850
840
830
0 20 40 60 80 100 120 140 160 180 200 220
MASS VS TIME
TIME - SECONDS

Figure A-13

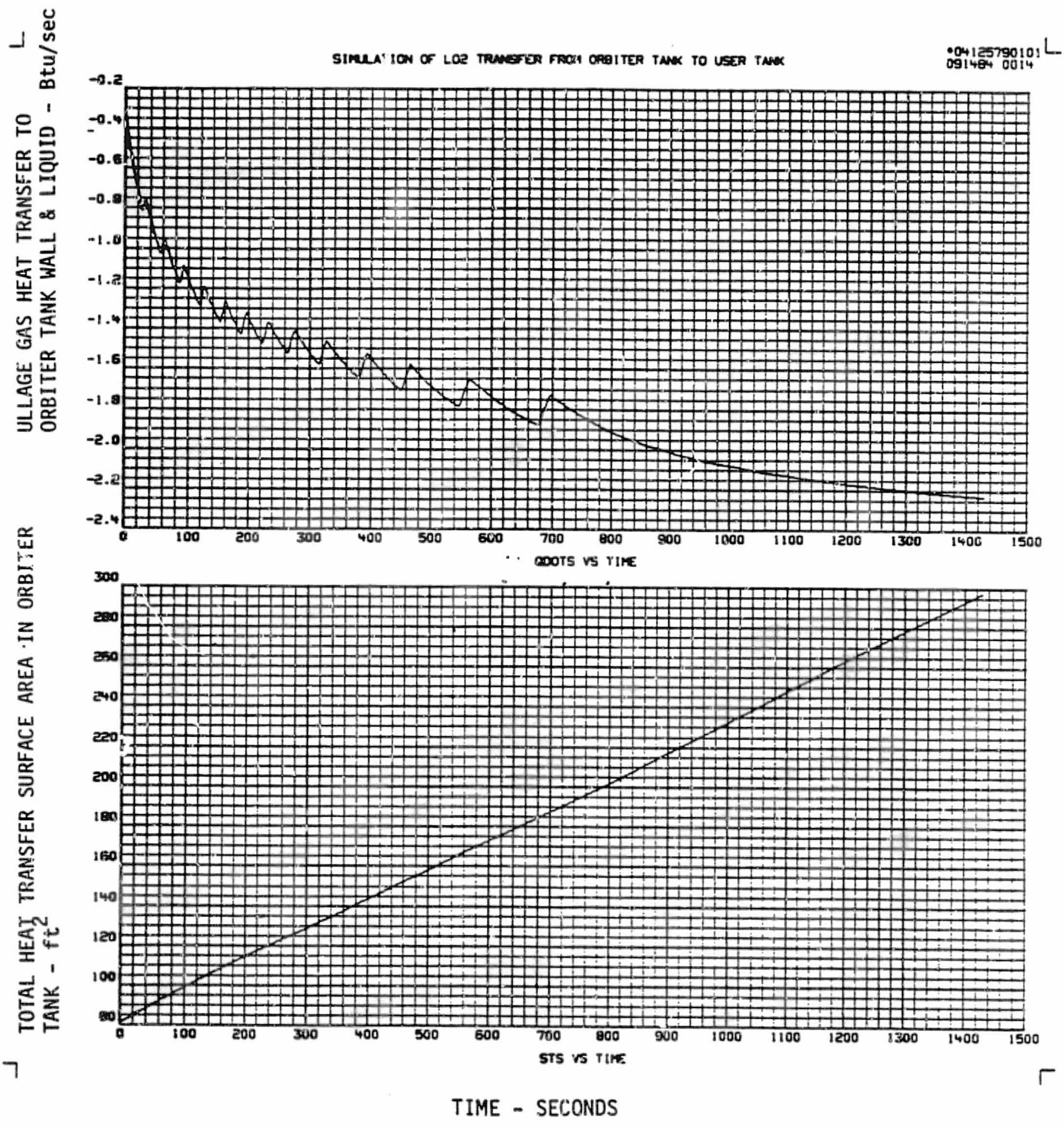

SIMULATION OF LO2 TRANSFER FROM ORBITER TANK TO USER TANK
*04125790101
091484 0014
ULLAGE GAS HEAT TRANSFER TO
ORBITER TANK WALL & LIQUID - Btu/sec
-0.2
-0.4
-0.6
-0.8
-1.0
-1.2
-1.4
-1.6
-1.8
-2.0
-2.2
-2.4
0
100
200
300
400
500
600
700
800
900
1000
1100
1200
1300
1400
1500
QDOTS VS TIME
TOTAL HEAT TRANSFER SURFACE AREA IN ORBITER
TANK - ft2
300
280
260
240
220
200
180
160
140
120
100
80
STS VS TIME
TIME - SECONDS

Figure A-14

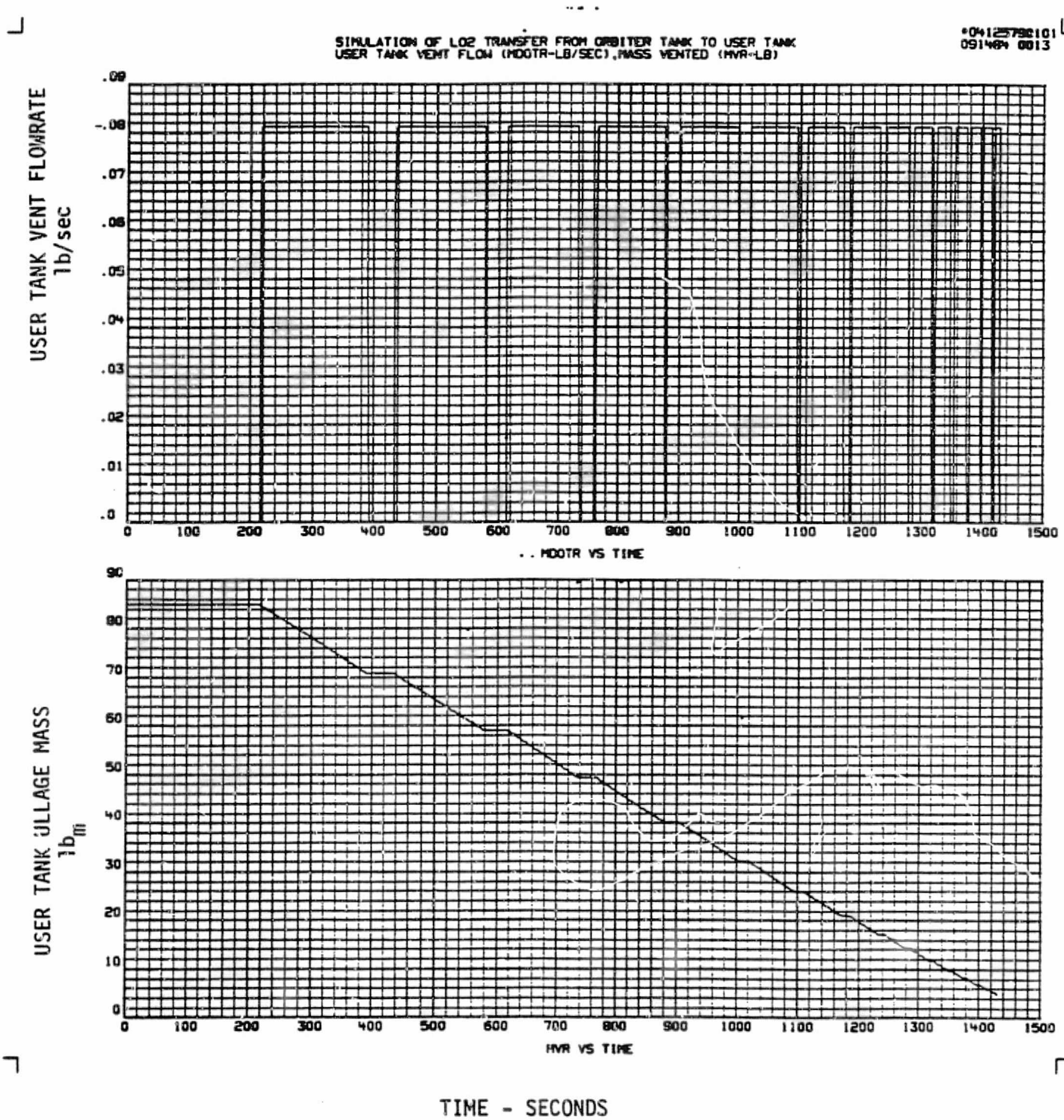

SIMULATION OF LO2 TRANSFER FROM ORBITER TANK TO USER TANK
USER TANK VENT FLOW (MDOTR-LB/SEC),MASS VENTED (MVR-LB)
USER TANK VENT FLOWRATE lb/sec
.. MDOTR VS TIME
USER TANK ULLAGE MASS lb_m
MVR VS TIME
TIME - SECONDS

Figure A-15

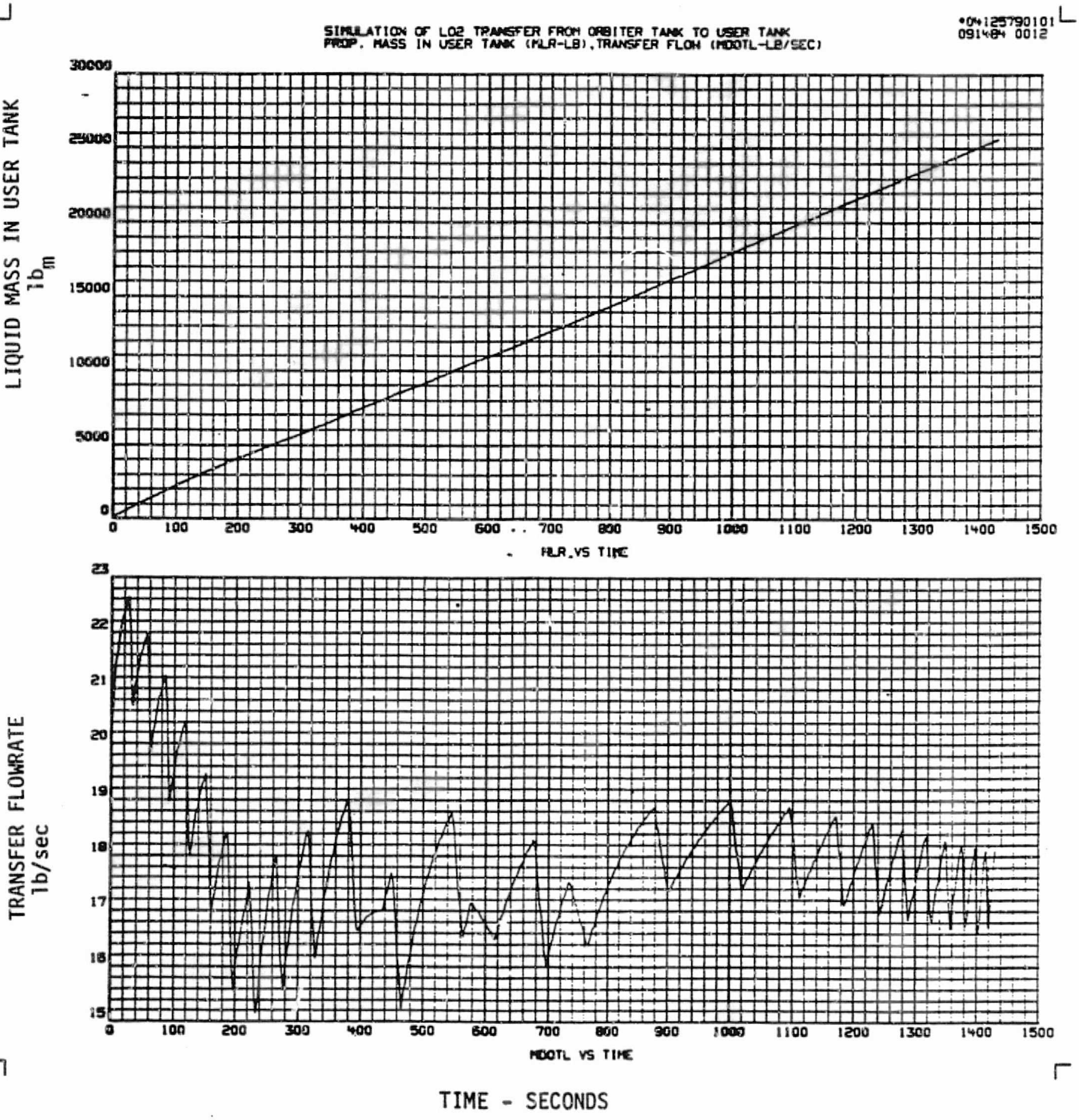

SIMULATION OF LO2 TRANSFER FROM ORBITER TANK TO USER TANK
PROP. MASS IN USER TANK (MLR-LB),TRANSFER FLOW (MDOTL-LB/SEC)
*04125790101
091484 0012
LIQUID MASS IN USER TANK
lb_m
30000
25000
20000
15000
10000
5000
0
0 100 200 300 400 500 600 700 800 900 1000 1100 1200 1300 1400 1500
MLR VS TIME
TRANSFER FLOWRATE
lb/sec
23
22
21
20
19
18
17
16
15
0 100 200 300 400 500 600 700 800 900 1000 1100 1200 1300 1400 1500
MDOTL VS TIME
TIME - SECONDS

Figure A-16

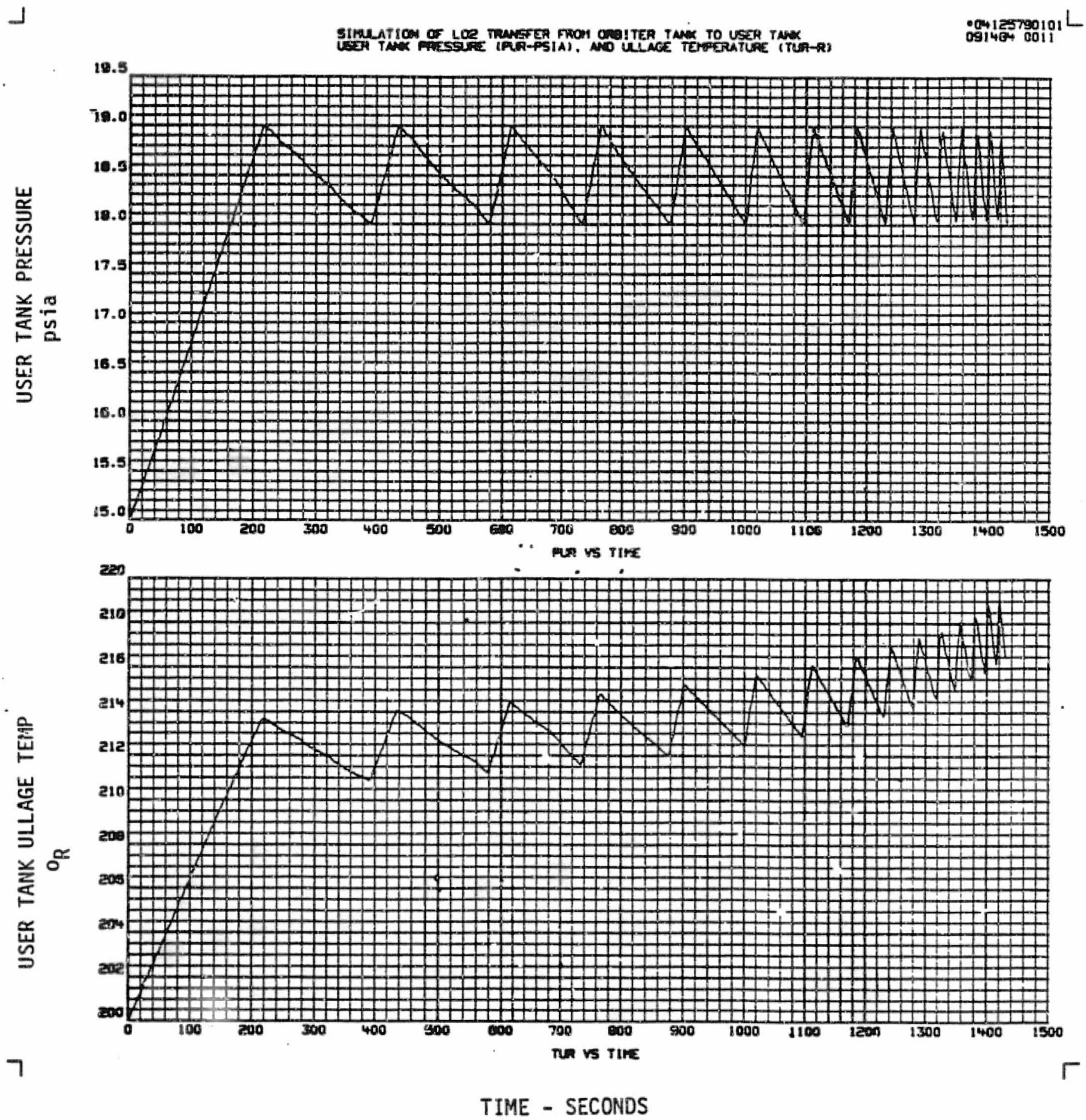
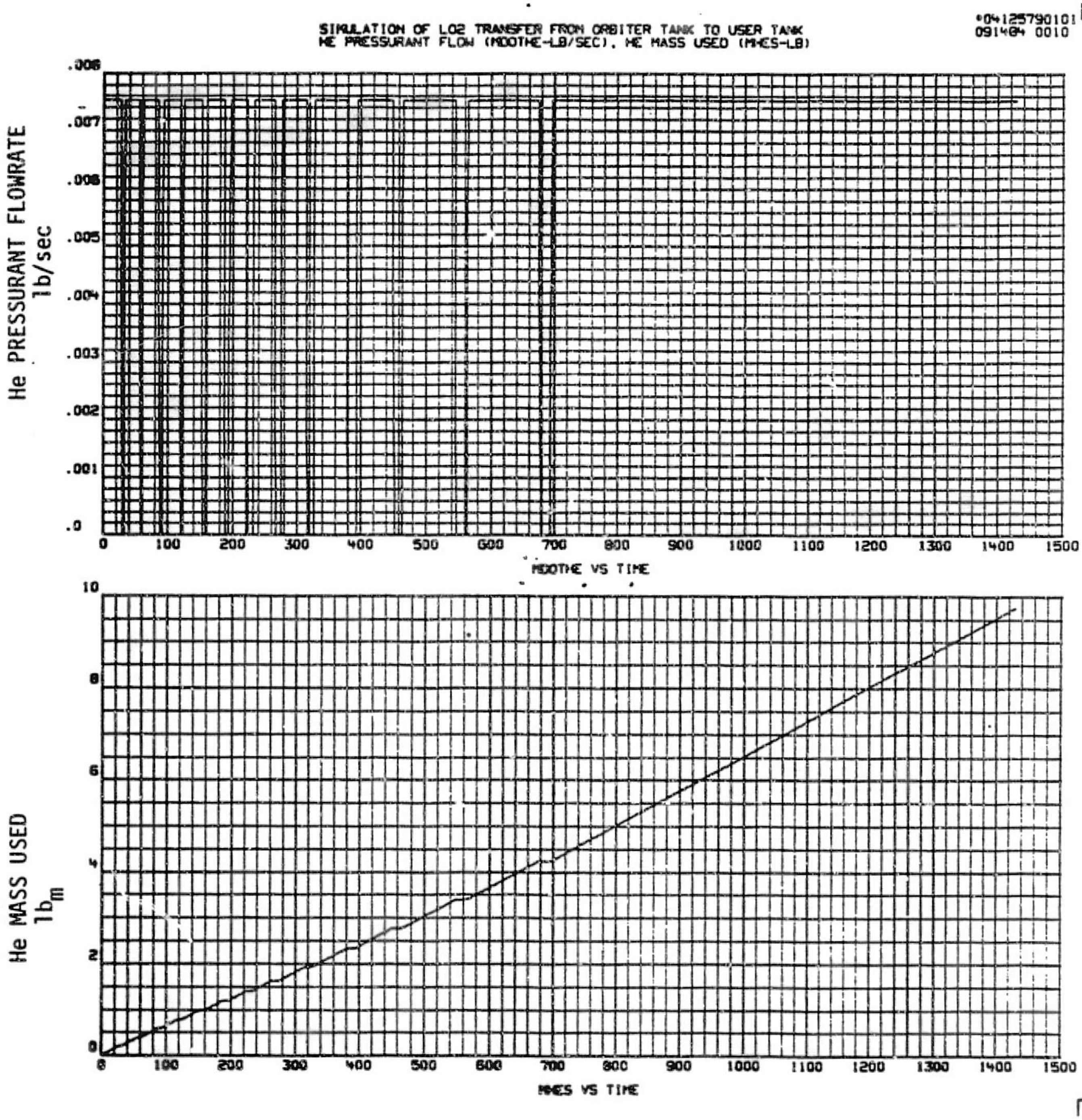
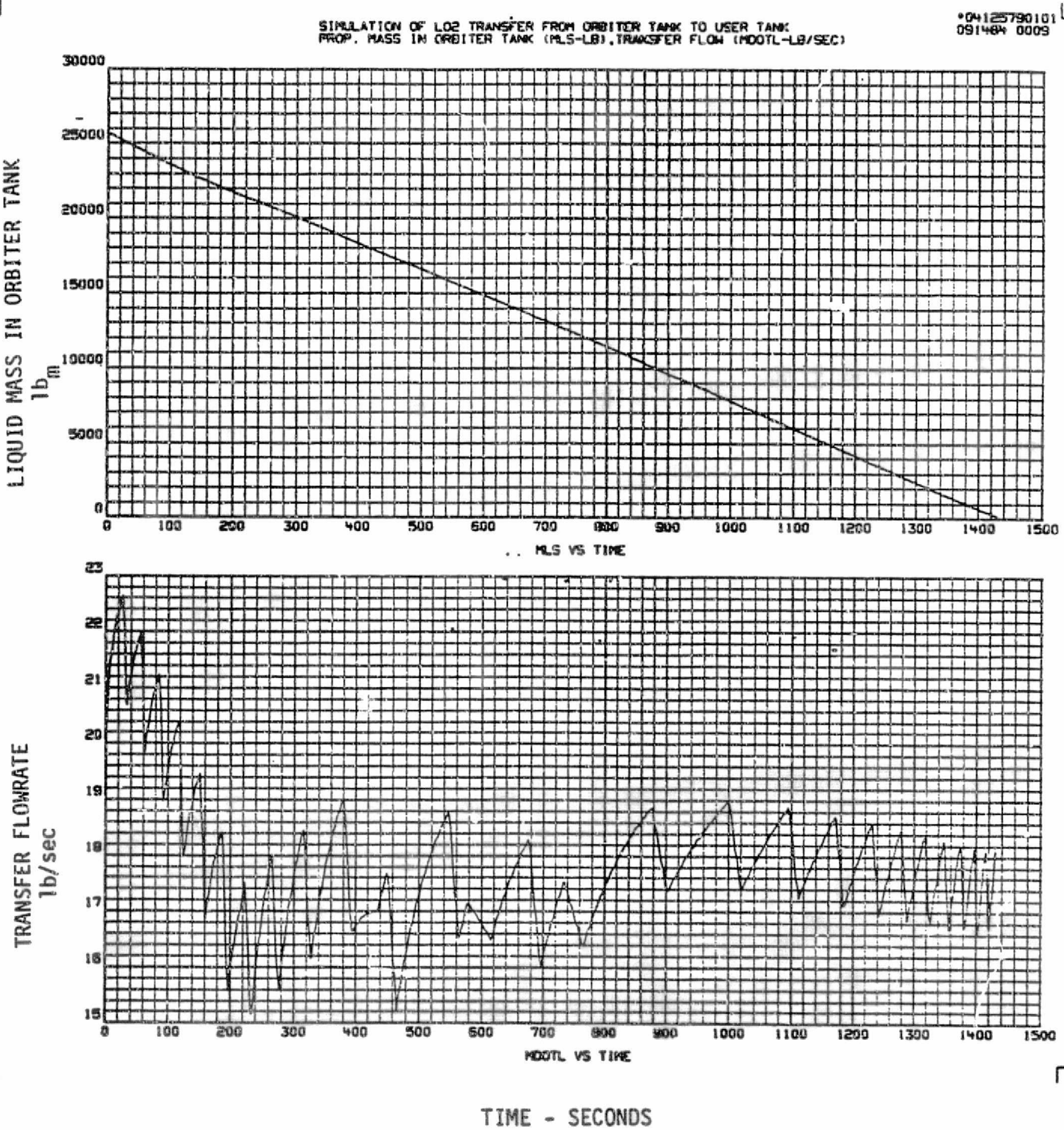

SIMULATION OF LO2 TRANSFER FROM ORBITER TANK TO USER TANK
USER TANK PRESSURE (PUR-PSIA), AND ULLAGE TEMPERATURE (TUR-R)
USER TANK PRESSURE psia
PUR VS TIME
USER TANK ULLAGE TEMP ^oR
TUR VS TIME
TIME - SECONDS

Figure A-17



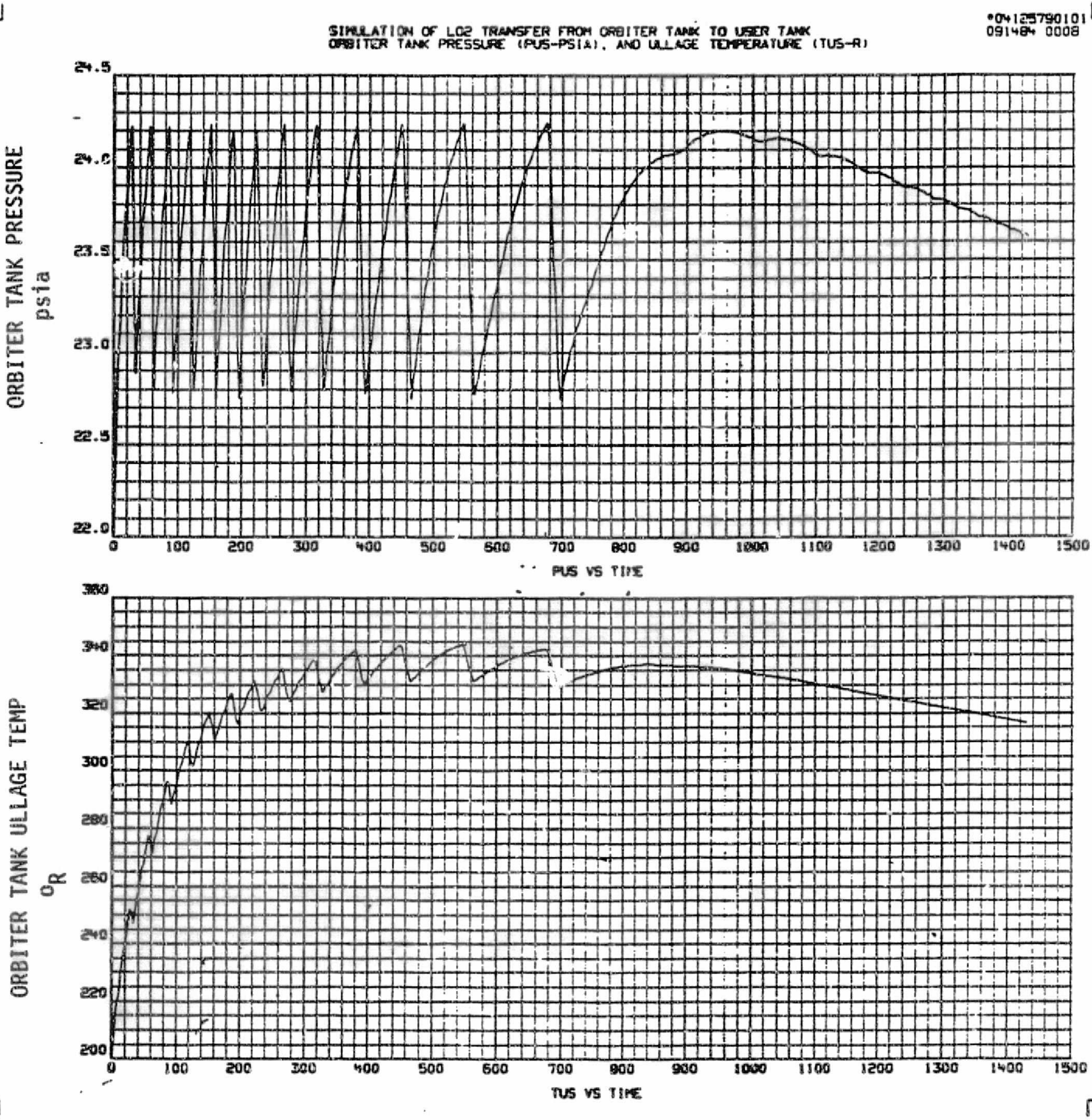
TIME - SECONDS

Figure A-18



SIMULATION OF LO2 TRANSFER FROM ORBITER TANK TO USER TANK
PROP. MASS IN ORBITER TANK (MLS-LB),TRANSFER FLOW (MDOTL-LB/SEC)
*04125790101
091484 0009
LIQUID MASS IN ORBITER TANK
lb_m
MLS VS TIME
TRANSFER FLOWRATE
lb/sec
MDOTL VS TIME
TIME - SECONDS

Figure A-19



TIME - SECONDS

Figure A-20

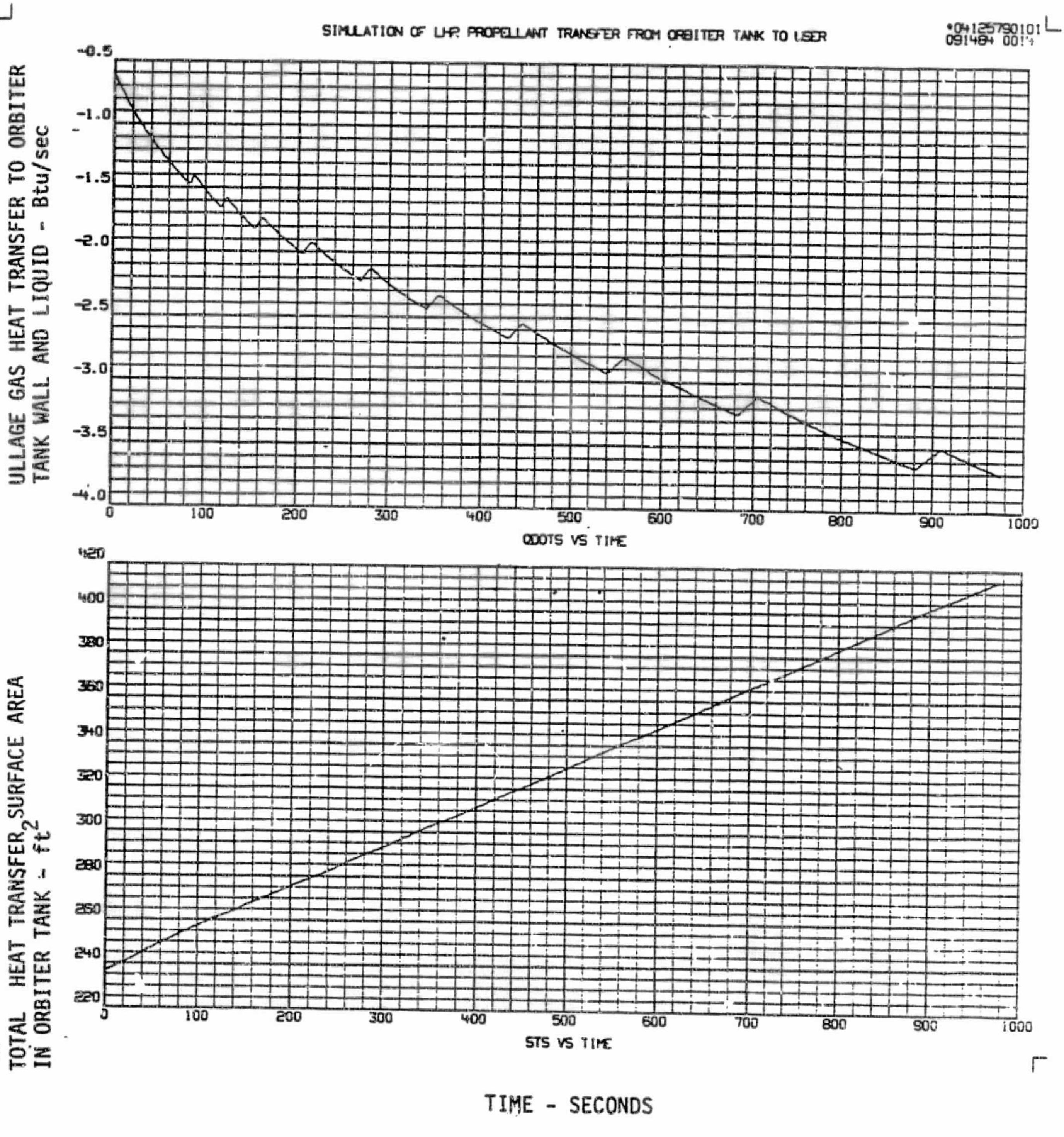
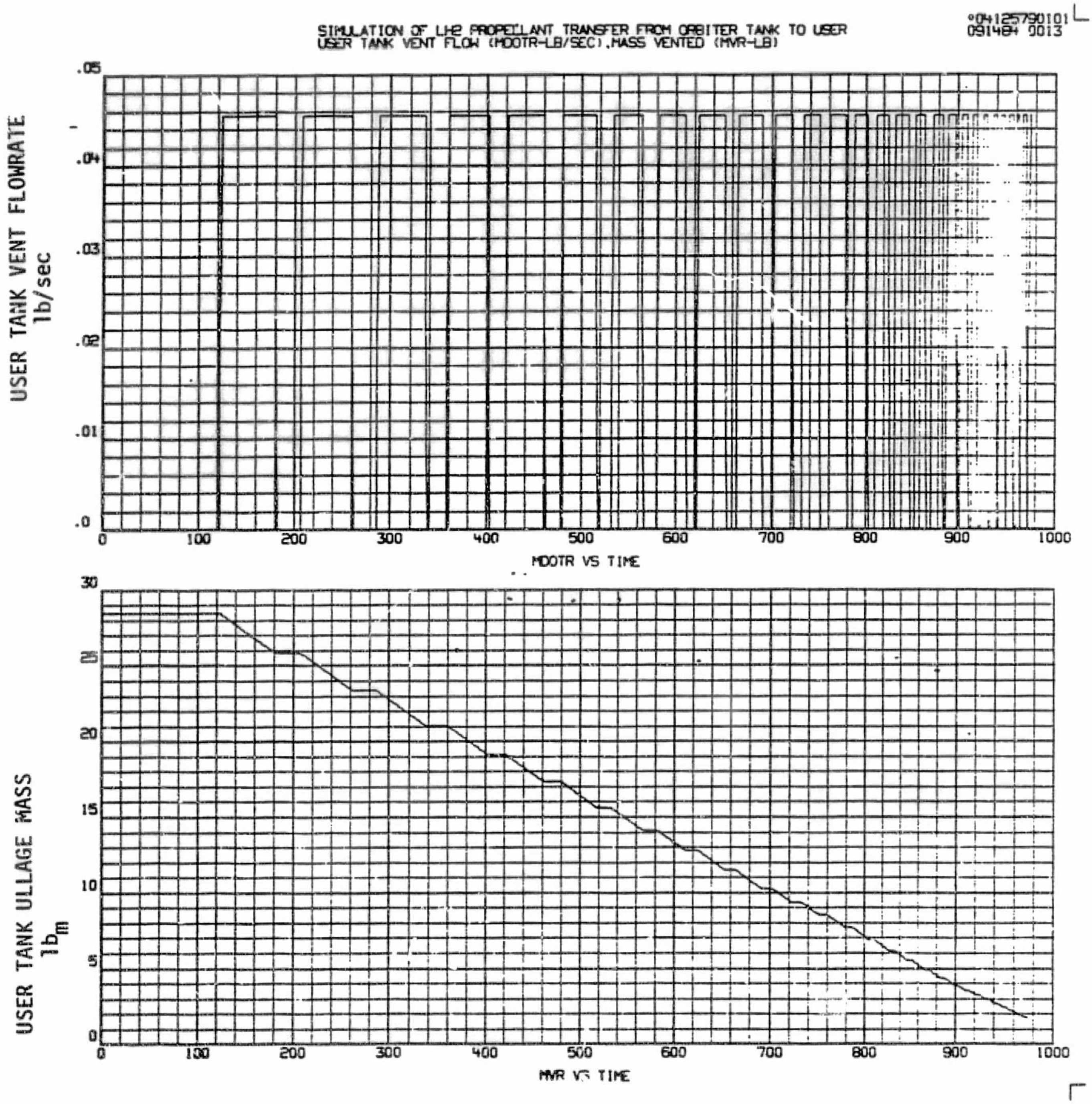
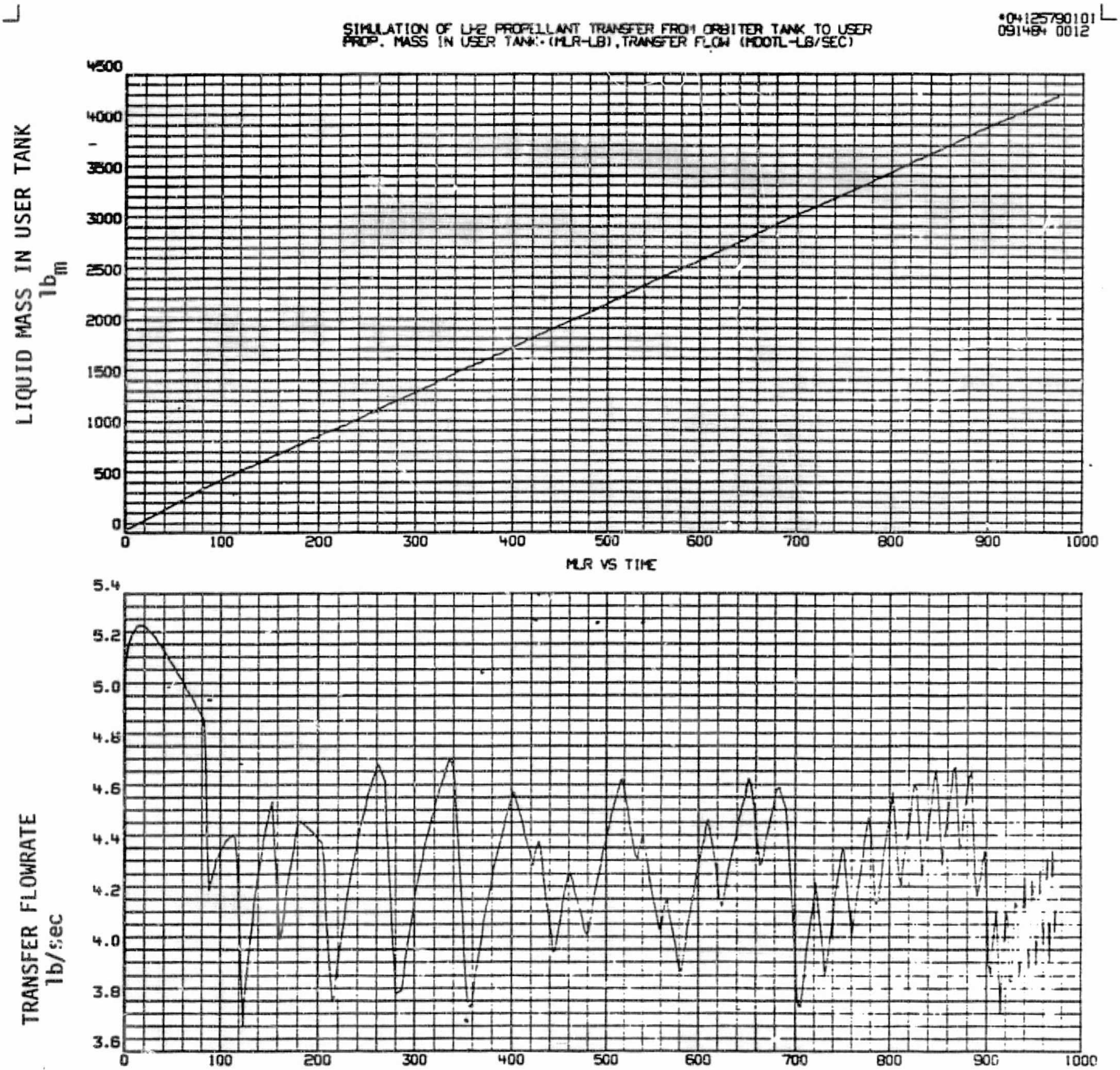

SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
ULLAGE GAS HEAT TRANSFER TO ORBITER TANK WALL AND LIQUID - Btu/sec
QDOTS VS TIME
TOTAL HEAT TRANSFER SURFACE AREA IN ORBITER TANK - ft^2
STS VS TIME
TIME - SECONDS

Figure A-21



TIME - SECONDS

Figure A-22



SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
PROP. MASS IN USER TANK-(MLR-LB),TRANSFER FLOW (MDOTL-LB/SEC)
*04125790101
091484 0012
LIQUID MASS IN USER TANK
lb_m
4500
4000
3500
3000
2500
2000
1500
1000
500
0
0
100
200
300
400
500
600
700
800
900
1000
MLR VS TIME
TRANSFER FLOWRATE
lb/sec
5.4
5.2
5.0
4.8
4.6
4.4
4.2
4.0
3.8
3.6
MDOTL VS TIME
TIME - SECONDS

Figure A-23

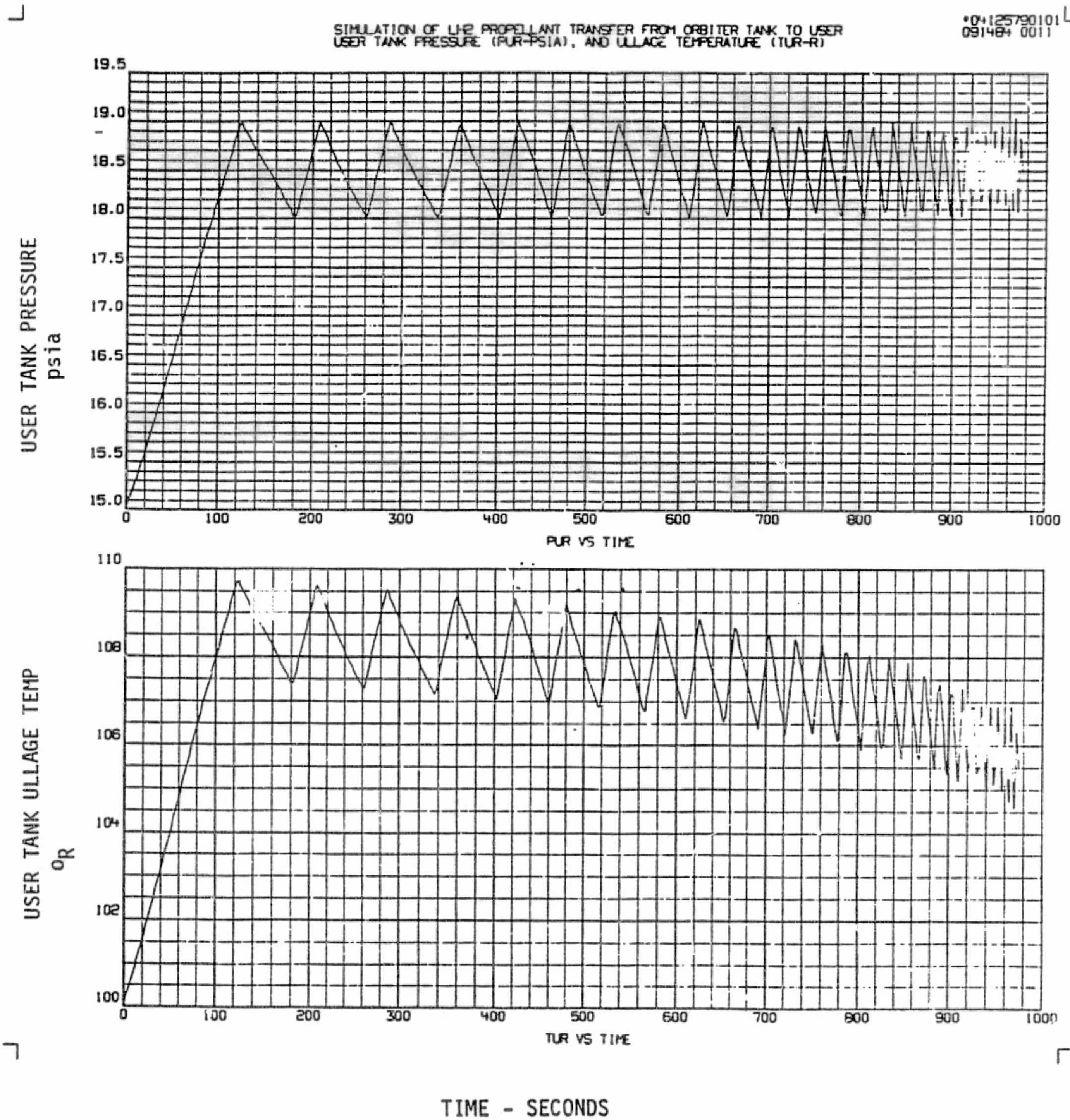

SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
USER TANK PRESSURE (PUR-PSIA), AND ULLAGE TEMPERATURE (TUR-R)
*04125790101
091484 0011
USER TANK PRESSURE
psia
PUR VS TIME
USER TANK ULLAGE TEMP
oR
TUR VS TIME
TIME - SECONDS

Figure A-24

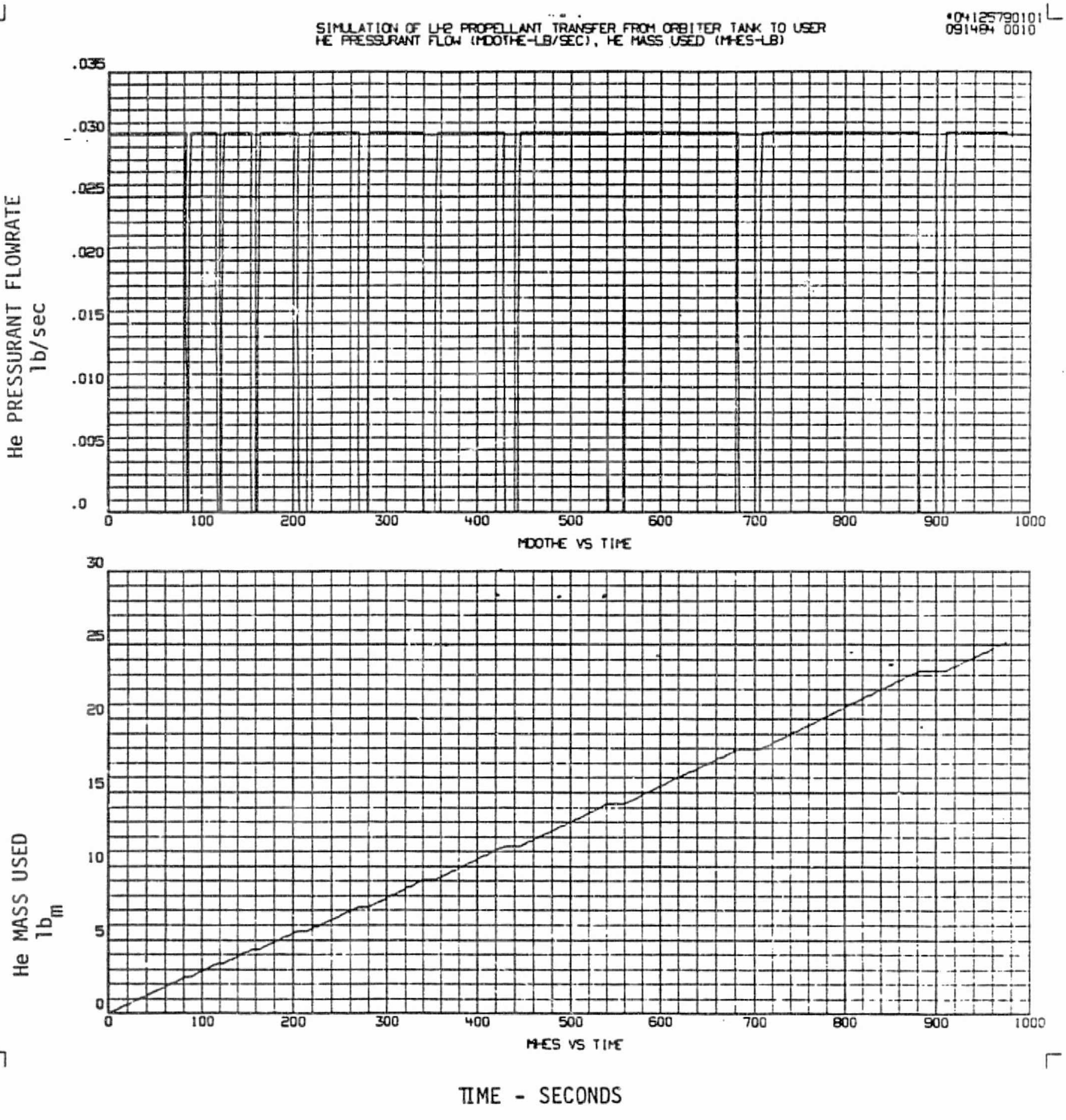
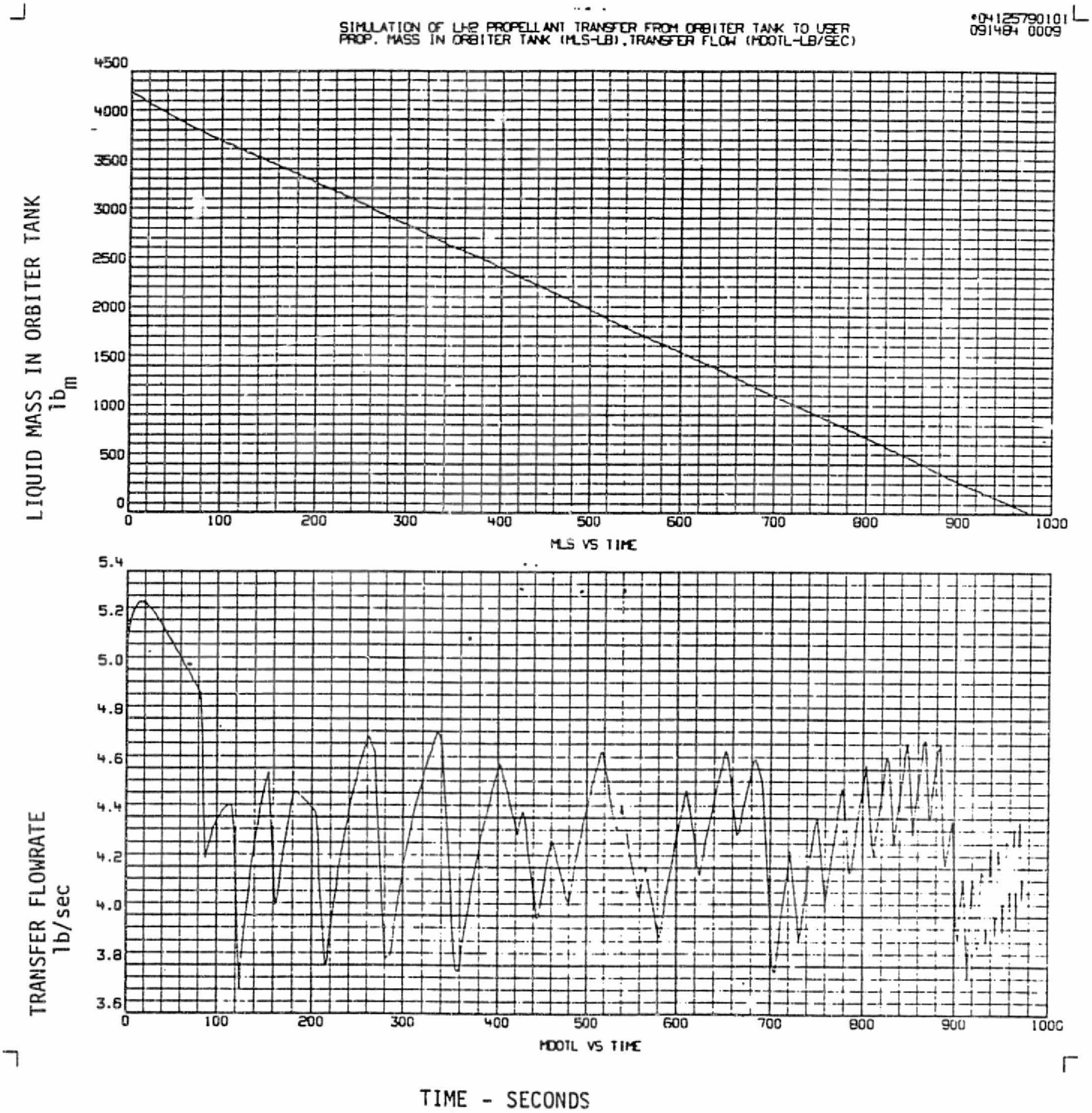

SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
HE PRESSURANT FLOW (MDOTHE-LB/SEC), HE MASS USED (MHES-LB)
*04125790101
091484 0010
He PRESSURANT FLOWRATE
lb/sec
.035
.030
.025
.020
.015
.010
.005
.0
0
100
200
300
400
500
600
700
800
900
1000
MDOTHE VS TIME
He MASS USED
lb_m
30
25
20
15
10
5
0
MHES VS TIME
TIME - SECONDS

Figure A-25



SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
PROP. MASS IN ORBITER TANK (MLS-LB), TRANSFER FLOW (MDOTL-LB/SEC)
*04125790101
091484 0009
LIQUID MASS IN ORBITER TANK
lb_m
MLS VS TIME
TRANSFER FLOWRATE
lb/sec
MDOTL VS TIME
TIME - SECONDS

Figure A-26

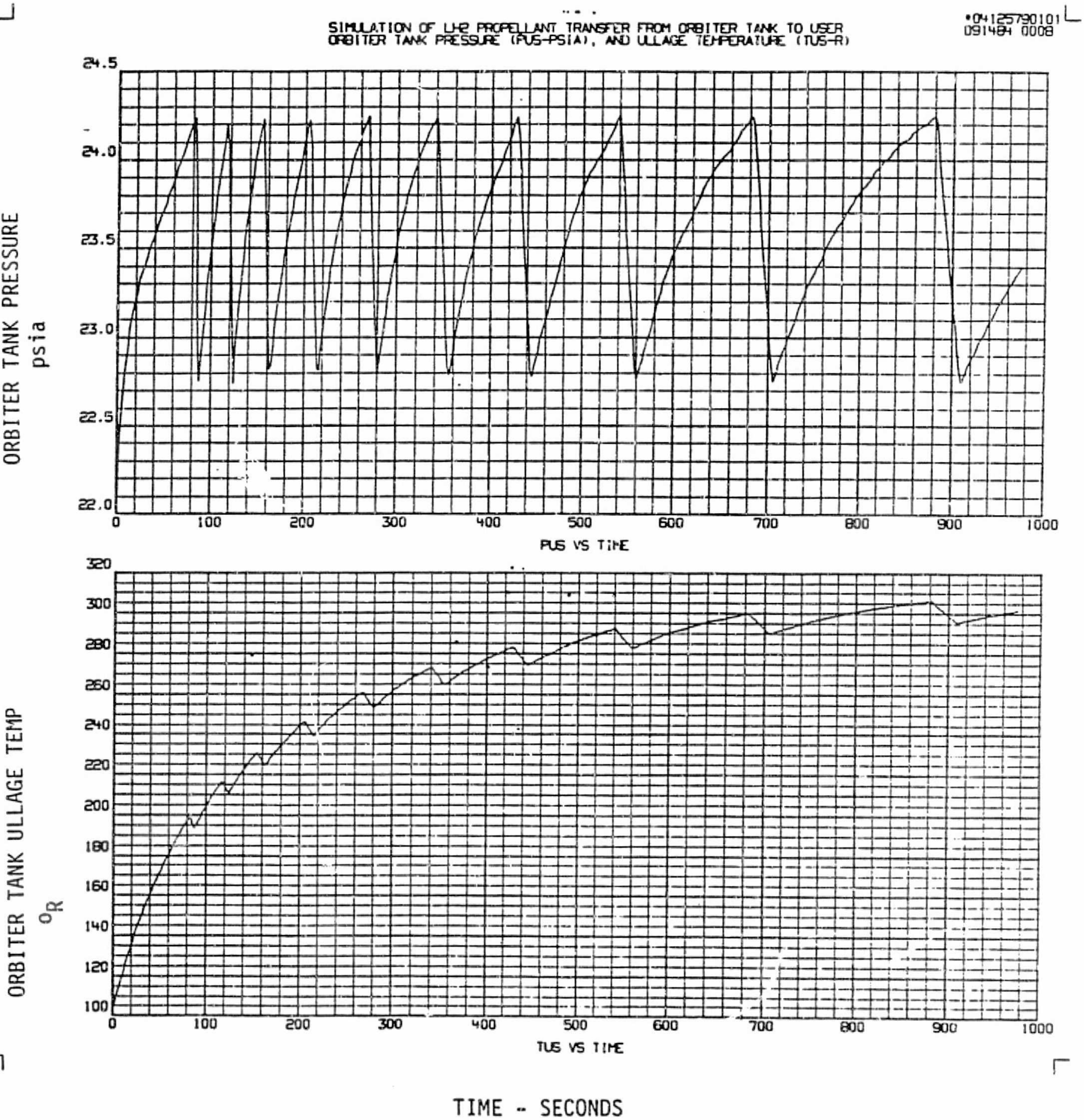

SIMULATION OF LH2 PROPELLANT TRANSFER FROM ORBITER TANK TO USER
ORBITER TANK PRESSURE (PUS-PSIA), AND ULLAGE TEMPERATURE (TUS-R)
*04125790101
091484 0008
ORBITER TANK PRESSURE psia
24.5
24.0
23.5
23.0
22.5
22.0
0
100
200
300
400
500
600
700
800
900
1000
PUS VS TIME
ORBITER TANK ULLAGE TEMP ^oR
320
300
280
260
240
220
200
180
160
140
120
100
TUS VS TIME
TIME - SECONDS

Figure A-27

A.3 STORABLE PROPELLANT SCAVENGING SCHEMATICS

Figure A-28 is the basic schematic for OMS propellant and pressurant scavenging without the payload bay tank system. The addition of gas compressers to assist pressurant transfer from the OMS helium bottles and of a QD purge network is shown in Figure A-29. Figure A-30 shows the inclusion of waste disposal tanks for the QD purge residuals.

PROPELLANT SCAVENGING SCHEMATIC

OMS SCAVENGING
NO PURGE
HELIUM RESUPPLY - BLOWDOWN

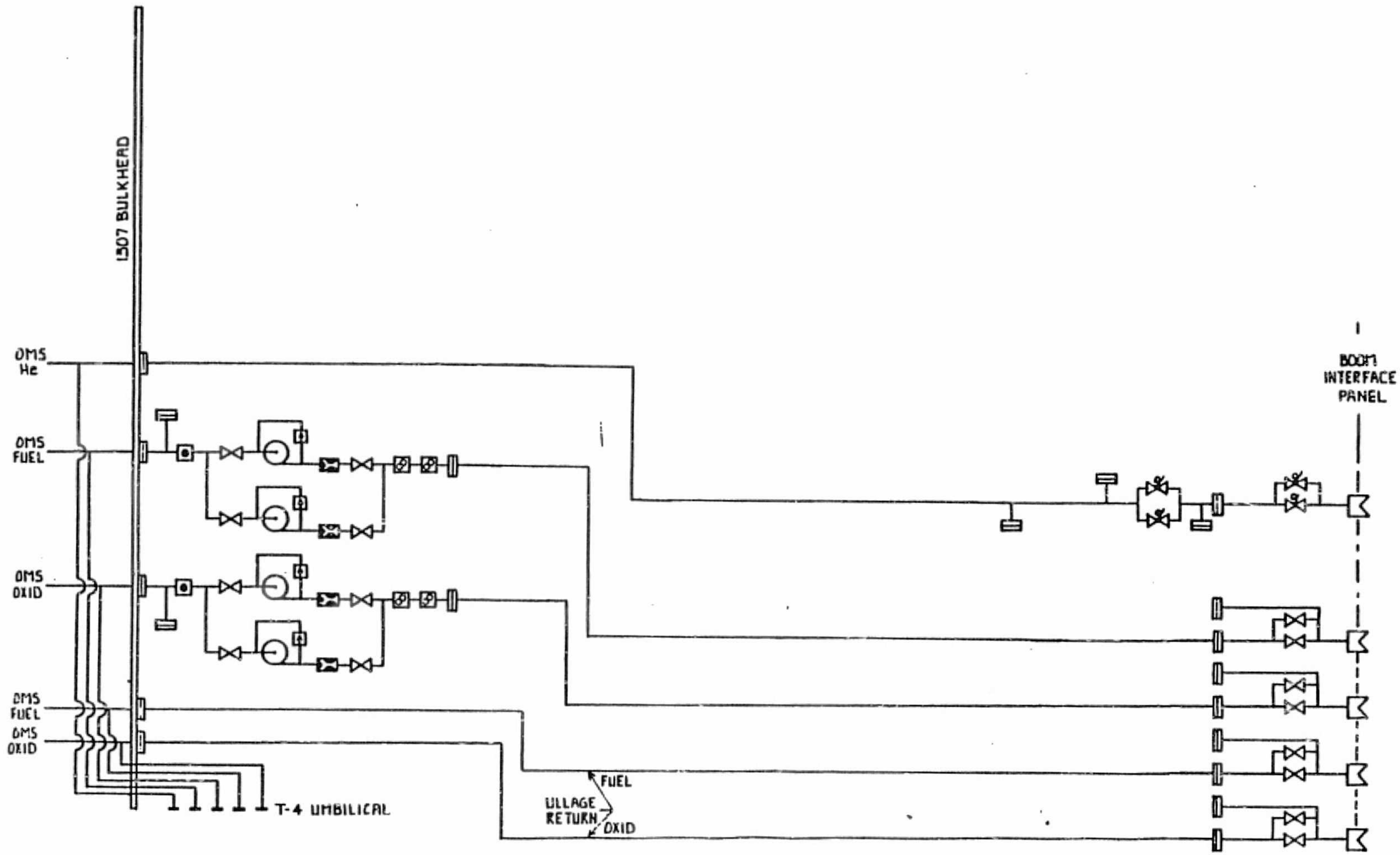


Figure A-28

PROPELLANT SCAVENGING SCHEMATIC

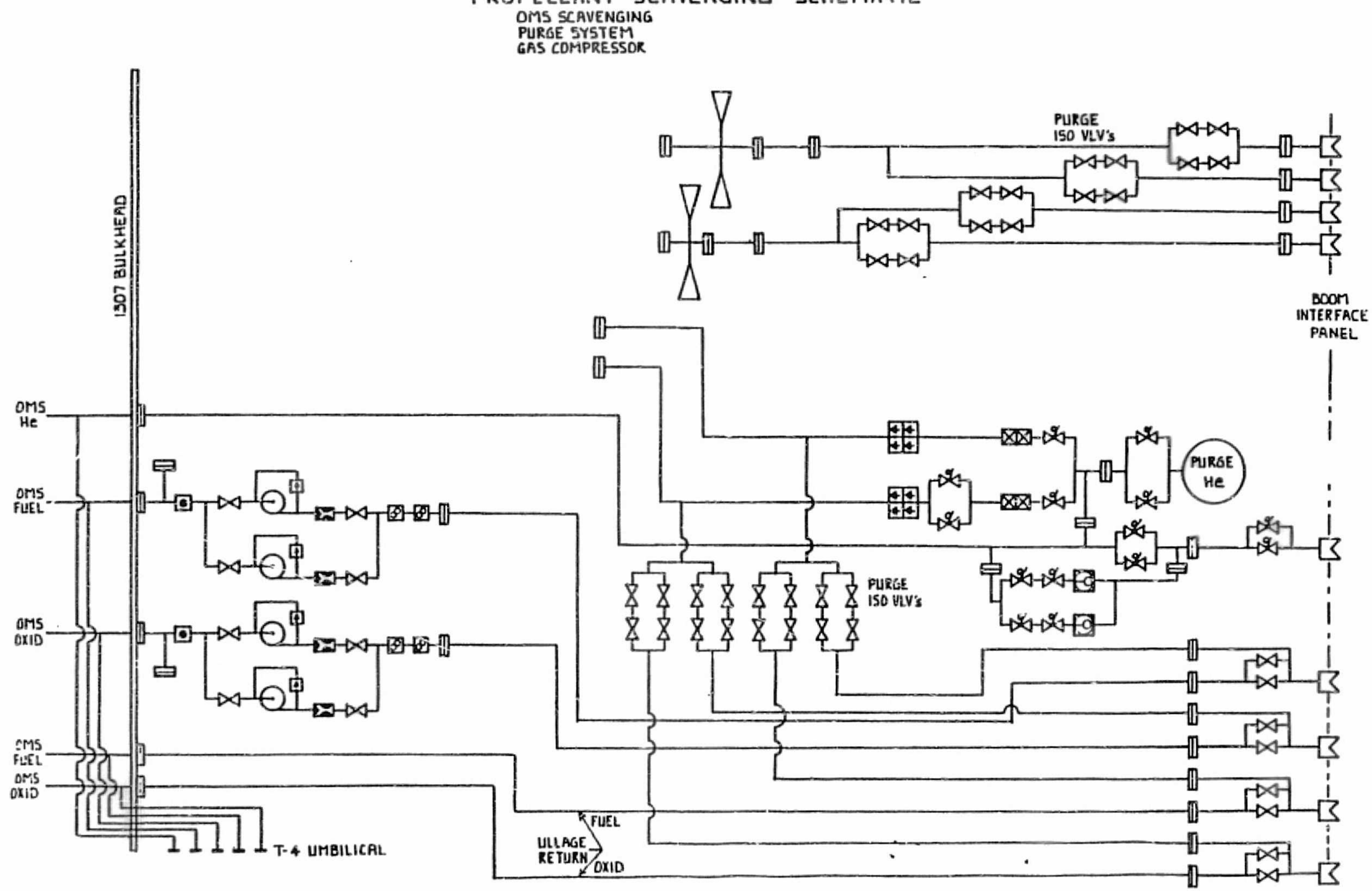


Figure A-29

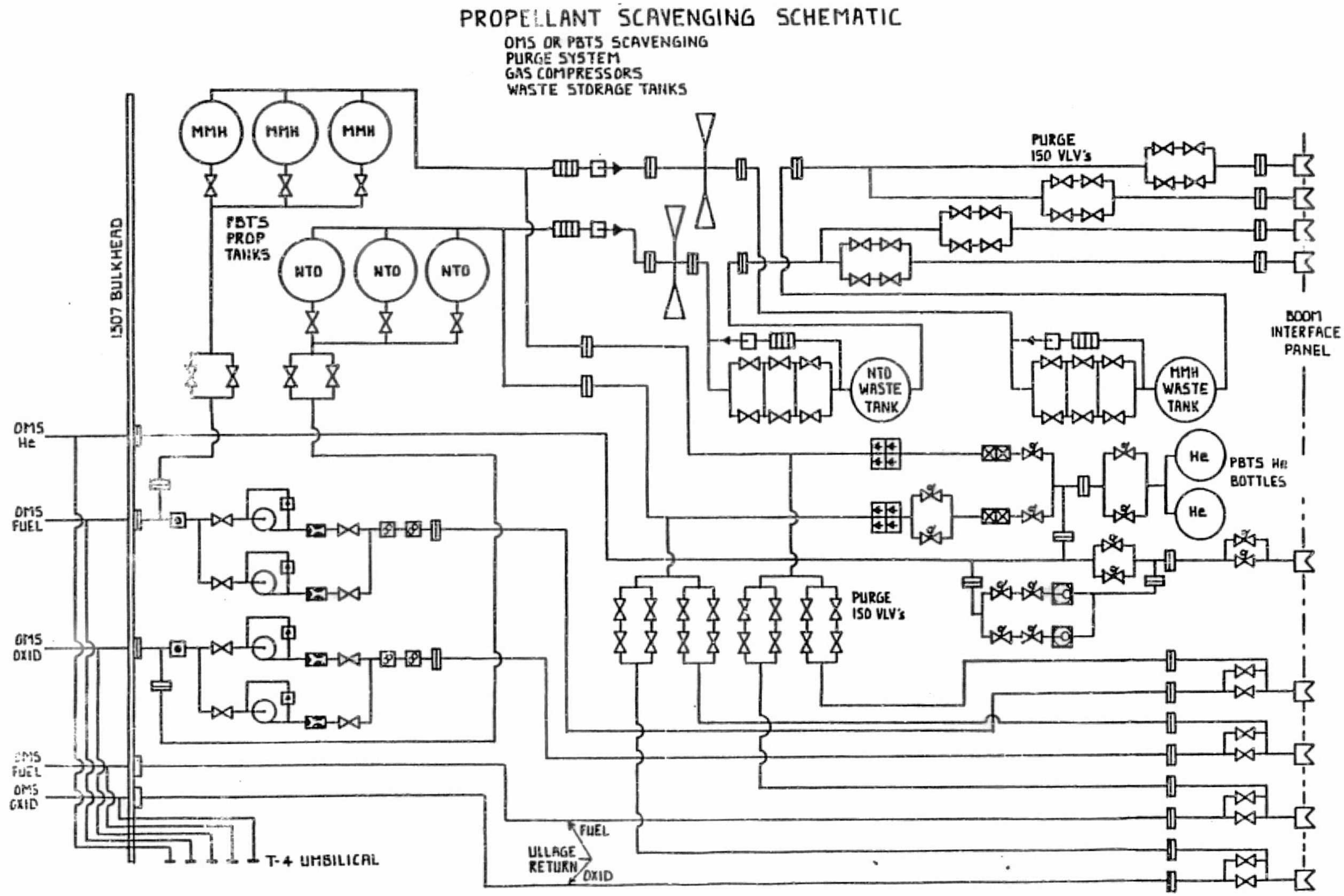

PROPELLANT SCAVENGING SCHEMATIC
OMS OR PBTS SCAVENGING
PURGE SYSTEM
GAS COMPRESSORS
WASTE STORAGE TANKS
MMH
MMH
MMH
PBTS
PROP
TANKS
NTO
NTO
NTO
1307 BULKHEAD
PURGE
ISO VLV's
BOOM
INTERFACE
PANEL
NTO
WASTE
TANK
MMH
WASTE
TANK
He
He
PBTS He
BOTTLES
PURGE
ISO VLV's
OMS
He
OMS
FUEL
OMS
OXID
OMS
FUEL
OMS
OXID
T-4 UMBILICAL
FUEL
ULLAGE
RETURN
OXID

Figure A-30

A.4 CRYOGENIC COMPONENT DESCRIPTIONS

Physical descriptions of the cryogenic components and their performance characteristics are provided in this section.

Check Valves- All check valves (ME284-0472-00XX) are fully certified for Shuttle use for a minimum of 100 missions and are manufactured by Circle Seal. To assure optimum systems performance and compatibility, each valve will be re-evaluated and some design improvements will be considered. If warranted, delta qualification testing will be performed.

For physical description and performance characteristics, refer to the following pages.

COMPONENT: VALVE, CHECK, HELIUM

(MC284-0472)

DESCRIPTION:

THE COMPONENT IS A SPRING-LOADED CLOSE, PRESSURE ACTUATED OPEN CHECK VALVE WHICH INCORPORATES A SELF-CENTERING TEFLON "O" RING TRAPPED BETWEEN TWO CONICAL SURFACES TO PERFORM THE SEALING. THE SEAL SELF-CENTERING CAPABILITY, COUPLED WITH THE RELATIVE SOFTNESS OF THE MATERIAL, PROVIDE GOOD SEALING PERFORMANCE AT AMBIENT AND CRYOGENIC OPERATING TEMPERATURES.

FUNCTION:

TYPE I - 1/4" DIAMETER CHECK VALVE

THE TYPE I CHECK VALVE IS USED IN THE LO_2 AND LH_2 REPRESSURIZATION SYSTEM WHICH INTERCONNECTS THE PREPRESSURIZATION LINES TO THE FEEDLINE MANIFOLD. AFTER LH_2 LOADING AND THE START OF THE LH_2 RECIRCULATION, THE CHECK VALVE PREVENTS THE ENTRANCE OF LIQUID HYDROGEN INTO THE LH_2 REPRESSURIZATION REGULATOR. DURING MAIN ENGINE START, THE CHECK VALVE PREVENTS PRESSURIZING OF THE MANIFOLDS THROUGH THE REPRESS SYSTEM.

Space Transportation System
Development & Production Division
Space Systems Group
Rockwell International

COMPONENT: VALVE, CHECK, HELIUM

(MC284-0472)

FUNCTION:

TYPE II - 3/8" DIAMETER CHECK VALVE

THE TYPE II CHECK VALVE IS USED IN THE HELIUM STORAGE FILL, THE SSME ENGINE PNEUMATIC PANELS AND THE VALVE ACTUATION PNEUMATIC PANELS. A TOTAL OF 16 VALVES ARE USED. IN THE HELIUM STORAGE FILL SYSTEM, THE CHECK VALVES PREVENT THE LOSS OF HELIUM IN THE STORAGE TANKS IN THE EVENT OF A LEAK UPSTREAM OF THE CHECK VALVES. IN THE SSME PNEUMATIC AND VALVE ACTUATION PANELS, THE CHECK VALVES ALLOW THE PANELS TO BE INTERCONNECTED SO THE HELIUM SUPPLY IN ALL THE STORAGE TANKS CAN BE USED INDIVIDUALLY, INDEPENDENTLY OR INTERCONNECTED AS DEMANDED OR REQUIRED.

TYPE III - 1/2" DIAMETER CHECK VALVE

THE TYPE III CHECK VALVE IS USED IN THE LO_2 MANIFOLD AND FEEDLINE REPRESSURIZATION SYSTEM AND IN THE VALVE ACTUATION PNEUMATIC PANEL. IN THE REPRESSURIZATION SYSTEM, THE CHECK VALVE PREVENTS THE ENTRANCE OF LIQUID OXYGEN FROM THE LO_2 FEEDLINE MANIFOLD TO THE REPRESS REGULATOR.

IN THE VALVE ACTUATION SUPPLY SYSTEM, THE CHECK VALVE IS USED TO ISOLATE THE SUPPLY LINE SURGE CHAMBERS FROM THE MAIN STORAGE SUPPLY. THE SURGE CHAMBER SUPPLY IS PROVIDED FOR OPERATION OF THE 17" DISCONNECTS AND THE PREVALVES IN THE EVENT THAT THE MAIN SUPPLY PRESSURE IS NOT AVAILABLE DURING MECO.

COMPONENT: VALVE, CHECK, HELIUM

(MC284-0472)

TYPE IV - 3/4" DIAMETER CHECK VALVE

THE TYPE IV CHECK VALVE IS USED IN THE VALVE ACTUATION PNEUMATIC SUPPLY PANEL, THE LH_2 MANIFOLD AND FEEDLINE REPRESSURIZATION, AND THE RTLS SYSTEM. IN THE VALVE ACTUATION PNEUMATIC SUPPLY PANEL, THE CHECK VALVE ISOLATES THE VALVE ACTUATION REGULATOR FROM VALVE ACTUATION LINE FEED SYSTEM.

IN THE LH_2 MANIFOLD REPRESSURIZATION SYSTEM, THE CHECK VALVE ISOLATES THE REPRESSURIZATION REGULATOR FROM LIQUID HYDROGEN IN THE LH_2 MANIFOLD. IN THE RTLS, THE CHECK VALVE ISOLATES THE RTLS HELIUM SUPPLY SYSTEM FROM LIQUID HYDROGEN DURING A VENT CYCLE OF THE LH_2 MANIFOLD RELIEF VALVE.

TYPE IV (MODIFIED) 3/4" DIAMETER CHECK VALVE

THE TYPE IV (MODIFIED) CHECK VALVE IS USED IN THE SSME PNEUMATIC SUPPLY PANELS AND ISOLATES THE PNEUMATIC PANEL REGULATOR FROM THE ENGINE HELIUM SUPPLY LINES. IN THE EVENT OF A LEAK UPSTREAM OF A CHECK VALVE, THE LOSS OF HELIUM IS PREVENTED.

TYPE V - 1.0" DIAMETER CHECK VALVE

THE TYPE V CHECK VALVE IS USED IN THE GO_2 AND GH_2 HELIUM PREPRESSURIZATION SYSTEM TO PROVIDE REDUNDANCY TO THE CHECK VALVE IN THE T-0 UMBILICAL DISCONNECT. AFTER ENGINE START, THE CHECK VALVES PREVENT THE LOSS OF GO_2/GH_2 PRESSURANT IN THE EVENT OF A LEAK IN THE LINES OR DISCONNECT UPSTREAM OF THE CHECK VALVES.

COMPONENT: VALVE, CHECK, HELIUM

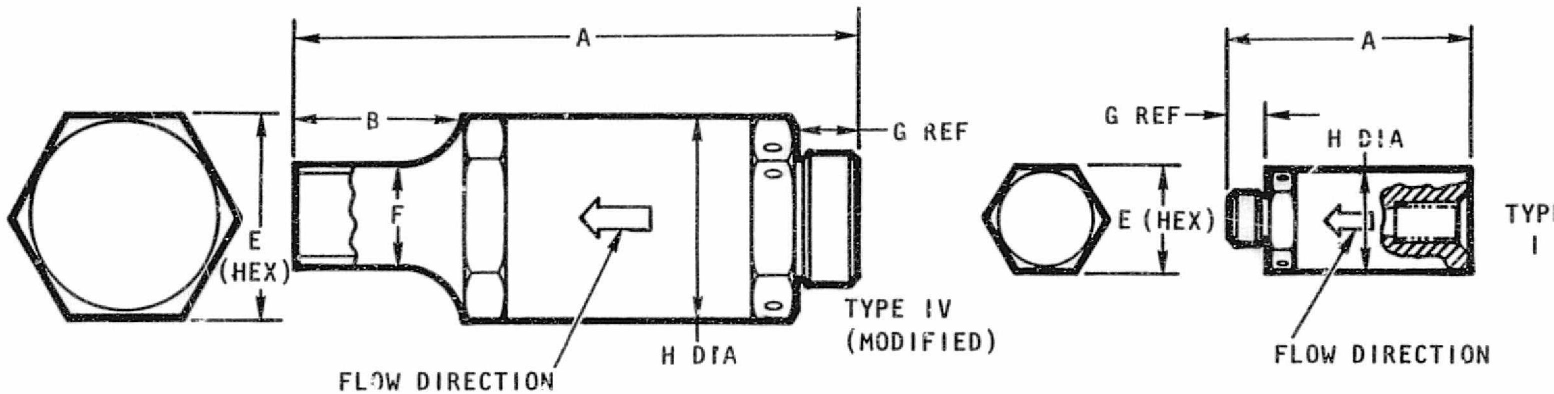
(MC284-0472)

DESIGN PARAMETERS:

TYPE:	I	II	III	IV	IV (MODIFIED)	V
CRACKING PRESSURE	5 PSID MAX.	5 PSID MAX.	5 PSID MAX.	5 PSID MAX.	.06 PSID MAX.	5 PSID MAX.
RESEAT PRESSURE	2 PSID MIN.	2 PSID MIN.	2 PSID MIN.	2 PSID MIN.	.01 PSID MIN.	2 PSID MIN.
OPERATING PRESSURE (PSIG)	650	4500	850	850	850	750
OPERATING TEMPERATURE (°F)	-423 TO +250	-160 TO +250	-320 TO +250	-423 TO +250	-160 TO +250	-100 TO +250
FLUID TEMPERATURE (°F)	-160 TO +130	-160 TO +130	-160 TO +140	-160 TO +140	-160 TO +140	+20 TO +120
FLOW RATE (HELIUM, LBS/SEC)	.005	.05	.08	0.202	.202	1.5
PRESSURE DROP AT RATE FLOW	11 PSID MAX.	10 PSID MAX.	10 PSID MAX.	44 PSID MAX.	15 PSID MAX.	167 PSID MAX.

Space Transportation System
Development & Production Division
Space Systems Group
Rockwell International

COMPONENT: VALVE, CHECK, HELIUM
(MC284-0472)



NOTE: VALVE TYPES SPECIFIED SHALL FUNCTION WHEN ORIENTED IN ANY POSITION WITHIN VEHICLE

VALVE ENVELOPE

TYPE	CONTROL NO.	I&T	A	B	OUTLET	INLET	E MAX	F DIA	G REF	H DIA	T THICKNESS
I	ME284-0472	T_S	1.850 MAX 1.760 MIN	-- --	MP273-0002-1004	MS33649-04	0.77	--	0.290 MAX 0.260 MIN	0.755 MAX 0.735 MIN	--
IV	ME284-0472	T_S	4.347 MAX 4.307 MIN	1.393 MAX 1.333 MIN	BRAZE STUB	MP273-0002-1012	1.875	0.754 MAX 0.750 MIN	0.427 MAX 0.397 MIN	1.890 MAX	0.028

Pneumatic Solenoid Valves - The pneumatic solenoid valves selected are MC284-0403 (2-way) and MC284-0404 (3-way) types. These solenoids are fully qualified for 100 shuttle missions and are fabricated by Wright Components. The valves will be utilized in the pneumatic pressurization system and to operate the various 2-inch and 3-inch control valves. A limited pressure drop analyses is proposed if filters are incorporated on the inlet side of the valves to prevent contamination.

For physical description and performance characteristics see the following pages.

COMPONENT: VALVE, SOLENOID, TWO WAY

(MC284-0403)

DESCRIPTION:

THE VALVE IS A TWO-WAY SOLENOID VALVE, NORMALLY CLOSED, REQUIRING 28 VOLT DC POWER SUPPLY.

FUNCTION:

THE VALVES ARE USED IN THE MAIN PROPULSION HELIUM SYSTEM. TYPE I VALVE ISOLATES THE HELIUM STORAGE FROM THE SYSTEM. TYPE II VALVE CONTROLS HYDROGEN VENTING PRESSURE FOR PNEUMATICALLY ACTUATED VALVES AND THE HELIUMS SUPPLY BLOWDOWN. TYPE III VALVE CONTROLS FUEL AND OXIDIZER FEEDLINE REPRESSURIZATION. THE TYPE V VALVE PROVIDES HELIUM GAS TO THE SSME.

DESIGN PARAMETERS:

VALVE RESPONSE: 100 MILLISECONDS MAXIMUM

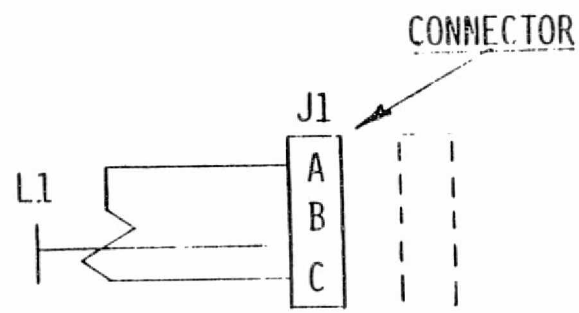
LIFE CYCLE: THE VALVE HAS A USEFUL LIFE OF 10,000 CYCLES OR A 100 ORBITAL MISSION EQUIVALENT

	TYPE 1
PROOF PRESSURES:	9000
BURST PRESSURE:	18000

OPERATING TEMPERATURE: -100°F TO +250°F (MEDIA)
-160°F TO +175°F (ENVIRONMENT)

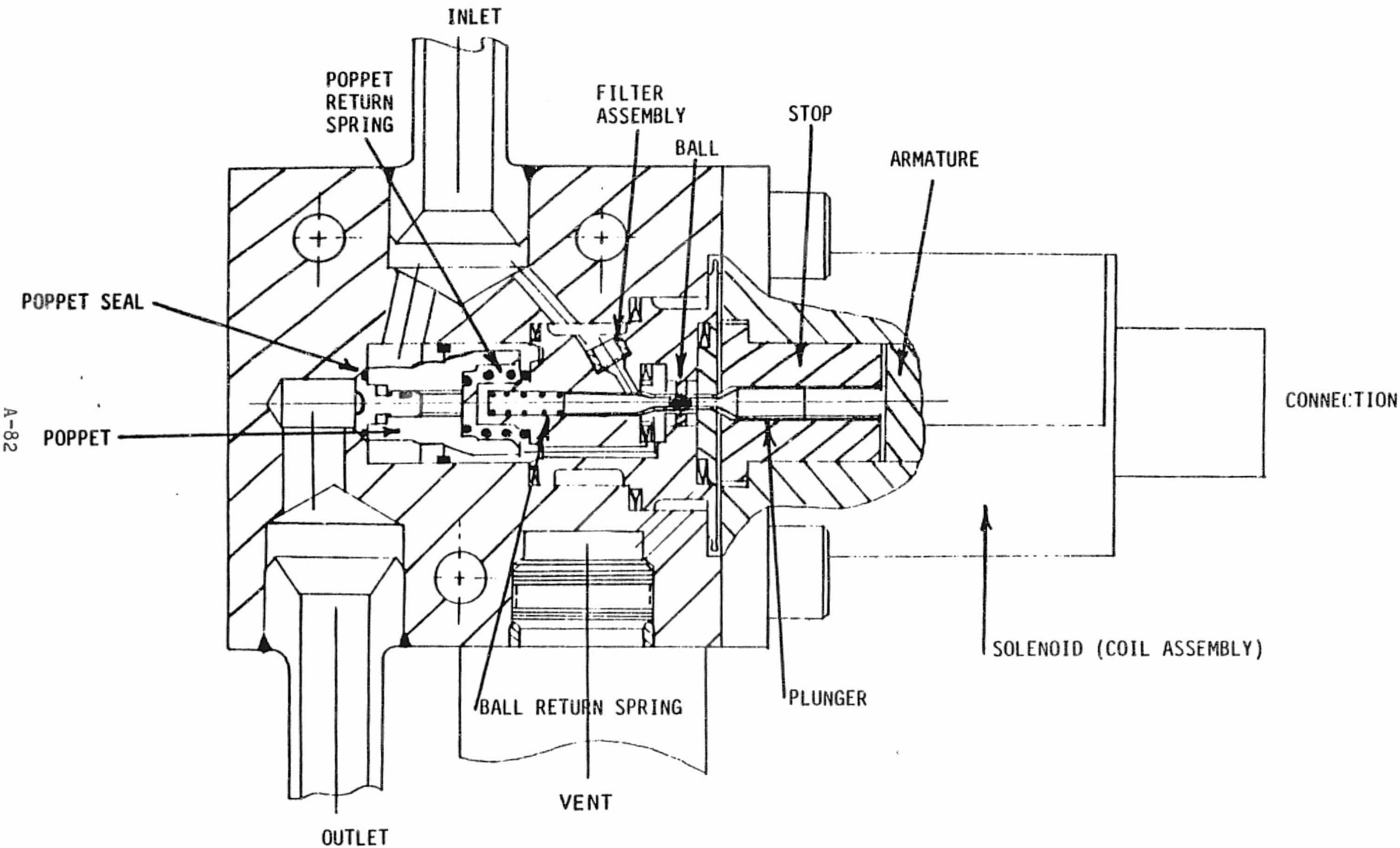
WEIGHT: TYPE 1 2.0 LBS

LINE SIZE: TYPE I 3/8 INCH INLET/OUTLET



COMPONENT: VALVE, SOLENOID, TWO WAY, TYPE I

(MC284-0403)



Space Transportation System
Development & Production Division
Space Systems Group
Rockwell International

COMPONENT: VALVE, SOLENOID THREE WAY

(MC284-0404)

DESCRIPTION:

THE VALVE IS A THREE-WAY, NORMALLY CLOSED, TWO POSITION, CONTINUOUS DUTY TYPE, WITH HERMETICALLY SEALED ELECTRICAL COMPONENTS. THE VALVE IS SPRING-LOADED SO THAT IN THE DE-ENERGIZED POSITION THE PRESSURE PORT IS CLOSED AND THE ACTUATION PORT IS OPEN TO THE VENT PORT. WITH THE VALVE ENERGIZED, THE VENT PORT IS CLOSED AND THE PRESSURE PORT IS OPEN TO THE ACTUATION PORT. THE VALVE SHALL BE SO CONSTRUCTED THAT ALL INTERNAL LEAKAGE IS DIRECTED TO THE VENT PORT. A VENT PORT PROTECTIVE DEVICE IS REQUIRED AS A REMOVABLE ASSEMBLY TO PROVIDE VENT PORT, MOISTURE, AND CONTAMINATION PROTECTION.

FUNCTION:

THE VALVE IS A THREE-WAY, NORMALLY CLOSED, 28 VOLT DC SOLENOID VALVE. THE VALVE IS USED IN THE ORBITER MAIN PROPULSION HELIUM SYSTEM TO SUPPLY PRESSURE TO THE PNEUMATICALLY ACTUATED VALVES OF THE MAIN PROPULSION SUBSYSTEM.

DESIGN PARAMETERS:

LIFE CYCLE: THE VALVE HAS A MINIMUM USEFUL LIFE OF 1000 ENERGIZED HOURS OR A 100 ORBITAL MISSION EQUIVALENT.

PROOF PRESSURE: 1540 PSIG OUTLET - 1700 PSIG INLET

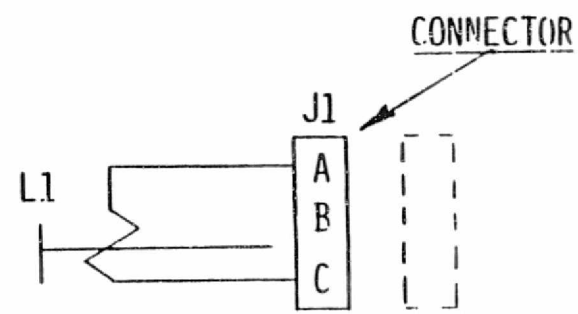
BURST PRESSURE: 3400 PSIG

OPERATING TEMPERATURE: -100°F TO +250°F (MEDIA)
-160°F TO +175°F (ENVIRONMENT)

WEIGHT: TYPE II 1.86 LBS.

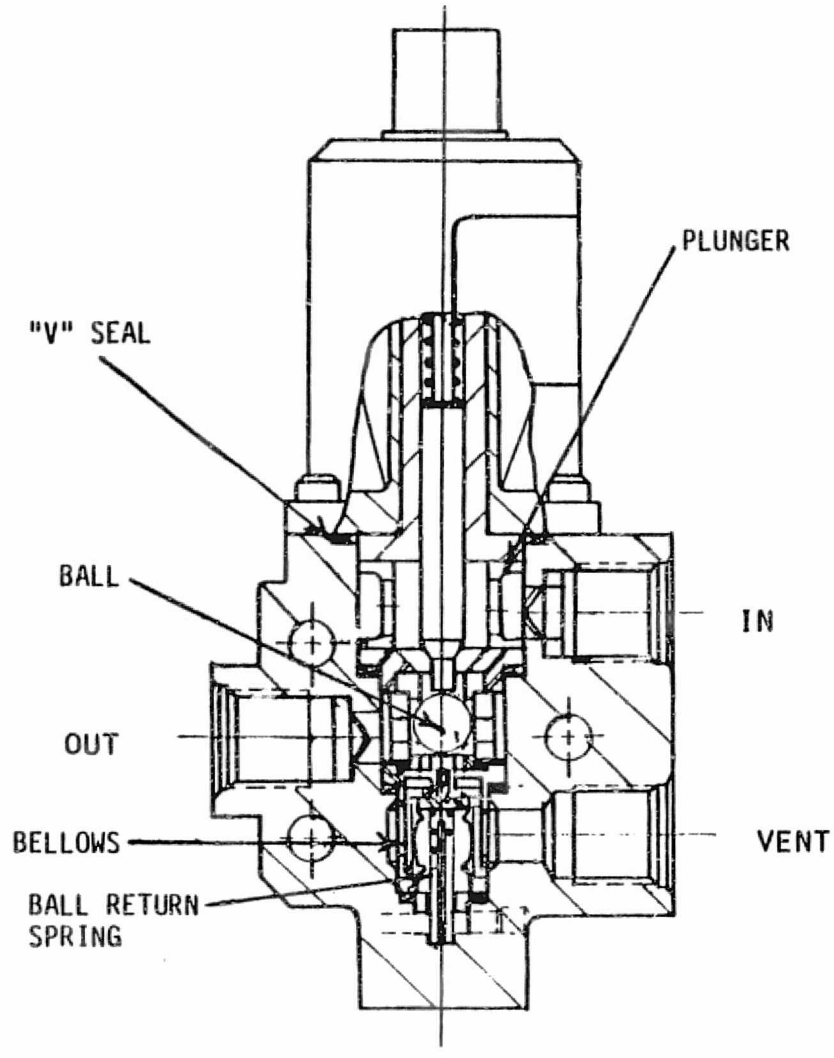
LINE SIZE: TYPE II 1/4 INCH INLET/OUTLET

OPEN RESPONSE: 100 MILLISECONDS MAXIMUM WHEN ENERGIZED
CLOSE RESPONSE: 100 MILLISECONDS MAXIMUM WHEN DE-ENERGIZED



COMPONENT: VALVE, SOLENOID, THREE WAY, TYPE II

(MC284-0404)



Regulator - 750 psig - The 750 psig Helium Pressure Regulator MC284-0533 is manufactured by Consolidated Controls Corporation. This regulator is currently used in the Shuttle Pneumatic System and is fully qualified for 100 Shuttle missions. The regulator will control the 4,500-psig pneumatic supply pressure to the required 750-psig system pressure.

For physical description and performance characteristics refer to the following pages.

COMPONENT: REGULATOR, HELIUM PRESSURE, 750 PSIG

(MC284-0533)

DESCRIPTION:

THE REGULATOR IS A PILOT-OPERATED, FAST-RESPONSE, HIGH FLOW HELIUM REGULATOR WITH AN INTERNAL ABSOLUTE PRESSURE REFERENCE. THE OUTLET PRESSURE IS REGULATED BETWEEN 715 PSIA AND 770 PSIA WITH AN INLET PRESSURE BETWEEN 900 PSIG AND 4500 PSIG AT FLOW RATES UP TO 1.0 POUNDS PER SECOND (PPS) MAXIMUM. THE REGULATOR IS AN ALL-WELDED UNIT WITHOUT ANY EXTERNAL ADJUSTMENT OR SENSING LINES.

FUNCTION:

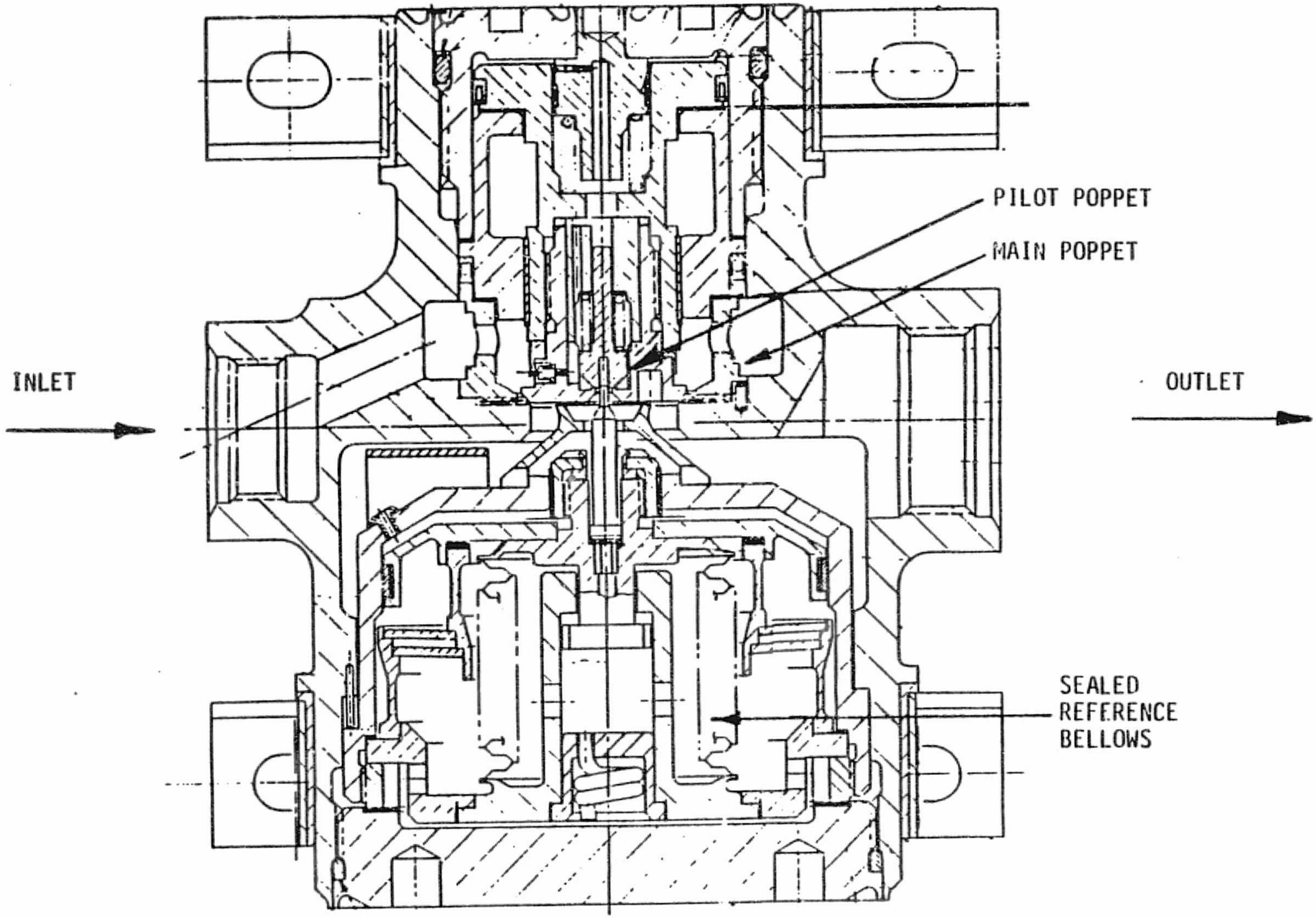
THE REGULATOR IS USED IN THE ORBITER PNEUMATIC SYSTEM TO REGULATE THE HIGH PRESSURE HELIUM SUPPLY TO A LOWER SYSTEM OPERATING PRESSURE FOR USE IN COMPONENT OPERATION, PROPELLANT FEED AND FILL SYSTEMS REPRESSURIZATION, PRESSURIZATION TO EXPEL PROPELLANTS FROM THE FEED SYSTEMS AND TO PROVIDE ENGINE PURGE GAS. THE REGULATOR IS USED PRIOR TO, DURING AND AFTER COMPLETION OF FLIGHT.

DESIGN PARAMETERS:

PROOF PRESSURE:	INLET - 9050	OUTLET - 1550
BURST PRESSURE:	INLET - 18,000	OUTLET - 3400
LINE SIZE:	INLET - 3/8 IN. BOSS	OUTLET - 1/2 IN. BOSS
WEIGHT:	3.3 LBS.	
OPERATING BAND:	700 TO 755 PSIG	
OPERATING TEMPERATURE:	900 TO 2000 PSIG, -160F TO +220F	
	2000 TO 4500 PSIG, -75F TO +220F	

COMPONENT: REGULATOR, HELIUM PRESSURE, 750 PSIG

(MC284-0533)



Helium Supply Tank - The 17.3-cu ft MC282-0082 helium supply tank is manufactured by Brunswick Corporation. This tank is currently used on the Shuttle and is fully qualified for 100 Shuttle missions. Helium supplied from the tank will provide actuation pressure for the operation of pneumatic activated control valves.

For physical description and performance characteristics see the following pages.

COMPONENT: HELIUM SUPPLY TANK

(MC282-0082)

DESCRIPTION:

Pressure vessels supply helium to pneumatically actuated valves and provide gas for engine and line purge. Vessels are spherical in shape and are comprised of a titanium 6AL-4V liner wound with epoxy impregnated Kevlar 49.

FUNCTION:

The He vessels supply actuation gas to valves and supply helium purge to the propulsion system.

DESIGN PARAMETERS:

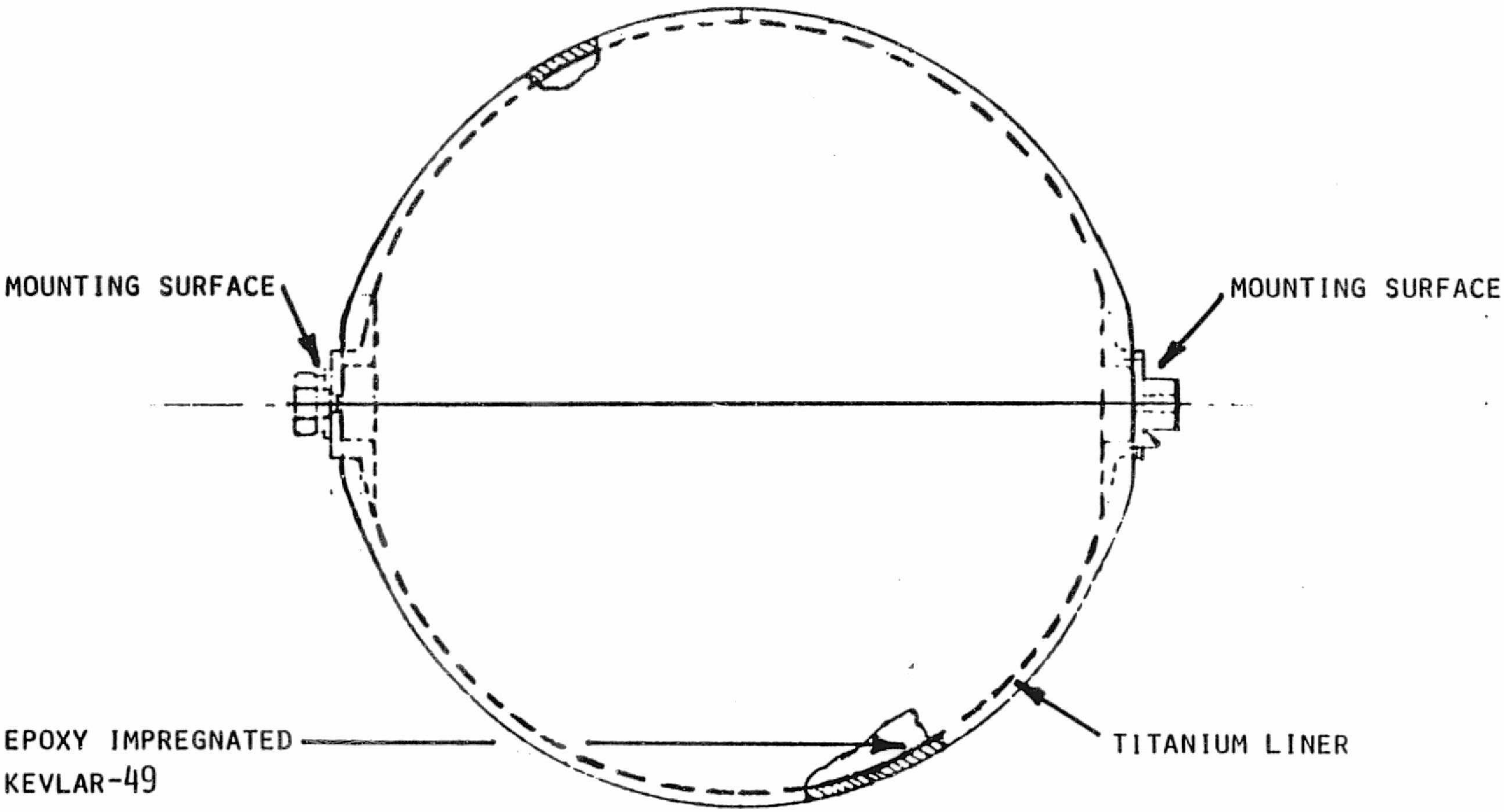
Tank	Max. O.D. at Zero Press.	Max. Press. at 40 to 200°F	Weight
29,930 cu. in.	40.3 in.	4,500 psig	289 lb

OPERATING LIFE:

250 pressure cycles from ambient to maximum operating pressure.

COMPONENT: HELIUM SUPPLY TANK

(MC282-0082)



Relief Valve, Helium Pressure, 850 PSIG - The helium relief valve MC284-0398-0005 is manufactured by Consolidated Controls Corporation. This valve is currently used in the Shuttle Pneumatic System and is fully qualified for 100 Shuttle missions. In the event of an upstream regulator failure, the valve would relieve excessive pressure and then reseat when reseat pressure is re-established. It is not a burst disc device.

For physical description and performance characteristics see the following pages.

COMPONENT: RELIEF VALVE, HELIUM PRESSURE, 850 PSIG

(MC284-0398)

DESCRIPTION:

THE 850 PSIG HELIUM RELIEF VALVE IS A FAST RESPONSE, PILOT OPERATED GAGE REFERENCE VALVE. THE VALVE HAS BOTH INTERNAL AND REMOTE SENSING TO AID STABILITY. THE PILOT IS PROTECTED BY A LOW PRESSURE OPERATED FLAPPER AT THE PILOT VENT OUTLET. THE MAIN POPPET FLOW EXITS THE VALVE BODY SEPARATELY FROM THE PILOT FLOW AND IS PROTECTED EXTERNALLY BY A PERFORATED COVER.

FUNCTION:

THE RELIEF VALVE IS USED IN THE ORBITER HELIUM PNEUMATIC AND PURGE SYSTEMS TO PROVIDE A PRESSURE RELIEF IN THE EVENT A SYSTEM MALFUNCTION RESULTS IN A PRESSURE INCREASE WHICH EXCEEDS THE PRESET RELIEVING PRESSURE. THE RELIEF VALVE SHALL CLOSE AND MEET THE LEAKAGE REQUIREMENTS WHEN THE PRESSURE HAS BEEN REDUCED TO A PRESET RESEAT PRESSURE. THE RELIEF VALVE IS FUNCTIONAL CONTINUOUSLY.

DESIGN PARAMETERS:

LIFE CYCLE: THE VALVE HAS A USEFUL LIFE OF 2000 CYCLES OR A 100-ORBITAL MISSION EQUIVALENT.

PROOF PRESSURE: 1750 PSIG

BURST PRESSURE: 3400 PSIG

WEIGHT: 2.27 LBS.

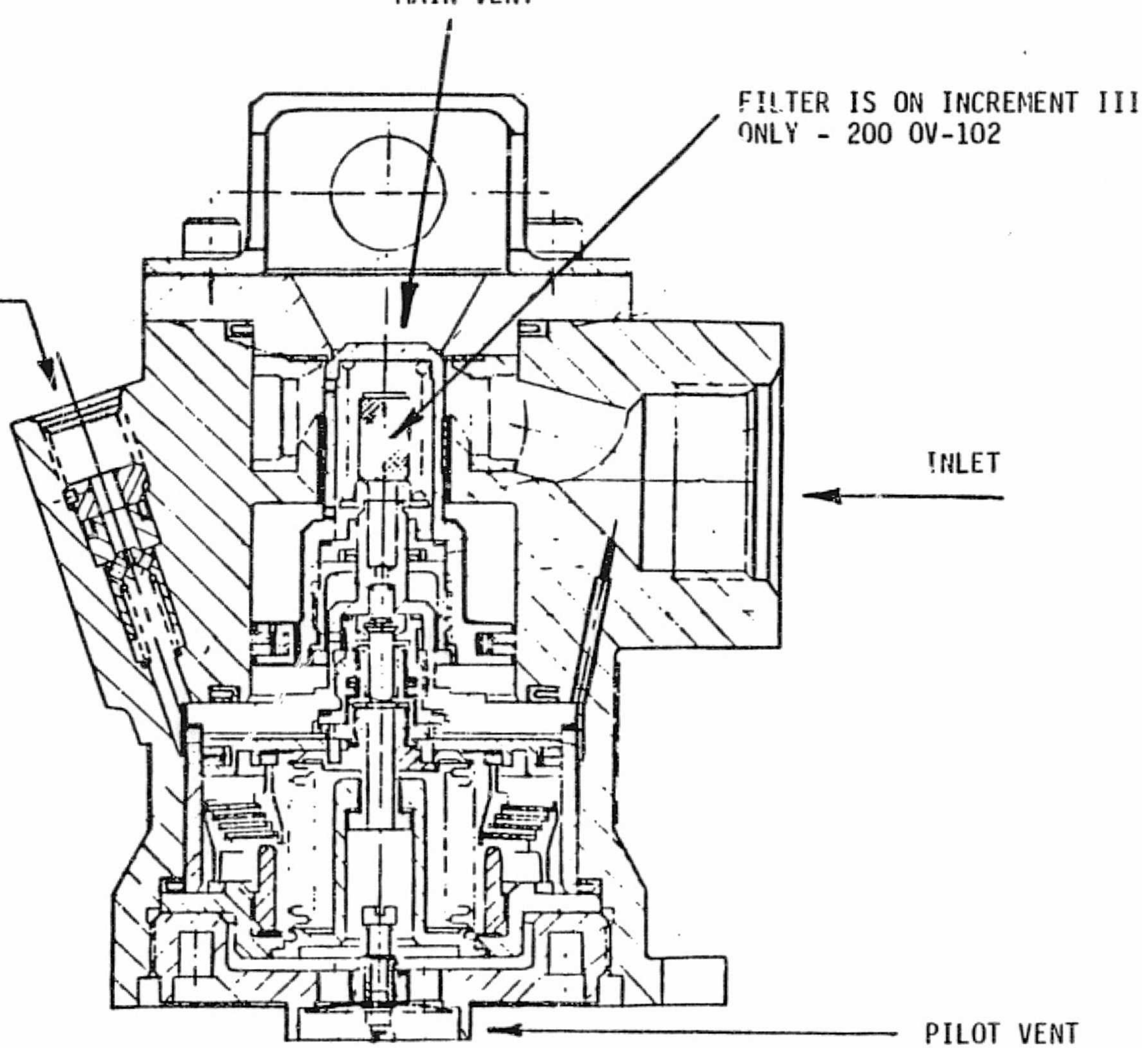
LINE SIZE: INLET, 0.75 INCHES: SENSE, 0.25 INCHES

OPERATING BAND: 785 TO 850 PSIG

OPERATING TEMPERATURE: -160F TO +220F (MEDIA)
-100F TO +220F (ENVIRONMENT)

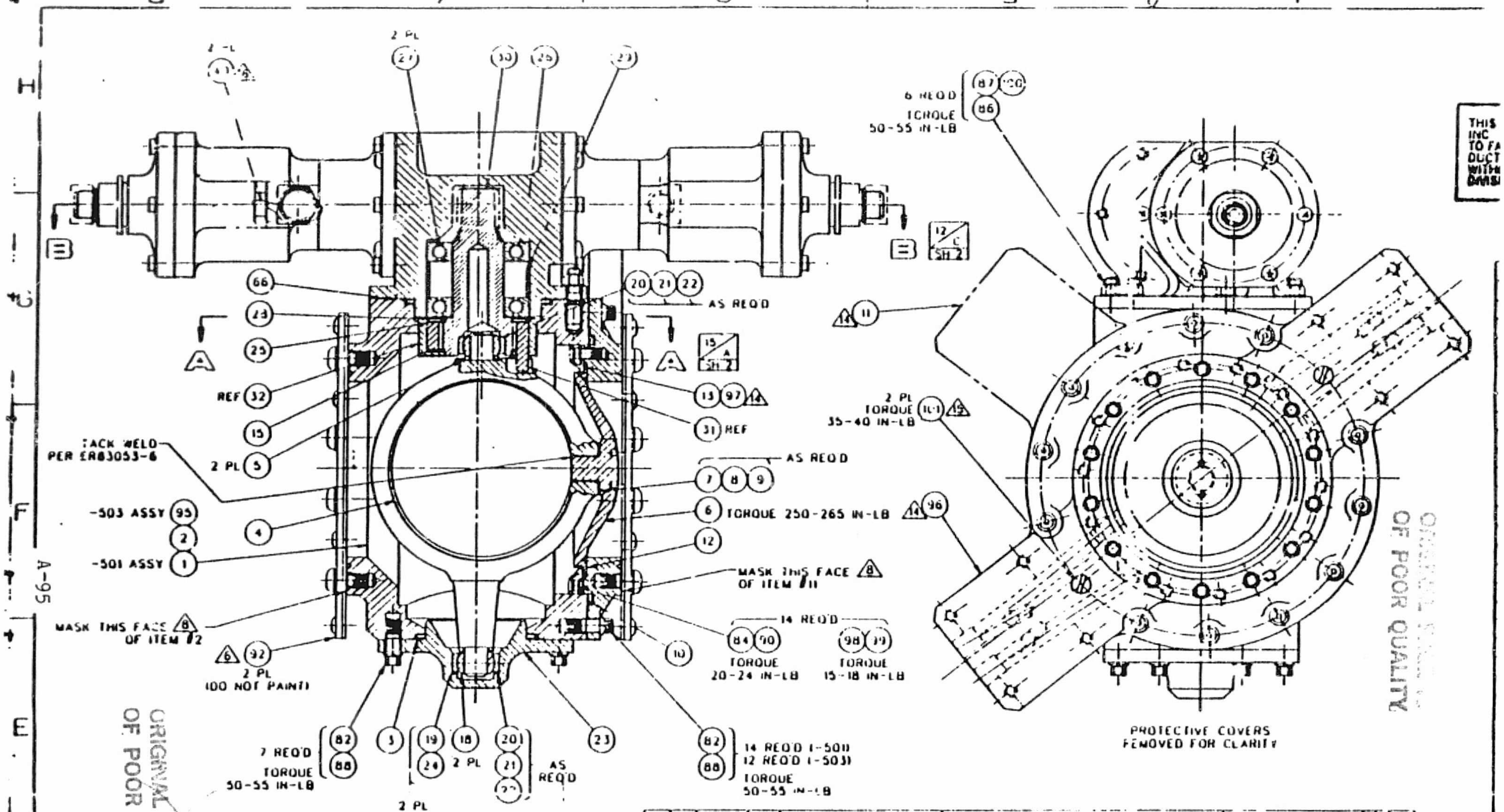
COMPONENT: RELIEF VALVE, HELIUM PRESSURE, 850 PSIG

(MC284-0398)



3.5-Inch Dump Valve - The proposed dump valve will be a derivative of the Centaur 3.5-inch shutoff valve currently under development by Fairchild. Additional safety features and redundancies that will be imposed on this valve, such as dual actuators, open and close application of redundant operating pressure, and the removal of opposing pressure before the valve is moved out of positive detents, will ensure complete safety of operation. Both inlet and outlet flanges of the valve body must be redesigned since the sealing surfaces of the Centaur valve are not compatible with the ME261-0045 cryogenic static face seals used on the Shuttle. A full qualification test for this critical component is required.

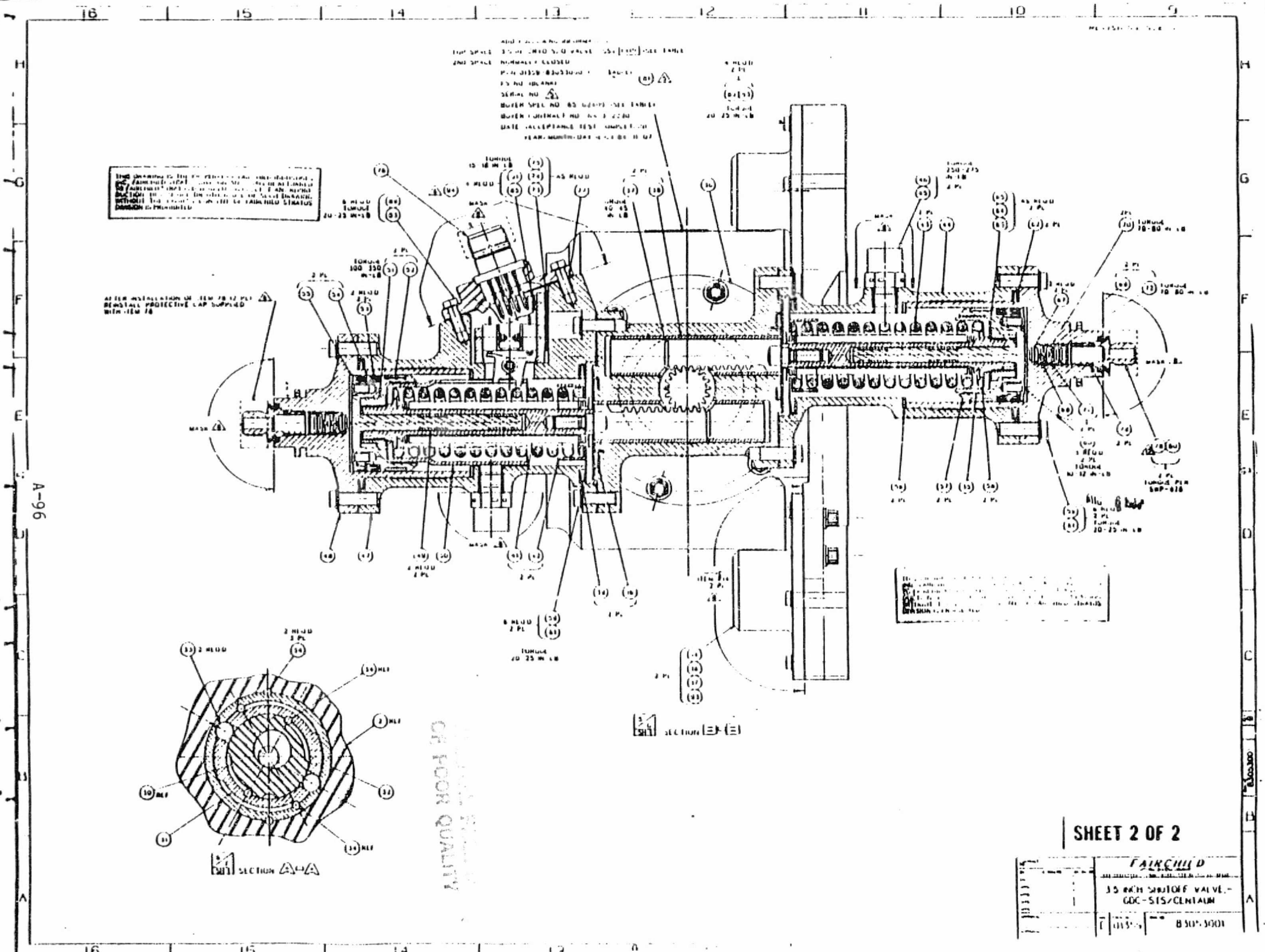
For physical description and performance characteristics see the following pages.



A-95

2	–			SCREW, FLAT HD 100°	NAS1102E4-12			[illegible]	101
6	–			NUT, SELF-LOCKING	NAS1291C4			[illegible]	100
14	–			WASHER	NAS620C6L			[illegible]	99
14	–			SCREW, FLAT FIL HD	NAS1101E06-6			[illegible]	98
1	–			SUPPORT, SEAL	83053064-3			[illegible]	97
1	–			FLANGE ASSY, SEAT	83053063-501			[illegible]	96
✓	–			ASSEMBLY (NC)	83053001-503			[illegible]	95
1	1			CAP, DUST	MS90376-12Y			[illegible]	94
2	–			VALVE ASSY, RELIEF	83059119-501			[illegible]	93
2	2			COVER ASSY, SHIPPING	83053022-501			[illegible]	92
4	4			WASHER	NAS620C6L			[illegible]	91
–	14			WASHER	NAS620C8L			[illegible]	90
14	14			WASHER	NAS620C8				89
19	21			WASHER	NAS1587-4C				88
6	6			WASHER	AN960C428			[illegible]	87
	6	56878		NUT, SELF-LOCKING	61170-428		15	[illegible]	86
[illegible]	[illegible]			[illegible]	[illegible]			[illegible]	85

3.5 INCH ... VALVE

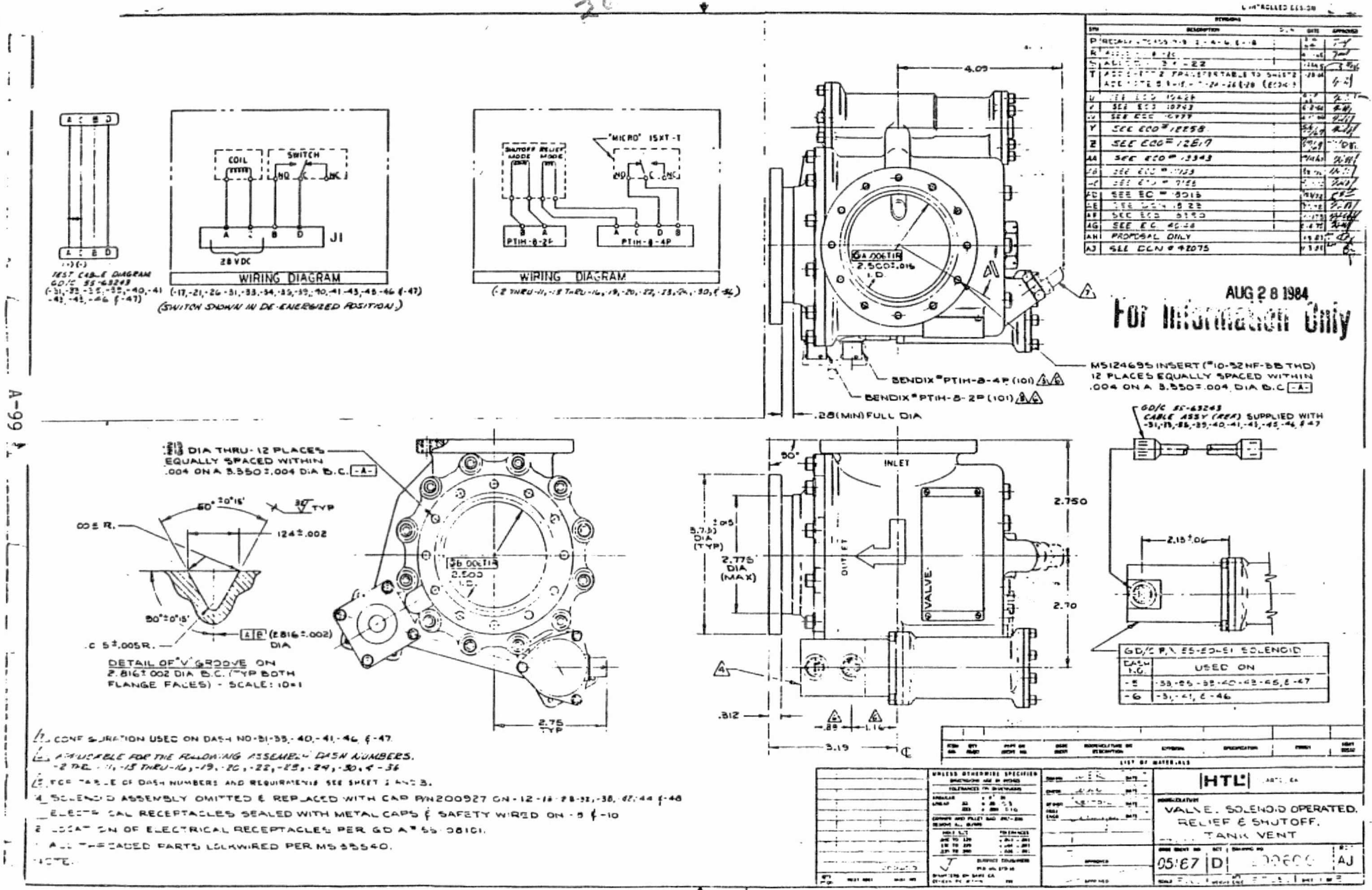

SECTION A-A
SECTION E-E
SHEET 2 OF 2
FAIRCHILD
3.5 INCH SHUTOFF VALVE - GDC-STS/CENTAUR
83053001
OF POOR QUALITY

Disconnect - The user inflight disconnect will be a new component especially designed to provide maximum safety and utility. The disconnects will have integral dual in-series poppets. These will insure positive sealing integrity in case of upstream user transfer valve failure. Either one of the two poppets will satisfy the systems leakage requirements. Positive latching capability, maximum angulation, and ease of engagement will be mandatory design features. The latest state of the art technology will be employed to accomplish the design. The disconnect, however, can utilize the same design concepts, materials and performance characteristics as currently used on the Shuttle and Centaur Programs.

Also, there are a number of seal designs currently used and therefore existing sealing capabilities can be easily incorporated into the design.

Relief Valve - This valve will be used to prevent overpressurization of the cryogenic tanks. The existing Shuttle vent/relief valves are too large for use in this system. However a valve now being used on the Centaur program can easily be adapted for use on this system. The Centaur valve, P/N 200600, is manufactured by HTL Industries, of Duarte, Ca. It will require some modifications and a limited delta qualification test in order to achieve a fully certified status. Both inlet and outlet flanges must be modified to accommodate the ME261-0045 cryogenic static face seals used on the Shuttle. The valves physical characteristics are as follows:

ORIGINAL PAGE IS OF POOR QUALITY



2-Inch Shutoff Valve - The scavenging system requires normally closed two inch valves for the fill valves, user transfer valves, vent valves and the TVS shutoff valves. To accommodate this requirement, the present Shuttle MC284-0345, Type II normally closed ball valves will be modified for these functions. This valve is fully qualified for 100 Shuttle missions and is manufactured by Consolidated Controls Corporation. The valve is pneumatically operated by 750 psi helium pressure, has position indicators and internal relief capability. To provide a normally closed valve the ball will be rotated 90 degrees. The new valve will be qualified by similarity to the present Shuttle valve. For a physical description and performance characteristics of the valve see the following pages.

An alternate valve which will be considered is a 2 inch Shuttle valve now being tested by Fairchild for the Centaur program. This valve has a gate which first lifts off the seat and then rotates to the open position. A trade study will be performed to determine the best unit for the scavenging systems.

COMPONENT: VALVE, SHUTOFF, LO_2 POGO

(MC284-0395)

TYPE 2

REFERENCE MAIN PROPULSION SYSTEM SCHEMATIC FOR LOCATION IN SYSTEM (PV20 AND PV21)

DESCRIPTION AND FUNCTION:

THE POGO VALVE IS A NORMALLY OPEN, SINGLE ACTING, PNEUMATICALLY ACTUATED, BALL VALVE. THE VALVE IS USED IN THE LIQUID OXYGEN POGO SUPPRESSION SYSTEM. TWO VALVES ARE USED IN THE POGO SYSTEM.

THE VALVE CONSISTS OF AN ACTUATOR ASSEMBLY, A RACK AND PINION BALL DRIVE, AND A OPEN AND CLOSED POSITION INDICATOR ASSEMBLY.

THE VALVE IS DESIGNED TO RELIEVE PAST THE MAIN BALL SEAL TO PREVENT THE BUILDUP OF PRESSURE BETWEEN THE POGO VALVE AND THE LO_2 BLEED VALVE, WHEN THE LO_2 BLEED VALVE AND THE POGO VALVE ARE BOTH CLOSED. THE VALVE RELIEVES INTO THE LO_2 MANIFOLD.

THE VALVE IS DESIGNED TO FAIL SAFE IN THE OPEN POSITION.

OPERATIONAL PARAMETERS AND FEATURES

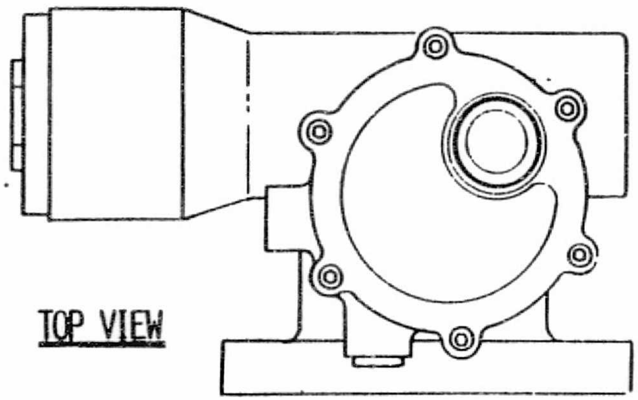
VALVE SIZE:	2 INCHES
OPERATING PRESSURE (BODY):	0 TO 400 PSIG
OPERATING PRESSURE (ACTUATOR):	500 TO 850 PSIG
OPERATING TEMPERATURE:	-297 F TO +170 F
WEIGHT:	7.8 POUNDS
BODY MATERIAL:	ALUMINUM ALLOY CASTING (A-356)
INLET FLANGE SIZE:	3.717 INCHES DIA.
OUTLET FLANGE SIZE:	4.747 INCHES DIA.
OPENING RESPONSE	.100-.750 SEC.
CLOSING RESPONSE	.100-.500 SEC.

ACTUATOR ASSEMBLY SAME FOR ALL MC284-0395 CONFIGURATIONS.

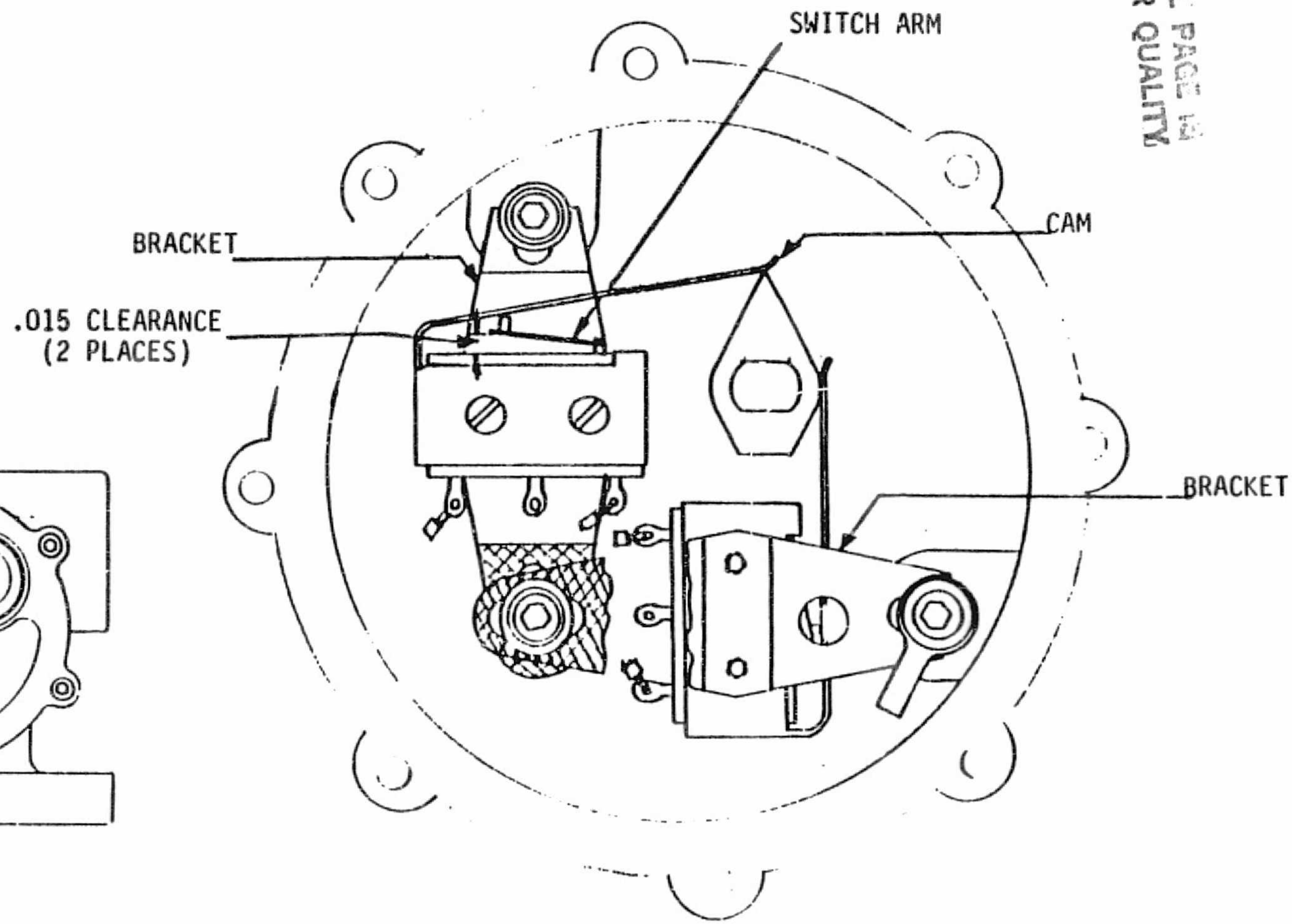
COMPONENT: VALVE, SHUTOFF, LH_2 OUTBOARD RTLS DUMP

(MC284-0395)

TYPE 1 THRU 4

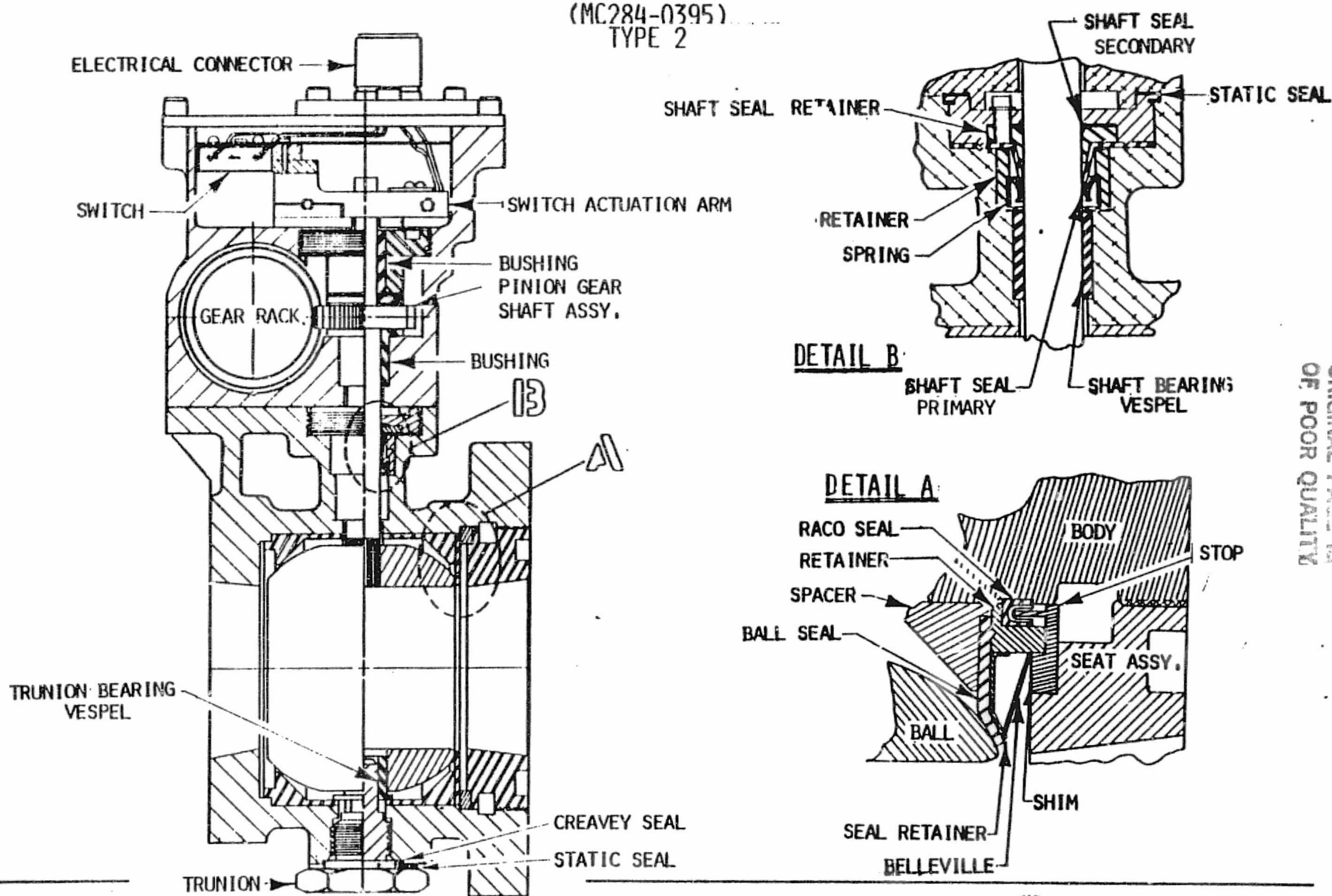


TOP VIEW



COMPONENT: VALVE, SHUTOFF, LO_2 POGO

(MC284-0395)
TYPE 2



Orbiter 8-Inch Fill and Drain Valve - The inboard Orbiter 8-inch fill and drain valve MC284-0397 is currently being qualified for 100 shuttle missions. The scavenging system, as proposed, requires a new fill and drain valve main housing to accommodate a new 2-inch relief shutoff valve. The redesigned fill and drain valve main housing will differ from the present shuttle valve only by a larger outlet boss at a new location for the new relief shutoff valve. The modification to the fill and drain valve could be accomplished by reworking two available spare valves with new housings. Upon incorporating the system into the shuttle the replaced inboard fill and drain valves may be allocated back to spares inventory. Qualification of the redesigned fill and drain valve may be accomplished by analysis and test.

Refer to following pages for physical description and performance characteristics of the shuttle fill and drain valve.

COMPONENT: VALVE, FILL AND DRAIN, PROPELLANT

(MC284-0397)

DESCRIPTION AND OPERATION:

THE COMPONENT, A BI-STABLE VALVE (OPEN AND CLOSED), IS ACTUATED TO THE OPEN OR CLOSED POSITION BY A PNEUMATIC ACTUATOR. ELECTRICAL SWITCHES PROVIDE OPEN AND CLOSED POSITION INDICATION. THE VALVES ARE CLASSIFIED INTO THREE TYPES: IA OXYGEN, IB OXYGEN, II HYDROGEN (TYPE IA AND IB ARE IDENTICAL EXCEPT THE ACTUATOR ON IB IS ROTATED 90° CLOCKWISE).

NOTE: VALVES WHICH HAVE ANTISLAM PROVISIONS ARE CAPABLE OF WITHSTANDING SLAMMING DUE TO IMPROPER APPLICATION/SEQUENCING OF ACTUATION PRESSURE.

FUNCTION:

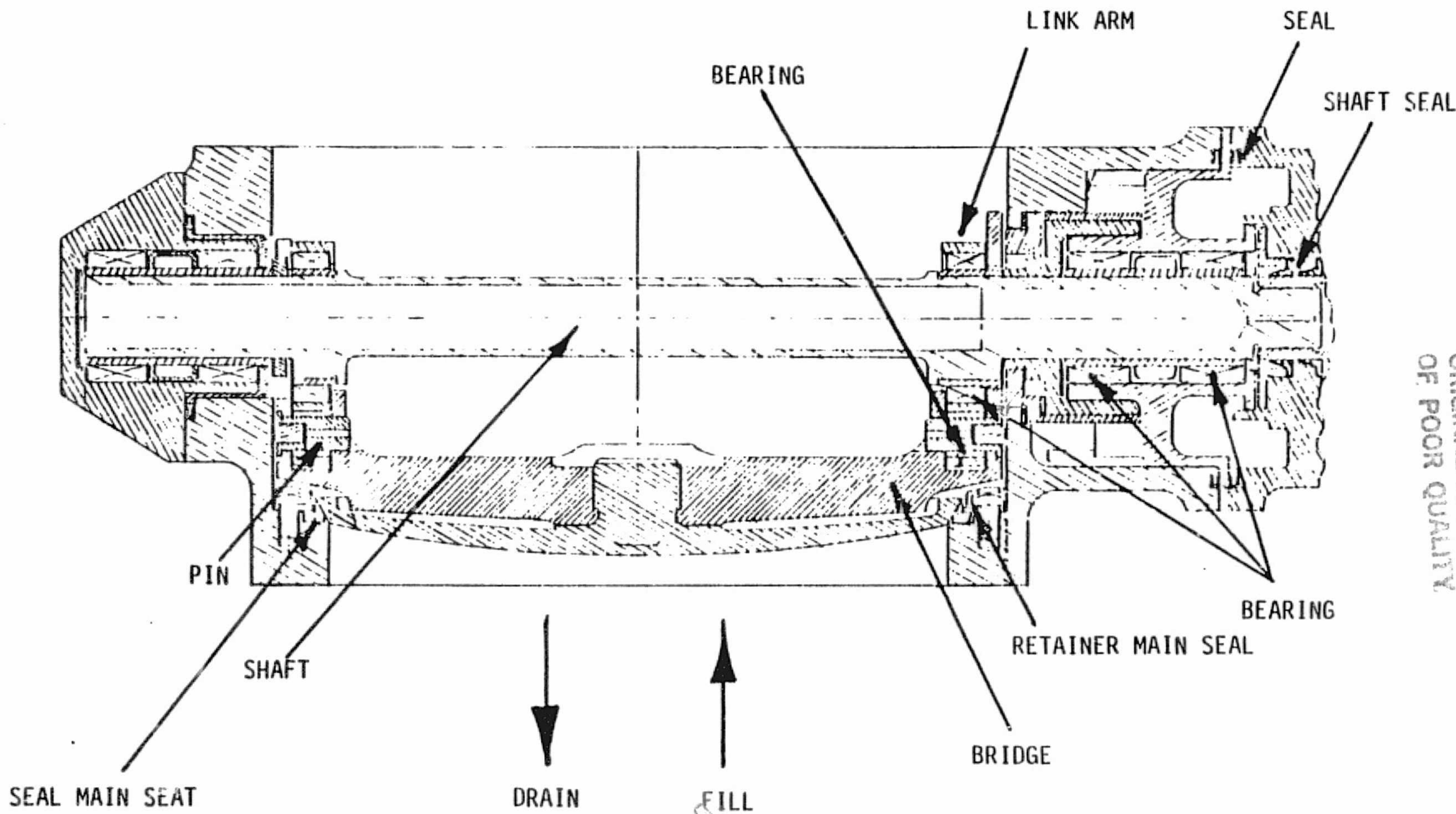
THE VALVE IS USED IN THE FUEL AND OXIDIZER FILL & DRAIN SYSTEMS TO FACILITATE FILL & DRAIN OF THE LIQUID OXYGEN AND HYDROGEN TANKS. THE VALVE REMAINS CLOSED DURING BOOST AND IS CYCLED IN ORBIT. THE VALVE REMAINS IN THE LAST POSITION SELECTED UNTIL ACTUATION PRESSURE IS APPLIED. THE VALVE HAS A RELIEF MECHANISM WHICH BYPASSES THE MAIN VALVE SEAT TO PREVENT EXCESSIVE PRESSURE ON THE UPSTREAM SIDE OF THE VALVE.

DESIGN PARAMETERS AND FEATURES:

PRESSURE OPERATING	0 TO 275 PSIA (OXYGEN) 0 TO 110 PSIG (HYDROGEN)	
TEMPERATURE OPERATING	150F TO -320F (OXYGEN) 150F TO -423F (HYDROGEN)	
TEMPERATURE NON-OPERATING	200F MAXIMUM	
FLOW (OXYGEN)	5,000 GAL/MIN @ 130 ± 5 PSIA,	P 5 PSI (FILL) P 3.3 PSI (DRAIN)
FLOW (HYDROGEN)	12,000 GAL/MIN @ 35 ± 5 PSIA,	P 1.8 PSI (FILL) P 1.2 PSI (DRAIN)
FLOW (RELIEF)	0.3 LBS/SEC LO2 AND GO2 @ 50 PSID MAXIMUM 0.1 LBS/SEC LH2 AND GH2 @ 50 PSID MAXIMUM	
CRACK & RESEAT (RELIEF)	15-50 PSID	
ACTUATION PRESSURE	740 ± 40 PSIG NOM., 850 PSIG MAX., 400 PSIG MIN.	
ACTUATION TEMPERATURE	-160 TO +130F (FLUID), +150 TO -423F (BODY) 200F MAX. NON-OPERATING	
RESPONSE TIME OPENING	7 ± 2 SECONDS	
RESPONSE TIME CLOSING	7 ± 2 SECONDS	
WEIGHT	TYPE I, 47.0 LBS.; TYPE II, 46.5 LBS. (WITHOUT ANTISLAM PROVISIONS) 49.0 42.5 LBS. (WITH ANTISLAM PROVISIONS)	

COMPONENT: VALVE, FILL AND DRAIN, PROPELLANT

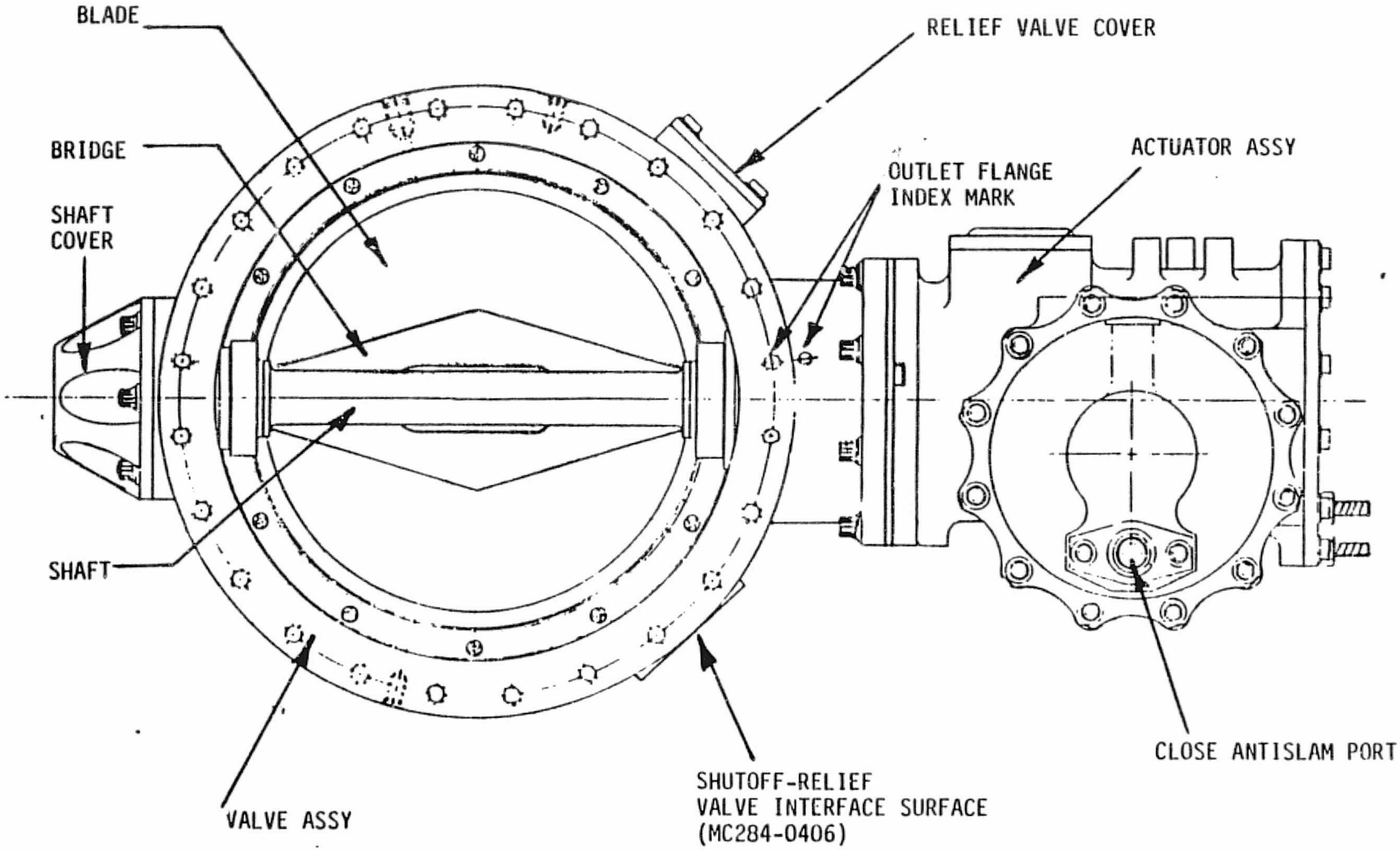
(MC284-0397)



COMPONENT: VALVE, FILL AND DRAIN, PROPELLANT

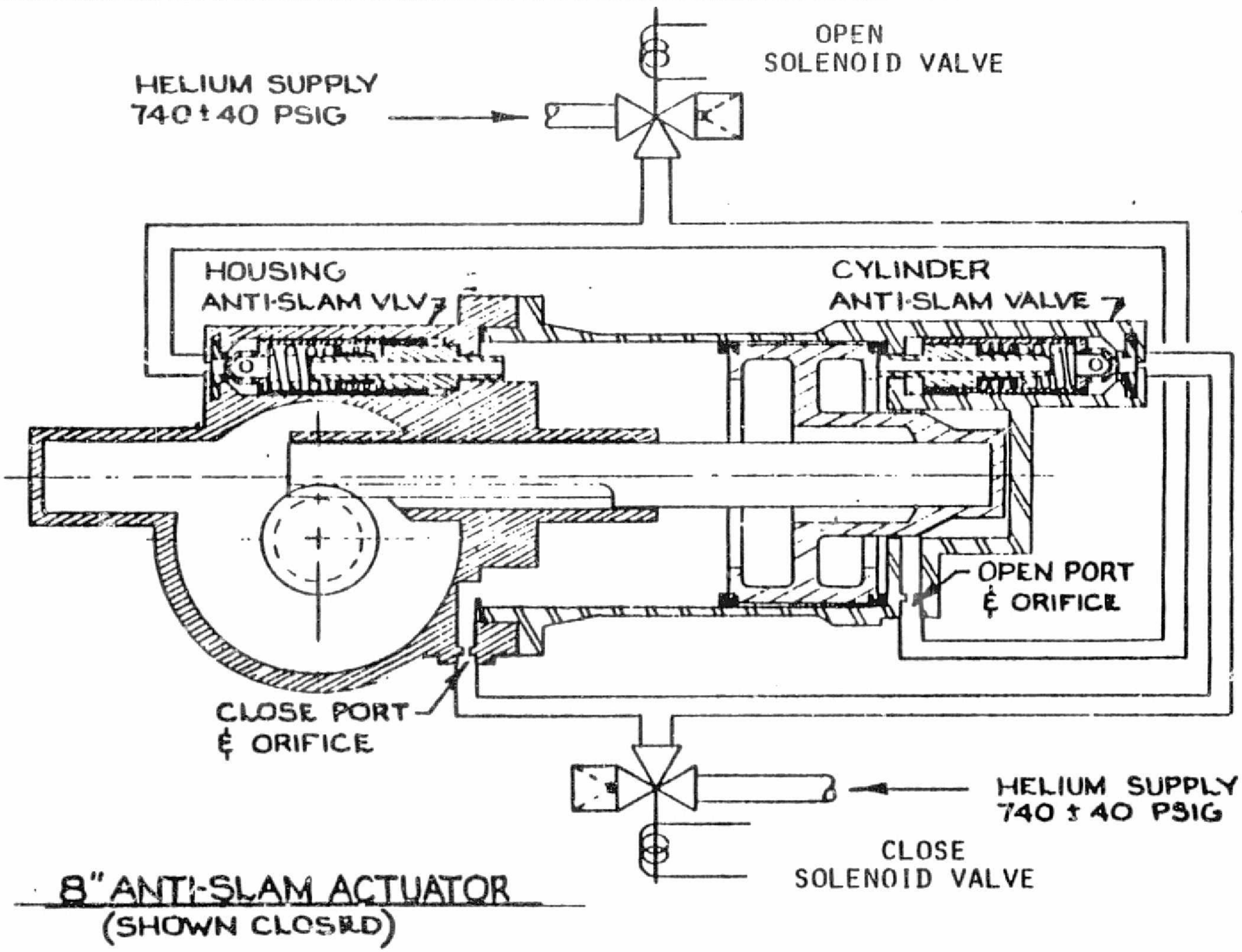
(MC284-0397)

(WITH ANTISLAM PROVISIONS)



COMPONENT: VALVE, FILL AND DRAIN, PROPELLANT

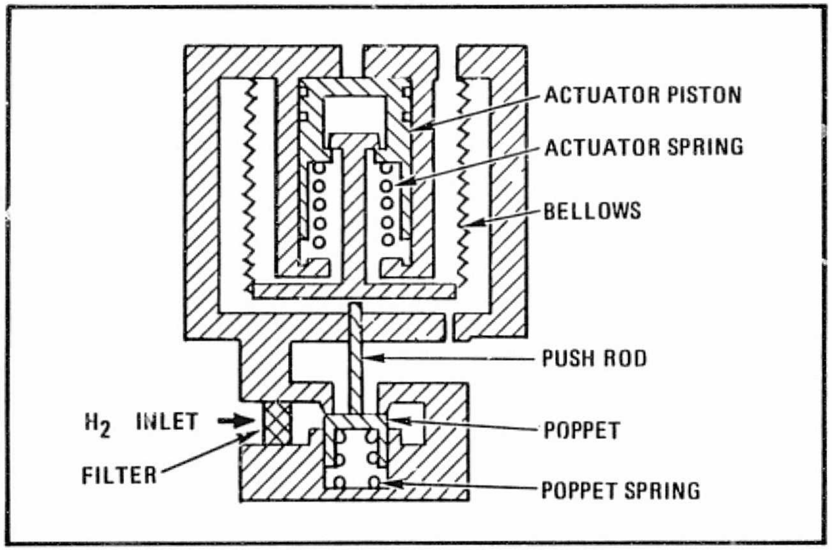
(MC284-0397)

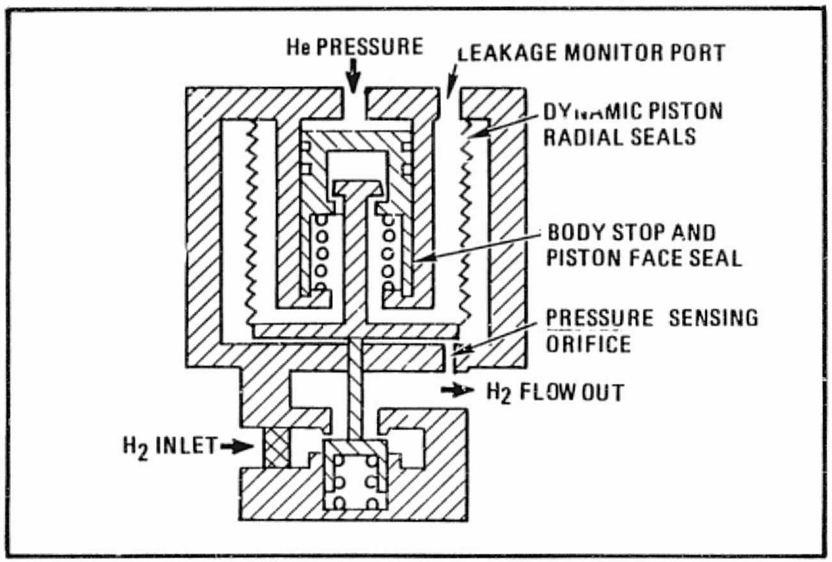


SCHEMATIC OF ANTI-SLAM ACTUATOR

Orbiter 2-Inch Shutoff Relief Valve - The scavenging system requires the present Orbiter normally open one-inch shutoff relief valve be changed to a two-inch diameter (ID) to allow a larger flowrate into the scavenging tank. To accommodate this requirement the present one-inch valve will be replaced by the existing normally open two-inch ball valve, P/N MC284-0395-0052, which will also be utilized in the scavenging system with a normally closed configuration. See the description for the two-inch shutoff valve for details.

LH2 Control Vent Module (CVM), Heat Exchanger, and Mixer Motor - The LH2 control vent module (CVM) consists of two parallel pressure regulators with integral pneumatically activated shutoff valves. The CVM is normally in the closed (shutoff) position. To activate the regulator, pneumatic pressure is applied to the pressure ports, and the CMV regulates LH2 at 20-25 psig inlet pressure to 6-7 psig outlet pressure. The CVM is manufactured by Consolidated Controls Corporation for the Centaur project and is partial qualified. The heat exchanger and mixer motor pump are manufactured by Sunstrand for the Centaur program and are partial qualified. Full qualification is expected within the next few months. See the following pages for a physical layout of the CVM module, mixer pump and heat exchanger.





- **NORMALLY CLOSED POSITION (NO He PRESSURE)**

INTEGRATED CVM DESIGN CONCEPT

- **REGULATING (He PRESSURE APPLIED)**

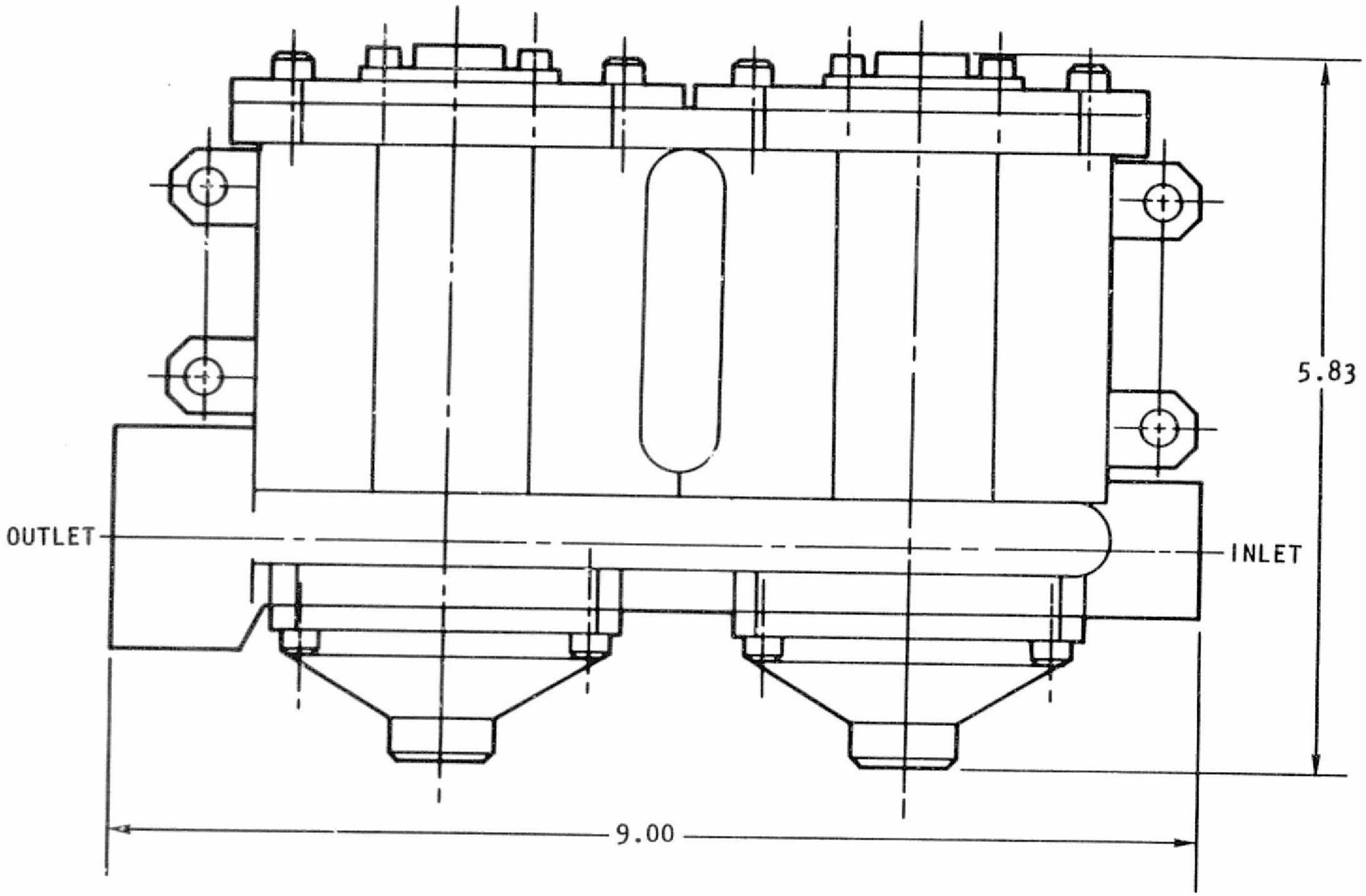
CONDEC Consolidated Controls

CVM REQUIREMENTS

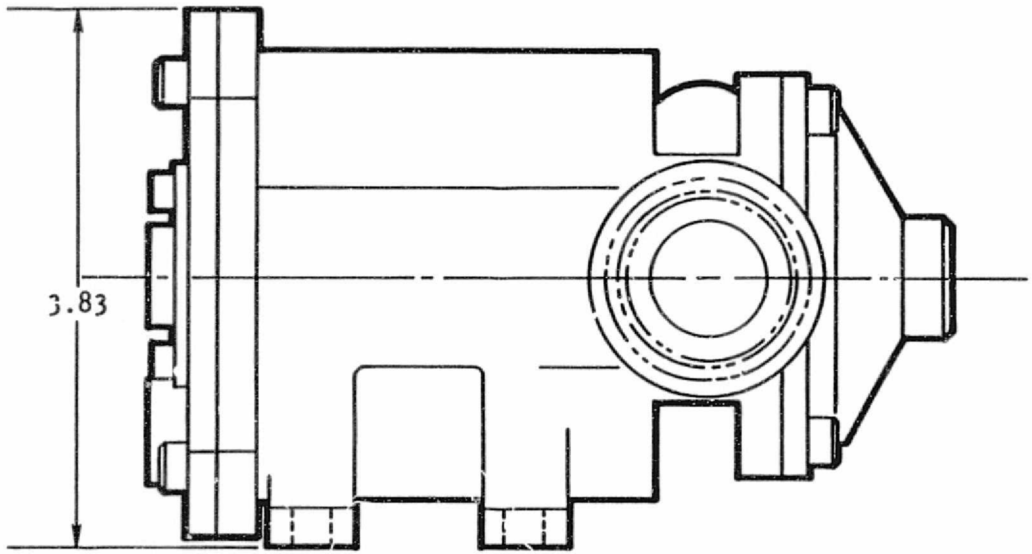
FLUID	FLOW (LBM) (HR)	INLET PRESSURE (PSIA)	OUTLET PRESSURE
LIQUID	40 ± 2	20 - 25	7.1
VAPOR	40^{+2}_{-12}	20 - 25	6.3

SUNDSTRAND

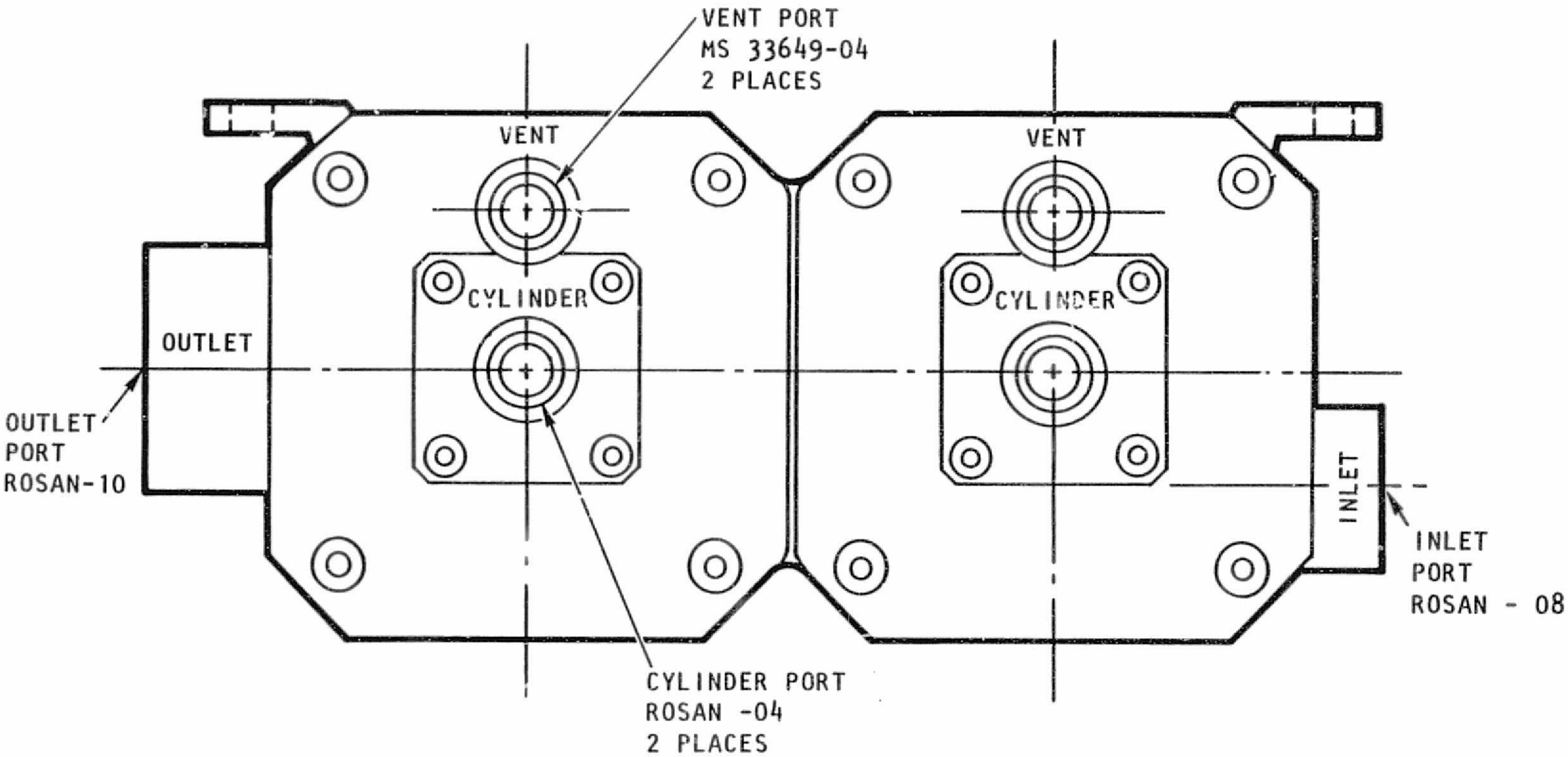
CONTROLLING VALVE MODULE



CONTROLLING VALVE MODULE

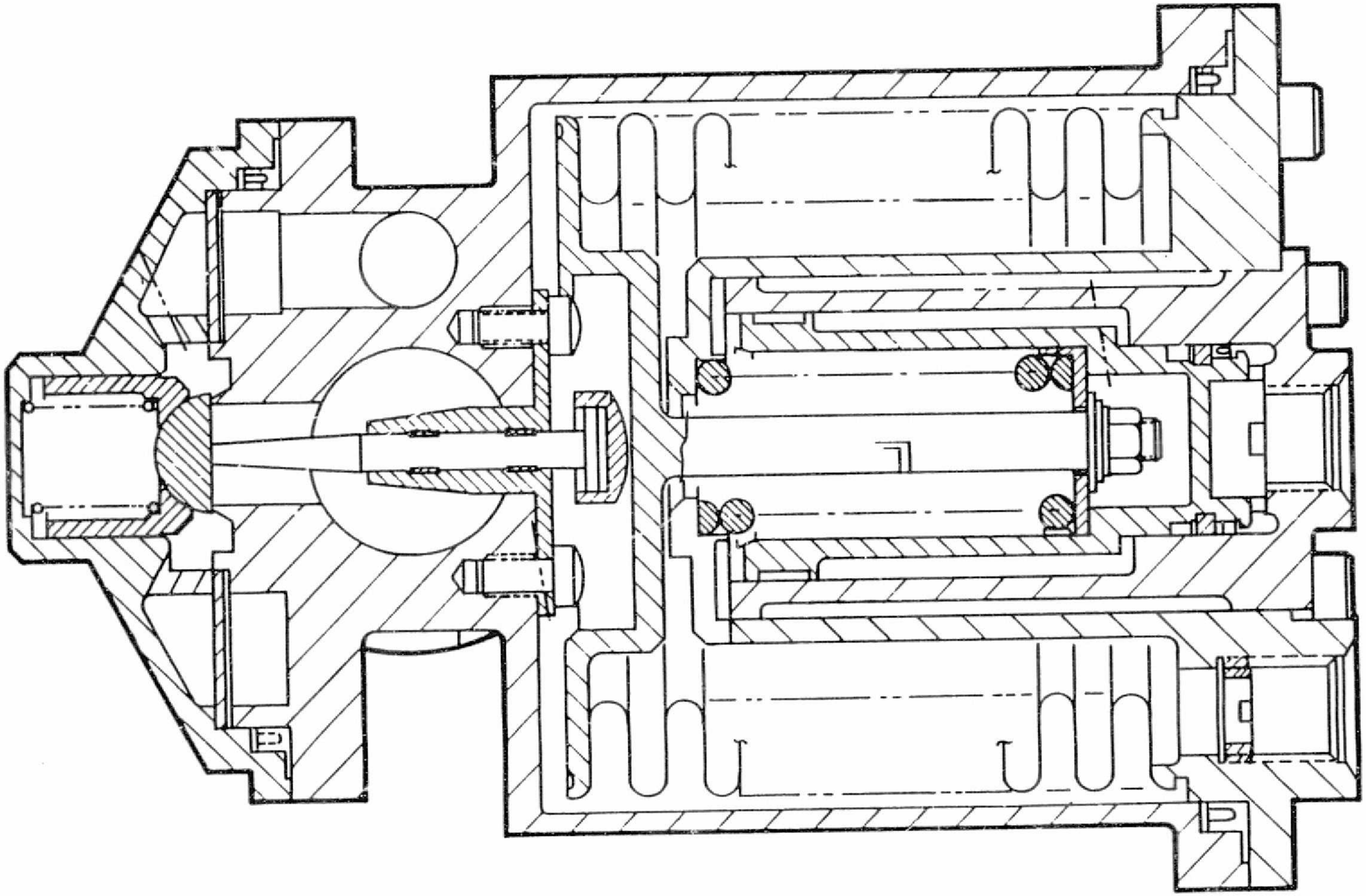


CONTROLLING VALVE MODULE

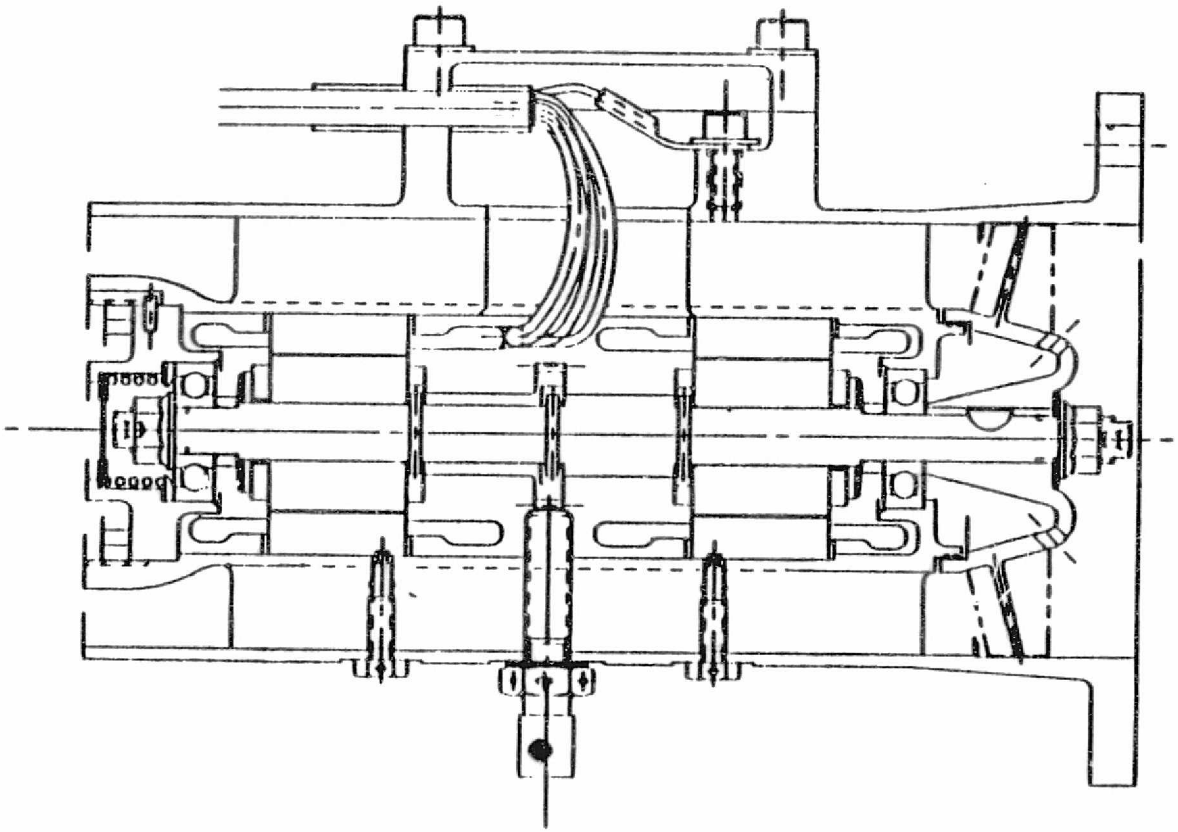


SUNDSTRAND

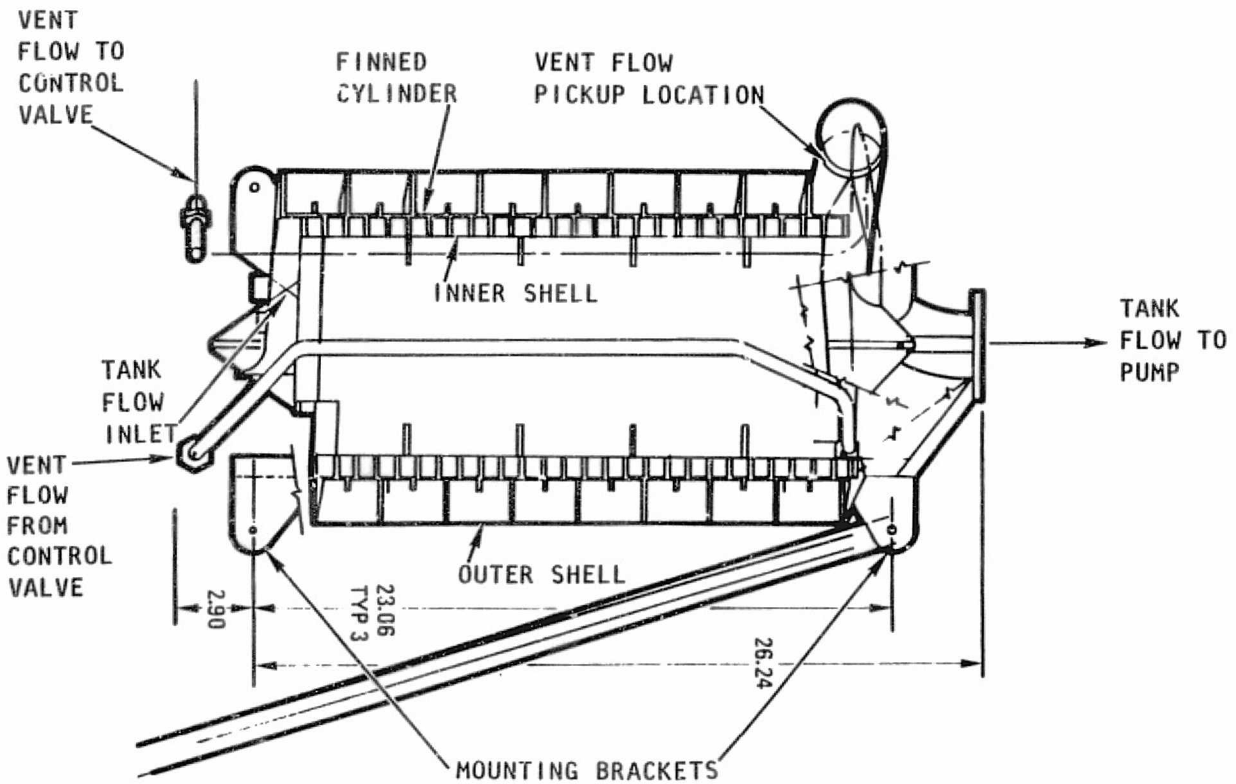
CONTROLLING VALVE MODULE



ZERO GRAVITY TVS MIXER PUMP



SUNDSTRAND HEAT EXCHANGER CONFIGURATION



Orifices - Orifices of various sizes will be necessary within the system to assure proper pressure drop and flow. The orifices may be off-the-shelf type or in-house fabricated type.

Unions, Fitting and Tubes - All hardware plumbing will be of the Shuttle approved standard type. None of these will require component level development and/or testing.