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# LOW THRUST CHEMICAL ORBIT TO ORBIT PROPULSION SYSTEM PROPELLANT MANAGEMENT STUDY

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LIST OF SYMBOLS USED IN MAIN TEXT

ACPS	Auxiliary Control Propulsion System
ACS	Attitude Control System
ASE	Airborne Support Equipment
C.G.	Center-of-Gravity
DOD	Department of Defense
EVA	Extra Vehicular Activity
GEO	Geosynchronous Orbit
Isp	Specific Impulse
K	Thermal Conductivity
LEO	Low Earth Orbit
LeRC	Lewis Research Center
LSS	Large Space Systems
LTPS	Low Thrust Chemical Propulsion System
MLI	Multilayer Insulation
MMU	Manned Maneuvring Unit
MSFC	Marshall Space Flight Center
NOF	Non-Optimum Factor
PP/LSSI	Primary Propulsion/Large Space System Interaction Study
RCS	Space Shuttle Reaction Control System
RP-1	Kerosene
RTLS	Return to Launch Site
SOFI	Spray-On-Foam Insulation
STS	Space Transportation System
T/M	Thrust to Mass Ratio, g
$\Delta V$	Velocity Change, M/S

## SUMMARY

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Inception of the Space Transportation System's (STS) operational flight capability will allow the launching of preconstructed large space platforms for deployment in orbits of various altitudes. For this study, the Large Space System (LSS) is to be placed in geosynchronous orbit by a low thrust chemical orbital transfer propulsion system (LTPS). A single Shuttle flight will launch the mated LTPS/LSS. The LSS is assumed to utilize the remainder of the 27,200 kg (60,000 lb<sub>m</sub>) payload limit and the volume in the orbiter payload bay not occupied by the LTPS.

The objectives of this program were to determine the propellant requirements, preferred propellant management techniques, propulsion system mass, and propellant management technology deficiencies for the LTPS.

Systems were evaluated to determine minimum length and maximum LTPS performance configurations. For the various systems, liquid oxygen (LO<sub>2</sub>) was employed separately with liquid hydrogen (LH<sub>2</sub>), liquid methane (LCH<sub>4</sub>) or kerosene (RP-1). These propellant combinations were held in various tank arrangements including toroidal, cylindrical with ellipsoidal domes, and ellipsoidal tanks. The three discrete thrust levels chosen for investigation were 445, 2225, and 4450 N (100, 500, and 1000 lb<sub>f</sub>). These were combined at nominal mixture ratios, with 1, 4, and 8 perigee burn LEO to GEO transfer strategies. The resulting matrix of systems was evaluated with Multilayer Insulation (MLI) and Spray-On-Foam Insulation (SOFI) Tank coverings. From this array of systems, promising concepts were selected for further refinement and Propellant Management Devices (PMD) were designed for each selected configuration. The techniques examined for propellant management were propellant settling using either the auxiliary propulsion system or main engine idle mode, total acquisition devices composed of screen covered channels, and partial acquisition devices or traps. After the refinement of the LTPS, a brief analysis of its accommodation with the LSS in the orbiter payload bay was completed. Finally, technology deficiencies with respect to the selected systems were determined along with possible methods of overcoming these drawbacks.

Results of system sizing indicated, as expected, that the shortest tankage combination consisted of a toroid mated with either an ellipsoidal or cylindrical-ellipsoidal domed tank. Superior insulation covering was the MLI which produced smaller tanks and resulting in vehicles that were 1,500 kg to 3,000 kg (3,300 lb<sub>m</sub> to 6,600 lb<sub>m</sub>) lighter than comparable systems utilizing SOFI. The use of LO<sub>2</sub>/LH<sub>2</sub> propellants produced the lightest LTPS, but these were also the longest systems (due to the low LH<sub>2</sub> density). The parallel tank arrangement and the tandem/toroidal configuration were evaluated with LO<sub>2</sub>/LCH<sub>4</sub> and both were found to be comparable in LTPS mass and space available for the LSS. Although some LO<sub>2</sub>/RP-1 systems were selected for further evaluation, they were the heaviest systems and are suitable only for a very low packaged density LSS. Evaluation of propellant management techniques resulted in an improved propulsive settling method using a simple surface tension device to delay gas ingestion into the outlet, it was preferred due to its minimum system weight penalty. The maximum performance configuration was found to be a conventional tandem tank arrangement using ellipsoidal tanks or cylindrical-ellipsoidal domed tanks. LO<sub>2</sub>/LH<sub>2</sub> was again the lightest system by approximately 2,000 kg (4,400 lb<sub>m</sub>); but this configuration was also 2 m (6.5 ft) longer than that employing LO<sub>2</sub>/LCH<sub>4</sub>. In the final portion of this study, the technology deficiencies of major concern were found to be the accuracy of propellant settling models and questions concerning surface tension device performance with cryogenics.

Although no one system can be chosen from the group as the best, a number of trends do appear: (1) Eight perigee burns result in considerable mass gains for the LSS over 1 and 4 burns. (2) Toroidal tanks must be developed for the LO<sub>2</sub>/LH<sub>2</sub> propellant combination. Due to the low density of LH<sub>2</sub>, conventional tank arrangements would require excessive orbiter payload bay volume; (3) When LCH<sub>4</sub> is the fuel, configurations using parallel tanks or tandem/toroidal tanks could be used. Less risk would be involved in the system development if the parallel tank configuration were used; (4) Propellant settling using a bubble trap type of screen device in the bottom of the tank is the simplest method of propellant management and has the lowest weight penalty; (5) The characteristics of the LSS will effect the final

choice of the matching LTPS. The  $LO_2/LH_2$  tandem/toroidal configuration is best suited for a shorter, high density LSS. Vehicles utilizing  $LO_2/LCH_4$  in either a tandem/toroidal or parallel tank arrangement would be required for low density LSS over 10 m (33 ft) in packaged length.



## I. INTRODUCTION

The availability of the Space Shuttle Transportation System (STS) in the early 1980s will make the production of on-orbit Large Space Systems (LSS) feasible. Studies performed by various agencies of government (NASA, DOD), Martin Marietta, and the remainder of the aerospace industry indicate that to meet future needs large antennas and platforms will be required either in Low Earth Orbit (LEO) or in Geosynchronous Earth Orbit (GEO). Specific applications, both civilian and military, have been identified in several recent studies.

In general terms large space structures are classified as either deployable or erectable, depending upon the process used to place them into operational status. With deployable structures, the entire manufacturing and assembly takes place on the ground, and the package in a high density form is flown into space where it is then deployed. The concept of erectable structures refers to assembly in space either by a building crew or by remote manipulation. Propulsion systems required to transfer these general types of structures from LEO to GEO can be either high or low thrust, depending upon the load bearing capability of the structure, which in turn depends upon the method and location selected for the final assembly. The objective of this study program was to address propulsion system concepts with low thrust levels using the specified conventional chemical propellants. Specifically, this study provided an evaluation of propellant management techniques for low thrust level chemical propulsion systems.

The specific objectives of this program were to determine propellant requirements, preferred propellant management techniques, propulsion system weights, and technology deficiencies for low thrust chemical orbit to orbit propulsion systems (LTPS) for LSS applications. The effort was divided into four tasks with the following individual objectives:

### Task I - Determination of Propellant Requirements

With the aid of an analytical computer model, 72 different propulsion systems were analyzed to determine the mass of propellant and tankage required by expendable low thrust chemical propulsion systems designed to transport the LSS from LEO to GEO. Each system was designed and sized to maximize the Shuttle cargo bay volume available to the LSS;

### Task II - Evaluation of Propellant Management Techniques

At the completion of Task I, attractive concepts for each propellant combination, and various thrust levels were selected for further study where three different propellant management schemes (propulsive settling, total and partial acquisition surface tension devices) were incorporated. The feasibility and weight of each system was assessed;

### Task III - Improved LTPS Concepts

Three promising LTPS concepts were further developed and optimized, paying particular attention to simplified propellant acquisition, improved LTPS/LSS packaging or integration, and further thermal insulation system optimization with the goal of increasing the available LSS weight; and

### Task IV - Technology Evaluation

The technology required for each of the identified LTPS vehicles was evaluated to determine the adequacy of current technology to permit detailed design and development of each concept.

## II. DETERMINATION OF PROPELLANT REQUIREMENTS

With the aid of an analytical computer model propulsion systems were analyzed to determine the weight of propellant and tankage required by expendable low thrust chemical propulsion systems (LTPS) designed to transport Large Space Systems (LSS) from low earth orbit (LEO) to geosynchronous orbit (GEO). Each system was designed and sized to maximize the Shuttle cargo bay volume available to the LSS.

### A. MISSION REQUIREMENTS

#### 1) Performance Specifications

Orbital transfer is accomplished by multiple perigee burns of the low thrust engine and a final burn at apogee that circularizes the orbit at the required altitude for GEO. Figure II-1 depicts a sequence of orbits resulting from an eight perigee burn strategy using a typical low-thrust propulsive system with an initial thrust to mass ratio of 0.01. Design points used for this study are shown in Table II-1; all data in the table were supplied by NASA-Lewis Research Center (LeRC). The combinations of propellants, engine thrust, and number of perigee burns were evaluated with various insulation concepts and tanking arrangements to determine the candidates chosen for further evaluation.

#### 2) Mission Timeline

The mission timeline was also specified by NASA-LeRC. Propellant topping is allowed to liftoff (T-zero) minus four minutes. Between T-zero and T plus 90 seconds the tank is locked-up with no venting of propellant vapor allowed. Any increase in pressure during the lockup period is not to exceed 41 kPa (6 psi); nominal pressure at T-zero is 124 kPa (18 psia). Space Transportation System (STS) launch, on-orbit checkout, and LTPS/LSS deployment from the orbiter cargo bay will require two hours. An additional 40 hours is required for erection and checkout of the LSS. The orbital transfer time from LEO to GEO, shown in Table II-1, depends on the propellant/thrust/burn strategy combination being evaluated for a particular case.

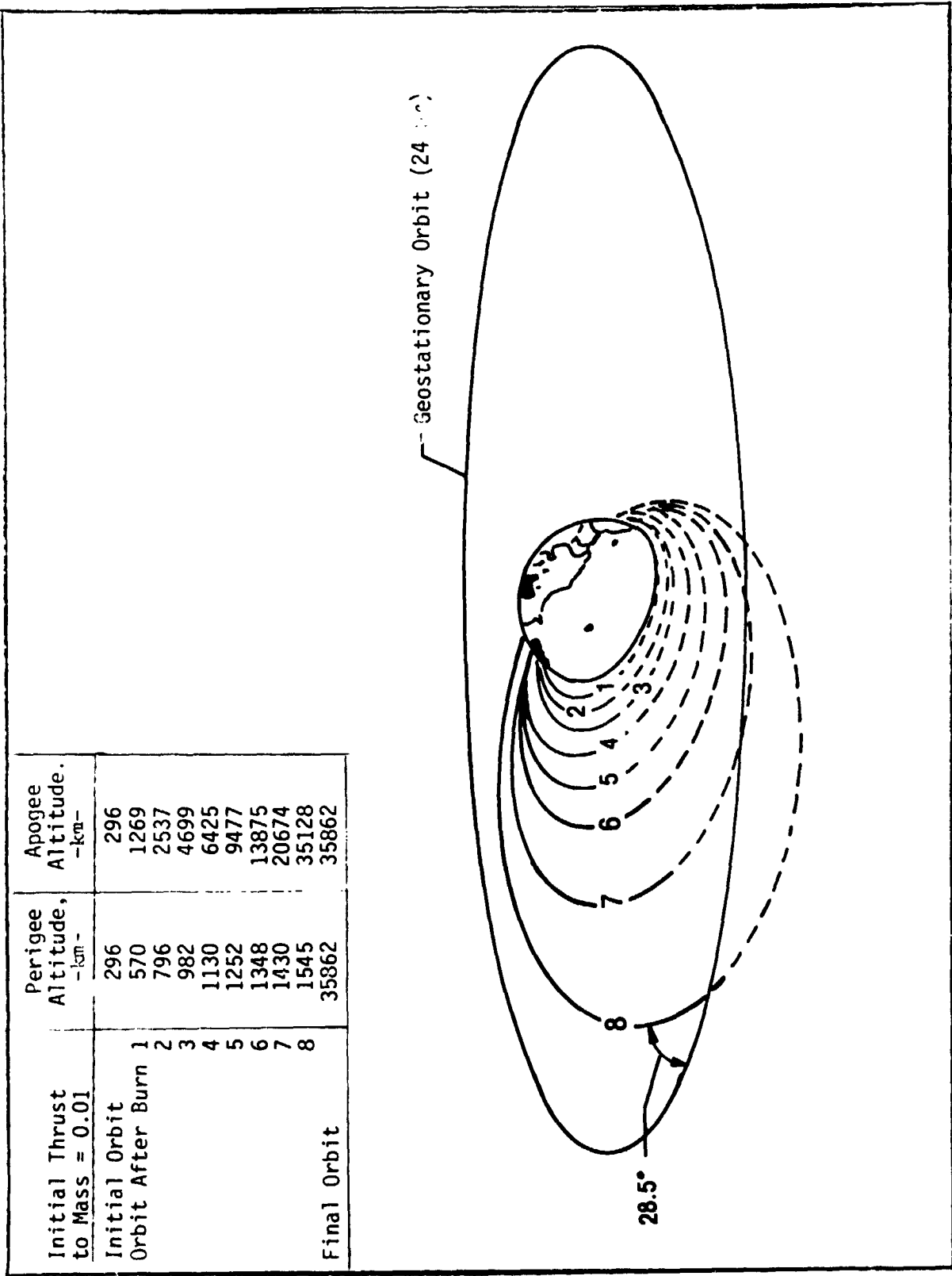


FIGURE II-1 SEQUENCE OF ORBITS FOR AN EIGHT BURN TRANSFER STRATEGY  
VIEWPOINT = 15°N, 135°W, T/H<sub>0</sub> = 0.01

TABLE II-1 SELECTED LTPS POINT DESIGN PARAMETERS*								
PROPELLANT COMBINATION	THRUST		NO. OF PERIGEE BURNS	Isp		TOTAL ΔV REQUIRED		LEO TO GEO TRANSFER TIME, hrs
	N	lbf		$\frac{N \cdot \text{sec}}{\text{kg}}$	$\frac{\text{lbf} \cdot \text{sec}}{\text{lbm}}$	m/sec	ft/sec	
LO <sub>2</sub> /LH <sub>2</sub> MR=6:1	445	100	1	4145	422.5	5537.1	18,166.3	59.21
			4			5271.5	17,294.8	61.36
			8			4983.4	16,349.9	72.37
	2225	500	1	4316	440.0	5289.0	17,352.4	16.89
			4			4855.8	15,931.2	19.83
			8			4448.2	14,593.9	31.76
	4450	1000	1	4405	449.0	5148.8	16,892.4	11.74
			4			4732.4	15,526.1	14.91
			8			4413.4	14,479.7	27.11
LO <sub>2</sub> /LCH <sub>4</sub> MR=3.7:1	445	100	1	3311	337.5	5524.9	18,126.3	52.85
			4			5261.7	17,262.8	55.37
			8			4976.3	16,326.6	66.74
	2225	500	1	3497	356.5	5260.4	17,258.6	15.77
			4			4838.5	15,874.2	18.83
			8			4441.4	14,571.4	30.87
	4450	1000	1	3572	364.5	5108.1	16,759.0	11.19
			4			4709.3	15,450.4	14.41
			8			4403.8	14,448.1	26.67
LO <sub>2</sub> /RP-1 MR=3:1	445	100	1	3115	317.5	5521.6	18,115.5	51.08
			4			5259.0	17,254.1	53.69
			8			4974.4	16,320.3	65.16
	2225	500	1	3272	333.5	5251.2	17,228.5	15.40
			4			4832.8	15,855.8	18.50
			8			4439.2	14,564.2	30.79
	4450	1000	1	3365	343.0	5096.5	16,720.9	11.03
			4			4702.7	15,428.8	14.27
			8			4410.0	14,438.9	26.53

\* As supplied by NASA LeRC (Customary units only)

During the 42 hours prior to the first LTPS burn, cryogenic propellant that evaporated would be vented, and thus, the mass of the LTPS/LSS at initial ignition would be less than the 27,220 kg (60,000 lb<sub>m</sub>) specified for all cases at STS liftoff.

#### B. PROPELLANT SYSTEM CHARACTERIZATION APPROACH

A simple analytical computer program to size the propulsion system was used to evaluate the candidates. This program (PROP) was written and checked out during the early Viking program and has been used many times since as a design and analysis tool. The program has four major system options. First, the choice of a monopropellant or bipropellant propulsion system using cryogenic and/or earth-storable propellants. Second, the pressurization system sizing includes either a blowdown or a regulated case; in addition, another mode bypasses the pressurization sizing loop and substitutes a fixed input mass to accommodate other types of systems (autogenous, etc). Third, available propellant tank shapes are: 1) spherical, 2) cylindrical with hemispherical ends, 3) cylindrical with  $\sqrt{2}$  ellipsoidal ends, 4)  $\sqrt{2}$  ellipsoidal and 5) toroidal. The fourth option allows the input/output units to be specified in one of four combinations: 1) English/English, 2) English/SI, 3) English/English and SI, and 4) SI/SI. Other options are chosen at input, such as to specify vehicle mass, delta-V, and Isp, allowing the computer to calculate the propellant mass; or to specify the mass of propellant burned. Also, the program will model a wide range of adiabatic or isothermal burns.

The program output includes a complete propellant inventory (including boil-off for cryogenic cases), pressurant and propellant tank dimensions for a given ullage, pressurant requirements, insulation requirements, and miscellaneous masses. The output also includes the masses of all tanks; the mass of the insulation, engines and other components; total wet system and burnout mass; system mass fraction; total impulse; and burn time.

In addition, a modification was programmed to provide the capability to calculate the remaining mass, volume, and ullage height at the beginning of all burns, for each propellant. The ullage height is the length of the inside of the tank minus the height of the propellant if it were all settled in the bottom of the tank. Also calculated at the initiation of each burn is the total system mass and acceleration along with the burn duration. The same variables, except ullage height and burn duration, are also computed at the end of the circularization burn. The final outputs are propellant tank dimensions. A simplified flow chart of the program appears in Figure II-2, and sample inputs and outputs are shown in Appendix A.

#### C. DESIGN CRITERIA

In the first phase of the analysis, the criterion was to design the propulsion systems to maximize Shuttle cargo bay volume available to the LSS. The resulting objective is to minimize LTPS length. From the original set of candidates, a selected number were chosen for further evaluation with the incorporation of propellant management schemes in subsequent studies.

In Section IV of this report, the emphasis is changed from maximizing cargo bay volume available for the LSS to maximizing mass available for the LSS.

#### D. CANDIDATES FOR STUDY

##### 1) Propellants

Three propellant combinations were chosen for study - two were cryogenic and one was a cryogen/storable combination. Liquid Oxygen ( $LO_2$ ) is the oxidizer used for all three combinations, and it is paired with Liquid Hydrogen ( $LH_2$ ), Liquid Methane ( $LCH_4$ ) and Kerosene (RP-1).

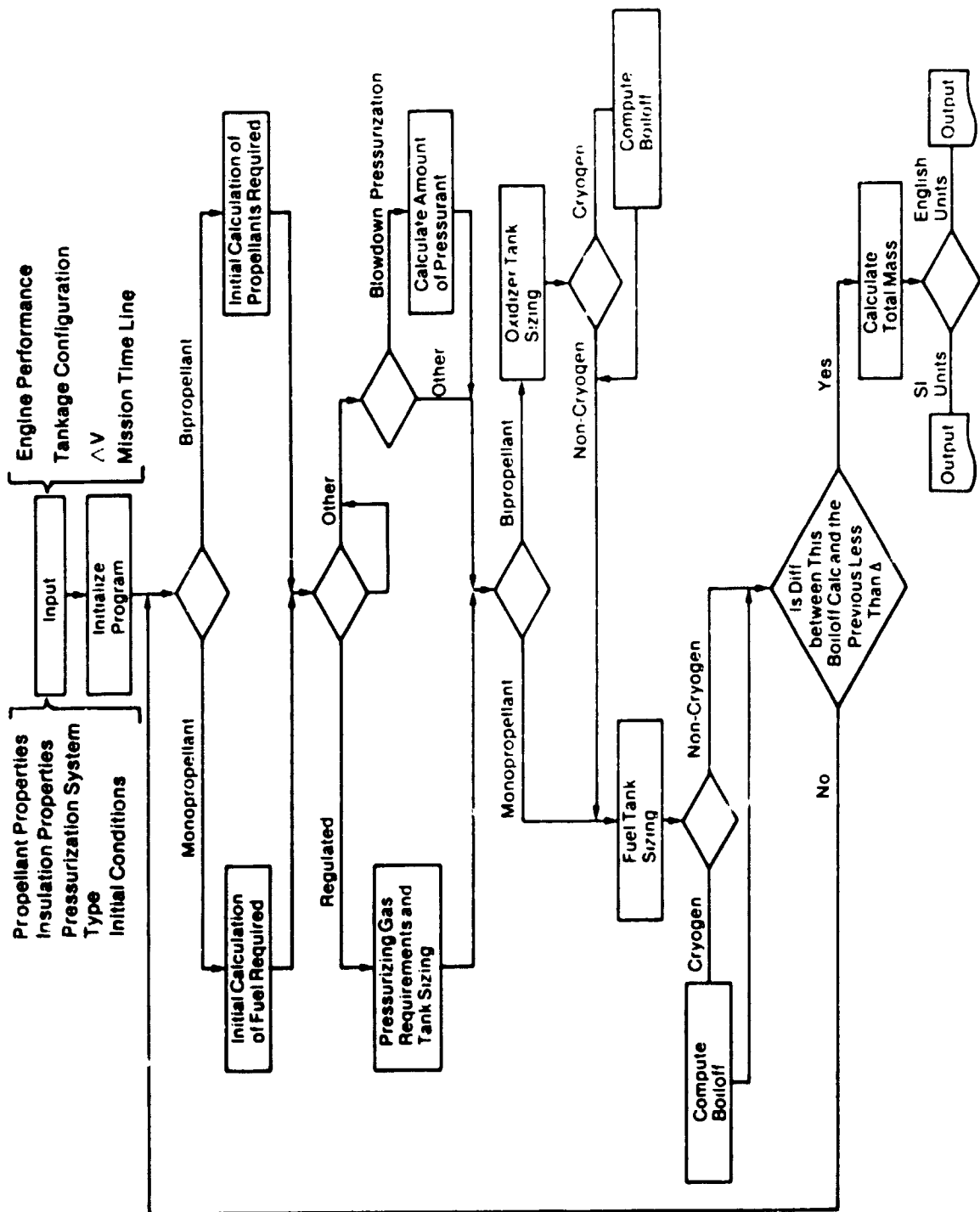


FIGURE II-2 PROP PROGRAM SUMMARY FLOW CHART



The  $\text{LO}_2/\text{LH}_2$  combination offers high specific impulse (Isp) [4150 to 4400 N-sec/kg (423 to 450  $\text{lb}_f\text{-sec}/\text{lb}_m$ )] and clean burning qualities important for engine restart capability. But the  $\text{LH}_2$  has a very low density [ $\sim 64 \text{ kg}/\text{m}^3$  ( $4 \text{ lb}_m/\text{ft}^3$ )], which represents a large volume penalty. Combining  $\text{LO}_2$  and  $\text{LCH}_4$  will provide two "soft" cryogenics, reasonable clean burning, and the  $\text{LCH}_4$  has an attractive density [ $413 \text{ kg}/\text{m}^3$  ( $26 \text{ lb}_m/\text{ft}^3$ )] compared to  $\text{LH}_2$ . This combination has a modest Isp [3310 to 3570 N-sec/kg (338-365  $\text{lb}_f\text{-sec}/\text{lb}_m$ )] resulting in a reduction in mass available for the LSS. The third combination is  $\text{LO}_2/\text{RP-1}$ . This fuel has a high density [ $806 \text{ kg}/\text{m}^3$  ( $50 \text{ lb}_m/\text{ft}^3$ )] and thermal insulation requirements are reduced because RP-1 is an earth storable. However, the coking problems caused by using a hydrocarbon fuel makes restart very difficult, and it has a relatively low Isp [3120 to 3370 N-sec/kg (318 to 343  $\text{lb}_f\text{-sec}/\text{lb}_m$ )].

## 2) Thrust Levels and Burn Strategy

Thrust levels and burn strategy influence both the total  $\Delta V$  requirements and total orbit transfer trip time. Three discrete thrust levels were chosen for evaluation: 445, 2225, and 4450 N (100, 500, and 1000  $\text{lb}_f$ ). Burn strategies of 1, 4 and 8 perigee burns were selected to be combined with the various thrust levels.

As the thrust and number of burns increases, the individual burn time at perigee decreases; the result is smaller gravity losses which decreases the total  $\Delta V$  requirements. In Figure II-3 the 8 perigee burn shows a considerable reduction in required velocity increment when compared to the single burn approach in the acceleration (T/M) range of  $10^{-1}$  to  $10^{-2} \text{ g's}$ . Lower  $\Delta V$  requirements result in smaller amounts of propellant. Boiloff of cryogenic propellants is directly related to orbital transfer trip time. Trip time starts to increase rapidly at a T/M of approximately 0.03g for both 1 and 8 burns in Figure II-4. At these T/M levels the difference in trip time for 1 or 8 burns is about 15 hours. As T/M increases above 0.03g, trip time for 8 burn stays almost constant while trip time for 1 burn continues to decrease making the difference between burn strategies even longer. As can already be

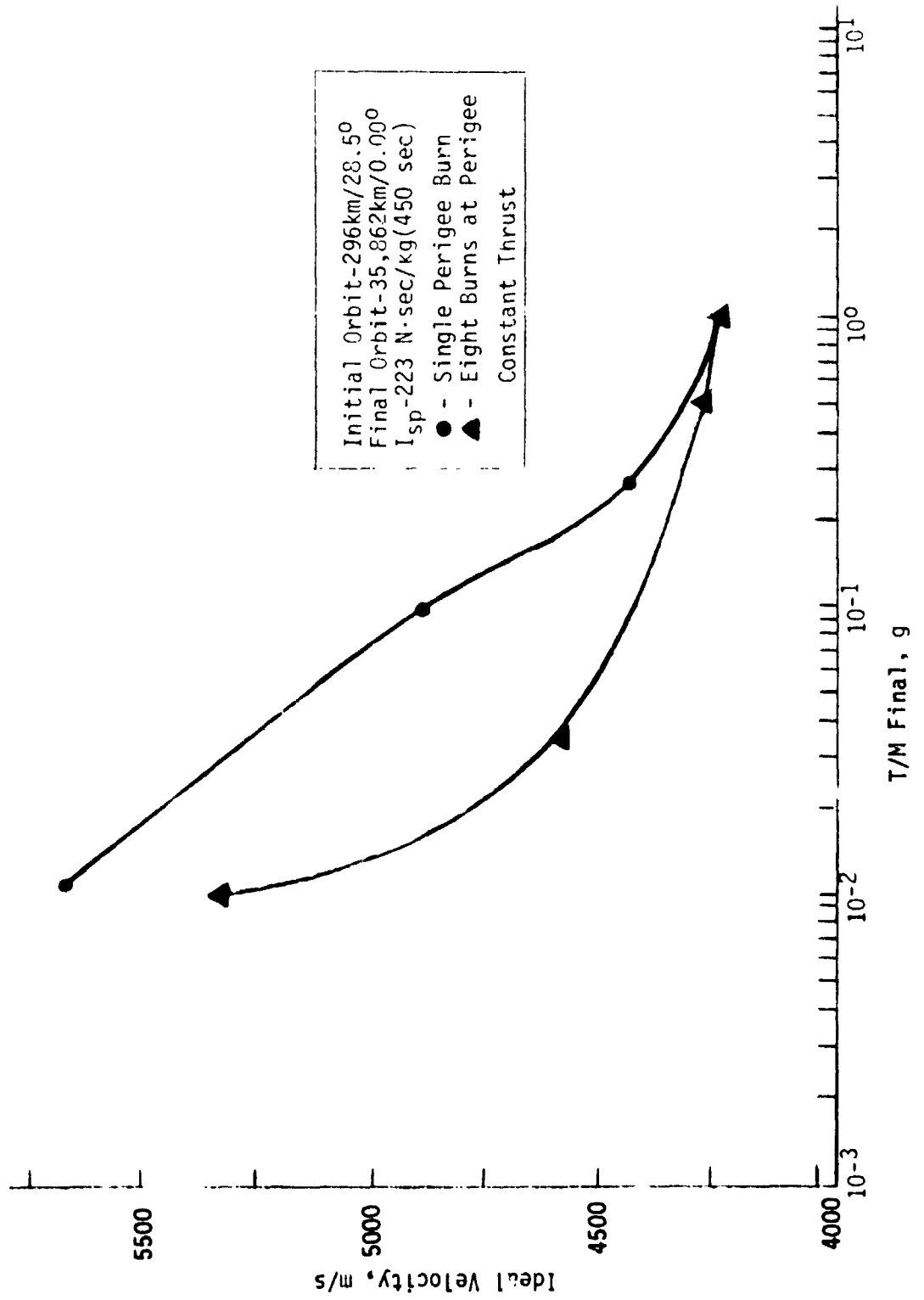


FIGURE II-3 TOTAL CHANGE IN VELOCITY REQUIREMENTS

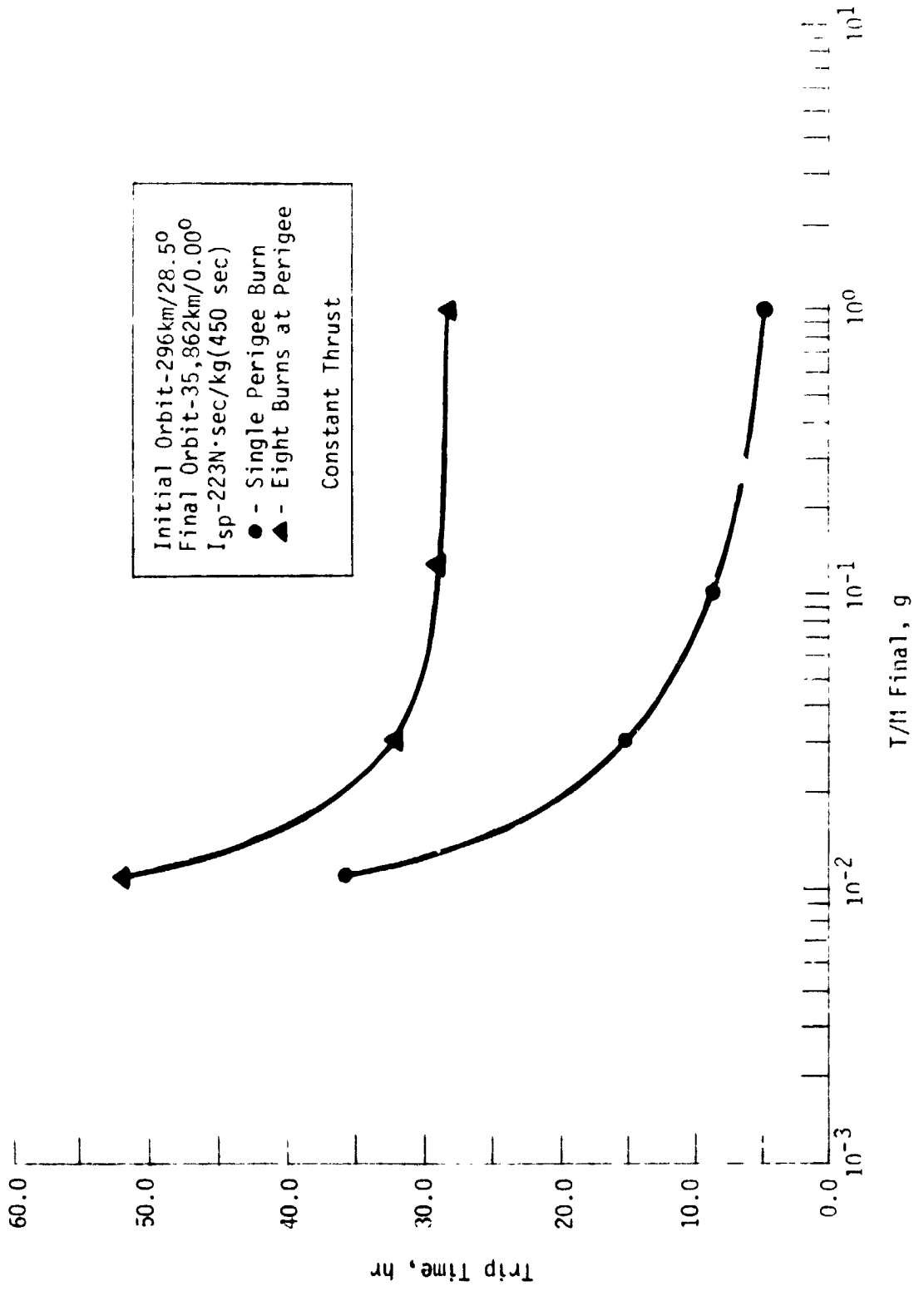


FIGURE II-4 TRIP TIME REQUIREMENTS

seen, increasing the thrust and the number of burns will decrease the mass of propellant needed. However, both improvements have attendant drawbacks. A T/M exists at which any increase in thrust will increase the structural requirements of the LSS, thus increasing the required structural mass. This problem was addressed by Martin Marietta in another LeRC contract (NAS3-21955), "Primary Propulsion/Large Space Systems Interactions Study". Engine long life and multiple restart capability will require advancement in engine technology.

### 3) Tank Insulation Concepts

A number of different insulation systems were considered as LTPS candidates. The two most promising concepts were a Multilayer Insulation system (MLI) with a helium purge bag and the Spray-On-Foam Insulation (SOFI) utilized on the Space Shuttle External Tank program. The SOFI (CPR-488) was compared with other foam insulations (Ref. 1), and it was selected because it had the best balance between low density and good thermal conductivity.

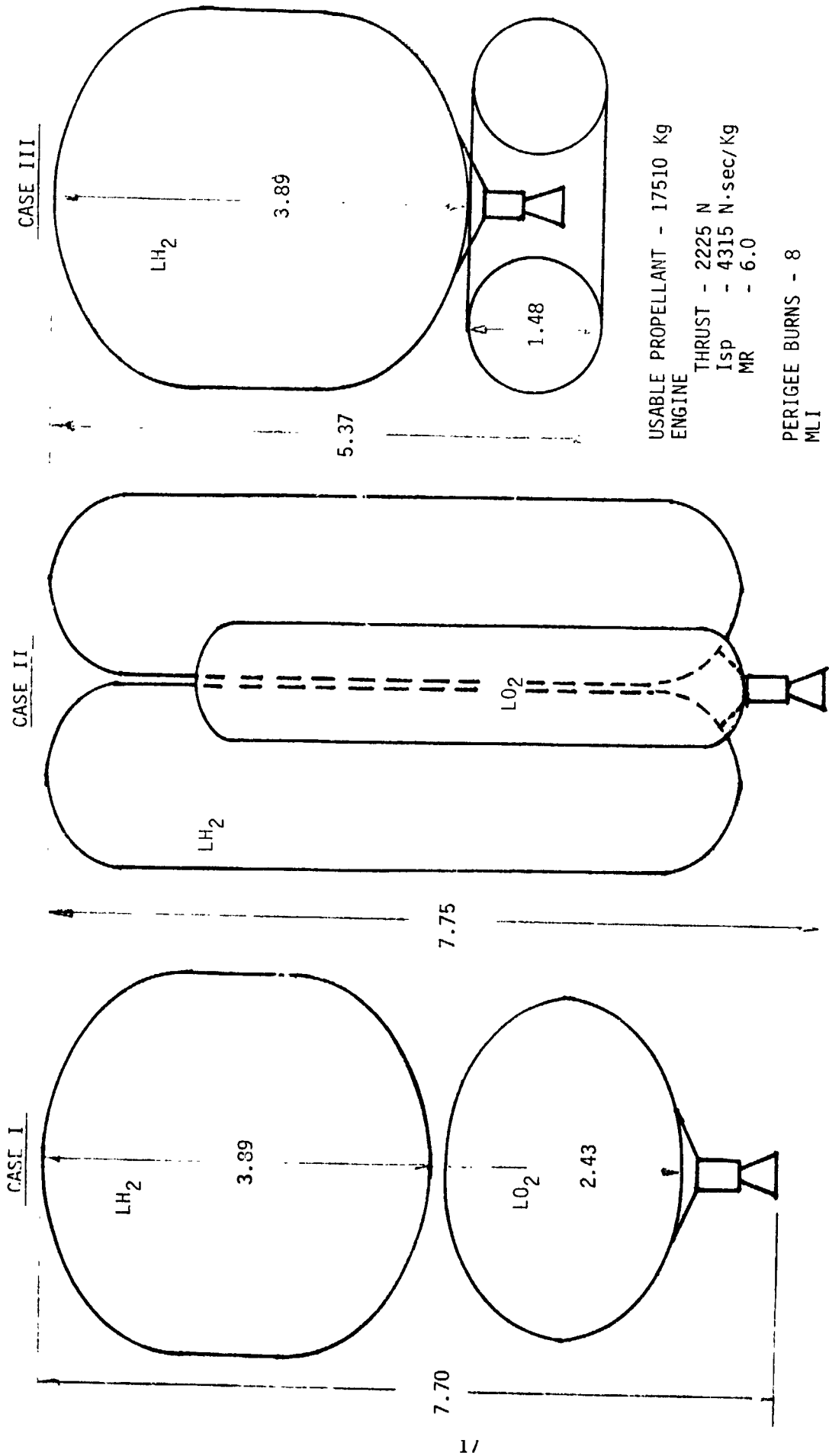
### 4) Tanks

Based on previous Tug studies (Ref. 2) several of the most promising configurations were chosen for this study, and in preparation for the propulsion system characterization studies using PROP, some preliminary configuration sizing calculations were performed. Each of the LTPS propellant combinations were evaluated for both maximum and minimum propellant loads. The usable propellant quantities were calculated using the ideal velocity equation and the velocity increments and specific impulses for each propellant combination, burn strategy, and thrust level (itemized in Table II-1). The minimum loads were derived from the maximum thrust, maximum  $I_{sp}$  and 8 perigee burn conditions; while the maximum loads were derived from the minimum thrust, minimum  $I_{sp}$  and 1 perigee burn conditions. For preliminary tank sizing calculations, four percent of usable propellant was added to account for trapped propellant, five percent for boiloff and a two percent ullage.

A typical example of the three different propulsion system configurations considered for each propellant combination is shown in Figure II-5. This example shows the  $LO_2/LH_2$  cases. Case I is the series "conventional" tankage configuration utilizing either ellipsoidal (for this study all ellipsoidal tanks have  $\sqrt{2}$  domes) or cylindrical/ellipsoidal domed tanks. Case II is the parallel tank configuration utilizing four cylindrical/ellipsoidal domed tanks. The specific oxidizer and fuel tank diameters for Case II were selected by using the analysis in Appendix B, assuming a distance of 0.15 m between adjoining tanks to allow for insulation and clearance. Case III is the series "non-conventional" tankage configuration utilizing a toroidal tank and either an ellipsoidal or a cylindrical/ellipsoidal domed tank. This case was expected to have the minimum performance (due to inefficiencies of toroidal tanks) and also minimum length, while Case I was anticipated to have the maximum performance and maximum length. For this preliminary sizing all tanks were contained inside a 4.27 m (14 ft) diameter package.

In comparing overall stage lengths among any three cases (for a given propellant combination and propellant load) the engine length can be a factor. For Case I and II the engine length always adds directly to the length of the tankage involved (two tanks for Case I or one tank for Case II). However, for Case III, if the torus diameter becomes large enough, the engine can become totally buried and the stage length will no longer be a function of the engine length. Thus, proper modeling of engine length can be an important factor in determining the shortest stage length. For this study NASA-LeRC supplied engine envelopes for all three thrust levels (this data is included in Table II-3).

Figures II-5 through II-7 show the results of the preliminary configuration sizing study for the various propellant combinations and loads. The shortest configuration for every propellant combination and load was Case III; however, the longest varied with the propellants. For  $LO_2/LH_2$ , Case II was longest while for  $LO_2/LCH_4$  and  $LO_2/RP-1$  Case I was longest. It was anticipated that Case I would have the best stage performance and Case III would have the worst. The final computer analysis provided the actual payload values for each case to better compare optimum performance and optimum packaging.



NOTE:  
DIMENSIONS IN METERS  
MAXIMUM DIAMETER = 4.27m  
ENGINE LENGTH = 1.07m

FIGURE II-5 TYPICAL PRELIMINARY TANKAGE CONFIGURATION - L02/LH2

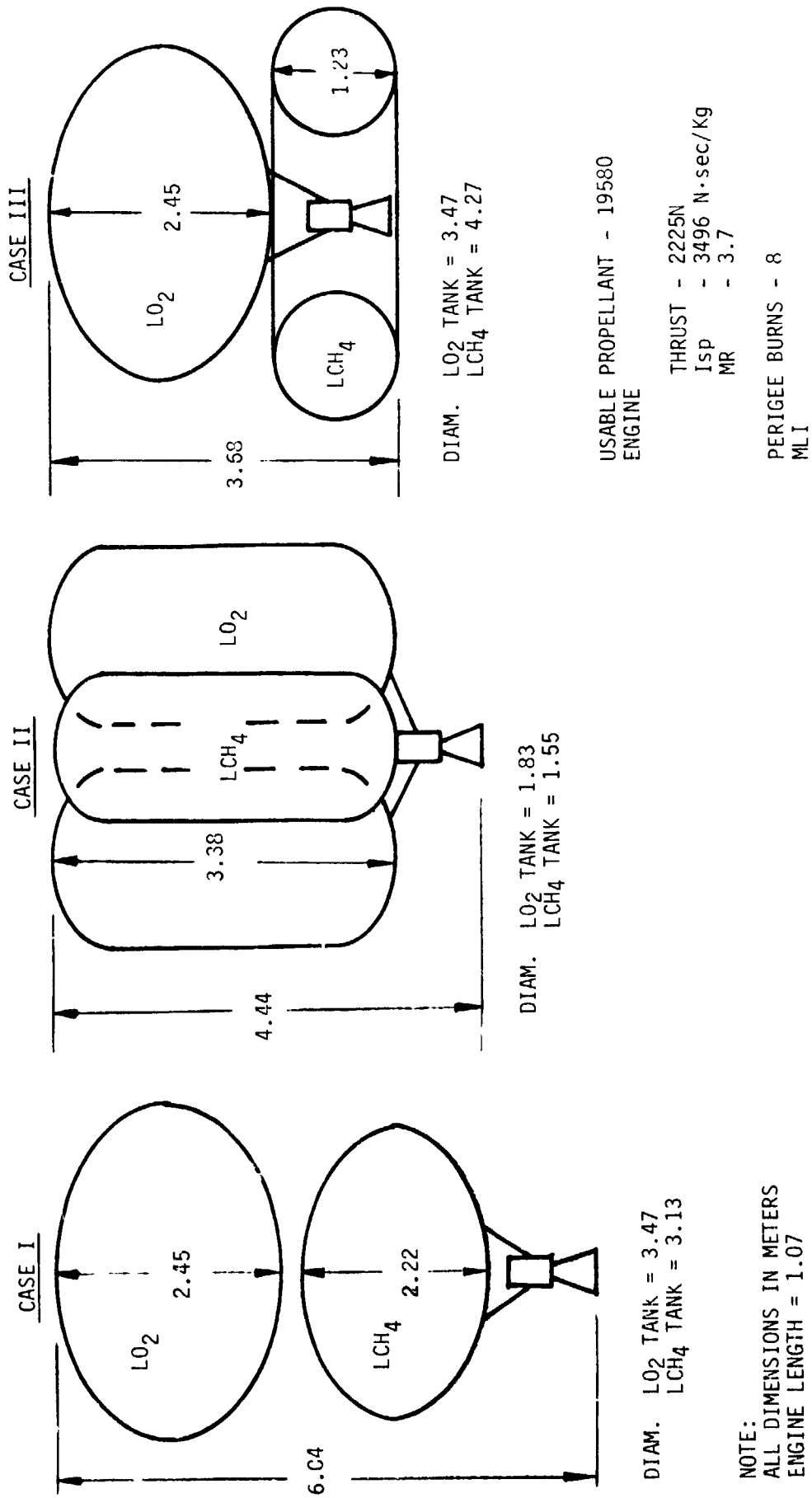
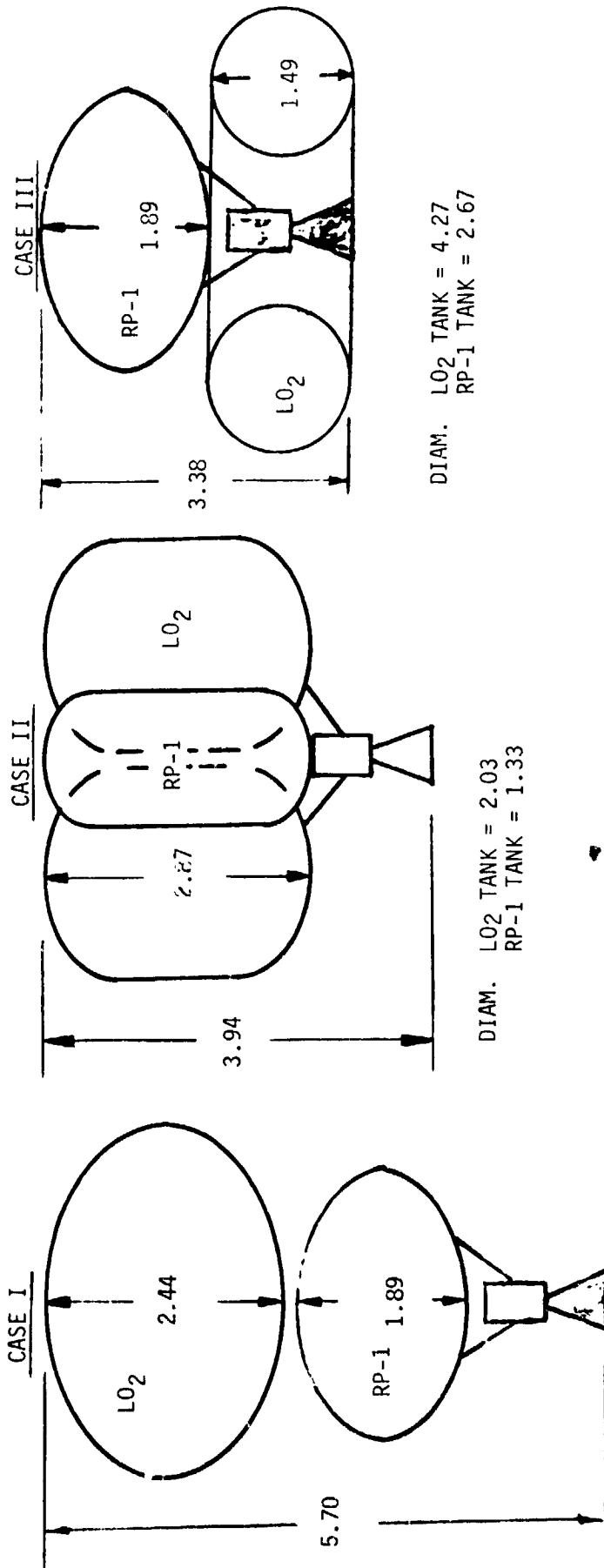


FIGURE II-6 TYPICAL PRELIMINARY TANKAGE CONFIGURATION - LO<sub>2</sub>/LCH<sub>4</sub>



DIAM. L02 TANK = 3.45  
 RP-1 TANK = 2.67

NOTE:  
 ALL DIMENSIONS IN METERS  
 ENGINE LENGTH = 1.07m

DIAM. L02 TANK = 4.27  
 RP-1 TANK = 2.67

DIAM. L02 TANK = 2.03  
 RP-1 TANK = 1.33

USABLE PROPELLANT - 20210 Kg  
 ENGINE

THRUST - 2225 N  
 Isp - 3270 N·sec/Kg  
 MR - 3.0

PERIGEE BURNS - 8  
 MLI

FIGURE II-7 TYPICAL PRELIMINARY TANKAGE CONFIGURATION - L02/RP-1



Although the tandem/toroidal tank combination was always shortest, it was decided to also evaluate the parallel tanks configuration with  $LO_2/LCH_4$ . Torodial tanks needed for the LTPS will require considerable developmental work and represent a new challenge in thermal and structural analysis. The cylindrical ellipsoidal domed tanks would be a lower developmental risk and a length penalty of only 70 cm for the  $LCH_4$  fueled concepts.  $LO_2/LCH_4$  is attractive for parallel tanks because the temperature difference between the cryogenics is about  $20^\circ C$  resulting in a small amount of radiative energy transfer and thermal conduction between propellant tanks.

Two alternative tank arrangements to the tandem/toroidal configuration were evaluated in an attempt to improve overall stage packaging efficiency by reducing length. The  $LO_2/LH_2$  maximum load case is presented as an example:

$$\begin{aligned}
 W_p &= 20,090 \text{ kg} \\
 V_{LH_2} &= 45.6 \text{ m}^3 \\
 V_{LO_2} &= 15.8 \text{ m}^3 \\
 \text{Maximum Stage Diameter} &= 4.27 \text{ m} \\
 \text{All domes are } \sqrt{2} \text{ semi-ellipsoid}
 \end{aligned}$$

(a) Parallel Tanks/Embedded Engine Concept

To embed the engine in the center space of the parallel tank arrangement, the individual tank diameters must be reduced to create a space for at least the engine thrust chamber assembly. To determine the corresponding increase in length of the tank requires calculating the volume as a function of the length. From Figure II-8.

$$\begin{aligned}
 V_{\text{Tank}} &= V_{\text{Cylinder}} + V_{\text{Domes}} \\
 &= \pi r^2 L_B + \frac{4}{3} \pi r^2 \frac{r}{\sqrt{2}}
 \end{aligned}$$

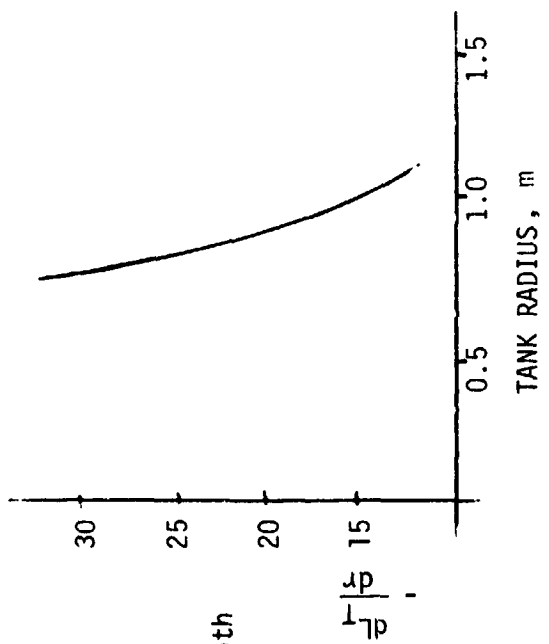
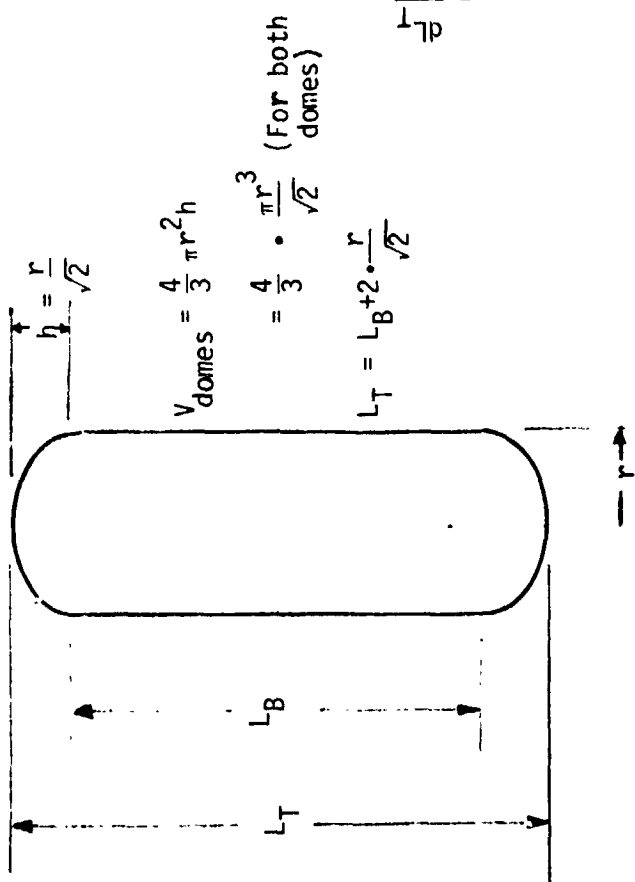


FIGURE I'-8 CYLINDRICAL TANK WITH  $\sqrt{2}$  ELLIPTICAL DOMES

FIGURE II-9 CHANGE IN TANK LENGTH DUE TO RADIUS VARIATION

or

$$\frac{V_T}{\pi r^2} = L_B + \frac{4}{3} \frac{r}{\sqrt{2}}$$

For the overall tank length ( $L_T$ ) as a function of  $r$ ,

$$\begin{aligned} \frac{V_T}{\pi r^2} &= \left[ L_B + \frac{4}{3} \frac{r}{\sqrt{2}} + \frac{2}{3} \frac{r}{\sqrt{2}} \right] - \frac{2}{3} \frac{r}{\sqrt{2}} \\ &= L_T - \frac{2}{3} \frac{r}{\sqrt{2}} \end{aligned}$$

or

$$L_T = \frac{V_T}{\pi r^2} + \frac{2}{3\sqrt{2}} r = \frac{V_T}{\pi r^2} + 0.4714r$$

To find the variation in tank length for a change in radius, the derivative of  $L_T$  with respect to  $r$  is

$$\left( \frac{dL_T}{dr} \right)_{\text{avg}} = -\frac{2V_T}{\pi r^3} + 0.4714$$

Since  $dL_T/dr$  is obviously nonlinear, an average value over some  $\Delta r$  can be found only by integrating. Thus

$$\left( \frac{dL_T}{dr} \right)_{\text{avg}} = \frac{1}{\Delta r} \int_{r_1}^{r_2} \frac{dL_T}{dr} dr$$

and

$$\begin{aligned} \Delta L_T &= \Delta r \left( \frac{dL_T}{dr} \right)_{\text{avg}} = \int_{r_1}^{r_2} \frac{dL_T}{dr} dr \\ &= \int_{r_1}^{r_2} \left[ -\frac{2V}{\pi r^3} + 0.4714 \right] dr \end{aligned}$$

$$\Delta L_T = \left[ \frac{V}{\pi r^3} + 0.4714 \right]_{r_1}^{r_2}$$

For the baseline case, the LH<sub>2</sub> tank determines the governing length. Each LH<sub>2</sub> tank will have a volume of 22.8 m<sup>3</sup> and a radius of 1.07 m giving a  $dL_T/dr = -12$ . As the radius decreases,  $-dL_T/dr$  increases sharply as shown in Figure II-9. The chamber diameter is added to an 8 cm clearance either side of the engine for insulation and to allow for gimbaling of the engine. Using this approach Table II-2 lists the revised stage length changes for a maximum and minimum case for each propellant combination. Embedding the engine a'ways results in a net gain stage length and so this arrangement is still longer than the tandem/toroid. The engine dimensions supplied by NASA LeRC are shown in Table II-3.

(b) Common Bulkheads

For this analysis, the same LO<sub>2</sub>/LH<sub>2</sub> example as in the previous case was utilized. This analysis uses a combination of a conventional ellipsoidal domed tank and an inverted ellipsoidal domed tank. The two variations considered are shown in Figure II-10. The overall stage length was calculated using (a) an inverted dome tank for the oxidizer tank with no change to the fuel tank, and (b) an inverted dome fuel tank with no change to the oxidizer tank. The shortest configuration was option (a), but it was still 0.52 m longer than Case III, the tandem/toroidal arrangement presented in Figure II-5. The concentric bulkhead tanks represent intermediate stage lengths. However, they also represent potential weight penalties due to extra stresses and resultant thickness increases in the inverted domes. Therefore no further consideration was given to common bulkheads, and the tandem/toroidal tank combination was used as the baseline to satisfy the minimum length constraint.

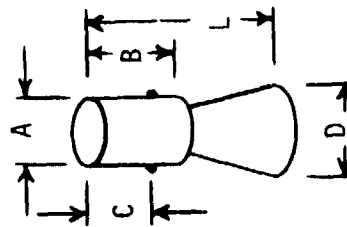
(c) Materials and Weights

All propellant tanks were assumed to be constructed of 2219-T87 aluminum and were designed for a maximum pressure of 165 kPa (24 psia), and a safety factor of 1.5 which is required for all STS propellant vessels. The tank shell mass is calculated by multiplying average tank thickness, tank surface area, and density of the tank material. This mass is then multiplied by a non-optimum factor (NOF) to account for welds, flanges and internal tank supports. The NOF for the ellipsoidal tank derived from previous experience with the ET and Titan tanks, was 1.3 (30% increase in mass). Toroidal tanks were estimated to have a NOF of 1.5.

TABLE II-2 CHANGE IN STAGE LENGTH DUE TO EMBEDDED ENGINE

Propellant Combination	Propellant Mass,		Thrust Level,	Increase in Tank Length,		Engine Length,		Increase in Stage Length,		
	kg	lbm		N	lb <sub>f</sub>	m	ft	m	ft	m
LO <sub>2</sub> /LCH <sub>4</sub>	22,090	48,700	445	100	1.29	4.2	0.91	3.0	+ 0.38	1.2
LO <sub>2</sub> /LCH <sub>4</sub>	19,280	42,500	4448	1000	1.42	4.7	1.22	4.0	+ 0.20	0.7
LO <sub>2</sub> /RP-1	22,590	49,800	445	100	1.24	4.1	0.91	3.0	+ 0.33	0.9
LO <sub>2</sub> /RP-1	19,280	42,500	4448	1000	1.44	4.7	1.22	4.0	+ 0.22	0.7
LO <sub>2</sub> /LH <sub>2</sub>	20,090	46,100	445	100	2.01	6.6	0.91	3.0	+ 1.10	3.6
LO <sub>2</sub> /LH <sub>2</sub>	17,240	38,000	4448	1000	2.15	7.1	1.22	4.0	+ 0.93	3.1

TABLE II-3 LOW THRUST ENGINE DIMENSIONS SUPPLIED BY NASA LeRC

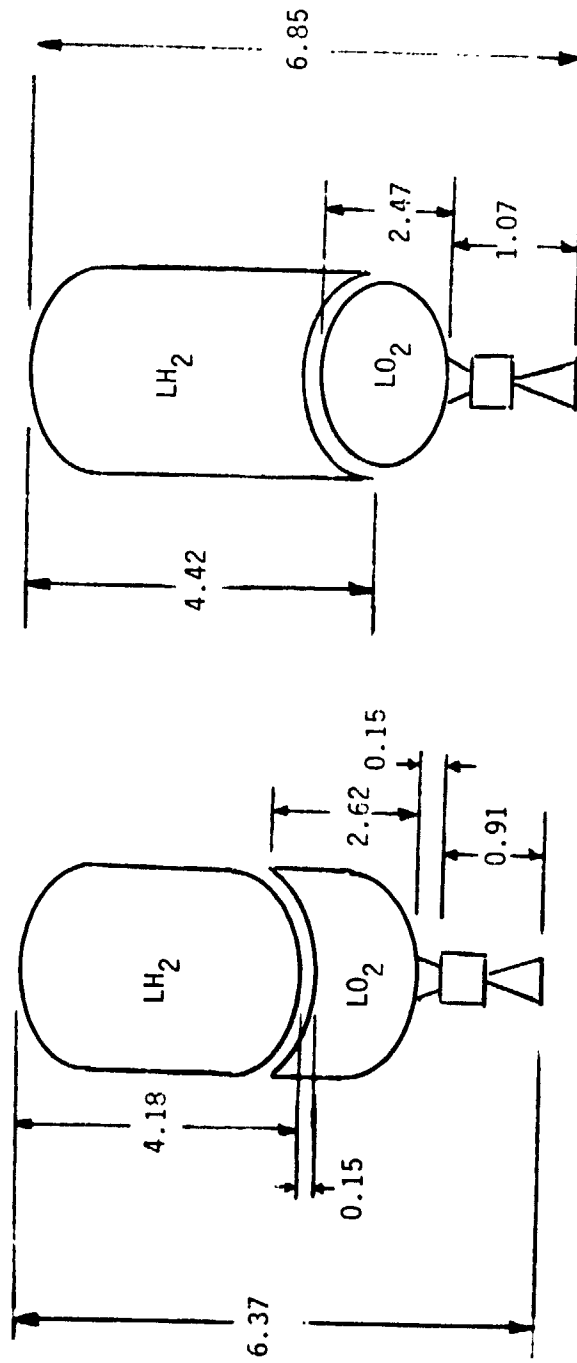


- A - Chamber Diameter
- B - Length of Cylinder
- C - Length from top of Cylinder to Gimbal Point ( $\pm 7^\circ$ )
- D - Exit Diameter
- L - Overall Length
- $\epsilon$  - 400:1

\*All linear dimensions are in centimeters

<u>F(N)</u>	<u>D*</u>	<u>L</u>	<u>A</u>	<u>B</u>	<u>C</u>	<u>Mass (kg)</u>
445	43	91	30	46	20	11
2225	51	107	36	53	25	36
4450	58	122	41	61	30	66

It was assumed that for a particular thrust level engines were the same size for all three propellant combinations.



(a) Inverted Oxidizer Tank Dome

(b) Inverted Hydrogen Tank Dome

FIGURE II-10 COMMON BULKHEAD TANK ARRANGEMENTS  
(All Dimensions in Meters)

## 5) Tank Pressurization

The pressurization system assumed was a constant mass system, most probably an autogenous system using propellant to repressurize during a burn. Due to long coast time and slow drainage rates only a small system would be required.

### E. TANK SHELL JUSTIFICATION

During the Space Tug studies conducted in 1973 by McDonnell Douglas and General Dynamics on cryogenic ( $\text{LO}_2/\text{LH}_2$ ) stage configurations (Ref. 3, 4) both contractors selected structural tankage arrangements that were suspended from the body structure. This makes the tanks non-load carrying during Shuttle boost. This is the maximum load condition (3.2 g's) independent of vehicle thrust. The suspended tank arrangement provides a number of advantages over integral load carrying structural arrangements for cryogenic propellents. The suspended tanks decouple intertank and body structure thermal stresses. The body structure or outer shell provides a mounting location for avionics, decoupling the warm electronics from the cold tanks and also providing meteoroid protection. Another advantage is the application of the helium purged, tank-mounted MLI system. The suspended tanks reduce the tank interface and sealing problems on the purge bag. For these reasons the suspended tank configuration was selected as the baseline for parametric study of the cryogenic propellant candidates.

### F. PROPELLANT INVENTORY

The elements of a typical propellant inventory are listed below:

#### 1) $\Delta V$ or Usable

Calculated from the ideal velocity equation using the velocity change and  $I_{sp}$  given in Table II-1.

## 2) Performance Reserve

Two percent of usable propellant, needed to cover possible mixture ratio and Isp variations during burns. This was based on previous Centaur experience.

## 3) Start/Shutdown Losses -

Propellant	Loss per Burn, kg
LO <sub>2</sub>	1.1
LH <sub>2</sub>	0.5
LCH <sub>4</sub>	1.1
KP-1	0.9

These propellants are included to account for chilldown at ignition and engine tailoff losses, they are representative values for the engine configurations under study.

## 4) Boiloff

Boiloff was calculated in PROP by assuming that all the heat leaking into the tank through the insulation and the support struts resulted in propellant evaporation. Calculations of the thermal energy passing through the insulation was performed for two different environments, ground hold and on-orbit, since these two environments result in different values for thermal conductivities of the insulation. For the helium purged MLI the heat input during ascent decreases from a high value on the ground to a low value in orbit. To accommodate this change in heat input the ascent heating was considered to be given by an equivalent ascent time at the ground-hold heat rate. This equivalent ascent time is totaled with the actual ground-hold time before launch and this time period is used for the length of time the ground-hold heating rate is in effect. A one-dimensional model was used to determine the heat conduction rate with the tank wall assumed to be at the



temperature of the propellant and the temperature of the outer layer determined by the analysis discussed in section G-4. Penetrating strut heat leaks are explained in section G-5. The total boiloff was then determined by the sum of both heat leaks and the latent heat of vaporization of the propellant.

5) Line Trapped

Propellant Combination	FEEDLINE TRAPPED-PER BURN, kg		
	Thrust, N		
	445	2225	4450
LO <sub>2</sub> /LH <sub>2</sub>	0.14/0.01	0.54/0.01	0.86/0.03
LO <sub>2</sub> /LCH <sub>4</sub>	0.14/0.03	0.42/0.08	0.69/0.16
LO <sub>2</sub> /RP-1	0.14/0.05	0.54/0.15	0.86/0.30
LO <sub>2</sub> /LCH <sub>4</sub> (Parallel tanks)	0.18/0.07	0.73/0.12	1.0/0.20

The amounts shown in the above table represent the propellant that is remaining in the feed line at the end of each burn and consequently boils off during coast. This lost propellant is calculated by first sizing the feed lines and then determining the length of the line exposed. A maximum pressure drop of 7 kPa (1 psid) for the feed lines was selected. Shutoff valves are located at the engine manifold and at the tank outlet, these are used to isolate the exposed portion of the feed line from the propellant. This trapped propellant would then be allowed to escape through a zero-thrust vent to prevent line rupture.

The feed line arrangement for the tandem/toridal configuration is shown in the layout of a LO<sub>2</sub>/LH<sub>2</sub> system, in Figure II-11a. The line feeding propellant from the toroid is partially enclosed inside the tank, this part was assumed to stay filled with propellant during coast. Boiloff of propellant only occurs in that portion of the 3 m of feed line outside the tank. All of the 1.5 m of feed line from the ellipsoidal tank is considered exposed. In computing the pressure drops for a particular flow rate, the effect of valves, elbows, and changes in line size were considered.

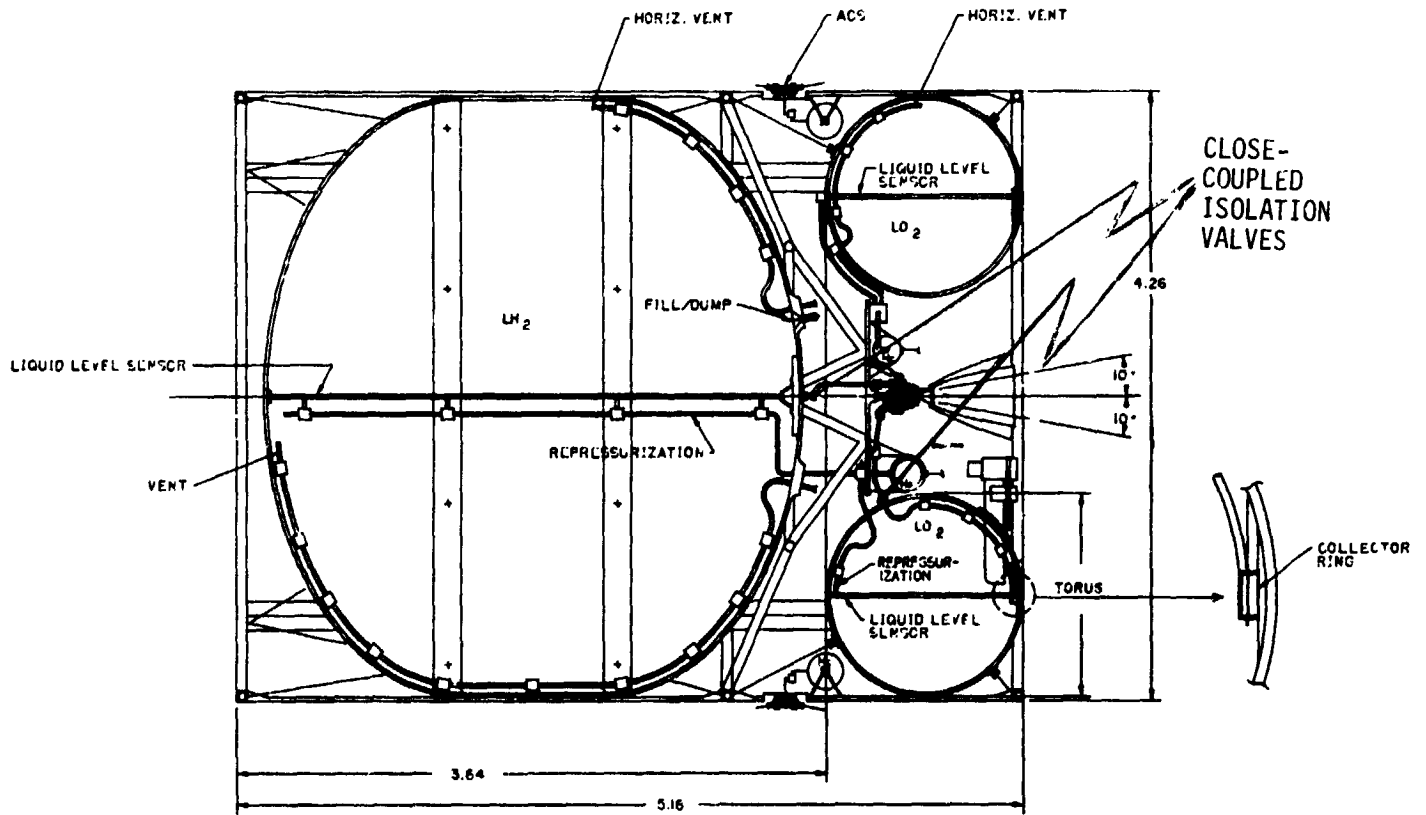


FIGURE II-11a FEEDLINE ARRANGEMENT FOR A TANDEM/TOROIDAL TANK CONFIGURATION (All dimensions in meters)

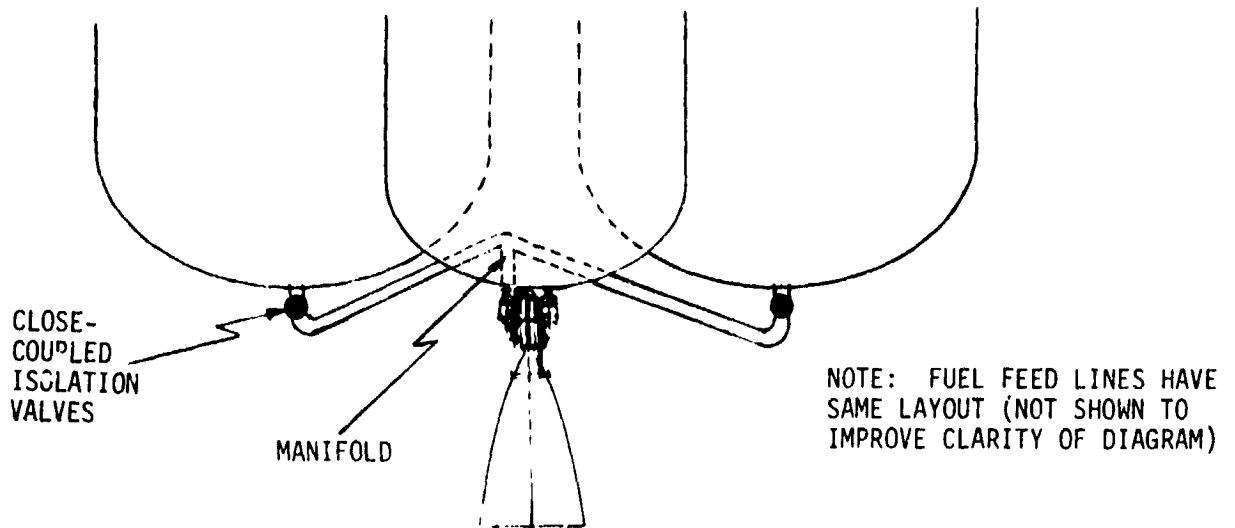


FIGURE II-11b FEEDLINE ARRANGEMENT FOR PARALLEL TANKS CONFIGURATION

The arrangement of lines for the parallel tanks is shown in Figure II-11b, each line from the tank to the engine manifold is 1.8 m long. The individual lines are allowed a 7 kPa (1 psi) pressure drop and again valves, bends, and diameter changes were considered.

The minimum line diameters calculated are shown in the table below. The  $\text{LO}_2/\text{LCH}_4$  tandem/toroid had the  $\text{LCH}_4$  in the toroid while the other two propellant combinations were designed with the  $\text{LO}_2$  in the toroid.

Propellant Combination	LINE DIAMETERS, cm		
	Thrust, N		
	445	2225	4450
$\text{LO}_2/\text{LH}_2$	1.0/0.8	2.0/1.3	2.4/1.8
$\text{LO}_2/\text{LCH}_4$	1.0/0.8	1.8/1.5	2.3/1.8
$\text{LO}_2/\text{RP-1}$	1.0/0.8	2.0/1.3	2.4/1.8
$\text{LO}_2/\text{LCH}_4$ (Parallel tanks)	0.8/0.8	1.5/1.0	1.8/1.3

6) Expulsion Efficiency - 98%

Estimate of the propellant that is drained from the tank. An accurate figure for propellant residuals was calculated for each propellant management technique and incorporated in the propellant inventory in the next section.

7) Loading Accuracy - 0.5%

This percentage of the total amount of propellant must be allowed due to limitations on accuracy of loading equipment and instrumentation and is representative of values achieved on previous programs.

G. THERMAL INSULATION STUDIES

1) Insulation Properties

a) Multilayer Insulation (MLI)

The multilayer insulation is composed of radiation shields of 0.006 mm (1/4 mil) double-aluminized Mylar separated with Dacron or silk net spacers (2 spacers per reflector) as shown in Figure II-12. The insulation has about 24 radiation shields per cm of thickness. All air will be purged from the insulation with helium prior to propellant loading and the purge will continue until shortly before lift-off. During ascent helium will outgas with a resulting decrease in conductivity as shown in Figure II-13. Because helium is trapped at atmospheric pressure on the ground, MLI conductivity before lift-off is essentially that of helium. To save weight the vehicle shell can be used as part of the "purge bag"; this arrangement is shown in Figure II-14.

Multilayer insulation results in a relatively light system with poor ground thermal conductivity but excellent on-orbit thermal conductivity. Thus, longer duration missions (i.e., multiple burn options which minimize  $\Delta V$  but require longer transit times) stand to benefit the most from a multilayer system. The actual insulation system mass is a function of the required insulation thickness and average density. The optimum thickness was determined by a trade-off between boiloff/vent losses and insulation mass.

b) Spray-On-Foam Insulation (CPR-488)

CPR-488 is a sprayable foam insulation utilized in low heating and shear applications as compared to ablator usage. Maximum design limits for CPR-488 are shown in the table below:

CPR-488 Maximum Design Limits

Parameter	Maximum Limits
Bondline Temperature	150°C
Maximum Heating Rate	113,000 W/m <sup>2</sup>
Maximum Shear	96 N/m <sup>2</sup>

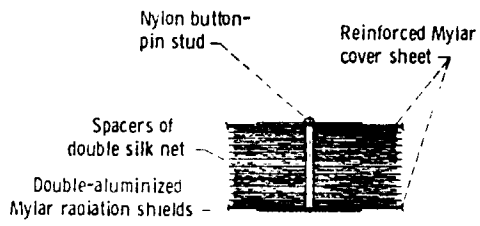


FIGURE II-12 TYPICAL SECTION OF MLI BLANKET (REF.5)

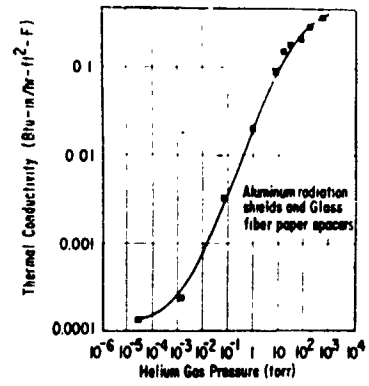


FIGURE II-13 EFFECTS OF HELIUM GAS PRESSURE ON THERMAL CONDUCTIVITY (REF.5)

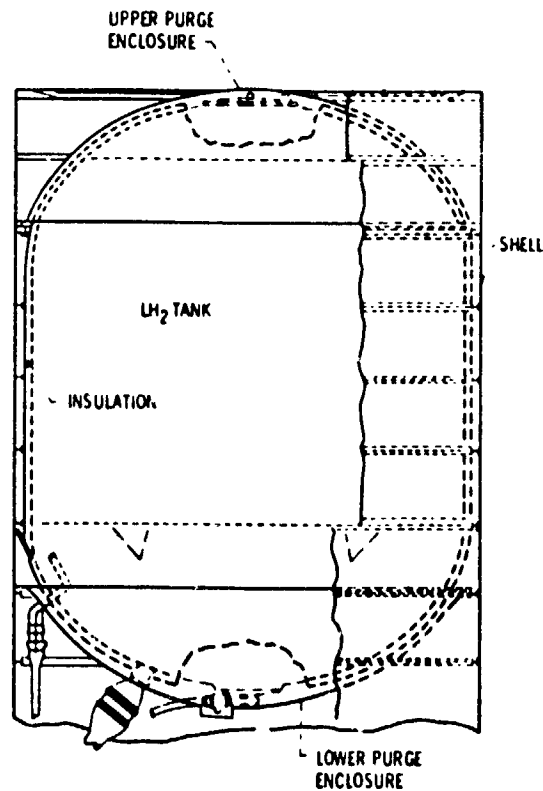


FIGURE II-14 HELIUM PURGE ENCLOSURE CONCEPT FOR SPACE TUG LIQUID HYDROGEN TANK (REF.5)

These design limits could require the use of an "undercoating" of another insulation in areas of high heating and/or shear stresses. The characteristics for both insulations appear in Table II-4.

## 2) Insulation Optimization Studies

Optimization of the insulation systems could be achieved by a repetitive use of the computer program PROP to analyze each propulsion system over a range of insulation thickness. However, because of the large number of cases involved in the initial screening process, a simpler and quicker method is required. For this reason analytical models were developed to predict insulation thicknesses that would minimize the LTPS length or mass. Each of the models involved some simplifying assumptions, consequently to establish the validity of the models, some of the optimum insulation thicknesses predicted by the models were compared with results from the computer program PROP. The models are described in the following subsections and the details of their derivations are contained in Appendixes C, D, and E.

### a) Length Optimized System

The propellant systems in the first phase of the program were to be of minimum length, therefore it was required to derive a length-optimization analytical model. Minimizing the system length is accomplished by optimizing the total volume with respect to insulation thickness for a constant outside diameter. Tank dimensions and propellant system masses for a typical  $\text{LO}_2/\text{LH}_2$  LTPS, as predicted PROP, are plotted as a function of insulation thickness in Figures II-15 and II-16 (in these runs the outside diameter of the tank plus insulation is maintained at a constant 4.32 m (170 in), and the tank diameter varies with insulation thickness). Optimum insulation thicknesses predicted to give minimum length tanks using the model derived in Appendix C are also shown on Figures II-15 and II-16. It can be seen that the predicted optimum insulation thickness values based on the analytical models are close to the optimum values that result from use of the computer program PROP.

TABLE II-4 BASELINE INSULATION CHARACTERISTICS

Parameter	Type	Multilayer Insulation (MLI)		Spray-on-Foam Insulation (CPR-488)
		Ground	On-Orbit	
Conductivity W/m-°K (BTU/hr-ft-OR)		0.606 (0.35)	4.6355T <sup>0.6</sup> X 10 <sup>-6</sup> (1.8824T <sup>0.6</sup> X10 <sup>-6</sup> )	(2.94 + 0.07639T) X 10 <sup>-3</sup> ((1.70 + 0.02452T) X 10 <sup>-3</sup> )
Density, kg/m <sup>3</sup> (lb <sub>m</sub> /ft <sup>3</sup> )		56.3* (3.51)	56.3* (3.51)	35.3 <sup>†</sup> (2.2)

\*Does not include protective cover sheet, fastening material, or purge system.  
 †Values at 289°K (520°R)

SOFI Data was from; MMC Dwg. No. 82600200102 "Thermal Data Book, External Tank Project". October 1979. Michoud Operations, Martin Marietta Corp., Denver Division, Denver, Colo 80201

Data for MLI was from; MCR-79-594 "Cryogenic Fluid Management Experiment, Thermal Analysis Report". June 1979. Martin Marietta Corp., Denver Division, Denver, Colo 80201

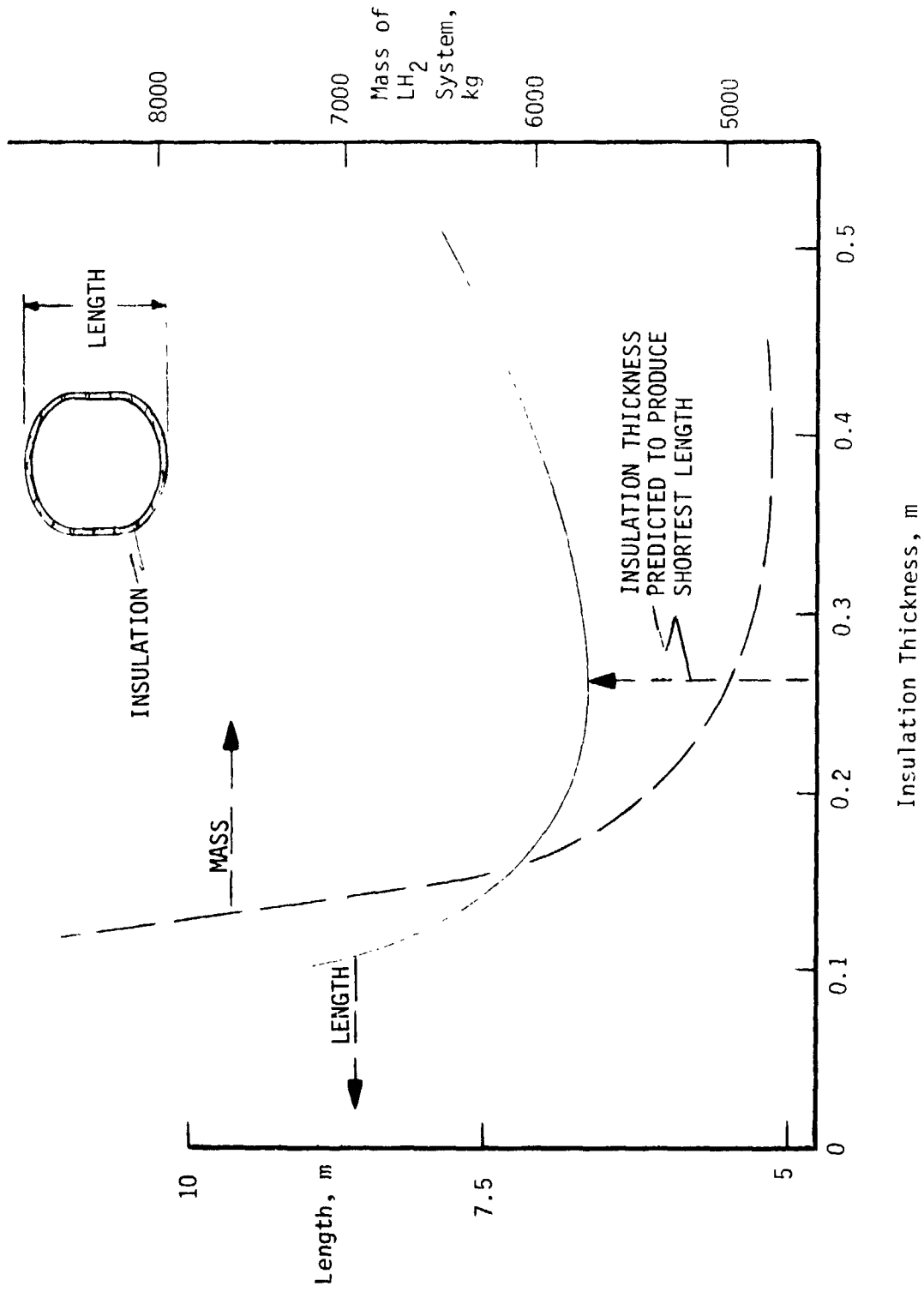


FIGURE II-15 LENGTH AND MASS FOR A SOFI COVERED LH<sub>2</sub> TANK AS A FUNCTION OF INSULATION THICKNESS



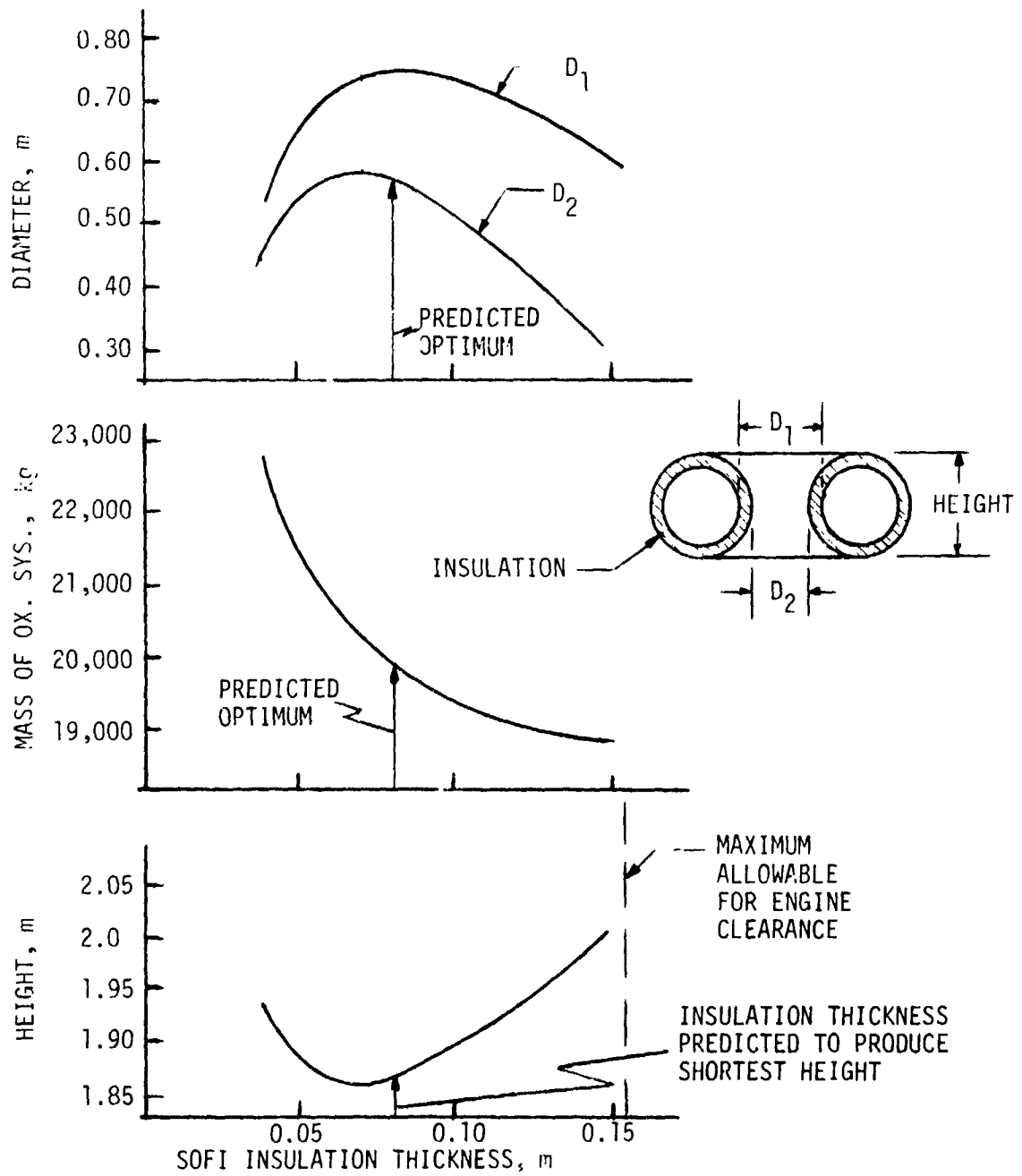


FIGURE II-16 SOFI COVERED TOROIDAL TANK CHARACTERISTICS AS A FUNCTION OF INSULATION THICKNESS

In Figure II-15 the two plots show length vs SOFI thickness (solid line) and mass vs thickness (broken line) for a cylindrical/ellipsoidal domed tank containing LH<sub>2</sub>. The SOFI thickness that produces the lightest propellant system is 0.43 m and a minimum length tank results at a SOFI thickness of 0.26 m. Decreasing the insulation thickness from 0.43 m to 0.26 m results in a decrease of 0.51 m in length and an increase in mass of about 200 kg for the LH<sub>2</sub> propellant system. This means that for the LH<sub>2</sub> tank a substantial reduction in length is accomplished without a large increase in mass. From Figure II-15 it can be seen that the insulation thickness predicted by Appendix C to minimize length would actually produce the shortest system.

The equation derived in Appendix C was also checked with toroidal tanks, this was done because of the different geometry of these tanks. The SOFI thickness predicted by the analytical model to minimize tank height is shown on Figure II-16 together with a plot of the results from several runs of the computer program PROP. The optimum insulation thickness, based on the analytical model produces a tank height only 0.8 percent taller than the actual optimum, based on the computer program results, but does produce a slightly lighter propellant system.

Consequently, from the results presented in Figures II-15 and II-16, all optimum SOFI thicknesses were selected using this tank length optimizing model.

b) Mass Optimized Insulation Thickness - Cylindrical/  
Ellipsoidal Domed Tanks

Optimum MLI thickness determined by the length optimization model produced propellant tanks that were only about 2 cm shorter than the corresponding minimum mass propellant systems but were over 100 kg heavier, thus mass optimization was used to find optimum MLI thicknesses.

This analytical model was designed to predict the insulation thickness that produces the lowest combined mass for the propellant, insulation, and tank at liftoff. The derivation of the equation used to predict thickness is presented in Appendix D. The curves plotted in Figure II-17 are from PROP outputs for a typical LTPS with ellipsoidal tanks. The predicted optimum insulation thicknesses based on the analytical model are marked on this figure for comparison. The LH<sub>2</sub> tank diameter was 4.27 m and the LO<sub>2</sub> tank diameter was 3.47 m. For the optimum MLI thickness predicted by the equation from Appendix D, the propellant system is 2 kg heavier than the optimum shown by PROP results but this is only 0.01 percent of the total LTPS mass and thus does not influence the comparative results. Consequently, MLI thicknesses predicted by the equation derived in Appendix D were used for all ellipsoidal shaped tanks.

c) Mass Optimized Insulation Thickness - Toroidal Tank

Due to the difference in the toroidal tank geometry, a separate insulation optimization analysis was performed and is described in Appendix E. The derivation followed the same initial approach presented in Appendix D; but the volume was initially maintained constant and a 5 percent boiloff was assumed. The optimum insulation thickness determined by the analytical model established the actual boiloff and the corresponding tank volume required. This new tank volume was then used to recalculate an improved value for optimum insulation thickness. The recalculated value of the optimum insulation thickness differed by a maximum of one percent from the original prediction for the cases tested. Since this corresponded to less than one layer of MLI, the initial prediction for optimum thickness was accepted.

A comparison between this predicted optimum insulation thickness based on the analytical model and the corresponding results from PROP are shown in Figure II-18. The predicted optimum insulation thickness produces a system 1.5 kg heavier than the lightest propellant system established by PROP, which amounts to 0.02 percent of the total system mass. Thus the equation developed in Appendix E was used to find the optimum insulation thickness for all MLI covered toroidal tanks.

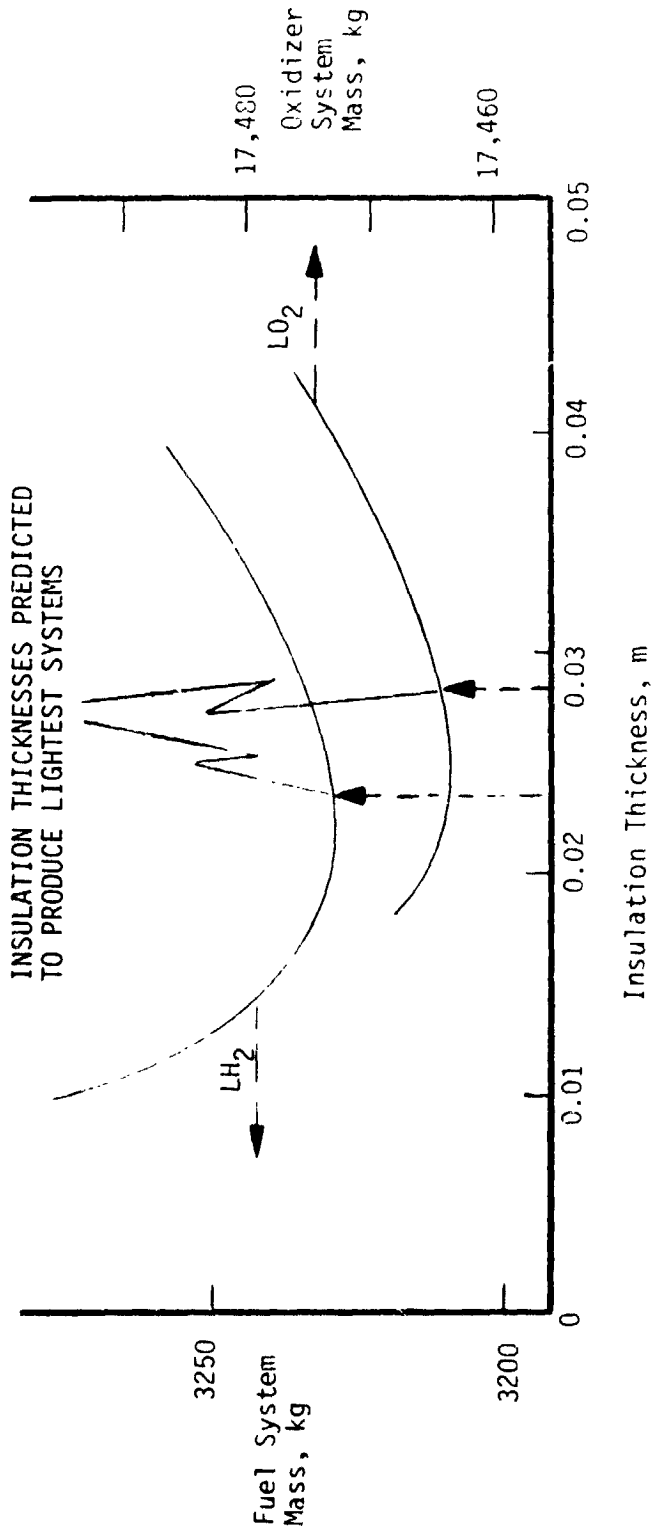


FIGURE II-17 EFFECT OF MLI THICKNESS ON SYSTEM MASS - LO<sub>2</sub>/LH<sub>2</sub>, 445 N THRUST, 4 PERIGEE BURNS.

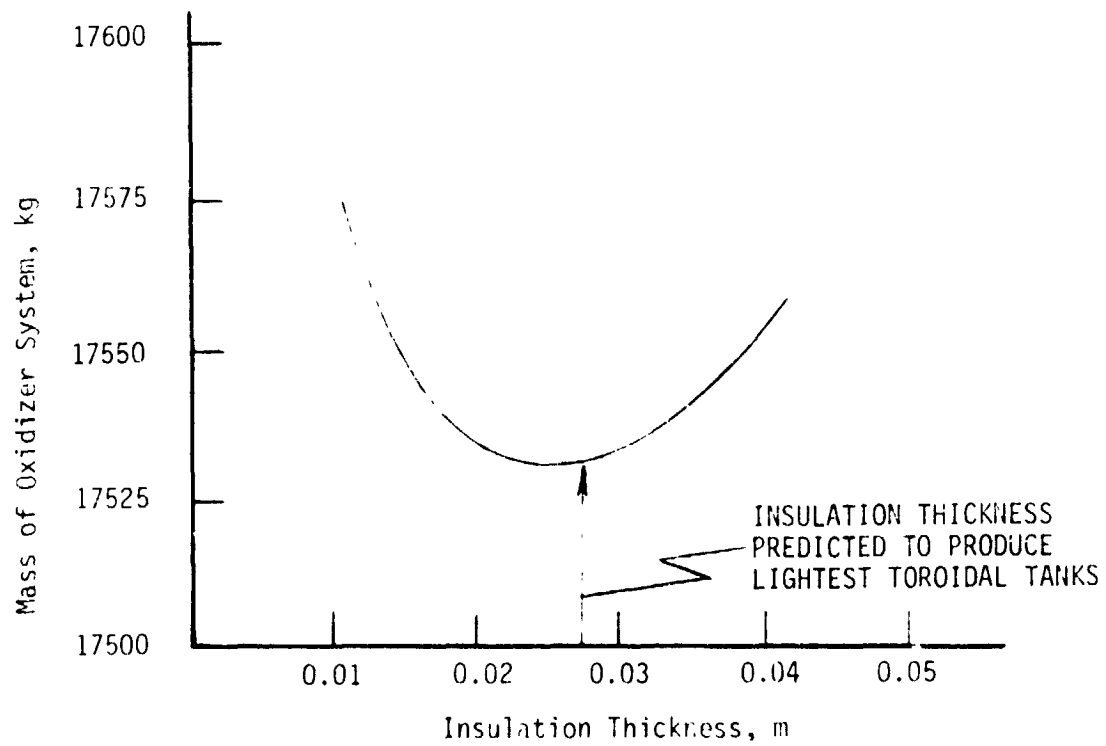


FIGURE II-18 TOROIDAL TANK OXIDIZER SYSTEM MASS VERSUS MLI INSULATION THICKNESS

### 3) External Shell Temperature

Boiloff is proportional to the heat flux into the propellant system; therefore, an estimate of the external skin temperature is required to calculate the losses. By considering average environmental temperatures associated with the baseline orbits a temperature of approximately 294°K (530°R) is predicted.

### 4) Insulation Outer Layer Temperature

The insulation outer layer temperature can be computed for steady state conditions by assuming the outside shell is an isothermal body at 294°K (530°R), and the tank wall is at the temperature of the liquid propellant (see Figure II-19). Both MLI and SOFI systems were considered to have an outer layer of aluminized Mylar for radiation reflection since at 294°K the shell would be radiating in far-infrared range ( $\lambda_{max} = 10\mu$ ) and the SOFI would have an absorbtivity of about 0.9.

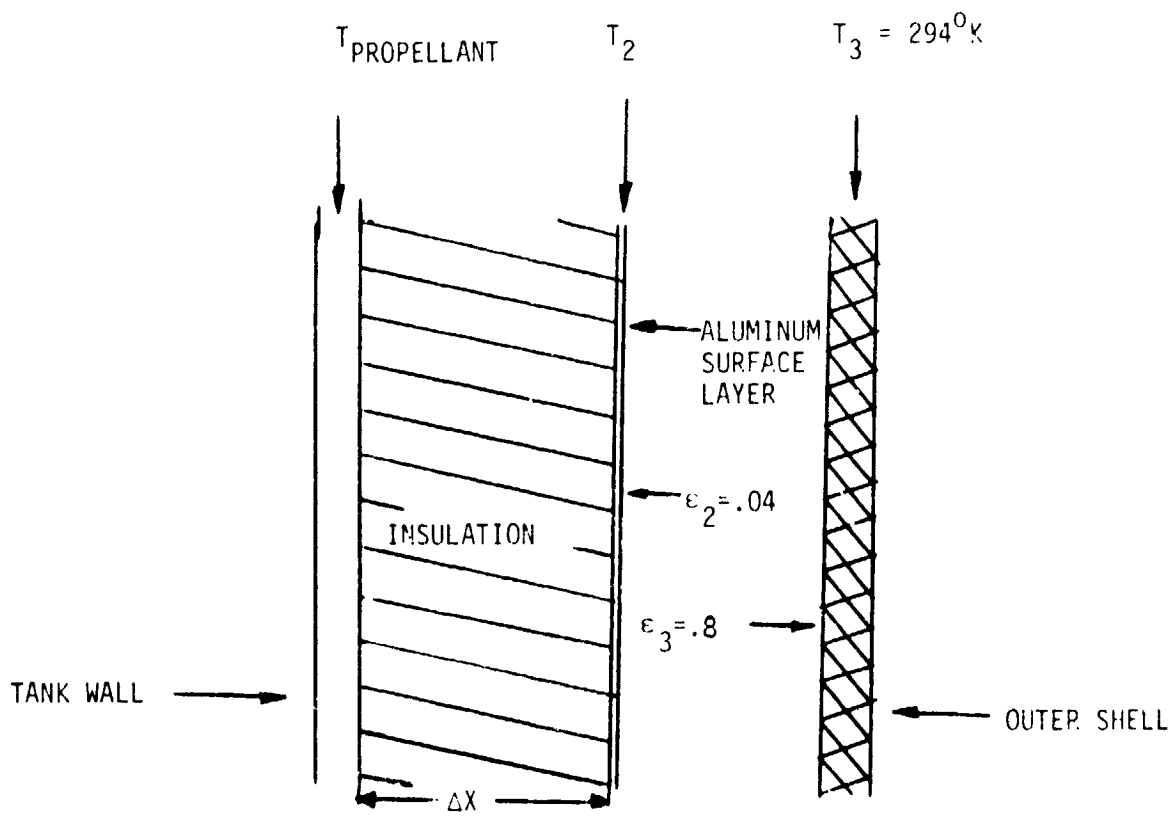
Under steady-state conditions, the radiation rate from the shell to the insulation outer surface must equal the insulation heat transfer rate to the tank wall,

or

$$\left(\frac{q}{A}\right)_{\text{conduction thru insulation}} = \left(\frac{q}{A}\right)_{\text{radiation}}$$

and

$$k \left( \frac{T_2 - T_{\text{PROP}}}{\Delta x} \right) = \sigma \frac{(T_3^4 - T_2^4)}{\frac{1}{\epsilon_2} + \frac{1}{\epsilon_3} - 1}$$



FOR STEADY STATE CONDITIONS

$$\left(\frac{\dot{q}}{A}\right)_{\text{CONDUCTION THRU INSULATION}} = \left(\frac{\dot{q}}{A}\right)_{\text{RADIATION FROM SHELL}}$$

$$\text{OR } k \left( \frac{T_2 - T_{\text{PROP}}}{\Delta X} \right) = \frac{\sigma (T_3^4 - T_2^4)}{\frac{1}{\epsilon_2} + \frac{1}{\epsilon_3} - 1}$$

FIGURE II-19 STEADY STATE HEAT TRANSFER ARRANGEMENT FOR THE LTPS

From the second equation the outer layer temperature can be calculated for a particular insulation and propellant temperature. For MLI systems the difference between shell and insulation surface temperature is  $2.8^{\circ}\text{K}$  ( $5^{\circ}\text{R}$ ) and with SOFI the difference is about  $128^{\circ}\text{K}$  ( $230^{\circ}\text{R}$ ), using a  $294^{\circ}\text{K}$  shell temperature.

#### 5) Penetrating Strut Heat Leak

The struts providing support for the tanks from the outside shell are direct heat leaks to the tank shell. An estimate of the thermal energy entering the propellant was needed to determine boiloff. The heat input rate per unit area was calculated assuming hollow graphite/epoxy struts 0.30 m long, with a thermal conductivity (K) of  $40 \text{ W/m}^{\circ}\text{K}$ . The total cross sectional area of the struts is assumed to be  $0.0005 \text{ m}^2$ , which is representative of tank support approaches utilized in Tug Studies (Ref. 3).

For the  $\text{LH}_2$  Tanks

$$\frac{\dot{q}}{A} = K \cdot \frac{\Delta T}{\Delta X} = \frac{(40 \text{ W/m}^{\circ}\text{K})(294^{\circ}\text{K} - 24^{\circ}\text{K})}{(0.30\text{m})} = 36,000 \text{ W/m}^2$$

(11,400 Btu/hr-ft<sup>2</sup>)

and for the  $\text{LO}_2$  tanks

$$\frac{\dot{q}}{A} = \frac{(40)(294 - 96)}{(0.30)} = 26,400 \text{ W/m}^2 \text{ (8,400 Btu/hr-ft}^2\text{)}$$

finally for the  $\text{LCH}_4$  tanks

$$\frac{\dot{q}}{A} = \frac{(40)(294 - 119)}{(0.30)} = 23,300 \text{ W/m}^2 \text{ (7,400 Btu/hr-ft}^2\text{)}$$



## H. ITEMIZED PROPELLANT INVENTORY

An itemized propellant inventory appears in Table II-5 for the  $LO_2/LH_2$ , 2223 N thrust, 8 burn, MLI cylindrical/toroidal tank configuration. The boiloff losses are divided into those attributed to the heat leaks through the insulation and through the tank-support penetrating struts. Also shown are losses due to start/shutdown transients. The propellant total masses included residuals which do not vary with each burn.

At the beginning of the first burn the vehicle mass is below 27,200 kg due to boiloff during ground hold and ascent plus a 40 hour erection time. Times between burn initiations are taken from Task III of PP/LSSI Study (Contract NAS3-21955). Propellant mass required per burn is calculated using the ideal velocity equation. Boiloff is calculated by times between burn initiations rather than an equal time split. Shown below is the boiloff broken down into ground and ascent boiloff and losses due to on-orbit erection time.

MODE	PROPELLANT	BOILOFF, kg	
		INSULATION HEAT LEAK	STRUT HEAT LEAK
ON GROUND AND DURING ASCENT	LH <sub>2</sub>	46	0.15
	LO <sub>2</sub>	41	0.2
40 HR ON-ORBIT ERECTION TIME	LH <sub>2</sub>	13	40
	LO <sub>2</sub>	10	53

The results predict that more boiloff is associated with the strut heat leak than with the on-orbit insulation heat leak.

## I. BASELINE TANK DIAMETER

For the preliminary tank screening a tank diameter of 4.27 m (14 ft) was assumed. The sketch in Figure II-20 depicts the reasoning for this choice of diameter. Starting with the maximum cargo bay diameter of 4.57 m (15 ft) an allowable stage diameter of 4.42 m (14.5 ft) was determined using inputs from Martin Marietta's Payload Integration Contract (F04701-77-7-C-0183). The external skin arrangement, constructed of graphite epoxy composite material, was determined from Space Tug Study results (Ref. 3, 4). The 3.5 cm MLI thickness resulted from the insulation studies previously discussed. By

TABLE II-5 ITEMIZED PROPELLANT INVENTORY FOR LO<sub>2</sub>/LH<sub>2</sub>, 2225 N, THRUST, 8 BURNS, MLI

TOTAL  $\Delta V = 4448$  m/s ( 14,594 ft/s ); EACH PERIGEE BURN  $\Delta V = 327$  m/s ( 1,074 ft/s );  
 CIRCULARIZATION BURN  $\Delta V = 1829$  m/s ( 6,000 ft/s )

BURN NO.	PROPELLANT MASS REQUIRED PER BURN, kg	APPROXIMATE TIME BETWEEN BURN INITIATION, sec	BOILOFF BETWEEN BURN INITIATIONS, kg				START/SHUTDOWN LOSSES, kg		MASS OF TOTAL FUEL REMAINING, kg	MASS TOTAL OXIDIZER REMAINING, kg	TOTAL VEHICLE MASS, kg
			DUE TO INSULATION HEAT LEAK		DUE TO PENETRATING STRUT LEAK		F	O			
			FUEL	OXIDIZER	FUEL	OXIDIZER					
1	1972	6024	58.4 <sup>†</sup>	50.5 <sup>†</sup>	40.3 <sup>†</sup>	53.9 <sup>†</sup>	0	0	2650	15675	27013
2	1828	6990	0.52	0.41	1.84	2.45	0.9	2.3	2365	13979	25032
3	1694	8255	0.61	0.47	2.13	2.85	0.9	2.3	2100	12407	23195
4	1569	9972	0.72	0.61	2.52	3.37	0.9	2.3	1854	10949	21491
5	1454	12424	0.87	0.73	3.04	4.06	0.9	2.3	1625	9597	19909
6	1346	16198	1.08	0.92	3.79	5.08	0.9	2.3	1412	8345	18441
7	1247	22426	1.41	1.19	4.94	6.62	0.9	2.3	1212	7183	17077
8	1154	22279	2.00	1.66	6.85	9.16	0.9	2.3	1024	6101	15807
9	5051	9799*	1.95	1.64	6.80	9.07	0.9	2.3	849	5099	14630
END OF MISSION			0.86	0.66	2.73	3.65	0.9	2.3	123	763	9569

† BOILOFF BETWEEN LIFTOFF AND FIRST BURN

\* TIME FOR CIRCULARIZATION BURN

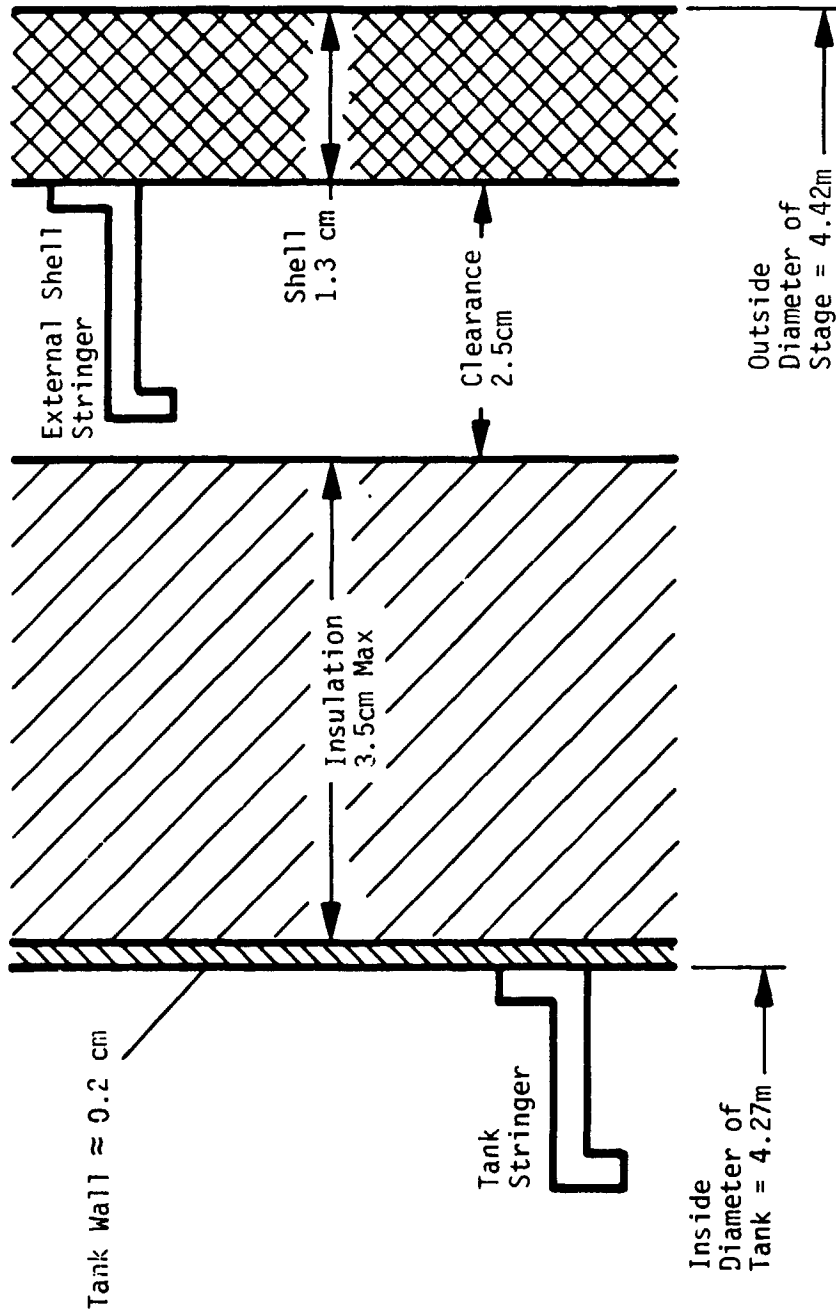


FIGURE II- 20 BASELINE TANK DIAMETER (MLI)

considering a typical tank wall thickness of 0.2 cm, an inside diameter of 4.27 m is derived for tank sizing. For the SOFI-covered tanks the outside diameter of the insulation is constrained to 4.32 m (14.2 ft), and the inside diameter of the tank will vary depending on the insulation thickness.

J. NON-TANK SYSTEM HARDWARE MASSES

To predict a value for usable payload mass requires an estimation of the mass of auxiliary systems required by the LTPS such as attitude control propulsion system (ACPS), external shell, purge system and avionics.

The overall stage mass will include the following constant masses:

<u>Mass (kg)</u>	<u>Components</u>	<u>Reference</u>
460	Structures (external shell, Shuttle I/F equipment, equipment mounting, etc).	IUS and TUG Studies
340	Avionics (data management devices, computer, fuel cell & communications)	Component masses & Tug Studies.
200	ACPS Components	Tug Studies
180	ACS Propellant	Estimate.
40	Purge System for LO <sub>2</sub> /RP-1 with MLI	Estimate.
70	Purge System for all other MLI Systems	Estimate.
0	SOFI System (no purge needed)	
45	Engine mounts and supports	Tug studies.
25	Components and lines	Tug studies.
90	Pressurant system mass	Estimate.
1380	LO <sub>2</sub> /RP-1 with MLI	
1410	All other MLI systems	
1340	SOFI systems	

In addition, the mass of the engines, as a function of thrust level supplied by NASA-LeRC, were:

<u>Thrust, N</u>	<u>Mass, kg</u>
445	11
2224	36
4448	66

#### K. INITIAL SYSTEM CHARACTERISTICS

The principal result of this first portion of the report is the selection of 26 propellant system configurations for further evaluation in Section III. Seventy-two candidates consisting of all thrust levels, burn strategies and insulation concepts were considered for the initial sizing using PROP. Fifty-four of the systems were arranged in the tandem/toroidal configuration employing all three propellant combinations and 18 were arranged in parallel tanks filled with  $LO_2$  and  $LCH_4$ .

The computer program PROP was described in Section II-B of this report. Sample PROP inputs and outputs are shown in Appendix A for the different types of configurations evaluated. Inputs for each concept were determined from data supplied by NASA-LeRC and information from the analyses described in the previous sections. System characteristics, calculated from PROP, for the 72 cases are shown in Tables II-6 through II-13 for the three propellant combinations, two insulation concepts, and two tank arrangements. The first five columns in the tables specify the configuration, and the rest are outputs from PROP. The rows labeled "F" are the fuel data, and those labeled "O" are oxidizer data.

The definitions of these columns have been previously discussed, except for overall length. For the tandem/toroidal tank configurations the length of the ellipsoidal tank (cylindrical with ellipsoidal domes for  $LH_2$ ) plus twice its insulation thickness is added to either the toroidal tank height plus twice its insulation thickness (Figure II-21a) or the engine length plus 0.15 m (6 in) for clearance purposes if the toroidal tank is not large enough in diameter to completely embed the engine as in Figure II-21b. The parallel tank configuration overall length is computed by adding the engine length, twice the insulation thickness, and the length of the tank.

TABLE II-6

## PROPELLANT SYSTEM CHARACTERISTICS

PROPELLANT COMBINATION:  $LQ_2/LH_2$  INSULATION CONCEPT: MLI  
 MIXTURE RATIO: 6:1 INITIAL VEHICLE MASS: 27216 kg

Thrust, N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total $\Delta V$ , m/sec	LEO to GEO Hr	PROPELLANT MASSES, (kg)				TANK			System Dry Mass, kg	System Wet Mass, kg	Payload Mass kg	Overall Length, m
					Usable ( $\Delta V$ )	Trapped, Start-S/D Losses	Boiloff Losses	Total	Volume, m <sup>3</sup>	Diameter, m	Length, m				
445 (100)	1		5537.1	59.2	F 2866 0 17197	116 713	183 196	21272	47.9 16.8	4.27 4.27	4.36 1.59	1783	23055	4161	6.07
	4	4143 (422.5)	5271.5	61.4	F 2799 0 16791	116 704	180 199	20789	46.9 16.4	4.27 4.27	4.28 1.57	1730	22569	4647	5.97
	8		4983.4	72.4	F 2720 0 16321	117 694	191 214	20257	45.9 16.0	4.27 4.27	4.21 1.54	1778	22035	5181	5.88
2224 (500)	1		5289.0	16.9	F 2747 0 16480	112 684	128 135	20286	45.2 16.0	4.27 4.27	4.17 1.54	1775	22061	5155	5.82
	4	4315 (440.0)	4855.8	19.8	F 2626 0 15757	110 562	129 139	19423	43.4 15.3	4.27 4.27	4.04 1.50	1766	21189	6027	5.65
	8		4448.0	31.8	F 2501 0 15007	108 641	141 155	18552	41.7 14.6	4.27 4.27	3.92 1.45	1761	20314	6901	5.48
4448 (1000)	1		5148.8	11.7	F 2680 0 16082	109 668	121 127	19789	44.1 15.6	4.27 4.27	4.09 1.52	1796	21585	5631	5.72
	4	4403 (449.0)	4743.3	14.9	F 2564 0 15384	107 647	123 131	18956	42.3 15.0	4.27 4.27	3.97 1.47	1787	20743	6473	5.55
	8		4413.4	27.1	F 2461 0 14766	107 632	135 148	18247	40.9 14.4	4.27 4.27	3.87 1.44	1785	20032	7183	5.42

\* Includes Insulation Thicknesses

TABLE II-7 PROPELLANT SYSTEM CHARACTERISTICS															
INSULATION CONCEPT: SOFI															
PROPELLANT COMBINATION: LO <sub>2</sub> /LH <sub>2</sub>															
INITIAL VEHICLE MASS: 27216 kg															
Thrust, N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LEO to GEO Hr	PROPELLANT MASSES, (kg)					TANK			Payload Mass kg	Overall Length,* m	
					Usable (ΔV)	Trapped, Start-S/D	Botloff Losses	Total	Volume, m <sup>3</sup>	Diameter, m	Length, m	Propellant System Dry Mass, kg			System Wet Mass, kg
445 (100)	1		5537.1	59.2	F 2866 O 17197	116 713	1168 2550	24609	62.9 18.9	3.80 4.16	6.45 1.81	2492	27102	113	8.94
	4	4143 (422.5)	5271.5	61.4	F 2799 O 16791	116 704	1169 2518	24095	61.9 18.5	3.79 4.16	6.38 1.77	2481	26578	637	8.84
	8		4983.4	72.4	F 2720 O 16321	117 694	1231 2702	23783	61.6 18.3	3.76 4.15	6.44 1.76	2522	26307	908	8.92
2224 (500)	1		5289.0	16.9	F 2747 O 16480	112 684	778 1868	22666	55.1 17.6	3.92 4.20	5.48 1.69	2192	24860	2356	7.69
	4	4315 (440.0)	4855.8	19.8	F 2626 O 15757	110 662	780 1947	21880	53.2 17.0	3.91 4.20	5.35 1.64	2178	24060	3156	7.52
	8		4448.0	31.8	F 2501 O 15007	108 641	850 2057	21162	52.4 16.4	3.87 4.13	5.37 1.61	2227	23391	3825	7.56
4448 (1000)	1		5148.8	11.7	F 2680 O 1602	109 668	714 1761	22014	53.1 17.1	3.94 4.20	5.29 1.65	2171	24137	3029	7.43
	4	4403 (449.0)	4743.3	14.9	F 2564 O 15384	107 647	723 1852	21275	51.4 16.6	3.93 4.20	5.17 1.61	2161	23438	3773	7.29
	8		4413.4	27.1	F 2461 O 14766	107 632	805 1987	20755	51.1 16.1	3.89 4.19	5.22 1.58	2213	22971	4245	7.37

\* Includes Insulation Thicknesses

TABLE II-8

PROPELLANT SYSTEM CHARACTERISTICS

PROPELLANT COMBINATION: LO<sub>2</sub>/LCH<sub>4</sub>      INSULATION CONCEPT: MLI  
 MIXTURE RATIO: 3.7:1      INITIAL VEHICLE MASS: 27216 kg

Thrust, N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LEO to GEO Hr	PROPELLANT MASSES (kg)				TANK			Payload Mass kg	Overall Length, m		
					Usable (ΔV)	Trapped, Start-S/D Losses	Botloff Losses	Total	Volume, m <sup>3</sup>	Diameter, m	Length, m			Propellant System Dry Mass, kg	System Wet Mass, kg
445 (100)	1		5524.9	52.9	F 4700 0 17389	200 709	76 173	23247	12.3 16.9	4.27 3.58	1.29 2.53	1596	24843	2373	3.93
	4	3310 (337.5)	5261.7	55.4	F 4509 0 17055	203 702	77 176	22823	12.0 16.6	4.27 3.55	1.27 2.51	1594	24417	2798	3.90
	8		4976.3	66.7	F 4503 0 16661	208 696	82 191	22342	11.8 16.2	4.27 3.53	1.26 2.49	1594	23936	3279	3.87
2224 (500)	1		5260.4	15.8	F 4505 0 16667	192 680	57 120	22220	11.7 16.2	4.27 3.52	1.26 2.49	1606	23826	3290	3.84
	4	3496 (356.5)	4838.5	18.8	F 4340 0 16056	193 662	58 123	21432	11.3 15.6	4.27 3.48	1.23 2.46	1601	23033	4182	3.79
	8		4441.4	30.9	F 4165 0 15411	195 646	62 138	20616	10.9 15.0	4.27 3.43	1.20 2.43	1603	22218	4997	3.73
4448 (1000)	1		5108.1	11.2	F 4403 0 16293	188 665	54 113	21717	11.5 15.8	4.27 3.50	1.24 2.47	1631	23347	3868	3.90
	4	3574 (364.5)	4709.3	14.4	F 4240 0 15687	189 648	56 116	20935	11.1 15.2	4.27 3.45	1.21 2.44	1627	22561	4654	3.87
	8		4403.8	26.7	F 4101 0 15175	192 636	61 132	20298	10.7 14.8	4.27 3.42	1.19 2.42	1627	21926	5291	3.84

\* Includes Insulation Thicknesses



PROPELLANT SYSTEM CHARACTERISTICS															
INSULATION CONCEPT: SOFI															
INITIAL VEHICLE MASS: 27216 kg															
PROPELLANT COMBINATION: LO <sub>2</sub> /LCH <sub>4</sub>															
MIXTURE RATIO: 3.7:1															
Thrust. N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LTO to GEO Hr	PROPELLANT MASSES (kg)					TANK			Payload Mass kg	Overall Length, m	
					Usable (ΔV)	Trapped, Start-S/D	Losses	Boiloff	Losses	Total	Volume, m <sup>3</sup>	Diameter, m			Length, m
445 (100)	1		5524.9	52.9	F 4700	200	783	25850	14.0	4.17	1.44	1628	27438	0	4.36
	4	3310 (337.5)	5261.7	55.4	F 4609	709	2069	25381	13.8	4.16	1.43	1631	27013	203	4.34
	8		4976.3	66.7	F 4503	702	2037	25035	18.3	3.67	2.60	1638	26674	542	4.34
2224 (500)	1		5260.4	15.8	F 4505	192	598	24112	13.0	4.20	1.37	1599	25713	1503	4.16
	4	3496 (356.5)	4838.5	18.8	F 4340	680	1472	23372	17.4	3.61	2.55	1599	24971	2235	4.11
	8		4441.4	30.9	F 4165	662	1522	22578	16.9	3.57	2.53	1618	24196	3019	4.10
4448 (1000)	1		5108.1	11.2	F 4403	188	569	23502	12.7	4.21	1.34	1618	25112	2104	4.10
	4	3574 (364.5)	4709.3	14.4	F 4240	665	1385	22778	17.0	3.58	2.53	1619	24397	2819	4.06
	8		4403.8	26.7	F 4101	648	1443	22272	16.5	3.54	2.51	1633	23905	3310	4.06

\* Includes Insulation Thicknesses

TABLE II-10

PROPELLANT SYSTEM CHARACTERISTICS (PARALLEL TANKS)

PROPELLANT COMBINATION: LO<sub>2</sub>/LCH<sub>4</sub>  
 MIXTURE RATIO: 3.7:1  
 INSULATION CONCEPT: MLI  
 INITIAL VEHICLE MASS: 27216 kg

Thrust, N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LEO to GEO Hr	PROPELLANT MASSES, (kg)						Total	TANK <sup>+</sup> , EACH			Propellant System Dry Mass, kg	System Wet Mass, kg	Payload Mass, kg	Overall Length, m
					Usable (ΔV)	Trapped, Start-S/D	Losses	Boiloff	Losses	Volume, m <sup>3</sup>		Diameter, m	Length, m					
445 (100)	1		5524.9	52.9	F	4664	221	112	23316	6.20	1.58	3.53	1647	24963	2253	4.50		
	4	3310 (337.5)	5261.7	55.4	0	17257	786	276	22897	8.41	1.86	3.52	1643	24541	2675	4.42		
					F	4575	223	114		6.09	1.58	3.47						
2224 (500)	1		5260.4	15.8	F	4469	212	79	22238	5.90	1.58	3.37	1650	23888	3328	4.52		
	4	3496 (356.5)	4838.5	18.8	0	16537	753	188	21457	8.03	1.86	3.39	1644	23100	4116	4.41		
					F	4306	211	80		5.70	1.58	3.27						
4448 (1000)	1		5108.1	11.2	F	4369	207	74	21729	5.77	1.58	3.30	1673	23402	3814	4.60		
	4	3574 (364.5)	4709.3	14.4	0	16166	736	177	20955	7.84	1.86	3.32	1667	22621	4595	4.49		
					F	4207	206	76		5.57	1.58	3.20						
2224 (500)	1		4403.8	26.7	F	4071	209	86	20338	5.42	1.58	3.12	1665	22003	5213	4.41		
	4	3574 (364.5)	4709.3	14.4	0	15567	716	182	20338	7.56	1.86	3.21	1665	22003	5213	4.41		
					F	4071	209	86		5.42	1.58	3.12						
2224 (500)	1		4441.4	30.9	F	4134	212	90	20663	5.50	1.58	3.17	1640	22303	4413	4.30		
	4	3496 (356.5)	4838.5	18.8	0	15296	713	219	21457	7.45	1.86	3.17	1640	22303	4413	4.30		
					F	4134	212	90		5.50	1.58	3.17						

+DIMENSIONS OF A SINGLE TANK

\*INCLUDES INSULATION THICKNESSES

TABLE II-11

PROPELLANT SYSTEM CHARACTERISTICS (PARALLEL TANKS)

PROPELLANT COMBINATION: LO<sub>2</sub>/LCH<sub>4</sub>  
 MIXTURE RATIO: 3.7:1  
 INSULATION CONCEPT: SOFI  
 INITIAL VEHICLE MASS: 27216 kg

Thrust, N (lb.)	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LEO to GEO hr	PROPELLANT MASSES (kg)				TANK +			Payload Mass, kg	System Wet Mass, kg	System Dry Mass, kg	Overall Length,* m	
					Usable (ΔV)	Trapped, Start-S/D Losses	Boiloff Losses	Total	Volume, m <sup>3</sup>	Diameter, m	Length, m					
445 (100)	1		5524.9	52.9	F 0	4438 16421	21 760	744 2653	25237	6.70	1.56	3.88	26840	1602	376	4.91
	4	3310 (337.5)	5261.7	55.4	F 0	4358 16126	223 754	748 2666	24874	6.61 8.98	1.56 1.83	3.83 3.85	26474	1600	742	4.86
	8		4976.3	66.7	F 0	4272 15806	228 749	791 2802	24645	6.56 8.89	1.56 1.83	3.81 3.82	26251	1606	965	4.83
2224 (500)	1		5260.4	15.8	F 0	4209 15573	212 718	535 1914	23160	6.15 8.36	1.56 1.83	3.59 3.61	24739	1578	2477	4.75
	4	3496 (356.5)	4838.5	18.8	F 0	4068 15053	211 701	534 1919	22487	5.97 8.12	1.56 1.83	3.50 3.52	24062	1575	3154	4.66
	8		4441.4	30.9	F 0	3930 14540	212 688	581 2076	22024	5.86 7.95	1.56 1.83	3.44 3.46	23604	1580	3612	4.61
4448 (1000)	1		5108.1	11.2	F 0	4111 15211	207 701	500 1783	22513	5.98 8.13	1.56 1.83	3.50 3.53	24112	1598	3104	4.82
	4	3574 (364.5)	4709.3	14.4	F 0	3972 14696	206 685	503 1799	21861	5.81 7.89	1.56 1.83	3.41 3.44	23456	1595	3760	4.73
	8		4403.8	26.7	F 0	3866 14306	209 677	554 1985	21595	5.74 7.79	1.56 1.83	3.38 3.40	23198	1602	4018	4.70

\*DIMENSIONS OF A SINGLE TANK

\*INCLUDES INSULATION THICKNESSES

PROPELLANT SYSTEM CHARACTERISTICS																	
INSULATION CONCEPT: MLI																	
INITIAL VEHICLE MASS: 27216 kg																	
Thrust, N (lb <sub>f</sub> )	Number of Burns	Isp, N-sec (sec)	Total ΔV, m/sec	LEO to GEO hr	PROPELLANT MASSES, (kg)					Total	Volume, m <sup>3</sup>	TANK		Propellant System Dry Mass, kg	System Wet Mass, kg	Payload Mass, kg	Overall Length, m*
					Usable (ΔV)	Trapped, Start-S/D Losses	Botloff Losses	Losses	Losses			Volume, m <sup>3</sup>	Diameter, m				
445 (100)	1		5521.6	51.1	F	5619	267	0	23787	7.49	2.73	1.93	1797	25584	1632	3.57	
	4	3114 (317.5)	5259.0	53.7	F	5520	262	0	23365	7.36	2.71	1.92	1814	25179	2037	3.55	
	8		4974.4	65.2	F	5401	264	0	22907	7.21	2.69	1.90	1813	24721	2495	3.52	
2224 (500)	1		5251.2	15.4	F	5408	257	0	22823	7.21	2.69	1.90	1806	24630	2586	3.49	
	4	3270 (333.5)	4832.8	18.5	F	5223	249	0	22055	6.96	2.66	1.88	1802	23857	3359	3.43	
	8		4439.2	30.8	F	5026	247	0	21275	6.71	2.63	1.86	1799	23074	4142	3.37	
4448 (1000)	1		5096.5	15.4	F	5279	251	0	22274	7.04	2.67	1.89	1831	24106	3110	3.45	
	4	3364 (343.5)	4702.7	18.5	F	5095	243	0	21510	6.79	2.64	1.87	1827	23336	3880	3.39	
	8		4401.0	30.8	F	4938	243	0	20900	6.59	2.61	1.85	1825	22724	4492	3.34	

\* Includes Insulation Thicknesses

TABLE II-13

## PROPELLANT SYSTEM CHARACTERISTICS

PROPELLANT COMBINATION: LO<sub>2</sub>/RP-1  
 MIXTURE RATIO: 3:1  
 INSULATION CONCEPT: SOFI  
 INITIAL VEHICLE MASS: 27216 kg

Thrust, N (lb)	Number of Burns	I <sub>sp</sub> (sec)	Total ΔV, m/sec	LEI to GEO Hr	PROPELLANT MASSES, (kg)						Volume, m <sup>3</sup>	TANK		Propellant System Dry Mass, kg	System Wet Mass, kg	Payload Mass, kg	Overall Length,* m
					Usable (ΔV)	Trapped, Start-S/D Losses	Botloff Losses	Total	Diameter, m	Length, m							
445 (100)	1		5521.6	51.1	F 5506 0 16519	267 765	0 1504	24561	7.35 17.3	2.71 4.17	1.91 1.68	1800	26361	855	3.59		
	4	3114 (3175)	5259.0	53.7	F 5412 0 16236	262 753	0 1495	24157	7.22 17.0	2.69 4.16	1.90 1.66	1799	25956	1260	3.56		
	8		4974.4	65.2	F 5299 0 15897	264 747	0 1616	23823	7.08 16.77	2.67 4.15	1.89 1.65	1800	25623	1593	3.54		
2224 (500)	1		5251.2	15.4	F 5265 0 15796	731 257	0 1174	23223	7.03 16.3	2.67 4.20	1.89 1.59	1783	25007	2209	3.48		
	4	3270 (333.5)	4832.8	18.5	F 5092 0 15277	249 708	0 1180	22506	6.80 15.8	2.64 4.20	1.87 1.56	1781	24287	2929	3.43		
	8		4439.2	30.8	F 4916 0 14749	247 694	0 1262	21868	6.57 15.3	2.61 4.18	1.84 1.53	1787	23655	3561	3.37		
4448 (1000)	1		5096.5	15.4	F 5141 0 15424	251 714	0 1079	22608	6.26 15.8	2.65 4.20	1.87 1.56	1807	24416	2860	3.43		
	4	3364 (343.5)	4702.7	18.5	F 4963 0 14888	243 690	0 1134	21917	6.62 15.7	2.62 4.20	1.85 1.53	1802	23719	3496	3.38		
	8		4401.1	30.8	F 4828 0 14484	243 681	0 1217	21452	6.15 15.0	2.60 4.16	1.83 1.52	1809	23261	3954	3.35		

\* Includes Insulation Thicknesses

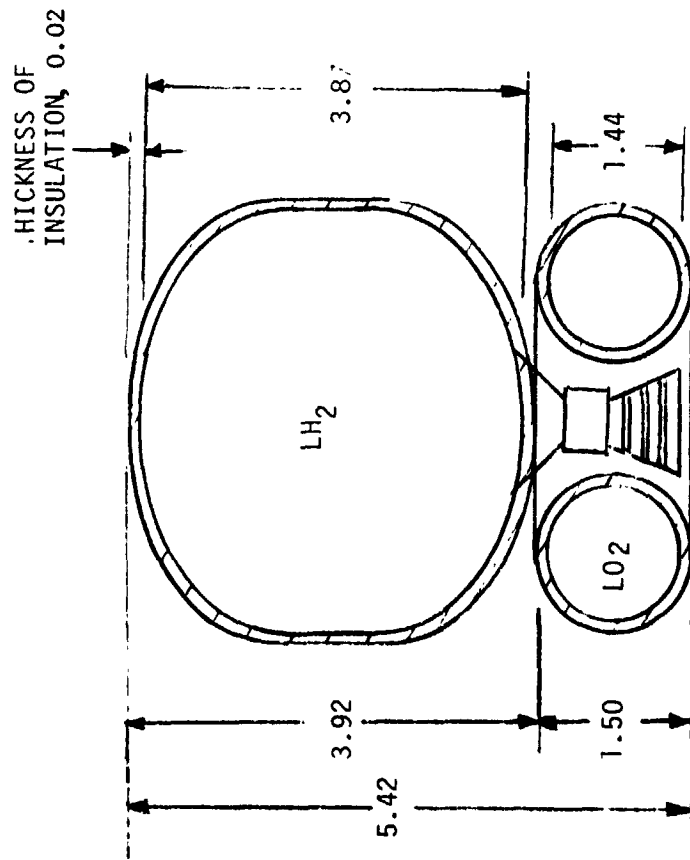
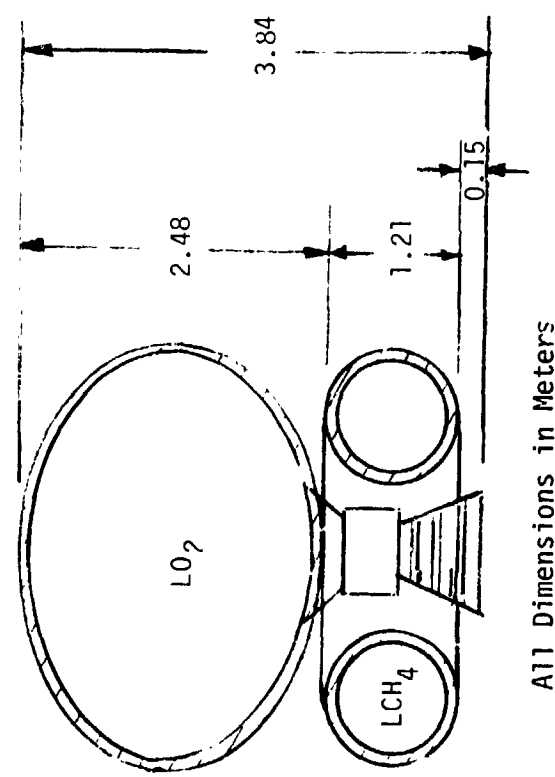


FIGURE II-21a LO<sub>2</sub>/LH<sub>2</sub>, 4450 N THRUST, 8 BURNS, MLI, TANDEM/TOROIDAL CONFIGURATION



All Dimensions in Meters

FIGURE II-21b LO<sub>2</sub>/LCH<sub>4</sub>, 4450 N THRUST, 8 BURNS, MLI, TANDEM/TOROIDAL CONFIGURATION

For ease of comparison, the payload mass and LTPS overall length for each concept sized is shown by the bar charts in Figures II-22 through II-25. Each chart shows the 18 combinations of thrust, insulation, and burn strategy for a particular propellant and tank configuration. Systems which minimized LTPS length and maximized the mass available to the LSS were chosen for further evaluation. Since the reduced complexity of this insulation concept merits further evaluation some SOFI configurations were chosen even though they did not satisfy the aforementioned criteria. Selected configurations are noted on the bar charts by the circled burn numbers.

The criterion for this portion of the study requires a minimum length system. Thus, the thicknesses of SOFI were sized for optimum tank length rather than optimum mass. However, the MLI systems were mass optimized for the reasons explained in Section II-G-2. Even when SOFI was length optimized, it still required a thickness of about 0.26 m (10 in) on the LH<sub>2</sub> tank. The increase in tank length over the MLI systems can be graphically seen in Figure II-22. The SOFI systems are longer than the MLI systems for three reasons (1) more propellant is required because boiloff is greater; (2) thicker insulation adds length to the system; and (3) as the insulation thickness increases the tank diameter must decrease. This decrease in tank diameter also causes an increase in tank length (e.g., each 10 cm decrease in LH<sub>2</sub> tank diameter produces a length increase of 28 cm for the LO<sub>2</sub>/LH<sub>2</sub> combination). No SOFI cases were chosen for LH<sub>2</sub>-fueled systems due to this large length increase.

All selected systems were 4 and 8 perigee burn configurations because of payload penalties associated with the large gravity losses of a single perigee burn. Among systems of similar propellant combination and tank arrangements higher thrust levels increased LSS lengths by at most 12 percent for LO<sub>2</sub>/LH<sub>2</sub> with MLI, 7 percent for LO<sub>2</sub>/LCH<sub>4</sub>, and 7 percent for LO<sub>2</sub>/RP-1. However, an increase in thrust from 445 N to 4450 N will increase the mass available for the payload considerably more - from 30 percent to 60 percent.

As expected, the LO<sub>2</sub>/LH<sub>2</sub> combination produced the lightest propulsion systems. In fact, each 445 N thrust systems using MLI allowed a heavier LSS payload than the comparable 4450 N thrust systems with LO<sub>2</sub>/LCH<sub>4</sub>. For this reason, two configurations from all three thrust levels were chosen from the LO<sub>2</sub>/LH<sub>2</sub> candidates. Eight were selected from each tank arrangement

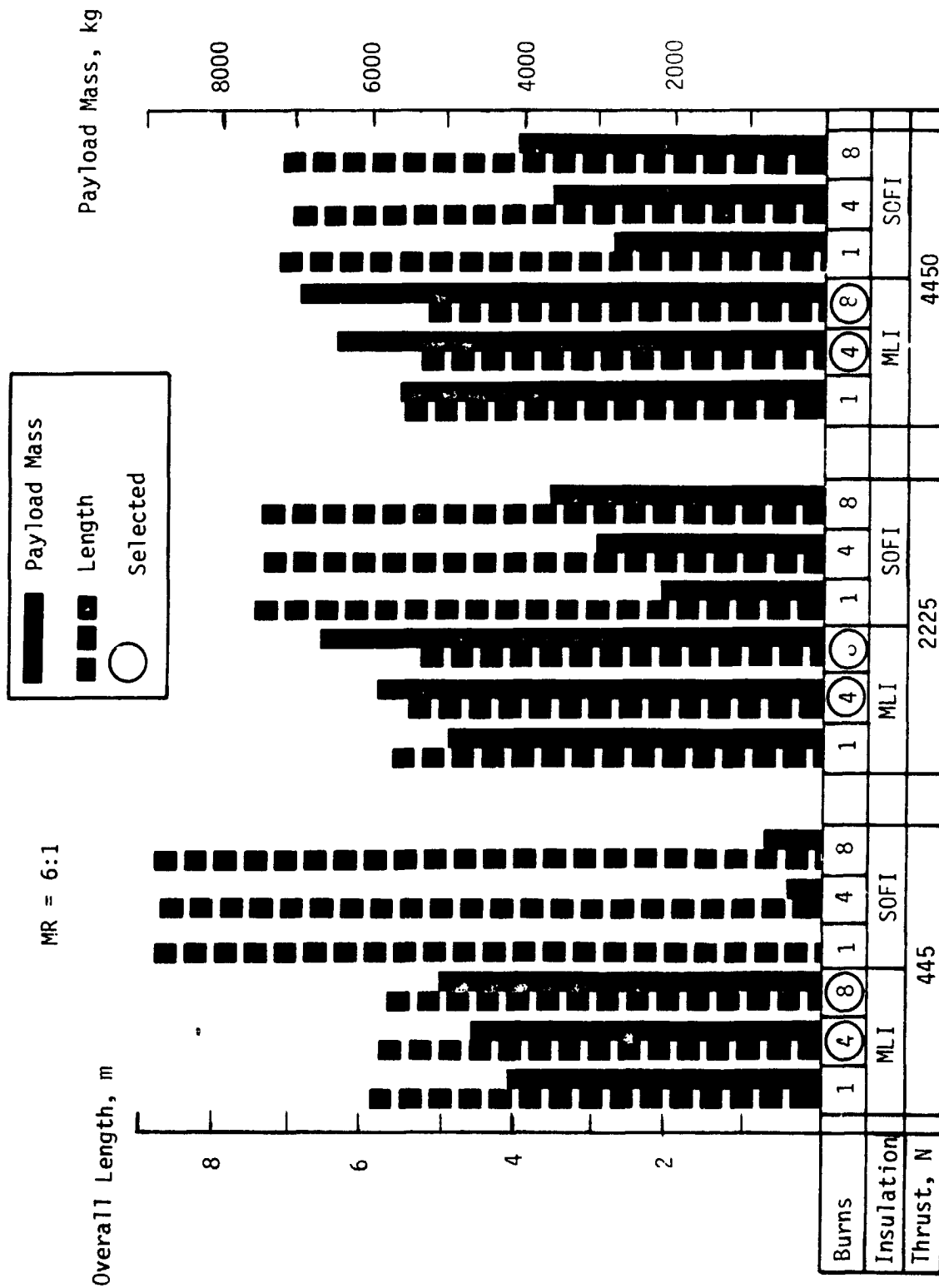


FIGURE II- 22 LO<sub>2</sub>/LH<sub>2</sub> PROPELLANT SYSTEM LENGTH AND PAYLOAD MASS



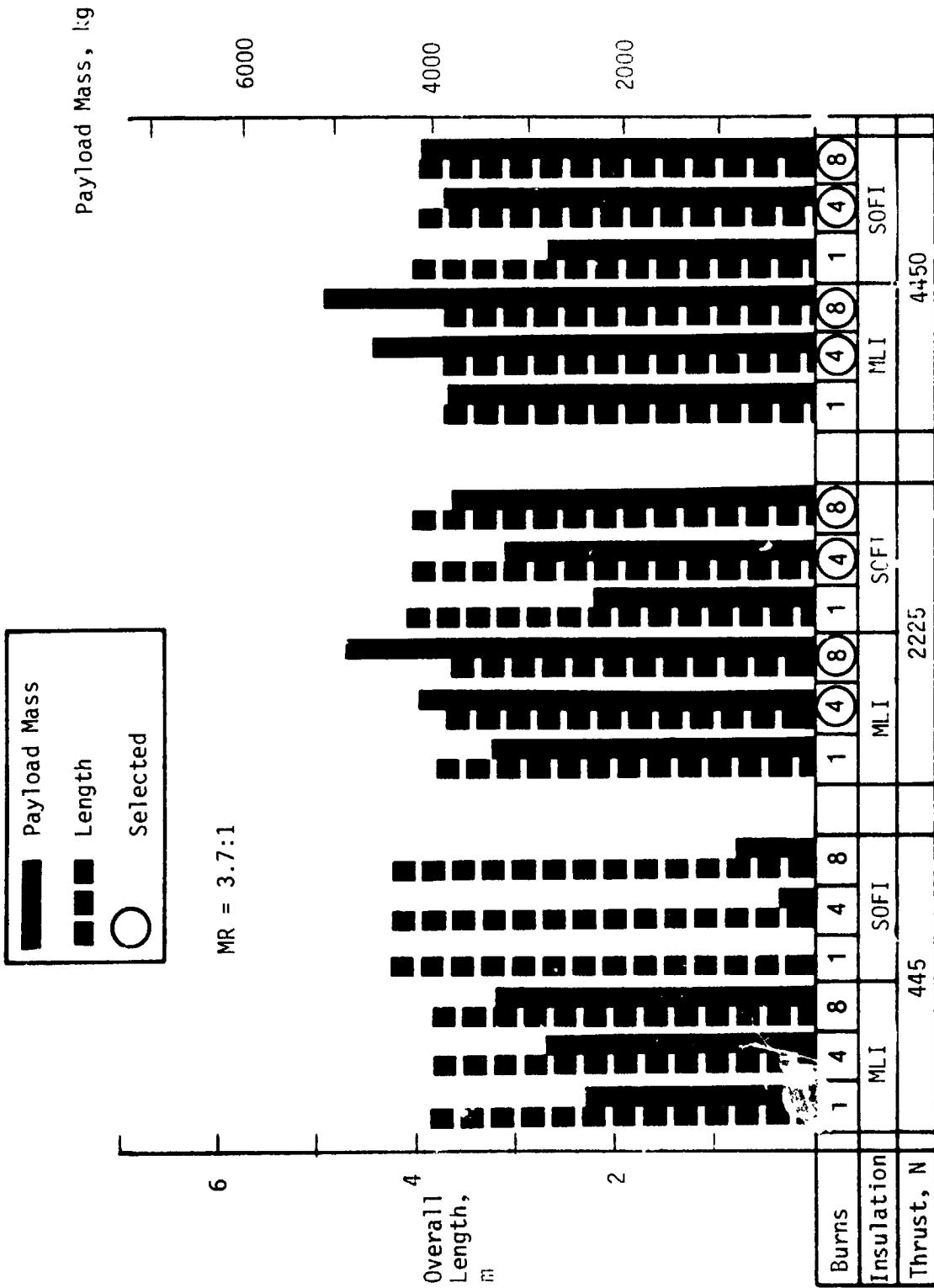


FIGURE II-23 LO<sub>2</sub>/LCH<sub>4</sub> PROPELLANT SYSTEM LENGTH AND PAYLOAD MASS

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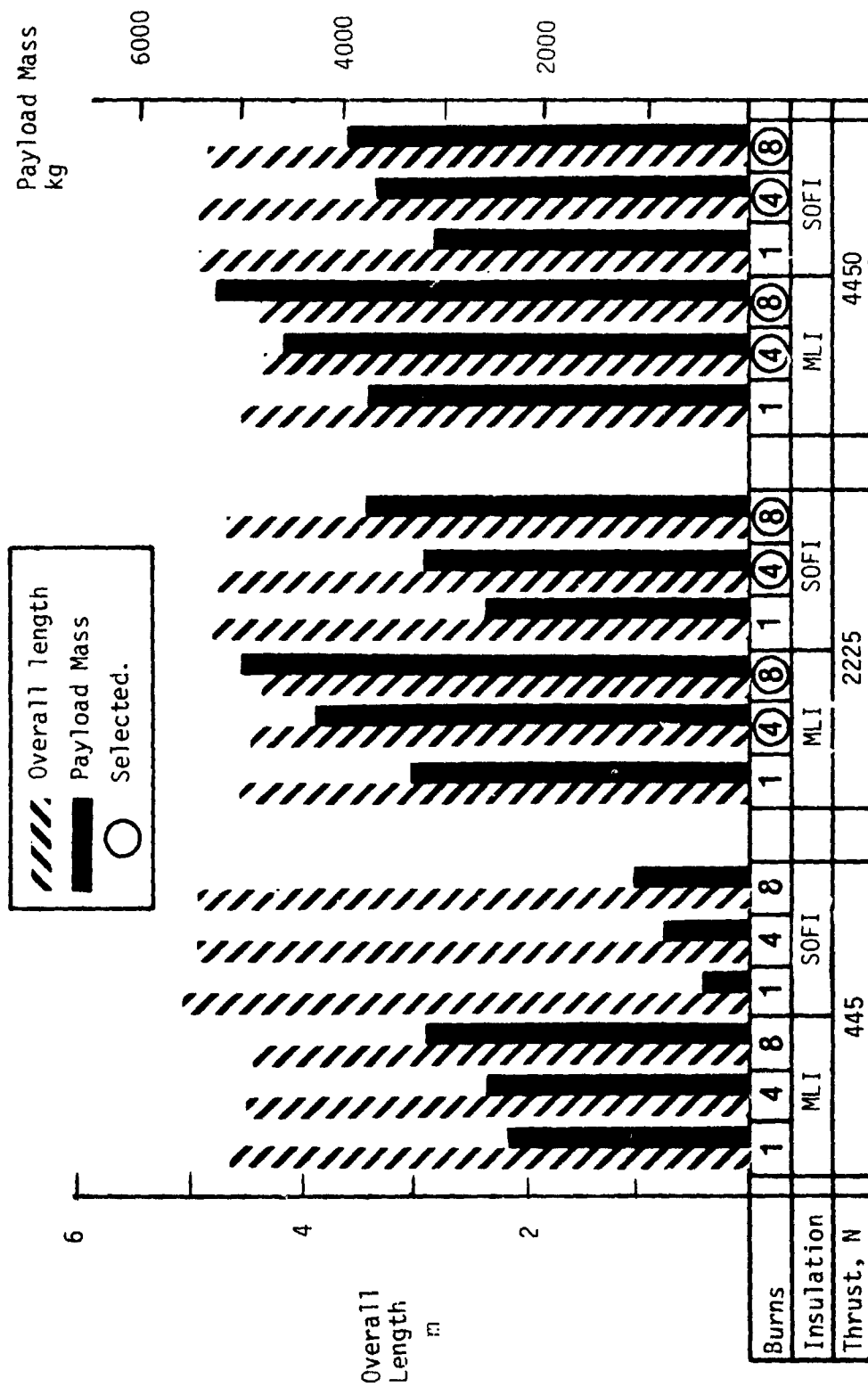


FIGURE II - 24 LO<sub>2</sub>/LCH<sub>4</sub> PROPELLANT SYSTEM LENGTH AND PAYLOAD MASS FOR PARALLEL TANKING CONFIGURATION

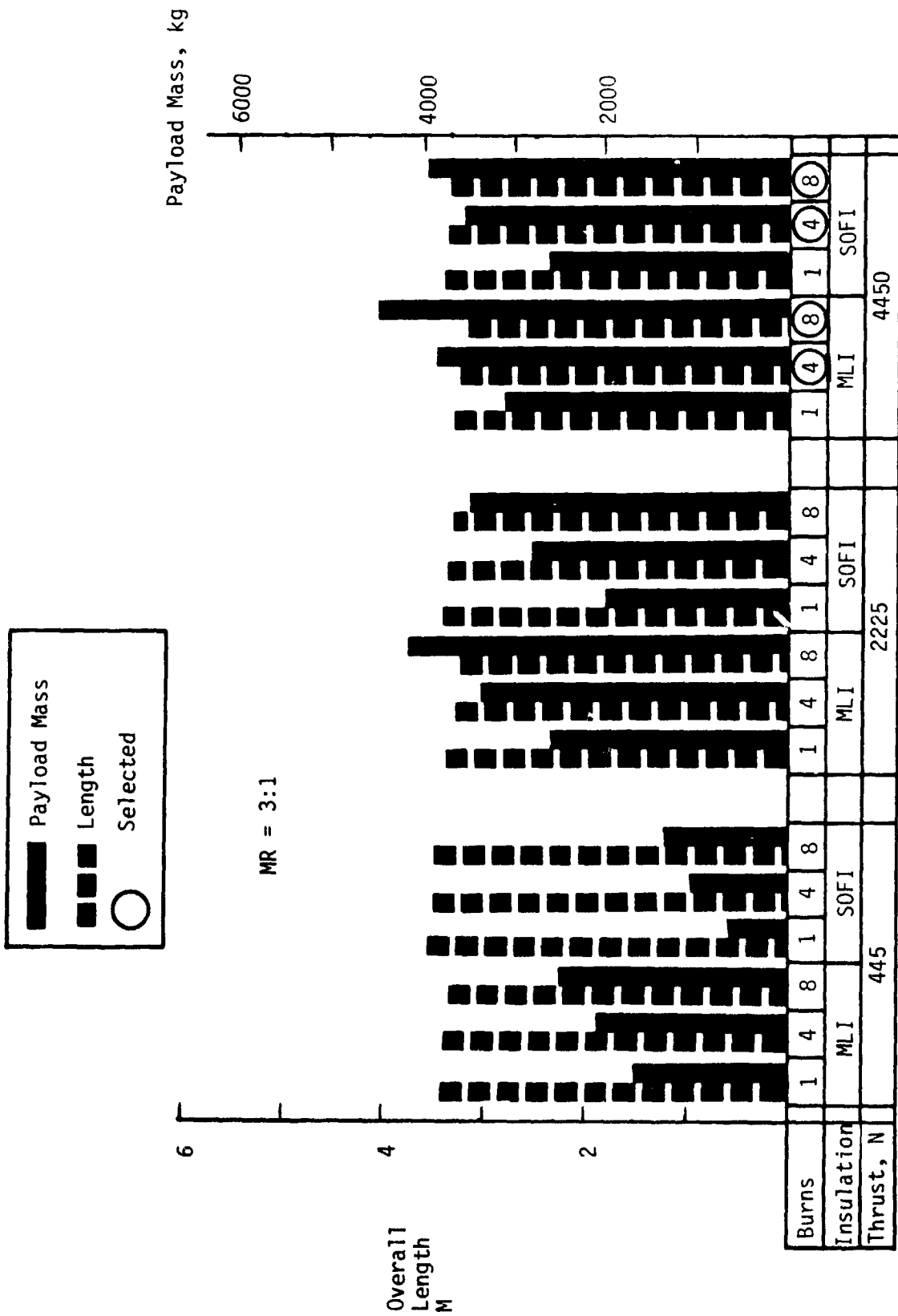


FIGURE II-25 LO<sub>2</sub>/RP-1 PROPELLANT SYSTEM LENGTH AND PAYLOAD MASS

using  $LO_2/LCH_4$  - four SOFI and four MLI. Thrust levels of 2225 N and 4450 N produced comparable LTPS lengths and masses, but 445 N systems were considerably heavier and none were chosen. The  $LO_2/RP-1$  systems were the shortest, but due to the low performance of this propellant combination, only four configurations (all 4450 N thrust) were chosen for further evaluation.

The 26 chosen configurations were then carried into the next section of the study for incorporation of the three different propellant management techniques and further refinement of the propellant requirements. Configurations were numbered 1 through 26 (Table II-14) for ease of identification.

TABLE II-14 SELECTED PROPELLANT SYSTEM CONFIGURATIONS

CONFIG- URATION	PROPELLANT COMBINATIONS	THRUST		NO. OF BURNS	INSULATION SYSTEM	TOROID/ ELLIPSOIDAL TANK COMBINATION
		N	T <sub>b</sub> f			
1	LO <sub>2</sub> /LH <sub>2</sub> ↓	445	100	4	MLI ↓	TOROID/ ELLIPSOIDAL TANK COMBINATION ↓
2		445	100	8		
3		2225	500	4		
4		2225	500	8		
5		4450	1000	4		
6		4450	1000	8		
7	LO <sub>2</sub> /LCH <sub>4</sub> ↓	2225	500	4	MLI MLI SOFI SOFI MLI MLI SOFI SOFI	
8		↓	↓	8		
9		↓	↓	4		
10		↓	↓	8		
11		4450	1000	4		
12		↓	↓	8		
13		↓	↓	4		
14		↓	↓	8		
15	LO <sub>2</sub> /RP-1 ↓	4450	1000	4	MLI MLI SOFI SOFI	
16		↓	↓	8		
17		↓	↓	4		
18		↓	↓	8		
19	LO <sub>2</sub> /LCH <sub>4</sub> ↓	2225	500	4	MLI MLI SOFI SOFI MLI MLI SOFI SOFI	
20		↓	↓	8		
21		↓	↓	4		
22		↓	↓	8		
23		4450	1000	4		
24		↓	↓	8		
25		↓	↓	4		
26		↓	↓	8		

### III. EVALUATION OF PROPELLANT MANAGEMENT TECHNIQUES

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In order to further develop the propulsion system concepts selected in Section II, preliminary designs of propellant management devices were prepared for each of the propulsion systems. These designs were of sufficient detail to determine the feasibility and the weight penalty of the propellant management techniques. Three propellant management techniques were identified as being appropriate for propulsion systems of this size: propulsive settling, partial acquisition devices, and total acquisition devices. Propulsive settling makes use of an auxiliary propulsion system to produce an acceleration that will position the propellant at the outlet of the main propulsion system tanks. Both partial and total acquisition devices make use of the maturing technology of surface tension propellant management devices. These devices are made with fine-mesh screen and make use of the surface tension of the propellant to expel liquid in preference to gas.

The approach used to design the propellant management concepts and determine their feasibility and weight penalties is described in this section. At the end of this chapter the calculated weight penalties for propellant management were substituted for the previous estimates as part of the process of establishing a weight estimate for the total LTPS. Certain propulsion system and mission parameters were required to perform this analysis, such as tank geometry, flowrates, acceleration, and propellant remaining for each engine burn. These parameters were computed using the computer model (PROP) described in Section II.

#### A. PROPULSIVE SETTLING

Propulsive settling is a rather straight-forward method of providing propellant to an engine so that it can start in low-g. Propulsive settling is a proven technique, having been used for propellant management on the Transtage, Centaur, and Apollo space vehicles and is only applicable to a propulsion system that will maintain the propellant in the settled condition once settling has been achieved (such as the main propulsion system of a spacecraft). Since the LTPS is such a system, propulsive settling was applicable and further evaluation established that it was feasible.

The propulsive settling method of propellant management requires an auxiliary propulsion system that will orient the propellant over the tank outlet prior to each main engine start. It was assumed that an auxiliary propulsion system was available, including thrusters, any required tankage, and its own propellant management system. Therefore, only the propellant used by the auxiliary thrusters for the purpose of propulsive settling contributed to the weight penalty. It was also assumed that the thrust of the auxiliary thrusters could be selected solely on the basis of the propulsive settling requirements.

Two types of auxiliary propulsion systems were considered. One type used the same propellants as the main engines, except that the specific impulse was degraded by 10 percent. The second type had its own supply of earth storable propellants:  $N_2O_4$  and MMH with a specific impulse of 2750 N-sec/kg (280  $lb_f$ -sec/ $lb_m$ ).

#### 1) Propellant Settling Time

The key to the design of a propulsive settling system is the time required to settle the propellant. The time required to settle the propellant determines how long the auxiliary thrusters must operate, and hence the amount of propellant they consume and that contribution to the weight penalty. A number of studies have been performed investigating the manner and rate of propellant settling under various conditions. Off-axis accelerations and unsymmetrical conditions have been shown to have a significant influence on the manner of propellant motion during settling (Ref. 6). One of the more recent studies, performed at NASA-LeRC, established an approach for optimizing the time required to settle the propellant (Ref. 7).

An analytical approach presented in that study was used, where applicable, to select an optimum value for the settling thrust and to predict the settle time. The NASA study determined that increasing the settling acceleration decreases the reorientation time to the point where geysering and splashing at the tank outlet occur, which cause an increase in the settle time. The  $\Delta V$  of the settling thrusters, which is a function of the settling acceleration and settle time, can be minimized for any given tank size fill volume, and propellant. Minimizing the  $\Delta V$  also minimizes the propellant usage.

Propellant settling in a representative LTPS LH<sub>2</sub> tank was analyzed to illustrate the optimization approach (Figure III-1). The  $\Delta V$  required to achieve reorientation was plotted versus the Bond number (Bo). The fill fraction and Weber number (We) were the independent parameters.

The lines of constant fill fraction show that there was a minimum  $\Delta V$  as We and Bo were varied. From the figure, it appears that the  $\Delta V$  could be minimized for the full range of fill fractions at a Bo of about four. A recent study has substantiated this result for the general case of reorientation in a cylindrical tank (Appendix B of Ref. 8). The applicability of this analytical approach is limited to cylindrical tanks with relatively long barrel sections and conditions that yield low values of Bo and We (<1000).

For those conditions where the above approach was not applicable (e.g., ellipsoidal tanks, toroidal tanks, and higher Bond numbers) an alternative approach based on free-fall periods was used. Multiples of the time required for a particle to fall from the initial interface position to the tank bottom provided an estimate of the settle time (Ref. 9). Comparisons between the optimized approach and the free-fall approach indicated that both approaches yielded similar results and provide a fair representation for the weight penalty of the propulsive settling technique.

The application of these methods of computing the settle time as based on the following considerations:

- a) The settling acceleration should yield a Bond number between four and five to produce the most efficient settling of the propellant;
- b) The acceleration must be large enough to make the propellant interface unstable, so that settling will occur in both the fuel and oxidizer tanks (Bond number greater than 1.5); and



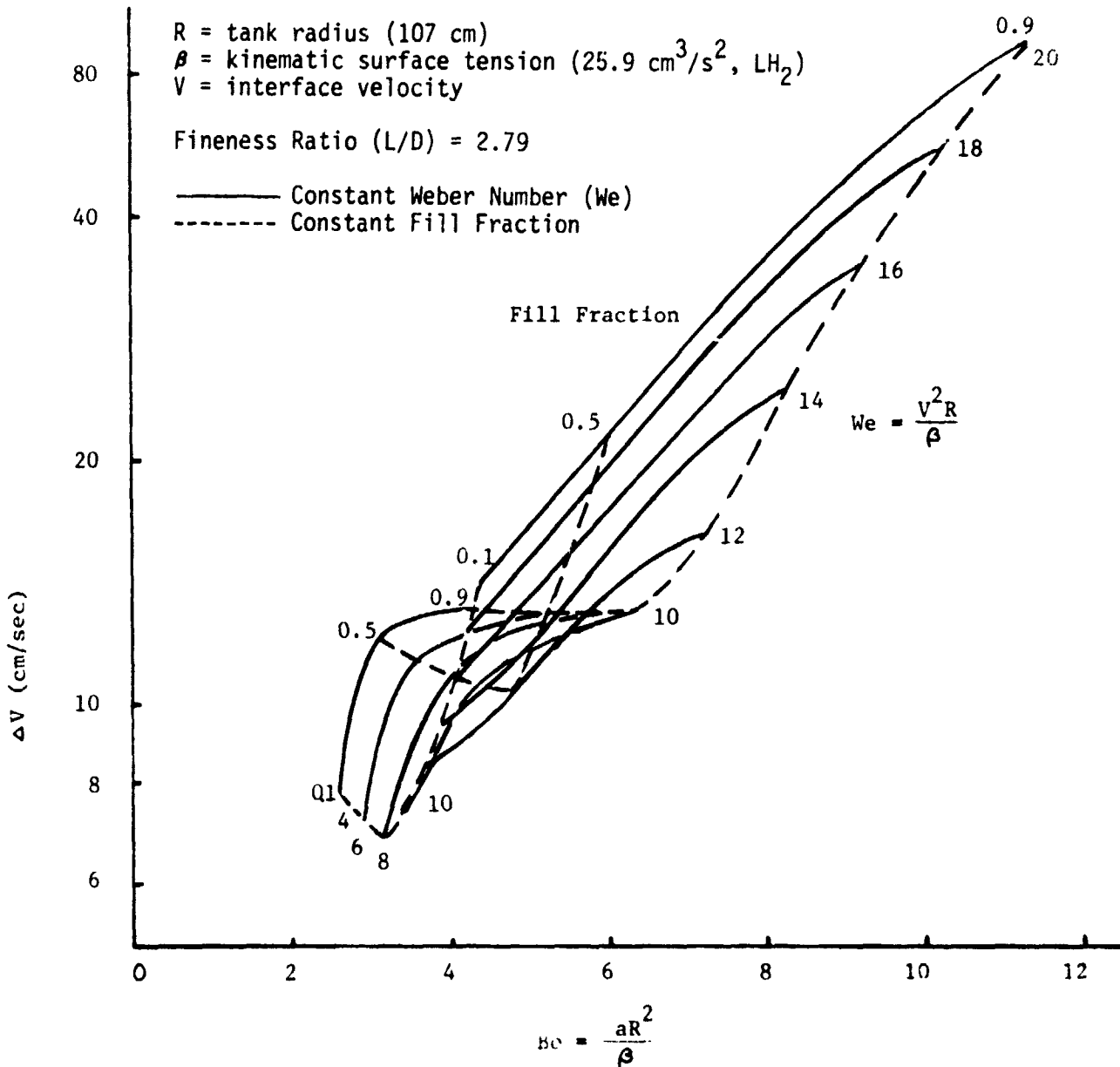


FIGURE III-1 PROPELLANT REORIENTATION OPTIMIZATION

- c) For the first burn, the atmospheric drag acceleration is at a maximum and opposes applied acceleration, so the applied acceleration must exceed the sum of the drag and the acceleration required for settling.

These requirements conflict in some respects. The acceleration necessary to cause interface instability in one tank may yield a  $Bo$  greater than five in the other tank. In this case the requirement that the interface be unstable in both tanks had precedence, and a less efficient settling condition had to be accepted.

Other conflicts arose due to the variability of the atmospheric drag. There are daily variations in the atmospheric density and variations due to solar activity at any orbital altitude. A settling system will have to be designed for the maximum drag, and the variability of the density could yield an actual drag that is up to a factor of five less, making the settling acceleration applied to the propellant exceed the optimum range. For any given payload and orbital altitude the atmospheric drag can be calculated. For the purpose of this study, representative payloads were considered so that a typical value of the drag could be calculated. A large space structure that fits into the Shuttle cargo bay can have a frontal area of between 700 and 7000  $m^2$  (8,000 and 80,000  $ft^2$ ). Using the larger area and a deployment altitude of 370 km (200 n.mi.) a drag acceleration of  $2.2 \times 10^{-5}g$  was calculated. This value was used for analyzing all the propulsive settling systems.

After the first burn, the drag will be insignificant due to the higher orbital altitude. A settling system designed to provide sufficient acceleration prior to the first burn will be oversized for settling prior to subsequent burns, when the drag can be neglected and the spacecraft mass is less. Our approach was to assume that there were a number of RCS thrusters available to perform the settling, and the number fired could be varied in increments to obtain a settling acceleration near the optimum value.

The manner of calculating the settle time depended upon the quantity of propellant in the tank. For large fill levels the settling was based on the motion of the ullage bubble. Assuming the worst case initial condition of the ullage bubble over the tank outlet, the bubble had to be displaced by the settling acceleration a distance sufficient to prevent the bubble from being drawn into the outlet at main engine start. When the ullage could no longer be represented as a bubble, the time required for the propellant to flow down the tank wall and collect sufficiently at the outlet to allow main engine start was calculated.

## 2) Weight Penalty for Propulsive Settling

The weight penalty for propulsive settling consists of the propellant used by the auxiliary propulsion system in settling the main engine propellant and the propellant that cannot be drained from the main tanks. The propellant required for settling was calculated from the settle time, thrust of the auxiliary propulsion system, and the specific impulse of the propellants being used.

The residual propellant in the tank is determined by the point at which gas is drawn into the tank outlet, so that gas-free propellant is no longer being supplied to the engine. The best available correlations for this suction dip phenomena were used to predict the residual propellant mass. The accelerations for the final burn of the LTPS were large enough to make it a high-g draining condition, so the influence of surface tension was negligible. For the tanks with ellipsoidal domes the following correlation from Reference 10 was used.

$$\frac{h_{vi}}{r_o} = 1.03 \cdot \left[ \left( \frac{R}{r_o} \right)^2 \cdot \left( \frac{v_o^2}{2 g_o r_o} \right) \right]^{0.143}$$

where  $h_{vi}$  = vapor ingestion height,  
 $r_o$  = outlet radius,  
 $R$  = tank radius,  
 $v_o$  = velocity in outlet line, and  
 $g_o$  = accelerations,

This correlation was developed for a tank with a hemispherical dome, but the differences in tank geometry were accounted for in the analysis. When the volumetric flowrate was substituted into this correlation, it was found that the vapor ingestion height is independent of the outlet radius ( $r_o$ ). For the toroidal tanks the residuals were scaled from test data presented in Reference 11. The acceleration was assumed to be parallel with the tank axis, and the toroidal tank had only one outlet. Tank draining was considered in more detail for the improved LTPS concepts in Section V.

The pertinent parameters for the propulsive settling technique, when applied to the 26 propulsion systems, are summarized in Table III-1. It was found that the propellant required for settling was an almost insignificant contribution to the total weight penalty. While improvements in the technology regarding the prediction of settling time are necessary, it appears that conservative approaches to determining the settling requirements are acceptable.

The draining residual essentially determined the weight penalty for propulsive settling. These residuals became very large at the higher thrust levels due to a greater influence of flowrate in comparison to acceleration. The residuals were much greater for the toroidal tanks. Methods of reducing the draining residual were considered for the improved LTPS concepts in Section IV.

#### B. PARTIAL ACQUISITION DEVICES

Partial acquisition device is one general type of surface tension propellant management device. The fine-mesh screen used to fabricate the device preferentially orients a portion of the propellant at the tank outlet for the purpose of engine start. This device is only applicable to a propulsion system that will settle the propellant at the outlet and maintain that orientation throughout the engine burn. This type of device is applicable to an LTPS, and a feasible concept is described in the following paragraphs. One type of partial acquisition device has been in use for a number of years on the Agena, and the Space Shuttle Orbital Maneuvering System (Ref. 12) uses another type of partial acquisition device.

TABLE III-1 PARAMETERS FOR PROPELLANT SETTLING

Configuration*	Settling Thrust Per Engine N (lbf)	Maximum Number of Thrusters	Total Settling Impulse N sec (lbf sec)	Mass of Propellant Required for Settling		Draining Residual		Total Weight Penalty	
				N <sub>2</sub> O <sub>4</sub> /MMH, kg (lb <sub>m</sub> )	Primary Propellants, kg (lb <sub>m</sub> )	Fuel, kg (lb <sub>m</sub> )	Oxidizer, kg (lb <sub>m</sub> )	Using N <sub>2</sub> O <sub>4</sub> /MMH, kg (lb <sub>m</sub> )	Using Primary Propellants, kg (lb <sub>m</sub> )
1	0.4 (0.1)	16	3620 (814)	1.3 (2.9)	1.0 (2.1)	8.2 (18)	66 (146)	76 (167)	75 (166)
2	0.4 (0.1)	16	6670 (1500)	2.4 (5.3)	1.8 (3.9)	8.2 (18)	64 (141)	74 (164)	74 (163)
3	0.4 (0.1)	16	4800 (1080)	1.8 (3.9)	1.2 (2.7)	13 (29)	166 (365)	181 (398)	180 (397)
4	0.4 (0.1)	16	10500 (2360)	3.8 (8.4)	2.7 (6.0)	14 (30)	177 (391)	195 (429)	194 (427)
5	0.4 (0.1)	16	7210 (1620)	2.6 (5.8)	1.8 (4.0)	16 (35)	250 (551)	269 (592)	268 (590)
6	0.4 (0.1)	16	12200 (2740)	4.4 (9.8)	3.1 (6.8)	16 (36)	240 (530)	261 (576)	260 (573)
7	2.2 (0.5)	4	4980 (1120)	1.8 (4.0)	1.6 (3.5)	154 (339)	87 (191)	242 (534)	242 (534)
8	2.2 (0.5)	4	8230 (1850)	3.0 (6.6)	2.6 (5.8)	148 (326)	88 (195)	240 (528)	239 (527)
9	2.2 (0.5)	4	4540 (1020)	1.6 (3.6)	1.5 (3.2)	146 (321)	83 (182)	230 (507)	230 (506)
10	2.2 (0.5)	4	7250 (1630)	2.6 (5.8)	2.3 (5.1)	141 (311)	85 (188)	229 (505)	229 (504)
11	2.2 (0.5)	4	5120 (1150)	1.9 (4.1)	1.6 (3.5)	255 (562)	105 (232)	362 (798)	362 (798)
12	2.2 (0.5)	4	8180 (1840)	3.0 (6.6)	2.5 (5.6)	247 (545)	105 (232)	356 (784)	355 (783)
13	2.2 (0.5)	4	4580 (1030)	1.7 (3.7)	1.5 (3.2)	254 (559)	101 (222)	356 (785)	356 (784)
14	2.2 (0.5)	4	7380 (1660)	2.7 (5.9)	2.3 (5.1)	249 (549)	104 (229)	356 (784)	355 (783)
15	0.4 (0.1)	16	5290 (1190)	2.0 (4.3)	1.8 (3.9)	31 (69)	252 (556)	285 (629)	285 (629)
16	0.4 (0.1)	16	7300 (1640)	2.7 (5.9)	2.0 (4.3)	32 (70)	254 (561)	289 (637)	288 (636)
17	0.4 (0.1)	16	4890 (1100)	1.8 (3.9)	1.6 (3.6)	31 (68)	245 (541)	278 (613)	278 (613)
18	0.4 (0.1)	16	6850 (1540)	2.5 (5.5)	2.3 (5.0)	31 (68)	254 (560)	288 (634)	287 (633)
19	1.3 (0.3)	6	12800 (2880)	4.7 (10.3)	4.4 (9.6)	10 (23)	44 (97)	59 (131)	59 (130)
20	1.3 (0.3)	6	15500 (3490)	5.7 (12.5)	5.3 (11.6)	11 (24)	45 (99)	61 (135)	61 (134)
21	1.3 (0.3)	6	11000 (2480)	4.0 (8.9)	3.8 (8.3)	10 (21)	40 (89)	54 (119)	54 (119)
22	1.3 (0.3)	6	18300 (4110)	6.7 (14.7)	6.2 (13.7)	10 (22)	41 (90)	58 (127)	57 (126)
23	1.3 (0.3)	6	11800 (2660)	4.3 (9.5)	3.7 (8.1)	13 (28)	53 (117)	70 (155)	70 (154)
24	1.3 (0.3)	6	18600 (4180)	6.8 (14.9)	5.8 (12.7)	13 (29)	54 (120)	74 (163)	73 (161)
25	1.3 (0.3)	6	13400 (3010)	4.9 (10.8)	4.2 (9.2)	13 (28)	50 (110)	68 (149)	67 (147)
26	1.3 (0.3)	6	17300 (3880)	6.3 (13.9)	5.4 (11.8)	13 (28)	50 (111)	69 (153)	68 (151)

\* See Table II-14 for definitions of configurations

## 1) Partial Acquisition Device Concept

A reservoir, fabricated with a fine-mesh screen, holds propellant over the tank outlet so that it is available for engine start. After the engine has been started, the propellant outside the reservoir settles and sustains propellant feed. One approach is to design the reservoir so that it will refill during each burn. Refill can take place if the hydrostatic pressure of the settled propellant exceeds the retention capability of the screen that forms the reservoir, so that gas can escape from within the reservoir (Ref. 13). Due to the low accelerations of the LTPS, the pores in the screen that allows refill would have to be large (typically a coarse square weave screen is required). Such screen material would severely degrade the ability of the reservoir to remain wetted during the coast periods, when retention of propellant in the reservoir is required. Our conclusion was that refill is not feasible for the LTPS application. Therefore, the approach of designing the reservoir so that it will hold enough propellant to perform all the engine starts was the only feasible approach for a partial acquisition device.

The reservoir must contain sufficient propellant to perform every engine start. At the beginning of each burn a portion of that propellant is consumed. The volume of the trap must take into account the following requirements: 1) the quantity of propellant required to start the main engine and maintain operation until the propellant settles at the beginning of each burn, 2) the propellant required to fill the feed line prior to each engine burn, 3) the propellant required for chilldown of the main engine, and 4) the propellant lost from the reservoir due to vaporization.

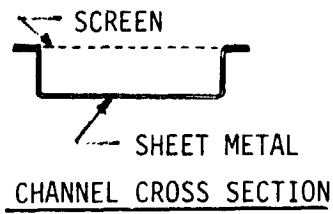
The settling requirement was determined by calculating the settle time based on methods described for the propulsive settling technique. With a partial acquisition device, settling does not have to be as complete as it has to be for the propulsive settling system, since the screen of the partial acquisition device will filter out any gas entrained in the settled propellant. The quantities required for line fill and chilldown were those

used for the sizing of the propulsion system (see Section II). The amount lost due to vaporization was a fraction of the total boiloff from the tank. That fraction was determined from the percentage of the mission during which the reservoir may not be in contact with the bulk propellant and the ratio of the reservoir surface area to the bulk liquid surface area.

While the reservoir holds propellant in the vicinity of the outlet, it also retains an increasing quantity of gas as that propellant is used. A means of feeding only liquid from inside the reservoir to the outlet must be provided. This was done by adding a simple fine-mesh screen channel network inside the reservoir that was connected to the outlet. The channel network was configured inside the reservoir so that some portion of it will always be in contact with the liquid.

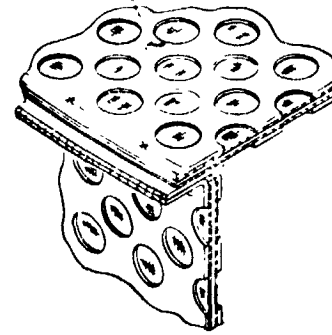
Basic configurations for the partial acquisition devices were selected for ellipsoidal and toroidal tanks (Figures III-2 and III-3). For an ellipsoidal tank a cylindrical reservoir configuration was selected. This is a compact configuration, easy to manufacture and integrate with the tank, and provides good communication with the bulk propellant during settling and terminal drain. The height of the reservoir was kept to a minimum to reduce the effects of hydrostatic pressure on the retention capability of the screens, but the proportions of the reservoir were also considered to limit the surface area and weight of the device. The same factors influenced the selection of a truncated, wedge-like sector for the reservoir in the toroidal tanks. This shape simplifies fabrication and fits compactly over the tank outlet. The dimensions of each reservoir were selected, trading off these factors, so as to obtain the required reservoir volume, including a 1.5 factor of safety. The surface of the reservoir was a sandwich of perforated plate and screen, which aids in keeping the screen in a wetted condition throughout the mission. Gas will bubble through the screen when liquid is withdrawn or evaporated from the reservoir, but the screen must rewet so the reservoir will continue to retain liquid.

The reservoir would not rest on the tank wall but would be spaced so as to avoid excessive heat inputs. If too much heat enters the reservoir, vaporization of liquid within the reservoir could cause the pressure to rise and result in liquid being forced out to the bulk region.

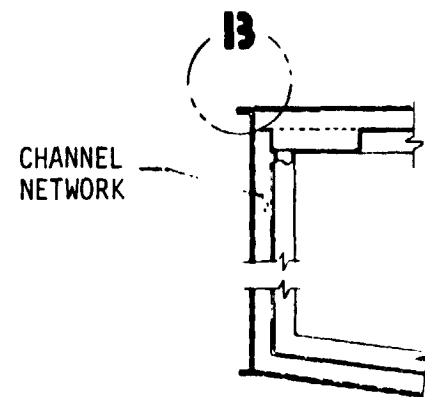
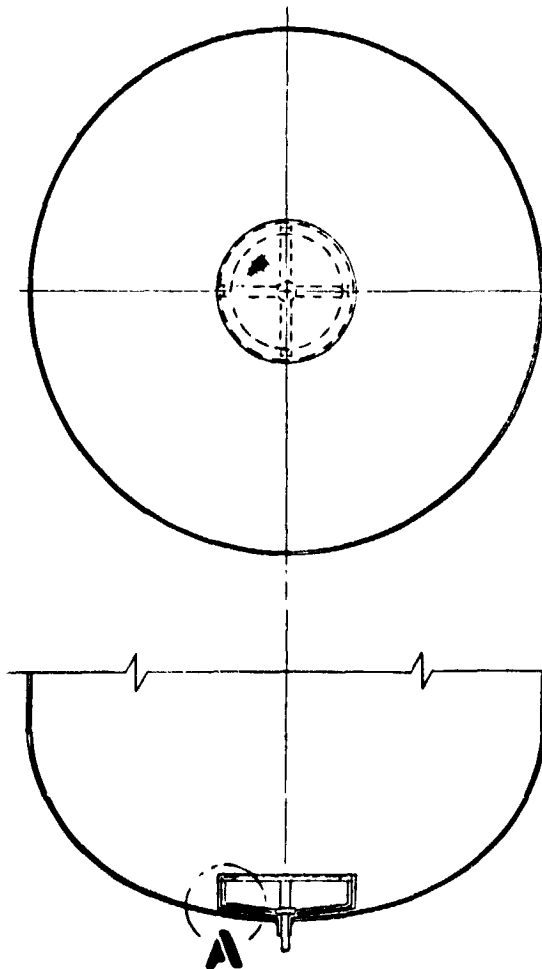


RESERVOIR CONSTRUCTION

PERFORATED PLATE - SCREEN -  
SPACER - SCREEN - PERFORATED PLATE  
SANDWICH



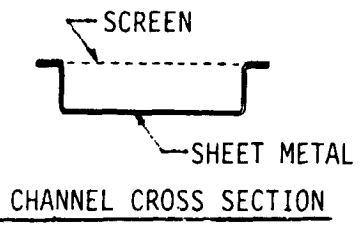
**DETAIL B**



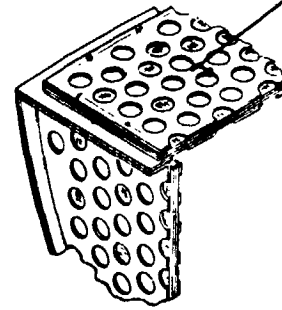
**DETAIL A**

FIGURE III-2 PARTIAL ACQUISITION DEVICE FOR ELLIPSOIDAL TANK

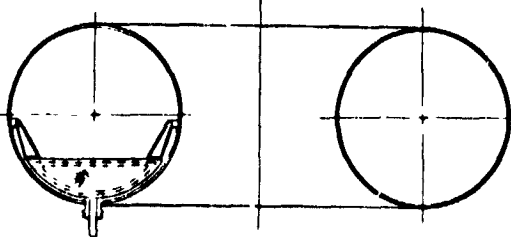
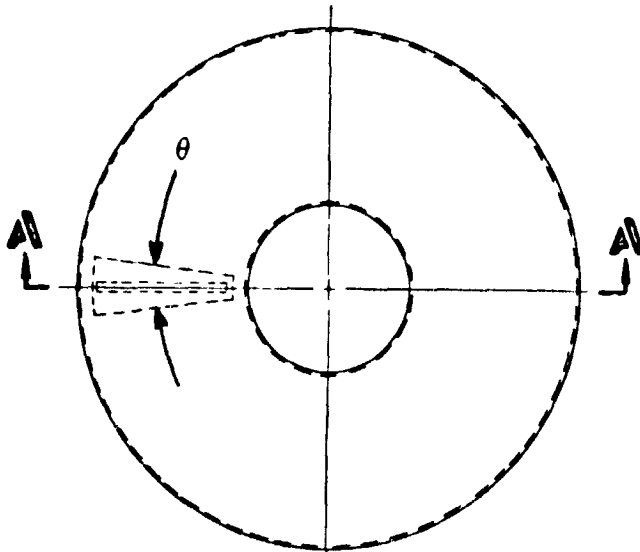




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SCREEN-PERFORATED PLATE SANDWICH



**VIEW 13**



**SECTION A-A**

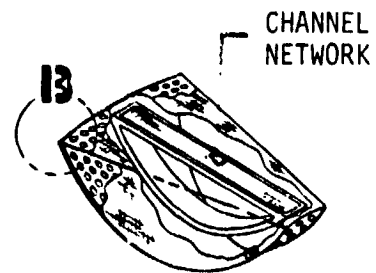


FIGURE III-3 PARTIAL ACQUISITION DEVICE FOR TOROIDAL TANK

Dryout of the screens is another concern. As long as the vaporization occurs on the outer screen surface of the reservoir, (due primarily to heat transfer with the ullage gas) it will function properly. The loss of liquid due to vaporization tends to lower the pressure inside the reservoir.

## 2) Weight Penalty for Partial Acquisition

The weight penalty for partial acquisition was determined by the weight of the device and the weight of the propellant that cannot be expelled from the tank. The weight of the device was determined by designing a device for each of the propulsion system concepts. The reservoir was sized to meet the requirements described in the previous section, and the internal flow channels were sized for the propellant flowrate and effective expulsion of the reservoir. The structure needed to attach the device to the tank was also considered. Gas-free expulsion of propellants will cease when gas begins to be ingested into the channels within the reservoir as the bulk propellant level falls below those channels. The propellants remaining within the channels and the puddle below the channels determined the total propellant residual.

The pertinent parameters for the partial acquisition devices are listed in Table III-2 for the series tankage concepts and Table III-3 for the parallel tankage concepts. For the series tanks, the weight penalty varied from 40 to 80 kg (90 to 180 lbm) with little noticeable influence of thrust or number of burns on the result. The  $LO_2/LCH_4$  concepts were lighter than the others and all the  $LO_2/LH_2$  and  $LO_2/RP-1$  concepts had similar weight penalties. The weight penalty for the parallel tank concepts had a similar range of variation, but a stronger influence of the SOFI versus MLI could be seen.

The allowance for vaporization in sizing the reservoir was one of the most significant factors influencing the weight penalty. The vaporization loss accounted for one-third to one-half of the volume, being greatest for the concepts with SOFI. The contributions to the reservoir volume for the settling requirement and engine chilldown were of equal magnitude.

TABLE III-2 PARAMETERS FOR PARTIAL ACQUISITION DEVICES, SERIES TANKS

CONFIG #	ELLIPSOIDAL TANKS						TOROIDAL TANKS					
	HEIGHT cm (in.)	RADIUS cm (in.)	VOLUME, m <sup>3</sup> (ft <sup>3</sup> )	WEIGHT OF RESIDUALS, kg (lb <sub>m</sub> )	DEVICE WEIGHT, kg (lb <sub>m</sub> )	HEIGHT cm (in.)	ANGLE, θ,(+) (deg.)	VOLUME m <sup>3</sup> (ft <sup>3</sup> )	WEIGHT OF RESIDUALS, kg (lb <sub>m</sub> )	DEVICE WEIGHT, kg (lb <sub>m</sub> )	TOTAL WT. PENALTY, kg (lb <sub>m</sub> )	
1	33.5 (13.2)	53.8 (20)	0.27 (9.6)	0.45 (1)	19 (41)	39.1 (15.4)	7.0	0.06 (2.2)	42 (93)	10 (21)	70.8 (156)	
2	32.3 (12.7)	61.0 (24)	0.38 (13.3)	0.45 (1)	24 (52)	38.6 (15.2)	9.6	0.08 (3.0)	42 (93)	10 (23)	76.7 (169)	
3	25.1 (9.9)	50.8 (20)	0.20 (7.2)	0.45 (1)	18 (40)	37.6 (14.8)	8.9	0.07 (2.6)	43 (94)	10 (23)	71.7 (158)	
4	30.2 (11.9)	61.0 (24)	0.35 (12.4)	0.91 (2)	24 (53)	36.3 (14.3)	13.8	0.11 (3.9)	43 (94)	12 (26)	79.4 (175)	
5	28.2 (11.1)	50.8 (20)	0.23 (8.1)	0.91 (2)	20 (45)	36.8 (14.5)	10.7	0.09 (3.1)	44 (98)	12 (26)	77.6 (171)	
6	35.1 (13.8)	61.0 (24)	0.41 (14.4)	0.91 (2)	27 (60)	35.8 (14.1)	16.7	0.13 (4.6)	44 (97)	13 (29)	85.3 (188)	
7	18.0 (7.1)	30.5 (12)	0.05 (1.9)	8.6 (19)	10 (21)	30.7 (12.1)	12.9	0.08 (2.8)	15 (34)	10 (22)	43.5 (96)	
8	21.1 (8.3)	35.6 (14)	0.08 (3.0)	9.1 (20)	11 (25)	30.0 (11.8)	19.3	0.11 (4.0)	15 (34)	12 (26)	47.6 (105)	
9	18.3 (7.2)	40.6 (16)	0.09 (3.3)	9.5 (21)	13 (28)	33.5 (13.2)	17.8	0.12 (4.3)	15 (34)	12 (26)	49.4 (109)	
10	22.4 (8.8)	45.7 (18)	0.15 (5.2)	10 (22)	15 (34)	33.0 (13.0)	26.6	0.18 (6.2)	15 (34)	15 (32)	55.3 (122)	
11	20.5 (8.2)	30.5 (12)	0.06 (2.2)	11 (24)	11 (24)	30.2 (11.9)	14.4	0.09 (3.1)	16 (35)	11 (24)	48.5 (107)	
12	23.5 (9.0)	40.6 (16)	0.10 (3.7)	12 (27)	15 (32)	29.7 (11.7)	22.5	0.13 (4.6)	16 (35)	13 (29)	55.8 (123)	
13	19.1 (7.5)	40.6 (16)	0.10 (3.5)	12 (27)	14 (31)	33.0 (13.0)	18.2	0.12 (4.3)	16 (35)	13 (28)	54.9 (121)	
14	21.1 (8.3)	50.8 (20)	0.17 (6.0)	14 (31)	19 (42)	32.5 (12.8)	29.1	0.19 (6.8)	16 (35)	16 (35)	64.9 (143)	
15	12.2 (4.9)	30.5 (12)	0.04 (1.3)	7.7 (17)	10 (22)	36.8 (14.5)	8.8	0.07 (2.5)	49 (107)	10 (22)	76.2 (168)	
16	17.5 (6.7)	30.5 (12)	0.05 (1.8)	7.7 (17)	10 (23)	36.1 (14.2)	13.5	0.10 (3.7)	49 (107)	11 (25)	76.5 (172)	
17	12.4 (4.9)	30.5 (12)	0.04 (1.3)	7.7 (17)	10 (22)	37.8 (14.9)	11.2	0.09 (3.3)	48 (106)	11 (24)	76.7 (169)	
18	16.3 (6.6)	30.5 (12)	0.05 (1.7)	7.7 (17)	10 (23)	37.3 (14.7)	16.3	0.13 (4.7)	48 (106)	13 (28)	78.9 (174)	

(+) Defined on Figure III-3 \* See Table II-14 for definition of configurations

TABLE III-3 PARAMETERS FOR PARTIAL ACQUISITION DEVICES, PARALLEL TANKS

* No	OXIDIZER TANKS						FUEL TANKS						Total Wt. Penalty, kg (lb <sub>m</sub> )
	Height, cm (in.)	Radius, cm (in.)	Volume m <sup>3</sup> (ft <sup>3</sup> )	Wt. of Res- iduals, kg (lb <sub>m</sub> )	Device Weight, kg (lb <sub>m</sub> )	Radius, cm (in.)	Height, cm (in.)	Volume m <sup>3</sup> (ft <sup>3</sup> )	Wt. of Res- iduals, kg (lb <sub>m</sub> )	Device Weight, kg (lb <sub>m</sub> )			
19	13.0 (5.1)	25.4 (10.0)	0.03 (0.9)	9.7 (21)	9.3 (20)	30.5 (12.0)	12.7 (5.0)	0.04 (1.3)	3.6 (8)	12 (26)	34.6 (76.2)		
20	15.7 (6.2)	30.5 (12.0)	0.05 (1.6)	11 (23)	13 (28)	30.5 (12.0)	18.8 (7.4)	0.05 (1.9)	3.7 (8)	13 (30)	40.5 (89.2)		
21	18.8 (7.4)	45.7 (18.0)	0.12 (4.4)	13 (29)	24 (53)	45.7 (18.0)	18.0 (7.1)	0.12 (4.2)	4.5 (10)	24 (53)	65.6 (144.6)		
22	19.8 (7.8)	61.0 (24.0)	0.23 (8.2)	15 (34)	38 (84)	61.0 (24.0)	18.8 (7.4)	0.22 (7.8)	5.4 (12)	38 (83)	96.6 (213.0)		
23	15.5 (6.1)	25.4 (10.0)	0.03 (1.1)	9.9 (22)	9.7 (21)	30.5 (12.0)	11.9 (4.7)	0.03 (1.2)	3.6 (8)	12 (26)	35.0 (77.2)		
24	19.6 (7.7)	30.5 (12.0)	0.06 (2.0)	11 (24)	14 (30)	30.5 (12.0)	21.3 (8.4)	0.05 (2.2)	3.8 (8)	14 (31)	42.2 (93.0)		
25	15.7 (6.2)	45.7 (18.0)	0.10 (3.6)	13 (28)	23 (51)	45.7 (18.0)	15.2 (6.0)	0.10 (3.5)	4.5 (10)	23 (51)	63.4 (139.8)		
26	19.6 (7.7)	61.0 (24.0)	0.23 (8.1)	15 (34)	38 (84)	61.0 (24.0)	17.3 (6.8)	0.20 (7.1)	5.4 (12)	37 (82)	95.8 (211.2)		

\* See Table II-14 for definition of configurations

### C. TOTAL ACQUISITION DEVICES

Total acquisition is another general category of surface tension propellant management devices. The device is configured such that it is always in contact with the bulk propellant regardless of its orientation. The device forms a flow passage from the bulk propellant to the tank outlet, so that gas-free propellant can always be supplied to the engine. This concept is not dependent upon settling, so the device will provide more flexibility and capability than is required for the LTPS application. Total acquisition devices are well suited to applications such as attitude control systems, where propellant must continue to be supplied as the maneuvers are performed. Total acquisition devices have been flight-proven; the Intelsat V communication satellite being the one most recently launched (Ref. 14). The Space Shuttle Reaction Control System (RCS) also uses a total acquisition device (Ref. 15).

#### 1) Total Acquisition Device Concept

The concept selected for the LTPS application uses a simple channel configuration. For the ellipsoidal tank four channels are mounted on the tank wall as shown in Figure III-4. The channels are manifolded at the outlet and terminated slightly below the initial ullage level. For the toroidal tank, the channels are configured as shown in Figure III-5. The devices will be submerged during launch so that it will not be vulnerable to the associated acceleration, thermal, and vibration environments.

The flow area of the channels, screen area, and screen mesh were selected so that liquid would be retained throughout the mission, with the final draining of the tank presenting the worst case condition. At that point a hydrostatic pressure differential acts along the length of the channels and the pressure differential due to flow through the screen continues to increase due to the decreasing area of screen within the settled liquid. Dynamic head and friction have smaller contributions to the total pressure differential acting across the screen. The channels would be filled with liquid when the tank is loaded and must remain free of gas until reaching very small residuals (0.5 percent of the load or less). When the pressure differential across the screen due to flow and acceleration reaches the retention capability of the screen, gas-free expulsion of propellant will no longer be possible. A

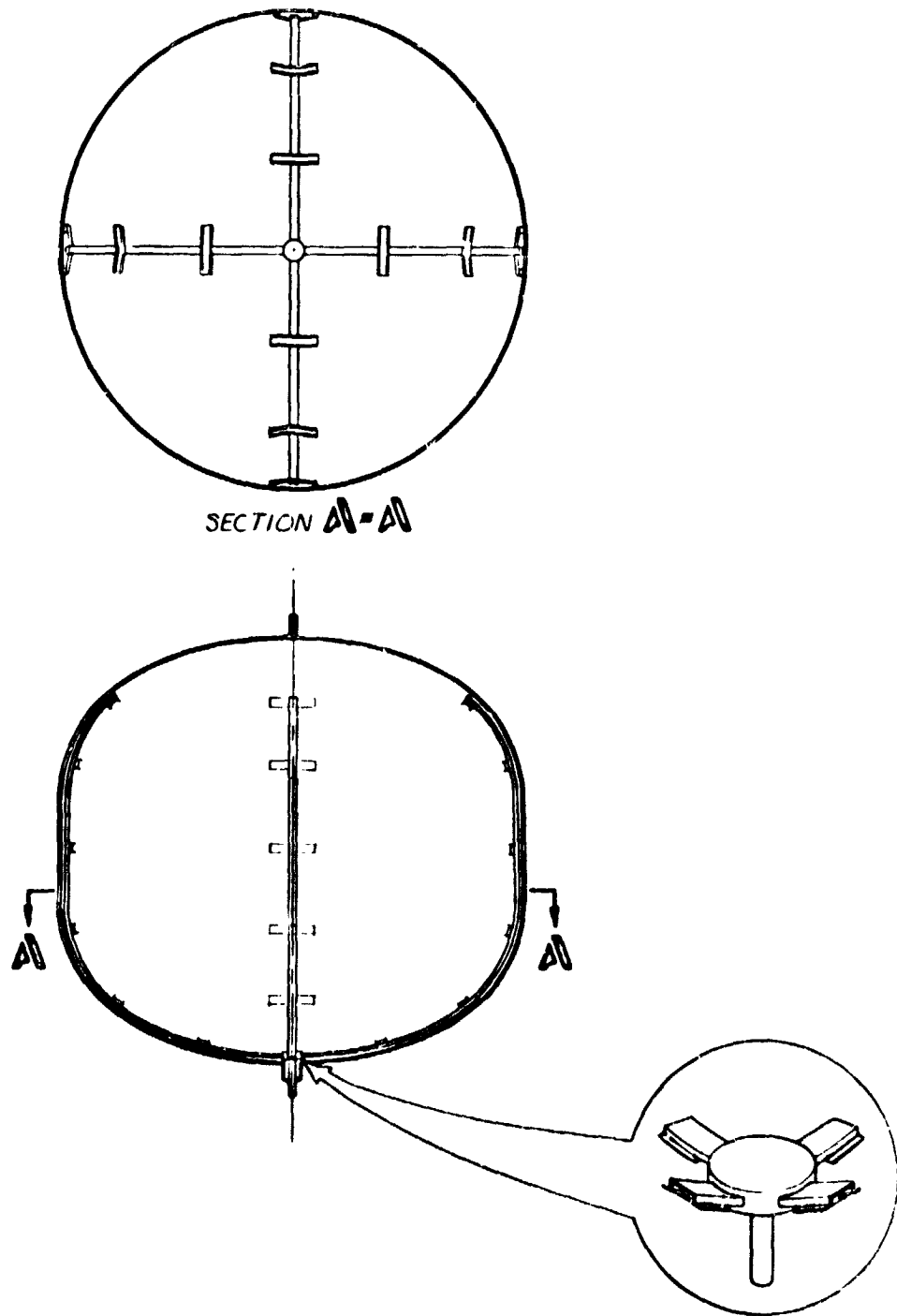


FIGURE III-4 TOTAL ACQUISITION DEVICE FOR ELLIPSOIDAL TANK

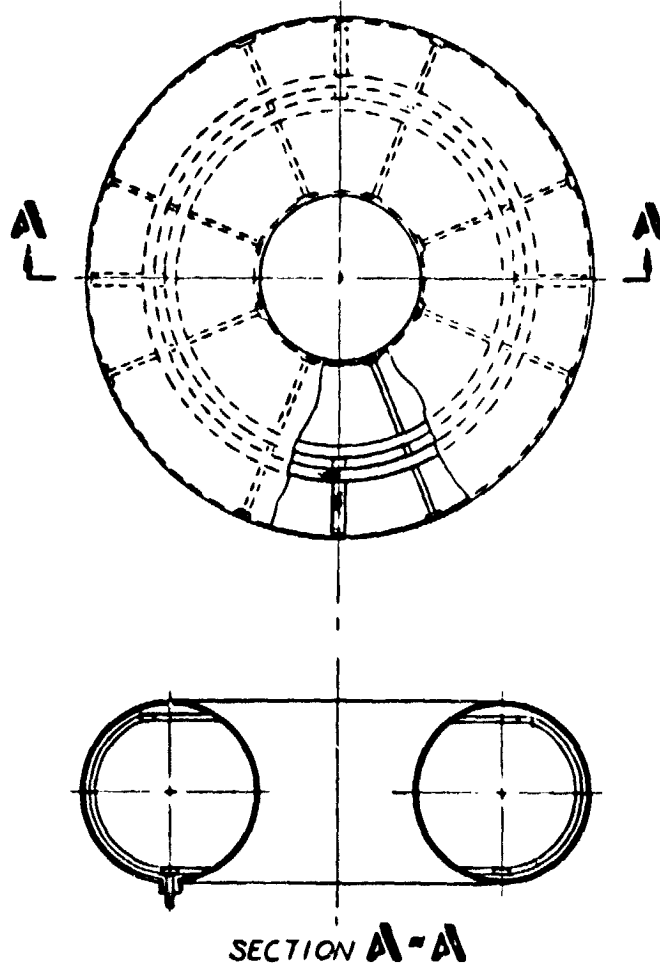
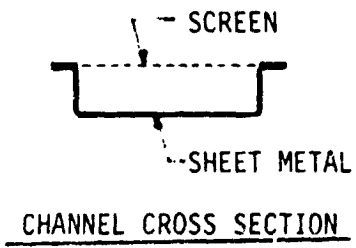


FIGURE III-5 TOTAL ACQUISITION DEVICE FOR TOROIDAL TANK

very-fine mesh screen was selected: 325 x 2300 mesh Dutch twill screen. Increasing the retention capability of the screen increases the performance of this device and the 325 x 2300 screen is a practical limit for the largest possible retention capability.

For the parallel tanks with the 4450 N engine (configurations 23 through 26) the hydrostatic pressure differentials alone exceeded the screen retention capability so gas-free expulsion at low fill levels would not be possible. Therefore, total acquisition was not considered to be feasible for those configurations. Methods of overcoming this problem, such as shortened channels, multiple screen layers or compartmenting the tank were not considered appropriate, due to their impact on device weight and complexity, for this application. For all the other configurations the total acquisition device was considered to be applicable and feasible.

The channels of the device must be thermally isolated from the tank walls, but must also be adequately supported. Thermal isolation is required to prevent boiling of the liquid within the channels. Vaporization of liquid at the screen surface can be accommodated, but boiling puts vapor into the channels, which is not acceptable. Potential designs for the tank support structure were evaluated so that their mass could be estimated.

## 2. Weight Penalty for Total Acquisition

The weight penalty consisted of the device and the propellant residuals. The mass of the device was calculated based on the preliminary design prepared for each configuration. The cross-section of the channel was selected to provide adequate flow area and screen area. The width of the channel, plus the manifold where the channels join at the tank outlet, determined the area of screen in contact with the bulk propellant as it drained. A channel width (and therefore screen area) was selected which prevented gas ingestion into the channels until the bulk propellant was drained to a level just touching the channels. The channel internal flow area was less critical, so a minimum practical channel thickness of 1.3 cm (0.5 in) was used for all the devices. This thickness, in conjunction with the selected channel width, gave a flow area that was more than adequate. The weight of the device was calculated from the channel dimensions and the structural configuration. Once gas



enters the channels, gas-free expulsion of propellant can no longer be guaranteed so the residuals consisted of the propellant within the channels and the propellant puddle left below the device. The pertinent parameters are summarized in Table III-4.

As the thrust and flowrate increased, the size of the device increased and the residuals were also increased, with the mass of the residuals increasing at a much greater rate than the device mass. Doubling the number of devices increased the weight penalty for parallel tanks, even though the flowrate per tank was halved.

#### D. SUMMARY OF WEIGHT PENALTIES

The weight penalties resulting from this analysis are summarized for the three propellant management techniques in Table III-5. The propulsive settling technique usually gave the largest weight penalty, although there were some exceptions with the parallel tank concepts. There was an insignificant difference due to whether the primary propellants or  $N_2O_4/MMH$  were used in the auxiliary propulsion system. The weight penalty for propulsive settling was mostly due to the draining residual. There are schemes for reducing the draining residual but they were not considered at this point in the evaluation. The approach was based on an available auxiliary propulsion system that did not add to the weight penalty. Only if this is true can the propulsive settling technique be competitive with the two surface tension device concepts.

The partial acquisition system was the lightest weight propellant management system, with the exception of configurations 1 and 2. The weight was primarily a function of the reservoir volume, which was highly dependent upon the loss due to vaporization.

The total acquisition devices usually ranged from 1.5 to 2 times the weight of the partial acquisition device. Flowrate and tank configuration were the primary factors influencing the device weight.

TABLE III-4 - PARAMETERS FOR TOTAL ACQUISITION DEVICES

* CONFIG.	Series Tanks - Ellipsoidal Tank				Series Tanks - Toroidal Tank				Total Weight Penalty, kg (lb <sub>m</sub> )
	Channel Width, cm (in.)	Channel Thickness, cm (in.)	Mass Of Residuals, kg (lb <sub>m</sub> )	Mass Of Device, kg (lb <sub>m</sub> )	Channel Width, cm (in.)	Channel Thickness, cm (in.)	Mass Of Residuals, kg (lb <sub>m</sub> )	Mass Of Device, kg (lb <sub>m</sub> )	
1	2.5 (1.0)	1.3 (0.5)	0.9 (2)	20 (44)	2.5 (1.0)	1.3 (0.5)	17 (38)	15 (34)	53.5 (118)
2	2.5 (1.0)	↓	0.9 (2)	20 (44)	2.5 (1.0)	↓	17 (38)	15 (34)	53.5 (118)
3	5.1 (2.0)	↓	1.4 (3)	24 (53)	5.1 (2.0)	↓	27 (60)	20 (44)	72.6 (160)
4	5.1 (2.0)	↓	1.4 (3)	24 (53)	5.1 (2.0)	↓	27 (60)	20 (44)	72.6 (160)
5	10 (4.0)	↓	2.7 (6)	33 (72)	10 (4.0)	↓	46 (102)	29 (64)	111 (244)
6	10 (4.0)	↓	2.7 (6)	32 (71)	10 (4.0)	↓	46 (102)	29 (64)	110 (243)
7	5.1 (2.0)	↓	20 (45)	20 (44)	5.1 (2.0)	↓	10 (22)	20 (44)	70.3 (155)
8	5.1 (2.0)	↓	20 (44)	20 (44)	5.1 (2.0)	↓	10 (22)	20 (44)	69.9 (154)
9	5.1 (2.0)	↓	20 (45)	20 (45)	5.1 (2.0)	↓	10 (22)	20 (44)	70.8 (156)
10	5.1 (2.0)	↓	20 (45)	20 (45)	5.1 (2.0)	↓	10 (21)	20 (43)	69.9 (154)
11	10 (4.0)	↓	34 (74)	26 (58)	10 (4.0)	↓	17 (38)	29 (64)	106 (234)
12	↓	↓	34 (74)	26 (58)	↓	↓	17 (38)	29 (64)	106 (234)
13	↓	↓	35 (78)	27 (59)	↓	↓	17 (37)	29 (63)	108 (237)
14	↓	↓	35 (77)	27 (59)	↓	↓	17 (37)	29 (63)	107 (236)
15	↓	↓	20 (43)	22 (49)	↓	↓	46 (101)	29 (63)	116 (256)
16	↓	↓	20 (43)	22 (49)	↓	↓	46 (101)	29 (64)	117 (257)
17	↓	↓	20 (43)	22 (49)	↓	↓	46 (101)	29 (63)	116 (256)
18	↓	↓	20 (43)	22 (49)	↓	↓	46 (101)	29 (63)	116 (256)
	PARALLEL TANKS - OXIDIZER				PARALLEL TANKS - FUEL				
19	5.1 (2.0)	↓	32 (70)	39 (86)	5.1 (2.0)	↓	11 (24)	38 (84)	120 (264)
20	5.1 (2.0)	↓	31 (68)	39 (86)	5.1 (2.0)	↓	11 (24)	38 (84)	119 (262)
21	5.1 (2.0)	↓	33 (72)	40 (88)	5.1 (2.0)	↓	12 (26)	39 (86)	123 (272)
22	5.1 (2.0)	↓	33 (72)	40 (88)	5.1 (2.0)	↓	12 (26)	39 (86)	123 (272)
23	NOT FEASIBLE	↓							
24	↓	↓							
25	↓	↓							
26	↓	↓							

\* See Table II-14 for definition of configurations

TABLE III-5 WEIGHT PENALTY FOR PROPELLANT MANAGEMENT CONCEPTS

CONFIG. *	WEIGHT PENALTY, kg ( $1b_m$ )			
	SETTLING		PARTIAL ACQUISITION	TOTAL ACQUISITION
	$N_2O_4$ /MMH	PRIMARY PROPELLANTS		
1	76 (167)	75 (166)	71 (156)	54 (118)
2	74 (154)	74 (163)	77 (169)	54 (118)
3	181 (398)	180 (397)	72 (158)	73 (160)
4	195 (429)	194 (427)	79 (175)	73 (160)
5	269 (592)	268 (590)	78 (171)	111 (244)
6	261 (576)	260 (573)	85 (188)	110 (243)
7	121 (267)	121 (267)	44 ( 96)	70 (155)
8	123 (271)	122 (270)	48 (105)	70 (154)
9	113 (250)	113 (249)	49 (109)	71 (156)
10	116 (256)	116 (255)	55 (122)	70 (154)
11	166 (366)	166 (366)	49 (107)	106 (234)
12	157 (346)	156 (345)	56 (123)	106 (234)
13	149 (329)	149 (328)	55 (121)	108 (237)
14	152 (336)	152 (335)	65 (143)	107 (236)
15	285 (629)	285 (629)	76 (168)	116 (256)
16	289 (637)	288 (636)	78 (172)	117 (257)
17	278 (613)	278 (613)	77 (169)	116 (256)
18	288 (634)	287 (633)	79 (174)	116 (256)
19	59 (131)	59 (130)	34 ( 76)	120 (264)
20	61 (135)	61 (134)	40 ( 89)	119 (262)
21	54 (119)	54 (119)	66 (145)	123 (272)
22	58 (127)	57 (126)	97 (213)	123 (272)
23	70 (155)	70 (154)	35 ( 77)	Not Feasible
24	74 (163)	73 (161)	42 ( 93)	↓
25	68 (149)	67 (147)	64 (140)	
26	69 (153)	69 (151)	96 (211)	

\* See Table II-14 for definition of configurations

While all of these propellant management techniques have been used in some form on flight proven systems, only the propulsive settling technique has been used with a cryogenic system.

While the technology for fine-mesh screen devices continues to grow and the number of flight-proven systems continues to increase, their application to very large cryogenic systems still requires some development. The technology deficiencies are discussed in detail in Chapter VII.

#### IV. REFINED LTPS CONFIGURATIONS

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##### A. PROPELLANT DENSITIES

An analysis was performed to account for changes in cryogenic propellant densities due to boiling of the propellant prior to and during launch. For the initial sizing in Section II-K the propellant densities were considered at saturation conditions and 165 kPa (24 psi). Since the heat leak to the LTPS during the ground hold time and launch is large enough to produce boiling in the cryogenics, the decrease in density must be integrated into the system sizing. The decrease in the average density caused by boiling would require an increase in tank volume which, in turn, would increase tank length. The analysis in Appendix F predicted densities slightly lower than comparable Centaur data. This was to be expected since in this evaluation it was assumed that all heat leaks create vaporization only, which is not true under actual conditions.

Densities resulting from the analysis are shown in Table IV-1. Comparing the first 18 configurations, all tandem/toroidal tank arrangements, the SOFI Systems have less density change from saturation density due to a much lower value of  $K/\Delta X$  (thermal conductivity divided by insulation thickness). The lower value is because on-ground K for SOFI is about half of the value for MLI and the on-orbit requirements demand a thick layer of insulation because of the poorer K for SOFI on-orbit than MLI. However, for the parallel tanks, configurations 19 through 26, densities are lower than the first 18 systems due to a larger surface area to volume ratio and generally longer tanks.

These values of propellant density were used in the final evaluation of configurations 1 through 26.

##### B. RESIZING OF SELECTED SYSTEMS

Using the predicted propellant management weight penalties, the inputs to PROP were modified to reflect an accurate assessment of the amount of propellant trapped in the tanks at burnout and any additional hardware that would be required. Each configuration was sized with all three propellant management techniques. The resulting LTPS masses are shown in Table IV-2.

TABLE IV-1 TANKING DENSITIES PREDICTED BY ANALYSIS

CONFIG. #	FUEL DENSITY		OXIDIZER DENSITY	
	kg/m <sup>3</sup>	lb <sub>m</sub> /ft <sup>3</sup>	kg/m <sup>3</sup>	lb <sub>m</sub> /ft <sup>3</sup>
1 (MLI)	67.25	4.198	1106	69.04
2 (MLI)	67.28	4.200	1109	69.22
3 (MLI)	67.12	4.190	1107	69.12
4 (MLI)	67.20	4.195	1108	69.14
5 (MLI)	67.11	4.189	1107	69.12
6 (MLI)	67.19	4.194	1108	69.16
7 (MLI)	409.5	25.56	1106	69.01
8 (MLI)	409.6	25.57	1106	69.04
9 (SOFI)	412.7	25.76	1114	69.51
10 (SOFI)	412.8	25.77	1114	69.54
11 (MLI)	409.3	25.55	1105	68.99
12 (MLI)	409.5	25.56	1106	69.01
13 (SOFI)	412.7	25.76	1114	69.52
14 (SOFI)	412.8	25.77	1114	69.53
15 (MLI)	805.7	50.30	1106	69.02
16 (MLI)	↓	↓	1107	69.13
17 (SOFI)	↓	↓	1114	69.54
18 (SOFI)	↓	↓	1114	69.56
19 (MLI)	404.3	25.24	1098	68.55
20 (MLI)	404.8	25.27	1099	68.61
21 (SOFI)	410.9	25.65	1110	69.26
22 (SOFI)	411.2	25.67	1110	69.30
23 (MLI)	404.5	25.25	1098	68.55
24 (MLI)	404.8	25.27	1099	68.61
25 (SOFI)	410.9	25.65	1110	69.26
26 (SOFI)	411.1	25.66	1110	69.29

TABLE IV-2 LTPS MASSES, kg

CONF.	SETTLING	TOTAL ACQUISITION	PARTIAL ACQUISITION
1	22603	22579	22597
2	22096	22074	22097
3	21296	22177	21176
4	20464	20340	20347
5	20931	20753	20737
6	20249	20077	20070
7	23042	22991	22964
8	22266	22212	22190
9	23850	23805	23783
10	23361	23312	23292
11	22620	22555	22501
12	22006	21950	21904
13	23315	23270	23221
14	23020	22966	22927
15	23247	23075	23017
16	22651	22476	22425
17	23919	23752	23700
18	23625	23447	23402
19(7)*	22983	23044	22958
20(8)	22204	22262	22183
21(9)	23871	23939	23881
22(10)	23407	23471	23444
23(11)	22525	NOT	22489
24(12)	21923	FEASIBLE	21891
25(13)	23298	↓	23292
26(14)	23022	↓	23047

1 kg = 2.205 lb<sub>m</sub>

\* Numbers in parentheses represent corresponding systems with different tank arrangements

Propellant settling, using the main propellants or the ACS propellants, was considered as one group since the weight penalty due to either system differed by a maximum of approximately 1 kg. For the 8 parallel tanks cases, the MLI-covered tanks favor partial acquisition while the SOFI-covered tanks favor propellant settling. This is due to an increase in the size of the device when SOFI is used. This is because of an increase in boiloff which must be accommodated in the device. In the column headed "CONFIG." in the table, the numbers in parentheses are the LTPS configurations that have the same propellants, thrust level, burn strategy, and insulation concept but differing in tank configuration. For most of the minimum length configurations, the partial acquisition method was the system with the least mass. The mass available for the LSS (payload) is shown in Table IV-3.

The resulting LTPS lengths for each of the 26 configurations are shown in Table IV-4. The propellant management technique used on a particular configuration did not change the length of the system by more than 3 cm for any of the selected cases. Propellant settling always created the longest LTPS since the weight penalty was due to additional propellant, which is less dense than the additional metal parts that comprise a large portion of the weight penalties for the surface tension devices. Thus, no propellant management method produced a clear length advantage.

These final results for the minimum length systems will be compared to the maximum performance results at the end of the next section.



TABLE IV-3 LSS PAYLOAD MASS, kg

CONFIG- URATION	PROPELLANT SETTLING	TOTAL ACQUISITION	PARTIAL ACQUISITION
1	4613	4636	4617
2	5120	5142	5118
3	5920	6039	6039
4	6751	6876	6869
5	6285	6463	6479
6	6967	7138	7146
7	4173	4225	4252
8	4950	5003	5026
9	3365	3411	3432
10	3854	3904	3923
11	4595	4661	4714
12	5209	5266	5312
13	3900	3945	3994
14	4196	4250	4289
15	3968	4140	4199
16	4564	4739	4790
17	3297	3463	3515
18	3591	3769	3813
19(7)*	4232	4172	4257
20(8)	5012	4954	5033
21(9)	3345	3276	3335
22(10)	3809	3744	3772
23(11)	4691	NOT	4727
24(12)	5293	FEASIBLE	5324
25(13)	3917	↓	3923
26(14)	4193	↓	4169

1 kg = 2.205 lb<sub>m</sub>

\* Numbers in parentheses represent corresponding systems with different tank arrangements

TABLE IV-4 LTPS LENGTH, m

CONFIGURATION	SETTLING	TOTAL ACQUISITION	PARTIAL ACQUISITION
1	5.98	5.96	5.97
2	5.89	5.87	5.88
3	5.65	5.62	5.62
4	5.49	5.47	5.47
5	5.55	5.52	5.52
6	5.43	5.40	5.40
7	3.78	3.78	3.78
8	3.73	3.72	3.72
9	3.89	3.88	3.88
10	3.87	3.86	3.86
11	3.86	3.86	3.86
12	3.84	3.84	3.84
13	3.89	3.89	3.89
14	3.89	3.88	3.88
15	3.39	3.38	3.37
16	3.35	3.33	3.33
17	3.43	3.42	3.41
18	3.41	3.40	3.40
19(7)*	4.34	4.34	4.34
20(8)	4.25	4.24	4.24
21(9)	4.51	4.50	4.50
22(10)	4.47	4.47	4.46
23(11)	4.43	NOT	4.42
24(12)	4.35	FEASIBLE	4.35
25(13)	4.57	↓	4.56
26(14)	4.56	↓	4.55

1 m = 3.281 ft

\* Numbers in parentheses represent corresponding systems with different tank arrangements

## V. IMPROVED LTPS CONCEPTS

In this section, three promising LTPS concepts, one for each propellant combination, were further developed and optimized. Particular attention was paid to simplified propellant acquisition and further thermal insulation system optimization. The goal was to increase the mass available for the LSS.

### A. SYSTEM DESIGN

Due to minimum stage mass requirements of this section, cylindrical tanks with ellipsoidal domes and/or ellipsoidal tanks were paired in a conventional tandem arrangement as shown in Figure V-1. All three propellant combinations were sized using 2225 N (500lb<sub>f</sub>) thrust, 8 perigee burns, and MLI covered tanks. The initial system characteristics were calculated with PROP using a similar approach to that used in Section II.

### B. PROPELLANT INVENTORY

For these maximum performance configurations the only part of the propellant inventory that is defined differently from Section II-F is the propellant trapped in the line. The amount of trapped propellant is estimated by using the tank arrangements shown in Figure V-1. As in the previous calculations of line trapped, the line diameters were sized using a maximum pressure drop of 1 psid. The length of line isolated between the aft tank and the engine at the end of each burn was 0.3m. From the forward tank to the engine, the feedline length was 50% of the aft tank perimeter plus 0.45m. The effect of valves, contractions, bends, and line length were all included in the pressure drop calculation. The following is a table of the feedline diameters and the amount of propellant trapped in the line at the end of each burn.

Propellant Combination	Feedline Diameters, cm	Line Trapped Per Burn, kg
LO <sub>2</sub> /LH <sub>2</sub>	1.0/1.8	0.03/0.09
LO <sub>2</sub> /LCH <sub>4</sub>	2.0/1.3	1.6/0.02
LO <sub>2</sub> /RP-1	2.0/1.3	1.5/0.03

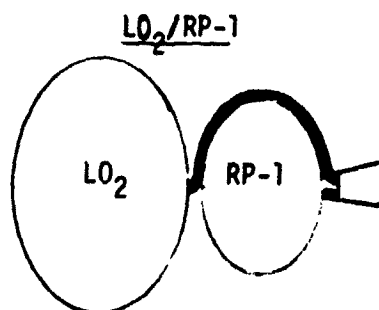
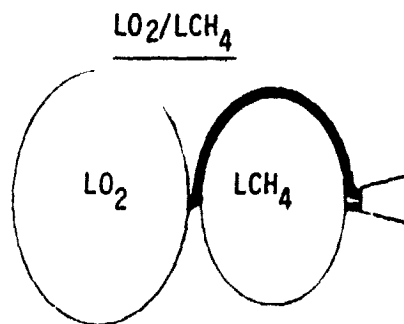
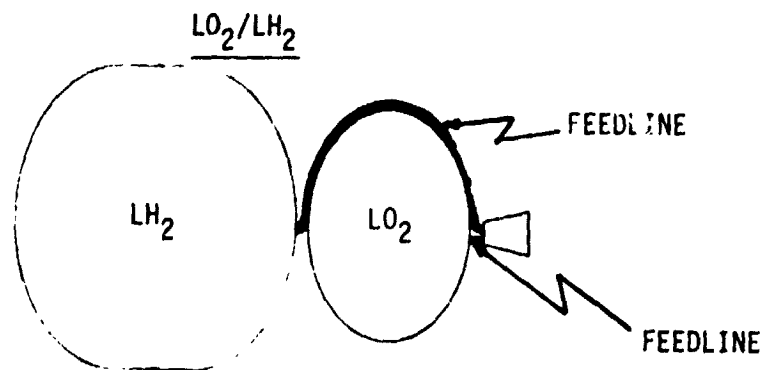


FIGURE V-1 TANK ARRANGEMENTS FOR MAXIMUM PERFORMANCE CONFIGURATIONS.

### C. INSULATION OPTIMIZATION

The optimized insulation thicknesses for the three propellant combinations were calculated by repetitive use of the computer program PROP. Each curve in Figure V-2 through V-4 was generated by inputting different insulation thicknesses to PROP then tabulating the mass of the propellant, plus its tank and insulation. As the insulation thickness was varied on one tank it was maintained constant on the other. The minimum point on the curve corresponds to the minimum tank system mass of each respective configuration. The insulation thickness that produced this minimum system mass was the thickness used to size the vehicle. As can be seen from the three sets of curves, the optimized insulation thickness for the LO<sub>2</sub> tanks were approximately 2.2 cm. A list of the optimized insulation thickness values used for the three maximum performance configurations is shown below.

#### OPTIMUM INSULATION THICKNESS, m

LO <sub>2</sub> /LH <sub>2</sub>	0.023/0.025
LO <sub>2</sub> /LCH <sub>4</sub>	0.022/0.018
LO <sub>2</sub> /RP-1	0.022/no insulation

It can be seen from Figures V-2, 3 and 4 that the curves are not very sensitive to insulation thickness around the optimum mass. A change of 0.5 cm, a change of approximately 15 percent, creates a change in system mass of at most 0.2 percent.

### D. PROPELLANT DENSITIES

The analysis in Appendix F was used to calculate on-ground tanking densities. The resulting propellant densities shown below were used to size the tanks:

Optimum Insulation Thickness, m  
 $\text{LO}_2$  : 0.023  
 $\text{LH}_2$  : 0.025

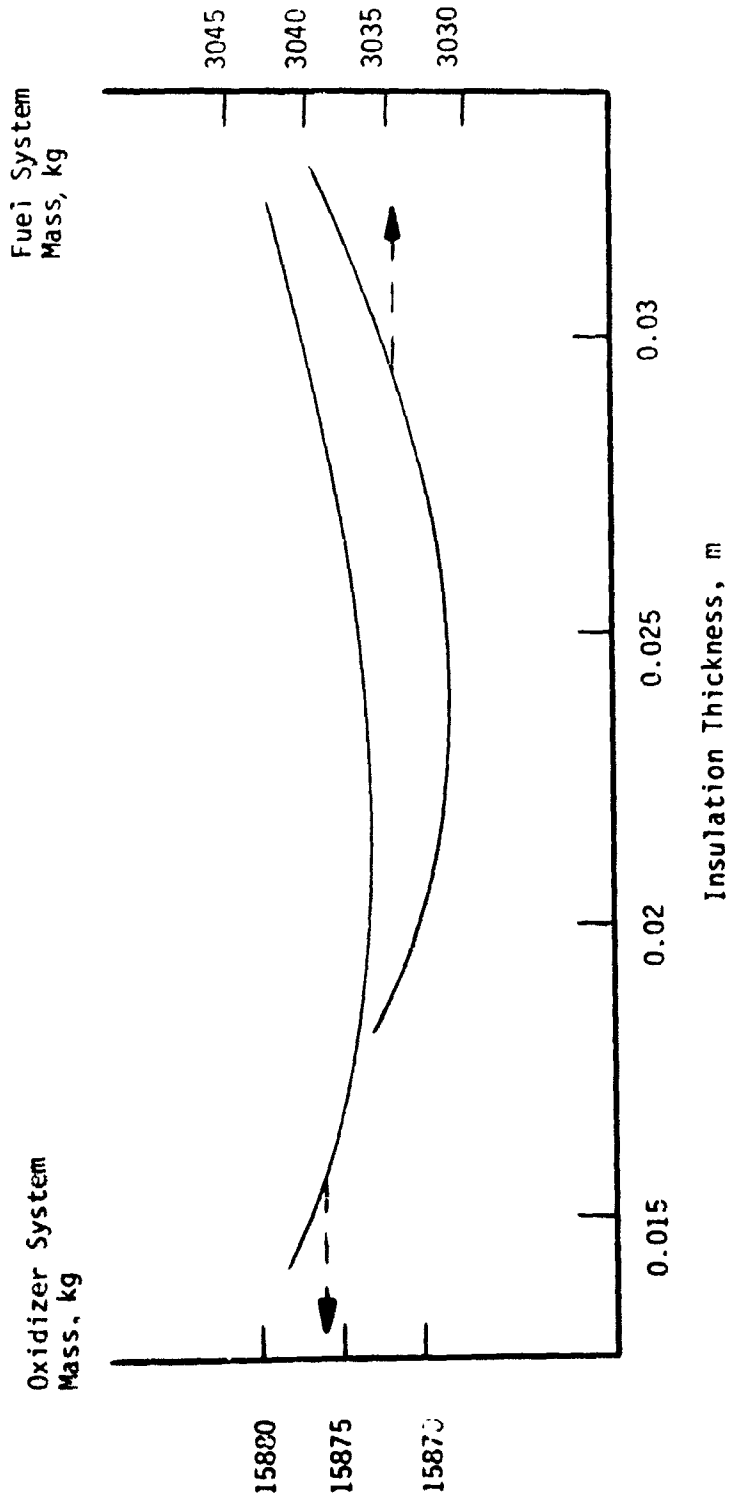


FIGURE V-2 SYSTEM MASS AS A FUNCTION OF INSULATION THICKNESS FOR  $\text{LO}_2/\text{LH}_2$ , HLI SYSTEMS

Optimum Insulation Thickness, m  
 $LO_2$  : 0.022  
 $LCH_4$  : 0.018

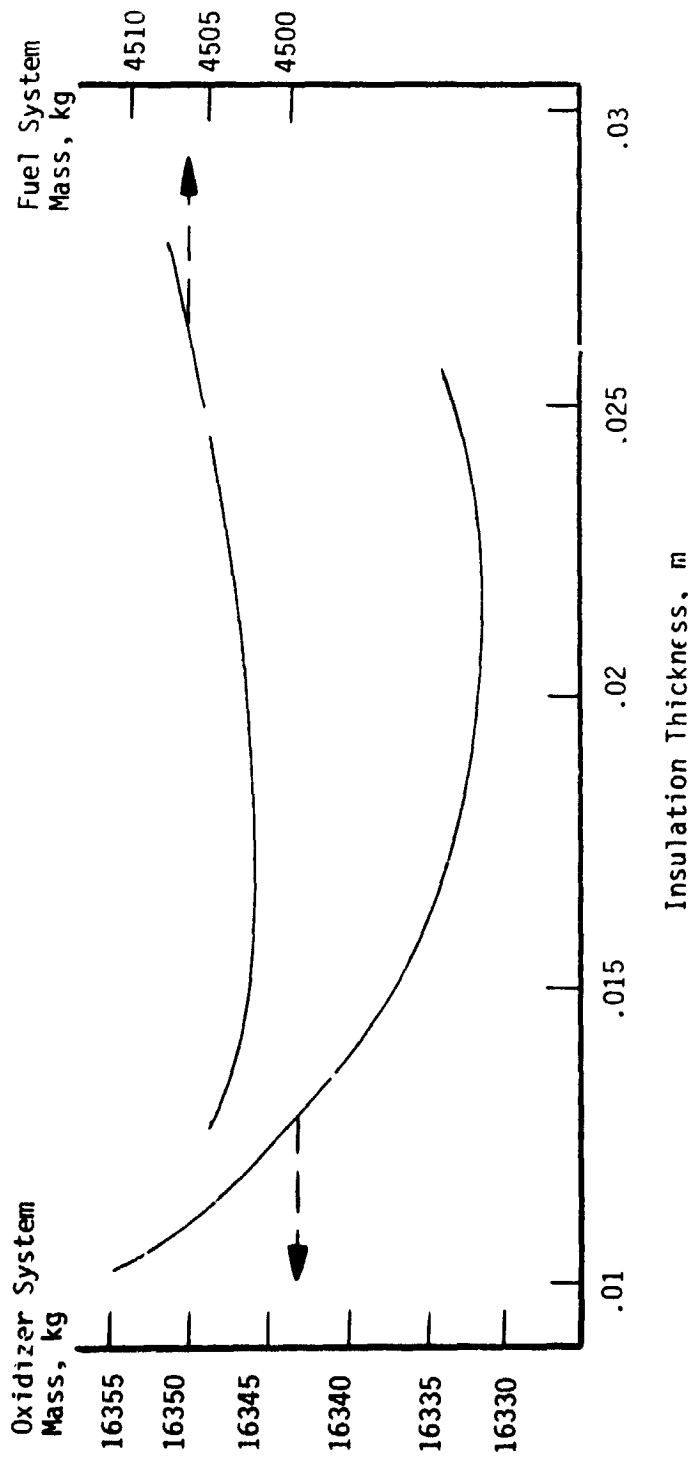


FIGURE V-3 SYSTEM MASSES AS A FUNCTION OF INSULATION THICKNESS FOR  $LO_2/LCH_4$ , MLI SYSTEMS

Optimum Insulation Thickness, m  
LO<sub>2</sub> : 0.022

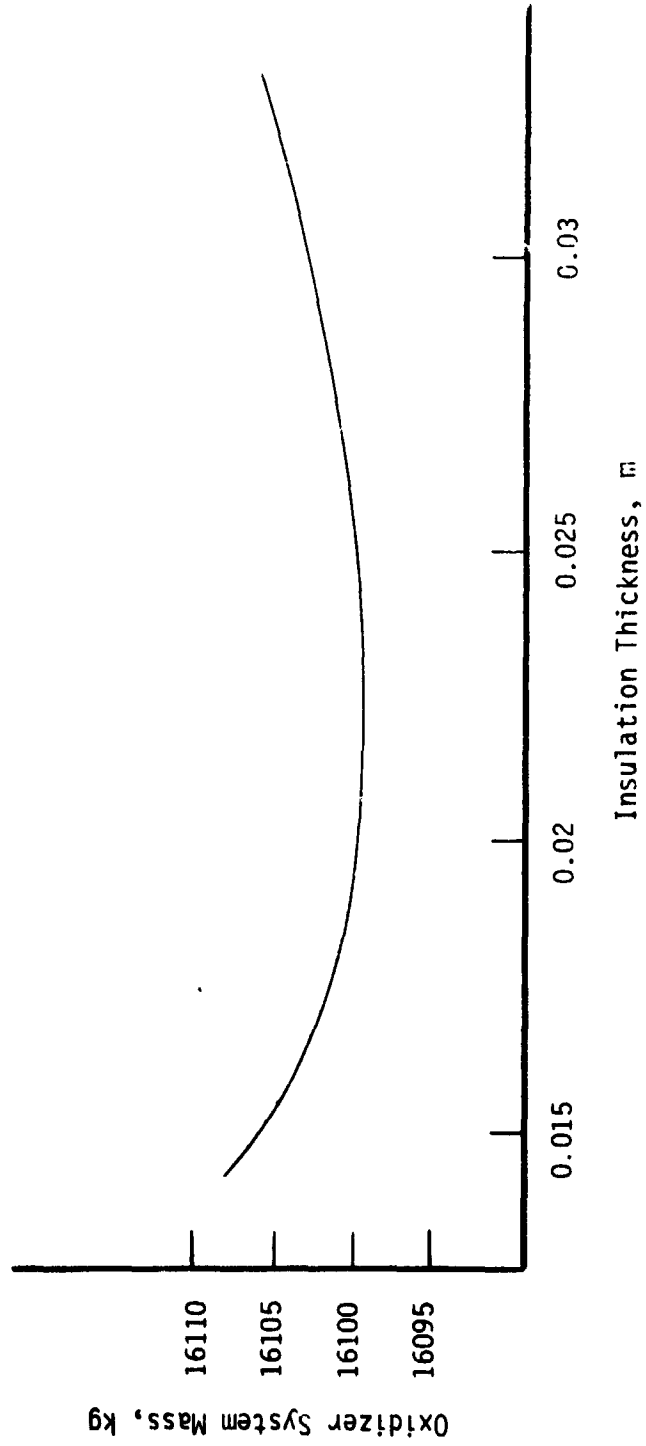


FIGURE V-4 OXIDIZER SYSTEM MASS AS A FUNCTION OF INSULATION THICKNESS FOR LO<sub>2</sub>/RP-1, MLI SYSTEM



		Oxidizer, kg/m <sup>3</sup> (lb <sub>m</sub> /ft <sup>3</sup> )	Fuel, kg/m <sup>3</sup> (lb <sub>m</sub> /ft <sup>3</sup> )
MLI 8 BURNS, 2225 N THRUST	LO <sub>2</sub> /LH <sub>2</sub>	1102 (68.80)	67.1 (4.19)
	LO <sub>2</sub> /LCH <sub>4</sub>	1101 (68.75)	408.4 (25.50)
	LO <sub>2</sub> /RP-1	1101 (68.75)	805.5 (50.3)

These densities are used as inputs to PROP. The tanks will be sized by calculating the maximum volume required to contain the propellant at lift off.

#### E. PROPELLANT MANAGEMENT TECHNIQUE

Propulsive settling was selected as the propellant management technique for the improved LTPS concepts. While the analysis of the concepts presented in the last section showed propulsive settling to be the heaviest of the approaches, improvements were possible. Further evaluation of propulsive settling established that the draining residual, the primary contribution to the weight penalty, could be significantly reduced by incorporation of a small surface-tension propellant management device. With this improvement the propulsive settling technique was established as the simplest and lightest weight method of propellant management.

The primary disadvantage of the fine-mesh screen partial and total acquisition devices was their vulnerability to the effects of heat and mass transfer. The fabrication and structural support of the devices was also a concern for tanks of the size considered in this study. It appears that considerable development will be required before fine-mesh screen systems can be applied to cryogenic systems of the size of the LTPS.

In comparison, the propulsive settling technique is essentially insensitive to thermal environment and tank size. The propellant settling times are scaled from small models, using technology that is fairly well developed. Conservative approaches to estimating the settle time do not significantly increase the weight penalty.

## 1. Propulsive Settling Concept

The draining residual was reduced by adding a bubble filter over the tank outlet. The filter is a simple screen device that delays gas ingestion into the tank outlet until the propellant reaches a small residual volume. Since the only function of the device is to exclude gas from the flow at the end of the last burn, it is not sensitive to the thermal environment as are the other surface tension propellant management devices evaluated in this study.

Some additional factors, neglected previously, were considered in analyzing the propulsive settling concept. One such factor was the gimbaling of the main engine. The center-of-gravity of the LSS payload will not be accurately known before it is deployed. Due to this uncertainty the main engine of the LTPS will be capable of gimbaling over a sufficient range so that the thrust vector will always be able to pass through the center-of-gravity. Gimbal angles as large as 10 degrees may be necessary and this angle will have to be maintained throughout the mission, including terminal drain. With the propellant displaced away from the tank outlet at the gimbal angle, the draining residuals will be increased. The bubble filter will help to maintain propellant feed despite the effect of gimbaling.

The bubble filter was a flat circle of screen, supported by perforated plate and mounted directly over the tank outlet. During terminal drain, suction dip will tend to draw the liquid interface downward toward the filter and gimbaling of the engine will displace the liquid so as to uncover the filter. The retention capability of the screen on the filter acts to prevent this gas that comes into contact with the filter from passing through. The portions of the filter, still submerged in liquid, can sustain liquid expulsion. When the retention capability of the screen can no longer balance the flow loss through the area of liquid in contact with the screen, then gas will begin to penetrate the filter. A filter design that permitted one-half the filter to be exposed to gas before gas began to penetrate the screen was selected. This approach yielded a 25 cm diameter filter using the fine-mesh 325 x 2300 Dutch twill screen. The propellant residual was based on the liquid position with a 10 degree gimbal angle and one-half the filter exposed to gas.

Another factor that was evaluated was engine chilldown. Prior to each main engine burn, propellant would be flowed through the engine, providing thermal conditioning to ensure satisfactory performance at the time of engine start. After settling was complete, chilldown would begin. It was conservatively assumed that the settled orientation would have to be maintained by continuing the settling thrust while chilldown was performed.

The quantity of propellant required for chilldown is dependent upon the initial pump temperature and the temperature, pressure, and flowrate of the propellant. The chilldown time is a function of the flowrate and the final engine temperature. As the flowrate is increased, the chilldown time decreases but the total quantity of propellant increases. There is a trade-off between the quantity of propellant required for chilldown and the quantity of propellant required to maintain settling during chilldown.

Various sources of information were surveyed to establish a realistic value for the chilldown time (e.g., RL-10 engine data, orbit-to-orbit engine studies, and low-thrust engine evaluation). A chilldown period of 50 seconds was selected for this evaluation.

## 2. Weight Penalty for Propellant Management

An auxiliary propulsion system, operating on either earth storable or the primary propellants, was assumed to be available. Our previous analysis has shown that the difference in the weight penalty between using earth storable and primary propellants is negligible. The easier to store earth storables may be preferred for such a system. The prior optimization of the settling acceleration was shown to be of little value since the quantity of propellant required to achieve settling was reasonably small. A thrust of 22N (5 lb<sub>f</sub>) was selected, being representative of a small attitude control thruster. The time required to settle the propellant was increased by 50 seconds for each burn to allow for engine chilldown. Following this approach the quantity of propellant required for propulsive settling was calculated.

The propellant residual was calculated based on the above described bubble filter configuration and a 10 degree gimbal angle at propellant depletion. The weight of the bubble filter was estimated. Each of the contributions to the weight penalty are summarized in Table V-1. Even though this improved propellant management concept was capable of satisfying more stringent requirements than the original concepts presented in Section III, the weight penalty was less.

#### F. PROPELLANT SYSTEM CHARACTERISTICS

The weight penalties predicted in the previous section were used to modify PROP inputs representing trapped and miscellaneous hardware. Only the propellant settling approach described in the previous section was used to size these three maximum performance configurations. The system characteristics are listed in Table V-2 and graphically displayed in Figure V-5. Overall length, for this conventional tandem tank arrangement, was computed by adding both tank lengths (including insulation), 0.15 m clearance between tanks, 0.15 m clearance between the aft tank and engine, plus the engine length.

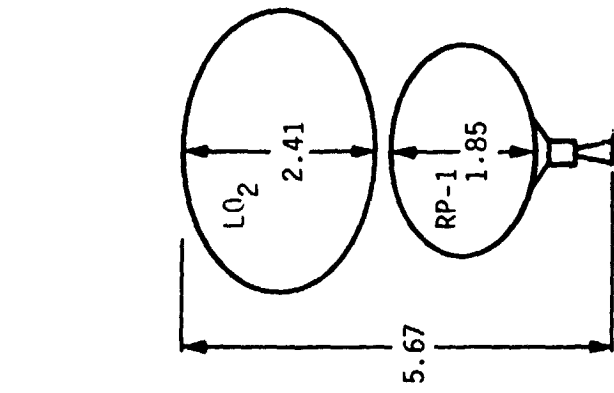
Three systems from the original selection of 26 cases were analyzed using the improved settling approach described in Section V E. The systems chosen were configuration numbers 4, 8 and 20. These were all 2225 N thrust, 8 perigee burn and MLI covered systems (as is the maximum performance configuration). The bubble filters were 25 cm diameter screen covered disks in the ellipsoidal tanks and the toroidal tanks has a ring-shaped screen covered channel connected to a single outlet. A 10-degree gimbal angle was assumed at propellant depletion. The result of this analysis can be seen in Table V-3. This improved settling produces systems lighter than either acquisition method or the settling technique used in Section III. This analysis provided a sampling of the influence of this improved propellant management concept on the weight penalty, but the trend indicates an improved LSS payload capability using this type of screen device.

TABLE V-1 WEIGHT PENALTY FOR PROPELLANT MANAGEMENT

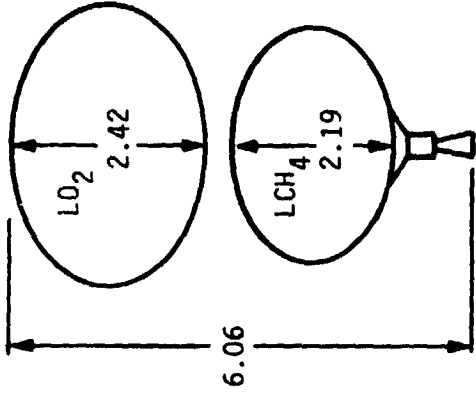
CONFIGURATION	PROPELLANT REQUIRED TO SETTLE, kg(1bm)		PROPELLANT RESIDUAL, kg(1bm)		BUBBLE FILTER WEIGHT, kg(1bm)	TOTAL WEIGHT PENALTY, kg(1bm)
	OXIDIZER	FUEL	OXIDIZER	FUEL		
1 LO <sub>2</sub> /LH <sub>2</sub>	10.5 (23.2)	1.8 (3.9)	14.5 (32.0)	1.8 (3.9)	1.2 (2.6)	29.8 (65.6)
2 LO <sub>2</sub> /LCH <sub>4</sub>	9.6 (21.1)	2.6 (5.7)	14.8 (32.7)	4.5 (10.0)	1.2 (2.6)	32.7 (72.1)
3 LO <sub>2</sub> /RP-1	9.7 (21.4)	3.2 (7.1)	9.5 (20.9)	5.5 (12.1)	1.2 (2.6)	29.1 (64.1)

TABLE V-2 PROPELLANT SYSTEM CHARACTERISTICS FOR CONVENTIONAL TANDEM TANK CONFIGURATIONS																
MLI INSULATION, 8 BURNS										INITIAL VEHICLE MASS: 27216kg						
PROPELLANT	MIXTURE RATIO	I <sub>sp</sub> N·sec (sec)	TOTAL ΔV, m/s	LEO TO GEO hr	PROPELLANT MASSES, kg				TOTAL	TANK			LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg	PAYLOAD MASS, kg	OVERALL LENGTH,* m
					USABLE (AV)	TRAPPED START-S/D LOSSES	BOILOFF LOSSES	VOLUME m <sup>3</sup>		DIAMETER, m	LENGTH, m					
LO <sub>2</sub> /LH <sub>2</sub>	6	4315 (440.0)	4448	19.8	F	2477	77	181	18237	41.8	4.27	3.93	2439	20207	7009	7.80
					O	14859	420	223		14.4	3.39	2.40				
LO <sub>2</sub> /LCH <sub>4</sub>	3.7	3496 (356.5)	4441	18.8	F	4134	136	83	20311	10.9	3.09	2.19	2323	22102	5114	6.06
					O	15295	437	226		14.9	3.42	2.42				
LO <sub>2</sub> /RP-1	3	3270 (333.5)	4439	18.5	F	5025	154	0	20905	6.59	2.61	1.85	2271	22635	4581	5.67
					O	15076	424	225		14.6	3.41	2.41				

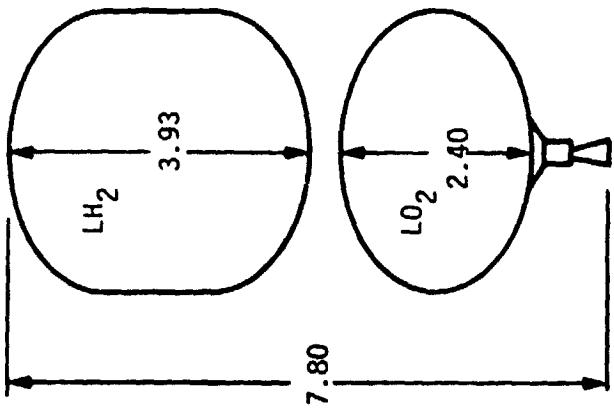
\*Includes Insulation Thickness and Engine Length



3272  
3:1  
22630  
4590



3497  
3.7:1  
22100  
5120



4316  
6:1  
20210  
7010

ISP, N-sec/kg  
Mixture Ratio  
LTPS Mass, kg  
LSS Mass, kg

Note:  
All Dimensions in meters  
Maximum Tank Diameter = 4.27m  
Engine Length = 1.07m

FIGURE V-5 LTPS MAXIMUM PERFORMANCE CONFIGURATIONS

TABLE V-3 LSS PAYLOAD MASS,

( 1 kg = 2.21 lb<sub>m</sub> )

CONFIG- URATION	PROPELLANT SETTLING	TOTAL ACQUISITION	PARTIAL ACQUISITION	PROPELLANT SETTLING WITH BUBBLE FILTER
1	4613	4636	4617	
2	5120	5142	5118	
3	5920	6039	6039	
4	6751	6876	6869	6931
5	6285	6463	6479	
6	6967	7138	7146	
7	4173	4225	4252	
8	4950	5003	5026	5031
9	3365	3411	3432	
10	3854	3904	3923	
11	4595	4661	4714	
12	5209	5266	5312	
13	3900	3945	3994	
14	4196	4250	4289	
15	3968	4140	4199	
16	4564	4739	4790	
17	3297	3463	3515	
18	3591	3769	3813	
19(7) *	4232	4172	4257	
20(8)	5012	4954	5033	5035
21(9)	3345	3276	3335	
22(10)	3809	3744	3772	
23(11)	4691	NOT	4727	
24(12)	5293	FEASIBLE	5324	
25(13)	3917	↓	3923	
26(14)	4193	↓	4169	
	TASK III	LO <sub>2</sub> /LH <sub>2</sub>	(4)	7008
	TASK III	LO <sub>2</sub> /LCH <sub>4</sub>	(8) (20)	5113
	TASK III	LO <sub>2</sub> /FP-1		4581

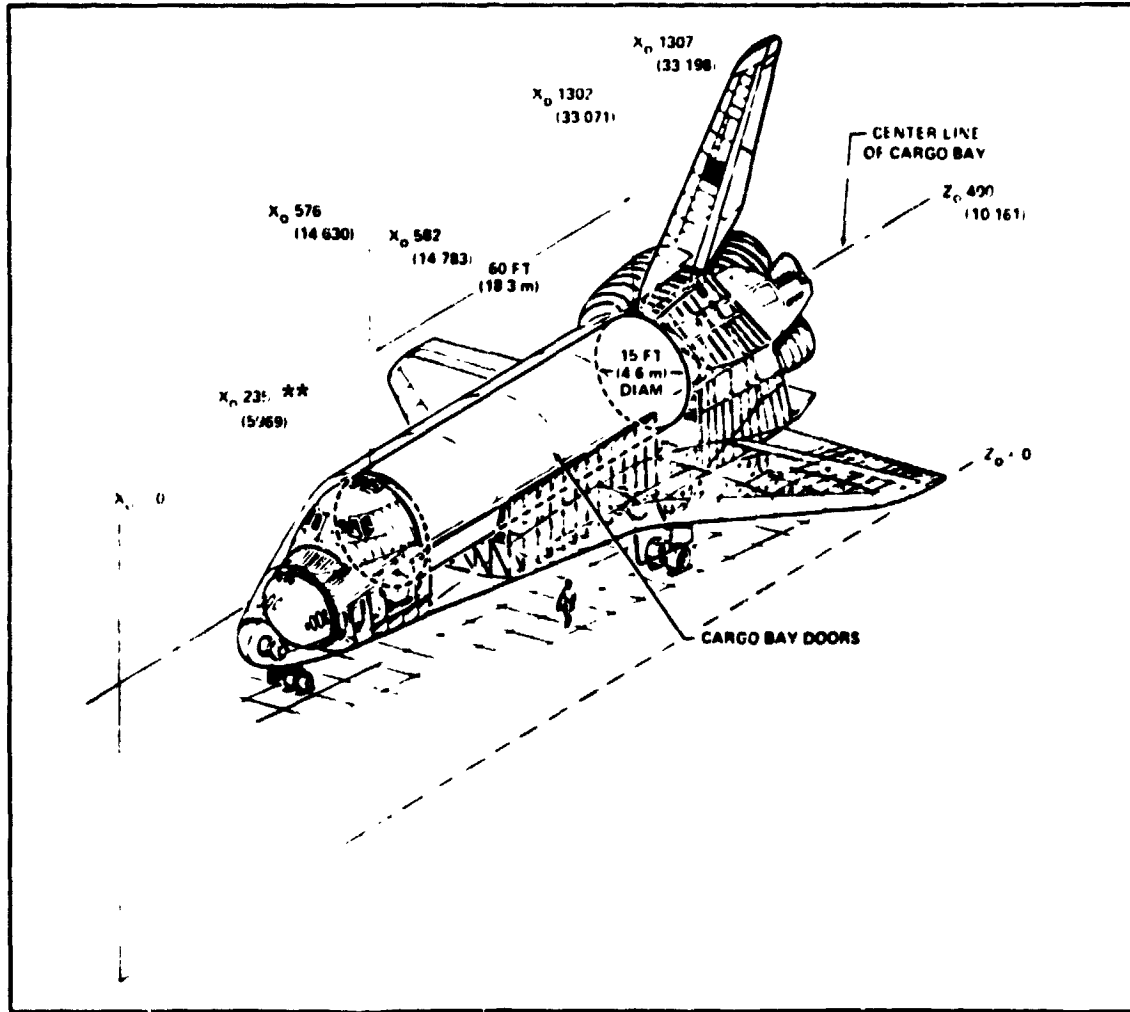
\* Numbers in parenthesis represent corresponding systems with different tank arrangements



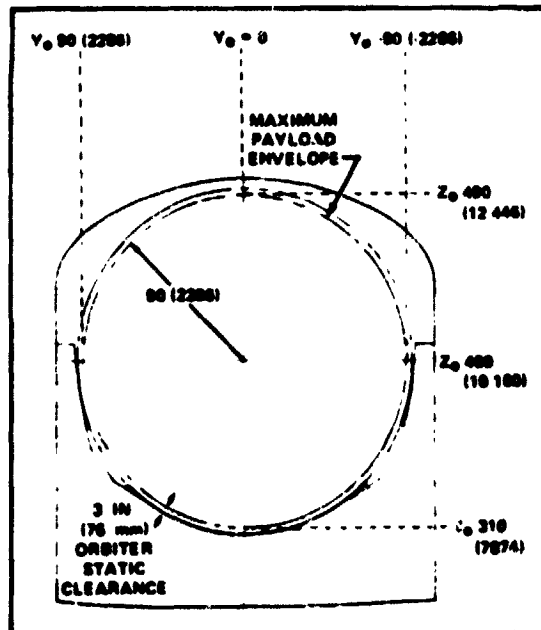
## VI. PAYLOAD ACCOMMODATIONS FOR THE LTPS/LSS IN THE ORBITER

Any payload intended to be launched by the STS must meet payload volume and mass constraints. The 18.28 m (60 ft) long by 4.57 m (15 ft) diameter payload envelope shown in Figure VI-1 has to accommodate payload and any clearances forward or aft of the payload. Forward clearances are for extra vehicular activity (EVA), Manned Maneuvering Unit (MMU), or any airborne support equipment (ASE). Two MMUs are included because one reason for LEO deployment of the LSS is for manned checkout of the structures. The ASE includes the mechanisms for payload activation monitoring, and deployment. Clearances aft of the payload are because of deployment constraints or ASE. A limit of 29,500 kg (65,000 lb<sub>m</sub>) mass exists for lift-off and a maximum design mass of 14,500 kg (32,000 lb<sub>m</sub>) for landing. Additional constraints exist for cargo mass distribution when landing and these center-of-gravity (C.G.) requirements are shown in Figures VI-2 and VI-3 for the three payload axes. If the payload cannot be deployed due to a flight abort or a problem on orbit, then the Shuttle can land with a payload larger than the 14,500 kg design limit but structural damage may occur. For all LTPS/LSS payloads evaluated in this study, a payload mass less than 14,500 kg can be reached by dumping only the oxidizer.

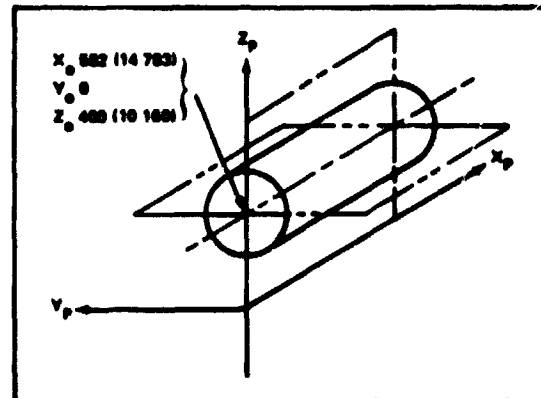
The payload positioning within the bay is determined by clearances aft and forward of the payload. The forward clearance is determined by the envelope required for storage and deployment of the MMU. To accommodate the MMUs, a clearance of 1.37 m (4.5 ft) aft of the flight deck is required on both sides of the payload bay. The clearance aft of the payload is due to the ASE, deployment procedure, and tank arrangement. The procedure chosen for this analysis is a fixed pivot point located at the engine exit similar to that used by General Dynamics in their Low Thrust Vehicle Concept Study for NASA/MSFC (Contract NAS8-33527, Task 7). A 75° deployment angle for the LTPS/LSS payload allows the LSS to be expanded while still attached to the Shuttle, see Figure VI-4. This method of deployment allows for erection and checkout while the unit is still fixed to the orbiter, thus the Shuttle RCS can be utilized for attitude control. This method also simplifies manned inspection. The tanking arrangement used changes the aft clearance because as



Orbiter coordinate system and cargo bay envelope. The dynamic clearance allowed between the vehicle and the payload at each end is also illustrated.



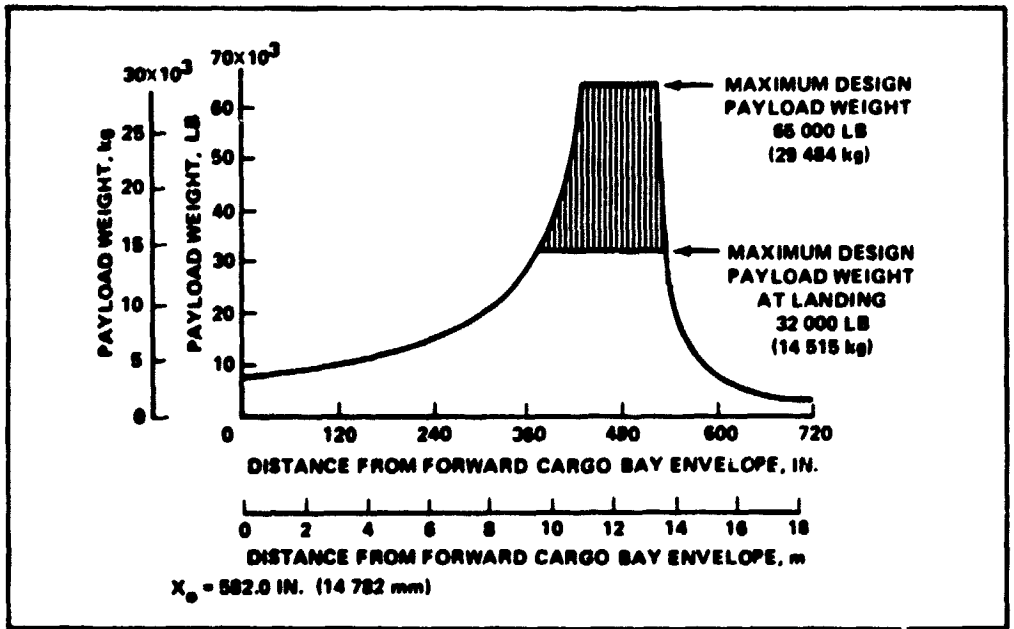
View of payload envelope looking aft.



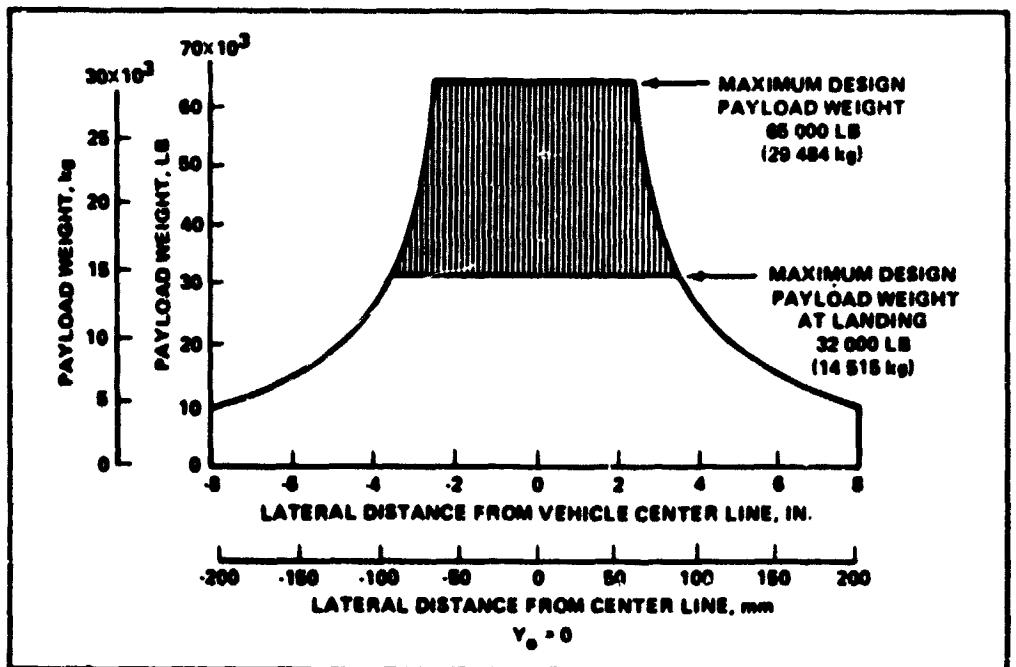
Payload coordinates, showing relationship to Orbiter station on each axis.

\*\* Orbiter coordinates are in inches (millimeters) unless otherwise stated

FIGURE VI-1 ORBITER PAYLOAD ENVELOPE (Ref. 16)



Payload center-of-gravity limits along the X-axis ( $X_G$ ) of the Orbiter.



Allowable center-of-gravity envelope along the Orbiter Y-axis ( $Y_G$ ).

FIGURE VI-2 PAYLOAD CENTER OF GRAVITY LIMITATIONS (Ref. 16)

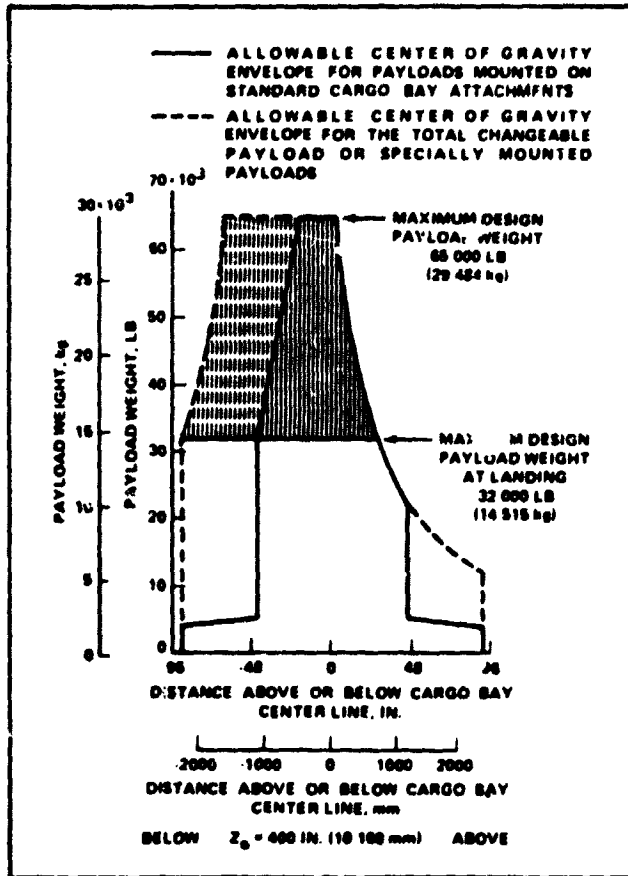


FIGURE VI-3 CENTER-OF-GRAVITY LIMITS OF CARGO, ALONG THE Z-AXIS ( $Z_0$ ) OF THE ORBITER (Ref. 16)

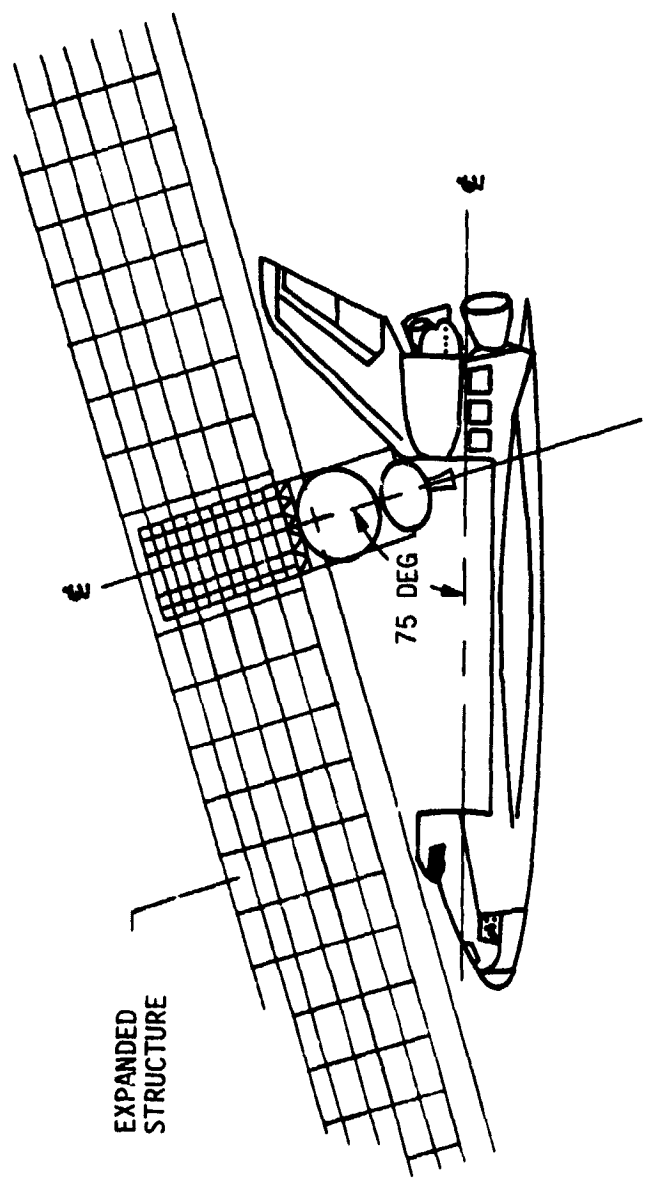


FIGURE VI-4 LTPS/LSS DEPLOYMENT ANGLE FOR SHUTTLE-ATTACHED DEPLOYMENT

the LTPS is rotated around the pivot point of  $75^\circ$  it must not hit the back of the cargo bay. To maximize usable space, the pivot point must be placed such that as the deployment angle reaches  $75^\circ$ , the edge of the tank touches the aft limit of the payload envelope. A scale drawing for three different tank configurations is shown in Figure VI-5 with minimum pivot point to aft payload limit distances. The drawing shows the 2225 N ( $500 \text{ lb}_f$ ) engine in the stored (dotted lines) and deployed positions, the outlines of the bottom of the tanks, and the relative positions of the aft payload limit (dashed vertical lines) with respect to the engine. The distances shown in Figure VI-5 were found graphically by locating the intersection of the tank perimeter (black curved lines) and the top of the payload envelope.

Using these restrictions on usable space payload envelopes were determined and C.G.s were calculated, these are shown in Figures V-6 through V-12. The C.G. was assumed to fall on the payload center line, with only variation along the X axis. To calculate the C.G. of the system, the sum of the moments of the components were divided by the total mass. In these C.G. calculations, the components are as follows:

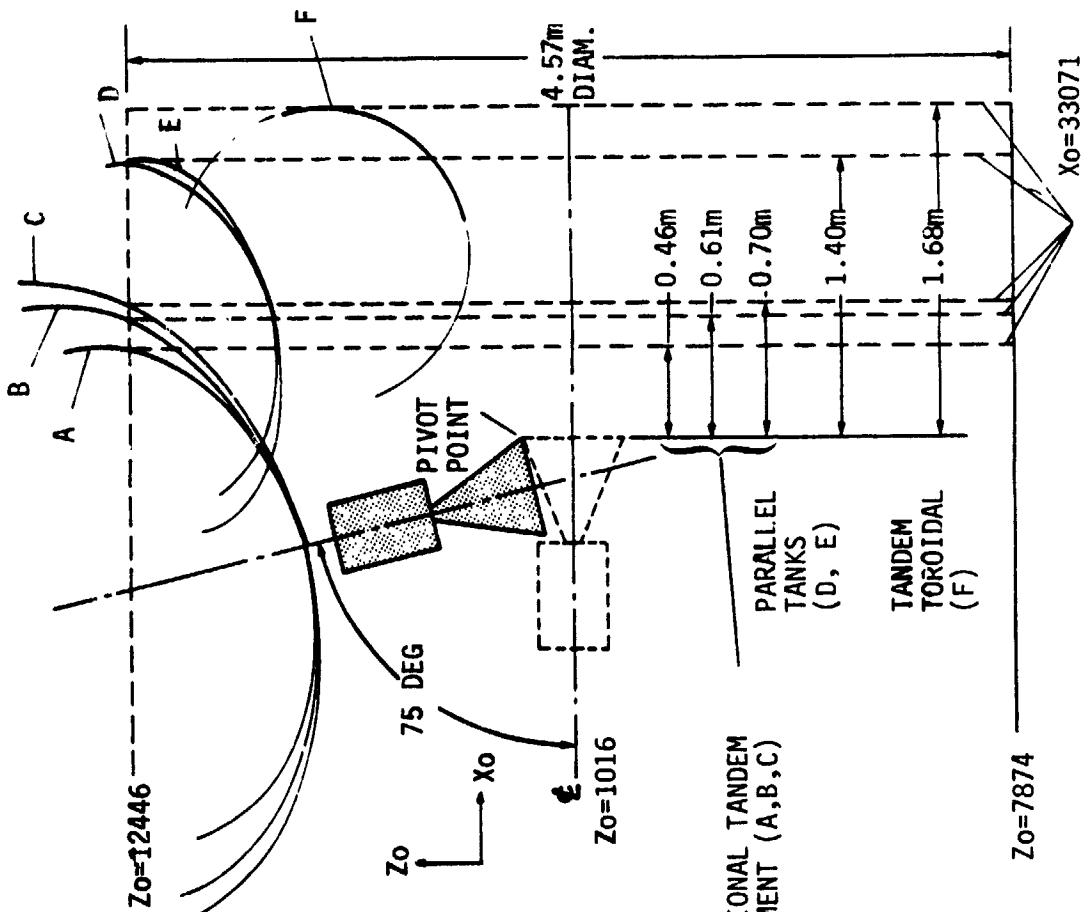
MMU - 460 kg; positioned forward of the payload.

ASE - 1810 kg; assumed distributed homogeneously in the aft of the bay.

Mass of Engine, Lines, and Hardware - determined by the engine thrust level.

Tanking System Mass - determined in PROP; the loaded values include total amounts of propellant, tank hardware and insulation. Unloaded values (in parenthesis) include tank hardware, insulation, and only the propellant considered as trapped.

Shell and Flight Hardware - this 680 kg was assumed to be evenly distributed within the shell.

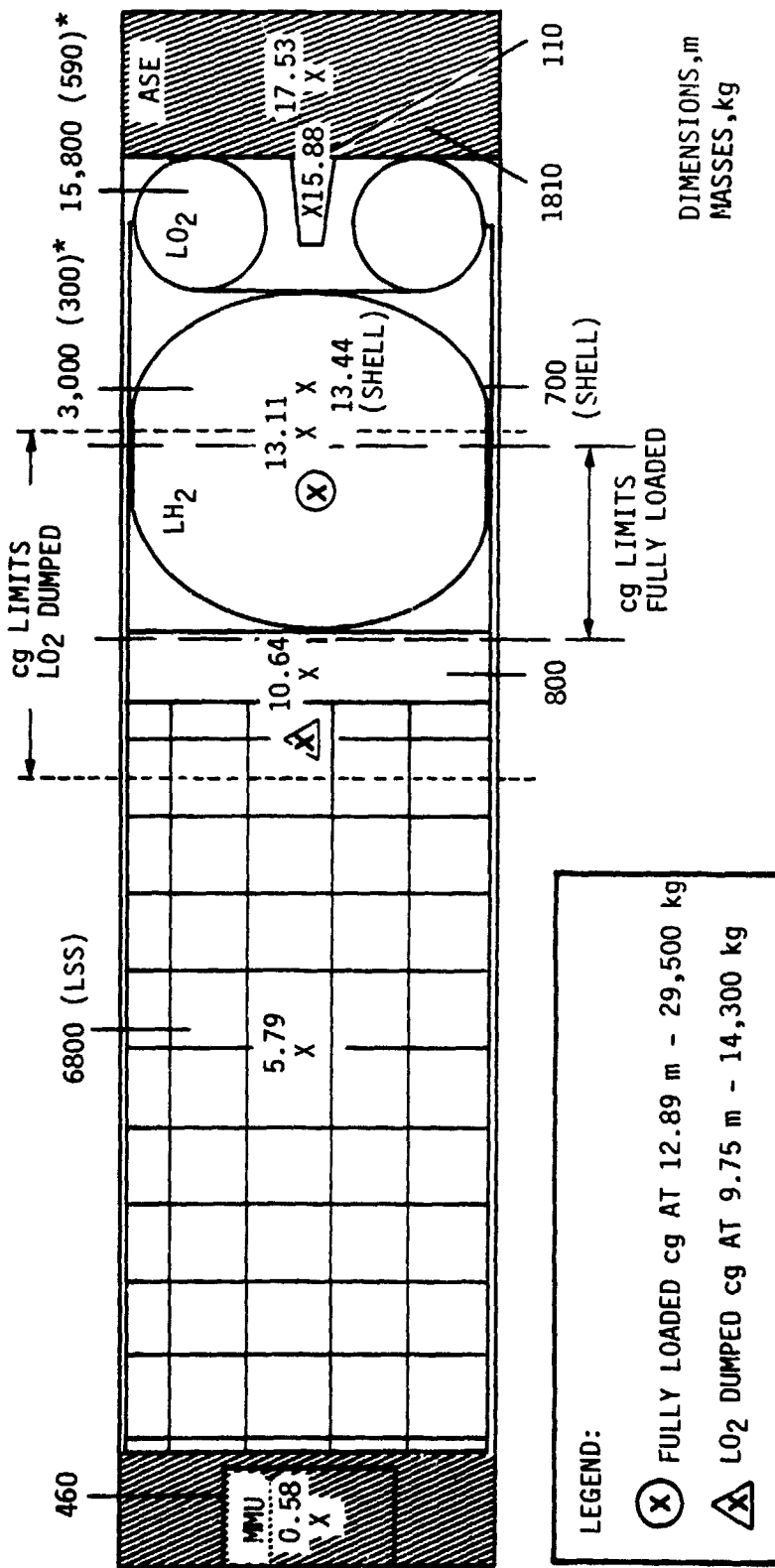


- A- RP-1 Tank, LO<sub>2</sub>/RP-1(Task III)
- B- LCH<sub>4</sub> Tank, LO<sub>2</sub>/LCH<sub>4</sub>(Task III)
- C- LO<sub>2</sub> Tank, LO<sub>2</sub>/LH<sub>2</sub>(Task III)
- D- LO<sub>2</sub> Tank } LO<sub>2</sub>/LCH<sub>4</sub>, 4450 N, 8 Burns
- E- LCH<sub>4</sub> Tank } LO<sub>2</sub>/LH<sub>2</sub>, 4450 N, 8 Burns
- F- LO<sub>2</sub> Tank, LO<sub>2</sub>/LH<sub>2</sub>, 4450 N, 8 Burns

FIGURE VI-5 REQUIRED DISTANCE OF ENGINE FROM AFT END OF PAYLOAD ENVELOPE FOR A 75° DEPLOYMENT USING VARIOUS TANKING ARRANGEMENTS

L02/LH2, MLI, 2225 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 5.55 m (18.2 ft)  
 LSS LENGTH = 8.90 m (29.2 ft)  
 LSS DENSITY = 51 kg/m<sup>3</sup> (3.2 lbm/ft<sup>3</sup>)

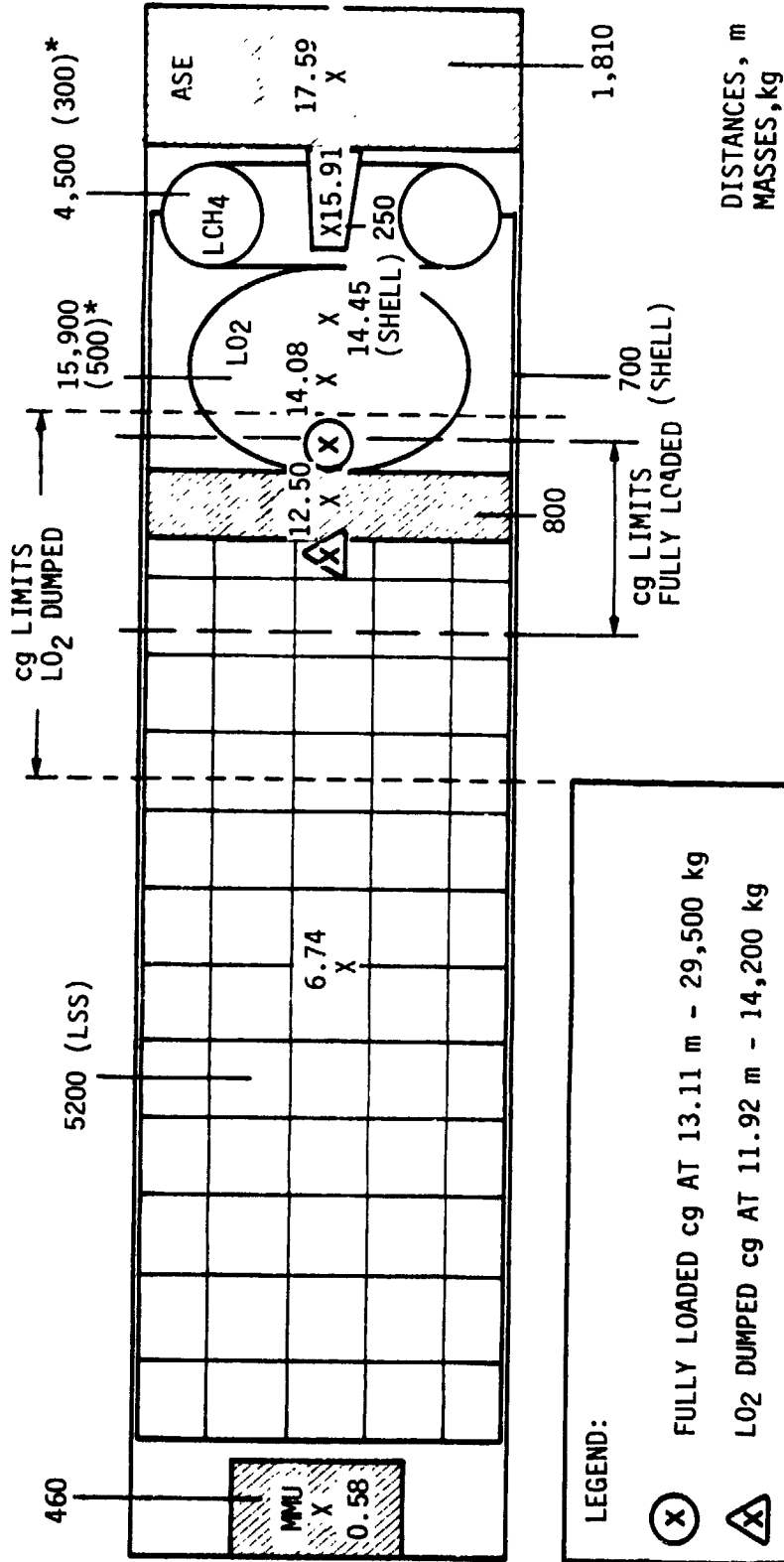
\*Masses in parentheses are for individual tank systems after the propellant has been dumped





L02/LCH4, MLI, 4450 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 3.84 m (12.6 ft)  
 LSS LENGTH = 10.67 m (35.0 ft)  
 LSS DENSITY = 34 kg/m<sup>3</sup> (2.1 lbm/ft<sup>3</sup>)

\*Masses in parentheses are for individual tank  
 systems after the propellant has been dumped

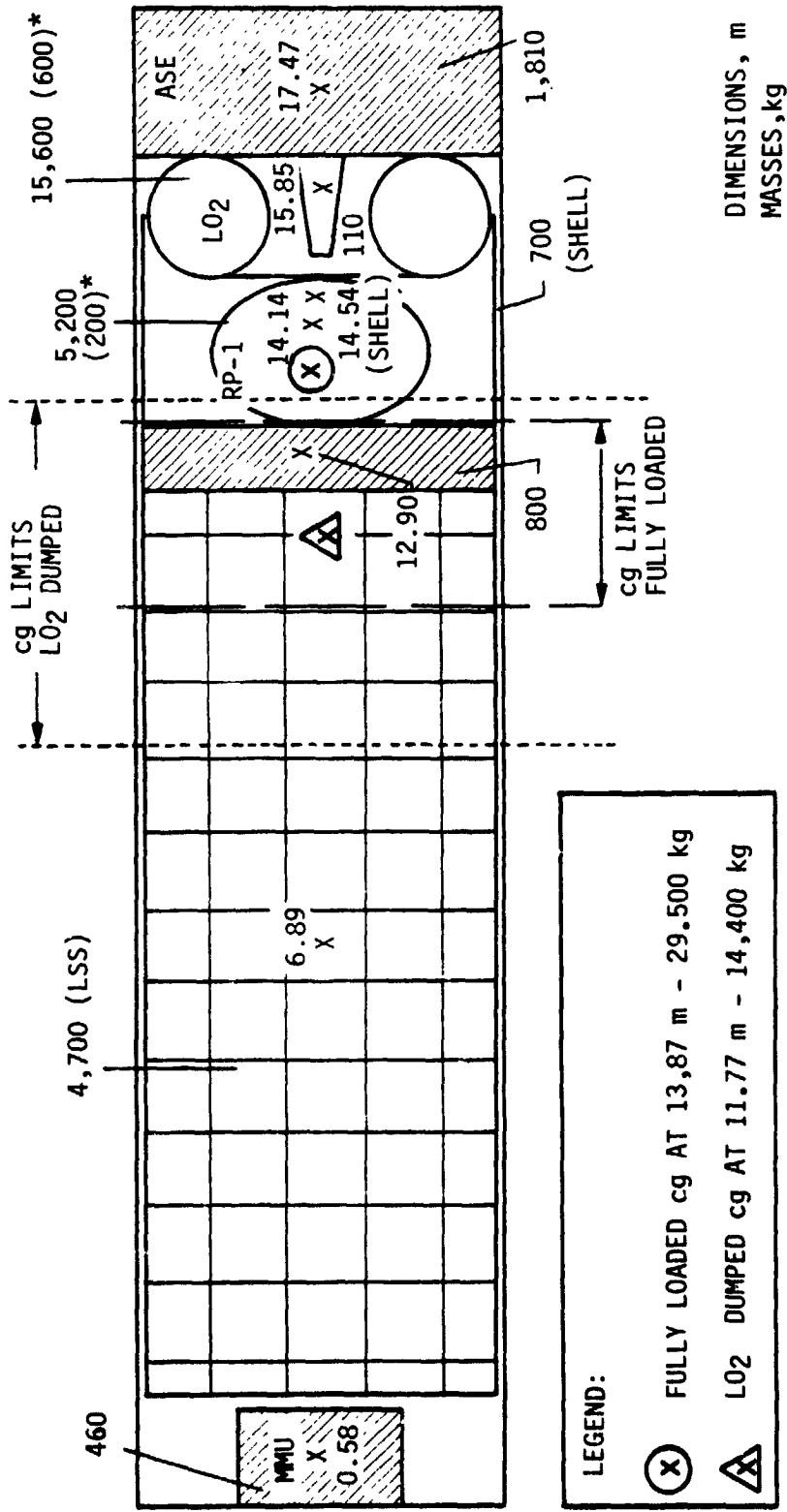


Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-7 SHUTTLE CARGO BAY PACKAGING OF LTPS/LSS - MINIMUM LENGTH L02/LCH4.

L02/RP-1, MLI, 4450 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 3.38 m (11.1 ft)  
 LSS LENGTH = 11.06 m (36.3 ft)  
 LSS DENSITY = 31 kg/m<sup>3</sup> (1.9 lbm/ft<sup>3</sup>)

\*Masses in parentheses are for individual tank systems after the propellant has been dumped

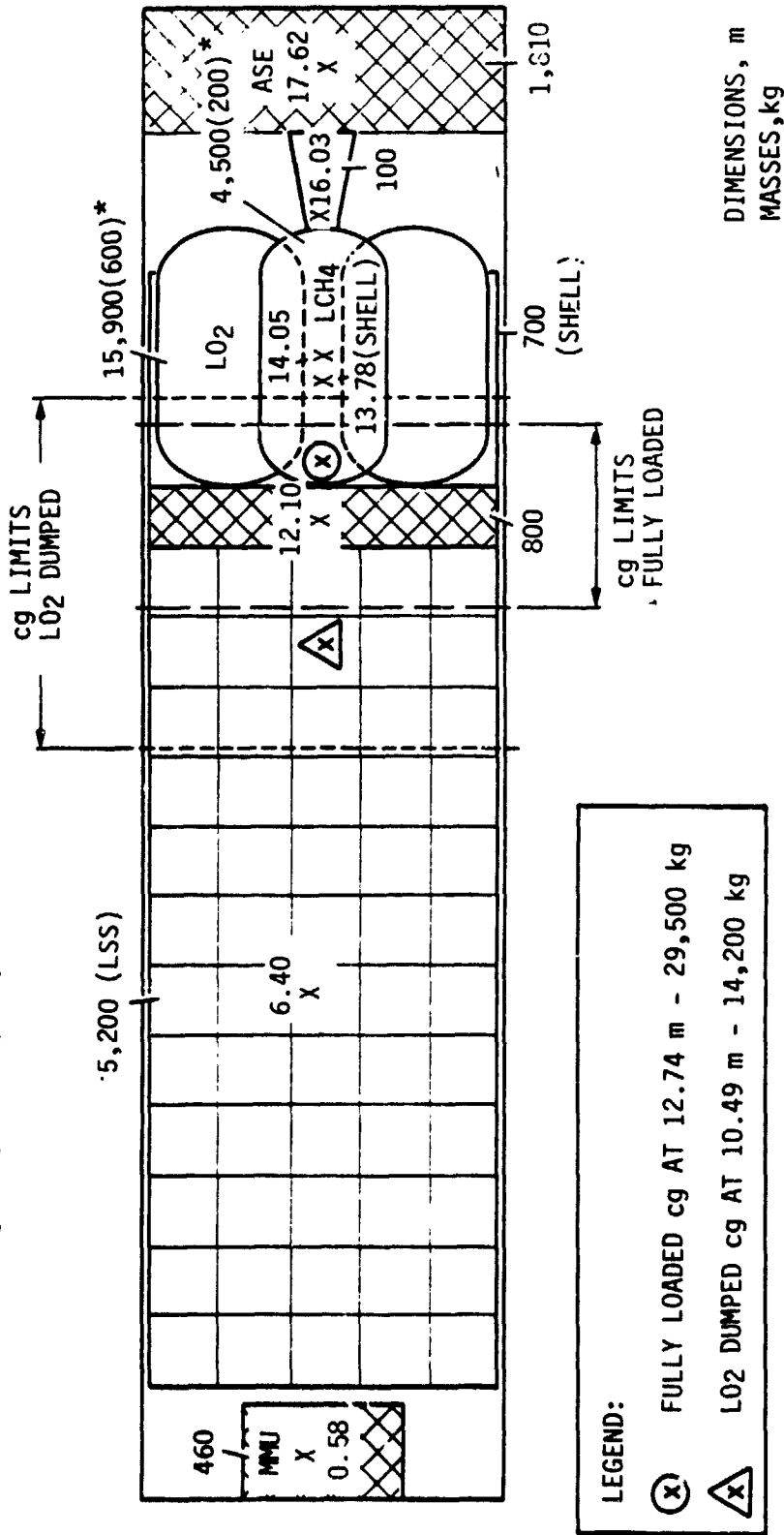


Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-8 SHUTTLE CARGO BAY PACKAGING OF LTPS/LSS - MINIMUM LENGTH L02/RP-1

L02/LCH4, MLI, 4450 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 4.36 m (14.3 ft)  
 LSS LENGTH = 10.36 m (34.0 ft)  
 LSS DENSITY = 35 kg/m<sup>3</sup> (2.2 lbm/ft<sup>3</sup>)

\*Masses in parentheses are for individual tank systems after the propellant has been dumped

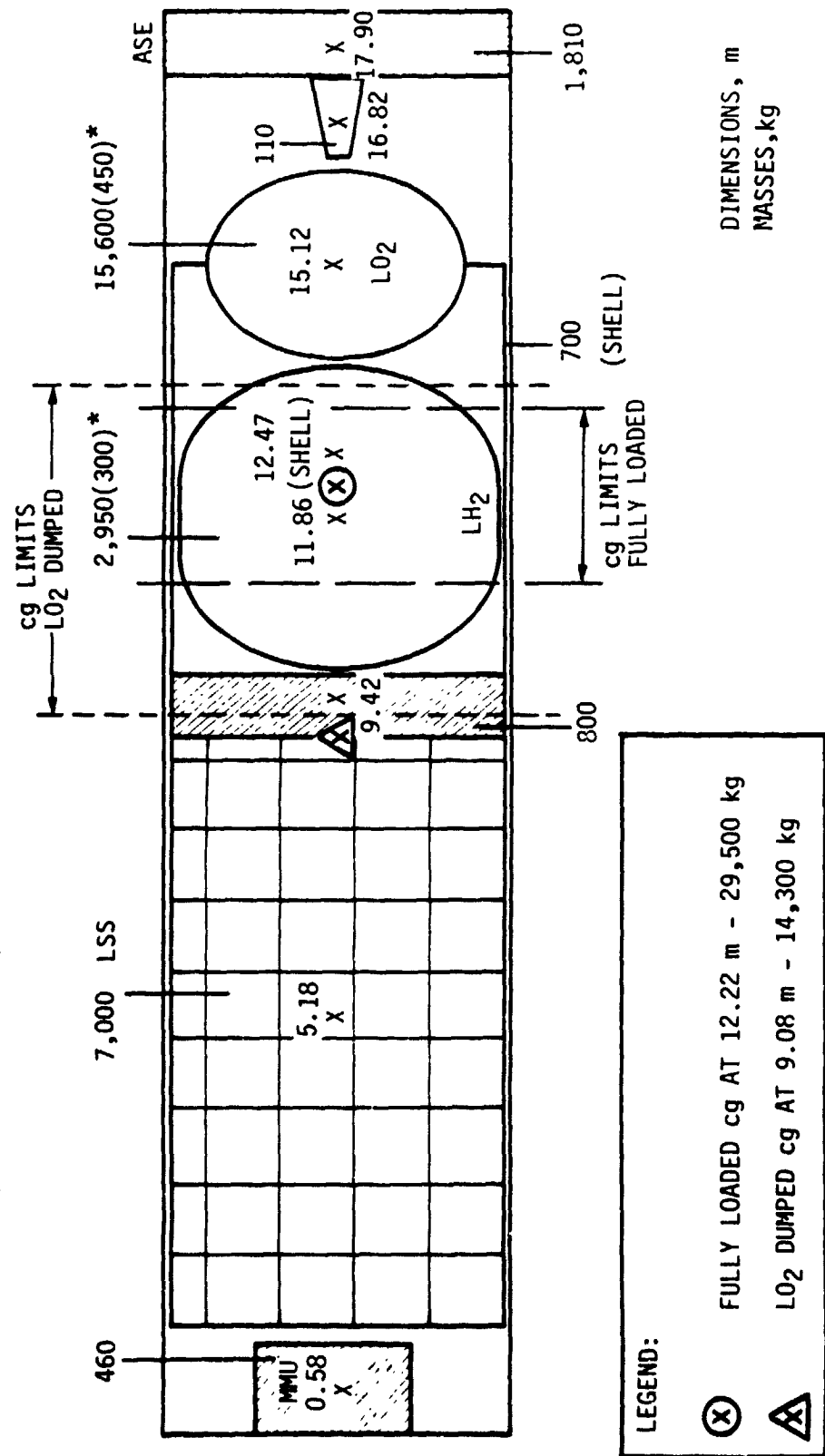


Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-9 SHUTTLE CARGO BAY PACKAGING OF LTPS/LSS - PARALLEL TANKS L02/LCH4

L02/LH2, MLI 2225 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 7.74 m (25.4 ft)  
 LSS LENGTH = 7.65 m (25.1 ft)  
 LSS DENSITY = 64 kg/m<sup>3</sup> (4.0 lbm/ft<sup>3</sup>)

\*Masses in parentheses are for individual tank systems after the propellant has been dumped



Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-10 SHUTTLE CARGO BAY PACKAGING OF LTPS /LSS -MAXIMUM PERFORMANCE L02/LH2

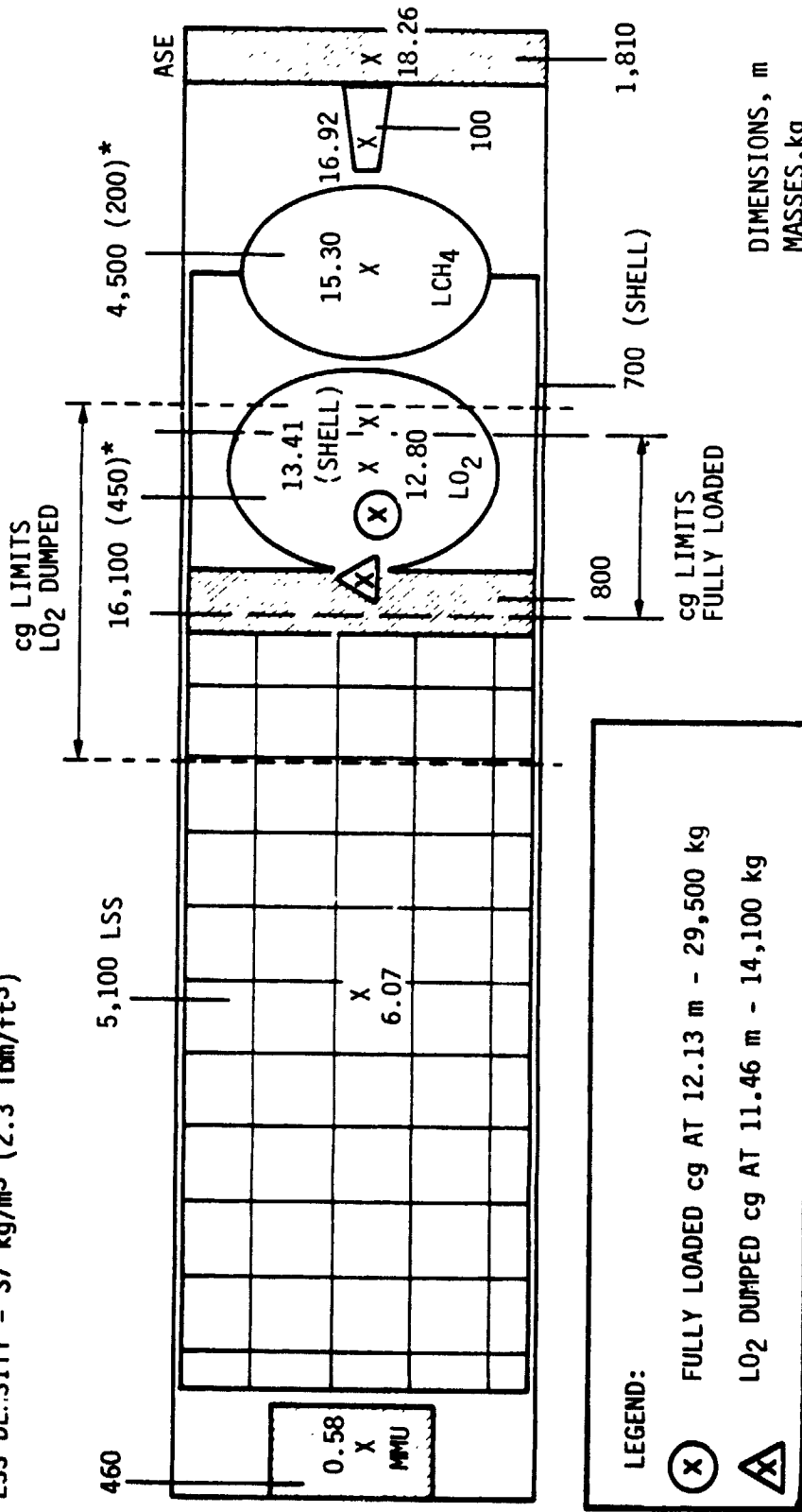
LO2/LCH4, TASK III, MLI, 2225 N, 8 BURNS  
75 DEG DEPLOYMENT

LTPS LENGTH = 6.07 m (19.9 ft)

LSS LENGTH = 9.45 m (31.0 ft)

LSS DENSITY = 37 kg/m<sup>3</sup> (2.3 lbm/ft<sup>3</sup>)

\*Masses in parentheses are for individual tank systems after the propellant has been dumped

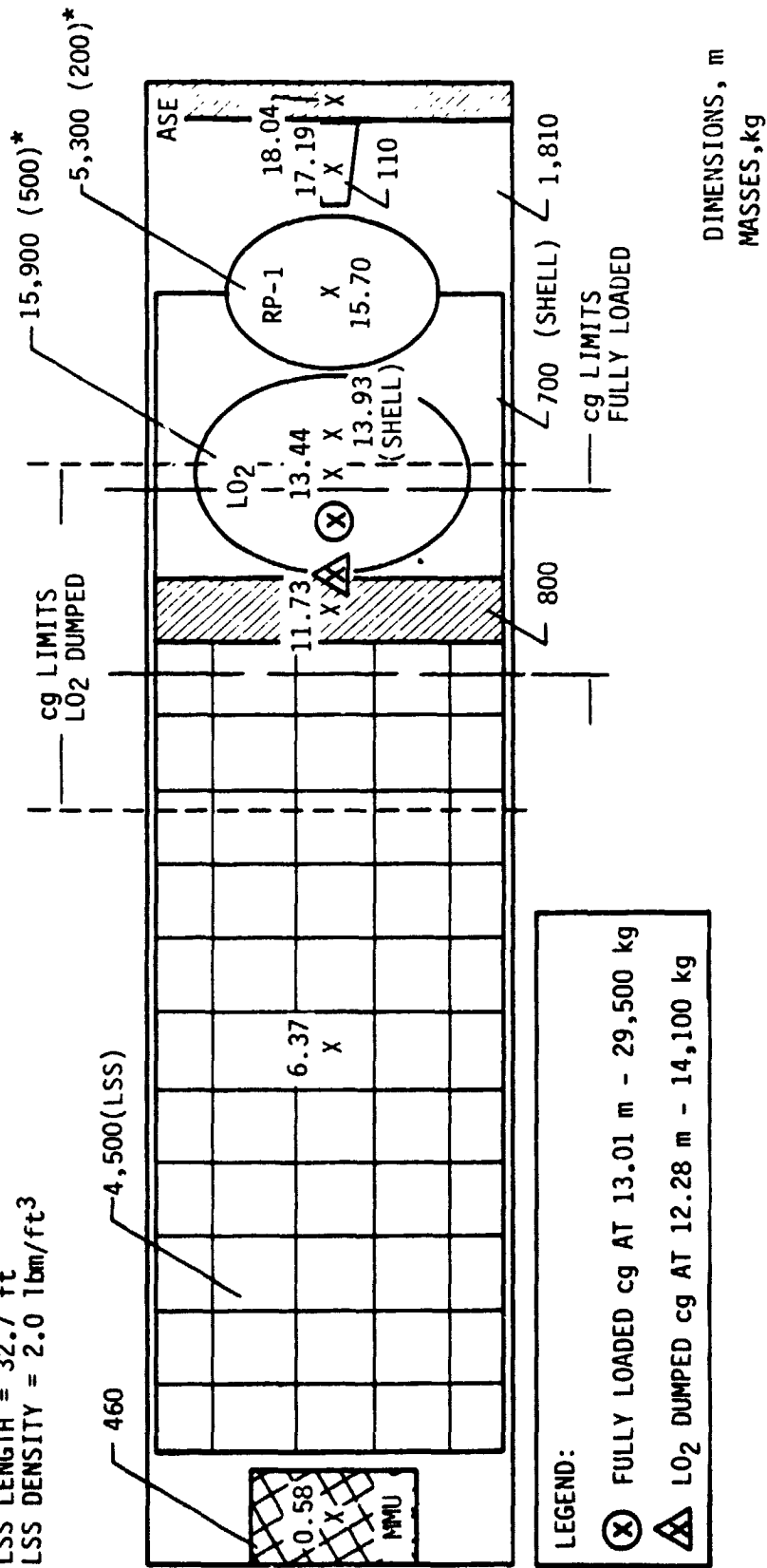


Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-11 SHUTTLE CARGO BAY PACKAGING OF LTPS/LSS - MAXIMUM PERFORMANCE L2O2/LCH4

\*Masses in parentheses are for individual tank systems after the propellant has been dumped

LO<sub>2</sub>/RP-1, TASK III, MLI, 2225 N, 8 BURNS  
 75 DEG DEPLOYMENT  
 LTPS LENGTH = 18.5 ft  
 LSS LENGTH = 32.7 ft  
 LSS DENSITY = 2.0 lbm/ft<sup>3</sup>



Distances along the payload bay centerline are measured from the forward payload limit

FIGURE VI-12 SHUTTLE CARGO BAY PACKAGING OF LTPS/LSS - MAXIMUM PERFORMANCE LO<sub>2</sub>/RP-1

Adapter Ring and Flight Hardware - some of the flight hardware is contained in this space forward of the tanks plus 230 kg (500 lb<sub>m</sub>) has been allowed for the ring itself. This mass for the adapter assumes a 0.76 cm thick aluminum ring 0.75 m long. This is oversized for transfer orbit longitudinal accelerations. But allowance must be made for bending and torsional stresses during launch in the Shuttle and during transfer orbit maneuvering.

LSS Payload - 29,500 kg (65,000 lb<sub>m</sub>) minus the sum of all other components. The mass is assumed to be distributed homogeneously with a density shown on the figure.

Manned Maneuvering Unit (MMU) - Two MMUs each weighing 230 kg (500 lb<sub>m</sub>) and occupying the space directly aft of the flight deck.

It should be noted that calculations for unloaded payload conditions include the dumping of only the oxidizer. This would represent RTLS where time permitted only the dumping of one propellant. From these calculations a range of payload densities is seen, the highest density payload uses a LO<sub>2</sub>/LH<sub>2</sub> in a conventional tandem configuration, the lowest density payload uses LO<sub>2</sub>/RP-1 in a tandem/toroidal configuration.

Finally, the C.G. limits shown in the diagrams of the payload are obtained from the data in Figure VI-2 and VI-3. Under the conditions of this study all configurations except the maximum performance LO<sub>2</sub>/LH<sub>2</sub> are within the mass and C.G. limits with the LO<sub>2</sub> dumped. Only the minimum length LO<sub>2</sub>/RP-1 falls outside the C.G. limits when fully loaded. Both of these could be corrected, the LO<sub>2</sub>/LH<sub>2</sub> payload would have to be reduced and the LO<sub>2</sub>/RP-1 vehicle could be moved further forward. But both of these fixes would reduce the length or mass of the LSS.

## VII. TECHNOLOGY EVALUATION

As with any new space system, certain improvements in technology must be attained before the vehicle is constructed. The technology problems facing the LTPS are briefly described in Table VII-1 and are discussed in detail in the following subsections.

### A. PROPELLANT MANAGEMENT

The adequacy of the technology for propellant management was evaluated and the deficiencies have been identified. In the following sections, the technology relevant to each of the three LTPS propellant management techniques is discussed. For further details of existing technology, the survey performed in Reference 17 provides a comprehensive summary of the state-of-the-art.

#### 1. Propulsive Settling System

Definition of the time required to settle propellant represents a key technology for propulsive settling. The available technology was discussed in Section III and is limited with regard to tank geometry and acceleration environment. Accurate prediction of the settle time requires that the influence of the following factors be understood in detail:

- o tank geometry, including stringers, ribs, and slosh baffles;
- o fill fraction and initial liquid orientation; and
- o degree of settling (i.e., bubble entrainment, geysering, splashing, etc).

More investigations of the type performed by Sumner (Ref. 7), which attempt to establish correlations that account for a wide range of variables, are required. The value of an approach that optimizes the settling acceleration needs further investigation. If toroidal and ellipsoidal tanks are to be used, investigations using these tank geometries are also required.



TABLE VII-1 TECHNOLOGY DEFICIENCIES

SYSTEM	MAJOR TECHNOLOGY CONCERN
<p>PROPULSIVE SETTLING</p> <p>FINE-MESH SCREEN AQUISITION DEVICES</p>	<ul style="list-style-type: none"> <li>o Experimental Verification and Refinement of Analytical Techniques</li> <li>o Screen Dryout</li> <li>o Thermal Isolation of Device</li> <li>o Structural Design of Attachments</li> <li>o Integration with Pressure Control Systems</li> </ul>
<p>TOROIDAL TANKS</p>	<ul style="list-style-type: none"> <li>o Propellant Slosh Modes</li> <li>o Residual Prediction Techniques</li> </ul>
<p>TANK INSULATION</p>	<ul style="list-style-type: none"> <li>o Performance of Combined SOFI/MLI Systems</li> </ul>
<p>PROPELLANT DUMPING</p>	<ul style="list-style-type: none"> <li>o Impact on Propellant Management</li> </ul>
<p>PROPELLANT GAGING</p>	<ul style="list-style-type: none"> <li>o Insufficient Acceleration for Conventional Techniques</li> </ul>

Investigation of propellant settling times have been based upon test data which is usually obtained with subscale tanks and referee liquids. Scaling of the test conditions is required to apply the results to full-size tanks and the actual propellants. Drop tower tests have been used extensively for settling studies, but the test times are limited. A fluid physics module is planned for Spacelab, which will be able to investigate propellant settling (Ref. 8). Such tests are recommended to further the technology investigation. Development of adequate correlation methods and scaling approaches should continue.

The other aspect of propulsive settling that requires further investigation is the design and performance of bubble filters. The technology of screen performance is well understood, but its specific application to tank draining needs to be investigated. The velocity field due to draining and propellant motion induced by settling will influence the effectiveness of the bubble filter in delaying gas ingestion. A refined and experimentally verified analytical approach to selecting the screen mesh and flow area is needed. Tests of prototype configurations under simulated draining conditions will be required.

One-g draining tests with a subscale tank model could investigate the effects of draining. Test method, scaling, and correlation would be similar to conventional tests. The screen area and mesh, test liquid, and its flowrate would be varied. Drop tower tests simulating the propellants settling and draining would add the effects of the liquid motion and reduced acceleration.

## 2. Partial Acquisition Devices

The time required to settle propellant, discussed above under "Propulsive Settling", is also pertinent to partial acquisition devices.

Prediction of the quantity of propellant lost from the device due to vaporization is essential to sizing the reservoir. The continued development of thermodynamic models, capable of predicting mass transfer under low-g conditions, is needed to perform this analysis. Investigations aimed at

providing heat transfer correlations for low-g conditions are recommended. Such investigations are one of the basic needs for not only propellant management but for the design of any type of low-g fluid storage, supply or transfer system as well as other fields, such as materials processing in space. Investigations such as those described in Reference 8 are currently being planned for Spacelab. Verification of the predictions will require tests of prototype devices. One-g tests will provide some insight, but low-g tests of prototype systems, including the acquisition device, tank, and thermal control system will be necessary.

Screen dryout is another area where there is a deficiency in demonstrated technology. A number of studies, sponsored by NASA-LeRC, have been performed (Ref. 18 and 19). Another effort entitled "Vapor Inflow Study" was recently initiated. These studies have been addressing the influences of heat input rate, the rate at which vapor flows through screen, and the configuration and mesh of the screen. Reduction of the tank pressure by venting must also be evaluated, since it will produce vaporization, or possibly boiling, at the screen surfaces. It is recommended that these studies continue, including tests of prototype devices under realistic operating conditions. The above described low-g test of a prototype system would also provide data on screen dryout.

As part of these test programs, the basic screen performance parameters - retention capability and pressure drop due to flow through the screen - should be verified. Some verification of these parameters has been done for oxygen and hydrogen, but there are little data for the other propellants that were considered in this study: methane and RP-1. This technology need is also applicable to the bubble filters for a propulsive settling system and for total acquisition devices.

The structural design of these devices is also a concern. Methods of fabricating the device to provide the structural support required to withstand the launch load and vibration environment and to provide isolation from the thermal environment need to be developed. Candidate concepts must be selected

and analyzed. This is another area where testing of prototypes is essential. Static load and vibration tests would verify the structural capability, and the effectiveness of the thermal isolation would be measured under typical operating conditions.

### 3. Total Acquisition Devices

Screen dryout, discussed under "Partial Acquisition Devices", is also a concern for total acquisition devices. For this case, vaporization within the device must be avoided. Studies similar to those that are being performed for partial acquisition devices are needed for total acquisition devices. The point at which boiling will occur inside the channels based on the heat input from the ullage gas, and the attachments to the tank wall would be established. Again, tests of prototype devices under one-g and low-g conditions are recommended.

The total acquisition device represents more complex structural design problems than the partial acquisition device. The long, narrow channels must be strong enough to withstand launch and must be thermally isolated from the tank wall. Prototypes should be designed and tested, measuring heat input and strength.

The Cryogenic Fluid Management Experiment (NAS3-21591), a Spacelab experiment being designed by Martin Marietta Denver Aerospace for NASA-LeRC, will make a significant contribution to this technology. The experiment will have a total acquisition device that will expel a liquid cryogen (LH<sub>2</sub>) under low-gravity conditions. The tank diameter will only be one meter, but the thermal conditions should be representative of the LTPS application.

### B. TANKS

Toroidal tanks are necessary to utilize the superior payload capability of the LO<sub>2</sub>/LH<sub>2</sub> propellant combination. These large tanks (4.3m diameter) have problems that can be divided roughly into two areas of concern - technology deficiencies and those that are associated with vehicle development. Some of these developmental problems of the toroids are also shared with the conventional tanks.

Technology deficiencies of the toroidal tank originate from its geometry and because it is untested at sizes required for the LTPS, therefore, the following areas of concern need investigation:

- o The effect of the number of outlets on propellant residuals and tank complexity; and
- o Determination of propellant sloshing modes and their interaction with the thin wall structure.

The solutions to the above would entail scale model tests of outflow in low-g, vibration testing, and the associated analyses.

Other concerns exist with the toroid but these can be described more accurately as design problems associated with the construction of a full size flight tank. Structural analysis and testing would provide information on the following design problems:

- o The internal support required for a thin walled toroidal tank with diameters as large as 4.3 m; and
- o Design and construction of baffles to reduce slosh.

Developmental problems that exist for both the conventional and toroidal tanks are as follows:

- o Structural supports for thin walled tanks inside the STS payload bay;
- o Reliability of tanks exposed to the STS launch environment; and
- o The compatibility of a composite overwrap with cryogenics to reduce the tank weight.

As with any new design, use of these large diameter-thin walled tanks in a flight vehicle would require an extensive test program.

## C. THERMAL ISOLATION

### 1. Tank Insulation Covering

Concerns associated with insulation of the cryogenics can also be considered to fall into one of the two categories mentioned in the previous section, technology deficiencies and developmental problems.

MLI would be the first choice for tank insulation due to its very low thermal conductivity when it is in a vacuum. Unfortunately, the increased complexity of using this system instead of a simple system such as the SOFI, would create certain developmental problems that would require overcoming. These concerns are:

- o Application to the large ellipsoidal and toroidal tanks needed for the LTPS;
- o Implementation of a ground purge system in the Orbiter payload bay;
- o A faster purge of the insulation so that the vacuum operating conditions can be reached sooner; and
- o Layer density control during STS launch, since compression of layers would result in degraded thermal performance.

Previous tests have established the reliability and excellent thermal characteristics of a multilayer system so only questions of application and implementation to individual systems remain.

An alternative system may be able to reduce the complexity of an MLI system. Some SOFI systems were chosen in the 26 selected configurations because of the reduced complexity of these systems due to the lack of purge requirements and potential ease of application. If a layer of SOFI was installed under the MLI, low thermal conductivity could possibly be combined

with the reduced complexity and improved ground-hold thermal characteristics of SOFI. This combination would require reduction of outgassing from the SOFI since the amount of gas given off is enough to seriously reduce the effectiveness of the MLI.

## 2. Support Struts

The support struts from the outer LTPS shell to the propellant tanks represent a direct thermal conduction path. From Table II-5 it can be seen that, on orbit, this heat leak through the struts is considerably larger than the sum of the corresponding heat leak through the MLI. Thus, the design of the support struts to minimize any heat leak to the cryogenics is an important factor in reducing boiloff losses. Design of supports for cryogenic payloads in the Shuttle are part of the task in the two contracts "Cryogenic Fluid Management Experiment" (NAS3-21591) and "Conceptual Design and Analysis of Orbital Cryogenic Liquid Storage and Supply Systems" (NAS3-22264).

## D. PROPELLANT DUMPING

Aborting a mission at any time would require dumping of one or more propellants to lower the Orbiter payload mass to less than 14,200 kg. As described in Section V, dumping of only the  $LO_2$  would bring the LTPS/LSS payload within mass and C.G. limits. For safety reasons, both propellants may have to be dumped and the tanks inerted. If this is the case,  $LO_2/LCH_4$  and  $LO_2/RP-1$  systems will still fit within the C.G. limits but the  $LO_2/LH_2$  configuration will be outside the landing limits described in Section V. A RTLS abort would place the most stringent requirements on propellant management. The difficulties of this abort are the short period of time that exists for propellant dumping overboard and the varying accelerations and directions. The pressurization concerns during abort are being addressed by the "Low Thrust Chemical Propulsion System Propellant Expulsion and Thermal Conditioning Study" (NAS3-22650). The impact of abort on propellant management needs to be examined as this may determine the technique used rather than any optimized systems as described in this report.

#### E. PROPELLANT GAGING

Continuous gaging of the propellant would not appear to be necessary, but monitoring of the propellant level during main engine burns would be adequate for updating the propellant utilization predictions. Even though the engine thrust is relatively low, the minimum accelerations are large enough to make acceleration forces dominate surface tension forces so the propellant interface within the tank during a main engine burn is essentially flat. However, the acceleration may not be large enough to make acceleration forces dominate in the vicinity of the sensing probes of the gaging system. A local distortion of the interface or clinging of the liquid within the sensor can result in erroneous propellant level readings. The operation of such sensors will have to be verified for the accelerations and propellants of the LTPS to ensure such gaging systems are suitable for this application. More sophisticated methods of gaging which are independent of gravity level, and are also less developed, may be needed (e.g. nuclear gaging with a radiation source and detector).

#### F. FACILITIES REQUIRED

A top priority for test facilities would be a precision model shop and a cryogenic propellant laboratory. These would be required for scale model tests of propellant management, propellant outflow tests, liquid sloshing, screen performance, structural tests, and tank support strut design evaluation. Drop tower tests would be required for low-g draining simulations. Vibration test facilities to simulate STS launch environment are also needed. Full scale fabrication capability should exist to evaluate manufacturing problems of toroidal and ellipsoidal tanks with thin walls. A vacuum chamber large enough to test MLI application to LTPS sized tanks and a clean room to assemble and test screen devices in scale model test tanks may be required. Many of these tests could be combined into one program if the facilities exist in one area. This could reduce cost and possible duplication of tests.



## VIII. RESULTS AND CONCLUSIONS

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The primary objectives of this study were to size various vehicle configurations, determine preferred propellant management techniques, and to assess the adequacy of current technology for low-thrust chemical propulsion system development.

### A. LTPS VEHICLE SIZE

Propellant requirements, system masses, and dimensions of tanks and the stage are included in Tables VIII-1 through 5. The vehicle size was the determining factor in the volume and mass available for the LSS, assuming a single shuttle flight with a mated LTPS/LSS payload. The approach used in Section VI on payload accommodation was followed to determine the maximum length available for the packaged LSS. The results are listed in Table VIII-6 along with LSS mass and packaged density. This density was calculated by using the maximum allowable payload length, a 4.27 m (14 ft) diameter packaged structure, and the maximum allowable LSS mass. From work done by Martin Marietta on the Primary Propulsion/LSS Interaction Study (NAS3-21955), a density range of 24 to 56 kg/m<sup>3</sup> (1.5 to 3.5 lb<sub>m</sub>/ft<sup>3</sup>) was predicted for deployable solar arrays, mesh antenna and radar. The vast majority of these predicted LSS payloads based on LTPS capability fall within the LSS density limits, see Figure VIII-1. Therefore, if the actual packaged LSS length is equal to or less than the maximum length available for a selected LTPS, propulsion system/payload compatibility has been achieved.

Selection of an LTPS is highly dependent on the LSS payload. Both the length and mass of the undeployed structure would determine the vehicle needed. But general trends for various configurations can be predicted. In Figure VIII-2 the LTPS vehicle lengths are charted in descending order and the LSS lengths available from mating with a particular vehicle are charted in the same format in Figure VIII-3. The configuration refers to the LTPS vehicle; those identified with an asterisk are the maximum performance configurations described in Section V of this report. For the LO<sub>2</sub>/LH<sub>2</sub> systems it can be

TABLE VIII-1 PROPELLANT SYSTEM CHARACTERISTICS		Tanking Configuration: Tandem/Toroidal Initial Vehicle Mass: 27216 kg													
CONFIG.	THRUST, N (lb <sub>f</sub> )	NUMBER OF BURNS	INSULATION CONCEPT	PROPELLANT MASS, kg						TANKS				PAYLOAD MASS, kg	OVERALL LENGTH,* m
				USABLE (ΔV)	TRAPPED, START-S/D	LOSSES	BOILOFF LOSSES	TOTAL	VOLUME, m <sup>3</sup>	DIAMETER, m	LENGTH, m	LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg		
1	445 (100)	4	MLI	F	2771	86	232	20532	4.71	4.27	4.30	2661	22602	4613	5.97
				O	16628	521	293		16.2	4.27	1.55				
2	→	3	→	F	2644	88	248	20030	46.2	4.27	4.23	2642	22096	5120	5.88
				O	16165	417	317		15.7	4.27	1.52				
3	2224 (500)	4	→	F	2600	87	162	19247	43.5	4.27	4.05	2722	21296	5920	5.65
				O	15599	603	197		15.2	4.27	1.49				
4	→	8	→	F	2477	88	179	18425	41.8	4.27	3.93	2697	20464	6751	5.48
				O	14860	598	224		14.5	4.27	1.44				
5	4448 (1000)	4	→	F	2538	88	153	13864	42.4	4.27	3.97	2809	20931	6285	5.55
				O	15229	670	186		14.8	4.27	1.47				
6	→	8	→	F	2437	90	171	18186	41.2	4.27	3.89	2779	20249	6967	5.43
				O	14620	656	212		14.3	4.27	1.43				

\*INCLUDES INSULATION THICKNESS AND ENGINE LENGTH.

CONFIG.	THRUST N (10 <sup>4</sup> )	NUMBER OF BURNS	INSULATION CONCEPT	PROPELLANT MASS, kg					TOTAL	TANKS			LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg	PAYLOAD MASS, kg	OVERALL LENGTH,* m
				USABLE (ΔV)	TRAPPED, START-S/D	LOSSSES	BOILOFF	LOSSES		VOLUME, m <sup>3</sup>	DIAMETER, m	LENGTH, m				
7	2224 (500)	4	MLI	F	4308	166	84	21187	11.4	4.27	1.24	23042	4173	3.79		
				O	15939	512	178		15.4	3.47	2.45					
8	→	8	MLI	F	4136	169	93	20410	11.0	4.27	1.21	22266	4950	3.73		
				O	15302	508	203		14.8	3.42	2.42					
9	→	4	SOFI	F	4105	162	590	22065	12.1	4.20	1.30	23845	3371	3.90		
				O	15189	495	1523		15.8	3.50	2.47					
10	→	8	SOFI	F	3961	166	640	21570	11.8	4.18	1.29	23354	3862	3.89		
				O	14654	495	1654		15.5	3.47	2.45					
11	4448 (1000)	4	MLI	F	4209	190	80	20740	11.2	4.27	1.22	22620	4595	3.85		
				O	15572	521	168		15.1	3.44	2.43					
12	→	8	MLI	F	407 <sup>2</sup>	185	90	20126	10.9	4.27	1.20	22006	5209	3.83		
				O	15067	519	194		14.6	3.41	2.41					
13	→	4	SOFI	F	4006	177	561	21511	11.8	4.20	1.28	23315	3900	3.89		
				O	14823	504	1439		15.4	3.47	2.45					
14	→	8	SOFI	F	3897	182	619	21204	11.7	4.19	1.27	23013	4203	3.87		
				O	14418	508	1581		15.2	3.45	2.44					

\*INCLUDES INSULATION THICKNESS AND ENGINE LENGTH.

TABLE VIII-3 PROPELLANT SYSTEM CHARACTERISTICS															
Propellant Combination: LO <sub>2</sub> /RP-1 Mixture Ratio: 3:1				Tanking Configuration: Tandem/Toroidal Initial Vehicle Mass: 27216 kg											
CONFIG.	THRUST, N (10 <sup>4</sup> )	NUMBER OF BURNS	INSULATION CONCEPT	PROPELLANT MASS, Kg				TOTAL	TANKS			LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg	PAYLOAD MASS, kg	OVERALL LENGTH,* m
				USABLE (ΔV)	TRAPPED, START-S/D	BOILOFF LOSSES	LOSSES		VOLUME, m <sup>3</sup>	DIAMETER, m	LENGTH, m				
15	4448 (1000)	4	MLI	F 5095	182	0	21420	6.71	2.63	1.86	2663	23247	3968	3.38	
				0 15285	674	184		15.0	4.27	1.48					
16		8	MLI	F 4939	186	0	20826	6.52	2.60	1.84	2649	22651	4564	3.35	
				0 14815	674	212		14.5	4.27	1.45					
17		4	SOFI	F 4860	181	0	22140	6.41	2.59	1.83	2589	23910	3305	3.45	
				0 14579	658	1826		15.7	4.20	1.55					
18		8	SOFI	F 4733	185	0	21840	6.26	2.57	1.82	2588	23614	3602	3.44	
				0 14200	667	2056		15.6	4.20	1.54					

\*INCLUDES INSULATION THICKNESS AND ENGINE LENGTH.

TABLE VIII-4 PROPELLANT SYSTEM CHARACTERISTICS														
Tanking Configuration: Parallel Tanks														
Initial Vehicle Mass: 27216 kg														
CONFIG.	THRUST N (lb <sub>f</sub> )	NUMBER OF BURNS	INSULATION CONCEPT	PROPELLANT MASS, Kg				TANKS +			LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg	PAYLOAD MASS, kg	OVERALL, * LENGTH, m
				USABLE (ΔV)	TRAPPED, START-S/D	BOILOFF LOSSES	TOTAL	VOLUME, m <sup>3</sup>	DIAMETER, m	LENGTH, m				
19	2224 (500)	4	MLI	F	4306	136	80	21114	5.73	1.60	3.24	22983	4233	4.35
				O	15933	465	194		7.74	1.89	3.22			
20		8	MLI	F	4139	141	90	20339	5.53	1.60	3.14	22204	5012	4.27
				O	15295	459	219		7.45	1.88	3.12			
21		4	SOFI	F	4085	135	504	22079	5.89	1.58	3.38	23871	3345	4.52
				O	15116	448	1790		8.01	1.87	3.37			
22		8	SOFI	F	3944	141	546	21610	5.77	1.56	3.33	23407	3809	4.48
				O	14593	447	1942		7.84	1.86	3.32			
23	4448 (1000)	4	MLI	F	4207	136	76	20632	5.60	1.60	3.17	22525	4691	4.45
				O	15567	464	182		7.57	1.89	3.15			
24		8	MLI	F	4071	142	86	20032	5.44	1.60	3.10	21923	5293	4.38
				O	15062	463	209		7.34	1.89	3.07			
25		4	SOFI	F	3989	136	472	21486	5.73	1.58	3.29	23298	3918	4.58
				O	14758	446	1683		7.80	1.87	3.28			
26		8	SOFI	F	3881	142	521	21203	5.67	1.58	3.27	23022	4194	4.57
				O	14361	450	1849		7.69	1.86	3.26			

\*INCLUDES INSULATION THICKNESS AND ENGINE LENGTH.  
+EACH TANK

TABLE VIII-5 PROPELLANT SYSTEM CHARACTERISTICS FOR CONVENTIONAL TANDEM TANK CONFIGURATIONS

INITIAL VEHICLE MASS: 27216kg																
PROPELLANT	MIXTURE RATIO	Isp N·sec (sec)	TOTAL ΔV, m/s	LEO TO GEO hr	PROPELLANT MASSES, kg				TOTAL	TANK			LTPS BURNOUT MASS, kg	LTPS LIFTOFF MASS, kg	PAYLOAD MASS, kg	OVERALL LENGTH,* m
					USABLE (ΔV)	TRAPPED START-S/O LOSSES	BOILOFF LOSSES	VOLUME m <sup>3</sup>		DIAMETER, m	LENGTH, m					
LO <sub>2</sub> / LH <sub>2</sub>	6	4315 (440.0)	4448	19.8	F	2477	77	181	18237	41.8	4.27	3.93	2439	20207	7009	7.80
					O	14859	420	223		14.4	3.39	2.40				
LO <sub>2</sub> / LCH <sub>4</sub>	3.7	3496 (356.5)	4441	18.8	F	4134	136	83	20311	10.9	3.09	2.19	2323	22102	5114	6.06
					O	15295	437	226		14.9	3.42	2.42				
LO <sub>2</sub> / RP-1	3	3270 (333.5)	4439	18.5	F	5025	154	0	20905	6.59	2.61	1.85	2271	22635	4581	5.67
					O	15076	424	225		14.6	3.41	2.41				

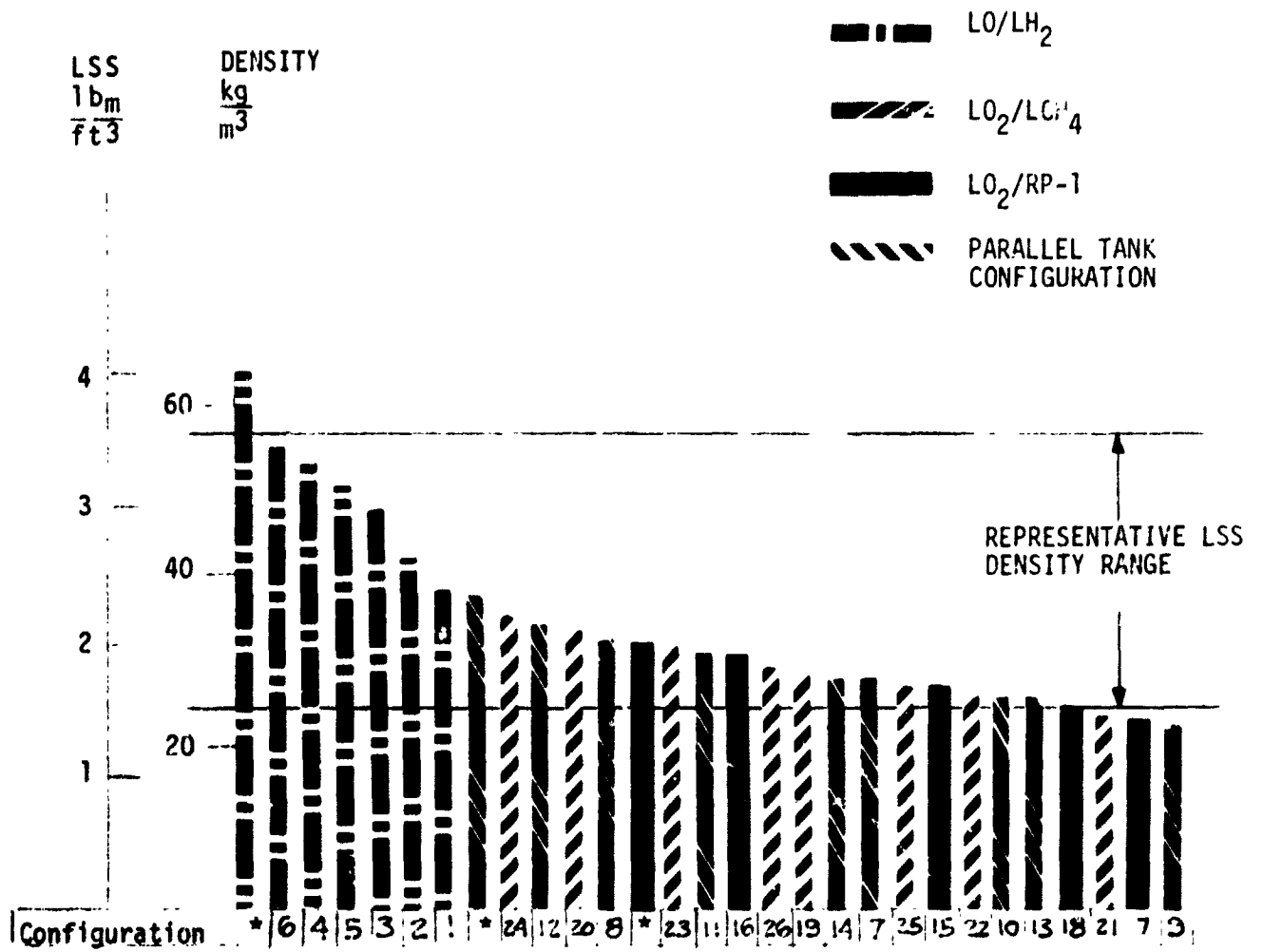
\*Includes Insulation Thickness and Engine Length

TABLE VIII-6 MASS AND LENGTH AVAILABLE TO THE LSS

LTPS Configuration					LSS Characteristic		
PROPELLANT COMBINATION	NUMBER	THRUST, N	NUMBER OF BURNS	INSULATION CONCEPT	AVAILABLE LENGTH, m	AVAILABLE MASS, kg	PACKAGED DENSITY, kg/m <sup>3</sup> (lbm/ft <sup>3</sup> )
LO <sub>2</sub> /LH <sub>2</sub>	1	445	4	MLI	8.50	4613	38 (2.4)
	2		8		8.60	5120	42 (2.6)
	3	2224	4		8.84	5920	47 (2.9)
	4		8		8.99	6751	53 (3.3)
	5	4448	4		8.96	6285	49 (3.1)
	6		8		9.08	6967	54 (3.3)
	TASK III	2224	8		7.65	7009	64 (4.0)
LO <sub>2</sub> /LCH <sub>4</sub>	7	2224	4	MLI	10.70	4173	27 (1.7)
	8		8		10.76	4950	32 (2.0)
	9	4448	4	SOFI	10.58	3371	22 (1.4)
	10		8		10.58	3862	26 (1.6)
	11	4448	4	MLI	10.61	4595	30 (1.9)
	12		8		10.64	5209	34 (2.1)
	13	4448	4	SOFI	10.58	3900	26 (1.6)
	14		8		10.61	4203	28 (1.7)
	19	2224	4	MLI	10.42	4233	28 (1.8)
	20		8		10.52	5012	33 (2.1)
	21	4448	4	SOFI	10.24	3345	23 (1.4)
	22		8		10.27	3809	26 (1.6)
	23	4448	4	MLI	10.33	4690	32 (2.0)
	24		8		10.39	5293	36 (2.2)
	25	4448	4	SOFI	10.18	3917	27 (1.7)
	26		8		10.18	4193	29 (1.8)
TASK III	2224	8	MLI	9.48	5113	38 (2.4)	
LO <sub>2</sub> /RP-1	15	4448	4	MLI	11.09	3968	25 (1.6)
	16		8		11.16	4564	29 (1.8)
	17	4448	4	SOFI	11.03	3305	21 (1.3)
	18		8		11.06	3602	23 (1.4)
	TASK III	2224	8	MLI	10.03	4581	32 (2.0)

1 m = 3.281 ft

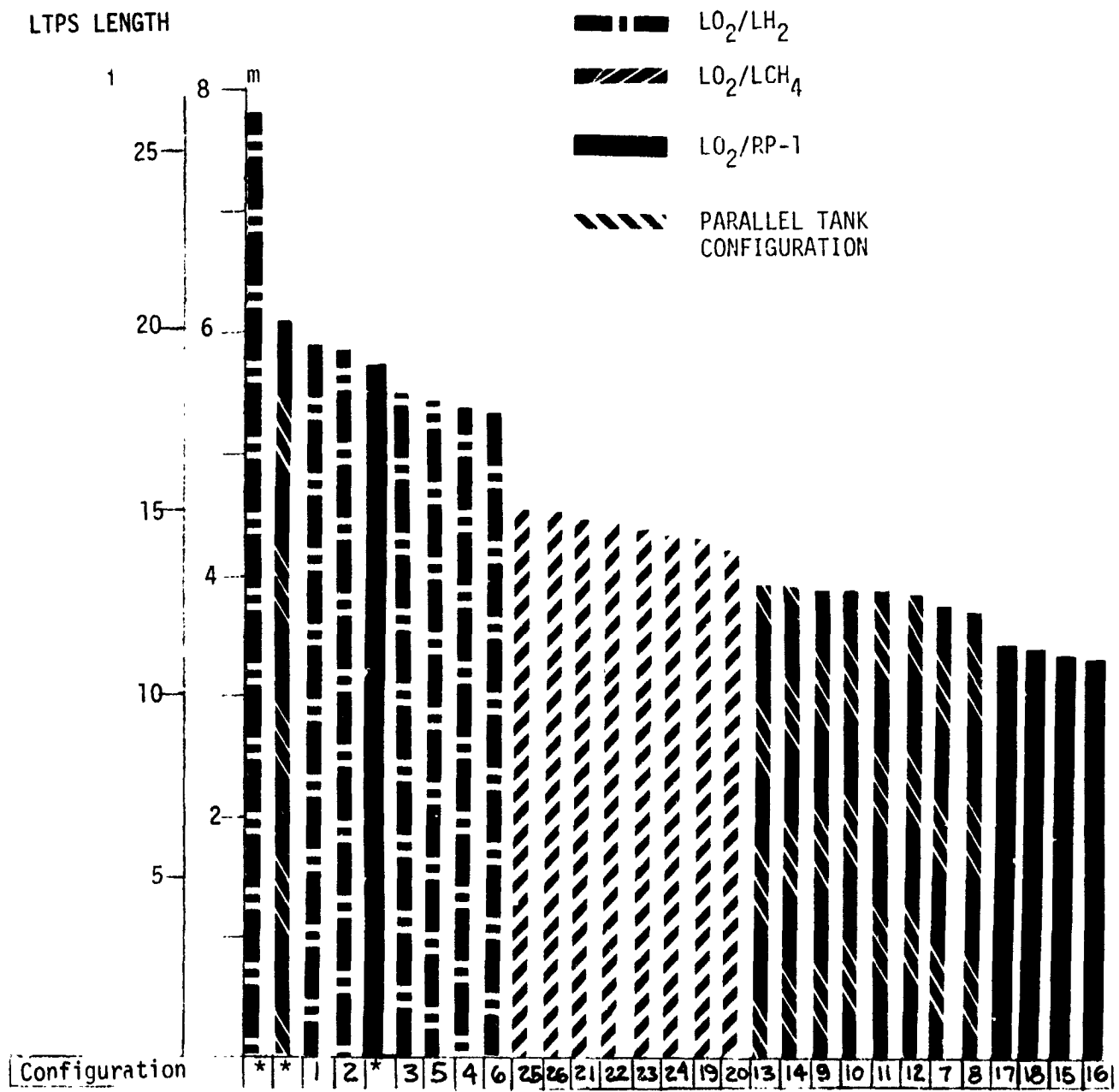
1 kg = 2.205 lb<sub>m</sub>



CONFIGURATIONS MARKED WITH AN ASTERISK (\*) DENOTE MAXIMUM PERFORMANCE CONFIGURATIONS DESCRIBED IN SECTION IV.

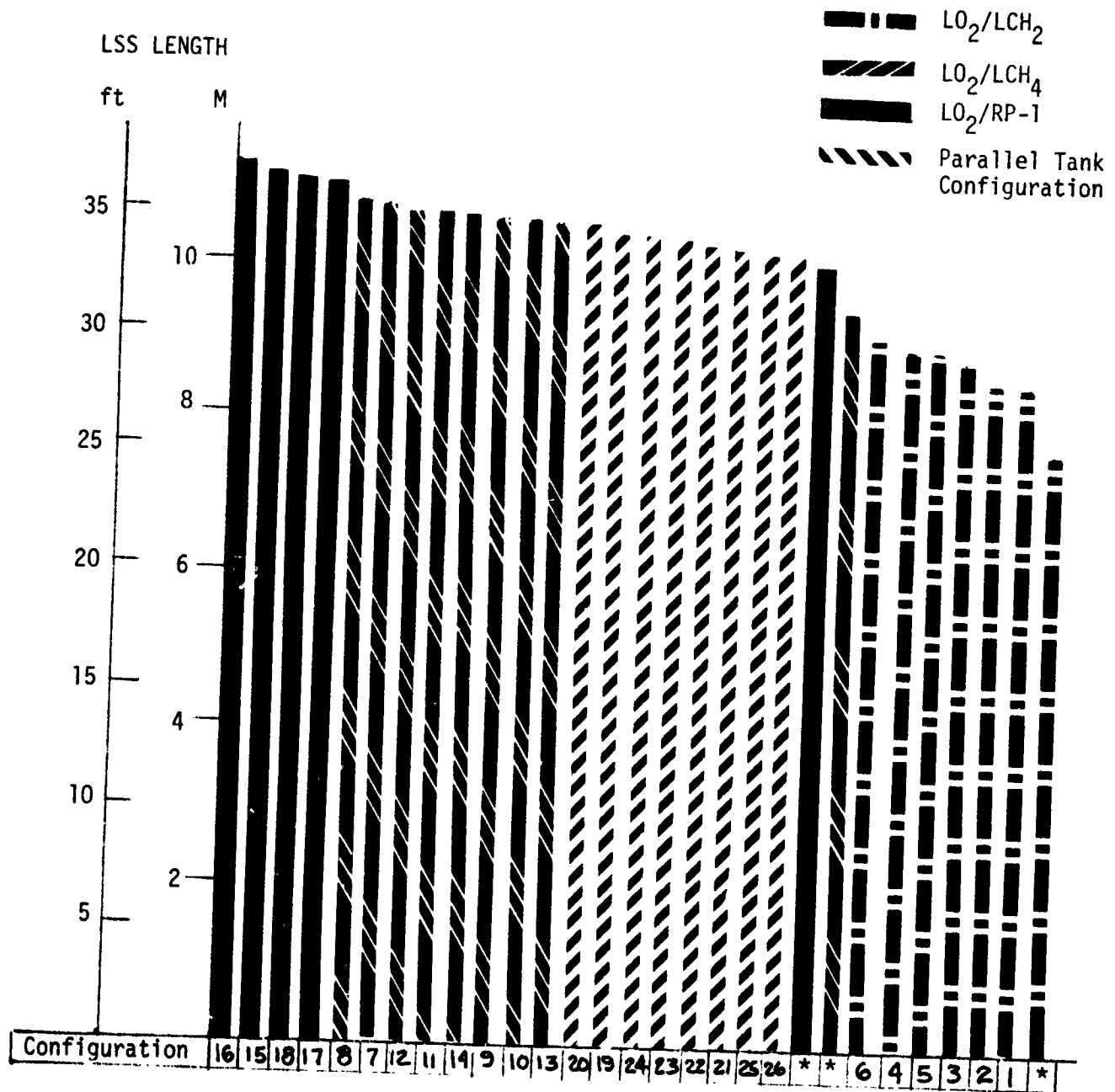
FIGURE VIII-1 LSS DENSITIES FOR SELECTED CONFIGURATIONS





CONFIGURATION MARKED WITH AN ASTERICK (\*) DENOTES MAXIMUM PERFORMANCE CONFIGURATIONS DESCRIBED IN SECTION IV

FIGURE VIII-2 LTPS LENGTH FOR SELECTED CONFIGURATIONS



CONFIGURATIONS MARKED WITH AN ASTERISK (\*) DENOTE MAXIMUM PERFORMANCE CONFIGURATIONS DESCRIBED IN SECTION IV.

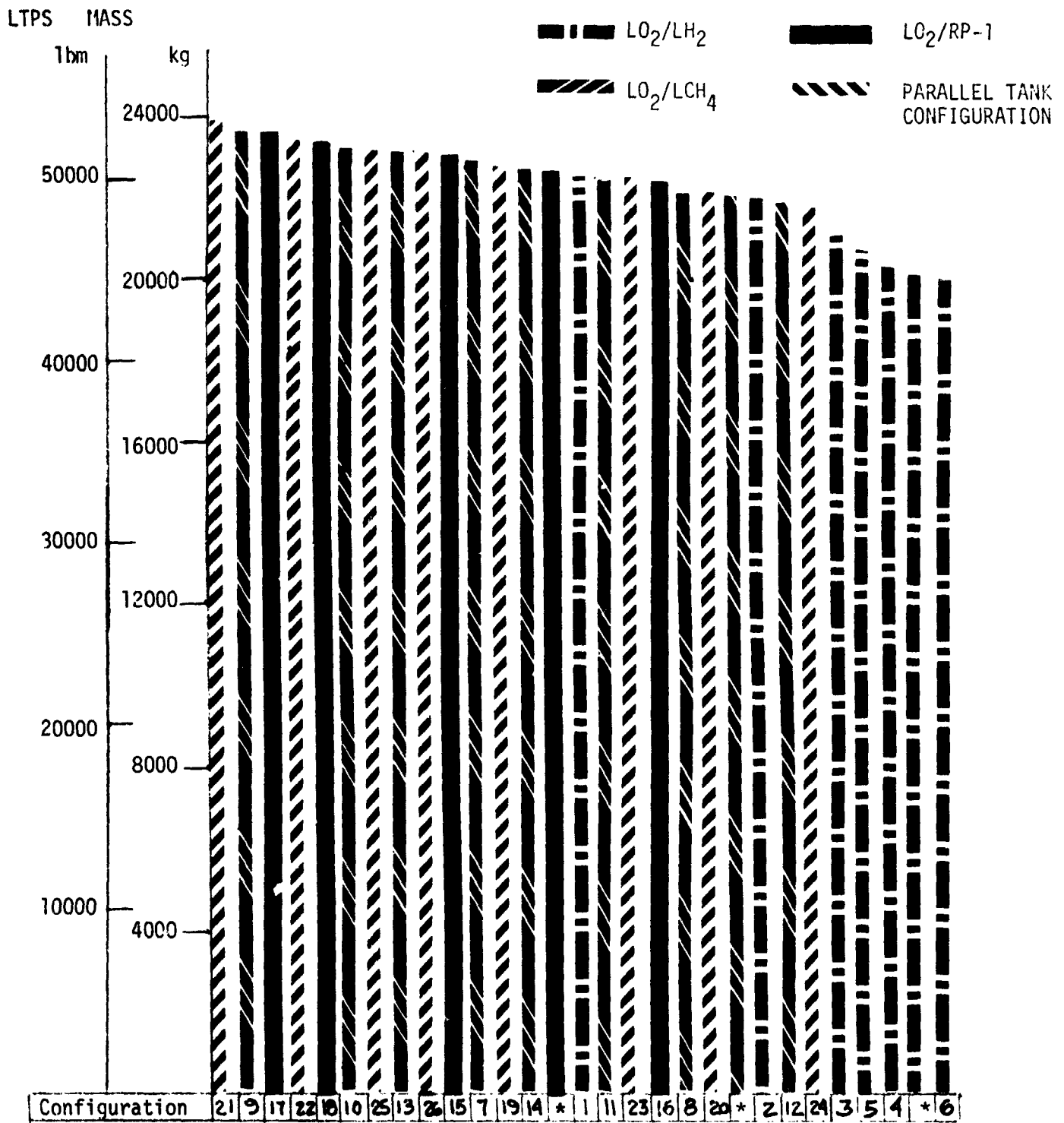
FIGURE VIII-3 LSS LENGTH FOR SELECTED CONFIGURATIONS

seen that the toroidal tank is needed to reduce overall vehicle length to provide sufficient room for the LSS. Direct comparison of LTPS lengths is not always an accurate method of determining comparative LSS lengths because of the varying aft clearance requirement in the Orbiter payload bay (see Section VI) which is a function of tank configuration. For example, the maximum performance  $LO_2/LCH_4$  vehicle is longer than all  $LO_2/LH_2$  minimum length systems but the  $LO_2/LCH_4$  vehicle would allow a longer LSS to be stowed with it in the Orbiter. Comparison of masses is straightforward since the mass available for an LSS is always 27,216 kg minus the mass of the LTPS. Both propulsion system and maximum allowable payload masses are displayed in Figures VIII-4 and VIII-5 respectively. In comparing vehicle lengths, the  $LO_2/RP-1$  tandem/toroidal systems produce very short vehicles but the mass available for the LSS payload is low.  $LO_2/LH_2$  systems produce opposite effects; they are long systems but are also the lightest. Both methane fueled tank arrangements analyzed produced systems similar in mass and space available for the payload. Since both systems could transfer a comparable LSS, the parallel tanks arrangement becomes very attractive because of reduced developmental problems.

The results predict the use of an  $LO_2/LH_2$ , tandem/toroidal arrangement for shorter, more dense payloads. While the lighter, longer payloads could be accommodated by a  $LO_2/LCH_4$  system using either a tandem/toroidal or parallel tanks configuration. Although the  $LO_2/RP-1$  system may reduce thermal problems, its low performance produces vehicles too heavy to allow full utilization of the Shuttle capabilities.

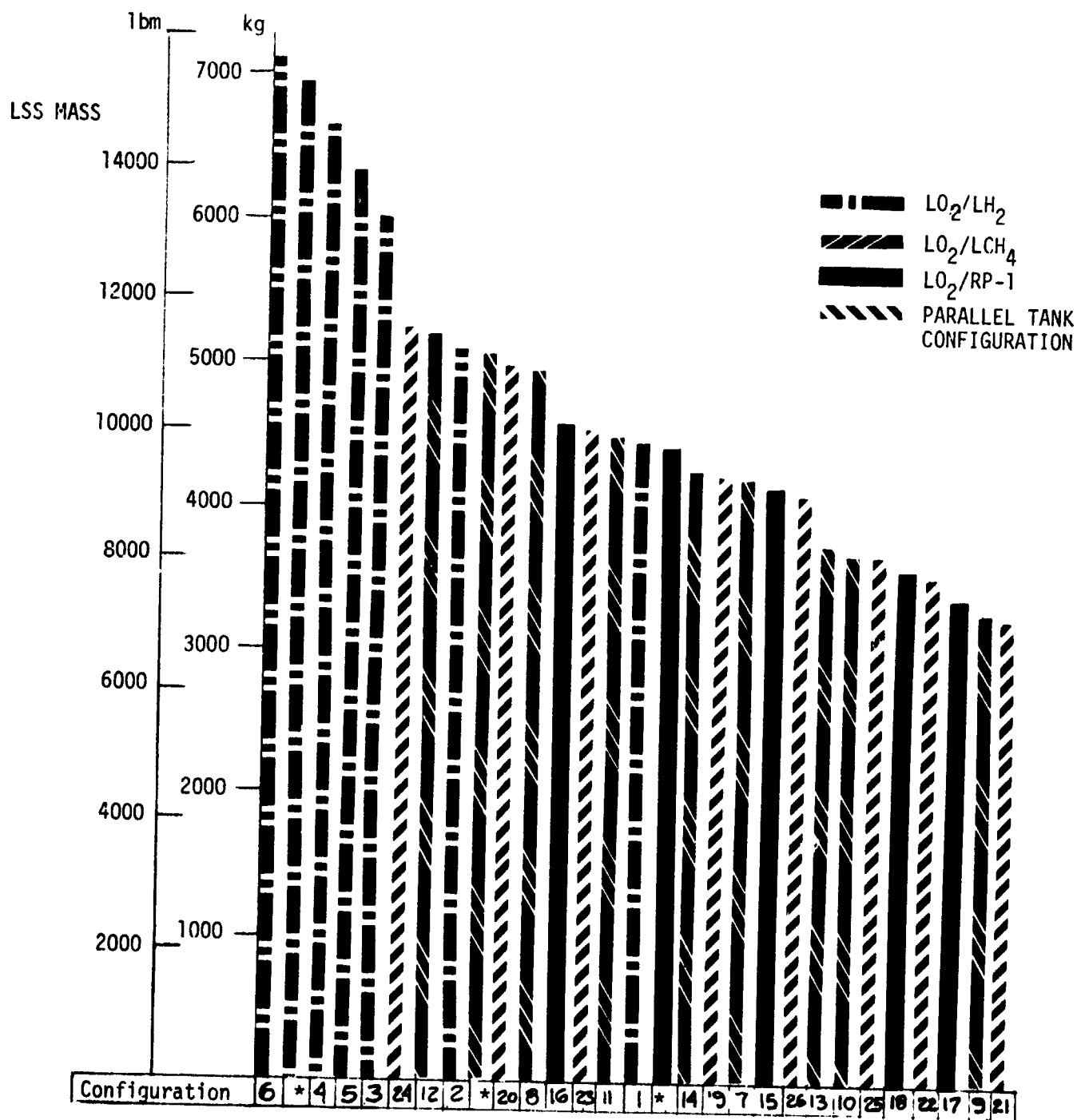
#### B. PROPELLANT MANAGEMENT

The length of the LTPS was not strongly affected by the propellant management approach but difference in system mass was as much as 200 kg. The approach that produced the lowest weight penalty was a combination of propulsive settling and screen devices introduced in Section V. The three short vehicles that were reevaluated with this combination produced a weight penalty that was lower than for any of the separate approaches. The improved approach combined propulsive settling and a screen over the outlet to delay



CONFIGURATIONS MARKED WITH AN ASTERICK (\*) DENOTE MAXIMUM PERFORMANCE CONFIGURATIONS DESCRIBED IN SECTION IV.

FIGURE VIII-4 LTPS MASS FOR SELECTED CONFIGURATIONS



CONFIGURATION MARKED WITH AN ASTERISK (\*) DENOTE MAXIMUM PERFORMANCE CONFIGURATION DESCRIBED IN SECTION IV

FIGURE VIII-5 LSS MASS FOR SELECTED CONFIGURATIONS

propellant dropout, in the tanks with an ellipsoidal shaped bottom, or a screen channel in the bottom of the toroid. This approach produced less propellant residuals, even when the engine was gimbaled at 10 degrees, than the simple settling approach used in Section III. These results point to a combination of settling and some form of screen device as the simplest and lightest approach for propellant management during orbital transfer.

C. TECHNOLOGY DEFICIENCIES

The problems that need to be solved if an LTPS vehicle is to be built are listed in Table VI-1. The two highest priority items would be tests to determine performance of screen devices with cryogenic propellants and development of improved propellant settling models.

## APPENDIX A

### SAMPLE PROP PRINTOUTS

The computer sizing program, PROP, was described in Section II-B. This appendix presents a dictionary of the input variables and sample inputs and outputs for a number of selected cases, each case has four pages of printout. The input dictionary follows on the next five pages and explains which variables are required for each option, what quantity the input label represents, and the units that the program assumes for each variable. Tables A-1 through A-4 follow the dictionary, and these tables show a representative case. The first sheet, Table A-1, lists the input variables and their values. Table A-2 is the second page of the output and this output predicts the remaining mass and volume of propellant, and ullage height, at the beginning of all burns for each propellant. The ullage height is the length of the inside of the tank minus the height of the propellant if it was all settled in the bottom of the tank. Also calculated at the initiation of each burn are the total system mass and acceleration along with the burn duration. The same variables, except ullage height and burn duration, are also computed at the end of the circularization burn. The final outputs in Table A-2 are the propellant tank dimensions. The third and fourth pages, Tables A-3 and A-4, show the results of the system sizing in English and SI units respectively.

The rest of the configurations presented in this appendix are configuration numbers 6, 16, 24, 26 and the three maximum performance configurations (see Table II-14 for the configuration numbers of various systems).

## PROP VARIABLE LABEL DICTIONARY

Variables appear in alphabetical order except for DVU, DVB, WPU, and WPB. The out of sequence order of these four variables is intended to make their explanations easier to follow. A variable in parentheses is the label for that variable when it is applied to the oxidizer or pressurizing gas system. Inside the square brackets following the explanation are the units that the program requires the input to be in. If the input is required in all cases an "-R-" follows the variable label while an "-O-" designates an optional input. Any cases where the optional variable becomes a required input are specified in the explanation given for that label.

The Fortran format for an input is 10F8.0. All input variables that are not required should be input as zero.

ATRPF (ATRPO)	-O-	A mass input for trapped and/or residual fuel (oxidizer). [lb <sub>m</sub> or kg]
BDR	-O-	Blowdown Ratio, required input only if system is a blowdown case.
BTRPF (BTRPO)	-O-	A fraction of the <u>total usable</u> propellant allocated for residual fuel (oxidizer).
CTRPF (CTRPO)	-O-	A fraction of the <u>total amount</u> of fuel (oxidizer) allowed for residuals.
DPRG	-R-	<ul style="list-style-type: none"><li>o Helium Pressurization System, DPRG is the pressure drop across the regulator [psi or Pa]</li><li>o Blowdown Case, DPRG=0.0</li><li>o All others, DPRG &lt; 0.0 (the computer assumes an external pressurization concept that requires no sizing by the program).</li></ul>
DVU	-O-	The total velocity change required for orbit transfer. Used to calculate the weight of usable propellant from the ideal velocity equation. [ft/sec or m/sec]



DVB -0- The amount of velocity change for the vehicle that is accomplished burning the propellant isothermally. The remaining propellant is assumed to be burned adiabatically. [ft/sec or m/sec]

D1F (D10) -0- Fuel (oxidizer) tank diameter. [in or m]

- o Cylindrical/domed tanks, required input.
- o Toroidal tanks, requires an input for D1F (D10) or D2F (D20).
- o Spherical or ellipsoidal tanks, no input required as the program calculates the diameter. (Note: if the cylindrical tank option is chosen and a spherical or ellipsoidal tank of the same volume can be sized with a diameter less than D1F (D10) then the program will default to the sphere or ellipsoid option.)

D2F (D20) -0- Inner diameter of fuel (oxidizer) toroidal tank. [in or m] - Toroidal tank must have an input for either D1F (D10) or D2F (D20).

ENGT -R- Total number of engines.

FCRYO (OCRYO) -R- Option to specify if fuel (oxidizer) is a cryogenic [1.0] or storable [0.0] propellant.

FNOFP (FNOPO, FNOFG) -R- Non-optimum factor applied to the fuel (oxidizer, gas) tank mass to account for welds, flanges or tank supports. [ $\geq 1.0$ ]

FNOFV (FNOFGT) -R- Non-optimum factor used in the propellant (gas) tank volumes to account for PMDs, internal stringers or other tank intrusions. [ $\geq 1.0$ ]

FSFT (FSOT, FSGT) -R- Safety factor for the fuel (oxidizer, gas) tank. [ $\geq 1.0$ ]

FU (OU) -0- Fraction of volume to be allowed for initial ullage inside fuel (oxidizer) tank.

GAM -R- Gamma, ratio of specific heats for pressurizing gas.

GR -R- Ratio of g for the mission divided by g for the earth.

ISP -R- Specific impulse [lb<sub>f</sub>-sec/lb<sub>m</sub> or N-sec/kg]

**MOB** -R- Mono or Bipropellant option.  
o Monopropellant, MOB=1.0  
o Bipropellant, MOB=2.0

**MOE** -R- Metric or English units option  
o English inputs/English outputs, MOE=1.0  
o English inputs/Metric outputs, MOE=2.0  
o English inputs/English and Metric outputs, MOE=3.0  
o Metric inputs/Metric outputs, MOE=4.0

**MR** -0- Mixture ratio, required if bipropellant option is used.

**MVOI2** -0- For engine weight calculation.

**NFSHAP (NOSHAP)** -R- Defines tank shape for fuel (oxidizer) tank.  
o Spherical Tank, 1.0  
o Cylindrical with Hemispherical Dome Ends, 2.0  
o Cylindrical with  $\sqrt{2}$  Ellipsoidal Dome Ends, 3.0  
o  $\sqrt{2}$  Ellipsoidal Tank, 4.0  
o Toroidal Tank, 5.0

**NFT (NOT,NGT)** -R- Number of fuel (oxidizer, gas) tanks.

**PC** -0- Engine chamber pressure, used when PROP is to size the engine, 1.0 otherwise. [psi or Pa]

**PGTI** -0- Initial pressure of the gas tank, required only if a regulated case is used. [psi or Pa]

**PMF (PMO)** -R- Maximum pressure that the fuel (oxidizer) tank must withstand. [psi or Pa]

**PUF1 (PUO1)** -R- Initial ullage pressure in fuel (oxidizer) tank. [psi or Pa]

**RG** -R- Gas constant of pressurizing gas.  
[ft-lb<sub>f</sub>/lb<sub>m</sub>-°R or m-N/kg-°K]

**RHOF (RHOO)** -R- Density of fuel (oxidizer). [lb<sub>m</sub>/ft<sup>3</sup> or kg/m<sup>3</sup>]

**RHOM (RHOMG)** -R- Density of material used to construct the propellant (gas) tanks. [lb<sub>m</sub>/in<sup>3</sup> or kg/m<sup>3</sup>]

**STARTS** -R- Number of perigee burn starts.

**SULT (SULTG)** -R- Ultimate strength of material used to construct propellant (gas) tanks. [psi or Pa]

**TB** -0- Burn Time, not required if engine weights are known. [sec]

**TG2** -0- Temperature of the gas tank environment at the end of the adiabatic burn. [°R or °K]

TMIN	-R-	Minimum allowable thickness of tank wall. [in or m]
TPER	-0-	Thrust per engine, not required if engine weight is known. [lbf or N]
TSI	-R-	Initial system temperature. [°R or °K]
TTW	-0-	Thrust to weight ratio, required only if engine weights are unknown.
VFT (VOT)	-0-	Volume of fuel (oxidizer) tank, may be input if known. [ft <sup>3</sup> or m <sup>3</sup> ]
VTOP	-R-	Tank volume option. o If tank volume is known, VTOP= 1.0 o If PROP is to calculate tank volume, VTOP=0.0
WENGT	-0-	Mass of engine. If no input then program will calculate engine mass. [lb <sub>m</sub> or kg]
WI	-R-	Initial mass of vehicle and payload at disconnect. Required input if WPU or WPB is unknown. [lb <sub>m</sub> or kg]
WMSC	-R-	Mass of miscellaneous propulsion system components. [lb <sub>m</sub> or kg]
WPU	-0-	Mass of usable propellant. Input if known otherwise input value for DVU. [lb <sub>m</sub> or kg]
WPB	-0-	Mass of usable propellant burned isothermally. Input if known, otherwise input value for DVB. The rest of the propellant is assumed to be burned adiabatically. [lb <sub>m</sub> or kg]
WPLUM	-R-	Mass of plumbing system for engines. [lb <sub>m</sub> or kg]
WPRESS	-0-	Mass of non-tank pressurization hardware. [lb <sub>m</sub> or kg]
WSTOPF (WSTOPO)	-R-	Mass of fuel (oxidizer) used at engine tailoff. [lb <sub>m</sub> or kg]
WSTRTF (WSTRTO)	-R-	Mass of fuel (oxidizer) required for engine chilldown or startup, prior to ignition. [lb <sub>m</sub> or kg]

The following properties are needed if either FCRYO=1.0 or OCRYO=1.0.

TGRND -R- On-ground temperature of external layer of insulation. [°R or °K]

TMEGND -R- Time during which on-ground thermal conditions exist. [hr]

TMELCO -R- Time on orbit before first ignition, for erection and checkout. [hr]

TMETST -R- Orbital transfer time. [hr]

TORB -R- On-orbit temperature of external layer of insulation. [°R or °K]

The following properties are needed if FCRYO= 1.0 (OCRYO=1.0).

ACONDF (ACONDO) -R- Total cross-sectional area for heat conduction through the fuel (oxidizer) support struts. [ft<sup>2</sup> or m<sup>2</sup>]

HFGF (HFGO) -R- Latent heat of vaporization for fuel (oxidizer). [Btu/lb<sub>m</sub> or J/kg]

KGRNDF (KGRNDO) -R- Thermal conductivity of fuel (oxidizer) tank insulation when the vehicle is on-ground. [Btu/hr-ft-°F or W/m-°C]

KORBF (KORBO) -R- Thermal conductivity of fuel (oxidizer) tank insulation when the vehicle is on orbit. [Btu/hr-ft-°F or W/m-°C]

RHOINF (RHOINO) -R- Density of insulation covering the fuel (oxidizer) tank. [lb<sub>m</sub>/ft<sup>3</sup> or kg/m<sup>3</sup>]

THKINF (THKINO) -R- Thickness of insulation covering on the fuel (oxidizer) tank. [in or M]

TPROFF (TPROPO) -R- Fuel (oxidizer) temperature at tank liftoff pressure. [°R or °K]

QCONDF (QCONDO) -R- Penetrating strut heat leak rate per unit area for fuel (oxidizer) supports. [Btu/hr-ft<sup>2</sup> or W/m<sup>2</sup>]

TABLE A-1 PROP INPUTS

Configuration Number  
 Propellant Combination  
 Insulation  
 Perigee Burns  
 Propellant Management Approach

```

*1 LOX/LM2 MLI 100 LBF THRUST 4 BURN - PROPELLANT SETTLING
***INPUT***
OVJ = 17294.80 RMOJ = 4 198 PMD = 24 0 AIRPF = 20 500 MR = 6 000 FSFT = 1.500
OV8 = 17294.80 RMO8 = 69.040 PG11 = 0 BIRPF = 020 FU = 020 FSOT = 1.500
TSP = 422 50 RMO4 = 103 T51 = 530 000 CIRPF = 005 UU = 020 FSG1 = 0.000
WI = 60000 00 RMO0 = 0.000 T1W = 0 000 ATRPU = 198 500 NFI = 1 FNDPF = 1.700
WPU = 0 00 GAM = 1 660 ENGT = 1 000 BIRPO = 020 NG1 = 0 FNDFO = 1.500
WPB = 0 00 RG = 386 000 WENGT = 25 000 CIRPU = .005 NGT = 0 FNDPG = 0.000
TB = 0 00 GR = 1.000 TPER = 100 000 MOSHAP = 5 D10 = 168 000 FNDPV = 1.005
MOB = 2 DRRG = -1 0 MVO12 = 0 NFSHAP = 3 D1F = 168 000 FNDFT = 0.000
SUL1 = 69E+05 PDR = 0 000 WPLUM = 50 000 VFT = 0 000 D2U = 0 000 VTOP = 0.000
SULTG = 0. PMF = 24 0 WMSL = 3250 000 VDT = 0 000 D2F = 0 000 TMIN = 0.020

PC = 1 0 WSTRIF = 1.000 WSTR... = 2 500 WSTOPF = 1 000 WSTOP = 2.500 STARTS = 4.
WFRESS = 200 000 MDE = 3 FCRVC = 1 0 OCRYD = 1.0 TG2 = 530.000 PUF1 = 24.000
TMELCO = 42.00 TMELIST = 61 35 TMGRND = .15 TGRND = 530 00 TORB = 570.00
PHOINF = 3.510 THAKINF = 1.11000 TPRODF = 39 69 KGRNDF = 3500E-01 KORBF = 4900E-04 UCOND = 11270 0
ACONDF = .050 HEGF = 187 04 TPROFD = 171 28 KGRNDD = 3500E-01 KOREB = 4100E-04 UCOND = 8257 0
PHOIMO = 3 510 THAKIMO = 1.32000
ACONDO = 050 HEGO = 89 39
    
```

TABLE A-2 CONDITIONS AT THE INITIATION OF EACH BURN

CONFIGURATION	1.	THRUST	100.0 LBF.	OX. FLOW RATE	.2029 LBM/SEC.	FUEL FLOW RATE	.0338 LBM/SEC.	ISP	422.5 SEC	
BURN NUMBER										
	1	38140.2	555.2	3.99	6543.3	1566.5	22.59	47087	59417.1	.17E-02
	2	28499.3	414.9	19.45	4888.4	1170.3	58.18	38135.	48121.4	21E-02
	3	20674.5	301.0	29.29	3536.1	846.6	83.42	30863.	38944.3	.26E-02
	4	14325.2	208.5	37.12	2429.8	581.7	104.07	24954.	31488.6	.32E-02
	5	9174.5	133.6	43.80	1523.2	364.7	121.12	38343	25431.4	39E-02
END OF MISS'ON		1123.0	16.3	---	180.1	42.9	---	---	16036.8	.62E-02
OXIDIZER TANK SHAPE IS TOROIDAL										
TANK VOL = 571.0 FT <sup>3</sup> . INNER DIAM = 45.6 IN. OUTER DIAM = 168.0 IN. TANK HEIGHT = 61.2 IN										
FUEL TANK SHAPE IS CYL/SORT2 DCME										
TANK VOL = 1662.7 FT <sup>3</sup> . DOME VOL = 508.0 FT <sup>3</sup> . TANK DIAM = 168.0 IN. TANK LENGTH = 169.2 IN. BARREL LENGTH = 50.42 IN										

TABLE A-3 PROPELLANT AND SYSTEM CHARACTERISTICS-ENGLISH UNITS

#1	LOX/LH2	MLI	100 LBF THRUST	4 BURN	PROPELLANT SETTLING
VEHICLE MASS	=60000.0 LBM		DELTA V=	17294.8 FPS	AVE. ISP= 422.5 SEC.
TOTAL PROPELLANT				45265.47 LBM	
USABLE FUEL				6109.92	
USABLE OXIDIZER				36659.52	
FUEL TRAPPED				180.14	
OXID TRAPPED				1123.01	
FUEL START-S/D LOSSES				10.00	
OXID START-S/D LOSSES				25.00	
FUEL BOILOFF				512.02	
OXIDIZER BOILOFF				646.73	
OXIDIZER TANKS (NO = 1)				232.52	
(TOROIDAL)					
INNER DIA=	45.627 IN				
OUTER DIA=	168.000 IN				
HEIGHT =	61.186 IN				
VOLUME =	570.995 FT <sup>3</sup>				
AVG THK =	.02333 IN				
FS = 1.50.	FNOP= 1.50				
FUEL TANKS (NO = 1)				411.00	
(CYLINDRICAL/SORT(2) ELLIPTICAL)					
DIAMETER=	168.000 IN				
LENGTH =	169.211 IN				
VOLUME =	1662.726 FT <sup>3</sup>				
DOME THK=	.02645 IN				
CYL THK =	.04383 IN				
FS = 1.50.	FNOP= 1.30				
PRESSURANT				740	
PRESSURANT SYSTEM MASS				200.000	
FUEL TANK INSULATION				222.25	
OXIDIZER TANK INSULATION				172.95	
ENGINES (NO = 1)				25.00	
(THRUST/ENG= 100.0 LBF)					
COMPONENTS AND LINES				50.00	
ENG. MOUNTS,SUPPORTS				3250.00	
TOTAL WET SYSTEM MASS				49829.9	
TOTAL BURNOUT MASS				5866.7	
(INCL NON-USABLE PROPEL AND GAS)					
MASS FRACTION				858	
TOTAL IMPULSE				18070086.2 LBF-S	
PRESSURE SCHEDULE(PSI ) AT T=530.0 R					
GAS TANK LOCK-UP PRESSURE	= 0			INITIAL CHAMBER PRESSURE	= 1.000
INITIAL OX SYS PRESSURE	= 24.00			FINAL OX SYS PRESSURE	= 24.00
INITIAL FU SYS PRESSURE	= 24.00			FINAL FU SYS PRESSURE	= 24.00
BURN TIME=	180700.86 SEC				

TABLE A-4 PROPELLANT AND SYSTEM CHARACTERISTICS - METRIC UNITS

#1 LOX/LH2 MLI 100 LBF THRUST 4 BURN - PROPELLANT SETTLING

VEHICLE MASS = 27215.5 KG DELTA V= 5271.5 M/S AVE. ISP=4143.1 N-S/KG

TOTAL PROPELLANT 20532.07 KG

USABLE FUEL 2771.41

USABLE OXIDIZER 16628.48

FUEL TRAPPED 81.71

OXID TRAPPED 509.39

FUEL START-S/D LOSSES 4.54

OXID START-S/D LOSSES 11.34

FUEL BOILOFF 232.25

OXIDIZER BOILOFF 293.35

OXIDIZER TANKS (NO = 1) 105.47  
(TOROIDAL)

INNER DIA= 1.159 M

OUTER DIA= 4.267 M

HEIGHT = 1.554 M

VOLUME = 16.169 M3

AVG THK = .00059 M

FS = 1.50, FNOP= 1.50

FUEL TANKS (NO. = 1) 186.43  
(CYLINDRICAL/SQRT(2) ELLIPTICAL)

DIAMETER= 4.267 M

LENGTH = 4.298 M

VOLUME = 47.083 M3

DOME THK= .00067 M

CYL THK = .00111 M

FS = 1.50, FNOP= 1.30

PRESSURANT 336

PRESSURANT SYSTEM MASS 90.718

FUEL TANK INSULATION 100.81

OXIDIZER TANK INSULATION 78.45

ENGINES (NO) = 1) 11.34  
(THRUST/ENG= 444.8 N)

COMPONENTS AND LINES 22.68

ENG. MOUNTS, SUPPORTS 1474.18

TOTAL WET SYSTEM MASS 22602.5

TOTAL BURNOUT MASS 2661.1  
(INCL. NON-USABLE PROP AND GAS)

MASS FRACTION 858

TOTAL IMPULSE 80379718.8 N-S

PRESSURE SCHEDULE(N/M2 ) AT 1=294.4 K

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 6895

INITIAL OX SYS PRESSURE = .1655E+06 FINAL OX SYS PRESSURE = .1655E+06

INITIAL FU SYS PRESSURE = .1655E+06 FINAL FU SYS PRESSURE = .1655E+06

BURN TIME=180700.86 SEC



#6 LDX/LM2 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

\*\*\*INPUT\*\*\*

DVU = 14479.70 RH01 = 4.189 PMU = 24.0  
 DVH = 14479.70 RH02 = 69.160 P411 = 0  
 ISP = 449.00 RH03 = .103 TS1 = 5.30 060  
 WI = 6000.00 RH04 = 0.000 TTW = 0.000  
 WPU = 0.00 GAM = 1.660 ENGI = 1.000  
 WFB = 0.00 RG = 386.000 WENG1 = 1.15 000  
 TB = 0.00 GR = 1.000 TPER = 1000.000  
 MOB = 0.00 2 DPRG = -1.0 MVO12 = 0  
 SULT = .69E+05 BOR = 0.000 WPLUM = 50.000  
 SULTG = 0. PMF = 24.0 WMSC = 3250.000

MR = 6.000 FSF1 = 1.500  
 FIJ = 0.20 FSU1 = 1.500  
 UU = 0.20 FSU2 = 0.000  
 NF1 = 1 FNQ1F = 1.300  
 N01 = 1 FNQ1O = 1.500  
 NG1 = 0 FNQPG = 0.000  
 D10 = 168.000 FNQPV = 1.005  
 D1F = 168.000 FNQPGT = 0.000  
 D20 = 0.000 VTOP = 0.000  
 D2F = 0.000 TMIN = .020

PC = 1.0 WSTRF = 1.000 WSTRD = 2.500 WSTOPF = 1.000 WSTOP = 2.500 STARTS = 8.  
 WPRESS = 200.000 MOE = 3 FCYD = 1.0 OCRYD = 1.0 TG2 = 530.000 PUJ1 = 24.000  
 TMELCO = 42.00 TMETST = 27.11 TMEGND = 15 TGRND = 530.00 TORB = 530.00  
 RHOINF = 3.510 THKINF = 1.01000 TPROPF = 39.69 KGRNDF = .3500E-01 KORB = 4900E-04  
 ACOND = .050 HGF = 187.04 KGRNDO = 3500E-01 KORB = 4900E-04  
 PHOTO = 3.510 HKJND = 1.21000 TPROPU = 171.28 KORB = 4900E-04  
 ACOND = 0.50 HFGU = 89.39

CONFIGURATION 6, THRUST 1000 0 LBF, OX.FLOW RATE 1.9090 LBM/SEC, FUEL FLOW RATE .3182 LBM/SEC, ISP 449.0 SEC

BURN NUMBER	MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	33827.3	491.6	3.77	5677.5	1362.1	22.19	1887.	59410.6	.17E-01
2	30201.1	438.9	10.74	5061.6	1214.3	37.60	1752.	55168.6	.18E-01
3	26832.2	389.9	15.80	4488.6	1076.9	49.19	1627.	51226.7	.20E-01
4	23702.4	344.4	20.07	3955.4	949.0	59.26	1511.	47563.6	.21E-01
5	20794.6	302.2	23.86	3459.3	829.9	68.54	1403	44159.7	.22E-01
6	18093.3	262.9	27.31	2997.5	719.1	77.18	1302	40996.7	.24E-01
7	15583.8	226.5	30.50	2567.7	616.0	85.22	1209	38057.4	.26E-01
8	13252.5	192.6	33.49	2167.7	520.1	92.70	1122	35326.0	.28E-01
9	11046.8	161.7	33	1795.2	430.7	99.68	5004	32787.9	30E-01
END OF MISSION	1401.1	20.3	---	179.7	42.9	---	---	21486.7	.47E-01

OXIDIZER TANK SHAPE IS TOROIDAL  
TANK VOL = 506.1 FT3, INNER DIAM = 55.3 IN, OUTER DIAM = 168.0 IN, TANK HEIGHT = 56.3 IN

FUEL TANK SHAPE IS CYL/SORT2 DOME  
TANK VOL = 1455.2 FT3, DOME VOL = 508.0 FT3, TANK DIAM = 168.0 IN, TANK LENGTH = 153.0 IN, BARREL LENGTH = 34.24 IN

#6 LOX/LH2 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING  
 VEHICLE MASS = 60000.0 LBM DELTA V = 14179.7 FFS AVE. ISP = 449.0 SEC

TOTAL PROPELLANT 40094.12 LBM  
 USABLE FUEL 5372.20  
 USABLE OXIDIZER 32233.22  
 FUEL TRAPPED 179.74  
 OXID TRAPPED 1401.05  
 FUEL START-S/D LOSSES 13.00  
 OXID START-S/D LOSSES 45.00  
 FUEL BOILOFF 376.84  
 OXIDIZER BOILOFF 468.08

OXIDIZER TANKS (NO = 1) 202.34  
 (TOROIDAL)  
 INNER DIA = 55.313 IN  
 OUTER DIA = 168.000 IN  
 HEIGHT = 56.344 IN  
 VOLUME = 506.137 FT3  
 AVG THK = .02109 IN  
 FS = 1.50, FNOP = 1.50

FUEL TANKS (NO = 1) 360.91  
 (CYLINDRICAL/SQRT(2) ELLIPTICAL)  
 DIAMETER = 168.000 IN  
 LENGTH = 153.037 IN  
 VOLUME = 1455.246 FT3  
 DOME THK = .02645 IN  
 CYL THK = .04383 IN  
 FS = 1.50, FNOP = 1.30

PRESSURANT .650

PRESSURANT SYSTEM MASS 200.000  
 FUEL TANK INSULATION 184.71  
 OXIDIZER TANK INSULATION 152.61

ENGINES (NO = 1) 145.00  
 (THRUST/ENG = 1000.0 LBF)  
 COMPONENTS AND LINES 50.00  
 ENG. MOUNTS, SUPPORTS 3250.00

TOTAL WET SYSTEM MASS 44640.3  
 TOTAL BURNOUT MASS 6127.0  
 (INCL. NON-USABLE PROP AND GAS)

MASS FRACTION .842  
 TOTAL IMPULSE 16884833.5 LBF-S

PRESSURE SCHEDULE (PSI) AT T = 530.0 R

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 1.000  
 INITIAL OX SYS PRESSURE = 24.00 FINAL OX SYS PRESSURE = 24.00  
 INITIAL FU SYS PRESSURE = 24.00 FINAL FU SYS PRESSURE = 24.00

BURN TIME = 16884.83 SEC

#6 LOX/LH2 MLI 1000 LBF THRUST 8 BURN PROPELLANT SETTING  
 VEHICLE MASS = 27215.5 KG DELTA V- 4413.4 M/S AVE. ISP=4403 0 N-S/KG

TOTAL PROPELLANT 18186.39 KG  
 USABLE FUEL 2436.79  
 USABLE OXIDIZER 14620.74  
 FUEL TRAPPED 81.53  
 OXID TRAPPED 635.51  
 FUEL START-S/D LOSSES 8.16  
 OXID START-S/D LOSSES 20.41  
 FUEL BOILOFF 170.93  
 OXIDIZER BOILOFF 212.32

OXIDIZER TANKS (NO. = 1) 91.78  
 (TOROIDAL)  
 INNER DIA= 1.405 M  
 OUTER DIA= 4.267 M  
 HEIGHT = 1.431 M  
 VOLUME = 14.332 M3  
 AVG THK = .00054 M  
 FS = 1.50, FNOP= 1.50

FUEL TANKS (NO. = 1) 163.71  
 (CYLINDRICAL/SQRT(2) ELLIPTICAL)  
 DIAMETER= 4.267 M  
 LENGTH = 3.887 M  
 VOLUME = 41.208 M3  
 DOME THK= .00067 M  
 CYL THK = .00111 M  
 FS = 1.50, FNOP= 1.30

PRESSURANT .295

PRESSURANT SYSTEM MASS 90.718  
 FUEL TANK INSULATION 83.79  
 OXIDIZER TANK INSULATION 69.22

ENGINES (NO. = 1) 65.77  
 (THRUST/ENG= 4448.2 N )  
 COMPONENTS AND LINES 22.68  
 ENG MOUNTS,SUPPORTS 1474.18

TOTAL WET SYSTEM MASS 20248.5  
 TOTAL BURNOUT MASS 2779.2  
 (INCL NON-USABLE PROP AND GAS)

MASS FRACTION .842  
 TOTAL IMPULSE 75107453.9 N-S

PRESSURE SCHEDULE(N/M2 ) AT T=294.4 K

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 6895.  
 INITIAL OX SYS PRESSURE = .1655E+06 FINAL OX SYS PRESSURE = .1655E+06  
 INITIAL FU SYS PRESSURE = .1655E+06 FINAL FU SYS PRESSURE = .1655E+06

BURN TIME= 16884.83 SEC

#16 LOX/RP-1 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

\*\*\*INPUT\*\*\*

DVU = 14438.90 RHDF = 50.700 PMD = 24.0 AIRPF = 95.600 MR = 3.000 FSFI = 1.500  
 DVB = 14438.90 RH00 = 69.130 PGTI = 0.0 BIRPF = .020 FU = 0.020 FSOT = 1.500  
 I5P = 343.00 RHOM = 10.3 LSI = 530.000 CIRPF = .005 OU = 0.020 F5GT = 0.000  
 WI = 60000.00 RHOMG = 0.000 TIW = 0.000 ATRPF = 615.700 PFI = 1.000 FNDPF = 1.300  
 WPU = 0.000 GAM = 1.660 ENGT = 1.000 RTRPO = .020 NGT = 1.000 FNDPO = 1.500  
 WPB = 0.000 RC = 386.000 WENGT = 145.000 CRPO = .005 U10 = 168.000 FNDPG = 0.000  
 TB = 0.000 GR = 1.000 IPER = 1000.000 NDSHAP = 5 U1F = 170.000 FNDPV = 1.005  
 MOB = 0.000 DPRG = -1.0 MVO12 = 0 NFSHAP = 3 U20 = 0.000 FNDPGT = 0.000  
 SULT = .69E+05 BDR = 0.000 WPLUM = 50.000 VFT = 0.000 D2F = 0.000 VTOP = 0.000  
 SULTG = 0. PMF = 24.0 WMSC = 3200.000 VOT = 0.000 D2F = 0.000 TMIN = .020

PC = 1.0 WSTRF = 2.000 WSTRD = 2.500 WSTOP = 2.000 WSTOP = 2.500 STARTS = 8.  
 WPRESS = 200.000 WOE = 3 FCR10 = 0.0 OCRYD = 1.0 IG2 = 530.000 PUF1 = 24.000  
 TMELCO = 42.000 TME1ST = 26.53 TMEGND = 15 TGRND = 530.00 TORB = 5.00  
 RHOINO = 3.510 THKINO = 1.20000 TPRPO = 171.28 KGRND0 = .3500E-01 KORBO = 4900E-04 OCUNDU = 8257.0  
 ACOMDO = 0.50 HFGO = 89.39

CONFIGURATION 16, THRUST 1000 0 LBF, OX.FLOW RATE 2.1866 LBM/SEC, FUEL FLOW RATE .7289 LBM/SEC, ISP 343.0 SEC

BURN NUMBER	MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	34293.7	498.6	2.80	1297.7	225.7	6.02	1866.	59678.3	.17E-01
2	30190.3	438.9	11.52	9937.6	198.6	16.88	1695	54214.9	.18E-01
3	26460.5	384.7	16.98	8702.0	173.9	23.35	1540.	49249.5	.20E-01
4	23070.1	335.4	21.52	7579.6	151.4	28.48	1399.	44736.7	.22E-01
5	19988.3	290.6	25.48	6560.1	131.1	32.86	1271.	40635.4	.25E-01
6	17187.0	249.9	29.03	5634.0	112.6	36.75	1154.	36907.9	.27E-01
7	14640.4	212.8	32.26	4792.9	95.8	40.29	1048.	33520.2	.30E-01
8	12325.5	179.2	35.25	4028.9	80.5	43.58	952.	30441.4	.33E-01
9	10221.1	148.6	38.03	3335.2	66.6	46.67	3977.	27643.2	.36E-01
END OF MISSION	1441.2	20.8	---	374.2	7.4	---	---	15902.3	.63E 01

OXIDIZER TANK SHAPE IS TOROIDAL  
TANK VOL = 513.3 FT3, INNER DIAM = 54.2 IN, OUTER DIAM = 168.0 IN, TANK HEIGHT = 56.9 IN

FUEL TANK SHAPE IS ELLIPSOIDAL  
TANK VOL = 230.2 FT3, DOME VOL = 115.1 FT3, TANK DIAM = 102.4 IN, TANK LENGTH = 72.4 IN, BARREL LENGTH = 0.00 IN

#16 LOX/RP-1 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING  
 VEHICLE MASS = 27215.5 KG DELTA V = 4401.0 M/S AVE ISP = 3363.5 N-S/KG

TOTAL PROPELLANT 20825.80 KG  
 USABLE FUEL 4938.47  
 USABLE OXIDIZER 14815.40  
 FUEL TRAPPED 169.73  
 OXID TRAPPED 653.70  
 FUEL START-S/D LOSSES 16.33  
 OXID START-S/D LOSSES 20.41  
 OXIDIZER BOILOFF 211.75

OXIDIZER TANKS (NO = 1) 93.15  
 (TOROIDAL)  
 INNER DIA = 1.378 M  
 OUTER DIA = 4.267 M  
 HEIGHT = 1.445 M  
 VOLUME = 14.535 M3  
 AVG THK = .00054 M  
 FS = 1.50, FNOP = 1.50

FUEL TANKS (NO = 1) 32.49  
 (ELLIPSOIDAL)  
 DIAMETER = 2.602 M  
 LENGTH = 1.840 M  
 VOLUME = 6.520 M3  
 AVG THK = .00051 M  
 FS = 1.50, FNOP = 1.30

PRESSURANT .112

PRESSURANT SYSTEM MASS 90.718  
 OXIDIZER TANK INSULATION 68.97

ENGINES (NO. = 1) 65.77  
 (THRUST/ENG = 4448.2 N )  
 COMPONENTS AND LINES 22.68  
 ENG MOUNTS, SUPPORTS 1451.50

TOTAL WET SYSTEM MASS 22651.2  
 TOTAL BURNOUT MASS 2648.8  
 (INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .872  
 TOTAL IMPULSE 66445697 O N-S

PRESSURE SCHEDULE (N/M2 ) AT T = 294.4 K

GAS TANK LOCK-UP PRESSURE = 0 INITIAL CHAMBER PRESSURE = 6895.  
 INITIAL OX SYS PRESSURE = .1655E+06 FINAL OX SYS PRESSURE = .1655E+06  
 INITIAL FU SYS PRESSURE = .1655E+06 FINAL FU SYS PRESSURE = .1655E+06

BURN TIME = 14937.59 SEC

#16 LOX/RP-1 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING  
 VEHICLE MASS =60000.0 LBM DELTA V= 14438.9 FPS AVE. ISP= 343.0 SEC

TOTAL PROPELLANT 45913.04 LBM  
 USABLE FUEL 10887.46  
 USABLE OXIDIZER 32662.37  
 FUEL TRAPPED 374.20  
 OXID TRAPPED 1441.16  
 FUEL START-S/D LOSSES 36.00  
 OXID START-S/D LOSSES 45.00  
 OXIDIZER BOILOFF 466.84

OXIDIZER TANKS (NO. = 1) 205.36  
 (TOROIDAL)  
 INNER DIA= 54.247 IN  
 OUTER DIA= 168.000 IN  
 HEIGHT = 56.877 IN  
 VOLUME = 513.297 FT3  
 AVG THK = .02131 IN  
 FS = 1.50, FNOP= 1.50

FUEL TANKS (NO. = 1) 71.63  
 (ELLIPSOIDAL)  
 DIAMETER= 102.427 IN  
 LENGTH = 72.427 IN  
 VOLUME = 230.243 FT3  
 AVG THK = .02000 IN  
 FS = 1.50, FNOP= 1.30

PRESSURANT 246

PRESSURANT SYSTEM MASS 200.000  
 OXIDIZER TANK INSULATION 152.05

ENGINES (NO. = 1) 145.00  
 (THRUST/ENG= 1000.0 LBF)  
 COMPONENTS AND LINES 50.00  
 ENG. MOUNTS, SUPPORTS 3200.00

TOTAL WET SYSTEM MASS 49937.3  
 TOTAL BURNOUT MASS 5839.6  
 (INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .872  
 TOTAL IMPULSE 14937592.3 LBF-S

PRESSURE SCHEDULE (PSI) AT T=530.0 R

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 1.000  
 INITIAL OX SYS PRESSURE = 24.00 FINAL OX SYS PRESSURE = 24.00  
 INITIAL FU SYS PRESSURE = 24.00 FINAL FU SYS PRESSURE = 24.00

BURN TIME= 14937.59 SEC



#24 ADD-ON LOK/LCH4 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

\*\*\*INPUT\*\*\*

DVU = 14448.10 RHOF = 25.270 PMO = 24.0 ATR11 = 36.300 MR = 3.700 FSIT = 1.500  
 DVB = 14448.10 RHOD = 68.610 PGT1 = 0.0 BTR11 = .020 FU = .020 FSOT = 1.500  
 ISP = 364.50 RHOM = .103 TSI = 530.000 CTR11 = .005 UU = .020 FSOT = 0.000  
 WI = 60000.00 RHONG = 0.000 ITM = 0.000 ATR1U = 139.800 NFI = 2 FNOPF = 1.300  
 MPU = 0.00 GAN = 1.660 ENGT = 1.000 BTR1U = .020 NOI = 2 FNOPF = 1.300  
 MPB = 0.00 RG = 386.000 WENGT = 145.000 CTR1U = .005 NGT = 0 FNOPG = 0.000  
 TB = 0.00 GR = 1.000 TPER = 1000.000 NOSHAP = 3 D10 = 74.220 FNOPV = 1.005  
 MOB = .69E+05 DPRG = -1.0 MV012 = 50.000 WFSHAP = 3 D1F = 62.830 FNOPG = 0.000  
 SULT = 0.0 PMF = 24.0 WPLUM = 3250.000 VFT = 0.000 D20 = 0.000 VTUP = 0.000  
 SULTG = 0.0 WMSC = 3250.000 VOT = 0.000 D2F = 0.000 TMIN = .020

IC = 1.0 WSTRF = 2.500 WSTRIO = 2.500 WSTOPO = 2.500 STARTS = 8.  
 WPRESS = 200.000 MOE = 3 FCRYO = 1.0 OCRYO = 1.0 TG2 = 0.000 PUF1 = 24.000  
 TMELCO = 42.000 TMETST = 26.67 TMEGND = .15 TGRND = 530.00 TORB = 530.00  
 RHQINF = 3.510 THKINF = .74900 TPROPF = 212.20 KGRNDF = .3500E-01 KORBF = .4900E-04  
 ACONDF = .050 HFCE = 214.80 TPRDPO = 171.20 KGRNDO = .3500E-01 KORBO = .4900E-04  
 RHQIMO = 3.510 THKIMO = 1.22000 QCOND = 8257.0  
 ACONDO = .050 HFCE = 89.39

CONFIGURATION 24, THRUST 100%.0 LBF, OX.FLOW RATE 2.1598 LBM/SEC, FUEL FLOW RATE .5837 LBM/SEC, ISP 364.5 SEC

BURN NUMBER	OXIDIZER MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	17185.9	251.7	9.37	4672.1	185.8	9.63	1869.	59552.2	.17E-01
2	15155.9	222.0	23.56	4120.5	163.9	23.20	1707.	54389.0	.18E-01
3	13300.8	194.8	34.43	3616.3	143.8	34.38	1559.	49670.4	.20E-01
4	11605.7	170.0	44.34	3155.2	125.5	44.60	1424.	45358.1	.22E-01
5	10056.8	147.3	53.41	2733.7	108.7	53.94	1300.	41417.1	.24E-01
6	8641.4	126.6	61.69	2348.3	93.4	62.49	1187.	37815.5	.26E-01
7	7348.1	107.6	69.25	1995.8	79.4	70.30	1084.	34524.0	.29E-01
8	6166.4	90.3	76.17	1673.5	66.6	77.44	989.	31510.0	.32E-01
9	5086.6	74.5	82.48	1378.8	54.8	83.97	4199.	28767.0	.35E-01
END OF MISSION	488.2	7.1	----	134.2	5.3	----	----	17081.0	.59E-01

OXIDIZER TANK SHAPE IS CYL/SORT2 DOME  
 TANK VOL = 259.1 FT3, DOME VOL = 43.8 FT3, TANK DIAM = 74.2 IN, TANK LENGTH = 121.0 IN, BARREL LENGTH = 68.51 IN

FUEL TANK SHAPE IS CYL/SORT2 DOME  
 TANK VOL = 192.2 FT3, DOME VOL = 26.6 FT3, TANK DIAM = 62.8 IN, TANK LENGTH = 121.9 IN, BARREL LENGTH = 77.51 IN

#24 ADD-ON LOX/LCH4 MLI 1000 LBF THRUST 8 BURN - PROPELLANT JETTILING

VEHICLE MASS = 60000.0 LBM DELTA V = 14448.1 FPS AVE. ISP = 364.5 SEC

TOTAL PROPELLANT	44163.83 LBM
USABLE FUEL	8974.51
USABLE OXIDIZER	33205.70
FUEL TRAPPED	268.36
OXID TRAPPED	976.49
FUEL START-S/D LOSSES	45.00
OXID START-S/D LOSSES	45.00
FUEL BOILOFF	189.11
OXIDIZER BOILOFF	459.65

OXIDIZER TANKS (NO. = 2)	160.77
(CYLINDRICAL/SQRT(2) ELLIPTICAL)	
DIAMETER = 74.220 IN	
LENGTH = 120.989 IN	
VOLUME = 259.128 FT3	
DOM THK = .02000 IN	
CYL THK = .02000 IN	
FS = 1.50, FNOP = 1.30	

FUEL TANKS (NO. = 2)	135.85
(CYLINDRICAL/SQRT(2) ELLIPTICAL)	
DIAMETER = 62.850 IN	
LENGTH = 121.941 IN	
VOLUME = 192.221 FT3	
DOM THK = .02000 IN	
CYL THK = .02000 IN	
FS = 1.50, FNOP = 1.30	

PRESSURANT	.299
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PRESSURANT SYSTEM MASS	200.000
FUEL TANK INSULATION	77.18
OXIDIZER TANK INSULATION	148.78

ENGINES (NO. = 1)	145.00
(THRUST/ENG = 1000.0 LBF)	
COMPONENTS AND LINES	50.00
ENG. MOUNTS, SUPPORTS	3250.00

TOTAL NET SYSTEM MASS	48331.7
TOTAL BURNOUT MASS	5412.7
(INCL. NON-USABLE PROP. AND GAS)	

MASS FRACTION	.873
TOTAL IMPULSE	15374686.8 LBF-S

PRESSURE SCHEDULE (PSI ) AT T = 530.0 R

GAS TANK LOCK-UP PRESSURE = 0.	INITIAL CHAMBER PRESSURE = 1.000
INITIAL OX SYS PRESSURE = 24.00	FINAL OX SYS PRESSURE = 24.00
INITIAL FU SYS PRESSURE = 24.00	FINAL FU SYS PRESSURE = 24.00

BURN TIME = 15374.69 SEC

#24 ADD-ON LOX/LCH4 MLI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

VEHICLE MASS = 27215.5 KG DELTA V = 4403.6 M/S AVE. ICP = 3574.4 N-S/KG

TOTAL PROPELLANT 20032.38 KG  
USABLE FUEL 4070.77  
USABLE OXIDIZER 15061.85  
FUEL TRAPPED 121.73  
OXID TRAPPED 442.93  
FUEL START-S/D LOSSES 20.41  
OXID START-S/D LOSSES 20.41  
FUEL BOILOFF 85.78  
OXIDIZER BOILOFF 208.50

OXIDIZER TANKS (NO. = 2) 72.93  
(CYLINDRICAL/SQRT (2) ELLIPTICAL)  
DIAMETER = 1.885 M  
LENGTH = 3.077 M  
VOLUME = 7.338 M3  
DOME THK = .00051 M  
CYL THK = .00051 M  
FS = 1.50, FNOP = 1.30

FUEL TANKS (NO. = 2) 61.62  
(CYLINDRICAL/SQRT (2) ELLIPTICAL)  
DIAMETER = 1.596 M  
LENGTH = 3.097 M  
VOLUME = 5.443 M3  
DOME THK = .00051 M  
CYL THK = .00051 M  
FS = 1.50, FNOP = 1.30

PRESSURANT .136

PRESSURANT SYSTEM MASS 90.718  
FUEL TANK INSULATION 35.01  
OXIDIZER TANK INSULATION 67.49

ENGINES (NO. = 1) 65.77  
(THRUST/ENG = 4448.2 N )  
COMPONENTS AND LINES 22.68  
ENG. MOUNTS, SUPPORTS 1474.18

TOTAL WET SYSTEM MASS 21922.9  
TOTAL BURNOUT MASS 2455.2  
(INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .873  
TOTAL IMPULSE 68389989.1 N-S

PRESSURE SCHEDULE (N/M2 ) AT T=294.4 M

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = .895.  
INITIAL OX SYS PRESSURE = .1655E+06 FINAL OX SYS PRESSURE = .1655E+06  
INITIAL FU SYS PRESSURE = .1655E+06 FINAL FU SYS PRESSURE = .1655E+06

BURN TIME = 15374.69 SEC

#26 ADD-ON LOX/LCH4 SOFI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

\*\*\*INPUT\*\*\*

DVB = 14428.10 RHOF = 25.660 PMO = 24.0 ATRIF = 35.700 NR = 3.700 FSFT = 1.500  
 DVB = 14448.10 RHCO = 69.290 PGTI = 0.0 BTRIF = .020 FU = .020 FSUT = 1.500  
 ISP = 364.50 RHOM = .103 TSI = 530.000 CTRIF = .005 DU = .020 FNGT = 0.000  
 WI = 60000.00 RHOMU = 0.000 TFW = 0.000 ATRU = 130.500 NFI = 2 FNOPF = 1.300  
 WPU = 0.00 GAV = 1.660 ENGT = 1.000 BTRU = .020 NGI = 2 FNOPG = 1.300  
 WPR = 0.00 RG = 386.000 WENGI = 147.000 CTRU = .005 NGI = 0 FNOPG = 0.000  
 YB = 0.00 GR = 1.000 TPER = 1000.000 NOSMAP = 3 D10 = 73.310 FNOPV = 1.005  
 MOB = 0.00 2 DPRG = -1.0 MV012 = 50.000 NFSMAP = 3 D1F = 62.080 FNOPG1 = 0.000  
 SULT = .69E+05 BDR = 0.000 WPLUJ = 110.000 VFT = 0.000 D2U = 0.000 VTOP = 0.000  
 SULTG = 0.0 PMF = 24.0 WMSC = 110.000 VOT = 0.000 D2F = 0.000 TMIN = .020

PC = 1.0 WSTRTF = 2.500 WSTRIO = 2.500 WSTOPO = 2.500 STARTS = 8.  
 WPRESS = 200.000 MOE = 3 FCRYU = 1.0 OCRYO = 1.0 TG2 = 0.000 PUF1 = 24.000  
 TMELCO = 42.00 TMEIST = 26.67 TMEGND = .23 TGRND = 530.00 TORB = 300.00  
 RHOINF = 2.200 THKINF = 1.30000 TPROPF = 212.20 KGRNDF = .1350E-01 KORBF = .1070E-01 QCOND = 7314.0  
 ACONDF = .050 HFGF = 214.80 TPROVO = 171.20 KGRNDO = .1350E-01 KORBO = .1070E-01 QCONDO = 8257.0  
 RHOIND = 2.200 THKIND = 1.49000 HFGD = 89.39  
 ACONDO = .050

CONFIGURATION 26, THRUST 1000.0 LBF, OX.FLOW RATE 2.1598 LBM/SEC, FUEL FLOW RATE .5837 LBM/SEC, ISP 364.5 SEC

BURN NUMBER	MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	17109.1	248.2	18.02	4655.0	182.3	17.41	1782.	56780.8	.18E-01
2	15084.2	218.8	30.29	4104.9	160.8	29.77	1621.	51630.6	.19E-01
3	13233.9	191.9	41.28	3601.9	141.1	41.02	1473.	46924.0	.21E-01
4	11543.1	167.4	51.32	3142.0	123.1	51.30	1338.	42622.6	.23E-01
5	9998.0	145.0	60.49	2721.5	106.6	60.70	1214.	38691.6	.26E-01
6	8586.2	124.5	66.87	2337.1	91.5	69.30	1102.	35099.1	.28E-01
7	7296.2	105.8	76.53	1985.5	77.8	77.16	999.	31816.0	.31E-01
8	6117.5	88.7	83.53	1664.1	65.2	84.35	904.	28815.6	.35E-01
9	5040.5	73.1	89.93	1370.1	53.7	90.92	3806.	26073.5	.38E-01
END OF MISSION	473.2	6.8	---	133.9	5.2	---	---	14466.6	.69E-01

OXIDIZER TANK SHAPE IS CYL/SORT2 DOME  
TANK VOL = 271.7 FT3, DOME VOL =

42.2 FT3, TANK DIAM = 73.3 IN, TANK LENGTH = 128.5 IN, BARREL LENGTH = 76.66 IN

FUEL TANK SHAPE IS CYL/SORT2 DOME  
TANK VOL = 200.1 FT3, DOME VOL =

25.6 FT3, TANK DIAM = 62.1 IN, TANK LENGTH = 128.9 IN, BARREL LENGTH = 84.98 IN

#26 ADD-ON LOX/LCH4 SOFI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING

VEHICLE MASS = 60000.0 LBM DELTA V = 14448.1 FPS AVE. ISP = 364.5 SEC

TOTAL PROPELLANT 46744.58 LBM

USABLE FUEL	8557.14
USABLE OXIDIZER	31661.41
FUEL TRAPPED	267.76
OXID TRAPPED	946.45
FUEL START-S/D LOSSES	45.00
OXID START-S/D LOSSES	45.00
FUEL BOILOFF	1149.35
OXIDIZER BOILOFF	4075.47

OXIDIZER TANKS (NO. = 2) 167.94

(CYLINDRICAL/SQRT(2) ELLIPTICAL)  
DIAMETER = 73.310 IN  
LENGTH = 128.495 IN  
VOLUME = 271.671 FT3  
DOME THK = .02000 IN  
CYL THK = .02000 IN  
FS = 1.50, FNOP = 1.30

FUEL TANKS (NO. = 2) 141.39

(CYLINDRICAL/SQRT(2) ELLIPTICAL)  
DIAMETER = 62.080 IN  
LENGTH = 128.872 IN  
VOLUME = 200.111 FT3  
DOME THK = .02000 IN  
CYL THK = .02000 IN  
FS = 1.50, FNOP = 1.30

PRESSURANT .313

PRESSURANT SYSTEM MASS	200.000
FUEL TANK INSULATION	87.39
OXIDIZER TANK INSULATION	118.37

ENGINES (NO. = 1) 145.00

(THRUST/ENG = 1000.0 LBF)  
COMPONENTS AND LINES 50.00  
ENG. MOUNTS, SUPPORTS 3100.00

TOTAL WFT SYSTEM MASS 50755.6

TOTAL BURNOUT MASS 5222.2

(INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .792

TOTAL IMPULSE 14659660.3 LBF-S

PRESSURE SCHEDULE (PSI ) AT T=530.0 R

GAS TANK LOCK-UP PRESSURE = 0.	INITIAL CHAMBER PRESSURE = 1.000
INITIAL OX SYS PRESSURE = 24.00	FINAL OX SYS PRESSURE = 24.00
INITIAL FU SYS PRESSURE = 24.00	FINAL FU SYS PRESSURE = 24.00

BURN TIME = 14659.66 SEC

#26 ADD-ON LOX/LCH4 SQFI 1000 LBF THRUST 8 BURN - PROPELLANT SETTLING  
 VEHICLE MASS = 27215.5 KG DELTA V = 4403.8 M/S AVE. ISP = 3574.4 N-S/KG

TOTAL PROPELLANT 21202.99 KG  
 USABLE FUEL 3881.45  
 USABLE OXIDIZER 14361.37  
 FUEL TRAPPED 121.45  
 OXID TRAPPED 429.30  
 FUEL START-S/D LOSSES 20.41  
 OXID START-S/D LOSSES 20.41  
 FUEL BOILOFF 521.34  
 OXIDIZER BOILOFF 1848.60

OXIDIZER TANKS (NO. = 2) 76.18  
 (CYLINDRICAL/SQRT(2) ELLIPTICAL)  
 DIAMETER = 1.862 M  
 LENGTH = 3.264 M  
 VOLUME = 7.693 M3  
 DOME THK = .00051 M  
 CYL THK = .00051 M  
 FS = 1.50, FNOP = 1.30

FUEL TANKS (NO. = 2) 64.13  
 (CYLINDRICAL/SQRT(2) ELLIPTICAL)  
 DIAMETER = 1.577 M  
 LENGTH = 3.273 M  
 VOLUME = 5.607 M3  
 DOME THK = .00051 M  
 CYL THK = .00051 M  
 FS = 1.50, FNOP = 1.30

PRESSURANT .142

PRESSURANT SYSTEM MASS 90.718  
 FUEL TANK INSULATION 39.64  
 OXIDIZER TANK INSULATION 53.96

ENGINES (NO. = 1) 65.77  
 (THRUST/ENG = 4448.2 N)  
 COMPONENTS AND LINES 22.68  
 ENG. MOUNTS, SUPPORTS 1406.14

TOTAL WET SYSTEM MASS 23022.3  
 TOTAL BURNOUT MASS 2368.8  
 (INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .792  
 TOTAL IMPULSE 65209394.1 N-S

PRESSURE SCHEDULE (N/M2 ) AT T = 294.4 K

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 6895.  
 INITIAL OX SYS PRESSURE = .1655E+06 FINAL OX SYS PRESSURE = .1655E+06  
 INITIAL FU SYS PRESSURE = .1655E+06 FINAL FU SYS PRESSURE = .1655E+06  
 BURN TIME = 14659.66 SEC



TASK III LOX/LH2 MLI 500 LBF THRUST 8 BURN

\*\*\*INPUT\*\*\*

DVU = 14593.90	RHOF = 4.190	PMO = 24.0	ATRPF = 10.000	MR = 6.000	FSFT = 1.500
DVB = 14593.90	RHOO = 68.800	PGTI = 0.	BTRPF = .020	FU = .020	FSOT = 1.500
ISP = 440.00	RHOM = .103	TSI = 530.000	CTRPF = .005	DU = .020	FSGT = 0.000
WI = 60000.00	RHOMG = 0.000	TTW = 0.000	ATRPO = 55.300	NFT = 1	FNDPF = 1.300
WPU = 0.00	GAM = 1.660	ENGT = 1.000	BTRPO = .020	NGT = 1	FNDPO = 1.300
WPB = 0.00	RG = 386.000	WENGT = 80.000	CTRPO = .005	NGT = 0	FNDPG = 0.000
TB = 0.00	GR = 1.000	TPER = 500.000	NDSHAP = 3	D10 = 168.000	FNDPV = 1.005
MOB = 0.00	DPRG = -1.0	MVO12 = 0	NFSHAP = 3	D1F = 168.000	FNDPVT = 0.000
SULT = 69E+05	BDR = 0.000	WPLUM = 52.600	VFT = 0.000	D20 = 0.000	VTOP = 0.000
SULTG = 0.	PMF = 24.0	WMSC = 3250.000	VOT = 0.000	D2F = 0.000	TMIN = .020
PC = 1.0	WSTRIF = 1.000	WSTRTO = 2.500	WSTRPF = 1.000	WSTRPO = 2.500	STARTS = 8.
WPRESS = 200.000	MOE = 3	FCRYO = 1.0	OCRYO = .0	TG2 = 0.000	PUF1 = 24.000
TMELCO = 42.00	TMETST = 31.76	TMGND = 15	TGRND = 530.00	TORB = 530.00	PUD1 = 24.00
RHOINF = 3.510	THKINF = 1.00000	TPROPF = 39.69	KGRNDF = .3500E-01	KORBF = .4900E-04	QCONDF = 11270.0
ACONDF = .050	HFGF = 187.04	TPRDF0 = 171.20	KGRND0 = .3500E-01	KORR0 = 4900E-04	QCONDD = 8257.0
RHOINO = 3.510	THKINO = .90000				
ACONDO = .050	HFGO = 89.39				

THRUST 500.0 LBF. OX FLOW RATE .9740 LBM/SEC. FUEL FLOW RATE 1623 LBM/SEC. ISP 440.0 SEC

BURN NUMBER	MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	33857.7	494.6	9.56	5757.1	1380.9	22.32	3620.	59410.0	.84E-02
2	30109.9	439.8	21.87	5119.1	1227.9	38.16	3538.	55024.2	.91E-02
3	26636.8	389.1	29.78	4526.9	1085.8	50.06	3277.	50958.9	.98E-02
4	23418.4	342.1	36.17	3977.1	953.9	60.42	3035.	47190.6	.11E-01
5	20435.9	298.5	41.73	3466.6	831.5	69.96	2810	43697.7	.11E-01
6	17672.3	258.1	46.74	2992.6	717.8	78.83	2602	40460.0	.12E-01
7	15111.4	220.7	51.38	2552.4	612.2	87.06	2409	37459.0	.13E-01
8	12735.5	186.1	55.74	2143.6	514.1	94.70	2230.	34677.2	.14E-01
9	10539.9	154.0	59.93	1763.7	423.0	101.83	9758.	32098.7	.16E-01
END OF MISSION	880.5	12.8	---	152.7	36.4	---	---	20828.3	.24E-01

OXIDIZER TANK SHAPE IS ELLIPSOIDAL

TANK VOL = 509.2 FT3. DOME VOL = 254.6 FT3. TANK DIAM = 133.5 IN. TANK LENGTH = 94.4 IN. BARREL LENGTH = 0.00 IN

FUEL TANK SHAPE IS CYL/SQRT2 DOME

TANK VOL = 1475.0 FT3. DOME VOL = 508.0 FT3. TANK DIAM = 168.0 IN. TANK LENGTH = 154.6 IN. BARREL LENGTH = 35.79 IN

TASK III	LOX/LH2	MLI	500 LBF THRUST	8 BURN
VEHICLE MASS	=60000.0 LBM	DELTA V=	14593.9 FPS	AVE. ISP= 440.0 SEC
TOTAL PROPELLANT			40204.86 LBM	
USABLE FUEL		5459.88		
USABLE OXIDIZER		32759.27		
FUEL TRAPPED		152.68		
OXID TRAPPED		880.51		
FUEL START-S/D LOSSES		18.00		
OXID START-S/D LOSSES		45.00		
FUEL BOILOFF		398.59		
OXIDIZER BOILOFF		490.94		
OXIDIZER TANKS (NO. = 1)			127.74	
(ELLIPTICAL)				
DIAMETER=	133.450 IN			
LENGTH =	94.363 IN			
VOLUME =	509.209 FT3			
AVG THK =	.0210 IN			
FS =	1.50, FNOP= 1.30			
FUEL TANKS (NO. = 1)			365.69	
(CYLINDRICAL/SURT(2) ELLIPTICAL)				
DIAMETER=	168.000 IN			
LENGTH =	154.580 IN			
VOLUME =	1475.049 FT3			
DOME THK=	.02645 IN			
CYL THK =	.04383 IN			
FS =	1.50, FNOP= 1.30			
PRESSURANT			.657	
PRESSURANT SYSTEM MASS			200.000	
FUEL TANK INSULATION			184.54	
OXIDIZER TANK INSULATION			83.01	
ENGINES (NO. = 1)			80.00	
(THRUST/ENG= 500.0 LBF)				
COMPONENTS AND LINES			52.60	
ENG. MOUNTS, SUPPORTS			3250.00	
TOTAL WET SYSTEM MASS			44549.1	
TOTAL BURNOUT MASS			5377.4	
(INCL. NON-USABLE PROP. AND GAS)				
MASS FRACTION			.858	
TOTAL IMPULSE			16816424.9 LBF-S	
PRESSURE SCHEDULE(		)	A1 T=530.0 R	
GAS TANK LOCK-UP PRESSURE =	0.	INITIAL CHAMBER PRESSURE =	1.000	
INITIAL O <sub>2</sub> SYS PRESSURE =	24.00	FINAL O <sub>2</sub> SYS PRESSURE =	24.00	
INITIAL FU SYS PRESSURE =	24.00	FINAL FU SYS PRESSURE =	24.00	
BURN TIME =	33632.85 SEC			

TASK III	LOX/LH2	MLI	500 LBF THRUST	8 BURN
VEHICLE MASS	=27215.5 KG	DELTA V=	4448.2 M/S	AVE. ISP=4314.7 N-S/KG
TOTAL PROPELLANT			18236.62 KG	
USABLE FUEL		2476.56		
USABLE OXIDIZER		14859.36		
FUEL TRAPPED		69.25		
OXID TRAPPED		399.39		
FUEL START-S/D LOSSES		8.16		
OXID START-S/D LOSSES		20.41		
FUEL BOILOFF		180.80		
OXIDIZER BOILOFF		222.69		
OXIDIZER TANKS (NO. = 1)			57.94	
(ELLIPSOIDAL)				
DIAMETER=	3.390 M			
LENGTH =	2.397 M			
VOLUME =	14.419 M3			
AVG THK =	.00053 M			
FS = 1.50,	FNOP= 1.30			
FUEL TANKS (NO. = 1)			165.87	
(CYLINDRICAL/SQRT(2) ELLIPTICAL)				
DIAMETER=	4.267 M			
LENGTH =	3.926 M			
VOLUME =	41.769 M3			
DOMES THK=	.00067 M			
CYL THK =	.00111 M			
FS = 1.50,	FNOP= 1.30			
PRESSURANT			.298	
PRESSURANT SYSTEM MASS			90.718	
FUEL TANK INSULATION			83.71	
OXIDIZER TANK INSULATION			37.65	
ENGINES (NO. = 1)			36.29	
(THRUST/ENG= 2224.1 N )				
COMPONENTS AND LINES			23.86	
ENG. MOUNTS, SUPPORTS			1474.18	
TOTAL WET SYSTEM MASS			20207.1	
TOTAL BURNOUT MASS			2439.1	
(INCL. NON-USABLE PROP. AND GAS)				
MASS FRACTION			.858	
TOTAL IMPULSE			74803157.5 N-S	

PRESSURE SCHEDULE(N/M2 ) AT T=294.4 K

GAS TANK LOCK-UP PRESSURE	= 0.	INITIAL CHAMBER PRESSURE	= 6895.
INITIAL OX SYS PRESSURE	= .1655E+06	FINAL OX SYS PRESSURE	= .1655E+06
INITIAL FU SYS PRESSURE	= .1655E+06	FINAL FU SYS PRESSURE	= .1655E+06

BURN TIME= 33632.35 SEC

TASK III LOX/LCH4 MLI 500 LBF THRUST 8 BURN

\*\*\*INPUT\*\*\*

DVU = 14571.40 RHOE = 25.500 PMO = 24.0 ATRPF = 19.200 MR = 3.700 FSFT = 1.500  
 DVB = 14571.40 RHO0 = 68.750 PGTI = 0. BTRPF = .020 FSOT = 1.500  
 ISP = 356.50 RHO1 = .103 TSI = 530.000 CTRPF = .005 FU = .020 FSOT = 1.500  
 WI = 60000.00 RHO2 = 0.090 TTW = 0.000 ATRPO = 68.300 NFI = 1 FNDPF = 1.300  
 WPU = 0.00 GAM = 1.660 ENGT = 1.000 BTRPO = .020 NFI = 1 FNDPF = 1.300  
 WPB = 0.00 RG = 386.000 WENGT = 80.000 CTRPO = .005 NGT = 0 FNOPG = 0.000  
 TB = 0.00 GR = 1.000 TPER = 500.000 NDSHAP = 3 D10 = 168.000 FNOPV = 1.005  
 MDR = 0.00 2 DPRG = -1.0 MVO12 = 0 NFSHAP = 3 D1F = 168.000 FNOPGT = 0.000  
 SULT = .69E+05 BDR = 0.000 WPLUM = 52.600 VFT = 0.000 D20 = 0.000 VTOP = 0.000  
 SULTG = 0. PMF = 24.0 WMSC = 3250.000 VOT = 0.000 D2F = 0.000 TMIN = .020

PC = 1.0 WSTRF = 2.500 WSTRIO = 2.500 WSTOPF = 2.500 WSTOPO = 2.500 STARTS = 8.  
 WPRESS = 200.000 MOE = 3 FCRYO = 1.0 OCRYO = 1.0 TG2 = 0.000 PUF1 = 24.000  
 TMELCO = 42.00 TMETST = 30.87 TMEGND = .15 TGRND = 530.00 TORB = 530.00  
 RHOINF = 3.510 THKINF = .70000 TPROPF = 212.20 KGRND = 3500E-01 KORBF = .4900E-04 QCONDF = 7314.0  
 ACONDF = 050 HFGF = 214.80 TPRPO = 171.20 KGRND0 = 3500E-01 KORBO = .4900E-04 UCONDO = 8257.0  
 RHOIND = 3.510 THKIND = .85000 HFGO = 89.39

THRUST 500 0 LBF, OX.FLOW RATE 1 1041 LBM/SEC, FUEL FLOW RATE .2984 LBM/SFC, ISP 356.5 SEC

BURN NUMBER	OXIDIZER MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	34853.5	509.5	9.66	9476.1	373.5	9.21	3787.	59551.8	.84E-02
2	30646.3	448.0	22.95	8333.3	328.4	20.92	3446.	54201.8	.92E-02
3	26814.7	392.0	31.30	7292.0	287.4	28.40	3137	49329.0	.10E-01
4	23325.3	341.0	37.99	6343.1	250.0	34.41	2854	44890.7	11F 01
5	20147.4	294.5	43.74	5478.5	215.9	39.58	2597	40848.2	12F 01
6	17253.3	252.2	48.87	4690.6	184.9	44.20	2363	37166.2	.13E 01
7	14617.8	213.7	53.57	3972.6	156.6	48.44	2150	33812.6	15E-01
8	12217.7	178.6	57.95	3319.2	130.8	52.39	1956.	30758.1	16E-01
9	10032.0	146.6	62.10	2721.7	107.3	56.15	8125	27976.0	.18E-01
END OF MISSION	917.7	13.3	---	254.8	10.0	---	---	16394.8	30E-01

OXIDIZER TANK SHAPE IS ELLIPSOIDAL  
TANK VOL = 524.6 FT3, DOME VOL = 262.3 FT3, TANK DIAM = 134.8 IN, TANK LENGTH = 95.3 IN, BARREL LENGTH = 0.00 IN

FUEL TANK SHAPE IS ELLIPSOIDAL  
TANK VOL = 385.8 FT3, DOME VOL = 192.9 FT3, TANK DIAM = 121.7 IN, TANK LENGTH = 86.0 IN, BARREL LENGTH = 0.00 IN

TASK III LDX/LCH4 MLI 500 LBF THRUST 8 BURN  
 VEHICLE MASS =60000.0 LBM DELTA V= 14571.4 FPS AVE ISP 356 5 SEC

TOTAL PROPELLANT 44777 74 LBM  
 USABLE FUEL 9113.67  
 USABLE OXIDIZER 33720.60  
 FUEL TRAPPED 254.77  
 OXID TRAPPED 917.74  
 FUEL START-S/D LOSSES 45.00  
 OXID START-S/D LOSSES 45.00  
 FUEL BOILOFF 182.94  
 OXIDIZER BOILOFF 498.01

OXIDIZER TANKS (NO. = 1) 131.59  
 (ELLIPSOIDAL)  
 DIAMETER = 134.779 IN  
 LENGTH = 95.303 IN  
 VOLUME = 524.573 FT3  
 AVG THK = .02122 IN  
 FS = 1.50, FNOP = 1.30

FUEL TANKS (NO. = 1) 101.04  
 (ELLIPSOIDAL)  
 DIAMETER = 121.655 IN  
 LENGTH = 86.023 IN  
 VOLUME = 385.775 FT3  
 AVG THK = .02000 IN  
 FS = 1.50, FNOP = 1.30

PRESSURANT .302

PRESSURANT SYSTEM MASS 200.000  
 FUEL TANK INSULATION 53.65  
 OXIDIZER TANK INSULATION 79.97

ENGINES (NO. = 1) 80.00  
 (THRUST/ENG = 500.0 LBF)  
 COMPONENTS AND LINES 52.60  
 ENG MOUNTS, SUPPORTS 3250.00

TOTAL WET SYSTEM MASS 48726 9  
 TOTAL BURNOUT MASS 5121.7  
 (INCL. NON-USABLE PROP. AND GAS)

MASS FRACTION .879  
 TOTAL IMPULSE 15270417.5 LBF-S

PRESSURE SCHEDULE (PSI ) AT T=530.0 R

GAS TANK LOCK-UP PRESSURE = 0. INITIAL CHAMBER PRESSURE = 1.000  
 INITIAL OX SYS PRESSURE = 24.00 FINAL OX SYS PRESSURE = 24.00  
 INITIAL FU SYS PRESSURE = 24.00 FINAL FU SYS PRESSURE = 24.00

BURN TIME = 30540.83 SEC

TASK III	LOX/LCH4	MLI	500 LBF THRUST	8 BURN
VEHICLE MASS	=27215.5 KG	DELTA V=	4441.4 M/S	AVE. ISP=3495.9 N-S/KG
TOTAL PROPELLANT			20310.84 KG	
USABLE FUEL		4133.89		
USABLE OXIDIZER		15295.41		
FUEL TRAPPED		115.56		
OXID TRAPPED		416.28		
FUEL START-S/D LOSSES		20.41		
OXID START-S/D LOSSES		20.41		
FUEL BOILOFF		82.98		
OXIDIZER BOILOFF		225.89		
OXIDIZER TANKS (NO. = 1)			59.69	
(ELLIPSOIDAL)				
DIAMETER=	3.423 M			
LENGTH =	2.421 M			
VOLUME =	14.854 M3			
AVG THK =	.00054 M			
FS = 1.50,	FNOP= 1.30			
FUEL TANKS (NO. = 1)			45.83	
(ELLIPSOIDAL)				
DIAMETER=	3.090 M			
LENGTH =	2.185 M			
VOLUME =	10.924 M3			
AVG THK =	.00051 M			
FS = 1.50,	FNOP= 1.30			
PRESSURANT			.137	
PRESSURANT SYSTEM MASS			90.718	
FUEL TANK INSULATION			24.34	
OXIDIZER TANK INSULATION			36.27	
ENGINES (NO. = 1)			36.29	
(THRUST/ENG= 2224.1 N)				
COMPONENTS AND LINES			23.86	
ENG MOUNTS, SUPPORTS			1474.18	
TOTAL WET SYSTEM MASS			22102.2	
TOTAL BURNOUT MASS			2323.2	
(INCL. NON-USABLE PROP. AND GAS)				
MASS FRACTION			.879	
TOTAL IMPULSE			67926176.3 N-S	

PRESSURE SCHEDULE(N/M2 ) AT T=294.4 K

GAS TANK LOCK-UP PRESSURE =	0.	INITIAL CHAMBER PRESSURE =	6895.
INITIAL OX SYS PRESSURE =	.1655E+06	FINAL OX SYS PRESSURE =	.1655E+06
INITIAL FU SYS PRESSURE =	.1655E+06	FINAL FU SYS PRESSURE =	.1655E+06

BUFN TIME= 30540.83 SEC



TASK III LOX/RP-1 MLJ 500 LBF THRUST 8 BURN

\*\*\*INPUT\*\*\*

DVU = 14564.20 RHDF = 50.300 PNO = 24.0 AIRPF = 22.300 MR = 3.000 FSFI = 1.500  
 OVB = 14564.20 RHOD = 68.750 PGTI = 0.0 BTRPF = 0.020 FSOT = 1.500  
 ISP = 373.50 RHOM = .103 TSI = 530.000 CTRPF = .005 OUI = .020 FSGT = 0.000  
 WI = 60000.00 RHOMG = 0.000 ITW = 0.000 ATRPD = 51.600 NFI = 1.300  
 WPU = 0.00 GAM = 1.660 ENGT = 1.000 BTRPD = .005 NGT = 0.000 FNOPG = 1.300  
 WPB = 0.00 RG = 386.000 WENG1 = 80.000 CTRPD = .005 NGT = 0.000 FNOPG = 1.300  
 YB = 0.00 GR = 1.000 TPER = 500.000 NDSHAP = 3 D10 = 167.600 FNOPV = 1.005  
 MOB = .69E+05 BDR = -1.0 MVO12 = 0 NFSHAP = 3 D1F = 170.000 FNOPGT = 0.000  
 SULT = 0.000 PMF = 24.0 WPLUM = 52.600 VFT = 0.000 D20 = 0.000 VTOP = 0.000  
 SULTG = 0.000 WMSC = 3200.000 VOT = 0.000 D2F = 0.000 TMIN = .020

PC = 1.0 WSTRTF = 2.000 WSTRTO = 2.500 WSTOPF = 2.000 WSTOPO = 2.500 STARTS = 8.  
 WPRESS = 200.000 MOE = 3 FCRYO = 0.0 OCRYO = 1.0 TG2 = 0.000 PUF1 = 24.000  
 TMELCO = 42.00 TMETST = 30.79 TMEGND = .15 TGRND = 530.00 TORB = 530.00  
 RHOINO = 3.510 THK:NO = .85000 TPRNO = 171.28 KGRND0 = .3500E-01 KORBO = 4900E-04 OCONDO = 8257.0  
 ACONDO = .050 HFGO = 89.39

THRUST 500 0 LBF. OR. FLOW RATE 1 1244 LBM/SEC. FUEL FLOW RATE .3148 LBM/SEC. ISP 333.5 SEC

BURN NUMBER	OXIDIZER MASS (LBM)	OXIDIZER VOLUME (FT3)	HEIGHT (IN)	MASS (LBM)	FUEL VOLUME (FT3)	HEIGHT (IN)	BURN DURATION (SEC)	TOTAL MASS (LBM)	ACCEL (G)
1	34240.5	502.0	9 63	11419.6	228.2	6.0	3779.	59673.5	.84E-02
2	30064.9	439.5	23 16	10003.0	199.9	17.18	3419	53981.1	.93E-02
3	26194.5	382.9	31 61	8721.6	174.3	23.79	3093	48829.2	.10E 01
4	22690.9	331.7	38 35	7562.5	151.1	29.01	2797	44166.5	.11E 01
5	19519.4	285.7	44 12	6514.1	130.2	33.47	2530	39946.6	.13E 01
6	16648.4	243.4	49 26	5565.8	111.2	37.42	2288	36127.3	.14E 01
7	14049.4	205.4	53 95	4708.2	94.1	41.01	2069.	32670.7	.15E 01
8	11696.6	171.0	58 31	3932.6	78.6	44.34	1871	29542.3	.17E 01
9	9566.4	139.8	62.43	3231.3	65.6	47.48	7631	26711.0	.19E 01
END OF MISSION	886.6	12.9		304.4	6.1			15106.3	.33E 01

OXIDIZER TANK SHAPE IS ELLIPSOIDAL  
 TANK VOL = 516.9 FT3, DOME VOL = 258.5 FT3, TANK DIAM = 124.1 IN, TANK LENGTH = 94.8 IN, BARREL LENGTH = 0.00 IN

FUEL TANK SHAPE IS ELLIPSOIDAL  
 TANK VOL = 232.7 FT3, DOME VOL = 116.4 FT3, TANK DIAM = 102.8 IN, TANK LENGTH = 72.7 IN, BARREL LENGTH = 0.00 IN

TASK .11	LOX/RP-1	MLI	500 LBF THRUST	8 BURN
VEHICLE MASS	=60000.0 LBM	DELTA V-	14564.2 FPS	AVE ISP= 333.5 SEC
TOTAL PROPELLANT			46086.91 LBM	
USABLE FUEL		11079.17		
USABLE OXIDIZER		33237.52		
FUEL TRAPPED		304.43		
OXID TRAPPED		888.82		
FUEL START-S/D LOSSES		36.00		
OXID START-S/D LOSSES		45.00		
OXIDIZER BOILOFF		495.96		
OXIDIZER TANKS (NO. = 1)			129.67	
(ELLIPSOIDAL)				
DIAMETER=	134.119 IN			
LENGTH =	94.837 IN			
VOLUME =	516.909 FT3			
AVG THK =	.02112 IN			
FS = 1.50,	FNOP= 1.30			
FUEL TANKS (NO. = 1)			72.14	
(ELLIPSOIDAL)				
DIAMETER=	102.794 IN			
LENGTH =	72.687 IN			
VOLUME =	232.728 FT3			
AVG THK =	.02000 IN			
FS = 1.50,	FNOP= 1.30			
PRESSURANT			.248	
PRESSURANT SYSTEM MASS		200.000		
OXIDIZER TANK INSULATION		79.19		
ENGINES (NO. = 1)			80.00	
(THRUST/ENG= 500.0 LBF)				
COMPONENTS AND LINES		52.60		
ENG. MOUNTS,SUPPORTS		3200.00		
TOTAL WET SYSTEM MASS		49900.8		
TOTAL BURNOUT MASS		5007.1		
(INCL.NON-USABLE PROP. AND GAS)				
MASS FRACTION		.888		
TOTAL IMPULSE		14779615.6 LBF-S		
PRESSURE SCHEDULE(PSI )			AT T=530.0 R	
GAS TANK LOCK-UP PRESSURE	= 0.	INITIAL CHAMBER PRESSURE	= 1.000	
INITIAL OX SYS PRESSURE	= 24.00	FINAL OX SYS PRESSURE	= 24.00	
INITIAL FU SYS PRESSURE	= 24.00	FINAL FU SYS PRESSURE	= 24.00	
BURN TIME=	29559.23 SEC			

C-3

TASK III	LOX/RP-1	MLI	500 LBF THRUST	8 BURN
VEHICLE MASS	=27215.5 KG	DELTA V=	4439.2 M/S	AVE. ISP=3270.4 N S/KG
TOTAL PROPELLANT			20904.67 KG	
USABLE FUEL		5025.43		
USABLE OXIDIZER		15076.28		
FUEL TRAPPED		138.09		
OXID TRAPPED		403.16		
FUEL START-S/D LOSSES		16.33		
OXID START-S/D LOSSES		20.41		
OXIDIZER BOILOFF		224.97		
OXIDIZER TANKS (NO. = 1)			58.82	
(ELLIPSOIDAL)				
DIAMETER=	3.407 M			
LENGTH =	2.409 M			
VOLUME =	14.637 M3			
AVG THK =	.00054 M			
FS = 1.50,	FNOP= 1.30			
FUEL TANKS (NO. = 1)			32.72	
(ELLIPSOIDAL)				
DIAMETER=	2.611 M			
LENGTH =	1.846 M			
VOLUME =	6.590 M3			
AVG THK =	.00051 M			
FS = 1.50,	FNOP= 1.30			
PRESSURANT			.113	
PRESSURANT SYSTEM MASS			90.718	
OXIDIZER TANK INSULATION			35.92	
ENGINES (NO. = 1)			36.29	
(THRUST/ENG= 2224.1 N )				
COMPONENTS AND LINES			23.86	
ENG. MOUNTS, SUPPORTS			1451.50	
TOTAL WET SYSTEM MASS			22634.6	
TOTAL BURNOUT MASS			2271.2	
(INCL. NON-USABLE PROP. AND GAS)				
MASS FRACTION			.888	
TOTAL IMPULSE			65742981.6 N-S	

PRESSURE SCHEDULE(N/M2 ) AT T=294.4 K

GAS TANK LOCK-UP PRESSURE	= 0.	INITIAL CHAMBER PRESSURE	= 6895.
INITIAL OX SYS PRESSURE	= .1655E+06	FINAL OX SYS PRESSURE	= .1655E+06
INITIAL FU SYS PRESSURE	= .1655E+06	FINAL FU SYS PRESSURE	= .1655E+06

BURN TIME= 29559.23 SEC

## APPENDIX B

### PARALLEL TANK DIAMETER ANALYSIS

#### SYMBOLS

D	-	Diameter of stage
h	-	Tank dome height
$L_B$	-	Tank barrel section length
$L_T$	-	Total tank height
r	-	Tank radius
R	-	Radius of stage
$V_{CYL}$	-	Volume of cylindrical section of tank
$V_{DOMES}$	-	Combined volume of the upper and lower tank domes
$V_{TANK}, V_T$	-	Total volume of the tank
$V_{O_2}$	-	Volume of one of the equal-volume $LO_2$ tanks
$V_{CH_4}$	-	Volume of one of the equal-volume $LCH_4$ tanks
X	-	Insulation thickness

To determine the fuel and oxidizer tank diameters for the parallel tanks configurations there were two approaches used depending on the propellant combination.

1) LO<sub>2</sub>/LH<sub>2</sub> Tank Diameters

Due to the large volume of fuel involved when LH<sub>2</sub> was used the pair of fuel tanks alone determined the system length. A representative LO<sub>2</sub>/LH<sub>2</sub> case is shown in Figure B-1(a). The fuel tank diameter was found by subtracting twice the insulation thickness from 2.16m (85 in). The oxidizer tanks then filled the volume left inside the 4.32m (170 in) diameter shell to produce the arrangement shown in Figure B-1(a).

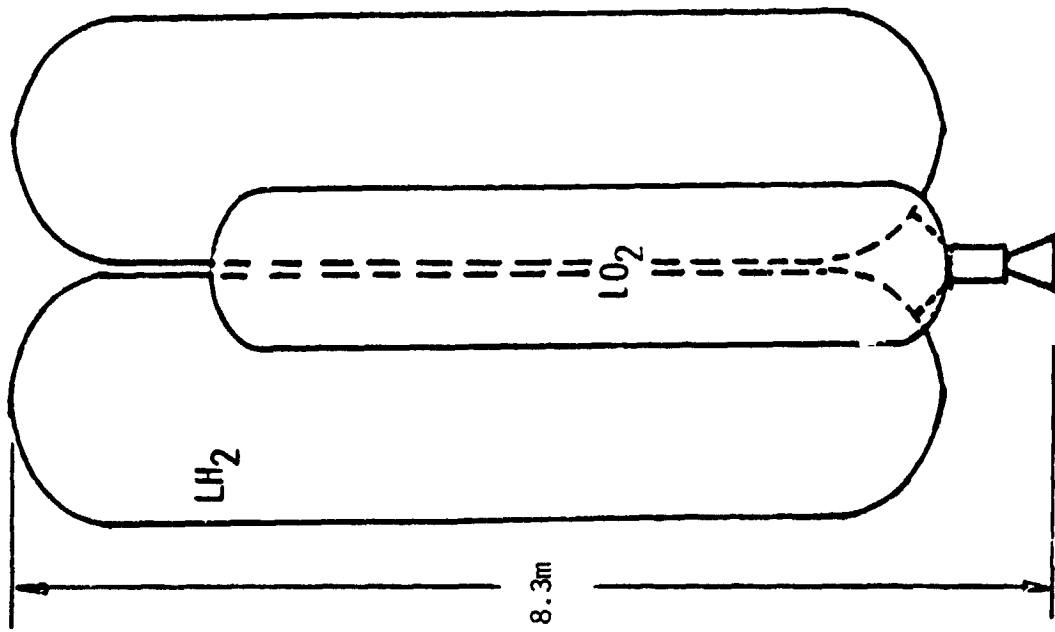
2) LO<sub>2</sub>/LCH<sub>4</sub> and LO<sub>2</sub>/RP-1 Tank Diameters

The arrangement shown in Figure B-1(b) is representative of both LCH<sub>4</sub> and RP-1 as fuel, only the dimensions differ. To minimize the stage length when using parallel tanks, the propellant should be equally divided between two tanks of equal length. It was assumed that the outside diameters (tank plus insulation) of a tank touches the outside diameter of the two adjacent tanks and the inside of the shell, as shown in Figure B-2.

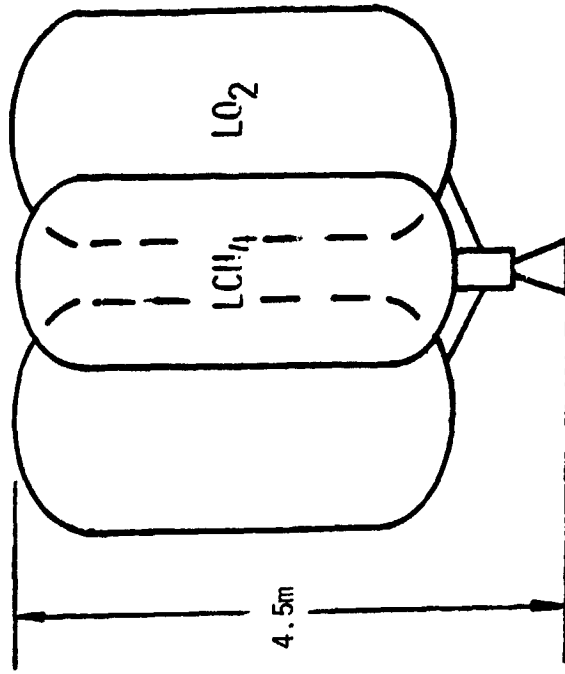
To calculate the tank radii, the tank volume was first calculated as a function of radius and tank length. Referring to B-3

where 
$$V_{\text{TANK}} = V_{\text{CYL}} + V_{\text{DOMES}} \quad (\text{B-1})$$

$$V_{\text{DOMES}} = \frac{4}{3}\pi r^2 h = \frac{4}{3} \cdot \frac{\pi r^3}{\sqrt{2}} \quad (\text{For both domes}) \quad (\text{B-2})$$



(a)  $LO_2/LH_2$



(b)  $LO_2/LCH_4$

FIGURE B-1 PARALLEL TANK CONFIGURATIONS FOR  $LO_2/LH_2$  AND  $LO_2/LCH_4$

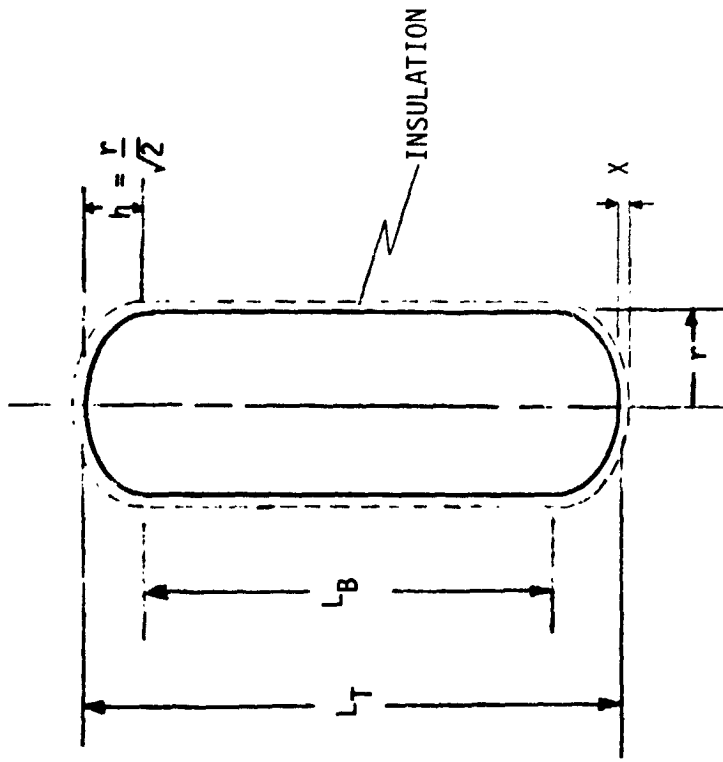


FIGURE B-3 CYLINDRICAL TANK WITH  $\sqrt{2}$  ELLIPTICAL DOMES

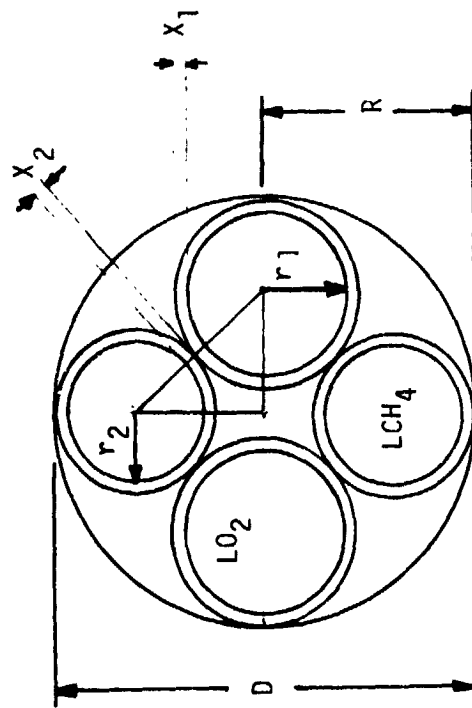


FIGURE B-2 CUTAWAY VIEW OF TANKS



and

$$V_{\text{TANK}} = \pi r^2 L_B + \frac{4}{3} \frac{\pi r^3}{\sqrt{2}} \quad (\text{B-3})$$

or

$$\frac{V_T}{\pi r^2} = L_B + \frac{4}{3} \frac{r}{\sqrt{2}} \quad (\text{B-4})$$

For the overall tank length ( $L_T$ ) as a function of  $r$ ,

$$\begin{aligned} \frac{V_T}{\pi r^2} &= \left[ L_B + \frac{4}{3} \frac{r}{\sqrt{2}} + \frac{2}{3} \frac{r}{\sqrt{2}} \right] - \frac{2}{3} \frac{r}{\sqrt{2}} \\ &= L_T - \frac{2}{3} \frac{r}{\sqrt{2}} \end{aligned} \quad (\text{B-5})$$

or

$$L_T = \frac{V_T}{\pi r^2} + \frac{2}{3\sqrt{2}} r = \frac{V_T}{\pi r^2} + .4714r \quad (\text{B-6})$$

Overall tank length =  $L_T + 2X$

$$= \frac{V_T}{\pi r^2} + 0.4714r + 2X \quad (\text{B-7})$$

where  $X$  is the insulation thickness.

For the minimum stage length, the overall lengths of each tank will be equal  
Therefore, allowing for the different clearances,

$$\frac{V_{\text{O}_2}}{2\pi r_1^2} + 0.4714r_1 + 2X_1 = \frac{V_{\text{CH}_4}}{2\pi r_2^2} + 0.4714r_2 + 2X_2 \quad (\text{B-8})$$

Now using the Pythagorean theorem

$$(R - r_1 - X_1)^2 + (R - r_2 - X_2)^2 = (r_1 + r_2 + X_1 + X_2)^2 \quad (\text{B-9})$$

which leads to

$$R^2 - Rr_1 - RX_1 - RX_2 - r_1X_2 - X_1X_2 = r_1r_2 + Rr_2 + r_2X_1 \quad (\text{B-10})$$

and

$$r_2 = \frac{R(R - r_1 - X_1 - X_2) - X_2(r_1 + X_1)}{R + r_1 + X_1} \quad (\text{B-11})$$

Combining equations B-8 and B-11

$$\begin{aligned} \frac{V_{O_2}}{2\pi r_1^2} + 0.4714r_1 + 2X_1 &= \frac{V_{CH_4}}{2\pi r_2^2} \left[ \frac{R + r_1 + X_1}{R(R - r_1 - X_1 - X_2) - X_2(r_1 + X_1)} \right]^2 \\ &+ 0.4714 \left[ \frac{R(R - r_1 - X_1 - X_2) - X_2(r_1 + X_1)}{R + r_1 + X_1} \right] + 2X_2 \quad (\text{B-12}) \end{aligned}$$

Values for  $r_1$  and  $r_2$  can be found using equations B-11 and B-12 that satisfy the equal length criteria for the full length of the tank, for any insulation thickness or shell diameter. The values of  $r_1$  and  $r_2$  will also result in the minimum length system.

APPENDIX C

OPTIMUM THICKNESS OF INSULATION - VOLUMETRIC CONSIDERATIONS

SYMBOLS

- $A_s$  -- Surface Area of Tank.
- $h_{fg}$  - Latent Heat of Vaporization.
- $K_g, K_o$  - Thermal Conductivity of Insulation During Ground Hold and Orbit.
- $t_g, t_o$  - Ground Hold and On-Orbit Time.
- $Thk_{tank}$  - Tank Wall Thickness.
- $\Delta T_g, \Delta T_o$  - Temperature Difference Between External Skin and Propellant On Ground and In Orbit.
- $V_B, V_{INS}, V_R, V_{TS}, V_U$  - Volume of Usable ( $\Delta V$ ) Propellant, Insulation, Residual Propellant, Tank Shell, and Ullage Respectively.
- $V_E, V_{EO}$  - Volume of Boiloff Due to Heat Leak Through Insulation and Struts.
- $V_{TOTAL}$  - Total Volume of Propellant and Tank Subsystems.
- $X_I$  - Thickness of Insulation.
- $\rho_P$  - Density of Propellant.

The total volume for the propellant tank subsystems can be calculated by summing all volumes:

$$V_{TOT} = V_B + V_E + V_{EO} + V_{INS} + V_{TS} + V_U + V_R \quad (C-1)$$

Differentiating with respect to insulation thickness,

$$\begin{aligned} \frac{dV_{TOT}}{dX_I} &= \frac{d}{dX_I} (V_E + V_{INS} + V_{TS}) \\ &= \frac{d}{dX_I} \frac{(Kt\Delta T)' A_S}{h_{fg}\rho_p X_I} + A_S X_I + A_S Thk_{tank} \end{aligned} \quad (C-2)$$

where

$$(Kt\Delta T)' = K_g t_g \Delta T_g + K_o t_o \Delta T_o \quad (C-3)$$

Assuming  $\frac{dA}{dX_I} \ll 1$

$$\frac{dV_{TOT}}{dX_I} = \frac{-(Kt\Delta T)' A_S}{h_{fg}\rho_p X_I^2} + A_S \quad (C-4)$$

Now to find the minimum volume, assume  $\frac{dV_{TOT}}{dX_I} = 0$

then

$$\frac{(Kt\Delta T)'}{h_{fg}\rho_p X_I} = 1 \quad (C-5)$$

or

$$X_I^2 = \frac{(Kt\Delta T)'}{h_{fg}\rho_p} \quad (C-6)$$

or

$$X_I = \sqrt{\frac{K_g t_g \Delta T_g + K_o t_o \Delta T_o}{h_{fg}\rho_p}} \quad (C-7)$$

APPENDIX D

OPTIMUM INSULATION THICKNESS - CYLINDRICAL/ $\sqrt{2}$  ELLIPSOIDAL TANK

Symbols

$A'$	Cross-sectional area of penetrating struts
$A_D, A_S$	Surface area of domes and tanks
$D_T$	Tank diameter
$f_R$	Fraction of propellant left as residual
$f_u$	Ullage fraction
$h_{fg}$	Latent heat of vaporization
$K_g, K_o$	Insulation thermal conductivity during ground hold and on-orbit
$L_B$	Length of barrel section
$q_s$	Heat input rate per unit area through struts
$Q_A, Q_G, Q_O$	Heat input to tank, through the insulation during period of ascent, ground-hold, and on-orbit
$Q_s$	Heat input to tank through the struts
$Q_I$	Total heat leak to tank
$\rho, \rho', \rho''$	Density of insulation, propellant, and tank material
$t'_A$	Equivalent ascent time
$t_g, t_o$	Time during which system is at ground hold or on-orbit environmental conditions
$T_{AG}, T_{AO}$	Ambient temperature during period of ground-hold or on-orbit

The following derivation is based on the use of a cylindrical tank with a constrained diameter, so that any growth required to accommodate additional propellant lost to evaporation is by increased length. It is further assumed that the initial ullage volume is a fixed fraction of the total tank volume, and that the residual propellant is a fixed fraction of the total propellant mass. This mass can be expressed as follows:

$$W_P = W_B + W_E + W_R = W_B + W_E + f_R W_P,$$

or

$$W_P = \frac{W_B + W_E}{1 - f_R},$$

and

$$V_T = \frac{W_P}{\rho_P (1 - f_u)}$$

where  $W_B$  is the mass of burned propellant,  $W_E$  is the evaporated propellant,  $W_R$  is the residual, and  $\rho_P$  is the evaporated propellant density (assumed constant).

The insulation is assumed to have a thermal conductivity on the ground which is different from that in orbit. It is further assumed that the ascent heating can be considered to be at the ground rate for some equivalent time which can be added to the locked-up ground hold time that, when multiplied by the ground hold heat rate than gives the ground hold plus ascent total heat input; i.e.,

$$Q_G + Q_A = q_G A_S (t_G + t'_A) = q_G A_S t'_G$$

where

$$q_G = \frac{K_G(T_{AG} - T_P)}{X_I}$$

The total heat input is given by the equation

$$Q_T = Q_A + Q_G + Q_O + Q_S$$

where

$$Q_O = \text{orbital heat input} = K_O A_S (T_{A_O} - T_P) t_O / X_I$$

$$Q_S = \text{solid conduction} = q_S (t_O + t'_G) A'$$

If the simplification that  $T_{AG} = T_{A_O}$  is made, then

$$Q_T = \frac{(K_G t'_G + K_O t_O) A_S (T_{A_O} - T_P)}{X_I} + q_S (t'_G + t_O) A'$$

and the weight of propellant evaporated (since the propellant temperature is assumed constant) is

$$W_F = Q_T / h_{fg}$$

where  $h_{fg}$  = latent heat of vaporization. The total tank surface area is given by

$$A_S = A_D + \pi D_T L_B$$

where

$$A_D = \text{dome surface area}$$

$$D_T = \text{tank diameter}$$

$L_B$  = barrel section length

Similarly, the tank volume can be expressed as

$$V_T = V_D + \pi D_T^2 L_B / 4,$$

from which

$$L_B = \frac{4(V_T - V_D)}{\pi D_T^2}$$

so that

$$A_S = A_D - \frac{4V_D}{D_T} + \frac{4V_T}{D_T} = A_D - \frac{4V_D}{D_T} + \frac{4W_P}{D_T \rho_P (1 - f_u)}$$

Let

$$A_D - \frac{4V_D}{D_T} = A_o \text{ and } \frac{4}{D_T \rho_P (1 - f_u)} = C_A;$$

then

$$A_S = A_o + C_A W_P$$

combining the above results, we get

$$W_E = \frac{\frac{(K_G t_G' + K_o t_o) (T_A - T_P) (A_o + C_A W_P)}{X_I} + q_s (t_G' + t_o)}{h_{fg}}$$



If we let

$$\frac{(K_G t_G + K_O t_O) (T_A - T_P)}{h_{fg}} = C_I$$

and

$$\frac{q_S (t_G + t_O)}{h_{fg}} = W_{EO}$$

then

$$W_E = \frac{C_I (A_O + C_A W_P)}{X_I} + W_{EO}$$

The total propellant mass then becomes

$$W_P = W_B + W_R + W_E = W_B + f_R W_P + \frac{C_I A_O}{X_I} + \frac{C_I C_A W_P}{X_I} + W_{EO}$$

or combining terms

$$W_P = \frac{W_B + W_{EO} + \frac{C_{I_O}}{X_I}}{1 - f_R - \frac{C_E}{X_I}}$$

where

$$C_{I_O} = C_I A_O \text{ and } C_E = C_I C_A$$

Since the tank must grow to accommodate the propellant lost to evaporation, its mass must be included also. This can be expressed as

$$W_T = W_D + \pi D_T L_T X_T \rho_T$$

where

$W_D$  = mass of the domes

$X_T$  = barrel section wall thickness

$\rho_T$  = barrel section density

In terms of previously defined variables, this becomes

$$W_T = W_D - \frac{4V_D X_T \rho_T}{D_T} + \frac{4X_T \rho_T W_P}{D_T \rho_P (1 - f_u)}$$

then if

$$W_D - \frac{4V_D X_T \rho_T}{D_T} = W_{T0}$$

and

$$\frac{4X_T \rho_T}{D_T \rho_P (1 - f_u)} = C_T$$

then

$$W_T = W_{T0} + C_T W_P$$

and the insulation mass is given by

$$W_I = A_S X_I \rho_I = X_I \rho_I (A_O + C_A W_P)$$

The combined propellant system mass

$$W_{PS} = W_P + W_T + W_I$$

can then be expressed as

$$W_{PS} = W_P + W_{T_0} + C_T W_P + X_I \rho_{I_0} A_0 + X_I \rho_{I_0} C_A W_P$$

$$W_{PS} = W_{T_0} + X_I \rho_{I_0} A_0 + (1 + C_T + X_I \rho_{I_0} C_A) W_P$$

$$W_{PS} = W_{T_0} + X_I \rho_{I_0} A_0 + (1 + C_T + C_A \rho_{I_0} X_I) \frac{W_B + W_{E_0} + \frac{C_{I_0}}{X_I}}{1 - f_R - \frac{C_E}{X_I}}$$

$$W_{PS} = W_{T_0} + X_I \rho_{I_0} A_0 + (1 + C_T + C_A \rho_{I_0} X_I) \left[ \frac{(W_B + W_{E_0}) X_I + C_{I_0}}{(1 - f_R) X_I - C_E} \right]$$

This can be simplified to

$$W_{PS} = \frac{aX_I^2 + bX_I + c}{dX_I - e} + f + gX_I$$

where

$$a = (W_B + W_{E_0}) C_A \rho_I$$

$$b = (W_B + W_{E_0}) (1 + C_T) + C_{I_0} C_A \rho_I$$

$$c = (1 + C_I) C_{I0}$$

$$d = 1 - f_R$$

$$e = C_E$$

$$f = W_{T0}$$

$$g = A_0 \rho_I$$

The optimum insulation thickness is obtained by setting

$$\frac{\partial W_{PS}}{\partial X_I} = 0$$

which gives the equation

$$\frac{2aX + b}{dX - e} - \frac{(aX^2 + bX + c)(d)}{(dX - e)^2} + g = 0$$

where

$$X = X_I \text{ opt}$$

After algebraic manipulation, this leads finally to

$$X = C_1 + \sqrt{\frac{C_1(C_1 C_2 + C_3) + C_4}{C_2 + C_5}}$$

where

$$C_1 = \frac{e}{d}, C_2 = \frac{a}{d}, C_3 = \frac{b}{d}, C_4 = \frac{c}{d}, \text{ and } C_5 = g$$

## APPENDIX E

### Optimum Insulation Thickness - Toroidal Tank

This Appendix presents a derivation for equations utilized in optimizations for minimum weight of toroidal vessels. Insulation thickness and all volumetric elements are included.

#### Symbols

$A'$	-	Total cross-sectional area of penetrating struts
$A_S$	-	Surface area of tank
$f_R$	-	Fraction of residual propellant
$f_u$	-	Ullage fraction
$h_{fg}$	-	Latent heat of vaporization
$K_G, K_O$		Thermal conductivity of insulation during ground hold and orbit
$q''_S$		Heat input rate per unit area
$q_G$		Heat input rate during ground hold
$Q_A, Q_G, Q_O$		Total heat input to propellant through $t$ insulation during ascent, ground hold and orbit
$Q_S$		Total heat input to propellant through penetrating struts
$\rho_{INS}, \rho_P, \rho_T$		Density of insulation, propellant and tank metal.
$t_A', t_G'$		Effective ascent and ground hold time
$t_G, t_O$		Ground hold and on-orbit time
$T_{AG}, T_{AO}$		Ambient temp on ground and in orbit
$T_P$		Propellant temperature
$\Delta T_G, \Delta T_O$		Temperature difference between external skin and propellant on ground and in orbit
$V_B, V_{INS}, V_R, V_T, V_{TS}, V_U$		Volume of usable ( $\Delta V$ ) propellant, insulation, residual propellant, inside of tank, tank shell and ullage, respectively
$V_E, V_{EO}$		Volume of boiloff due to heat leak through insulation and struts

$V_{TOTAL}$  Total volume of propellant and tank subsystem

$W_B, W_E, W_{INS}, W_P, W_R, W_T$  Mass of usable ( $\Delta V$ ) propellant, boiloff, insulation, total propellant, residual propellant and tank

$X_I$  Thickness of insulation

Total mass of propellant is:

$$W_P = W_B + W_E + W_R = W_B + W_E + f_R W_P \quad (E-1)$$

or 
$$W_P = \frac{W_B + W_E}{1 - f_R} \quad (E-2)$$

and 
$$V_T = \frac{W_P}{\rho_p(1-f_u)} \quad (E-3)$$

Now during ground hold and ascent

$$Q_G + Q_A = q_G A_S (t_G + t'_A) = q_G A_S t'_G \quad (E-4)$$

where

$$q_G = \frac{K_G (T_{AG} - T_P)}{X_I} \quad (E-5)$$

Total heat input is given by:

$$Q_T = Q_A + Q_G + Q_O + Q_S \quad (E-6)$$

where

$$Q_O = \frac{K_O A_S (T_{AO} - T_P)}{X_I} t_o \quad (E-7)$$

and 
$$Q_S = q_s'' (t_o + t'_G) A' \quad (E-8)$$

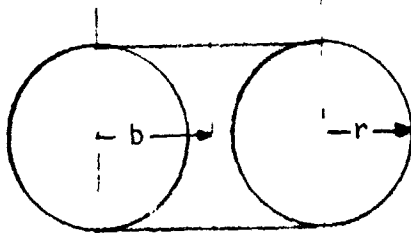
If we assume  $T_{AG} = T_{AO} = T_A$ , Then

$$Q_T = \frac{K_G (T_A - T_P) A_S t'_G}{X_I} + \frac{K_O A_S (T_A - T_P)}{X_I} t_o + q_s'' (t_o + t'_G) A' \quad (E-9)$$

$$= \frac{A_S (T_A - T_P) (K_G t'_G + K_O t_o)}{X_I} + q_s'' (t_o + t'_G) A' \quad (E-10)$$

$$W_E = \frac{Q_T}{h_{fg}} \quad (E-11)$$

Now for a toroidal tank



$$V_T = 2\pi^2 br^2 \quad (E-12)$$

$$A_S = 4\pi^2 br \quad (E-13)$$

$$\frac{V_T}{A_S} = \frac{r}{2}$$

$$\therefore A_S = \frac{2V_T}{r}$$

$$A_S = \frac{2W_p}{r\rho_p(1-f_u-f_r)} \quad (E-14)$$

Let  $C_A = \frac{2}{\rho_p(1-f_u-f_r)} \quad (E-15)$

$$A_S = C_A \frac{W_p}{r} \quad (E-16)$$

$$W_E = \left[ \frac{(K_G t_G' + K_O t_O)(T_A - T_P)(C_A W_p)}{r X_I} + q_s''(t_G' + t_O)A' \right] \cdot \frac{1}{h_{fg}} \quad (E-17)$$

Let  $\frac{(K_G t_G' + K_O t_O)(T_A - T_P)}{h_{fg}} = C_I \quad (E-18)$

and  $\frac{q_s''(t_G' + t_O)A'}{h_{fg}} = W_{EO} \quad (E-19)$

Then

$$W_E = \frac{C_I(C_A W_p)}{r X_I} + W_{EO} \quad (E-20)$$



$$W_P = W_B + W_R + W_E = W_B + f_R W_P + \frac{C_I C_A W_P}{r X_I} + W_{EO} \quad (E-21)$$

$$W_P = \frac{W_B + W_{EO}}{1 - f_R - \frac{C_I C_A}{r X_I}} = \frac{W_B + W_{EO}}{1 - f_R - \frac{C_E}{r X_I}} \quad (E-22)$$

Where  $C_E = C_I C_A$  (E-23)

Now the mass of the tank must be calculated also:

$$W_T = \rho_T X_T A_S = \frac{\rho_T X_T 2V_T}{r}$$

$$= \frac{\rho_T X_T 2W_P}{\rho_P (1 - f_u - f_R) r} \quad (E-24)$$

Now let

$$C_T = \frac{2\rho_T X_T}{\rho_P (1 - f_u - f_R)} \quad (E-25)$$

Then

$$W_T = \frac{C_T W_P}{r} \quad (E-26)$$

and the insulation mass is

$$W_{INS} = \rho_{INS} X_{INS} A_S = \frac{\rho_{INS} X_{INS} C_A W_P}{r} \quad (E-27)$$

Now the total mass of the system can be expressed as

$$W_{PS} = W_P + W_T + W_{INS} \quad (E-28)$$

or

$$\begin{aligned}
 W_{PS} &= W_P + \frac{C_T W_P}{r} + \frac{C_A \rho_{INS} X_{INS} W_P}{r} \\
 &= \left[ 1 + \frac{C_T}{r} + \frac{C_A \rho_{INS} X_{INS}}{r} \right] \left[ \frac{W_P + W_{EO}}{1 - f_R - \frac{C_E}{r X_I}} \right] \\
 &= \left[ 1 + \frac{C_T}{r} + \frac{C_A \rho_{INS} X_{INS}}{r} \right] \left[ \frac{(W_P + W_{EO}) X_I}{(1 - f_R) X_I - \frac{C_E}{r}} \right] \quad (E-29)
 \end{aligned}$$

Let  $a = \frac{(W_B + W_{EO})(C_A \rho_{INS})}{r}$  (E-30)

$$b = \left[ \frac{C_T}{r} + 1 \right] [W_B + W_{EO}] \quad (E-31)$$

$$c = 1 - f_R \quad (E-32)$$

$$d = \frac{C_E}{r} \quad (E-33)$$

and  $W_{PS} = \frac{aX_I^2 + bX_I}{cX_I - d}$  (E-34)

Now  $\frac{dW_{PS}}{dX_I} = 0$  (E-35)

would give the optimum thickness

$$\text{or } \frac{2aX_I + b}{cX_I - d} - \frac{(aX_I^2 + bX_I)c}{(cX_I - d)^2} = 0 \quad (E-36)$$

Therefore,

$$(2aX_I + b)(cX_I - d) = acX_I^2 + bcX_I \quad (\text{E-37})$$

and adding terms

$$acX_I^2 - 2adX_I - bd = 0 \quad (\text{E-38})$$

and using the quadratic equation

$$X_I = \frac{2ad \pm \sqrt{4a^2d^2 + 4abcd}}{2ac} \quad (\text{E-39})$$

$$X_I = \frac{d}{c} + \sqrt{\frac{d^2}{c^2} + \frac{bd}{ac}} \quad (\text{E-40})$$

APPENDIX F  
TANKING DENSITY

SYMBOLS

$a$	-	Acceleration
$B_o$	-	Bond number, ratio of gravitational effects to surface tension effects
$C_l$	-	Heat capacity of the liquid phase
$D_b$	-	Bubble diameter
$g$	-	Acceleration of gravity
$g_o$	-	Universal gravitational constant
$h_{fg}$	-	Latent heat of vaporization
$Ja^*$	-	Modified Jakob number, ratio of heat capacity of liquid to heat capacity of vapor at saturation
$M$	-	Total mass of liquid and vapor
$r_e$	-	Effective bubble radius
$T_{sat}$	-	Saturated liquid temperature
$V^*$	-	Total volume of liquid and vapor after boil off
$\mu$	-	Dynamic viscosity
$\rho_l$	-	Liquid density
$\rho_v$	-	Vapor density
$\bar{\rho}^*$	-	Bulk density
$\sigma$	-	Surface tension
$U_{min}$	-	Minimum bubble rise rate

While the STS is sitting on the launch site with the LTPS tanks loaded, there would be a large enough heat leak to cause boiling of the cryogenic propellants. The creation of bubbles in the liquid causes a decrease in bulk density of the liquid. For this analysis, it was assumed that the boiling rate depends on both the tank surface area and total heat influx.

Using configuration 1 (LO<sub>2</sub>/LH<sub>2</sub>, 100 lbf thrust, 4 burns, MLI) as an example, the method of analysis is as follows:

(a) Calculate the On-Ground Heat Leak Rate

Total Heat Leak = Strut Heat Leak + Insulation Heat Leak.

$$= 3780 \text{ W} \quad (\text{F-1})$$

(b) Calculation of Minimum Detachment Diameter ( $D_b$ )

A lower limit for the bubble diameter ( $D_b$ ) can be found by using the equations given by Rohsenow (Ref. 20) for the minimum bubble radius needed for the bubble to break loose.

$$B_0^{1/2} = \left[ \frac{g(\rho_l - \rho_v)}{g_0 \sigma} \right]^{1/2} \cdot D_b = (4.65 \times 10^{-4})(Ja^*)^{5/4} \quad (\text{F-2})$$

and

$$Ja^* = \frac{\rho_l C_l T_{SAT}}{\rho_v hfg} = 18.07 \quad (\text{F-3})$$

and

$$B_0^{1/2} = 1.733 \times 10^{-2} \quad (\text{F-4})$$

and

$$D_b = B_0^{1/2} \cdot \left[ \frac{g_0 \sigma}{g(\rho_l - \rho_v)} \right]^{1/2} \quad (\text{F-5})$$

$$= 0.028 \text{ mm}$$

$B_0$  - Bond number, ratio of gravitational effects to surface tension effects.

$Ja^*$  - Modified Jakob number, ratio of heat capacity of liquid to heat capacity of vapor at saturation.

(c) Calculate Minimum Bubble Rise Rate ( $V_{\min}$ )

To predict a maximum residency time for the rising vapor the minimum rise rate was chosen. The Log-Log plot in Figure F-1 shows the dependence of rise velocity on bubble diameter, with the plot being split into two regions depending on the effective radius of the bubble. For this analysis, the minimum velocity was chosen from Region II. This minimum was chosen because the volume of the bubbles are dependent on the cube of the radius and as the radius decreases by one or two orders of magnitude, the volume decreases by three to six orders of magnitude. Since the volume of the bubbles creates the density change these very small bubbles would have a very limited effect. Using the velocity relationship for Region II then

$$V_{\min} = 1.41 \left( \frac{\sigma a}{\rho} \right)^{1/4} \quad [\text{Ref. 17}] \quad (\text{F-6})$$

$$V_{\min} = 17.4 \text{ cm/sec}$$

and the corresponding radius is  $r_e = 0.15 \text{ cm}$

(d) Calculate Rise Time

$$\text{Rise Time} = \frac{\text{Depth of Liquid}}{\text{Rise Velocity}} = 24.6 \text{ sec} \quad (\text{F-7})$$

(e) Calculate Amount of Liquid Boiloff Under Steady State Conditions

$$\text{Mass of Vaporized Liquid} = 2.1 \text{ kg} \quad (\text{F-8})$$

$$\text{Volume of Vaporized Liquid} = 1.00 \text{ m}^3 \quad (\text{F-9})$$

$$\text{Volume of Liquid Lost Due to Vaporization} = 0.031 \text{ m}^3 \quad (\text{F-10})$$

(f) Calculate New Bulk Density

$$\text{New Bulk Density} = \frac{M}{V^*} = \frac{M}{V^*} \quad (\text{F-11})$$

$$\frac{M}{V^*} = \frac{(46.25 \text{ m}^3)(68.66 \text{ kg/m}^3)}{(46.25 - 0.031 + 1.00) \text{ m}^3} = 67.25 \text{ kg/m}^3 (= 0.9794\rho)$$

$$\text{From Centaur Data } \bar{\rho}^* = 67.40 \text{ kg/m}^3 = 0.9816\rho$$

Using the same method for the liquid oxygen gives

$$\bar{\rho}^* = 0.9910$$

$$\text{From Centaur Data } \bar{\rho}^* = 0.9957$$

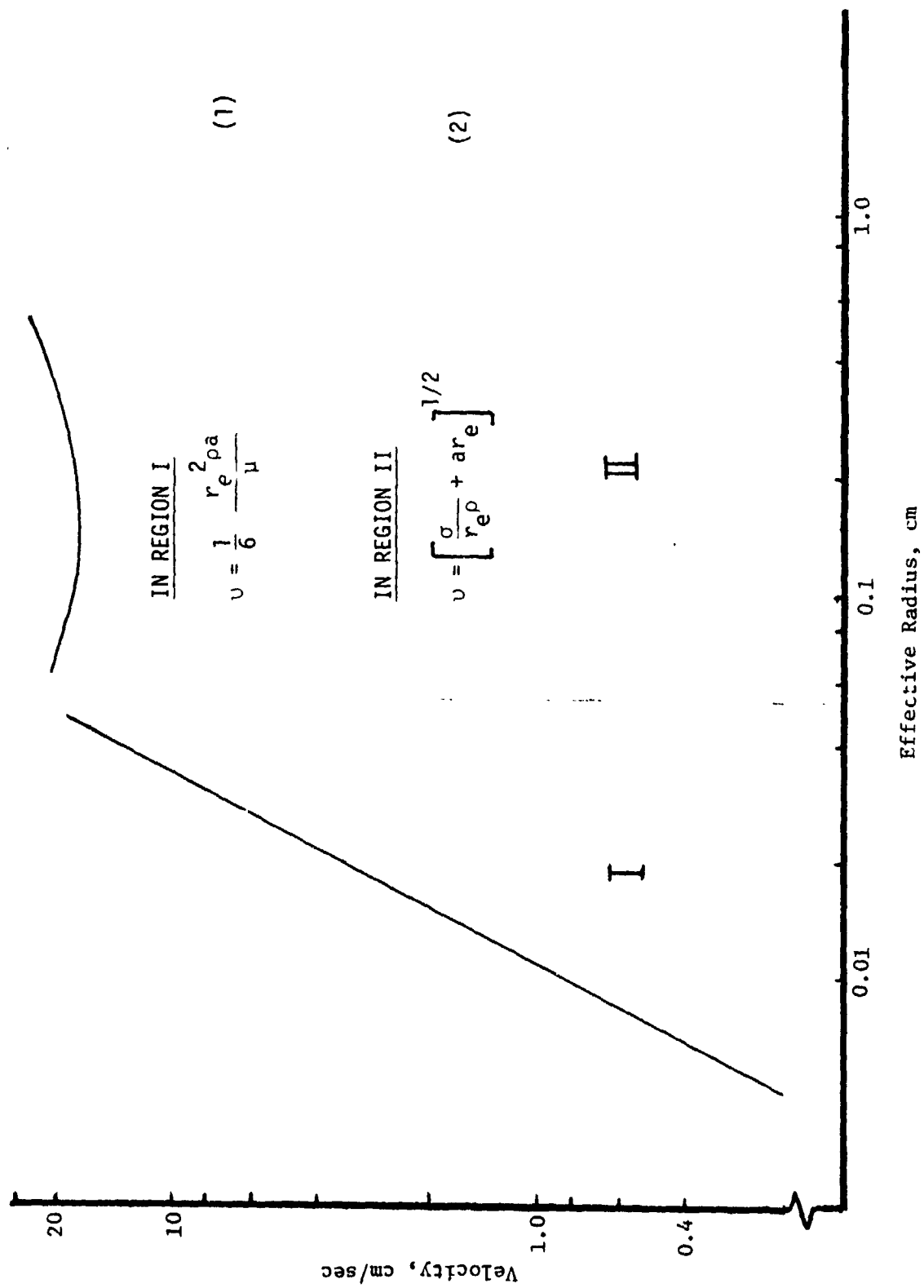


FIGURE F-1 BUBBLE RISE RATE VS EFFECTIVE BUBBLE RADIUS FOR HYDROGEN

The results of the analysis were expected to predict lower densities than the Centaur Data because it was assumed that all the heat leak created boiloff only and that all the tank surface area was in contact with the liquid (for all MLI Systems this was the case).



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