CONTRACT NAS8-27692
DR NO: MA-04

SD 72-SA-0053-3
JUNE 23, 1972

IN-SPACE PROPELLANT LOGISTICS AND SAFETY

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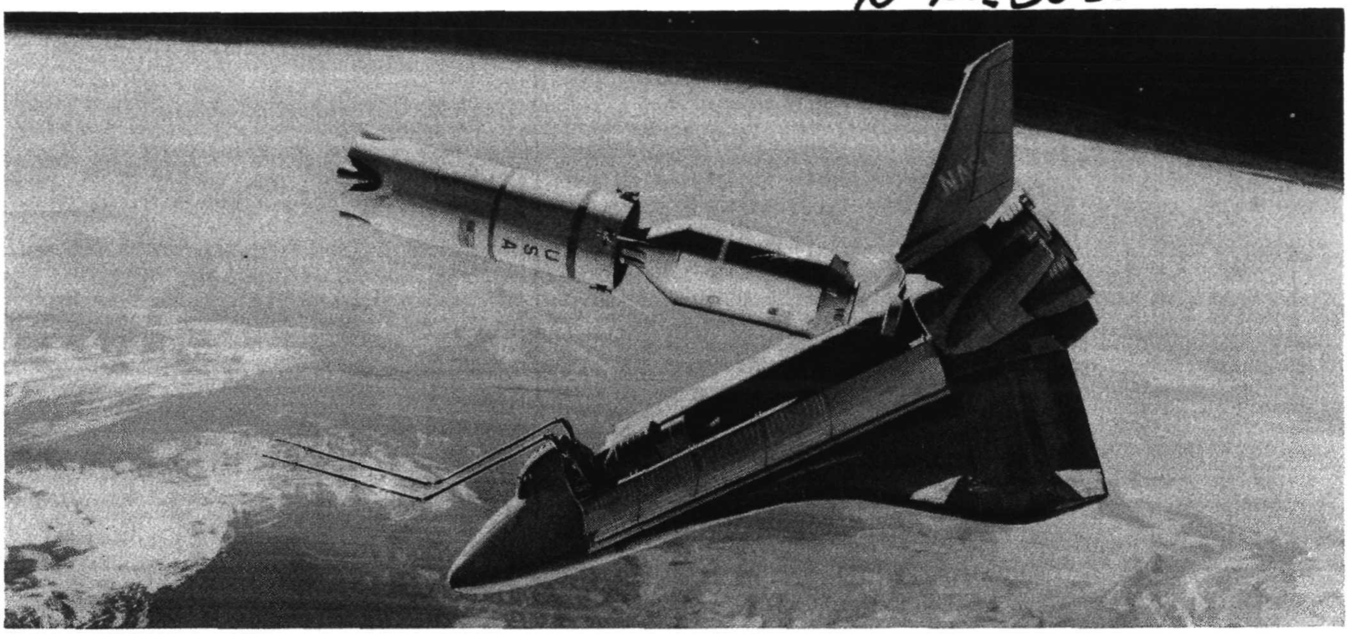

IN-SPACE PROPELLANT LOGISTICS

Volume III

TRADE STUDIES

Space Division
North American Rockwell

12214 Lakewood Boulevard, Downey, California 90241

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Volume III

TRADE STUDIES

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FOREWORD

This In-Space Propellant Logistics and Safety Study was performed by the Space Division of North American Rockwell Corporation for the National Aeronautics and Space Administration, Marshall Space Flight Center, under Contract NAS8-27692. The study was a twelve-month effort initiated on 25 June 1971 and completed on 23 June 1972.

The study was conducted as two separate but related projects. One project addressed the systems and operational problems associated with the transport, transfer, and storage of cryogenic propellants in low-earth orbits, while the other project addressed the safety problems connected with in-space propellant logistics operations. Correlation between the two projects was maintained by including safety considerations, resulting from the System Safety Analysis, in the trade studies and evaluations of alternate operating concepts in the Systems/Operations Analysis.

Walter E. Whitacre of Marshall Space Flight Center, Advanced Systems Analysis Office, was the Contracting Officer's Representative and provided technical direction to the overall contract and to the Systems/Operations Analysis project; Walter Stafford, of the same office, provided technical direction to the System Safety Analysis project. The contractor effort was under the direction of Robert E. Sexton, Program Manager; the Systems/Operations Analysis effort was led by Robert L. Moore and the System Safety Analysis effort was led by William E. Plaisted.

This document is Volume III of the following five volumes, which contain the results of the Systems/Operations Analysis:

Volume I	Executive Summary	(SD72-SA-0053-1)
Volume II	Technical Report	(SD72-SA-0053-2)
Volume III	Trade Studies	(SD72-SA-0053-3)
Volume IV	Project Planning Data	(SD72-SA-0053-4)
Volume V	Cost Estimates	(SD72-SA-0053-5)

The results of the System Safety Analysis portion of the study are contained in the following three volumes:

Volume I	Executive Summary	(SD72-SA-0054-1)
Volume II	System Safety Guidelines and Requirements	(SD72-SA-0054-2)
Volume III	System Safety Analysis	(SD72-SA-0054-3)

The trade studies accomplished in support of the development of cost effective in-space propellant logistic concepts are contained herein.

ACKNOWLEDGMENTS

The contributors to Volume III of this report include the following:

P. Sherman	Subsystem Trades
R. J. Milliken	Propellant Delivery Modes, Parking Orbit Location, and CIS/RNS Depot Requirements
D. A. Thome	Mini-Depot Definition
D. W. Triplett, K. L. Davis, and D. F. Gluck	Propellant Transfer Analysis
C. H. Martinez and J. F. Ross	Propulsion Systems for Propellant Transfer
H. S. Ishikawa	Logistic Tank Thermal Insulation
C. M. McCrary	Propellant Gauging
W. R. Beswick	Slush Hydrogen

CONTENTS

ILLUSTRATIONS

TABLES

LIST OF ABBREVIATIONS AND DEFINITIONS

APS	Auxiliary Propulsion System
AFRPL	Air Force Rocket Propulsion Laboratory
B_o	Bond Number
CER's	Cost Estimating Relationships
c.g.	Center of Gravity
CIS	Chemical Interorbital Shuttle
ESS	Expendable Second Stage
Fr	Froude Number
GH_2	Gaseous Hydrogen
GSE	Ground Support Equipment
I_{sp}	Specific Impulse
I_t	Total Impulse
KSC	Kennedy Space Center
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOX	Liquid Oxygen
MLI	Multilayer Insulation
NPSP	Net Positive Suction Pressure
OPD	Orbital Propellant Depot
OPSS	Orbital Propellant Storage System
OMS	Orbital Maneuvering System
RF	Radio Frequency
RNS	Reusable Nuclear Shuttle
S-II	Saturn Second Stage
SAK	Single Aluminized Kapton

SAM	Single Aluminized Mylar
SH_2	Slush Hydrogen
SMR	Specific Mass Requirements
SS	Space Shuttle
TFU	Theoretical First Unit (Used with Costs)
tug	Space Tug
W_p	Propellant Weight
Cargo sharing	The maximum utilization of a vehicle's payload volume or weight capability by carrying both propellant and dry payload
Cost effectiveness	A measure of the dollar cost of a system or program related to some measure of effectiveness, e.g. $ per lb. of propellant delivered to orbit. Cost effectiveness studies are conducted to compare the relative costs of alternate system programs or approaches in relation to measure of effectiveness
Hydrogen slush (slush hydrogen)	A mixture of small, solid hydrogen particles suspended in liquid hydrogen at the triple point
Linear Propellant Transfer	Acceleration of source tank and receiver tank in X axis direction to settle propellants and permit fluid transfer.
Liquid/Vapor Interface Control	Position management of the liquid to vapor boundaries in the propellant tank
Mass Fraction	The ratio of usable full thrust propellant to gross weight for a space vehicle
Modular transfer	The package exchange of cargo (fluids); i.e., the replacement of an empty tank by a like tank that is full
Operational effectiveness	Any measure of how well the operation carries out its objective used for comparative purposes. It is synonymous with "effectiveness" as used in the term cost effectiveness

Orbital storage	Sometimes referred to as storage. The accumulation and maintenance (saving) of fluid in earth orbit for subsequent transfer to a user vehicle
Program elements	Those propulsive vehicles and orbital stations which are the major hardware components of the space program
Propellant Logistics Module	Propellant tank and associated hardware fitting the shuttle orbiter cargo bay and employed for transporting propellant to the user vehicles
Propellant logistics system	That system which incorporates the transport from ground to space, transfer, and orbital storage (if required) for the purpose of propellant resupply of space-based user vehicles
Receiver tank	That tank accepting propellants in a propellant transfer operation
Rotational Propellant Transfer	Rotation of the propellant source tank and receiver tank about pitch axis to settle propellants and permit fluid pumping
Source tank	That tank supplying the propellants in a propellant transfer operation
Timelines	A sequence of activities in a mission with start and stop times (duration) of the activity defined
Traffic model	A description of the use of a particular vehicle or set of vehicles in terms of the number of trips per unit time, points of departure and destination, trip routes, and trip durations
User traffic model	Refers to the rate of flight of user vehicles
Logistic traffic model	Refers to the rate of flight of the propellant transport, transfer and storage vehicles defined in a propellant logistic system.

Traffic rate	The aspect of a traffic model description specifying the number of trips per unit time
Transport system	System for delivering propellants from earth-to-earth orbit. The tug is not considered part of the transport system for the purpose of this definition.
Transfer	The exchange of propellant or fluid from one vehicle or spacecraft to another vehicle or spacecraft
Tug	A propulsive vehicle for use in space for transporting payloads from one orbit to another. The vehicle may be ground based or space based. Vehicle size allows it to be transported to orbit in the shuttle cargo bay.
User vehicle	A space-based, propulsive vehicle requiring propellant refueling in earth orbit.

1.0 INTRODUCTION AND SUMMARY

This volume contains the details of major trade studies which were conducted in support of Project I, Systems and Operations Analysis, of the In-Space Propellant Logistics and Safety Study (ISPLS). The trade studies presented here are summarized in applicable portions of the Technical Report, Volume II.

Figure 1-1 shows the logic sequence for the conduct of the overall propellant logistic systems analysis study which is presented in Volume II and the relationship of the trade studies to this sequence. The trade studies included here are indicated in the darkened blocks of the figure. The following paragraphs identify the contents of each section of this volume.

Section 2 Propellant Delivery Modes - The delivery mode study is an evaluation of the cost of delivery of propellant to space on a dollars-per-pound basis for delivery by the shuttle, the shuttle in conjunction with the tug, and the use of the shuttle booster with an expendable second stage. The conclusion of the study is that propellant delivery by the shuttle alone with a propellant logistic module in its cargo bay is the most economical mode of delivery. This mode of delivery is used subsequently throughout the study.

Section 3 Parking Orbit Location - The parking orbit location study resulted in selection of 180 n mi at an inclination of 28.5 degrees as the preferred parking orbit location for a space-based tug and a supporting propellant depot. The same altitude is also the preferred altitude taken from the concurrent study for operation of the space-based chemical interorbital shuttle (CIS) and assumed herein for the reusable nuclear shuttle (RNS). These would be at an inclination of about 37 degrees.

Section 4 CIS/RNS Depot Requirements - The CIS/RNS depot study evaluated the requirement for relatively large depots which would be required to support a CIS or the RNS in their planned missions. The conclusion of the study is that CIS or RNS depots are not required. An indicated need for additional storage at high CIS or RNS flight rates could better be met by the use of two space-based CIS or RNS vehicles.

Section 5 Mini-Depot Definition - This section contains details of the mini-depot concepts evaluated during this study. The mini depot is a tug supportive depot developed and evaluated after it was concluded that a large CIS supportive depot was not required. The mini depot with other

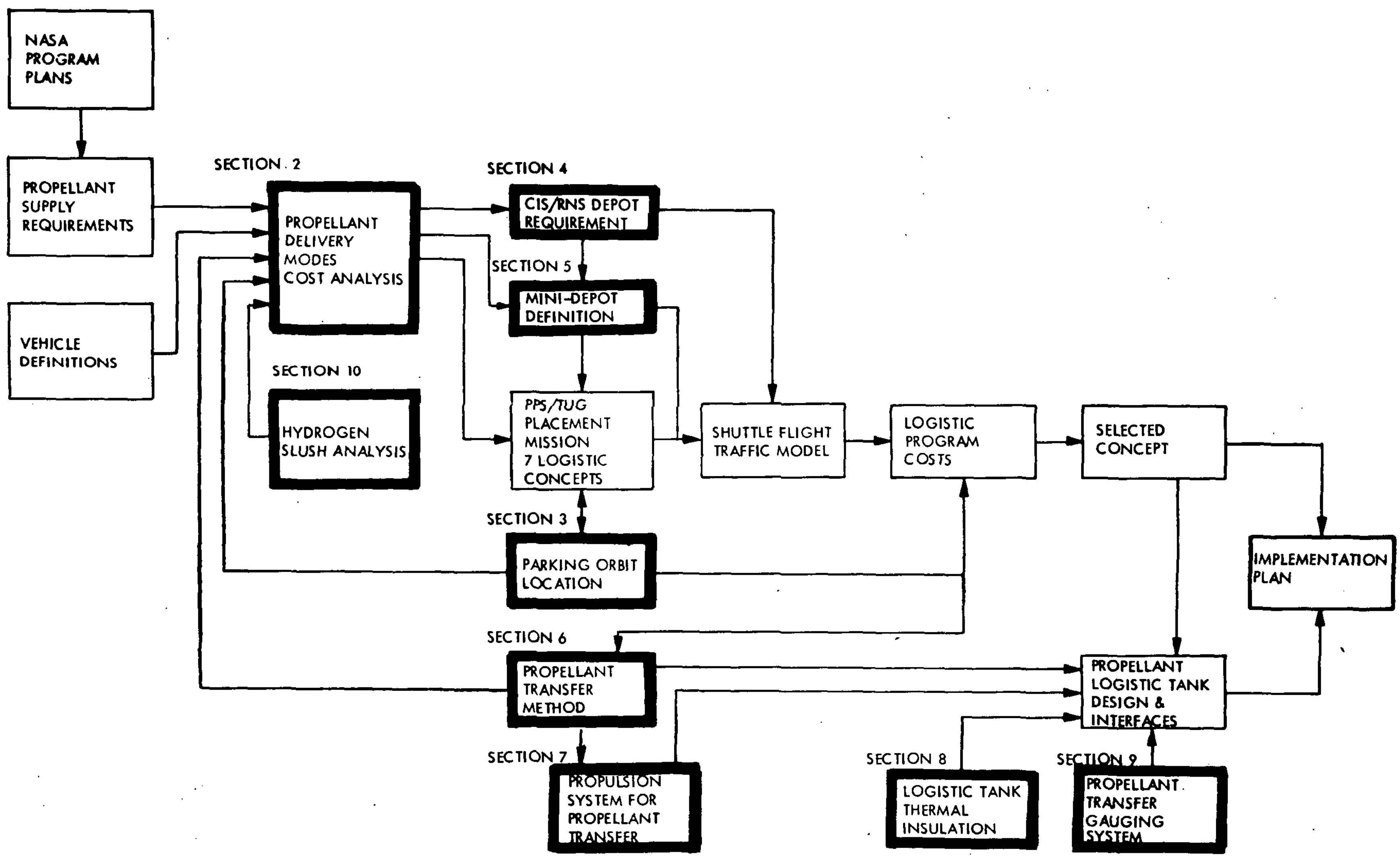


Figure 1-1 Trade Studies Interrelationship

candidate storage concepts were then incorporated into candidate operational concepts for conduct of the tug payload placement missions. The evaluation of the concepts comparing logistic program costs for each concept is contained in Section 7 of Volume II.

Section 6 Propellant Transfer Method - The propellant transfer study contains an evaluation of the requirements and feasibility of alternate concepts for the transfer of propellant in space. Propellant losses are analyzed in the study, and the study includes an optimization of propellant tank geometry for propellant transfer. After it was determined in other trade studies that depots were not required, and the shuttle with a propellant logistics module was the preferred mode for delivery of propellant to space, an analysis was made in the propellant transfer study to select the preferred method for transfer of propellant from the logistics module to the tug, CIS and RNS. The transfer mode using linear acceleration for propellant settling, with the logistic module attached to the receiver vehicle and the shuttle detached was selected as the preferred mode in all cases.

Section 7 Propulsion Systems for Propellant Transfer - This study deals with the selection and location of low-thrust jets required to provide acceleration for propellant settling during the propellant transfer operation. Jets which use the available LH_2 and LO_2 propellant and are mounted on the logistic module are selected as the preferred concept. The study was conducted primarily to support the development of the design concept for the logistic module presented in Section 9 of Volume II.

Section 8 Logistic Tank Thermal Insulation - Insulation alternatives for the propellant logistic module (Section 9, Volume II) are evaluated in this study. Multilayer insulation with provisions for purging is the selected concept.

Section 9 Propellant Gauging - Propellant gauging concepts for use in the low-g propellant transfer environment are investigated in this study in support of the definition of the logistics module presented in Section 9 of Volume II. A resistance wire with resistance characteristics modified by temperature and mounted normal to the propellant surface is selected as the preferred concept.

Section 10 Slush Hydrogen - The slush hydrogen study was conducted to evaluate the potential use of hydrogen slush to reduce losses in the hydrogen propellant systems. The basic conclusion of the study is that the dollar value of the potential propellant savings by the use of hydrogen slush over the life of the currently planned space program would be far exceeded by the development costs. The use of slush was, therefore, not included in the program propellant logistics concepts recommended in the ISPLS study.

2.0 PROPELLANT DELIVERY MODES

2.1 INTRODUCTION

This study consists of the determination of the minimum cost for delivery of propellant to space on a dollar-per-pound basis for three delivery modes under consideration. The three delivery modes consist of the space shuttle vehicle, the shuttle booster with an expendable second stage (ESS) and the shuttle plus the space tug. In the latter case, the shuttle delivers the propellant to an orbit at 100 n mi; the tug picks up the propellant payload and delivers it to the higher altitude. One hundred eighty nautical miles is selected as the terminal altitude because this is the indicated parking orbit altitude for the space based tug and CIS. The latest performance estimates for the space shuttle indicate that it can carry a full 65,000-pound payload to 180 n mi altitude. The study is based on the shuttle payload capability of 65,000 pounds at all altitudes from 100 n mi to 180 n mi. The effect of an increase in the cargo bay payload by a corresponding off-loading of the Orbital Maneuvering System (OMS) propellant for operation at 100 n mi is discussed in the study.

A summary of the study is presented in Section 7 of Volume II.

2.2 SUMMARY OF RESULTS

The delivery mode trade study is summarized in Table 2.2-1 which indicates that oxygen-hydrogen propellant in the tug/CIS supportive cases can be delivered to space by the space shuttle for $178 per pound. The addition of tug costs to the shuttle costs makes the shuttle-plus-tug mode a more costly mode within the 180 n mi range of the shuttle. At higher altitudes, beyond the 65,000-pound shuttle payload capability, the shuttle-plus-tug would become the cheaper mode. In the case of the booster-ESS, the increased payload capability of the ESS is not sufficient to offset the high cost of expending the stage and the mode is more expensive than the reusable shuttle modes.

In the case of hydrogen delivery alone to support an RNS, the shuttle cargo bay is volume limited and holds less than 35,000 pounds of hydrogen. In this case the booster-ESS would be the cheaper mode. The addition of supplemental external hydrogen tanks on the shuttle could, however, offset this advantage. It should be noted that only production and operational costs of the booster-ESS are included in this analysis. If the development costs for an ESS were included, the costs would be much higher.

2.3 SHUTTLE AND SHUTTLE-PLUS-TUG DELIVERY MODES

The compilation of costs for the shuttle direct and shuttle plus tug case are shown in Table 2.3-1. Costs of $10 million per flight for the shuttle and $1.3 million for the space-based

TABLE 2.2-1 PROPELLANT DELIVERY COST SUMMARY TO 180 N MI.

	OXYGEN-HYDROGEN (TUG/CIS SUPPORTIVE)	HYDROGEN (RNS SUPPORTIVE)
SHUTTLE DIRECT	$178 PER LB.	$314 PER LB. *
SHUTTLE + TUG	210 PER LB.	363 PER LB.
BOOSTER-ESS	245 PER LB.	237 PER LB.

*$235 PER LB. WITH EXTERNAL HYDROGEN TANK

Table 2.3-1 Propellant Delivery Costs - Shuttle Delivery Modes
(Dollars Per Pound)

	OXYGEN - HYDROGEN (TUG/CIS SUPPORTIVE)			HYDROGEN (RNS SUPPORTIVE)			HYDROGEN WITH EXTERNAL (SHUTTLE TANKS)	
	SHUTTLE DIRECT		SHUTTLE + TUG	SHUTTLE DIRECT		SHUTTLE + TUG	SHUTTLE DIRECT	
	100 nmi	180 nmi	180 nmi	100 nmi	180 nmi	100 nmi	100 nmi	180 nmi
VEHICLE COST PER FLIGHT	$M	$M	$M	$M	$M	$M	$M	$M
SHUTTLE	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
TUG	-	-	1.3	-	-	1.3	-	-
TANK	.11	.11	.11	.13	.13	0.17	1.6	1.6
TOTAL COST PER FLIGHT	10.11	10.11	11.41	10.13	10.13	11.47	11.6	11.6
PROPELLANT WEIGHT	K LBS	K LBS	K LBS	K LBS	K LBS	K LBS	K LBS	K LBS
SHUTTLE PAYLOAD CAPABILITY	65.0	65.0	65.0	65.0	65.0	65.0	65.0	65.0
TANK	5.1	5.1	5.1	7.0	7.0	8.7	13.0	13.0
PROPELLANT IN TANK	59.9	59.9	59.9	34.0	34.0	35.5	52.0	52.0
PROPELLANT USED BY TUG	-	-	2.6	-	-	2.2	-	-
PROPELLANT DELIVERED BEFORE LOSS	59.9	59.9	57.3	34.0	34.0	33.3	52.0	52.0
5% TRANSFER LOSS	3.0	3.0	2.9	1.7	1.7	1.7	2.6	2.6
PROPELLANT DELIVERED	56.9	56.9	54.4	32.3	32.3	31.6	49.4	49.4
DOLLARS PER POUND DELIVERED	$178	$178	$210	$314	$314	$363	$235	$235

tug have been used in the analysis. These costs are discussed in Section 7 of Volume II, and are the same costs used throughout the analytical portion of this study.

The $1.3 million for the tug is derived from tug purchase costs of $38 million prorated over a life of fifty missions plus an allowance for placing the tug in space and returning it to earth once every ten missions for maintenance. The tanks employed in the study are also illustrated in Section 7 of Volume II. Their costs include development costs as well as production and maintenance costs for a buy of five tanks prorated over an assumed life of 100 missions each.

In the shuttle-plus-tug cases, in which the tug picks up the propellant from the shuttle cargo bay at 100 n mi and delivers it to 180 n mi, the propellant used by the tug is subtracted from the total propellant delivered.

The weight and cost of the hydrogen tank in the RNS supportive case is more expensive and heavier than the oxygen-hydrogen tank because of its greater volume, 60 feet versus 38 feet long, although the 60-foot-long tank holds only 34,000 pounds of hydrogen.(1) In the RNS supportive shuttle-plus-tug case, a tank with a capacity of 1,900 pounds of oxygen and 33,600 pounds of hydrogen has been assumed in order to provide the oxygen as well as hydrogen required for the operation of the tug. The tug requires a mixture ratio of six pounds of oxygen to one of hydrogen.

2.3.1 Shuttle Payload Capability

The cost of delivery of propellant by the shuttle direct, $178 per pound in the oxygen-hydrogen case and $314 per pound in the all-hydrogen case, does not vary between 100 and 180 n mi altitude because of the limitation of 65,000 pounds on the payload in the shuttle cargo bay at both altitudes. Latest performance data estimated for the proposed shuttle now indicate that it will have the capability of delivering 65,000 pounds of payload or more to 50 x 100 n mi orbit in an easterly launch inclination from 28.5 degrees to about 37 degrees, and with 23,500 pounds of OMS propellant aboard for on-orbit maneuvers. The OMS propellant will then have the capability of delivering the shuttle with the 65,000-pound payload to about 180 n mi altitude. The OMS propellant is sufficient to de-orbit the shuttle with an empty propellant payload tank and retain 5000 pounds of propellant (equivalent to about 200 fps differential velocity (ΔV)) as a reserve for contingencies. This propellant delivery mission capability is based on the following differential velocity budget.

(1) The tank weights and capacities employed here were developed early in the ISPLS study. Values of weights and volumes changed when more detail tank designs were developed later in the study; however, the changes do not appreciably affect the conclusions of this section.

	Delta V
Transfer 50 x 100 n mi to 180 x 180 n mi	360 fps
Two rendezvous and dockings	180
De-orbit	320
Reserve for contingencies	200
	1,060 fps

The mission is based on the receiver vehicle being in a suitable or compatible orbit for rendezvous as no budget for turns or phasing orbits has been included. The estimated payload capability of the shuttle for this mission applicable to this study is presented in Figure 2.3-1.

If it were possible to increase the payload in the shuttle cargo bay above 65,000 pounds for operation to 100 n mi by a corresponding offloading of OMS propellant weight, the dollar-per-pound cost of delivery of propellant to 100 n mi would be reduced. For operation at 100 n mi, assume a delta-V budget of 690 fps:

	Delta V
Circularize at 100 n mi	90 fps
Two rendezvous and dockings	180
De-orbit	220
Reserve for contingencies	200
	690 fps

With the reduced delta-V budget for operation at 100 n mi as compared with 180 n mi, the on-board OMS propellant could be reduced by about 8000 pounds with a corresponding increase in propellant payload to 73,000 pounds. If this were the case, a net weight of about 64,000 pounds of propellant could be delivered to 100 n mi after an allowance for tank weight and five percent propellant losses. This would yield a cost of $158 per pound at 100 n mi. Then, if the tug picked up the tank with 64,000 pounds of propellant at 100 n mi and delivered it to 180 n mi, the shuttle-plus-tug delivery cost at 180 n mi would be $186 per pound after allowance for losses. It is noted that this cost is still considerably in excess of the shuttle direct cost.

2.3.2 Hydrogen Delivery with Supplemental External Tanks

In the case of the delivery of hydrogen in the RNS supportive cases, it has been noted above that the volume of the shuttle cargo bay is the limiting factor on payload so that the costs per pound delivered are higher than for oxygen-hydrogen combined. If there were

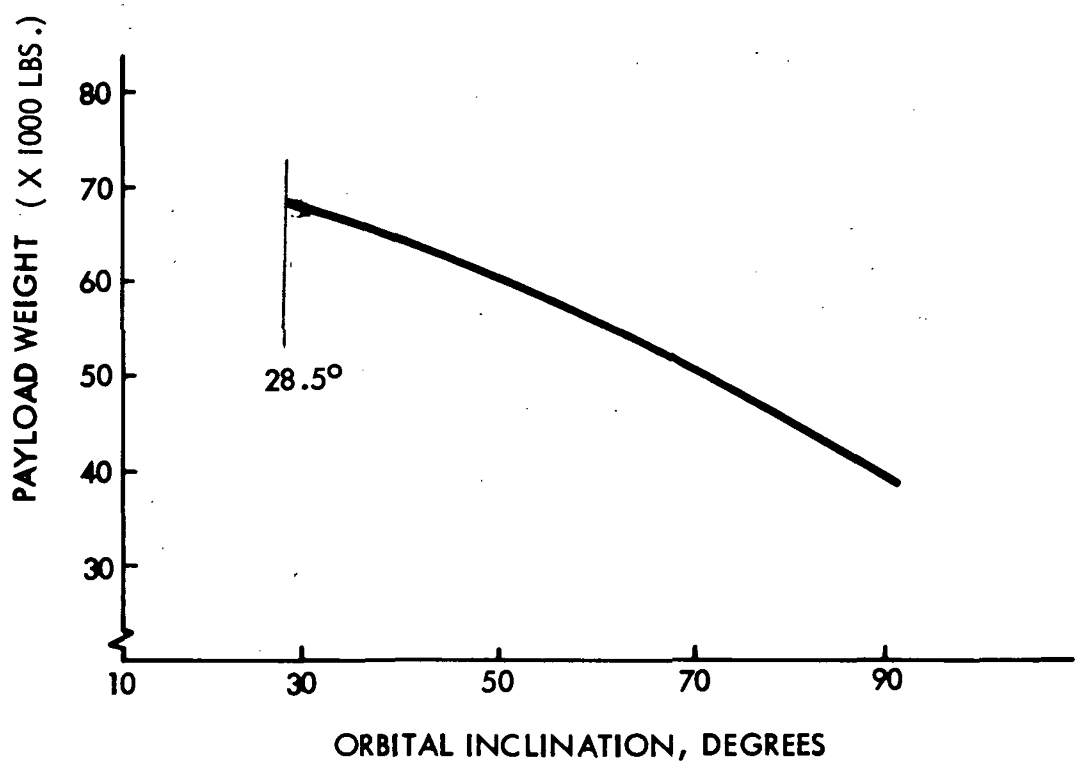


Figure 2.3-1 Shuttle Payload Capability to 180 nmi Propellant Delivery Mission

an RNS program which required large quantities of hydrogen in space and many repeated shuttle trips for its delivery, additional hydrogen storage could be provided on the shuttle. This could possibly be done by using strap-on tanks or by a humpback tank over the cargo bay. The case of "Hydrogen with External Shuttle Tanks" columns in Table 2.3-1 is based on the assumption that wing tanks would be used. An allowance was made for the weight and cost of such tanks which would permit the full 65,000-pound payload capability of the shuttle to be used. The estimates resulted in an allowance of 6,000 pounds weight and $1.5M cost for the expendable tanks with a capacity of 18,000 pounds of hydrogen. These figures yielded the indicated delivery costs of $235 per pound in Table 2.3-1. A detailed estimate was not made of either the tank weights or their costs. The values were derived by taking ratios of tank costs and weights with judgment allowances for differences in requirements.

Although the delivery cost estimates for the external hydrogen tank case are only approximate, they indicate the potential cost reduction which might be achieved by this technique.

2.4 SHUTTLE BOOSTER-ESS DELIVERY MODES

Estimates of the cost of delivery of propellant to space employing a combination of shuttle booster and expendable second stage are presented in Table 2.4-1. The estimates have been prepared in a manner comparable to that used for the reusable shuttle presented above in Table 2.3-1. The data indicate that the delivery of oxygen and hydrogen for the tug/CIS supportive cases is more expensive for the booster-ESS than for the reusable shuttle. The increased payload capability of the ESS is not sufficient to offset the estimated cost of $35.2M for the expended stage.

The delivery costs for all hydrogen are slightly less than for the oxygen-hydrogen mix because of the lower cost and weight of the all-hydrogen tank as opposed to two tanks for hydrogen and oxygen. The booster-ESS payload tank is not limited in volume for carrying hydrogen as is the shuttle cargo bay.

In the case of the booster-ESS plus tug, a space-based tug picks up the booster-ESS propellant tank at 100 n mi and transfers it to 180 n mi. Similar to the case of the shuttle delivery mode, the addition of the tug costs and the allowance for tug propellant use make this a more costly mode to 180 n mi than the use of the booster-ESS alone.

Two modes of operation are presented in Table 2.4-1, one with and one without retrieval of high cost components. In the without-retrieval case, the entire expendable stage, along with the payload tank, are destroyed on re-entry into the atmosphere. In the with-retrieval case, high cost components including the engines and

Table 2.4-1 Propellant Delivery Costs - Booster-ESS (Dollars per Pound)

	OXYGEN - HYDROGEN (TUG/CIS SUPPORTIVE)			HYDROGEN (RNS SUPPORTIVE)		
	BOOSTER - ESS		BOOSTER - ESS + TUG	BOOSTER - ESS		BOOSTER - ESS + TUG
WITHOUT RETRIEVAL OF HIGH COST COMPONENTS	100 nmi	180 nmi	180 nmi	100 nmi	180 nmi	180 nmi
VEHICLE COSTS PER FLIGHT	\$M	\$M	\$M	\$M	\$M	\$M
BOOSTER	3.3	3.3	3.3	3.3	3.3	3.3
ESS	35.2	35.2	35.2	35.2	35.2	35.2
TANK	4.5	4.4	4.5	3.6	3.5	3.6
TUG	-	-	1.3	-	-	1.3
TOTAL COST PER FLIGHT	43.0	42.9	44.3	42.1	42.0	43.4
PROPELLANT WEIGHT	K LBS	K LBS	K LBS	K LBS	K LBS	K LBS
BOOSTER-ESS PAYLOAD CAPABILITY	206.0	200.0	206.0	206.0	200.0	206.0
TANK WEIGHT	16.0	15.8	16.0	13.0	12.8	13.0
PROPELLANT IN TANK	190.0	184.2	190.0	193.0	187.2	193.0
PROPELLANT USED BY TUG	-	-	7.5	-	-	7.5
5% TRANSFER LOSS	9.5	9.2	9.5	9.5	9.4	9.7
TOTAL PROPELLANT DELIVERED	180.5	175.0	173.0	183.5	177.8	175.8
DOLLARS PER POUND DELIVERED	\$238	\$245	\$256	\$229	\$237	\$247
WITH RETRIEVAL OF HIGH COST COMPONENTS						
VEHICLE COSTS PER FLIGHT	\$M	\$M	\$M	\$M	\$M	\$M
BOOSTER	3.3	3.3	3.3	3.3	3.3	3.3
ESS	28.9	28.9	28.9	28.9	28.9	28.9
TANK	4.5	4.4	4.5	3.6	3.5	3.6
TUG	-	-	1.3	-	-	1.3
TOTAL COST PER FLIGHT	36.7	36.6	38.0	35.8	35.7	37.1
DOLLARS PER POUND DELIVERED	\$203	\$209	\$220	\$195	\$201	\$211

inertial electronics equipment are retrieved before the stage is destroyed. This mode was evaluated in the booster-ESS study, from which the booster-ESS data are taken. The retrieval would be accomplished by personnel operating from a shuttle vehicle. The retrieval would result in a net reduction in the cost of the expended stage from \$35.2M to \$28.9M. Consideration has not been given to the cost of the shuttle retrieval operation in the present analysis so that the data in Table 2.4-1 should be considered as indicating the maximum benefit which might be derived from the with-retrieval operation.

2.4.1 BOOSTER-ESS SOURCE DATA

For the present analysis, data on the booster-ESS were derived from the booster-ESS study contract which was part of the Phase B shuttle contract (NAS 9-10960). The booster-ESS study was undertaken because of interest in individual payloads which exceeded the 65,000 pound capability of the shuttle and could not be divided into lighter weight portions. The booster-ESS study was completed prior to initiation of the ISPLS study contract.

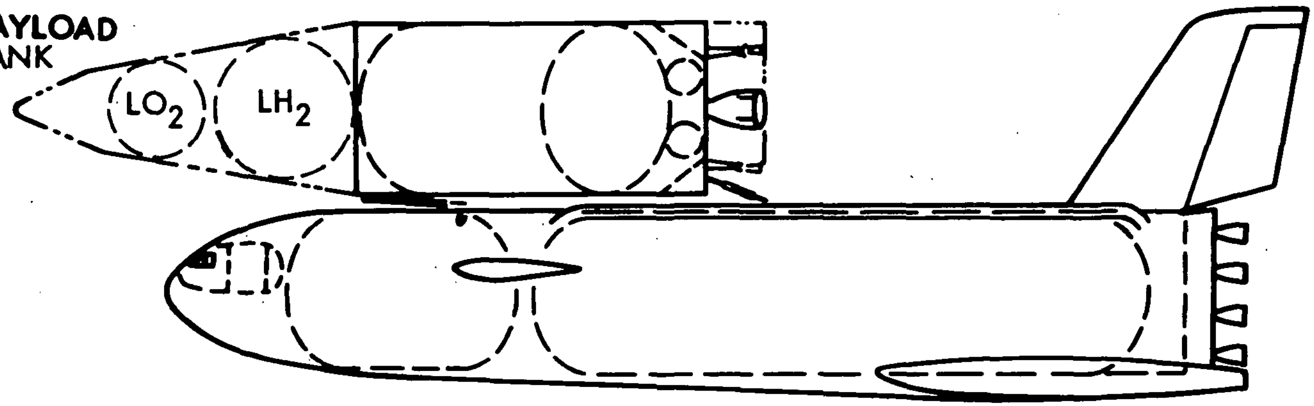
Data on the performance and configuration of the booster-ESS employed in the study are presented in Figures 2.4-1 and 2.4-2. The study was based on the flyback recoverable booster, B9U, which was the proposed concept at that time. The overall payload capability of the shuttle and booster have decreased slightly since that time so that results of the study may be considered as favoring the booster-ESS case slightly when considered in relation to later shuttle booster capability.

In an easterly launch of 28.5 degrees, the booster-ESS has a payload capability of 206,000 pounds to 100 n mi and 200,000 pounds to 180 n mi. Allowance for propellant tank weights, propellant used by the tug, if any, and a five percent transfer loss have been considered in calculating the net weight of propellant delivered as indicated in Table 2.4-1. The estimates of weights and costs for the propellant tanks to be used with the booster-ESS were made in this study. The tank costs include development costs as well as production and operations costs. The costs were based on fifty LO_2/LH_2 tanks and derived as follows:

Development	\$ 53M	
Production and launch preparation	170M	
	\$223M	÷ 50 = \$4.5M

Costs and weights for the all-hydrogen tanks are less because of the requirement for one fluid instead of two.

Costs for the expendable second stage were derived from the booster-ESS study and resulted in average cost of \$35.2M for the



BOOSTER:		ESS:		PAYLOAD CAPABILITY	
INERT WT	655,000 LB	INERT WT	103,000 LB	TO 100 nmi	206,000 LB
PROPELLANT (ASCENT)	3,382,000 LB	PROPELLANT (ASCENT)	677,000 LB	TO 180 nmi	200,000 LB
THRUST (SL)	6,600,000 LB	THRUST (VAC)	1,264,000 LB		

Figure 2.4-1 Shuttle Booster & ESS

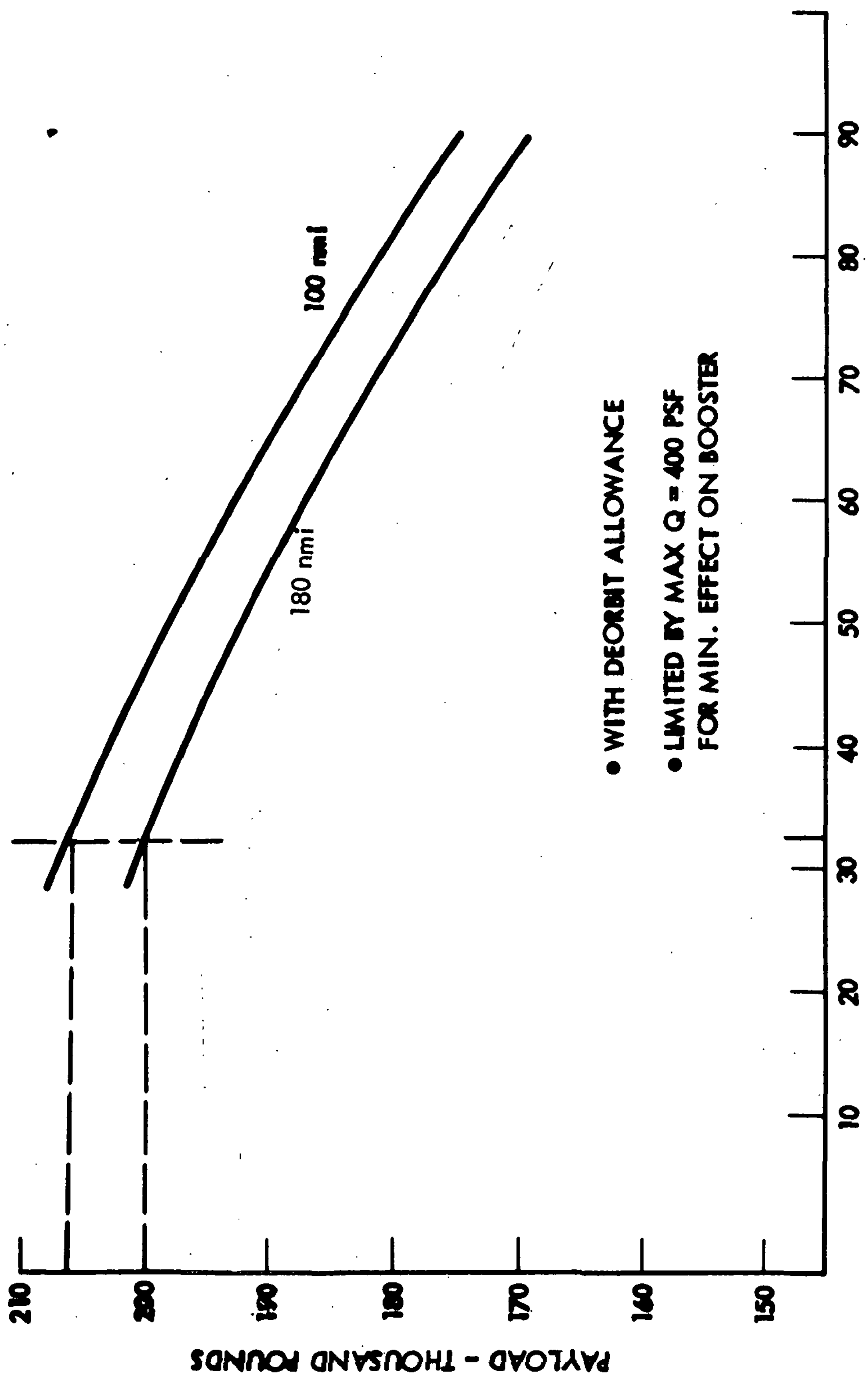


Figure 2.4-2 Booster-ESS Payload Capability

stage including its production and preparation for launch. Development costs are not included in this figure. As stated above, retrieval of the high cost components reduced the average figure for the study to $28.9M.

The costs for the shuttle booster of $3.3M per flight are the portion applicable to the booster of the $10M per shuttle orbiter and booster flight which is used throughout this study as follows:

Space Shuttle Program	Orbiter	Booster	Total
Production costs	$1562M	$ 674M	$2236M
Operations costs	1522	800	2322
	$3084M	$1474M	$4558M
Prorated over 444 operational flights in program.	$6.9M	$3.3M	$10.2M

These costs were calculated for the reusable pressure fed booster as of 15 February 1972 (See Section 7, Volume II). The $3.3M for the booster operation is higher than the value earlier estimated for the B9U flyback booster, which was $2.2M. The value of $3.3M is believed, however, to be more representative of actual anticipated costs for the booster. The sold rocket motor booster will of course result in further changes in the booster cost figure. It should be noted, however, that changes in the booster cost will have a small effect on the final result of the analysis because it is a small number compared with the $35M cost for the ESS.

3.0 PARKING ORBIT LOCATION

3.1 INTRODUCTION

The purpose of this study is to determine a best parking orbit altitude and inclination for the space-based tug and supporting mini-depot or other tug supportive propellant storage system. Although a mini-depot or other separate propellant storage system was not part of the recommended concept in the ISPLS study, the results of this study remain applicable to the space-based tug itself. Much of the study consists of reviewing data generated in other portions of this study for their effect on the parking orbit selection.

Consideration was given to the following factors in making the selection:

a. Rendezvous compatibility of the orbital location with shuttle launches from Kennedy Space Center (KSC).

b. Propellant requirements for orbital maintenance.

c. Shuttle payload capability for the delivery of propellant versus altitude.

d. The use of tug in a propellant transfer mode.

e. The effect of parking orbit location on tug propellant requirements in the tug missions.

3.2 SUMMARY OF RESULTS

An altitude of 180 n mi and inclination of 28.5 degrees is the recommended parking orbit location. The selection will support the majority of the ISPLS model tug missions, which lie at inclinations between 0 and 30 degrees. The altitude provides rendezvous compatibility with KSC launches every second day. It is high enough so that the propellant requirement for orbital maintenance is negligible. It is also within range of the shuttle capability to deliver a full 65,000-pound payload.

Rendezvous compatible orbits for 28.5 degrees from KSC exist on an every day basis at about 100 n mi and 260 n mi altitudes. The former was rejected because of drag and short orbital life, and the latter because it was too high for economical delivery of propellant.

3.3 ORBITAL LOCATION

3.3.1 Orbital Inclination

The following table indicates the approximate distribution of mission and propellant requirements in the ISPLS model with respect to orbital inclination.

	No. of Missions	Propellant Quantity	Propellant Percent
0 - 28.5°	90	5,554,334	78.0
28.5° - 30°	15	1,096,979	15.4
55°	6	160,914	2.3
90° - 100.7°	46	303,772	4.3
	157	7,115,999	100.0

Derived from Program Level C, space-based tug operations, 1985-1990

It was determined in Section 2.3.4 of Volume II, entitled "Polar Versus Easterly Missions", that the most economical mode of operation for the polar missions (90 to 100.7 degrees) was a ground-based vehicle so that the use of a space-based tug in polar orbit was ruled out. The few 55-degree missions can also be flown most economically by a ground-based tug. The propellant consumption to make the required turn from 28.5 degrees to either polar inclinations or to 55 degrees and return would be prohibitive. Propellant requirements for tug turns are indicated in Figure 3.3-1. A tug at an inclination of 28.5 degrees could serve all missions from 0 to 30 degrees. The 0 to 28.5 degree missions, of course, employ shuttle launches at 28.5 degrees from KSC. Basing a tug at zero degrees would require the shuttle to make the costly turn from 28.5 degrees to zero degrees and return. A tug based at 28.5 degrees can make the turn to 30 degrees and return for less than 4,000 pounds of propellant in a full load mission, which is about 7 percent of a shuttle payload and would, therefore, be practical. The inclination of 28.5 degrees was, therefore, selected as the space-based tug orbital inclination.

3.3.2 Orbital Altitude

A primary consideration in the selection of altitude was that the orbital location of the tug be compatible for rendezvous with shuttle launches from KSC. This would eliminate the requirement for costly phasing orbits to rendezvous with the space-based tug. Figure 3.3-2 presents orbital altitudes and inclinations for such rendezvous compatible orbits. The solid lines represent the location of orbits which provide compatibility for rendezvous at periodic intervals of the indicated number of days. A one-day rendezvous compatible orbit means that when KSC rotates into the plane of the tug orbit, the tug is simultaneously in the correct position in its orbit so that a shuttle launched from KSC will meet the tug without a requirement for a plane change or for a phasing orbit, once every day. In other words, the tug and KSC are synchronized in their rotations so that they are in the same position with respect to each other once every day. The calculation of the rendezvous compatible orbit locations involves consideration of the rotational period of the earth, the orbital period of the tug, and also the rate of precession of the tug orbital plane such that KSC and the tug are synchronized in the same relative position with respect to each other on a periodic basis.

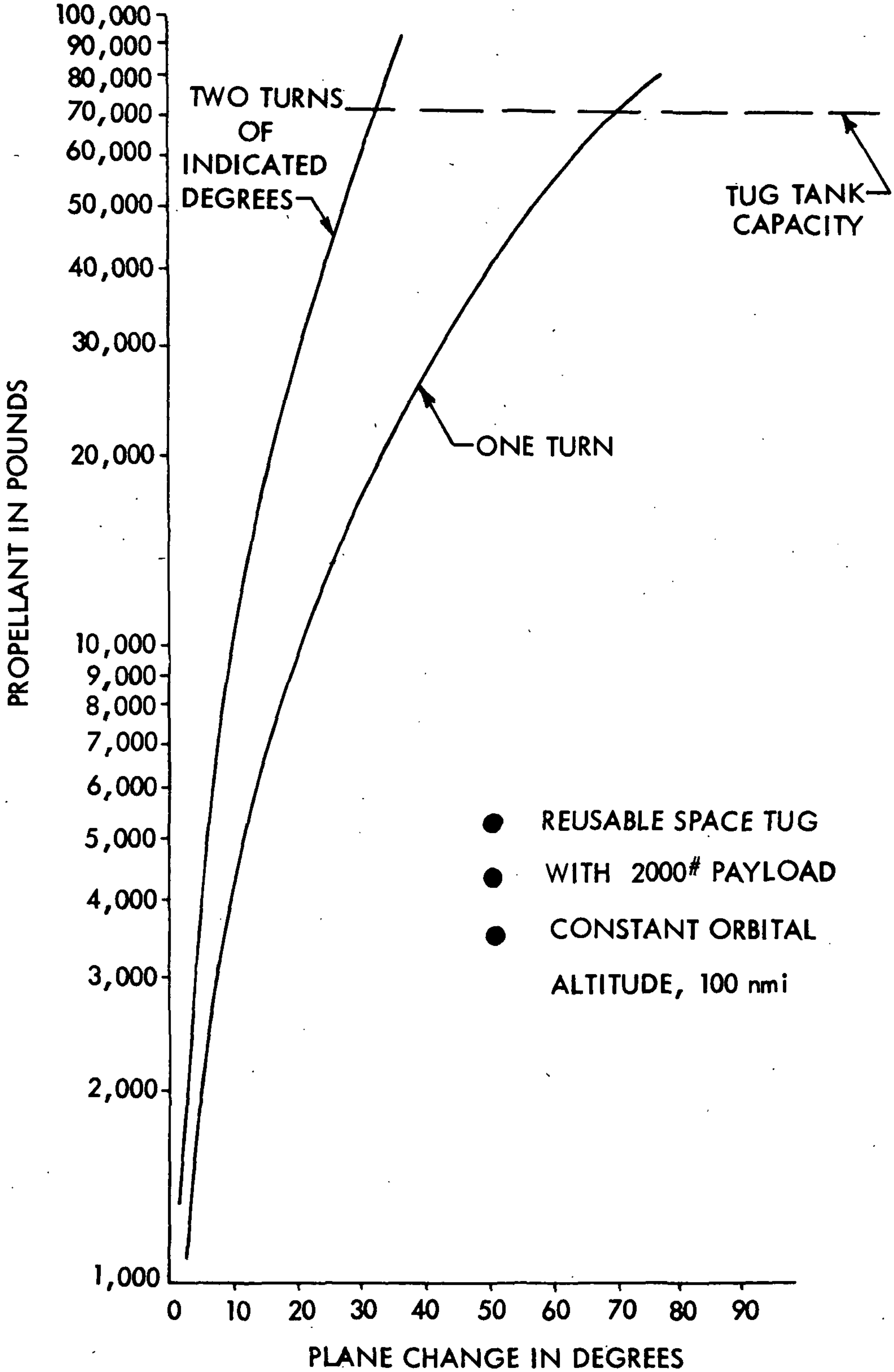


Figure 3.3-1 Propellant Requirements for Plane Changes

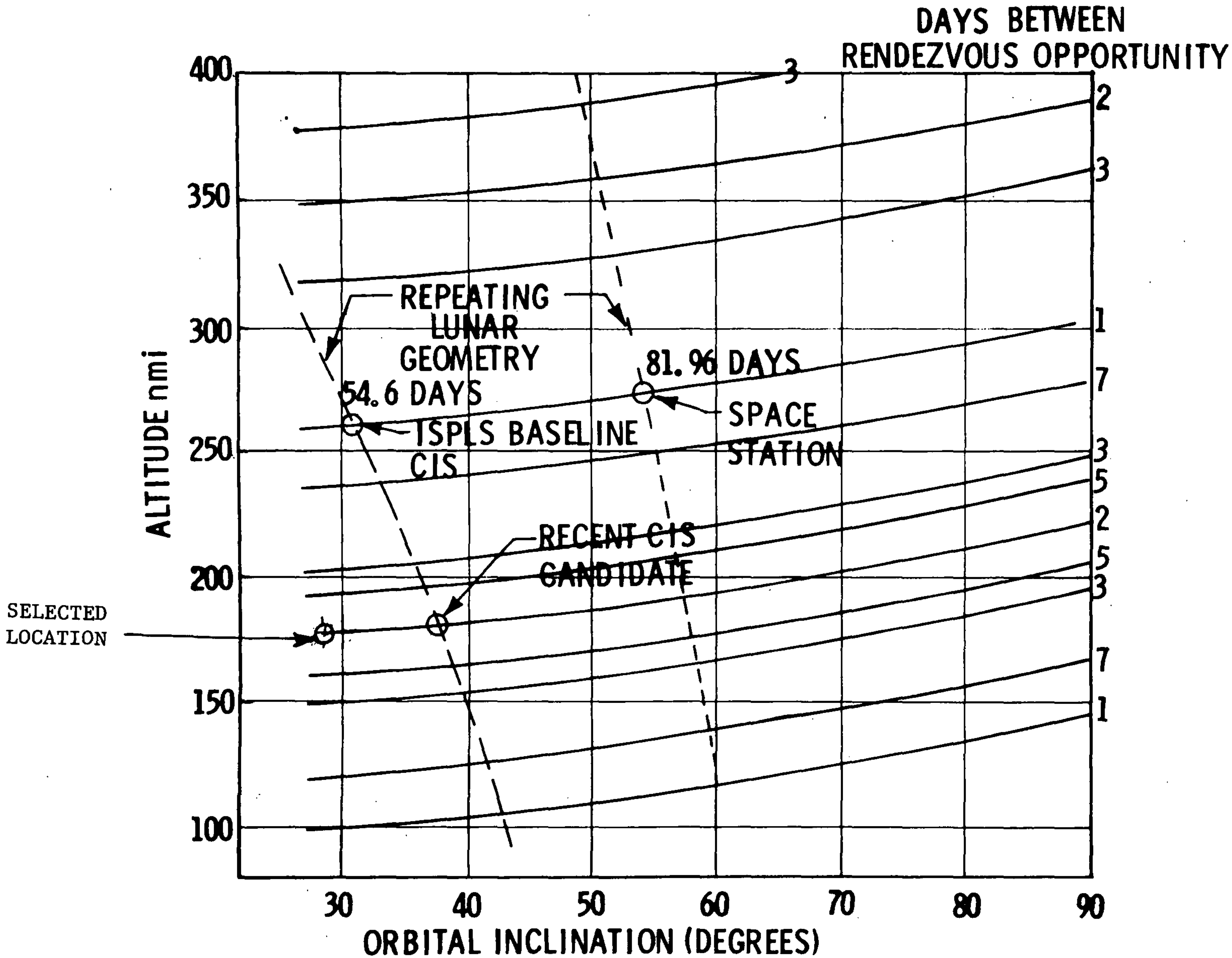


Figure 3.3-2 Rendezvous Compatibility Orbit Locations

The dashed lines in Figure 3.3-2 indicate orbital locations that are also synchronized with respect to the moon at intervals of 2 and 3 lunar months, 54.6 and 81.96 days, respectively.

Both of the CIS locations indicated in the figure are at altitudes and inclinations which are not only periodically synchronized with KSC launches, but are also synchronized with lunar positions so that uniform trajectories with uniform lighting conditions at the moon can be utilized periodically for the flights to and from the moon.

At the 28.5-degree inclination, three altitudes offer initial interest for the location of the space-based tug, 100 n mi and 260 n mi with one day rendezvous compatibility and 180 n mi with 2-day compatibility. Figure 3.3-3, which presents propellant requirements to overcome orbital drag, indicates that propellant requirements for orbital maintenance are negligible for orbits above about 150 n mi. The orbital maintenance requirements for the tug alone lie within the band of uncertainty indicated on the figure. The 100 n mi altitude is rejected not only because of the propellant consumption but because of the short orbital life which is estimated at one or two days for the tug at this altitude. In the event of a malfunction, this interval is considered too short to provide time for evaluation and decision and potential corrective action, if any.

The payload capability of the shuttle versus altitude is discussed in Section 2, Propellant Delivery Modes. It is shown in Figure 2.3-1 that the shuttle can deliver a full payload of 65,000 pounds to 180 nautical miles in the propellant delivery mission.

The results of the delivery mode study also indicate that the use of the shuttle vehicle is the most economical mode of propellant delivery to its altitude range of 180 n mi and that the costs are essentially constant from 100 n mi to 180 n mi delivery altitude. Once the shuttle is in orbit, its on-board hypergolic orbital maneuvering system propellant is used to take the full payload to 180 n mi. It is also indicated that it is not economical to use the tug to transfer propellant from a lower altitude to a higher altitude within the range capability of the shuttle.

The basing of the tug at an altitude as high as possible results in a nominal reduction in propellant requirements for the tug missions. Figure 3.3-4 indicates the savings in tug propellant requirement for a geosynchronous payload placement mission which would result from starting the mission at altitudes above 100 n mi as opposed to starting at 100 n mi. The figures indicates that about 1800 pounds of propellant would be saved by starting the mission from 180 n mi as opposed to 100 n mi. In other words, if 60,000 pounds of propellant is required for the mission starting at 100 n mi, 58,200 pounds would be required for the mission starting at 180 n mi.

A parking orbit altitude of 180 n mi was selected on the basis of the above considerations.

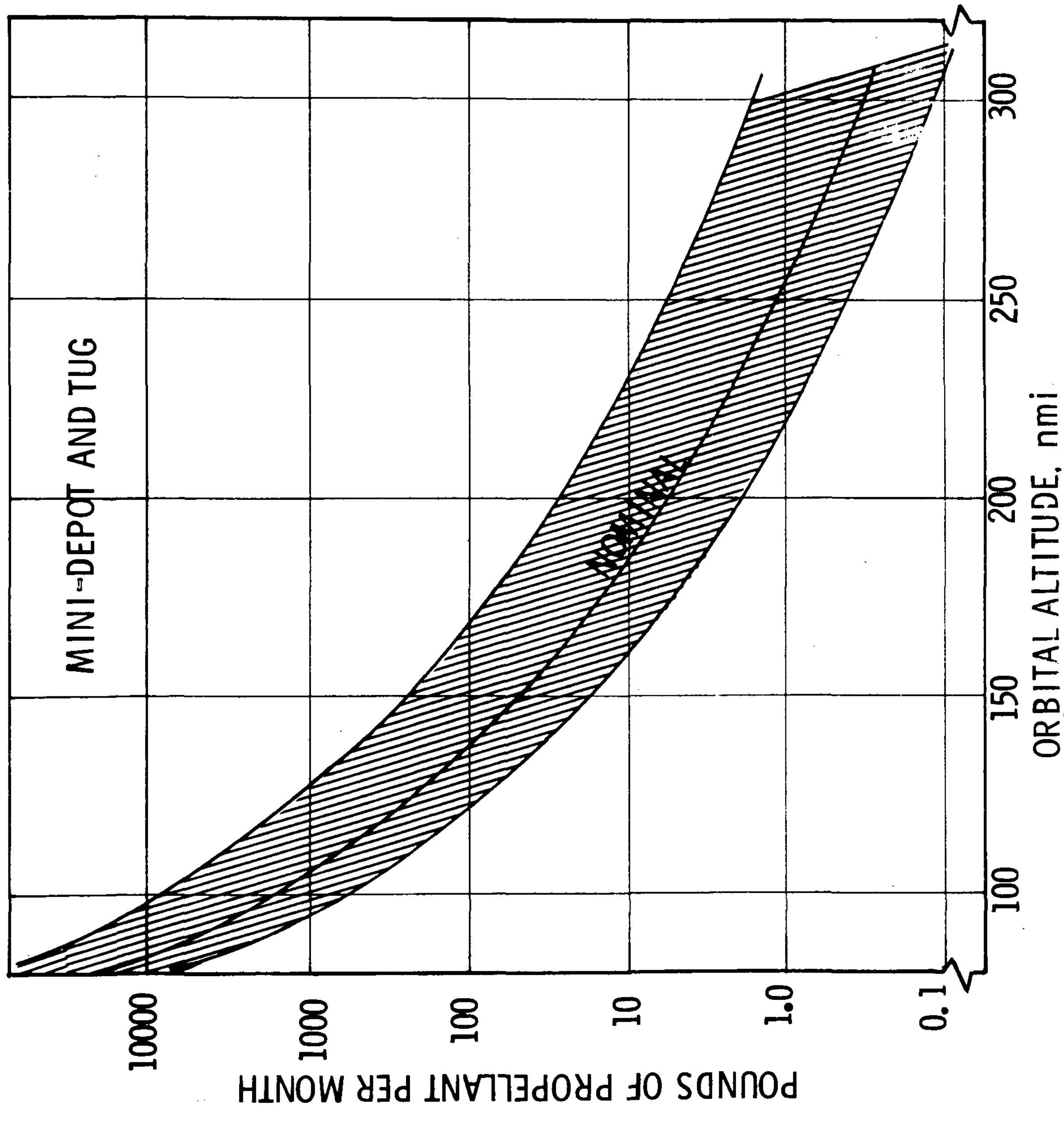


Figure 3.3-3 Propellant Requirements for Orbital Maintenance

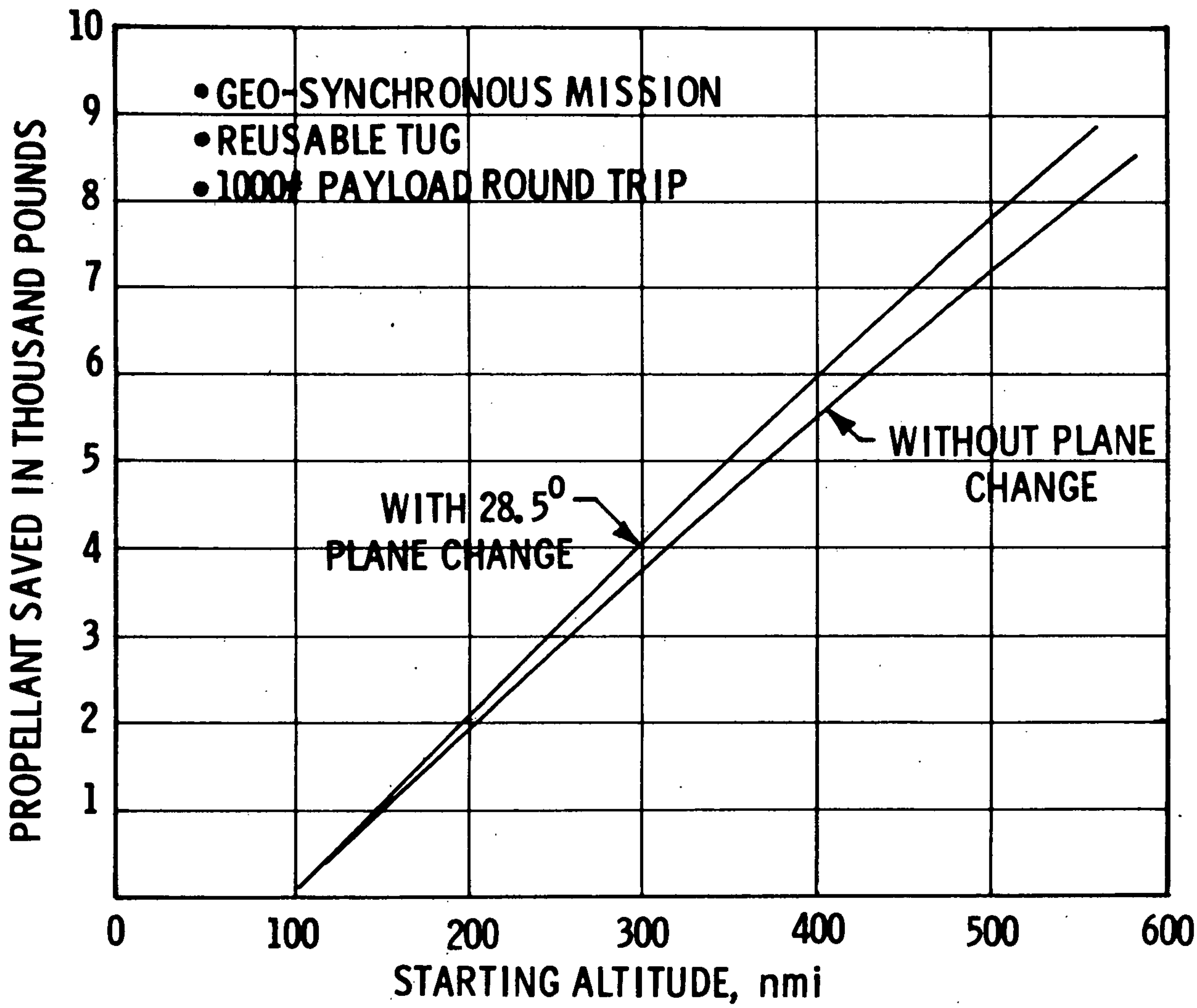


Figure 3.3-4 Savings in Tug Mission Propellant for Starting Altitudes Over 100 nmi

4.0 CIS/RNS DEPOT REQUIREMENTS

4.1 INTRODUCTION

The objective of this study is to determine if there is a need for an in-space propellant depot to support the space-based CIS or the RNS vehicles in their mission operations. If such a depot were required, it could also be available for use to support the space-based tug in its scientific payload placement missions. A depot to support the CIS or RNS would be a relatively large depot compared with a mini-depot which was considered later in the study to support the space-based tug missions. The mini-depot tank holds 60,000 pounds of propellant.

Space-based propellant depot concepts to support the CIS and RNS were defined in Reference 4.0-1, Orbital Propellant Feasibility Study, conducted just prior to this study. The CIS supportive depot had a capacity of 1,300,000 pounds of propellant and the estimated ten year depot program cost was about one billion dollars.

The CIS/RNS vehicles employed in the present study are described in Section 3 of Volume II of this report. The baseline CIS has a capacity of about one million pounds of propellant. It employs 990,000 pounds of propellants in its lunar mission while carrying a payload of 175,000 pounds to the moon and operating in Mode 1. The RNS would employ 300,000 pounds of propellant on the same mission.

The CIS/RNS depot requirement analysis presented here was conducted in the early phases of this study and is based on Mode 1 operation and propellant consumption which was the baseline at that time. Mode 1 employs a conventional earth to moon and return trajectory. The Mode 2 trajectory differs from Mode 1 in that the tug retrieves the CIS/RNS from an elliptical orbit about the earth on the return trip from the moon. Operation in Mode 2 would not significantly change the conclusions of the present analysis. A description and comparison of Mode 1 and Mode 2 propellant requirements and logistics program costs is contained in Section 7 of Volume II.

4.2 SUMMARY OF RESULTS

The results of the analysis indicate no requirement for a storage depot to support the CIS or RNS vehicles for the flight frequency established in this study. Although the study was conducted with respect to the CIS vehicle, the application of its logic and findings to the RNS vehicle indicate the same overall conclusions. There would be less need for a depot for RNS than for CIS because the RNS uses less propellant and requires fewer shuttle flights for support.

The analysis also indicates that some form of supplemental in-space propellant storage may be economically advantageous if the CIS flight rates were increased above the two lunar flights per year contained in the model used in this study. The economy of in-space storage results from the reduced number of dedicated shuttles to deliver the propellant since an optimum number of shuttles and flight schedules can be established, which is independent of the CIS availability for fueling when separate in-space storage exists. The two alternatives considered were the depot concept and the use of a second CIS. The two CIS vehicles in orbit would be used alternately so that as one CIS is performing the lunar mission, the other would be fueling in preparation for the next lunar mission, and hence, would be equivalent to in-space storage. The use of two CIS vehicles was found to be more economical than a depot, and furthermore, it could mean greater mission flexibility.

4.3 ANALYSIS OF THE REQUIREMENT FOR A DEPOT

Because the CIS can store and accumulate its own propellant in space, which are the functions of a depot, the need for a depot depends on the rate of CIS flights and its availability or non-availability for accumulating its required propellant. Approximately 19 shuttle flights are required to provide the propellant for each lunar flight. With an allowance for a two week turn-around time for one shuttle, 38 weeks would be required to fill the CIS. At high flight rates, the CIS would not be available for 38 weeks between flights to be refueled. The time required for refueling the CIS can be reduced by the use of additional shuttles. The number of shuttles required to fuel the CIS increases as the CIS flight rate increases and the time available between flights for fueling decreases.

If a depot were provided, it would be continually available in earth orbit to receive propellant, thus reducing the required number of shuttles. It could transfer its propellant to the CIS in a few days when the CIS returned from its trip to the moon. A tradeoff thus exists between the purchase and use of additional shuttles to support the CIS program and the cost of providing separate storage.

4.3.1 Relationship Between CIS/RNS Flights and Shuttle Flights

The number of shuttle flights required to support the CIS and RNS in each flight is shown in Figure 4.3-1. The curves are based on an overall propellant transfer loss of eight percent, with boiloff rates of 14 pounds per hour for the CIS and six pounds per hour for the RNS. The allowance for boiloff is dependent on the number of flights per year and decreases with increasing flights per year. An allowance of 5,100 pounds is made for the logistic tank weight so that after an eight percent transfer loss, one shuttle flight with a payload capability of 65,000 pounds delivers a net of 55,100 pounds of propellant. The 990,000 pounds of propellant plus an allowance of 61,000 pounds for boiloff at two flights per year used in the

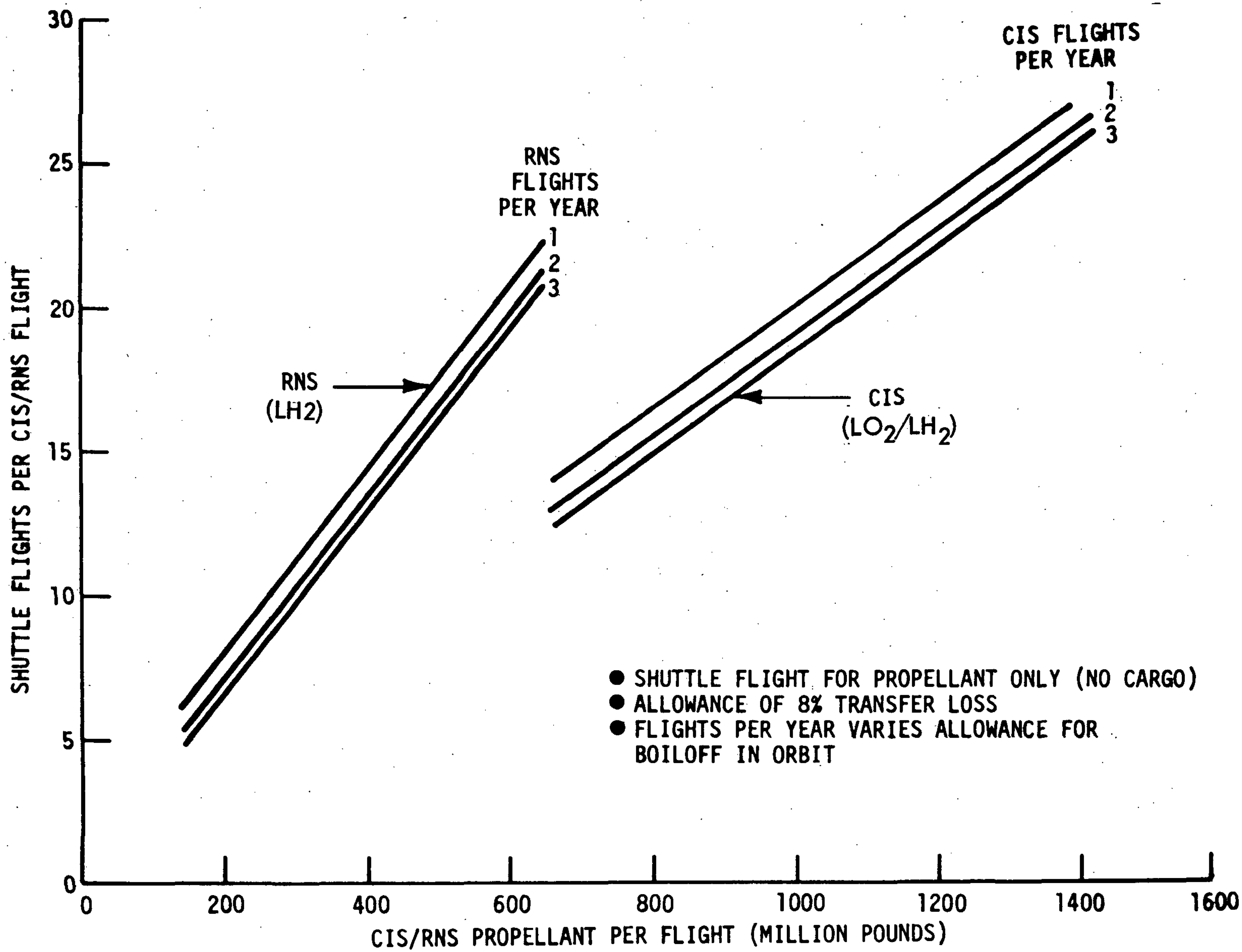


Figure 4.3-1 Shuttle Flights for CIS/RNS Propellant

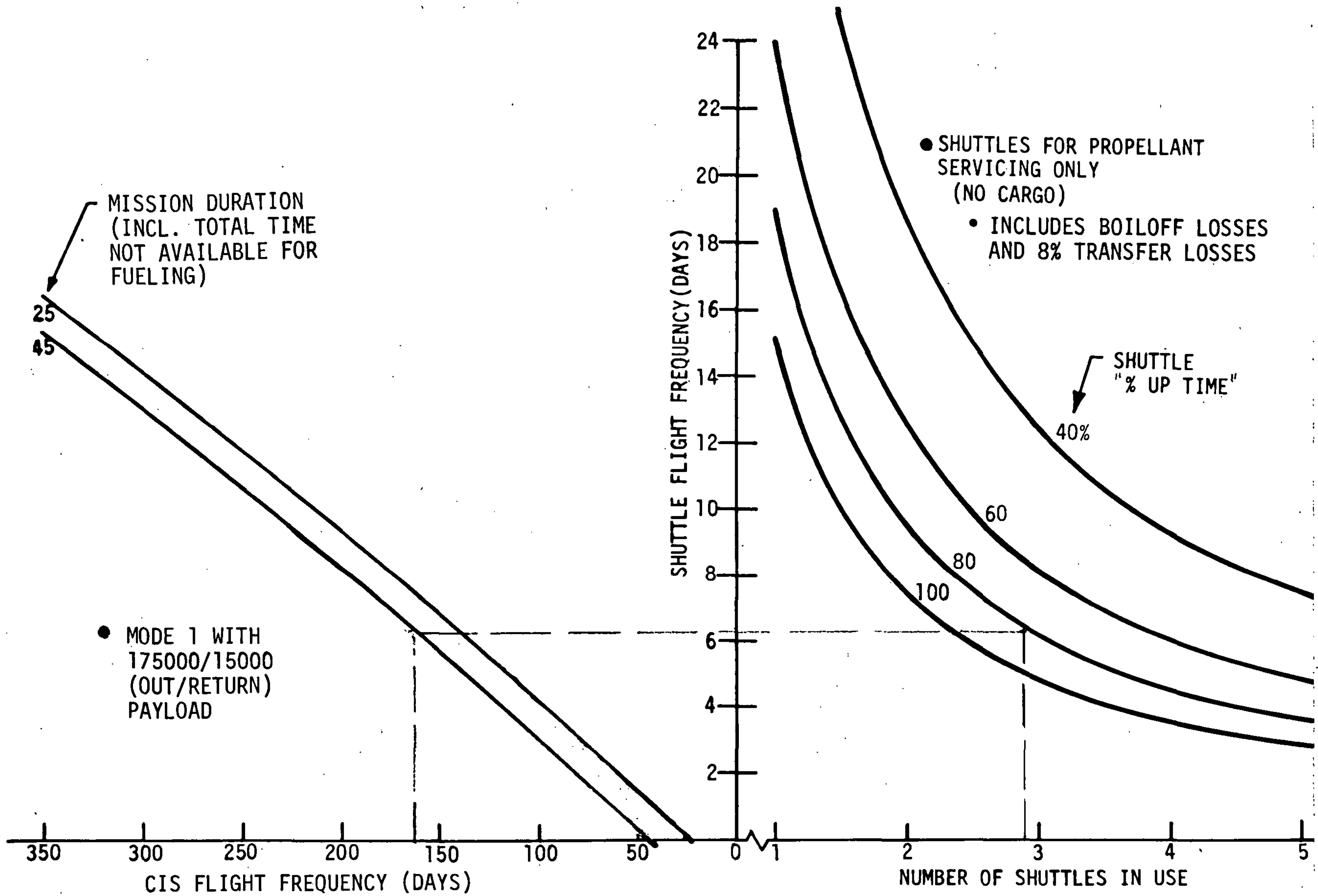


Figure 4.3-2 CIS and Shuttle Flight Relationships

analysis requires 19 shuttle flights. Figure 4.3-2 shows the relationship of shuttle flight frequency and number of shuttles in use for the fueling operation of the CIS as a function of its flight interval.

Shuttle "Up Time" refers to the percent of time that a shuttle is in use or ready for use. It is comparable to aircraft "In Commission" in Air Force terminology. Eighty percent up time means that, on the average, 80 percent of the shuttles are in actual flight or in their two week turn-around cycle between flights. Twenty percent of the shuttles are in maintenance and hence "down."

In the CIS study, a flight every six lunar months or 164 days was considered the baseline flight frequency. In addition, a 25 day flight plus approximately 20 days for maintenance and preparation in earth orbit when the CIS could not be fueled was established. A 164 day flight frequency and 45 day total "mission duration" leaves 119 days available for fueling. With 19 shuttle flights required, an average of one shuttle flight is needed every 6.3 days. At 80 percent shuttle up time, 2.9 shuttles which have to be interpreted as three shuttles, would be required full time to fuel the CIS in this four month interval. The additional shuttle flights required to transport the CIS payload to the CIS, as identified in the CIS study, increase the quantity of shuttles required by one so that a total of four shuttles would be required full time and dedicated to the CIS program in this operation mode.

4.3.2 Savings in Number of Shuttles by Depot or Second CIS

The number of shuttles required full time and dedicated to the CIS for propellant servicing during the servicing intervals is taken from Figure 4.3-2 and plotted in Figure 4.3-3 as a function of CIS flight frequency and mission duration. The longer mission durations (including maintenance and preparation time) require additional shuttles. When the mission duration approaches the flight frequency, no time remains for fueling the CIS (even if 19 shuttles were available at once) so the mission could not be flown. Also plotted on Figure 4.3-3 is the number of shuttles dedicated to the propellant servicing operation if a depot were available to provide separate accumulation and storage. This curve is based on the depot being available continuously to receive propellant except for a five day period which it is assumed necessary each time propellant is transferred to the CIS. The saving in the number of dedicated shuttles required that results from the existence of a depot can be determined by moving from the "without separate storage" curves vertically down to the "with depot" curve on Figure 4.3-3. For example, at an 80 day CIS flight frequency and 45 day mission duration, nine shuttles would be required. This could be reduced to four if a depot were available, a savings of five shuttles. It is noted that the existence of a second CIS would have essentially the same effect as the existence of a depot in that it could act as the propellant receiver and storage device.

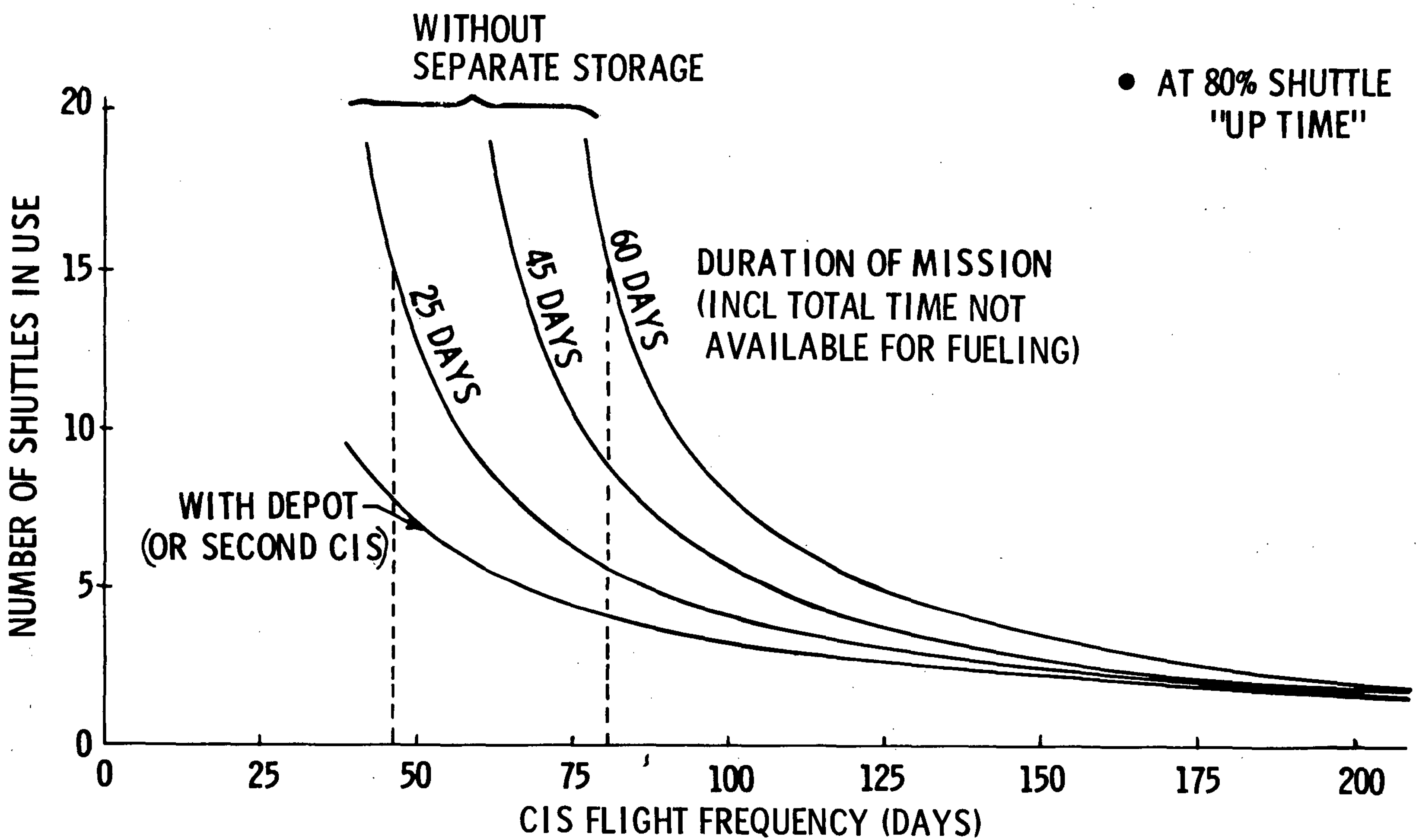


Figure 4.3-3 Shuttles Required to Support CIS Flights

The savings in the number of dedicated shuttles, achieved by the existence of a depot or a second space-based CIS, has been taken from Figure 4.3-3, priced in terms of dollar purchase price for these shuttles and plotted on Figure 4.3-4 as a function of the CIS flight frequency. Also approximate cost range anticipated for a depot program over a five year interval and the cost of purchasing one CIS, placing it in space and maintaining it for five years is indicated on Figure 4.3-4. The costs used in Figure 4.3-4 and their source are discussed below.

4.3.3 Conclusion

The data presented in Figure 4.3-4 indicates for a single CIS at a 45-day total mission duration, as defined above, consideration should be given to a depot when the CIS number of flights reached approximately three flights per year. In other words, the shuttles required might be more costly than a depot for more frequent flights. The use of a second CIS, however, should be considered if the number of CIS flights increases much above the two flights per year contemplated in the model (SPLS model) used during this study. The overall conclusion is that for operation with a single CIS, the cost of a CIS supportive depot program is greater than the indicated savings in shuttle costs unless CIS flight rates increase considerably above the two flights per year currently contemplated. In any case, the use of a second CIS would be cheaper than such a depot program. A second CIS would achieve the same advantage and would undoubtedly provide other advantages such as flexibility in program operations and a lunar rescue capability in case one CIS failed. As a consequence, no need has been indicated for a CIS supportive depot.

4.3.4 Costs Used in the Analysis

The above conclusions related to specific flights are, of course, dependent on the specific costs used in Figure 4.3-4 and would vary somewhat for changes in these costs and the manner in which they are used. However, the overall conclusions indicating no need for a depot and the benefit of a second CIS as flight rates increase are inescapable.

The cost savings in Figure 4.3-4 is based on the purchase cost of a shuttle vehicle with booster support at $400,000,000. This number may, of course, differ as shuttle program costs become firm and would affect the specific numbers quoted in the conclusions above. A lower number would indicate even less need for a depot than is indicated above. A portion of the production costs of the initial shuttles in the current shuttle program plan are included in the development costs because test vehicles are converted to operational vehicles in that plan. As a consequence, the purchase of additional shuttles would be more costly than the initial shuttle program production costs indicate. This fact was given consideration in the selection of the $400,000,000 figure for this analysis. Only the purchase cost

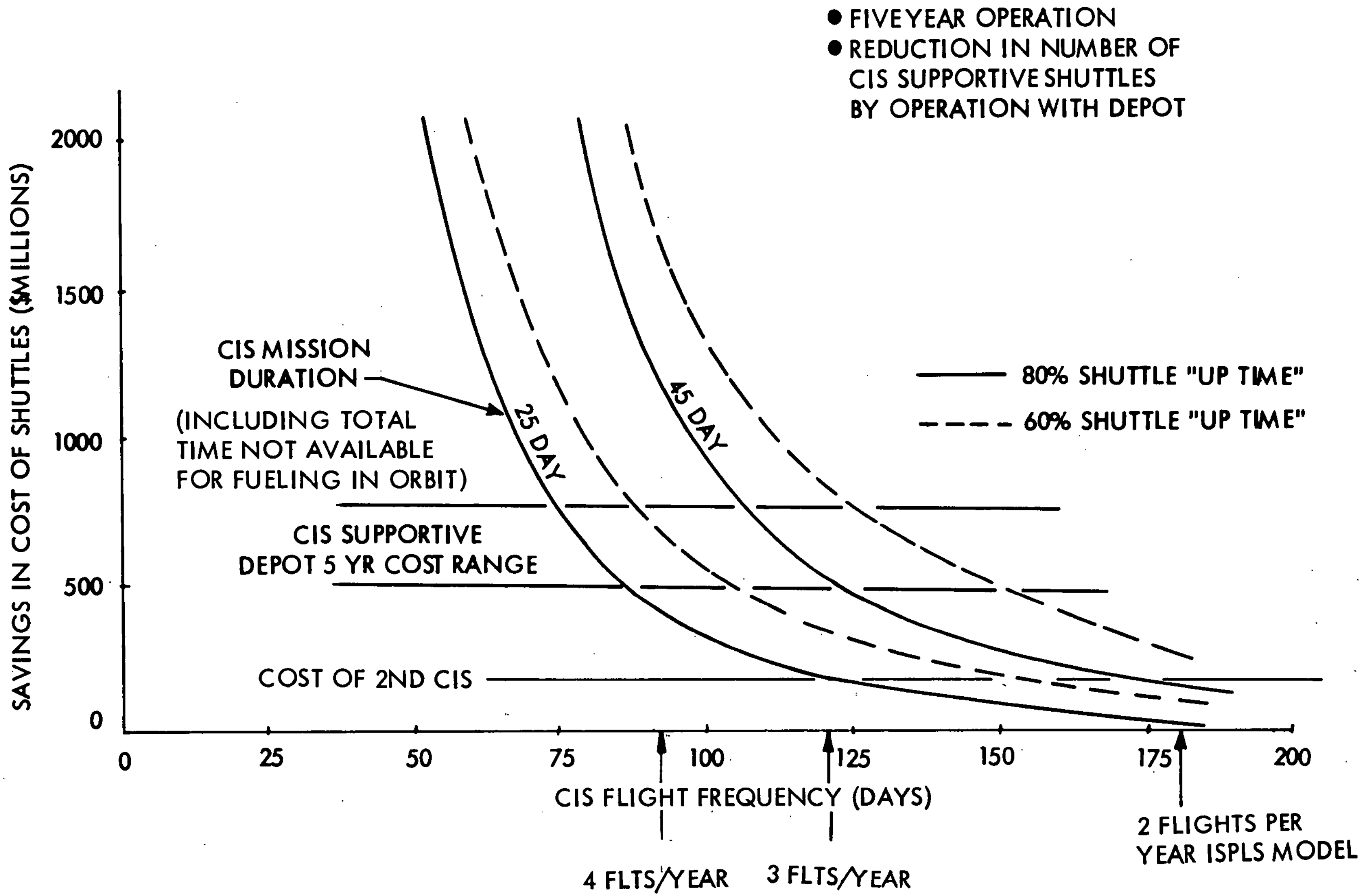


Figure 4.3-4 Savings in Shuttle Acquisition Costs by CIS Operation With Depot or Second CIS

of shuttles is considered because the operational costs of the shuttle are not affected by the existence or non-existence of the depot or the second CIS.

Five years is taken as the program interval because in all the cases considered, none of the shuttles would have to be replaced (at a 100 mission life each) before the five years was up. Five shuttles required for a 70-day CIS flight frequency, Figure 4.3-3, would fly approximately 500 missions in the five years. For other cases, the purchased shuttles would have differing portions of their lives remaining at the end of the five years. Nevertheless, it seems reasonable to equate the cost of the shuttles which would have to be purchased in a five-year interval with the cost of a depot program. In the case of the CIS, the costs are taken from Volume II, Section 7 of this study and include $110,000,000 for purchase of the second CIS and $88,000,000 for initially placing it in space and maintaining it for 10 flights. With a 10-flight life, two CIS can fly 20 flights or four flights per year in five years. At flight rates above four per year (every 91 days), consideration would need to be given to additional CIS costs. The depot program costs were taken from the Reference 4.0-1 study wherein costs ranged from one billion to 1.2 billion dollars for a ten-year, two-depot program including development costs. These costs have been approximately halved for the five-year program indicated in Figure 4.3-4. If full depot development costs were prorated to the five-year, instead of the 10-year program, depot costs would be greater than indicated and there would be less need for the depot than indicated.

4.4 REFERENCES

4.0-1 Orbital Propellant Storage System Feasibility Study, Report SD 70-554, dated March 31, 1971.

5.0 MINI-DEPOT AND LOGISTIC MODULE DEFINITION

5.1 INTRODUCTION

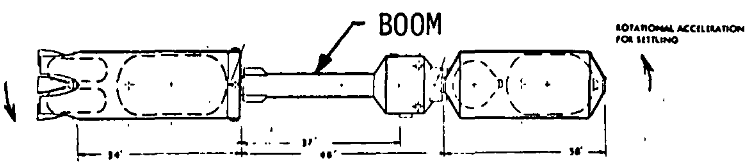
A mini-depot is an orbital propellant storage facility considerably smaller than those that would be able to refuel a CIS or RNS in one transfer operation. It was determined in Section 4 above that for the frequency of CIS or RNS flights in program levels D & E, a large Orbital Propellant Depot is not economically advantageous over accumulating propellant in the CIS or RNS. Were such a facility used for CIS or RNS, it would also supply propellant to a space-based tug. In the absence of such a facility, the mini-depot was conceived to provide orbital storage of propellant for the tug so that full utilization could be made of the shuttle orbiter delivery capability. With orbital storage, each shuttle flight would carry the scientific payload and the maximum propellant payload to fully utilize the shuttle 65,000 pound capability. Should that propellant quantity be more than is needed by the tug for a placement mission, the surplus could be stored in orbit in the mini-depot.

Four concepts of mini-depots are shown in Figure 5.1-1. They differ in the propellant transfer modes used. The first two receive propellant by a modular transfer of the logistics tank to the depot and the second two have fluid flow of propellant from the logistics tank to the permanent tanks of the mini-depot. Mini-depots 1a and 1b store propellant in the logistics tank delivered by the shuttle and have an equipment module providing orbital capability, power and systems for fluid transfer of propellant to the tug. The a and b versions differ in the propellant settling mode. The rotational depot uses less propellant to accomplish settling, but requires that the equipment module assume the configuration of a counter-weighted boom to prevent the combined vehicle center of gravity (cg) from falling within any of the tanks during transfer. The linear depot has a smaller, lighter equipment module that can share the initial launch in the shuttle with the logistic tank, but the settling mode uses much more propellant because the thrust must be applied continually during transfer.

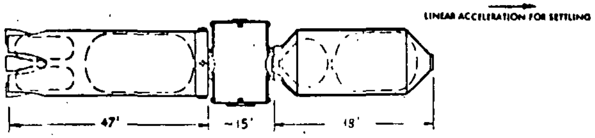
Mini-depot concepts 2a and 2b uses a similar logistic tank for propellant delivery but the storage tanks are a permanent part of the depot. This approach offers the lowest potential boiloff rate in tank design. In concepts 1a and 1b where the tanks used to deliver propellant also store it, their weight must be kept to a minimum to allow delivery of the maximum propellant. Because the mini-depot permanent propellant tanks (of 2a and 2b) are deployed once (empty) and remain in orbit, additional weight in thermal control systems and insulation can be included. Also, because the module can be launched empty in the shuttle, the tank supports can be minimized to reduce heat conduction paths between exposed structure and the tanks. The boiloff advantage is traded off against the propellant losses due to the fluid transfer of propellant to the depot.

The mini-depots with permanent tanks, 2a and 2b of Figure 5.1-1, were eliminated from further consideration at an early date in the study. It was determined that propellant boiloff losses were a small percentage of the total propellant requirement. Differences in losses would not be a driving cost factor. Also the permanent tankage mini-depots had the disadvantage of additional development, acquisition and launch costs. The propellant logistics tanks which delivered propellant to the mini-depot would be required in any case. Definitions of the operation and capabilities of the modular mini-depots were then used

1A MODULAR, ROTATIONAL MINI-DEPOT



1B MODULAR, LINEAR MINI-DEPOT

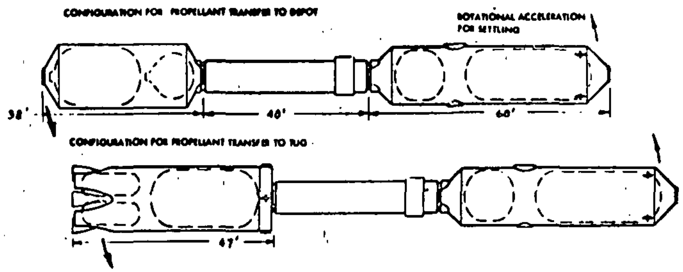


ELEMENTS: EQUIP MODULE & LOGISTICS/STORAGE TANK (1A)
FEATURES: ROTATIONAL SETTLING REDUCES PROP USE
EQUIP MOD REQS SEPARATE LAUNCH

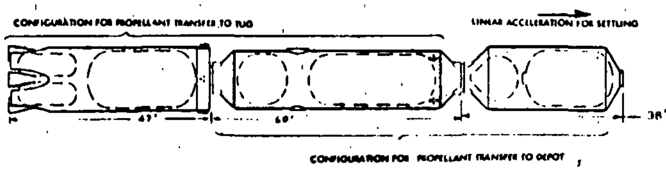
OPERATION: (BOTH)
ORBITER DELIVERS PROPELLANT TANK
TANK BECOMES PART OF DEPOT UNTIL EMPTY
EQUIP MOD PROVIDES TRANS & SETTLING CAPABILITY
PROP TRANS FLUID TO TUG FROM L/S TANK
ORBITER RETURNS TANK WHEN EMPTY (NEXT TRIP)

ELEMENTS: EQUIP MODULE & LOGISTICS/STORAGE TANK (1B)
FEATURES: EQUIP MOD & TANK CAN SHARE LAUNCH
SHORTER TRANS LINES; LESS BOILOFF

2A PERMANENT TANKAGE, ROTATIONAL MINI-DEPOT



2B PERMANENT TANKAGE, LINEAR MINI-DEPOT



ELEMENTS: DEPOT MODULE, BOOM & LOGISTICS TANK (2A)
FEATURES: ROTATIONAL SETTLING REDUCES PROP USE
PERMANENT TANK REDUCES BOILOFF
BOOM REQS SEPARATE LAUNCH

OPERATION: (BOTH)
ORBITER DELIVERS PROPELLANT TANK
TANK IS ATTACHED (TUG DETACHED) & PROP IS TRANSFERRED (FLUID) TO DEPOT
ORBITER RETURNS EMPTY TANK TO GROUND
TUG DOCKS RECEIVES PROP (FLUID TRANS) FROM DEPOT TANKS
DEPOT PROVIDES TRANS & SETTLING CAPABILITY

ELEMENTS: DEPOT MODULE & LOGISTICS TANK (2B)
FEATURES: SINGLE LAUNCH FOR DEPOT
SHORTER TRANS LINES; LESS BOILOFF

Figure 5.1-1 Mini-Depot Concepts Comparison in establishing the cost effectiveness analysis of the propellant logistics concepts. These two mini-depot definitions and the transfer capability tank definition used in costing the logistics concepts are presented in the following paragraphs.

5.2 MODULAR, ROTATIONAL MINI-DEPOT

The modular, rotational mini-depot is shown in Figure 5.2-1 in its operational configuration with the tug attached. The mini-depot consists of an equipment module and a logistics/storage tank module. The equipment module contains depot operation and transfer systems, including power and attitude control, and remains in orbit independently. For rotational propellant settling the combined tug and depot c.g., which is the center of rotation, will shift as the mass of propellant is transferred from the storage tank to the tug. For proper liquid interface control, the center of rotation must not fall within any tank involved in the transfer. The equipment module acts as a counterweighted boom to limit c.g. excursions. The weight of the module and offset c.g. location enable the module to prevent c.g. excursions into the tanks during transfer and at the same time keep the equipment module size and weight within the shuttle orbiter cargo capability for launch of the module. Propellant is delivered in the tank module which docks to the equipment module, then becomes part of the mini-depot until it is depleted and replaced with another tank from the next shuttle flight. There is the potential of leaving more than one tank module at the depot and thus increasing the storage capacity. This would increase the opportunities for full utilization of shuttle capability in delivering propellant. The transfer systems could include the capability to supply propellant from any tandem tanks but the c.g. excursion considerations of the rotational settling mode (as outlined on Figure 5.2-1) be a complicating factor.

5.2.1 Rotational Mini-Depot Equipment Module

Figure 5.2-2 gives the conceptual definition of the equipment module for the rotational, modular mini-depot. This module contains all the equipment required for orbital propellant transfer and to support independent operation in orbit. Its fuel cell and attitude control consumables are replenished by the frequent visits of the logistics tanks. The configuration of the module is determined (for the rotational propellant settling mode) by the necessity to control the location of the combined mini-depot and user c.g. to prevent rotation about a point that falls within the tanks involved in the transfer. The length, weight and location of the equipment module c.g. could be altered somewhat, but the combination chosen gives the required c.g. control and is compatible with shuttle launch. The high weight of the module will allow use of inexpensive boiler-plate type structure. The equipment module will be launched by the shuttle orbiter and remain in orbit for its nominal six-year life. Minimum on-board maintenance will be provided for, with return to the ground for refurbishment for any major unscheduled maintenance required.

5.2.2 Modular Mini-Depot Logistic and Storage Module

The logistic and storage tank module to be used with the rotational mini-depot, shown in Figure 5.2-3, is typical of the logistic modules required by both mini-depots. Its function is to bring propellant to the depot, to remain attached to the equipment module until it is empty, then to be returned to earth in the shuttle for recycle. Major design criteria are low residuals,

FOR ROTATIONAL PROPELLANT SETTLING THE COMBINED USER & DEPOT CG, WHICH IS THE CENTER OF ROTATION, WILL SHIFT AS THE MASS OF PROPELLANT IS TRANSFERRED FROM THE STORAGE TANK TO THE USER. FOR PROPER LIQUID INTERFACE CONTROL THE CENTER OF ROTATION MUST NOT FALL WITHIN ANY TANK INVOLVED IN THE TRANSFER. THE EQUIPMENT MODULE ACTS AS A COUNTER WEIGHTED BOOM TO LIMIT CG EXCURSION

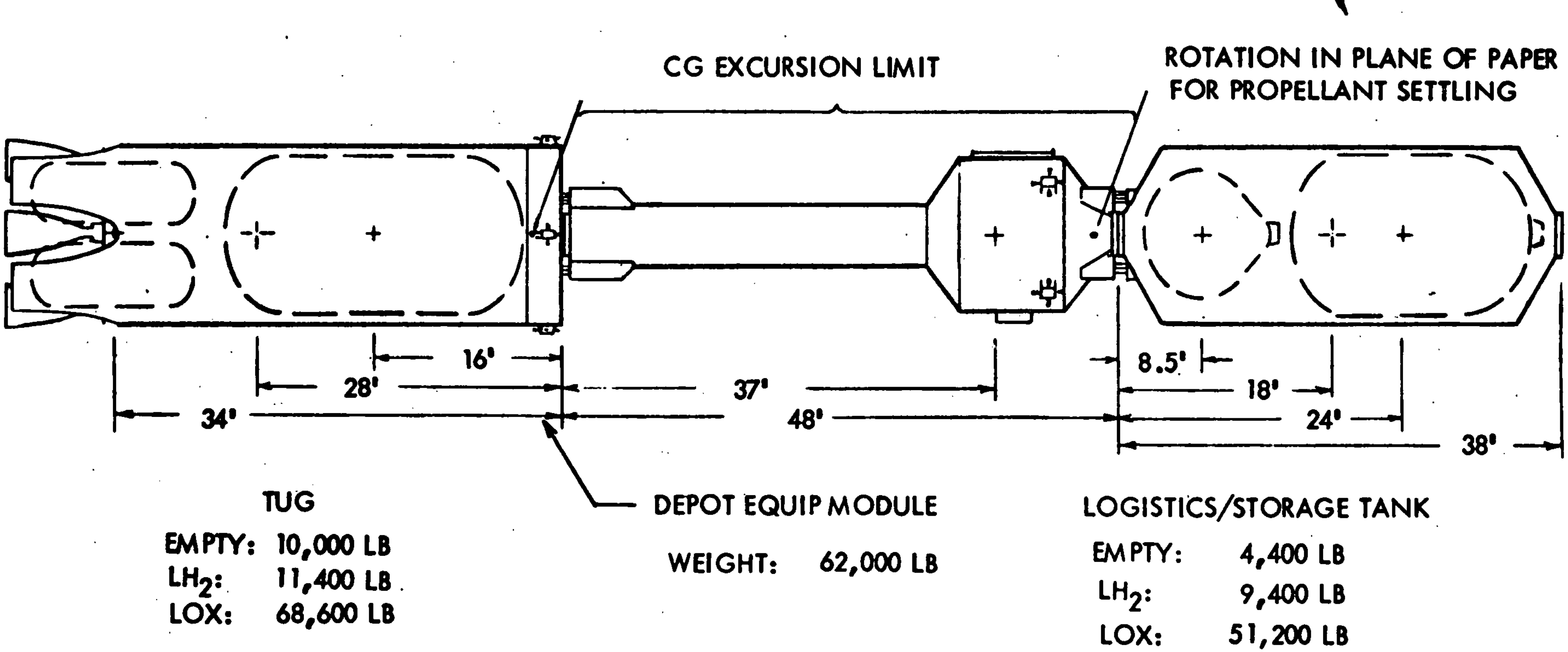
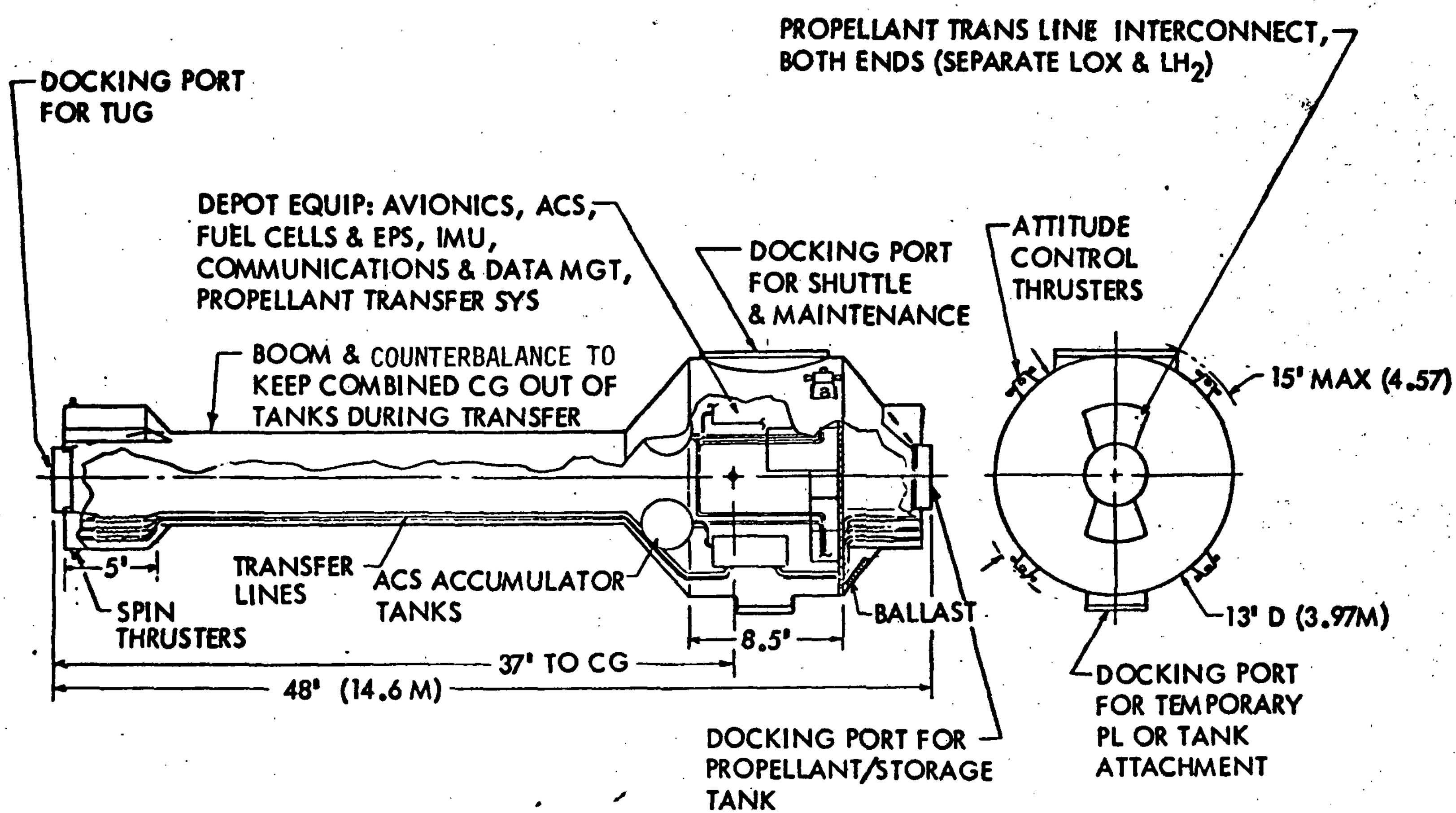


Figure 5.2-1 Modular, Rotational Mini-Depot



- MADEWITH BOILERPLATE TYPE STRUCTURE TO BUILD WEIGHT TO 62,000 LB (28,100 KG) FOR TRANSFER CG CONTROL

Figure 5.2-2 Rotational/Mini-Depot Equipment Module

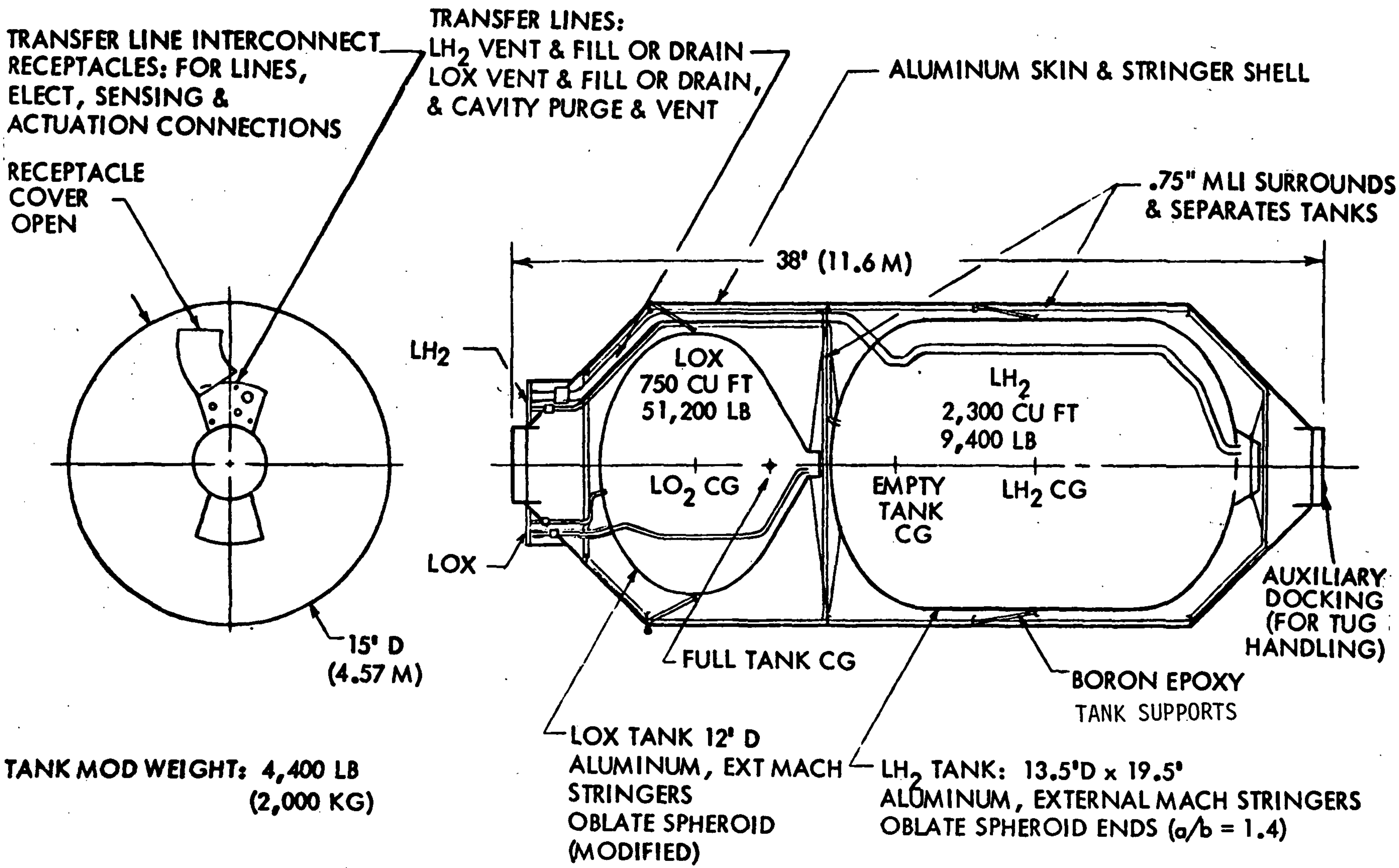


Figure 5.2-3 Modular Logistics and Storage Tank

low boil-off, and low inert weight. The tank module has no transfer equipment or orbital capability and must be dependent upon and under (attached) control of the shuttle, tug or equipment module at all times. The LO_2 and LH_2 tanks are sized for maximum utilization of the 65,000-pound shuttle cargo capability. The indicated tank weight includes allowances for cargo bay installation of umbilical and entry pressurization systems which are not physically a part of the tank module.

Preliminary definition of the tank has included shuttle interface considerations of cg location, cargo umbilicals and payload sharing. For the linear settling (modular mini-depot) option, the logistics/storage tank required is considered identical to this for evaluation purposes. The only discernible difference being that the tanks and lines would be oriented for propellant settling toward the propellant transfer docking port. The logistic tank modules for the permanent tankage depots are also considered similar enough to this one for present evaluation. In more detailed definition phases differences due to less emphasis on storage requirements (since the tank does not remain in orbit for an appreciable length of time) might reduce insulation or systems and allow a very slight increase in propellant capacity.

5.3 MODULAR, LINEAR MINI-DEPOT

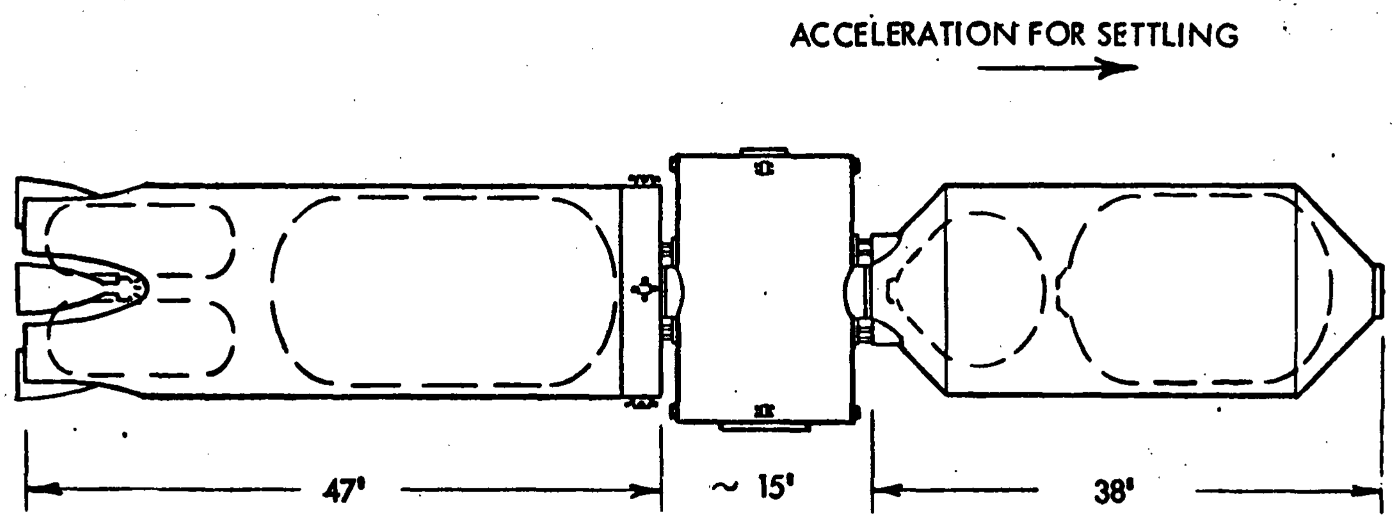
The modular, linear mini-depot is shown in Figure 5.3-1 in its operational configuration with the tug attached. The operational concept is much the same as for the rotational mini-depot, with an equipment module providing transfer and orbit-keeping capability and a docked logistics tank providing propellant storage and being replaced when empty (modular propellant transfer to the depot). Linear acceleration for propellant settling eliminates the cg excursion problem and allows for a lighter, more compact equipment module. The module length chosen provides clearance for docking to the end ports (as would be used for temporary placement during exchange of logistics tanks or for maintenance) and gives a satisfactory location for attitude control and settling thrusters. Also, the length is compatible with launch of both the logistics/storage tank and the equipment module in a single shuttle flight.

The logistics/storage tank for this mini-depot is quite similar to the rotational tank module. Structure, equipment, function, size, weight and cost are considered the same, though the transfer lines and tank ends will be reversed for settling of propellants toward the propellant transfer docking port.

5.3.1 Linear Mini-Depot Equipment Module

Figure 5.3-2 defines the equipment module for the modular, linear mini-depot. It has essentially the same complement of equipment and functions as the rotational equipment module and much of the same design rationale applies. Systems allow the module to function independently in orbit with all monitoring, communication, rendezvous and docking, and attitude control provisions in addition to the checkout and propellant transfer systems compatible with the tug. Fuel cells provide the power source and accumulation tanks, filled during transfer, hold the propellant for fuel cells and attitude control. The side docking ports have identical line interfaces and are

THE LOGISTICS/STORAGE TANK USED WITH THIS LINEAR MINI-DEPOT IS ESSENTIALLY THE SAME (CAPACITY, WEIGHT, COST) AS THE TANK FOR THE ROTATIONAL MINI-DEPOT EXCEPT SETTLING IS TOWARD TRANSFER PORT



TUG EMPTY:

EMPTY	10,000 LB
LH_2	11,400 LB
LOX	68,600 LB

DEPOT EQUIP MODULE WEIGHT: 5,700 LB

LOGISTICS/STORAGE TANK

EMPTY	4,400 LB
LH_2	9,400 LB
LOX	51,200 LB

Figure 5.3-1 Modular, Linear Mini-Depot

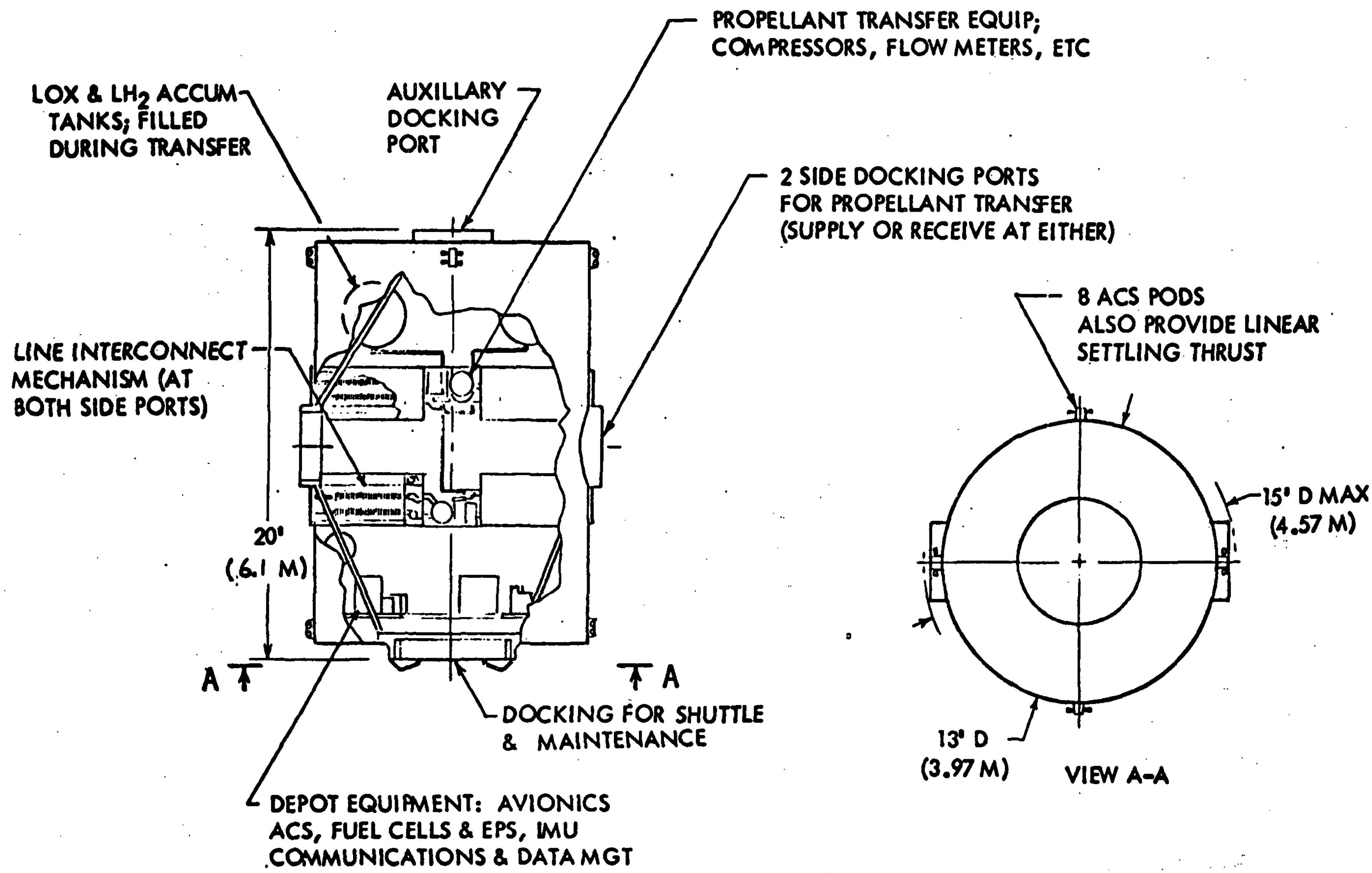


Figure 5.3-2 Linear Mini-Depot Equipment Module

interchangeable for docking with the logistics/storage tank or the tug. The volume of the module far exceeds that required by the equipment. This will allow use of many "shelf" components and simplify fabrication and maintenance tasks. The module is launched with a six year life expectancy with provisions to be returned to the ground for interim maintenance. The boom/counterweight approach is not required; and to reduce linear acceleration propellants expenditure, the structure would be of light-weight spacecraft design.

5.3.2 Linear Mini-Depot Logistic and Storage Module

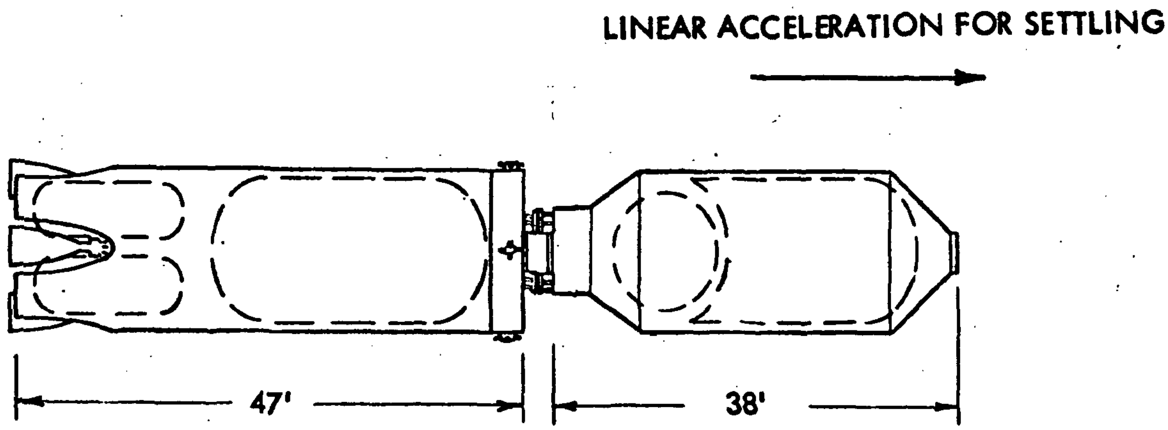
The same module as defined for the rotational mini-depot was used for the cost analysis of the logistic and storage module for the linear mini-depot.

5.4 DIRECT TRANSFER

The direct transfer operation given in Figure 5.4-1 is shown with the tug and a transfer capability logistics tank docked together. The tank module includes transfer line interconnects, compressors, other transfer related equipment and the settling acceleration thrusters. It would rely on the tug for electrical power, command, attitude control and data communication. The logistic tank module would be delivered by the shuttle, attached to the tug and the propellant transferred while the two modules are linearly accelerated for propellant settling. After transfer of mission propellant the shuttle would hold the tank module in orbit until the tug returned from the mission to receive the remaining propellant for stationkeeping.

5.4.1 Transfer Capability Logistic Module

The transfer capability logistics tank shown in Figure 5.4-2 would be used for direct transfer of propellant to the tug. Design criteria are basically the same as for other tanks, with the transfer systems added. Line interconnects engage the user receptacles and the necessary transfer compressors, lines, valves, actuation, flow metering and monitoring equipment are included with the tank. This equipment has brought the weight of this module to 400 pounds heavier than the logistic tank modules without transfer capability. For direct transfer the tank must be controlled by the tug during transfer and is dependent on the tug for power supply monitoring, actuation commands, and data management and transmittal. Since in this mode there is no depot module left in orbit, there would be no boom for cg control; and, therefore, settling is by linear acceleration. These requirements could impose some requirements on the tug beyond its basic mission design. The module length has been kept at 38 feet for a more equal comparison with the other logistic tanks and to allow the same payload sharing considerations to apply. The length required for installation of the transfer line interconnect mechanism has been acquired by using an inverted bulkhead on the LH_2 tank to allow nesting of the LO_2 tank. The inverted LH_2 bulkhead may aid in propellant settling and reduced LH_2 residuals. Tank weight again includes cargo bay umbilical and repressurization systems.



TUG	
EMPTY	10,000 LB
LH_2	11,400 LB
LOX	68,600 LB

TRANSFER CAPABILITY

LOGISTICS TANK	
EMPTY	4,800 LB
LH_2	9,300 LB
LOX	50,900 LB

Figure 5.4-1 Direct Transfer, Linear Acceleration

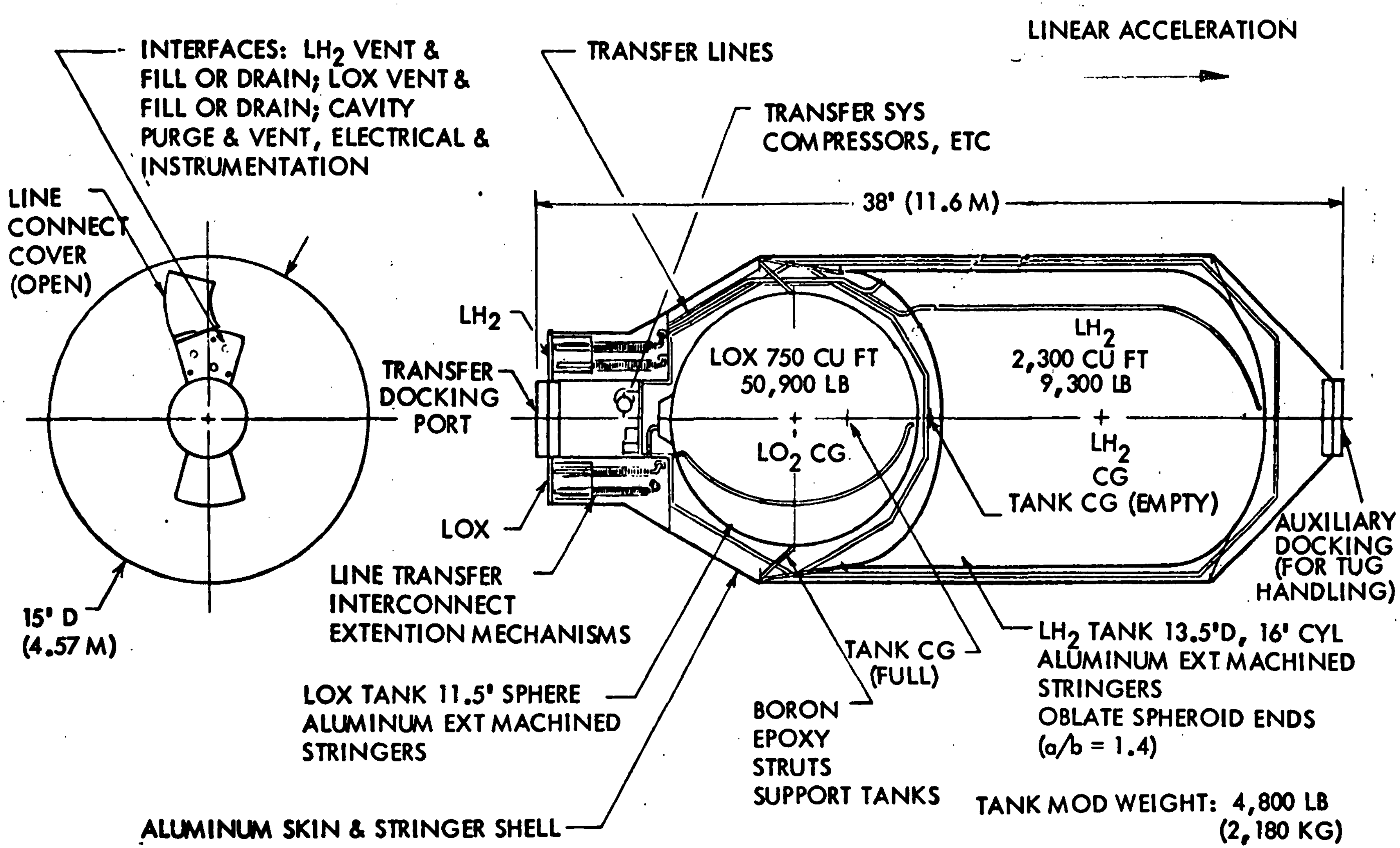


Figure 5.4-2 Transfer Capability Logistics Tank

5.4.2 Mini-Depot Module Weights

A listing and comparison of estimated weights of the mini-depot and transfer capability tank modules is given in Table 5.4-1. The transfer capability tank is slightly heavier than the logistic and storage tank because of the additional transfer equipment and systems. The equipment module for the modular, linear mini-depot does not have as stringent a design requirement for lowest possible weight as do the logistic tanks; therefore a simpler, slightly heavier structure is assumed. For the rotational mini-depot the equipment module is required to be extremely heavy to act as a counter weight. The design and manufacture should thus be modified to the point where this would be a less expensive module despite its weight. Some system weights are shown as higher in this module, this was to allow more use of "shelf" hardware or systems developed for other space program elements and should further reduce module cost.

5.5 MINI-DEPOT COST RATIONALE

The mini-depot is one of several modes of storing propellant in space that has been evaluated as part of the ISPLS study. This ability to store propellant in space allows for full utilization of the shuttle payload capability on each shuttle flight.

Two alternate mini-depots have been considered. The first is a rotational transfer depot and the second is a linear transfer depot. Each depot consists of an equipment module and a logistic tank. The costing approach on the different pieces of hardware is identical and is described below. In addition to the mini-depot logistic tank, the cost of a direct transfer tank has been included.

The non-recurring development costs for both mini-depot equipment module configurations are derived from both S-II and Apollo Cost Estimating Relationships (CER's). The CER's are applied at the subsystem level, in structures, insulation, docking, etc. Prior to their application these CER's were adjusted for "complexity" and "know-how." Complexity factors adjusted the CER based on a comparison between the work being proposed versus the work it is being related to. Know-how factors adjust the state of the art between the work proposed versus the work to be accomplished. These factors, along with weight scaling, adjust the "dollars per pound" of the CER's prior to their application.

The recurring production costs for both mini-depot equipment modules utilizes the same technique described above except that "know-how" is considered only for non-recurring development.

Table 5.4-1 Mini-Depot Module Weights

MINI-DEPOT MODULE WEIGHTS	Logistics Storage Tank	Transfer Capability Tank	Linear Equipment Module	Rotational (2) Equip Module
STRUCTURE (Total)	(2,870)	(3,100)	(2,840)	(58,360)
Shell	900	950	800	-
LOX Tank	300	300	60	150
LH_2 Tank	685	765	130	400
Tank Supports	150	150	40	100
Equip. House	-	-	-	24,300
Boom	-	-	-	8,550
Docking: EOS	-	-	400	500
" Trans.	120	120	120	200
" AUX	120	120	120	200
Bulkheads	50	50	200	2,000
Fairings	30	90	50	500
Support Bkts	10	20	200	800
Ballast	-	-	-	20,260
Growth	280	310	460	-
Meteoroid Protection	60	60	80	100
Orbiter Attach Ftgs.	165	165	180	300
INSULATION	300	250	100	200
TRANS SYSTEMS (Total)	(260)	(520)	(650)	(900)
Interconnect Mech.	30	180	360	500
Lines, Valves	60	70	90	200
Actuation, Monitor	60	60	100	100
Compressors, Meters	-	100	100	100
Prop. Gaging	50	50	-	-
Zero G Vent Sys.	30	30	-	-
Purge	30	30	-	-
ELECT POWER (Total)	NA	(70)	(620)	(850)
Fuel Cells, Lines			200	250
Batteries		60	120	150
Distribution		10	300	450
ATTITUDE CONTROL (Total)	NA	NA	(700)	(850)
Accum. Tanks, Lines			200	250
Thrusters			400	400
Settling Thrusters			100	200
AVIONICS (Total)	(10)	(10)	(590)	(590)
Instrumentation	10	10	100	100
DCM	-	-	150	150
IMU, GN&C	-	-	240	240
Rendez. & Docking	-	-	100	100
COMMUNICATIONS	NA	20	100	150
MODULE SUPPORT (Total)	(960)	(830)	(100)	(100)
Cargo Umbilicals	140	80	50	50
" Interconnect	120	50	50	50
(1) Repress Sys.	700	700	-	-
TOTAL:	4,400	4,800	5,700	62,000

(1) Includes tank inerting repress 450 lb. allowance (not req'd for current baseline)
(2) This module is not weight critical; therefore assume boilerplate structure and heavier, simplified systems.

Operations costs are based on the following rationale.

Rotational Equipment Module

Initial shuttle launch @ $10.0 million per launch	$10.0 M
Two shuttle returns for mini-depot refurbishment @ 50% of shuttle flight cost	$10.0 M
Two shuttle launches after refurbishment	$20.0 M
Miscellaneous launch operations cost	$ .7 M
Refurbishment costs based on a per cent of the first unit (TFU)	$ 2.9 M
Total Rotational Equipment Operations Costs	$43.6 M

Linear Equipment Module

Initial launch @ $10.0 million per launch	$10.0 M
Two shuttle returns for mini-depot refurbishment @ 50% of shuttle flight cost	$10.0 M
Two shuttle launches after refurbishment based on shuttle weight capability versus mini-depot weight	$ 1.6 M
Miscellaneous launch operation cost	$ .7 M
Refurbishment costs based on a per cent of TFU	$ 2.7 M
Total Linear Equipment Operations Costs	$25.0 M

A summary of the mini-depot equipment module costs is are follows:

	EQUIPMENT MODULE	
	Rotational	Linear
Development	$ 98.9 M	$ 90.2 M
Production	$ 14.5 M	$ 13.4 M
Operations	$ 43.6 M	$ 25.0 M
Total	$157.0 M	$128.6 M

The costs for the mini-depot logistic tank is $23.8 million dollars for design, development, test and evaluation (DDT&E). This effort was costed in the manner described previously, a complexity, know-how and weight scaling of selected CER's.

The production costs are based on a quantity of five tanks of a 90 per cent weight cost reduction curve. The first unit (TFU) costs equal $3.2 million and five tanks total $12.6 million.

Operations costs associated with the tanks are based on a factor of production costs developed from previous program experience and equals $10.7 million.

The costs for the direct transfer capability tank is $26.5 million for DDT&E. The DDT&E cost difference between the two tanks is the result of weight variances. The costing approach is identical. The production costs are based on a quantity of five tanks on a 90 per cent weight cost reduction curve. TFU costs equal $3.6 million and five tanks total $14.3 million.

Operations costs associated with the tanks are based on a factor of production costs developed from previous program experience and equals $12.1 million.

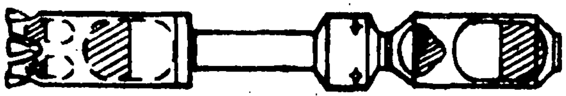
A summary of the mini-depot tank costs are as follows:

	TANKS	
	Rotational/Linear	Direct Transfer
Development	$23.8 M	$26.5 M
Production	$12.6 M	$14.3 M
Operations	$10.7 M	$12.1 M
Total	$47.1 M	$52.9 M

5.6 USE OF MINI-DEPOT EQUIPMENT MODULE IN PROPELLANT TRANSFER OPERATIONS

The space based mini-depot equipment module could be employed for use in transferring propellant from the logistic tank to the tug whether or not the mini-depot were required as a storage device. The equipment module would contain thrusting equipment to provide the low g acceleration for propellant settling for the transfer operation and would also contain propellant transfer equipment. In the case of direct transfer of propellant from the logistic tank to the tug, low thrust jets for propellant settling and propellant transfer equipment would have to be additional equipment on the logistic tank or on the tug.

The concepts for propellant transfer using the equipment module in rotational and linear acceleration modes and the direct tank-to-tug transfer in a linear acceleration mode are shown in Figure 5.6-1. Use of the equipment module in the rotational mode would result in a propellant loss of only 0.6 per cent as compared with 2.8 per cent for the direct transfer mode. The 2.8 per cent

	MINI-DEPOT		NO DEPOT	
	ROTATIONAL TRANSFER	LINEAR TRANSFER	DIRECT TRANSFER TANK, LINEAR	
				
TANK CAPACITY	60,000#	60,000#	59,600#	
TRANSFER LOSS	0.6%	2.4%	2.1%	
REDUCED TANK CAPACITY	—	—	.7%	
			2.8%	TOTAL
COST COMPARISON:				
LOSSES (TRANSFER & CAPACITY) SIX YEAR, LEVEL C *	$7.5M	$30.1M	$35.0M	
EQUIPMENT MODULE	157.0	128.6	—	
PROPELLANT TANK	47.1	47.1	52.9	
	$211.6 M	$205.8 M	$87.9M	

TANK COSTS

	ROTATIONAL/LINEAR	DIRECT TRANSFER
DEVELOPMENT	$23.8M	$26.5M
PRODUCTION (5 TANKS)	12.6	14.3
MAINTENANCE	10.7	12.1
	47.1 M	52.9 M

*BASED ON 7,024,000 LB PROPELLANT USED BY TUG IN 6 YEARS, 0-30° MISSIONS LEVEL C.
(% LOSS x 7,024,000 LB x $178/LB)

Figure 5.6-1 Use of Mini-Depot in Propellant Transfer Operation loss includes a reduced capabity of the tank of 0.7 per cent (treated as a loss) as compared with the mini-depot tank because of an allowance for the additional weight of 400 pounds for the transfer equipment and low thrust jets mounted on the tank.

The question arises whether the cost of propellant saved (by reduced losses) by the use of the mini-depot equipment module as a rotational transfer device would offset the cost of development and operation of the equipment module. The cost estimates and value of the propellant transfer losses presented in Figure 5.6-1 indicate that the cost of the propellant lost in the direct linear transfer mode is small ($35.0 M) as compared with the cost of development and operation of the equipment module ($157.0 M for the rotational version).

The overall conclusion derived from the data was that the mini-depot equipment module used as a propellant transfer device alone (if it were not required for propellant storage) would not be economical.

The transfer losses in the Figure were derived from the propellant transfer method trade study Section 6, Volume III. The value of propellant lost has been priced at $178 per pound which is the dollar value of propellant delivered to space calculated in the Propellant Delivery Mode Study, Section 2, Volume III. The propellant which could be transferred by the mini-depot module is 7,024,000 pounds and is the tug propellant for 0 to 30 degree missions in the six-year interval from 1985 to 1990 when the tug could be in the program in accordance with this study planning ground rules.

6.0 PROPELLANT TRANSFER METHOD

6.1 INTRODUCTION

The resupply of cryogenic propellant to space-based vehicles in earth orbit requires propellant transport from the surface of the earth and propellant transfer in the weightless environment of space. The transport system encompasses the transport vehicle, propellant container, and equipment for container deployment. The transfer system encompasses those subsystems necessary to transfer the propellant from the delivery container to the user vehicle. During the course of this trade study the space shuttle has been considered the primary transport vehicle utilizing a propellant logistics module in the cargo bay for propellant delivery. The user vehicles are baselined as the space-based tug, Chemical Interorbital Shuttle (CIS), and Reusable Nuclear Stage (RNS).

The purpose of the analysis presented in this section is the selection of the most favorable propellant transfer method in support of in-space propellant logistics. In the zero gravity environment of space, a satisfactory propellant transfer system must comprise an integrated set of subsystems providing:

- Liquid/vapor interface control
- Receiver tank thermodynamic control
- Expulsion
- Net positive suction pressure control

The presentation of this trade study is organized around these four subsystems. Each subsystem section identifies candidate concepts, provides a discussion of each candidate concept including an analysis of the salient functional and physical characteristics of each, and finally presents the concept selection along with the selection rationale. The selection of a baseline propellant transfer system, and the subsequent configurational and operational definition of the system are required to select and define the most favorable in-space propellant logistic concept and delivery vehicle propellant logistics module for use with the tug, CIS, and RNS.

6.2 SUMMARY OF RESULTS

To achieve satisfactory propellant transfer in the zero gravity environment of space, the following subsystems must be provided:

a. Liquid/Vapor Interface Control - This subsystem provides the necessary control to assure that the ullage and liquid are properly located to allow acceptable supplier tank outflow, liquid phase or acceptable propellant quality through the transfer lines, and acceptable receiver inflow conditions.

b. Receiver Tank Thermodynamic Control - This subsystem provides control to assure acceptable inflow characteristics, prevent unnecessary overboard venting of liquid or vapor, and maintain or establish receiver propellant thermodynamic conditions which fulfill the receiver vehicle's propulsion system outflow requirements.

c. Expulsion - This subsystem provides the energy and/or means of expelling the propellant from the supply container into the receiver.

d. Net Positive Suction Pressure Control - This subsystem provides vapor pressure control to establish subcooled or acceptable quality propellants to fulfill the requirements as established by the total transfer system.

The candidate concepts evaluated for liquid/vapor interface control are (1) linear acceleration, (2) radial acceleration, and (3) capillary devices. The application of each concept to propellant transfer from the logistic module to the tug, CIS, and RNS was analyzed. The criteria used for selection included propellant transfer losses, compatibility with user and logistic vehicle systems, development risk, and safety.

Considerable analytical work was accomplished to support the selection of the most favorable liquid/vapor interface control concept. Logistic module liquid residuals were computed parametrically as a function of acceleration level, flow rate, and tank geometry. Selected acceleration levels range from 10^{-5} g to 10^{-4} g. Propellant transfer flow rate throttling at a ratio of 10 to 1 for initial to final flow was found to significantly reduce residuals (by a factor of 6 to 20 depending on the propellant and tank geometry) and was, therefore, selected for baseline operations. Propellant residuals can be minimized through the use of a conical LO_2 bulkhead and a reversed (S-II type) LH_2 bulkhead. Propellant savings in the form of reduced residuals were computed to 700 pounds for LO_2 and 150 pounds for LH_2 as compared to standard elliptical bulkheads for the case of a 10-hour transfer to tug.

Propellant transfer times and acceleration levels were optimized for linear acceleration by minimizing propellant transfer losses. Optimum transfer times are 10, 15, and 10 hours for the tug, CIS, and RNS, respectively. Propellant transfer losses were computed to be in the range of 2.1 percent to 5.7 percent of the approximately 60,000 pounds of propellant transferred from the logistics module. These losses include propellant for residuals, auxiliary propulsion for acceleration, pressurization, pumping power, and transfer line chilldown.

Rotational rates and associated radial acceleration levels were optimized to minimize propellant losses for various rotational transfer concepts. Optimum rotational rates and acceleration levels range from 20 to 80 revolutions per hour and 8×10^{-4} g to 1.3×10^{-3} g, respectively, depending on the vehicle configuration involved. Propellant transfer losses are considerably less than for linear acceleration, ranging from 0.6 percent to 1.5 percent of the 60,000 pounds of propellant transferred. Rotation was not found to be practical for all configurations, since the center of gravity and, thus, the axis of rotation would enter the propellant tank. This condition leads to complex and unpredictable liquid orientation within the vehicle tank seriously compromising propellant gauging, ullage venting, and fluid inlet system requirements.

Two alternate concepts for linear acceleration thrust orientation were evaluated to determine their influence on orbital mechanics. In-plane thrusting was found to be unstable resulting in earth impact by the orbiting body after several revolutions. Cross-plane thrusting at a 180-nautical mile altitude causes a plane change orbit displacement resulting in a line-of-sight distance between the quiescent shuttle and the logistic module/user vehicle combination of only 7.1 miles after 15 hours of propellant transfer.

Because of the relatively high propellant losses for the CIS as compared to the tug, capillary devices were evaluated for their potential application to liquid/vapor interface control. It was found that some linear acceleration was still required for initial settling, propellant gauging, and for the last few logistic module loads when the CIS liquid level approached the vent outlet. For this reason and because of the higher development risk associated primarily with the integration of surface tension devices with a cryogenic thermal control system, a capillary system was not selected for liquid/vapor interface control.

On the basis of the selection criteria developed and the analytical data generated, linear acceleration was selected for the baseline propellant transfer system liquid/vapor interface control mode in support of tug, CIS, and RNS propellant resupply. This selection was made primarily on the basis of moderate program cost, minimum impact on shuttle and user vehicle configurational and operational requirements, and minimum development risk.

The candidate concepts evaluated for receiver tank thermodynamic control are: (1) overboard vent during transfer, (2) connected ullage, (3) overboard vent prior to transfer, and (4) use of the user vehicles existing thermodynamic vent system. Overboard vent during transfer was eliminated because of the attendant high propellant loss. Connected ullage, which allows the conservation of vent gases by returning them to the source tank, appears most favorable, but requires positive liquid/vapor interface control in the receiver tank. Full overboard venting of the receiver permits transfer without propellant settling. However, it is obvious that this method cannot be used for adding propellant to partially filled vehicles.

Use of the existing baseline design thermodynamic vent systems in connection with propellant transfer is feasible for only those receiver vehicles requiring multiple logistic module loads for a complete fill. The time between transfers is used to cool the tank and condense the vapors in the ullage to reduce the pressure increase resulting from the vaporization during the transfer. Thermodynamic venting does not require liquid/vapor interface control in the receiver vehicle since either liquid or gas can be efficiently utilized in the vent system. Impact to the use vehicle design results from a requirement for increased venting capacity, the need for bulk liquid mixers, and additional controls.

The connected ullage concept was selected as the baseline primarily on the basis of minimum propellant loss, system simplicity, and compatibility with the user vehicle configuration.

The candidate concepts evaluated for expulsion are: (1) liquid pump, (2) gas pump, (3) liquid-to-gas conversion pressure expulsion, (4) stored gas pressure expulsion, and (5) positive displacement. Power level and propellant consumption for power by a fuel cell were analyzed for both the liquid and gas pumps. The power level computed was 160 watts for transfer to tug, well within the tug power supply capability, and fuel cell consumption was found to be less than 0.1 percent of the 60,000 pounds of propellant transferred. No significant differences in power or propellant consumption were found between the liquid and gas pump.

Propellant consumption and system hardware weight penalties for LO_2 transfer were approximately 350 percent and 200 percent greater for liquid-to-gas conversion pressure expulsion and for stored gas pressure expulsion, respectively, primarily due to the additional pressurant and gas storage requirements, as compared to pump expulsion scavenging pressurant from the receiver tank.

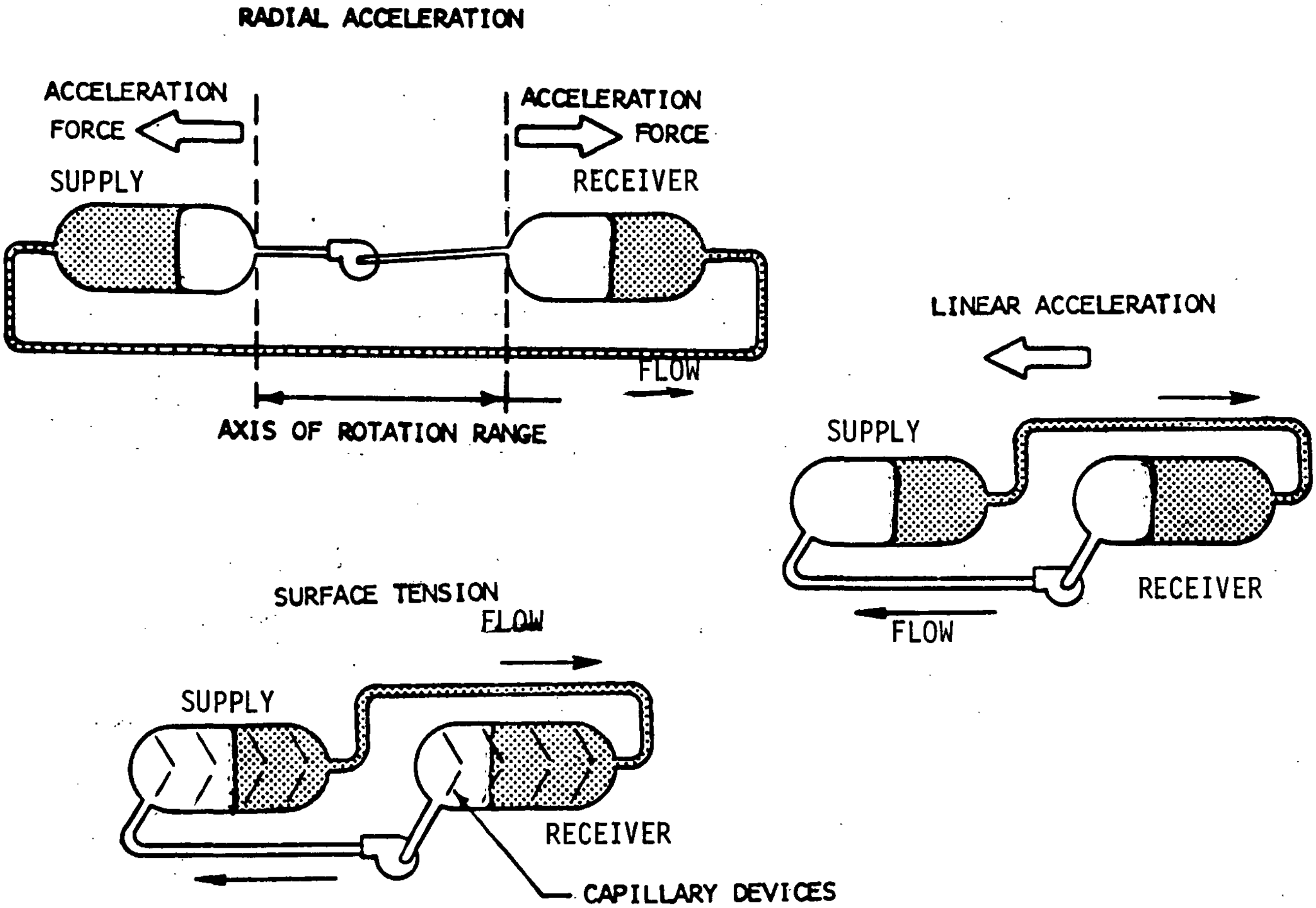
Evaluation of positive displacement expulsion revealed considerable development risk for this concept. Although this concept provides a degree of liquid/vapor interface control in the source tank, additional provisions would have to be made for suppression of gas formation under the bladder or bellows and for propellant gauging in the receiver tank. In addition to the reason discussed above, the gas pump concept was selected as the baseline expulsion technique because of its suitability to an in-line installation. As compared to an in-tank installation, an in-line gas pump provides improved accessibility and flow reversal capability through the use of appropriate valving and line routing.

The candidate concepts evaluated for net positive suction pressure control are: (1) self pressurization, (2) liquid-to-gas conversion using a gas generator and pump, (3) liquid-to-gas conversion using a solar heat exchanger and pump, and (4) stored gas. Because of long bubble collapse times in zero gravity (10 hours for a one foot spherical oxygen gas bubble), self pressurization was deemed unsatisfactory for refilling vehicles using capillary start baskets for zero g engine start. Use of a solar exchanger was eliminated because of the large surface area required. Stored gas was not found to be competitive in terms of weight penalty. For these reasons, liquid-to-gas conversion using a gas generator and pump was selected as the baseline concept for net positive suction pressure control.

6.3 LIQUID/VAPOR INTERFACE CONTROL

6.3.1 Candidate Concepts

Probably the most critical and most difficult requirement to achieve in support of orbital propellant transfer is liquid/vapor interface control. Previous studies have shown that the following concepts are most promising.



As can be seen the concepts employ either acceleration or surface tension for liquid/vapor interface control. Criteria used to select the technically preferred technique include:

- Propellant transfer losses
- Compatibility with user and logistic vehicle systems
- Development risk
- Safety

6.3.2 Discussion of Candidate Concepts

Considerable analysis has been completed during the course of this trade study to evaluate each of the candidate liquid/vapor interface control concepts in terms of the selection criteria identified. The areas of analysis include evaluation of:

a. Source tank residuals and the effect of flow rate throttling

b. Optimization of residual and thruster propellant losses for linear acceleration and the effect of modulated acceleration level

c. Optimization of residual and thruster propellant losses for radial acceleration

d. Effect of rotation on orbiter crew

e. Propellant orientation during rotational transfer

f. Orbital mechanics for linear acceleration

g. Characteristics of capillary propellant transfer

The details of these analyses and the influence of their results on the selection of a baseline liquid/vapor interface control concept are presented in this section.

Orbital transfer of propellant involves several types of propellant losses. These include such items as source tank residuals, acceleration propellant, pressurization power, transfer line heat leak, transfer line residuals, and tank and line chilldown.

Two of the most important losses are: residual propellant trapped in the logistic (supply) tanks due to ullage gas pull through, and auxiliary propulsion system (APS) thrusting propellant for vapor/liquid interface control. These two losses are related, as pull through residuals are a function of the acceleration field generated by APS thrusting. To this end, analysis was conducted with the following objectives: (1) develop the functional dependance of pull through residuals on flow rate, acceleration level, and tank geometry; (2) identify and define operational procedures, such as flow rate throttling or thrust level increase, and logistic tankage configurations to reduce residuals; (3) develop data for continuous linear acceleration vapor/liquid interface control; and (4) determine acceleration levels and transfer times to minimize combined APS propellant and residual propellant losses for refueling of the space-based tug, CIS, and RNS.

6.3.2.1 Source Tank Residuals

Vapor pull through is caused by the non-uniform velocity of the flow into the outlet line. As the liquid surface nears the outlet line, the non-uniformities cause the interface to deform, leading to gas ingestion, and concomitant trapped residual. Previous studies (References 6.0-1 and 6.0-2) have shown that liquid residuals due to pull through can be a large portion of total propellant losses incurred during transfer.

Liquid residual due to the pull through phenomenom is a function of Froude number, Bond number, and tank outlet geometry (Reference 6.0-3). Froude number, Fr, is a measure of the ratio of inertia to body forces, based, for example, on outlet conditions $Fr = v^2/ad$, where

v is the velocity in the outlet line

a is the acceleration

d is the outlet line diameter

Bond number, Bo, is a measure of the ratio of body to surface tension forces, based on tank diameter, $Bo = \rho a D^2/\sigma$, where

ρ is the liquid density

a is the acceleration

D is the tank diameter

σ is the surface tension

Generally speaking, prior experimental work on this problem has been done for high Bond numbers, i.e., $Bo > 300$ and for low Bond numbers, i.e., $Bo < 0.1$. Work on intermediate values of Bond numbers is generally lacking.

For the high Bond number condition, considerable work has been done, but Reference 6.0-4 is believed to be the most comprehensive. The NASA has recently completed an experimental study of pull through under normal gravity conditions (high Bond numbers) for hemispherically bottomed tanks (Reference 6.0-5). For low Bond numbers most of the work has been done by the NASA Lewis Research Center (References 6.0-5 and 6.0-6), although some work of this nature was accomplished in industry laboratories as reported in Reference 6.0-7. Applicable results are discussed below.

It should be noted that pull through is not the only cause of liquid residual. Other potential contributors to liquid residual are vortex formation and propellant slosh at low liquid levels. For booster and spacecraft propulsion systems, these problems have been prevented by vortex and low level slosh baffles; e.g., Reference 6.0-8. Similar hardware fixes are expected to be applicable to in-orbit propellant transfer; however, the low g levels, long transfer times, spacecraft perturbations, and orbital dynamics may require new baffle designs. In addition, these factors may entail operational constraints during transfer. Additional work should be done in subsequent studies to minimize and to develop predictive techniques for these other residuals.

The primary pull through data used in the pull through analysis was obtained from Reference 6.0-4. These data were obtained for a variety of tank outlet shapes and baffle configurations. Typical data from this reference are presented in Figure 6.3.2-1, as a plot of V/D^3 versus Froude number, v^2/ad. Here, using English units as typical,

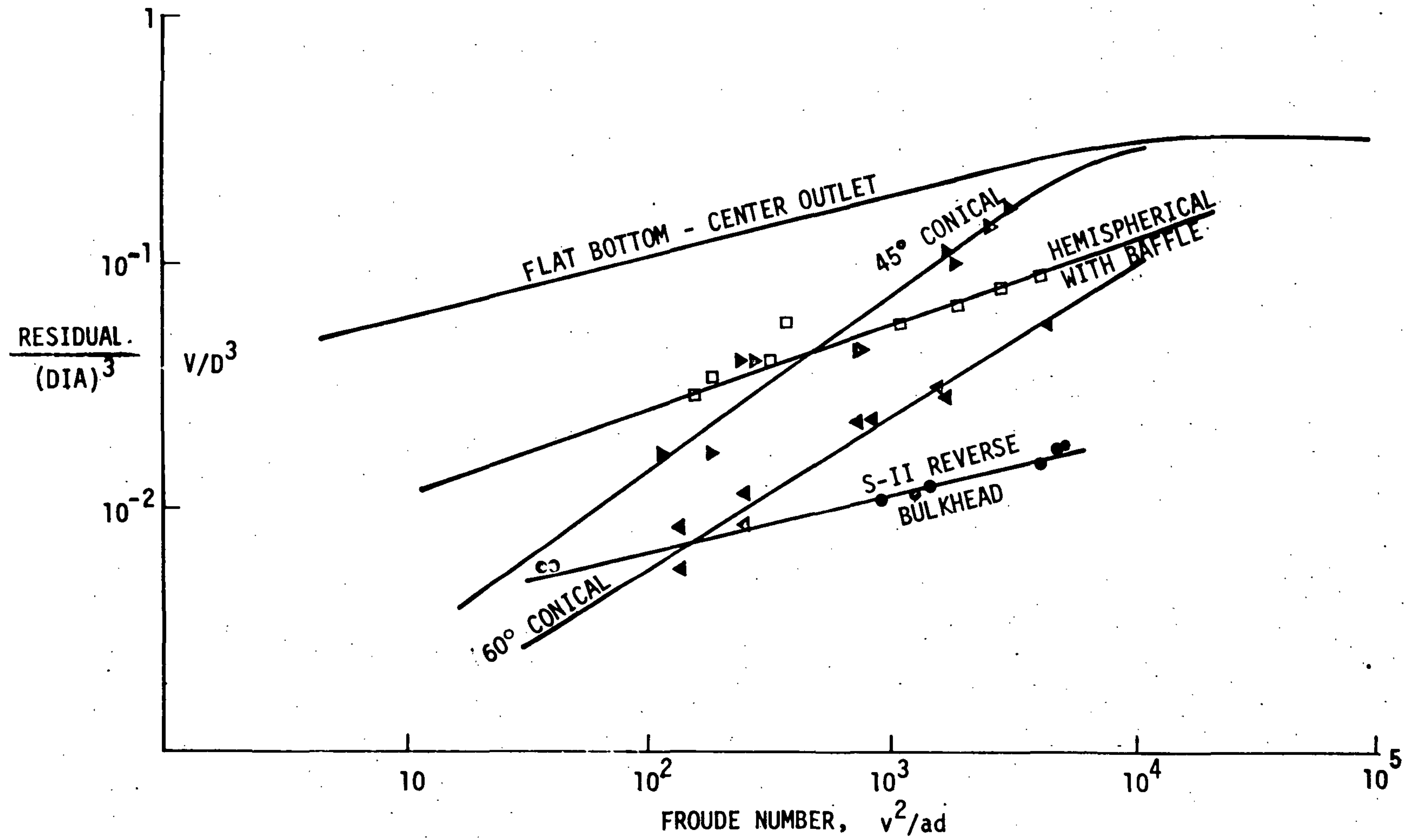


Figure 6.3.2-1 Residual Versus Froude Number for Various Tank Geometries

V is the volume of residual liquid, ft^3

D is the tank diameter, feet

v is the velocity in the outlet line, feet/second

a is the acceleration, ft/sec^2

d is the outlet line diameter, feet

Results are presented for a flat bottomed tank with center outlet, 45 degrees and 60 degrees conical tank bottoms, a hemispheric bottom with a pull through baffle, and a Saturn S-II reverse bulkhead. Conical bottoms are seen to be particularly attractive at low Froude number. The data are for Bond numbers greater than 300 and the ratio of tank to outlet diameter (D/d), greater than 10. Caution is urged in extrapolating to high Froude numbers as an effect of D/d is expected. Extrapolated results at high Froude numbers for the various configurations should fall below the asymptotic limit for the flat bottom, center outlet case. Extrapolation to lower Froude numbers is also hazardous because of the small residuals involved. In the work that follows straight line extrapolation was used at the lower Froude numbers.

Before proceeding, it should be pointed out that the data upon which the curves of Figure 6.3.2-1 are based were developed from small model experiments at one g. For most of the tank bottom configurations studied, residuals were very sensitive to changes in pull through height. Accurate measurement of pull through height is, therefore, critical. This is illustrated dramatically by a comparison of data from Reference 6.0-4 and Reference 6.0-5 for hemispherical bottomed tanks, as shown in Figure 6.3.2-2. Based on an outlet diameter of 0.5 feet and tank diameter of 11.5 feet, data from the two sources were used to calculate curves of residual LO_2 versus efflux rate, for various acceleration levels. Acceleration is given in terms of fraction of earth gravity field, g, where

g = a/gc, lbf/lbm

a is acceleration, ft/sec^2

gc is the gravitational conversion constant, 32.2 $\frac{lbm\ ft}{lbf\ sec^2}$

It is seen from the curves that the Reference 6.0-4 data yielded residuals a factor of two to four times as great as obtained from Reference 6.0-5. Within the scope of this effort it was not possible to reconcile the differences. Reference 6.0-4 data were used because they were more conservative.

At $g = 10^{-5}$, Bond numbers less than 300 were encountered. Curvature and surface tension effects on trapped residual were no longer negligible. Residuals were determined using Figure 6.3.2-1, to which was added residuals due to curvature, as determined from the vapor/liquid configurations of Reference 6.0-7. Results so obtained have been presented in Figure 6.3.2-2. In subsequent presented data residuals were also calculated with due regard to curvature effect. Figure 6.3.2-2 also shows data on the improvement obtained by using a pull through baffle.

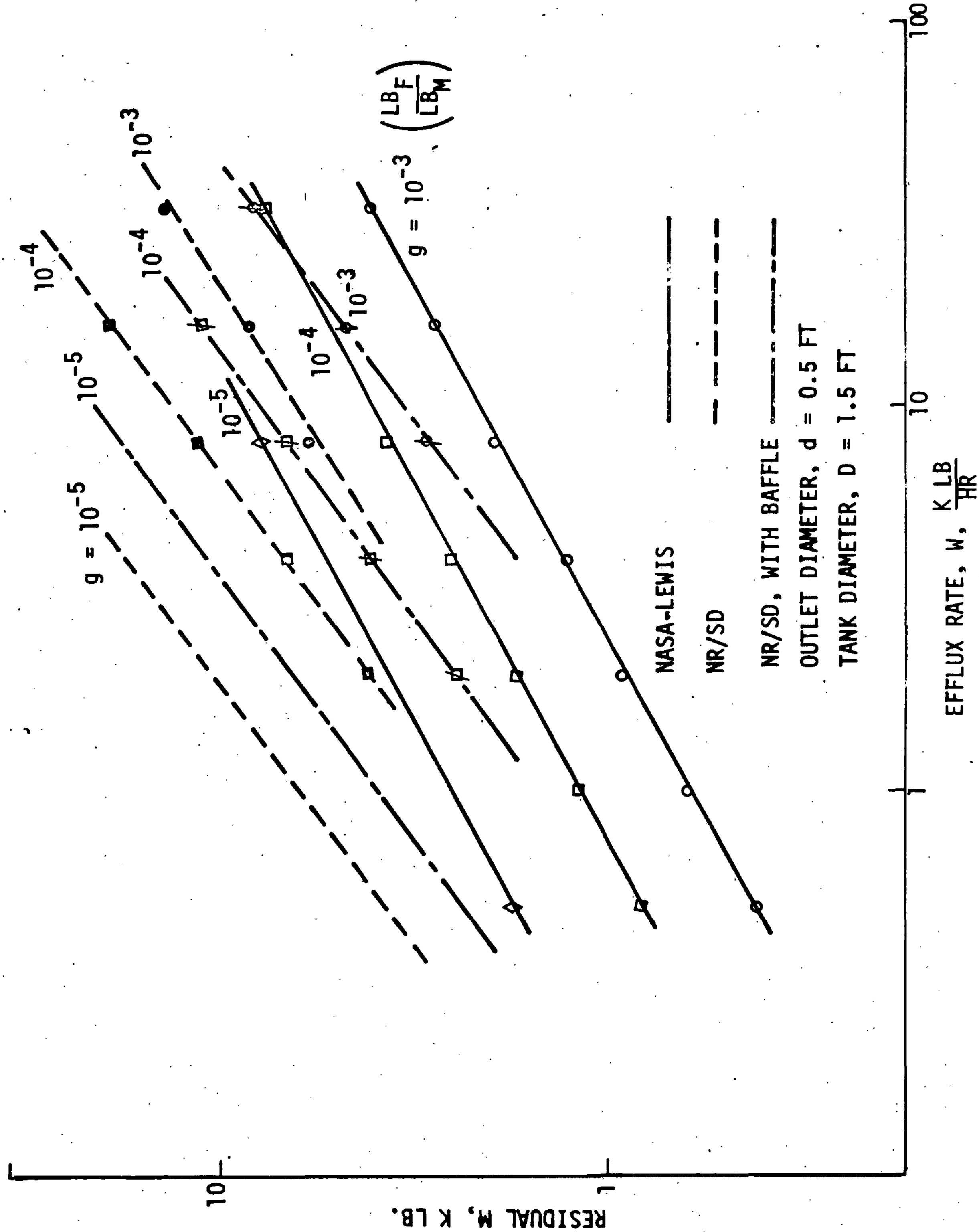


Figure 6.3.2-2 LO_2 Residuals: Hemispherical Bottomed Tank

Based on data from Figure 6.3.2-1 and Reference 6.0-7, propellant management techniques and trade studies were conducted to determine operational procedures and designs to decrease trapped residual. Variation in outlet diameter was not considered. For a given mass flow rate, trapped residual can be reduced by increasing the outlet diameter. However, there are practical limits to the size of the outlet diameter. Determination of the preferred outlet diameter was not attempted and in this analysis for all logistic tanks considered an outlet diameter of 0.5 feet was used.

Two basically different types of logistics tanks were considered and are shown in Figure 6.3.2-3. The first type has both oxygen and hydrogen tanks and is used for refueling the tug or the CIS. The approximate tank propellant capacity for the tug is 50,900 pounds of LO_2 and 9300 pounds of LH_2. The capacity for the CIS is 49,000 pounds of LO_2 and 11,000 pounds of LH_2. Differences are due to a number of factors, including allowance for LH_2 boiloff during extended refueling. These tanks are weight limited and do not require the entire volume of the space shuttle cargo bay. The tank diameters used were 13.5 feet for LH_2 and 11.5 feet for LO_2.

The second type of logistic tank contains LH_2 only and was used for refueling the RNS. This module is volume limited, requiring the total capacity of the space shuttle cargo bay. Tank diameter was 13.5 feet and LH_2 weight was 34,000 pounds. Figure 6.3.2-3 shows the tank bottom configurations selected as a result of this analysis. The rationale for this selection follows.

The dual propellant tank for refueling the tug and CIS is discussed first. The design of the LO_2 tank outlet region is most important, as in general the weight of the trapped LO_2 is several times greater than the weight of trapped LH_2. A reverse bulkhead similar to that used for the Saturn S-II was considered for the LH_2 tank. This design yields a compact logistic module and low LH_2 residuals.

Figure 6.3.2-4 presents LH_2 pull through residuals plotted against flow rate for acceleration levels of $g = 10^{-3}$, 10^{-4}, and 10^{-5}. Results were obtained using Figure 6.3.2-1 and Reference 6.0-7. The vapor/liquid interface shape data of this reference were important in establishing the residual at $g = 10^{-5}$, where capillary effects were important. Residuals are not a large fraction of the loaded propellant except at the highest flow rate and lowest acceleration level.

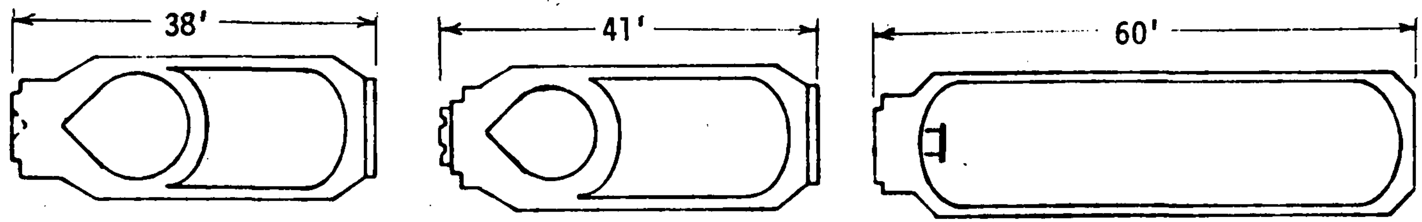
Attention is now directed to the design of the LO_2 tank outlet region. A number of possibilities were considered and are shown in Figure 6.3.2-5:

a. A 60-degree (angle made with horizontal) conical bulkhead

b. A 45-degree conical bulkhead

c. A hemispherical bulkhead with pull through baffle

d. A hemispherical bulkhead with sump and pull through baffle



PRELIMINARY PROPELLANT LOADING

	TUG	CIS	RNS
LH_2 (LBS)	9,300	11,000	34,000
LOX (LBS)	50,900	49,000	(VOLUME LIMITED)

Figure 6.3.2-3 Logistic Module Configurations

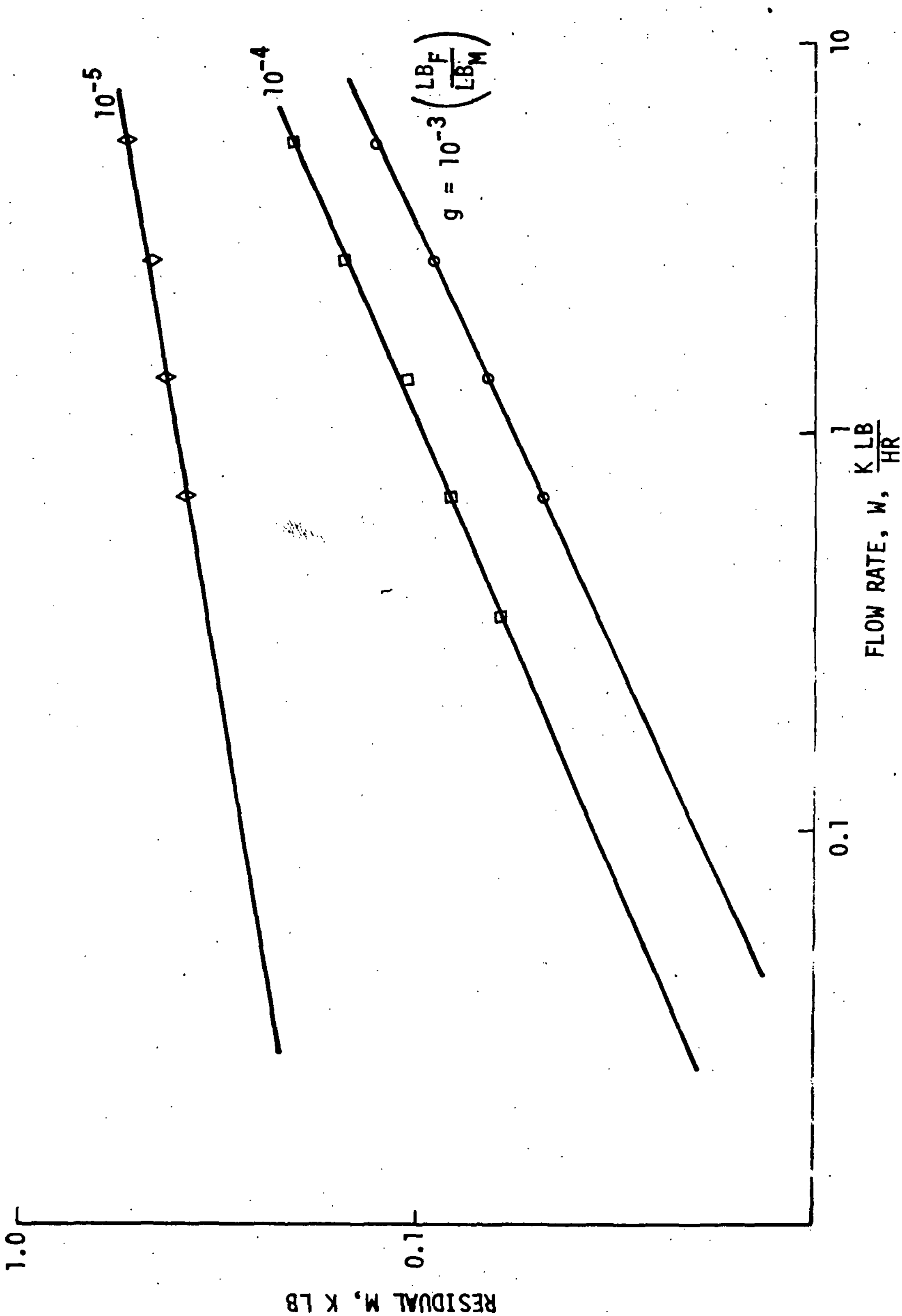


Figure 6.3.2-4 LH_2 Residuals: S-II Reverse Bulkhead

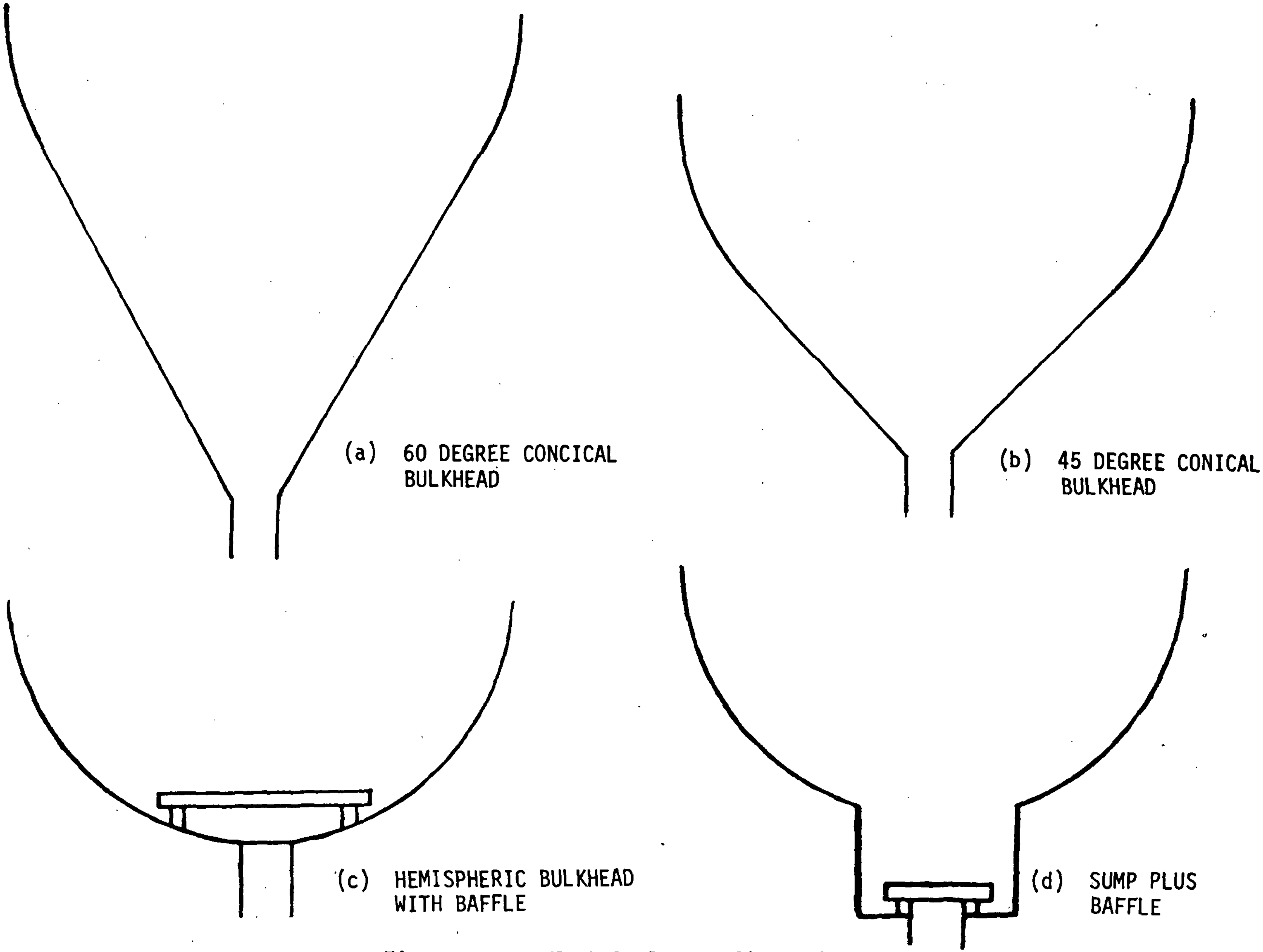


Figure 6.3.2-5Tank Outlet Configurations

Configuration (a) yielded low residuals, but would have required lengthening of the logistic shell. Configuration (d) appeared promising, but there were insufficient data for adequate evaluation. Sufficient data were available to evaluate Configurations (b) and (c), both of which are practical alternatives. In Figure 6.3.2-6, LO_2 residuals are plotted against mass flow rate for these two configurations, for accelerations of 10^{-3}, 10^{-4}, and 10^{-5}. The 45-degree conical bulkhead yielded lower residuals for low and moderate flow rates. A cross over occurred at higher flow rates, where the hemispherical bulkhead yielded lower residuals. Practical range of operation, as will be seen, is at the lower flow rates and hence the 45-degree conical bulkhead is preferred with regard to minimizing residuals.

The logistic module for the RNS is volume limited and, therefore, outlet region contouring and lengthening, as was done with the conical bulkhead, is unacceptable. Interior pull through baffles or pull through screens are the preferred design approach. Studies conducted have shown that an elliptical bulkhead with pull through baffle, when used in conjunction with flow rate throttling, constitutes an acceptable outlet region design. Flow rate throttling also enhances the feedout efficiency, i.e., reduces pull through residuals, for the dual propellant logistic modules.

Flow Rate Throttling During Transfer

Previous studies (References 6.0-1 and 6.0-2) have shown that it is advantageous to decrease the efflux rate at incipient vapor pull through, utilizing a preprogrammed flow rate reduction profile which avoided pull through until flow rate had decreased to a fraction of the initial value. Reference 6.0-1 considered flow rate throttling such that the ratio of initial to final mass flow rate (W_i/W_f), was 10 and 100. Both values of flow rate throttling yielded lower residuals and shorter minimum loss transfer times, with best results achieved with 100:1 throttling. However, 100:1 flow rate throttling is believed to be impractical. Since pressure drop varies with the square of the flow rate, 100:1 throttling implies a pressure drop variation of 10,000:1. An extremely sensitive control system would be required to effect transfer with 100:1 throttling. For this reason, 10:1 flow rate throttling was chosen as baseline for the study.

The benefits of flow rate throttling depend upon the particular conditions of transfer; e.g., acceleration level, propellant, and tankage configuration. Typically, 10:1 flow rate throttling decreases residuals by a factor of four and decreases the preferred transfer time by a factor of eight. Here, throttling decreases the preferred transfer time by making it advantageous to use higher initial flow rates. Of course, for a given initial flow rate, throttling increases the time required to effect transfer compared to non-throttled flow. This is shown in Figure 6.3.2-7 where initial flow rate is plotted against transfer time for LO_2 and LH_2, based on 10:1 throttling. Also shown on this figure is LO_2 flow rate versus transfer time for non-throttled flow. This data is based on the tug supportive logistic module described in Figure 6.3.2-3. Throttling has little effect on transfer time for low flow rates (where pull through is delayed), but at the higher flow rates (where pull through occurs relatively early in the transfer) throttling results in significant increases in transfer time. At the highest flow rate considered, transfer time was more than doubled by 10:1 flow rate throttling.

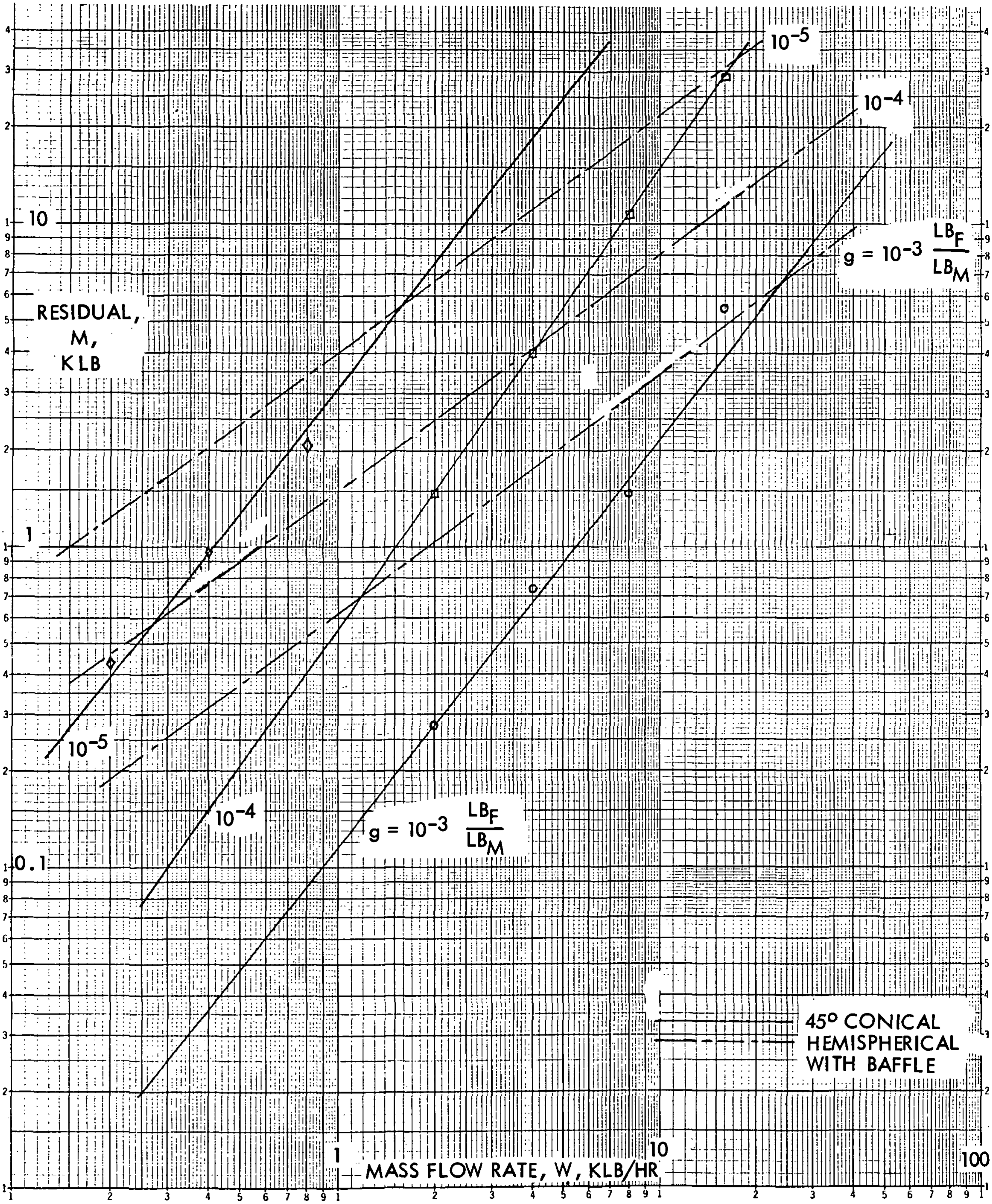


Figure 6.3.2-6 LO_2 Residuals: Various Tank Bottoms

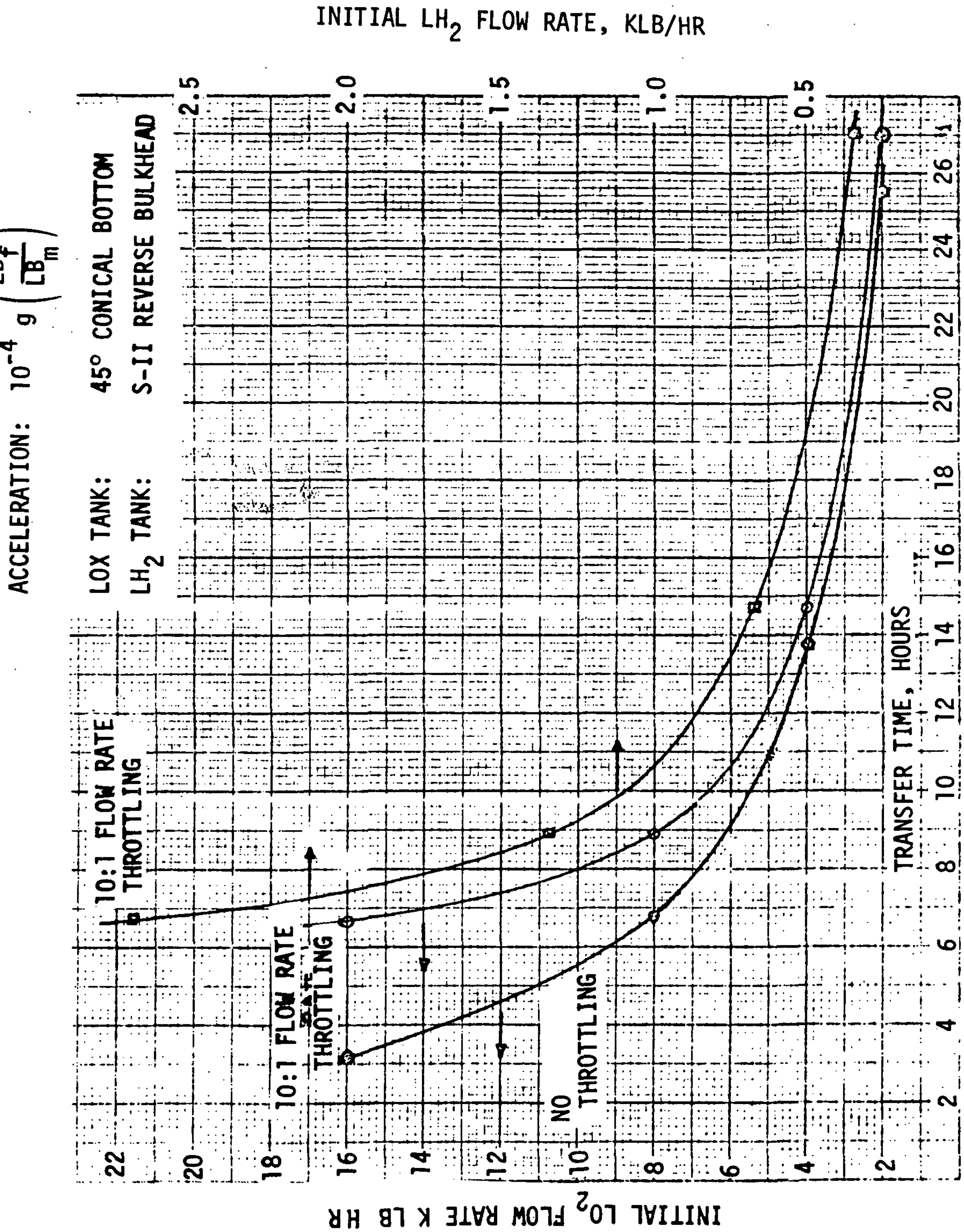


Figure 6.3.2-7 Initial Flow Rate Plotted Against Transfer Time

For the two classes of logistic modules of Figure 6.3.2-3, a parametric investigation was conducted yielding residual propellant as a function of transfer time and acceleration level. Results for the dual propellant type of module are presented in Figure 6.3.2-8 as curves of residual propellant versus transfer time relative to acceleration level. For $g = 10^{-3}$, 10^{-4}, and 10^{-5}, curves of both total propellant residuals ($LO_2 + LH_2$) and LO_2 residuals are presented; for $g = 10^{-1}$ and 10^{-2}, curves of total propellant are presented. The results show that at the lower acceleration levels, say $g = 10^{-3}$, residuals are quite sensitive to both transfer time and acceleration level, with residuals increasing slowly as both of these parameters decrease.

A similar study was conducted for the LH_2 logistic module for refueling the RNS (Figure 6.3.2-3). Again 10:1 flow rate throttling was utilized. The logistic module is cylindrical with 1.4:1 elliptical bulkhead. The outlet is covered with a pull through baffle of geometry and configuration similar to that shown in Figure 6.3.2-5 (c). Results are presented in Figure 6.3.2-9 as curves of LH_2 residual versus transfer time for a range of accelerations. For a case of practical interest, a transfer time equal to 10 hours with $g = 10^{-4}$, the residuals are less than one percent of the propellant transferred.

Although the presently preferred configuration of a dual propellant logistic module is of the type shown in Figure 6.3.2-3, this may not always be the case. Therefore, for completeness, results analogous to those of Figure 6.3.2-2 are presented in Figure 6.3.2-10 for tanks with an elliptical bulkhead and pull through baffle. As anticipated, residuals are higher than are those of Figure 6.3.2-8. Nevertheless, such tanks may be preferred for other reasons, such as structural weight or thermal control.

6.3.2.2 Optimization of Residual and Thruster Propellant Losses for Linear Acceleration

It was previously determined that 10:1 flow rate throttling was preferable to no throttling. However, preferred transfer time and acceleration level for refueling of the various user vehicles remain to be determined. In making this determination it is assumed that an auxiliary propulsion system is available to generate a continuous thrust level for linear acceleration liquid/vapor interface control during transfer. A specific impulse of 400 seconds is assumed based on a LO_2/LH_2 APS for the 1980's. It was seen in Figures 6.3.2-7 through 6.3.2-10 that residual propellant decreases as transfer time increases. APS propellant consumption increases linearly with transfer time. This is based on a continuous thrust requirement to obtain acceleration levels of 10^{-5}, 10^{-4}, and 10^{-3} for the tank to tug, tank to CIS, tank to RNS, and shuttle/tank attached to tug propellant transfer. Table 6.3.2-1 tabulates the thrust level required to accelerate the vehicles to the noted accelerations. For each of these transfer configurations propellant usage was determined and plotted on applicable graphs. The functional dependance with transfer time of these two losses (residuals and APS propellant) are opposed. Hence it is anticipated that there exists a transfer time for which the sum of the two losses is a minimum. This time is referred to as the preferred transfer time.

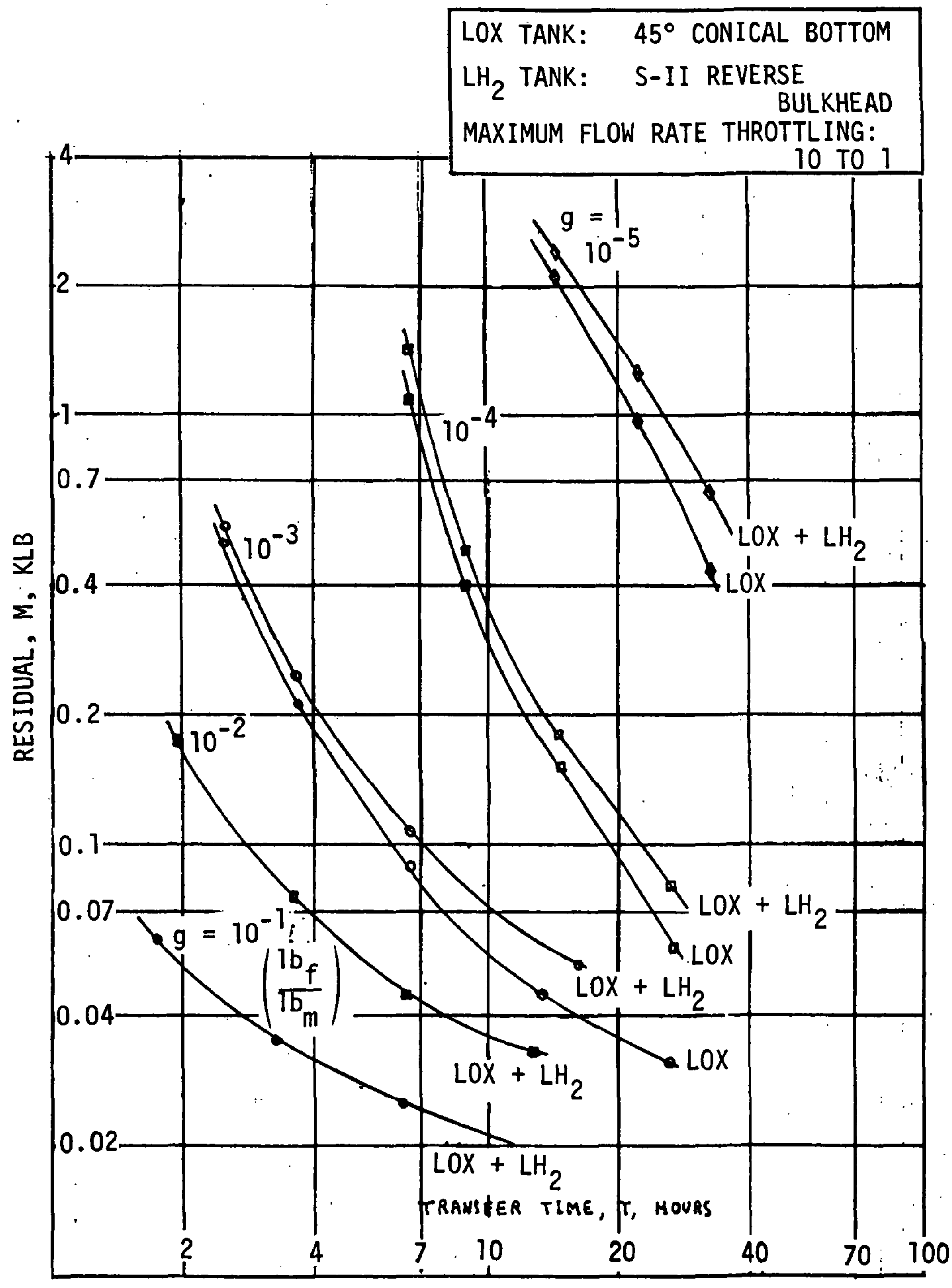


Figure 6.3.2-8 Residual as a Function of Transfer Time, with Flow Rate Throttling

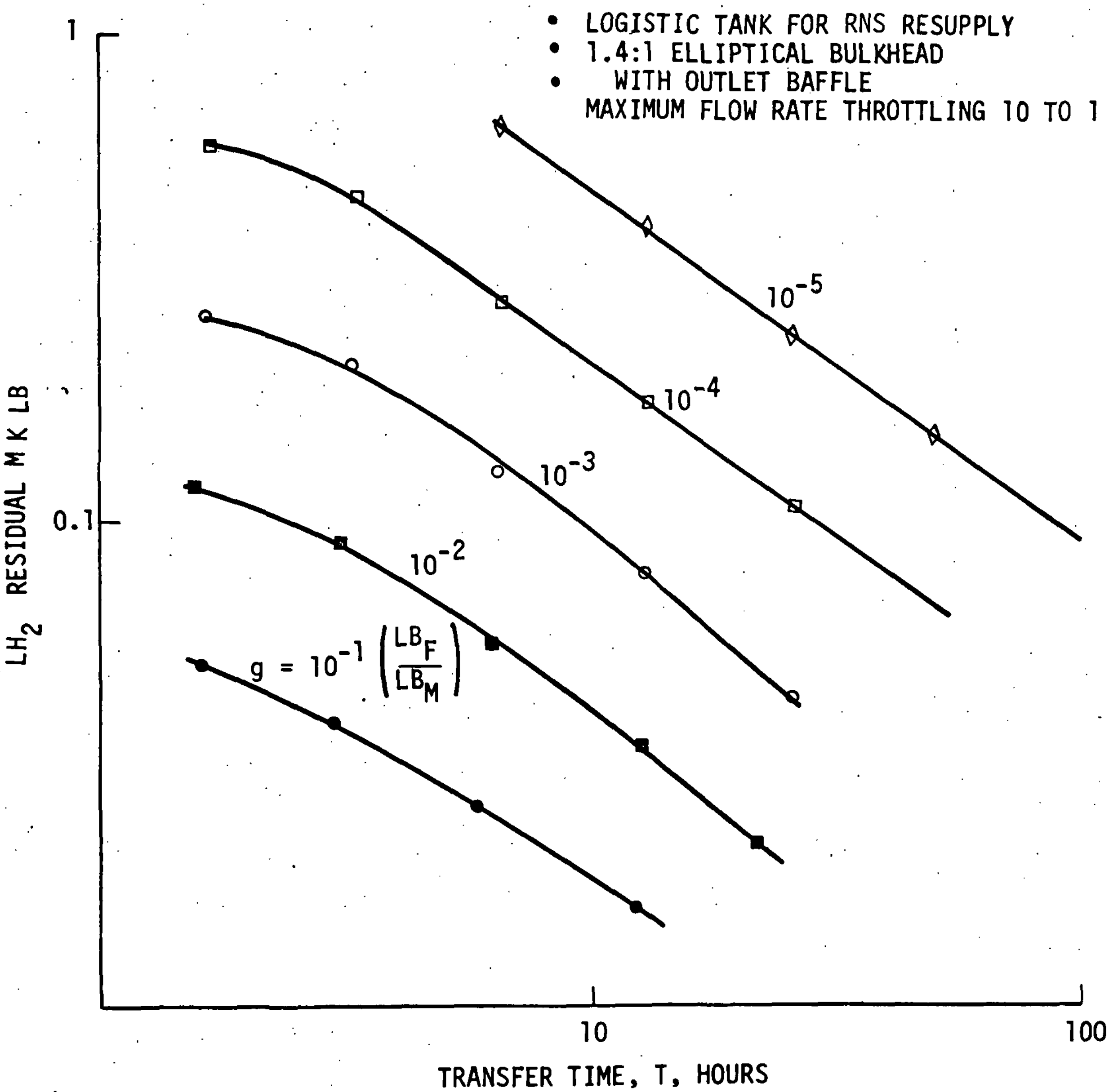


Figure 6.3.2-9 Residual as a Function of Transfer Time with Flow Rate Throttling – RNS

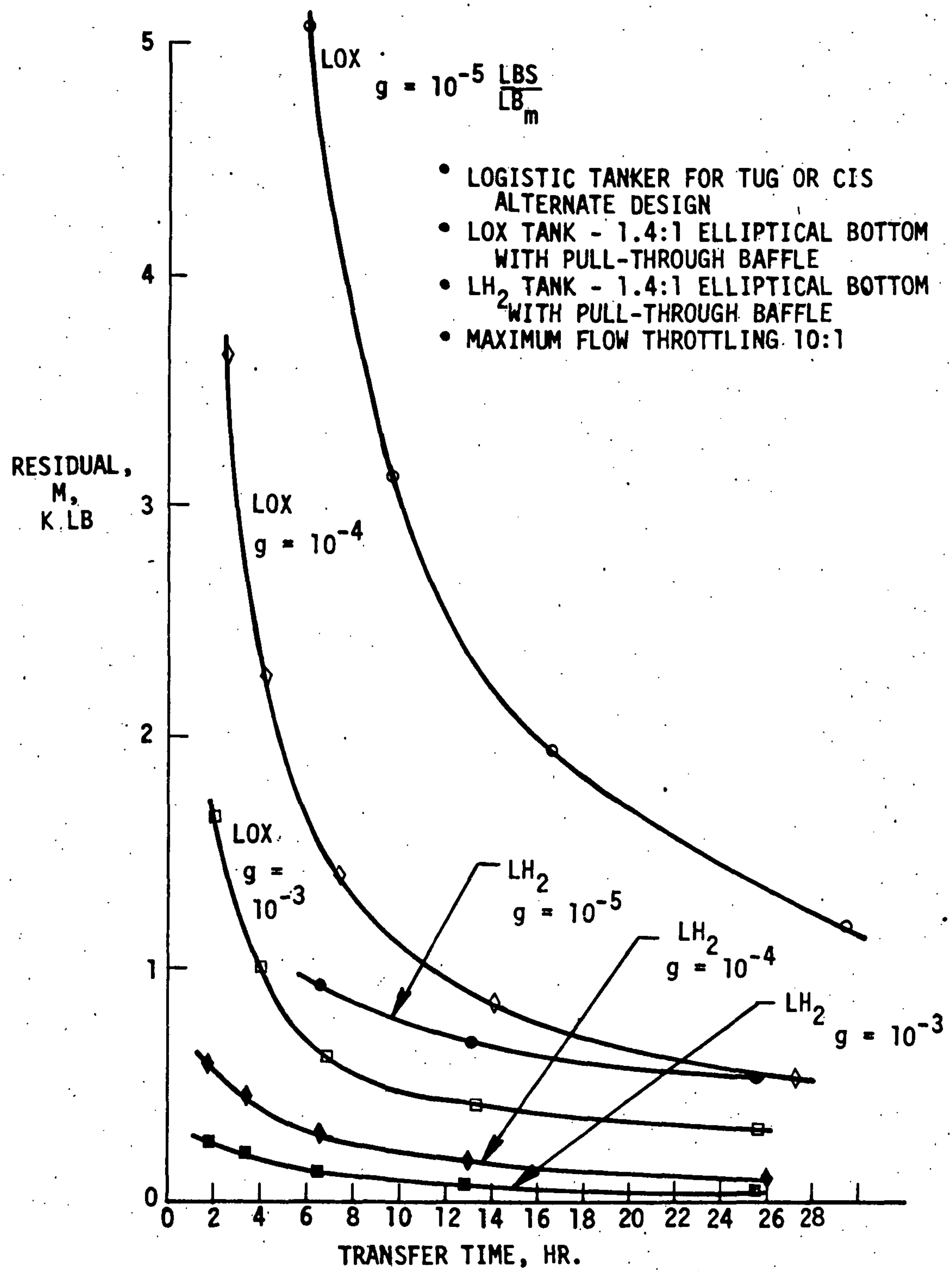


Figure 6.3.2-10 Residual as a Function of Transfer Time with Flow Throttling

Table 6.3.2-1. Thurst Levels Required for Acceleration Rates

ACCELERATION G	THRUST RANGE FOR ATTAINING ACCELERATION			
	TANK/TUG LB	TANK/CIS LB	TANK/RNS LB	TANK & SHUTTLE/TUG LB
10^{-5}	0.75	1.7 - 11	0.8 - 3.8	2.6
10^{-4}	7.5	17 - 110	8.0 - 38	26
10^{-3}	75	170 - 1100	80 - 380	260

Figures 6.3.2-11 through 6.3.2-13 present graphs of propellant losses versus transfer time for separate (space shuttle not attached) transfer from the logistic module to the tug. The logistic module and propellant are assumed to weight 65,000 pounds. The configuration of the LO_2 and LH_2 tanks in the logistic module is shown in Figure 6.3.2-3 (left hand sketch). In each case 10:1 flow throttling is used. As the acceleration level, g, decreases from 10^{-3} (Figure 6.3.2-11) to 10^{-5} (Figure 6.3.2-13), the preferred transfer time increases from 2.3 hours to a value greater than 40 hours. However, the total propellant loss decreases as acceleration level decreases. Results for $g = 10^{-4}$ (Figure 6.3.2-12) provide the best compromise, yielding moderate propellant losses (1.8 percent of 60,000 pounds) for a reasonable transfer time (10 hours).

Figures 6.3.2-14 and 6.3.2-15 present results for the same type of trade study. However, for this case the logistic module remains attached to the space shuttle during the transfer. Therefore, the mass which must be accelerated is greater than was the case for separate transfer. Results are presented for only $g = 10^{-4}$ and 10^{-5}, as losses at an acceleration level of 10^{-3} ar extremely large. Propellant losses are higher for attached transfer, which is one of the reasons that this technique was not selected for further study.

Figures 6.3.2-16 and 6.3.2-17 present propellant loss trade studies for propellant transfer from the logistic module to the CIS. The configuration of the tanks in the logistic module is shown in the center sketch of Figure 6.3.2-3. The problem of transfer to the CIS is considerably more complex than was transfer to the tug. Nineteen transfers are required to load the CIS, and thus the combined mass of the logistic module and the CIS with propellant varies by a factor of approximately six from the first to the last transfer. Figure 6.3.2-16 present the trades for the initial transfer at $g = 10^{-4}$. For the final transfer, only the APS propellant requirements are shown. At $g = 10^{-4}$ the sum of APS propellant and residual for the final transfer is clearly prohibitive. At an acceleration level of 10^{-5} total propellant losses are reduced, as shown in Figure 6.3.2-17. Here results for the initial and final transfer are shown.

If the acceleration level is maintained constant for each transfer, the thrust level generated by the APS must be increased with each successive transfer. To avoid the propulsion requirement for a variable thrust level, a single engine system, providing a single thrust level is recommended for each transfer. A thrust level which provides an acceleration range from approximately 10^{-4} for the initial transfer to 10^{-5} for the final transfer appears to represent the best compromise between propellant losses on the one hand and degree of interface control and transfer time on the other hand. For this single thrust level the total propellant loss for the initial transfer is given in Figure 6.3.2-16 while the total propellant loss for the final transfer (full CIS) is given in Figure 6.3.2-17.

Trade studies were also conducted on refueling of the RNS. Ten to one flow rate throttling was used as before, but the logistic module now has a single LH_2 tank with 1.4 to 1 elliptical bulkhead with baffle (shown in Figure 6.3.2-2). Results are generally similar to those obtained for the CIS, with lower propellant losses in this case because of the single tank with low density propellant. Results are shown in Figures 6.3.2-18, 6.3.2-19, and 6.3.2-20 for acceleration

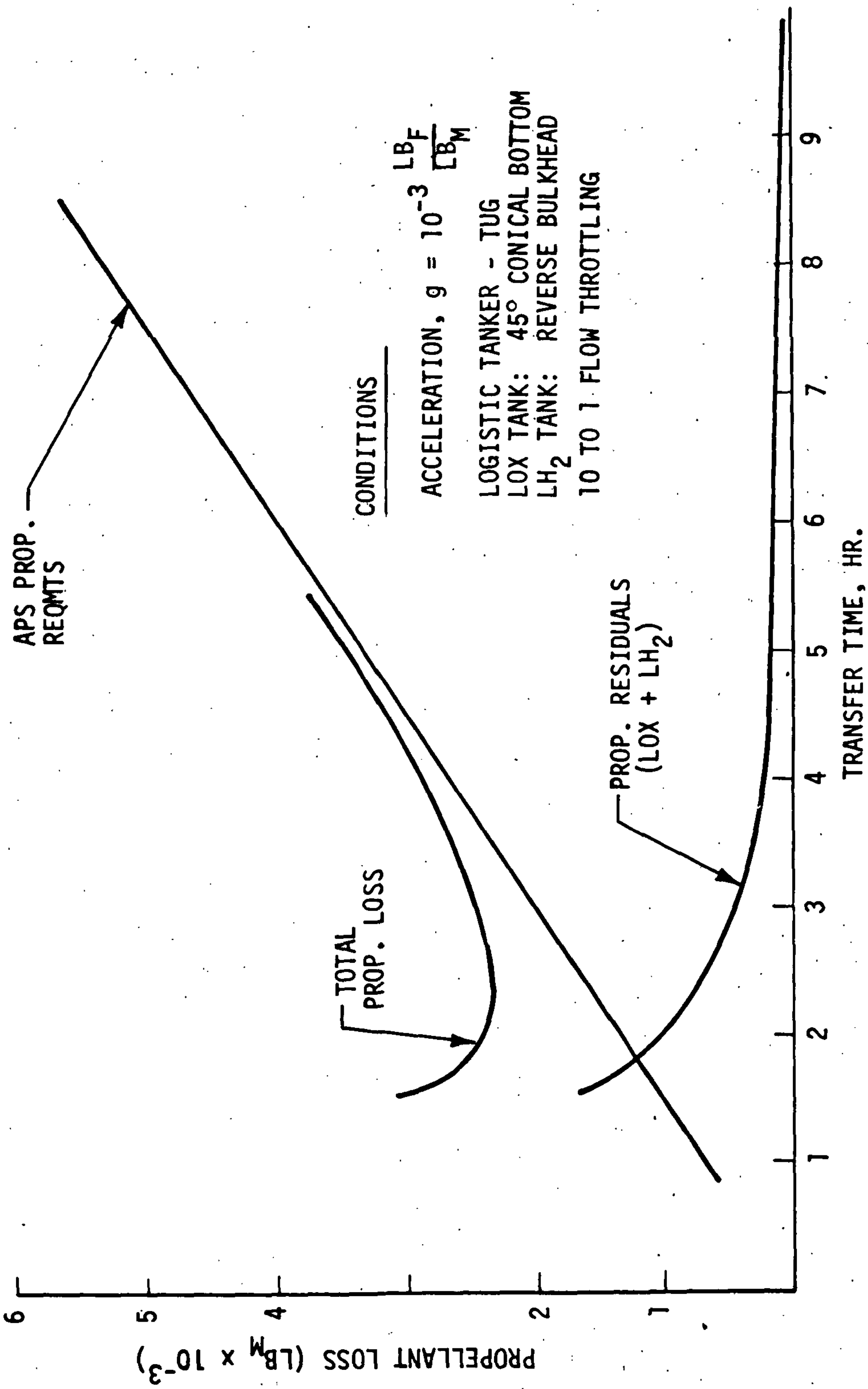


Figure 6.3.2-11 Propellant Loss Trade Study, g = 10^{-3}
Logistic Module - Tug

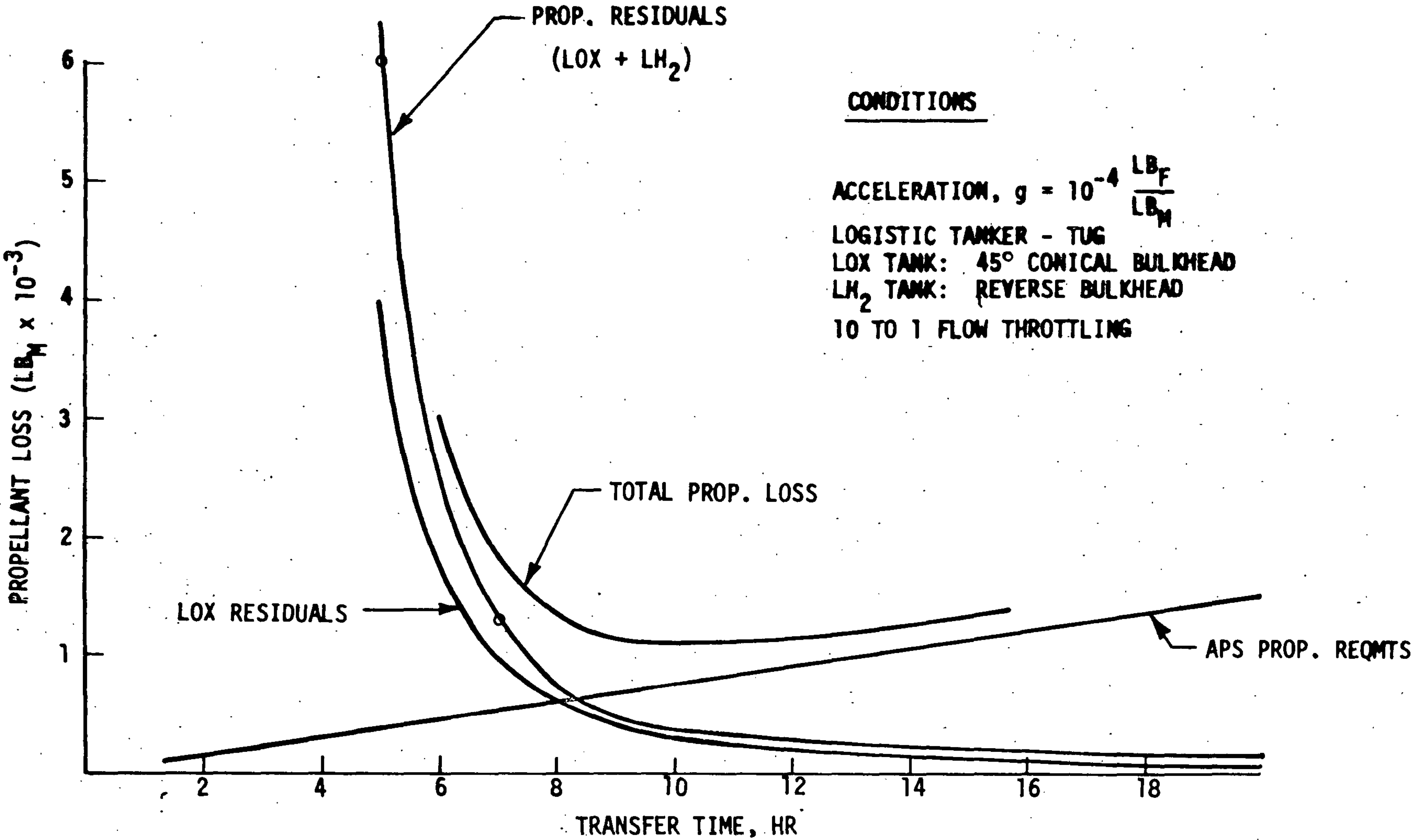


Figure 6.3.2-12 Propellant Loss Trade Study, $g = 10^{-4}$
Logistic Module - Tug

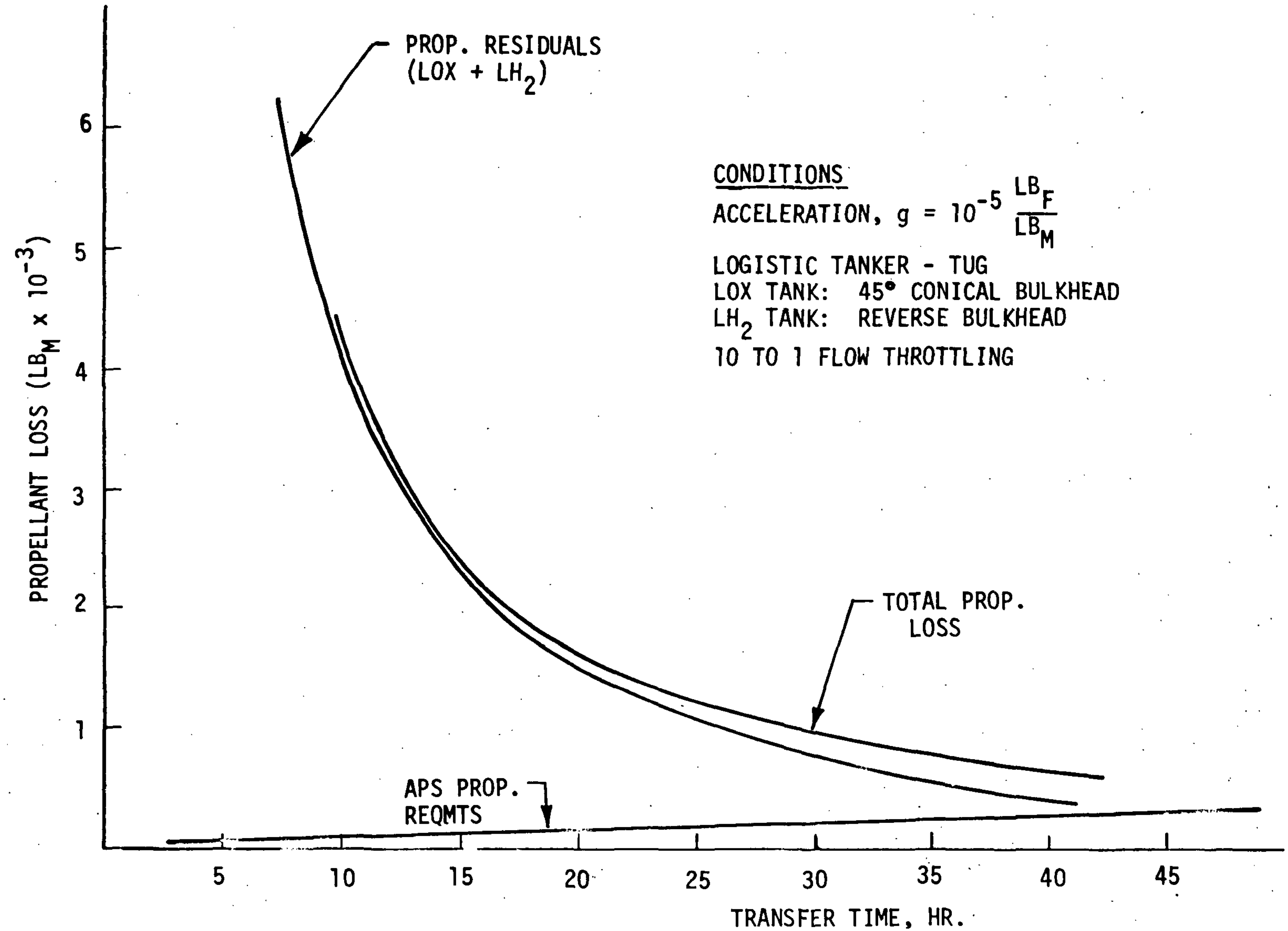


Figure 6.3.2-13Propellant Loss Trade Study $g = 10^{-5}$
Logistic Module - Tug

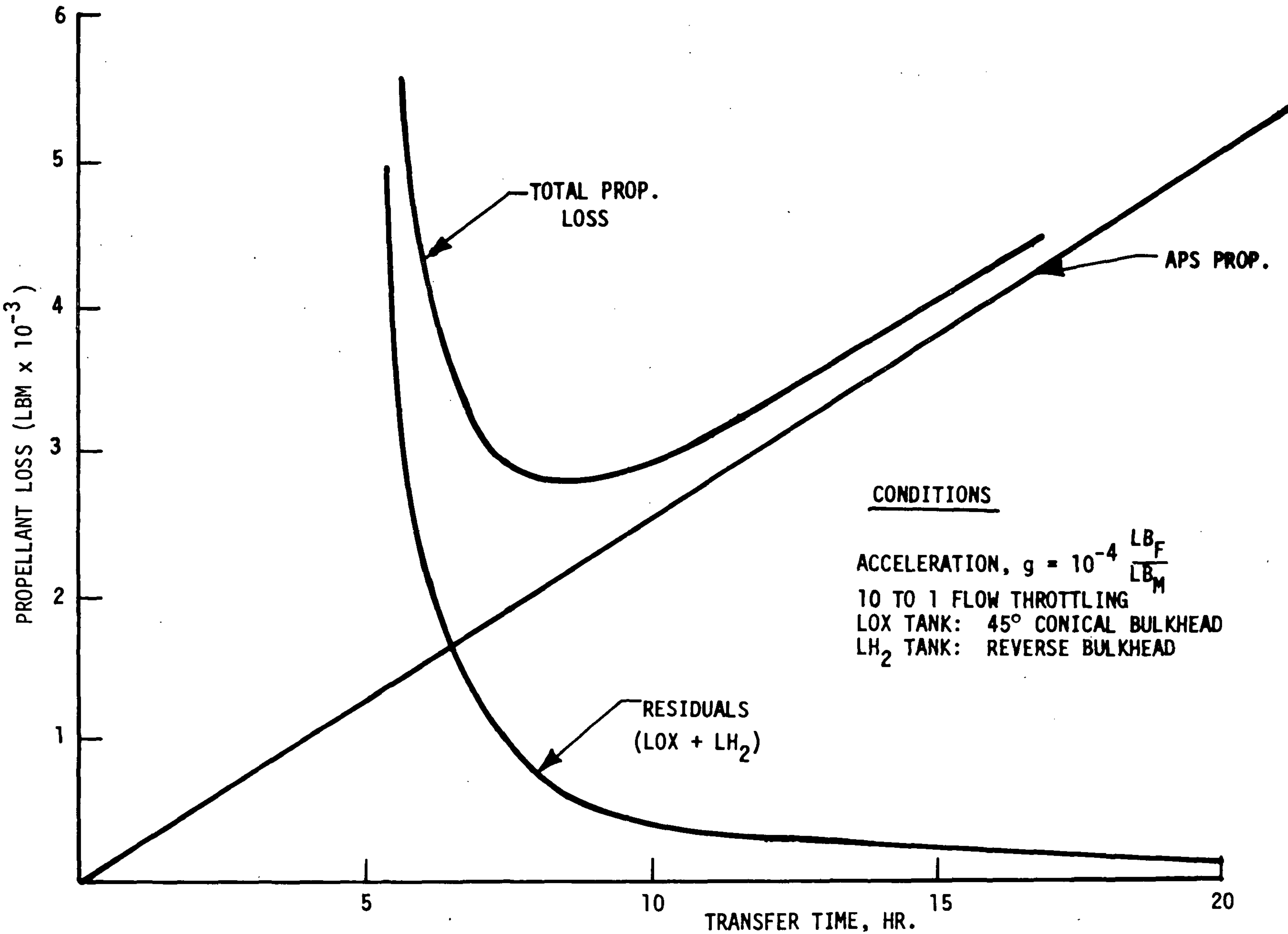


Figure 6.3.2-14 Propellant Loss Trade Study, g = 10^{-4}
Shuttle + Module + Tug

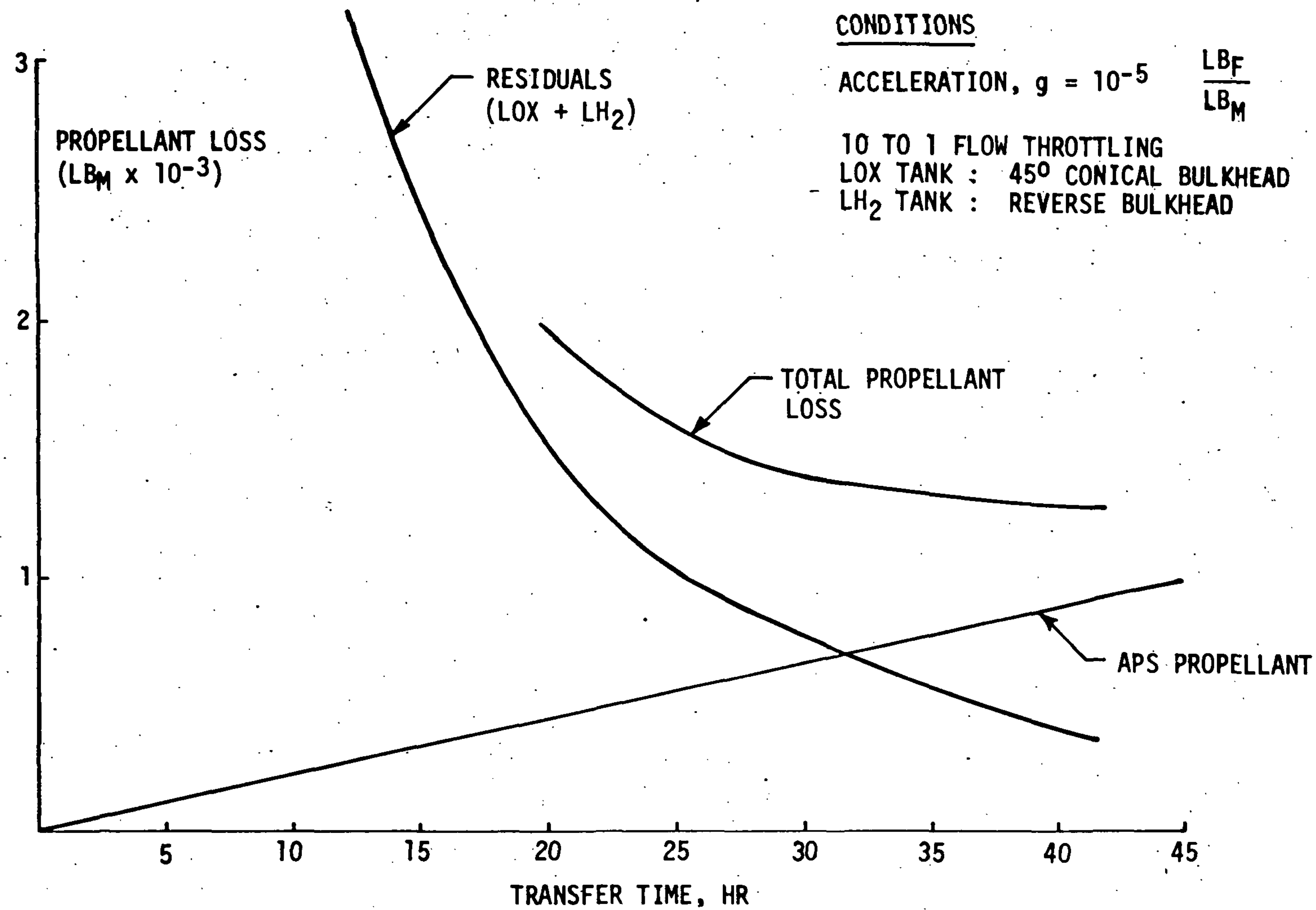


Figure 6.3.2-15 Propellant Loss Trade Study $g = 10^{-5}$
Shuttle + Module + Tug

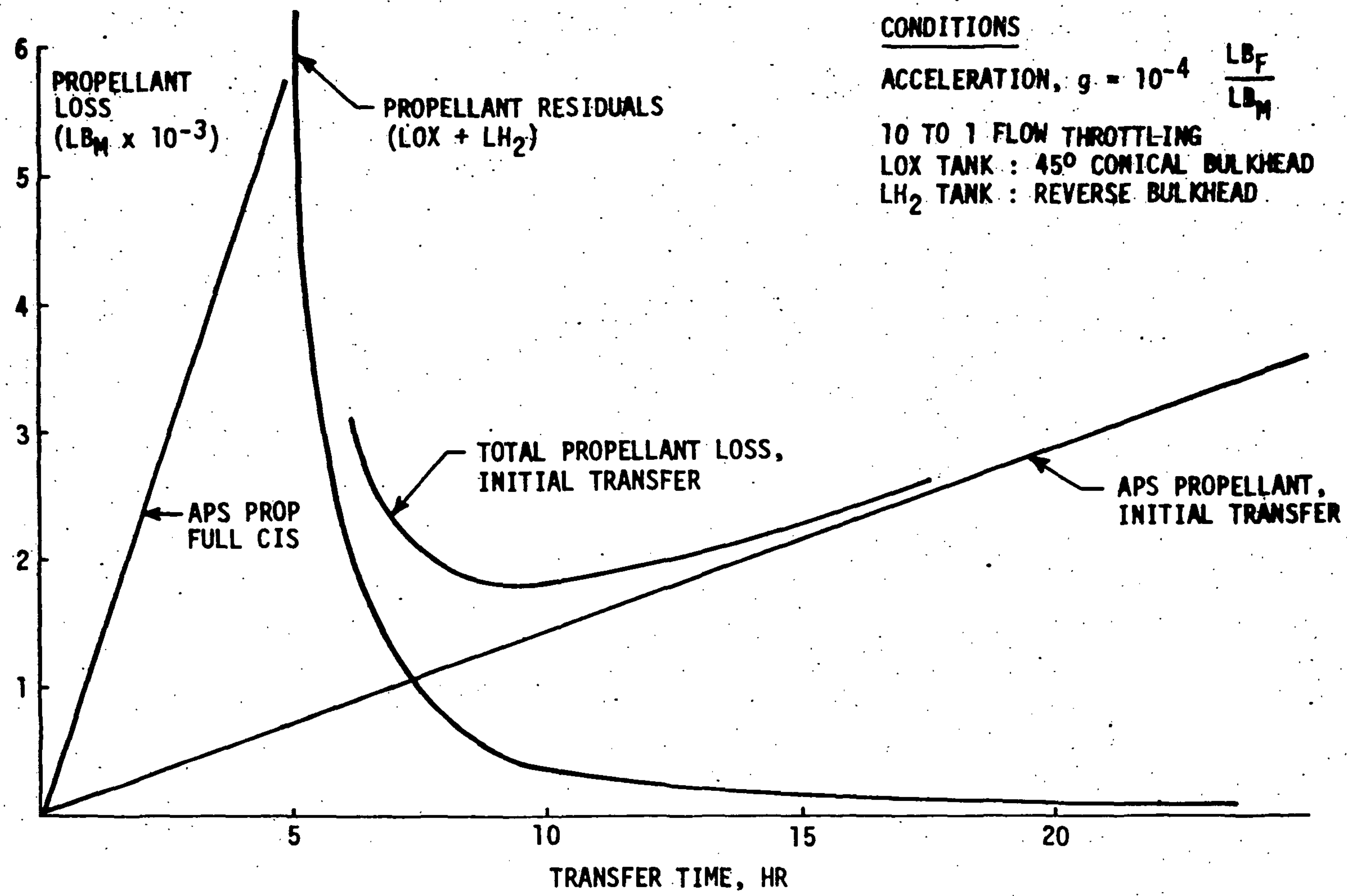


Figure 6.3.2-16 Propellant Loss Trade Study $g = 10^{-4}$
Logistic Module-CIS

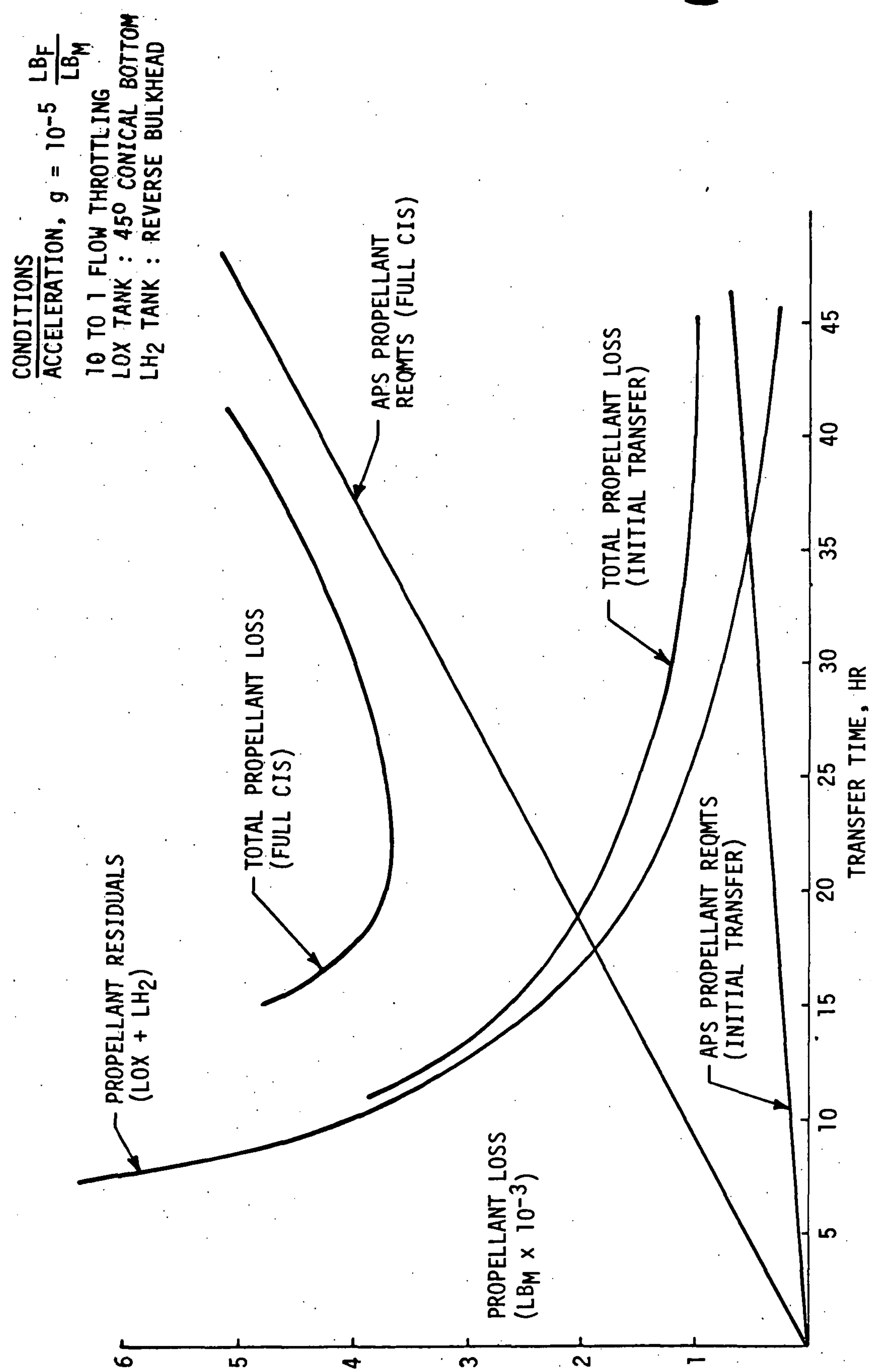


Figure 6.3.2-17 Propellant Loss Trade Study $g=10^{-5}$
Logistics Module - CIS

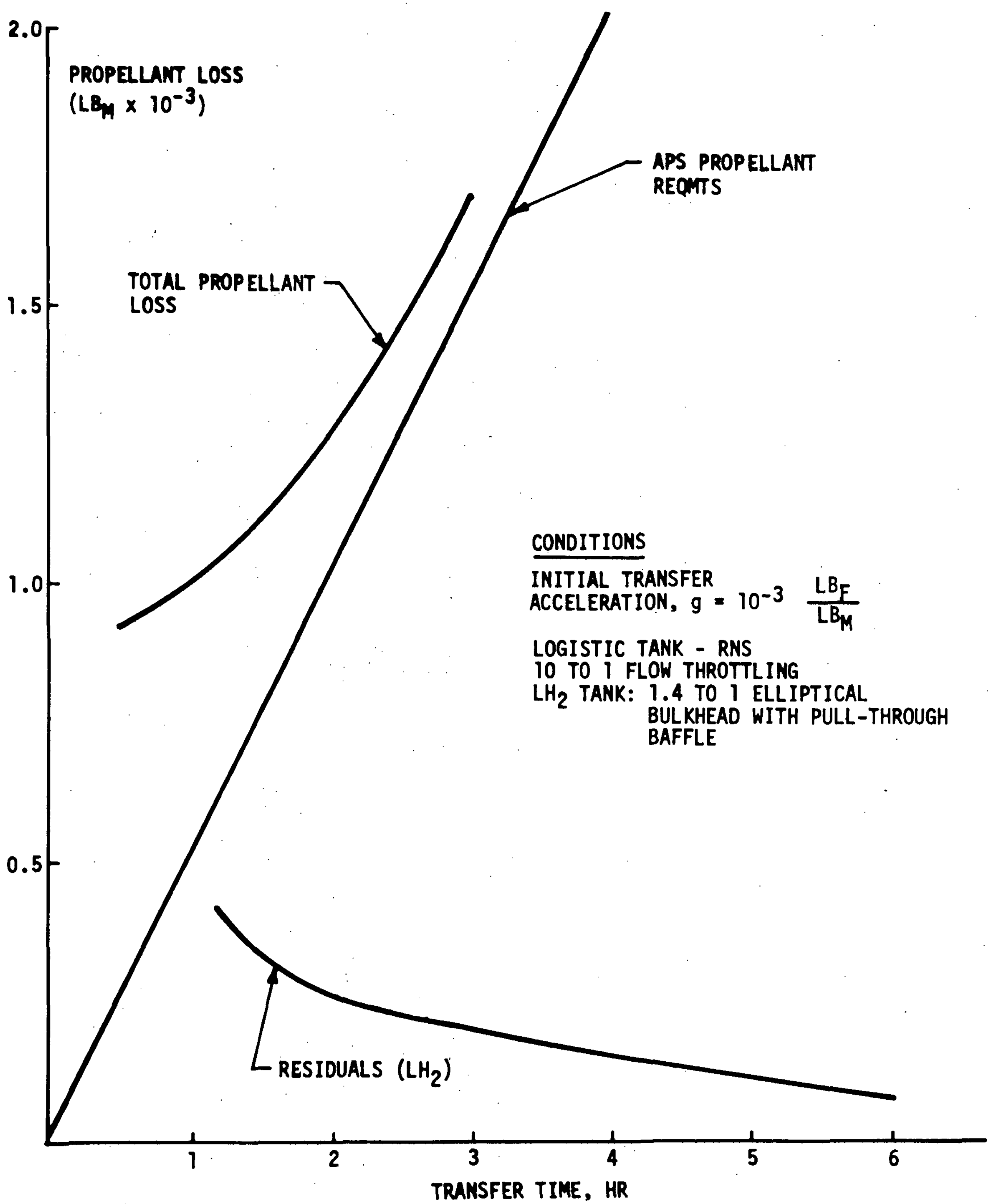


Figure 6.3.2-18 Propellant Loss Trade Study $g=10^{-3}$ Logistic Module - RNS

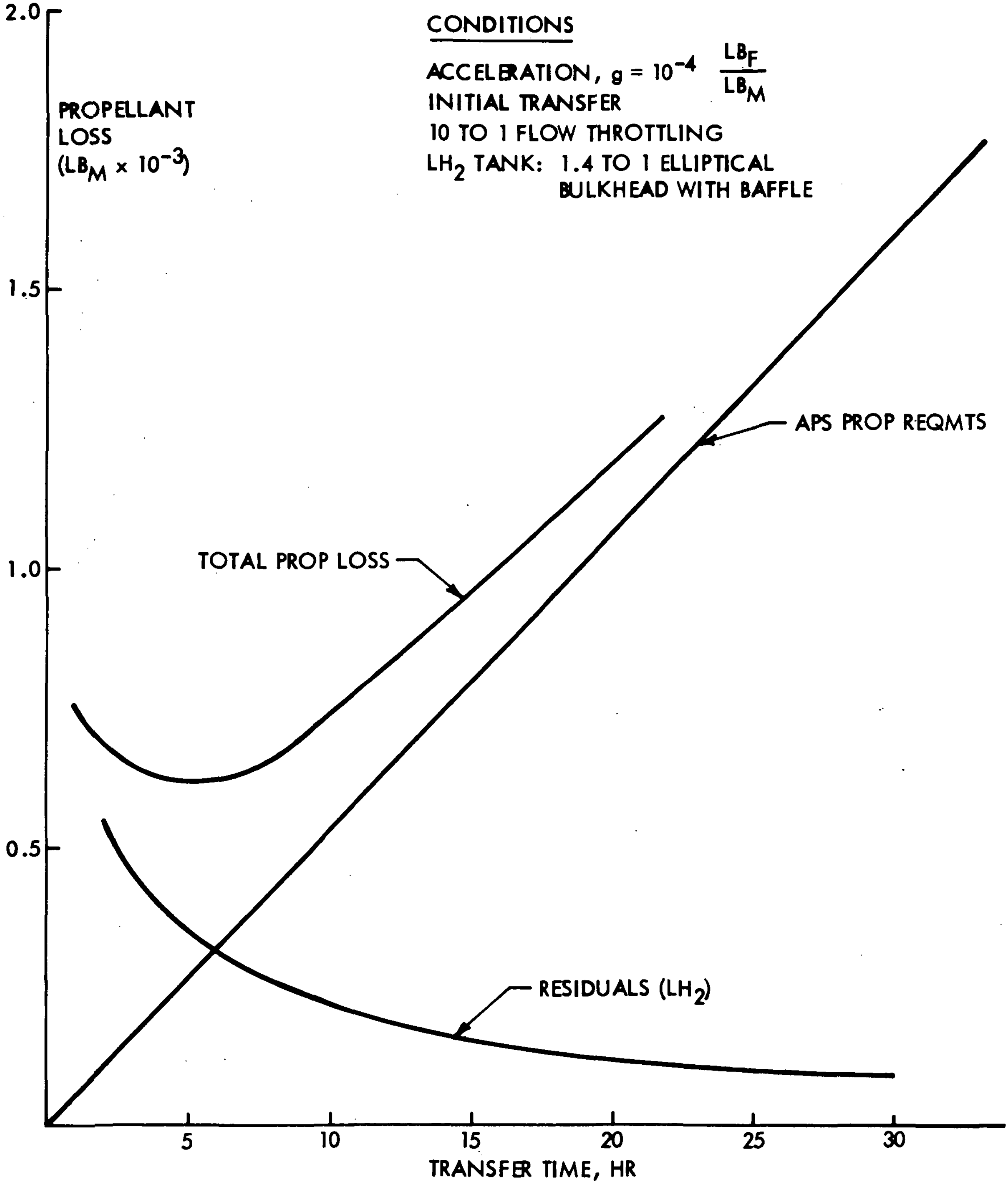


Figure 6.3.2-19 Propellant Loss Trade Study g = 10^{-4}
Logistic Module - RNS

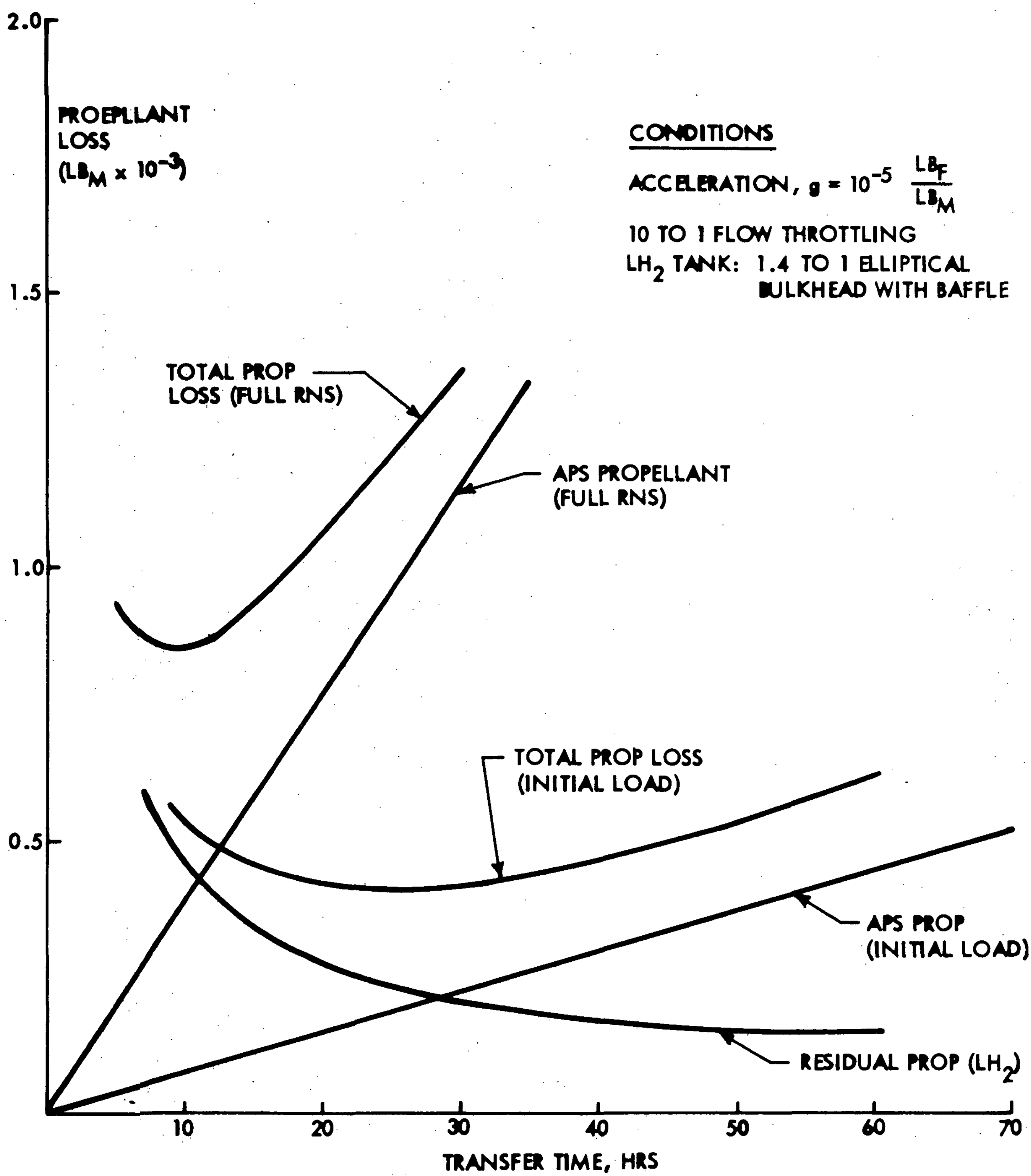


Figure 6.3.2-20 Propellant Loss Trade Study $g = 10^{-5}$
Logistic Module - RNS

levels of 10^{-3}, 10^{-4}, and 10^{-5}, respectively. Figures 6.3.2-18 and 6.3.2-19 show data for the initial transfer.

As with the CIS, it is recommended that a single engine system and thrust level be used for all of the propellant transfers. Acceleration level will vary from approximately 10^{-4} during the first transfer to 10^{-5} during the last transfer. From Figure 6.3.2-19 it is seen that for the first transfer, propellant losses are 700 pounds for the preferred transfer time of 5 hours, while from Figure 6.3.2-20 it is seen that for the last transfer propellant losses are 850 pounds for the preferred transfer time of 10 hours.

Effect of Modulated Acceleration Level During Transfer

As an alternative to flow rate throttling, an increase in acceleration level at incipient pull through will yield decreased residual. This approach does not increase transfer time, but does increase APS propellant consumption, and moreover, requires the capability to achieve a higher APS thrust level.

This latter requirement necessitates either variable thrust APS engines or additional engines to augment the initial thrust level. It is envisioned that the additional thrust could be preprogrammed or activated by liquid level sensors during the transfer.

This technique was studied for transfer to the tug, CIS and RNS. Results in general for this approach were poor being worse for the CIS and RNS than for the tug. Results for the tug are presented in Figures 6.3.2-21 through 6.3.2-23 where results are shown for both constant acceleration and a change to a higher acceleration at incipient pull through. The logistic module was identical to the one previously considered for flow rate throttling. For an initial acceleration level of 10^{-3}, Figure 6.3.2-21 shows an advantage to increasing the acceleration level to a value of 5×10^{-3} at incipient pull through. Both propellant losses and preferred transfer time were reduced. Similar, but improved, results were obtained for initial acceleration of 10^{-4} and 10^{-5} and a final acceleration of 5×10^{-3} as shown in Figures 6.3.2-22 and 6.3.2-23. A final acceleration level of 10^{-1} was also considered, but propellant losses were greater than for the final acceleration level of 5×10^{-3} as shown in Figure 6.3.2-22. Results were considerably better than those obtained for constant acceleration levels. However, comparison with Figure 6.3.2-22 shows that flow throttling is comparable to changing acceleration level as regards to propellant losses.

Flow throttling is preferred to changing the acceleration level for two primary reasons: (a) flow throttling is efficient as regards propellant losses over a wide range of parametric conditions and user vehicles, while change in acceleration level is efficient for only limited ranges of parameters and for small user vehicles; and (b) development of variable thrust engines or auxiliary engine systems would result in greater cost and development risk than the development of a modulating flow control valve.

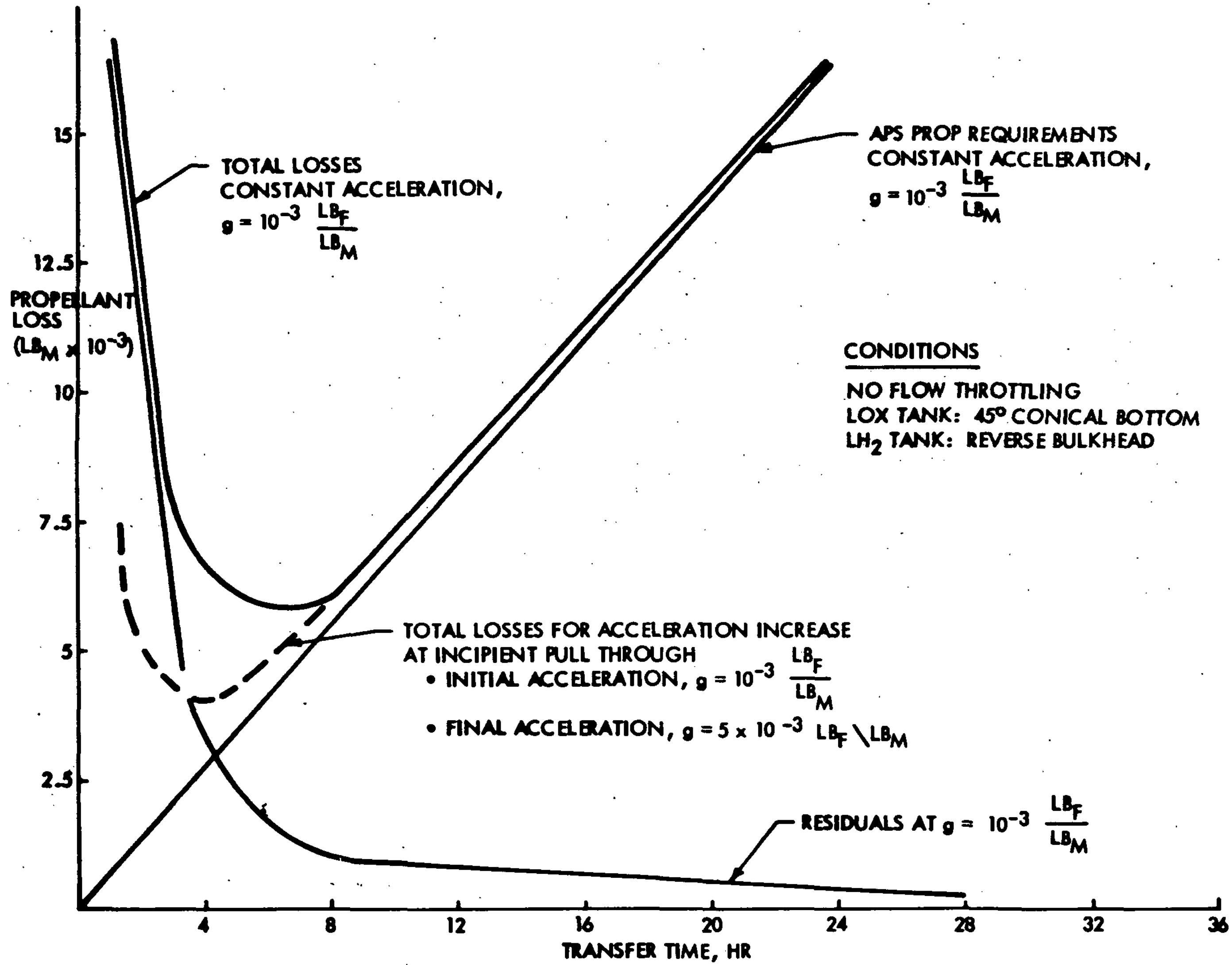


Figure 6.3.2-21 Effect of Acceleration Increase During Transfer on Propellant Losses Logistic Module - Tug $g = 10^{-3}$

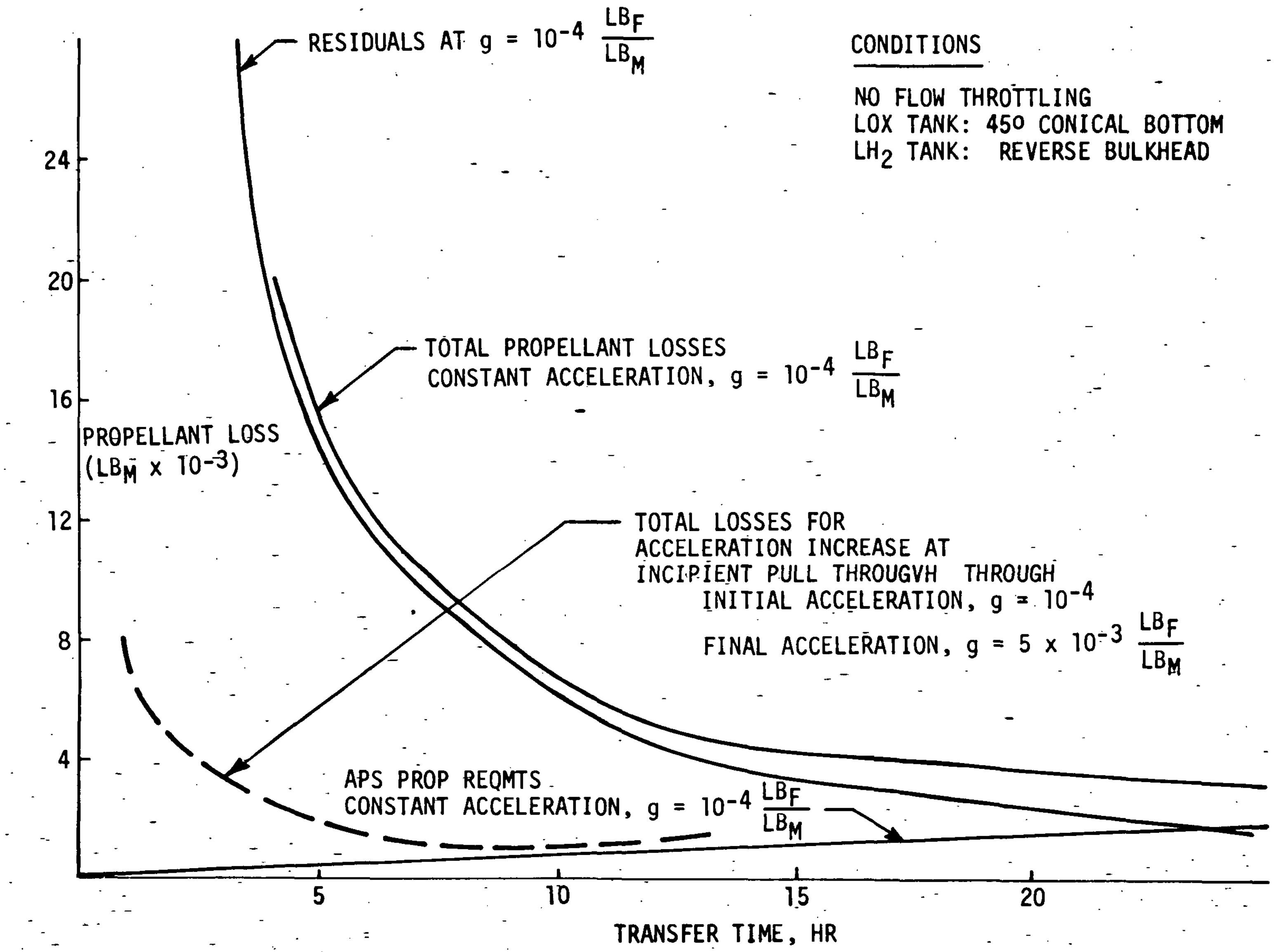


Figure 6.3.2-22 Effect of Acceleration Increase During Transfer on Propellant Losses. Logistic Module - Tug $g=10^{-4}$

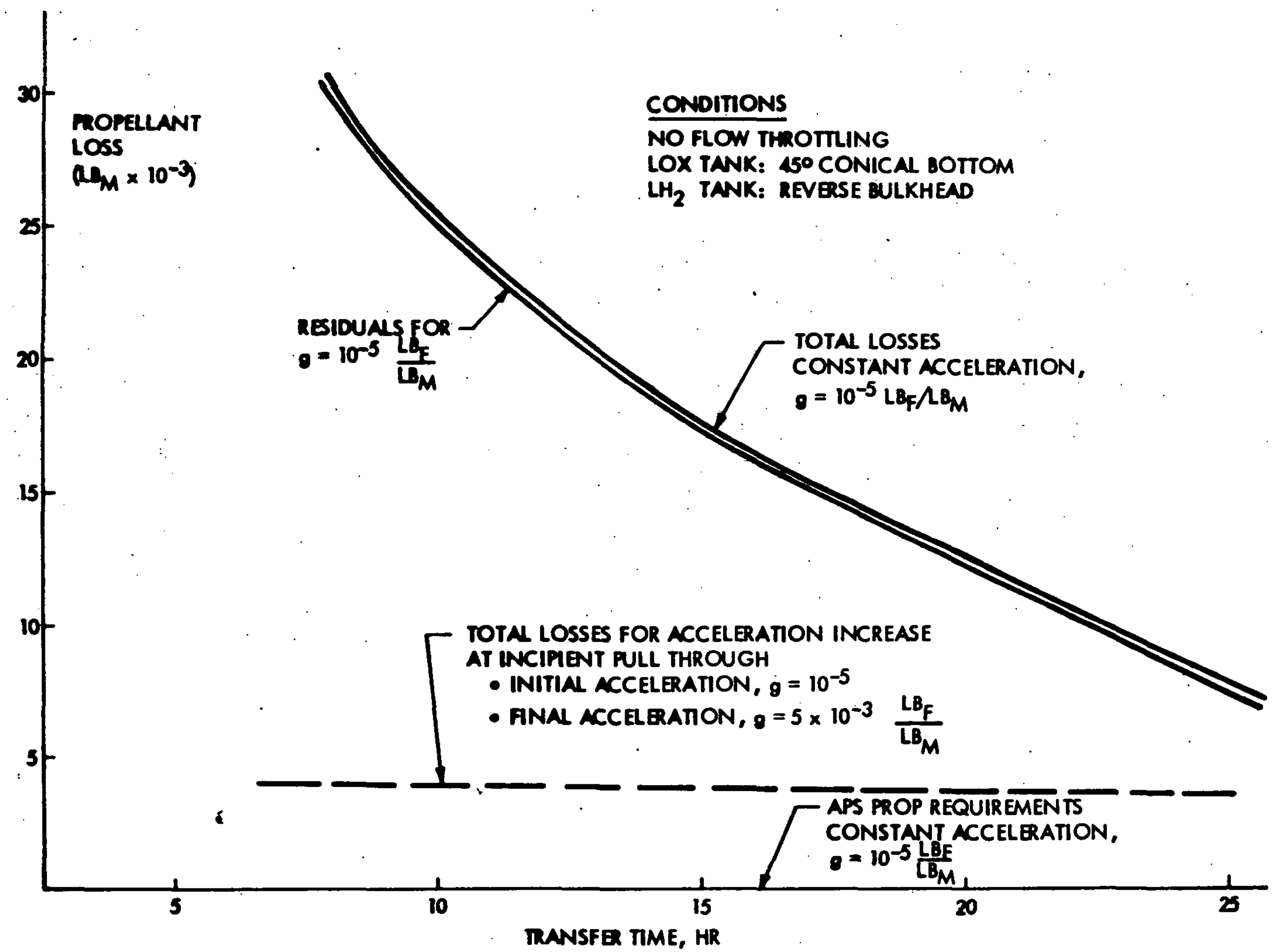


Figure 6.3.2-23 Effect of Acceleration Increase During Transfer on Propellant Losses Logistic Module - Tug $g = 10^{-5}$

6.3.2.3 Optimization of Residual and Thruster Propellant Losses for Radial Acceleration

Analyses were also conducted to determine the minimum propellant loss associated with providing the necessary acceleration in the rotational transfer mode. Residual data from Section 6.3.2.1 were used for this analysis. For the rotational mode the analysis covered a determination of rotation rates to provide the necessary acceleration in the feedout propellant, propellant consumption to induce rotation, propellant residuals, and acceleration levels to minimize the sum of residual and spin thruster propellant consumption.

Certain physical considerations must be noted and design study ground rules defined in connection with the problem of inducing rotation to assemblies in earth orbit. These are summarized as follows:

a. For most cases studied, thrust couples are provided on each side of the docking interface to minimize bending and translation insofar as practical. All thruster units were assumed to be constant thrust.

b. For propellant consumption calculations, thruster propellant is assumed to be the same as that utilized for other propulsion on the spacecraft where it is mounted, unless otherwise noted.

c. No additional spin impulse has been used during the propellant transfer period. In general, the propellant transfer increases the assembly moment of inertia thus slowing the rotation rate. Initial spin up velocities have been defined to account for this effect and to provide proper acceleration at the end of each transfer cycle.

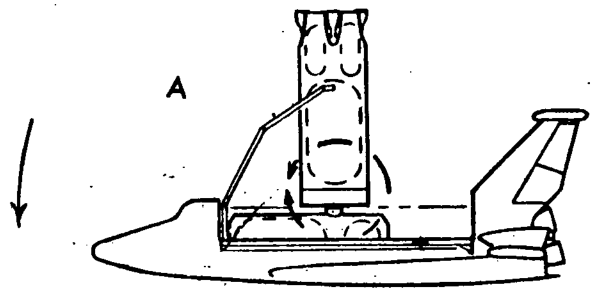
d. Current baseline design auxiliary propulsion system (APS) thrusters were included as a configuration option to perform the rotational acceleration function where practical.

e. In the cases of the CIS and RNS where numerous propellant loads are required to fill the user vehicle, the curves present the average servicing condition (one-half the total of the first and last transfers.

The docked vehicle configurations considered for rotational analysis are shown and briefly described below. For all the schematics, rotation is in the plane of the paper.

Configuration A

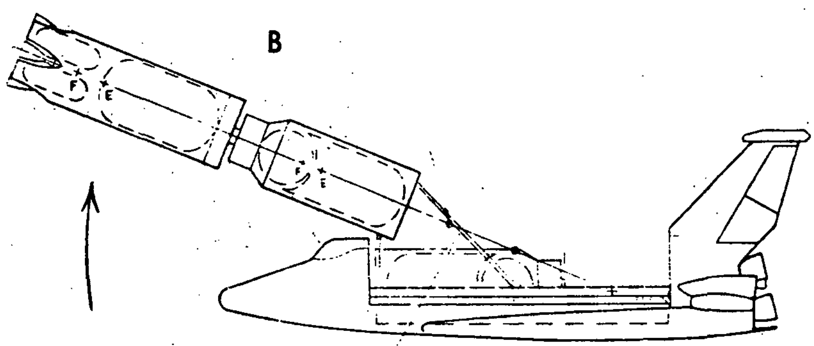
Tug serviced from a module mounted in the shuttle orbiter bay with the tug major axis normal to the orbiter axis. Rotation in pitch direction as shown.



Configuration B

Tug serviced from a module docked end-to-end and supported above the shuttle orbiter crew compartment with the tug and module major axis pointed through the orbiter cg. Rotation in pitch direction as shown.

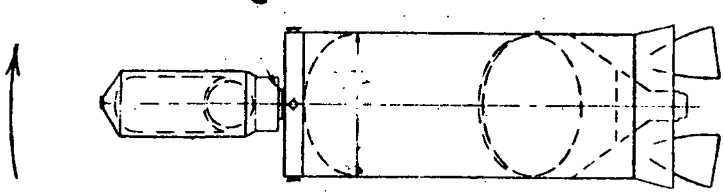
B



Configuration C

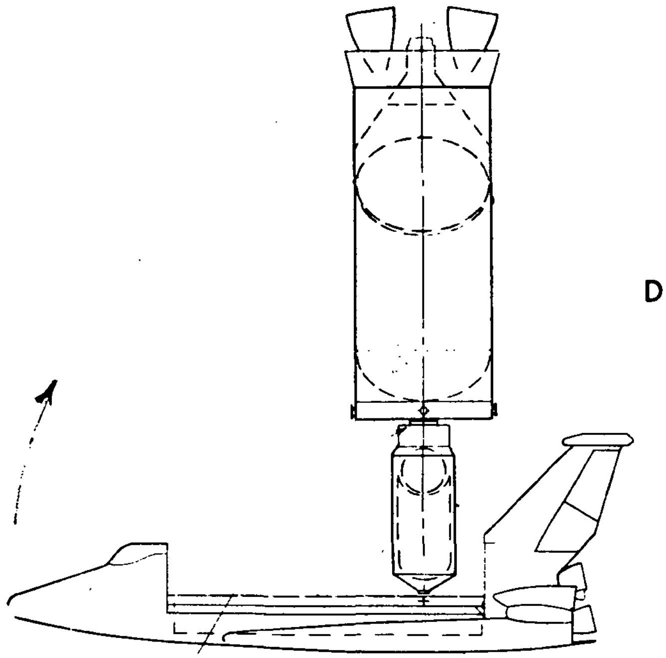
CIS serviced from module docked end-to-end as shown. Rotation on principal axis (end-over-end).

C



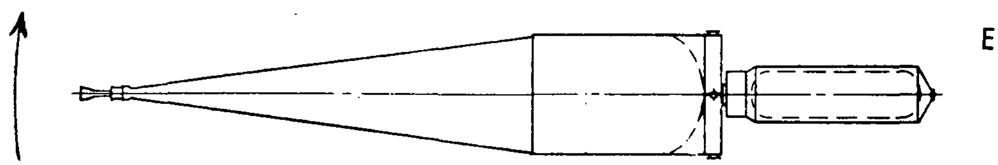
Configuration D

CIS serviced from a module mounted in the shuttle orbiter with the major axis pointing through the orbiter cg and normal to the major axis. Rotation in pitch direction as shown.



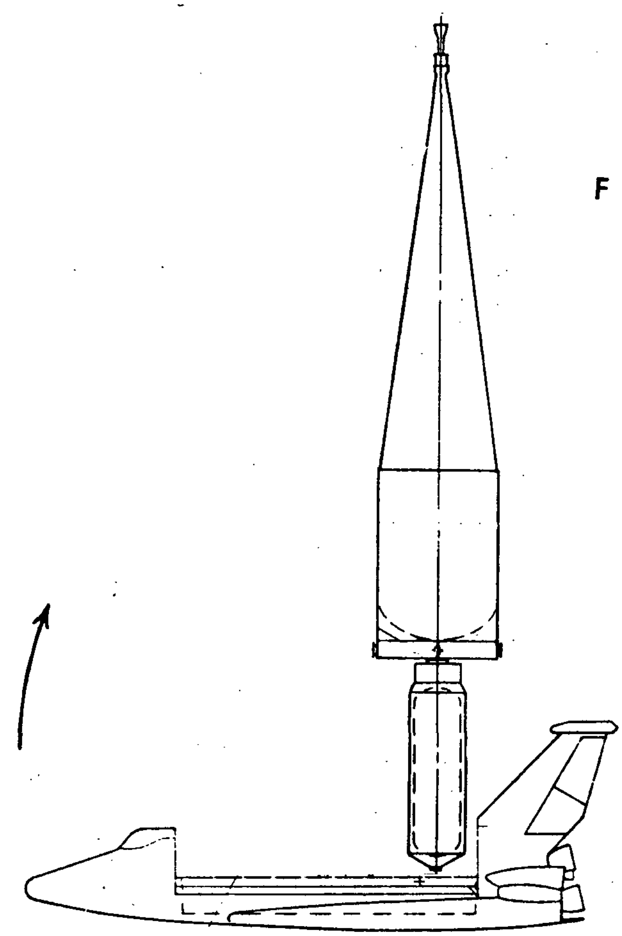
Configuration E

RNS serviced from module docked end-to-end. Rotation on principal axis (end-to-end) as shown.



Configuration F

RNS serviced from module mounted in shuttle orbiter with major axis pointing through orbiter cg and normal to main axis. Rotation in pitch direction as shown.



A wide range of engine arrangements to achieve combined vehicle pitch rotation is possible. In general, it is desired to utilize existing engines where available, to provide for balanced forces about the vehicle mating interface where practical, and to utilize the propellant available in the stage. In addition, where the option existed, the engines were assumed mounted on the ground-based vehicle to afford engine replacement or maintenance after each mission. No attempt was made to fine tune the engine balancing since it will be affected by changing vehicle weight, moments of inertia, and cg. Five thruster options were studied for Configuration A. These options are summarized in Table 6.3.2-2. Four of the options employ existing baseline design APS engines. Three options were studied for Configuration B and are summarized in Table 6.3.2-3. For Configuration B, all options provide for balanced momemts by adjusting lever arms or thrust level. Two options utilize existing baseline design APS engines.

A preliminary evaluation of propellant losses associated with the rotational mode of providing artificial g for propellant transfer from the logistics module was conducted on the various tug, CIS, and RNS mated configurations.

Table 6.3.2- 2
Configuration A
1 x 10^{-3} g For Propellant Settling
Spin Up Only

OPTION	THRUSTER DESCRIPTION				ON TIME	PROPELLENT USED	REMARKS
	ENGINES	THRUST	LEVER ARM	VEH MOUNTED			
1	1 N_2H_4	950#	61.5	Orbiter	10.9s	45.0#	Unbalanced Moments Existing Engine
2	2 GOX/GH_2	20	36.7	TUG	434	49.7	Unbalanced Moments Existing Engines
3	2 GOX/GH_2	20	36.7	TUG	10.7	1.2	Unbalanced Moments Existing Engines
	1 N_2H_4	950	61.5	Orbiter		44.0	
4	2 GOX/GH_2	20	36.7	TUG	162.5	18.6	New Engines on Orbiter - Existing Engines on TUG
	2 N_2H_4	20	61.5	Orbiter		28.3	
5	2 GOX/GH_2	2.5	36.7	TUG	1300	18.6	New Engines on Orbiter and TUG
	2 N_2H_4	2.5	61.5	Orbiter		28.3	

Table 6.3.2-3 Configuration B
1 x 10^{-3} g for Propellant Settling

OPTION	THRUSTER DESCRIPTION				ON TIME	PROPELLANT USED	REMARKS
	ENGINES TYPE	THRUST	LEVER ARM	VEH MOUNTED			
1	UP						
	2 GOX/GH_2	2.5#	37.3'	Tanker	2210s	79.8#	Balanced Moments
	2 N_2H_4	2.5	37.3	Orbiter			New Engines on Tanker & Orbiter
	Down						
	2 GOX/GH_2	2.5	48.0	Tanker	1695	58.5	
	2 N_2H_4	2.5	48.0	Orbiter			
2	UP						
	2 GOX/GH_2	20#	85.75'	TUG	120s	61.72#	Balanced Moments
	2 N_2H_4	46	37.3	Orbiter			New Engines on Orbiter & Existing Engines on TUG
	Down						
	2 GOX/GH_2	20	75.75	TUG	134.5	53.9	
	2 N_2H_4	32.9	46	Orbiter			
3	UP						
	2 GOX/GH_2	401#	85.75'	TUG	6.0s	38.4#	Balanced Moments
	1 N_2H_4	950	72.5	Orbiter			New Engines on TUG & Existing Engines on Orbiter
	Down						
	2 GOX/GH_2	500	75.75	TUG	5.4	37.7	
	1 N_2H_4	950	79.7	Orbiter			

These losses can be minimized by determining the optimum rotation speed and resulting acceleration level. APS consumption for vehicle spin up and spin down will increase and logistic module residuals will decrease with increasing spacecraft rotation rates. A summary of the minimum loss values for each of the configurations studied is presented in the following paragraphs.

For the 10-hour transfer period, the optimum acceleration for Configuration A was determined to be at approximately 2×10^{-3} g which results in losses totaling 233 pounds of propellant or 0.39 percent of the 60,000-pound propellant load. This can be seen in Figure 6.3.2-24. The thruster configuration was assumed to be Option 5 of Table 6.3.2-2, utilizing thrusters mounted near existing engines at tug station 143 and on the shuttle vertical stabilizer. The tug and shuttle orbiter vehicles will employ GO_2/GH_2 bipropellants and N_2H_4 monopropellant, respectively. The spin-up moment arms are 36.7 feet and 61.5 feet for the tug and orbiter, respectively. The spin-down lever arms are 21.4 and 57.5 feet, respectively.

For a 10-hour transfer period, the optimum point for Configuration B was established at about 8×10^{-4} g which results in losses totaling 210 pounds or 0.35 percent of the 60,000-pound propellant load. This is shown in Figure 6.3.2-25. In addition, optimum points for 5- and 20-hour transfer periods are shown as 1×10^{-3} and 4×10^{-4} g, respectively. The selected thruster configuration was Option 1 of Table 6.3.2-3 providing for balanced thruster couples mounted on the logistic module and shuttle orbiter vehicle. The module mounted thrusters utilize GO_2 and GH_2 propellants (I_{sp} 350) and the orbiter mounted thrusters utilize N_2H_4 monopropellants (I_{sp} 230). Spin-up thrust level arms on each side of the combined vehicle were assumed to be 48 feet. Spin-down moment arm was 37.3 feet.

Configuration C uses the 25-pound engines mounted on the CIS at Station 201 with two balancing engines at the extremity of the logistic module. Since filling of the CIS requires 19 module loads, the module engines were balanced for the initial and final conditions (3.1 and 21 pounds, respectively) and an average of all spin ups and spin downs were used for the propellant consumption calculations. The minimum loss of 190 pounds occurs at 4×10^{-4} g as shown in Figure 6.3.2-26. Variable thrust engines (perhaps preset on the ground) may be used to avoid excessive moments at the vehicle interface. Such a design concept will not affect the average values presented herein.

Configuration D is assumed to use existing baseline design CIS 25-pound engines at CIS Station 201 coupled with new orbiter mounted engines on a line through the cg from the CIS engines at orbiter-Z Station 96. Proper moment balance requires two engines of 49.2-pound thrust with CIS empty and 9.6 pounds with CIS full. They were assumed to utilize GO_2/GH_2 propellant. The minimum loss of 255 pounds occurs at 2×10^{-4} g as shown on Figure 6.3.2-26. Variable thrust engines would minimize moments as noted above and would not affect the average values presented.

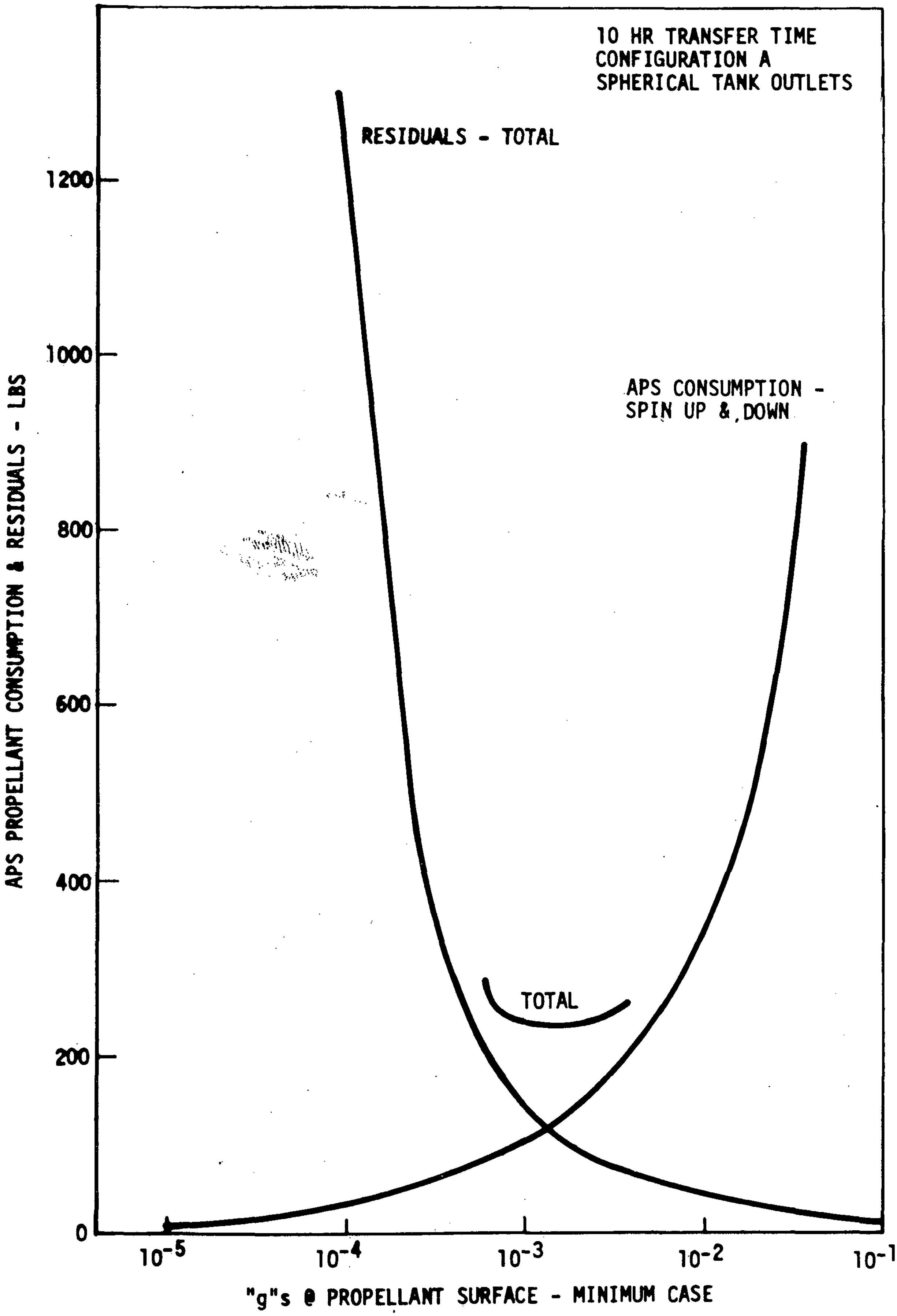


Figure 6.3.2-24 Propellant Consumption - Tug - Rotational Acceleration - Configuration A

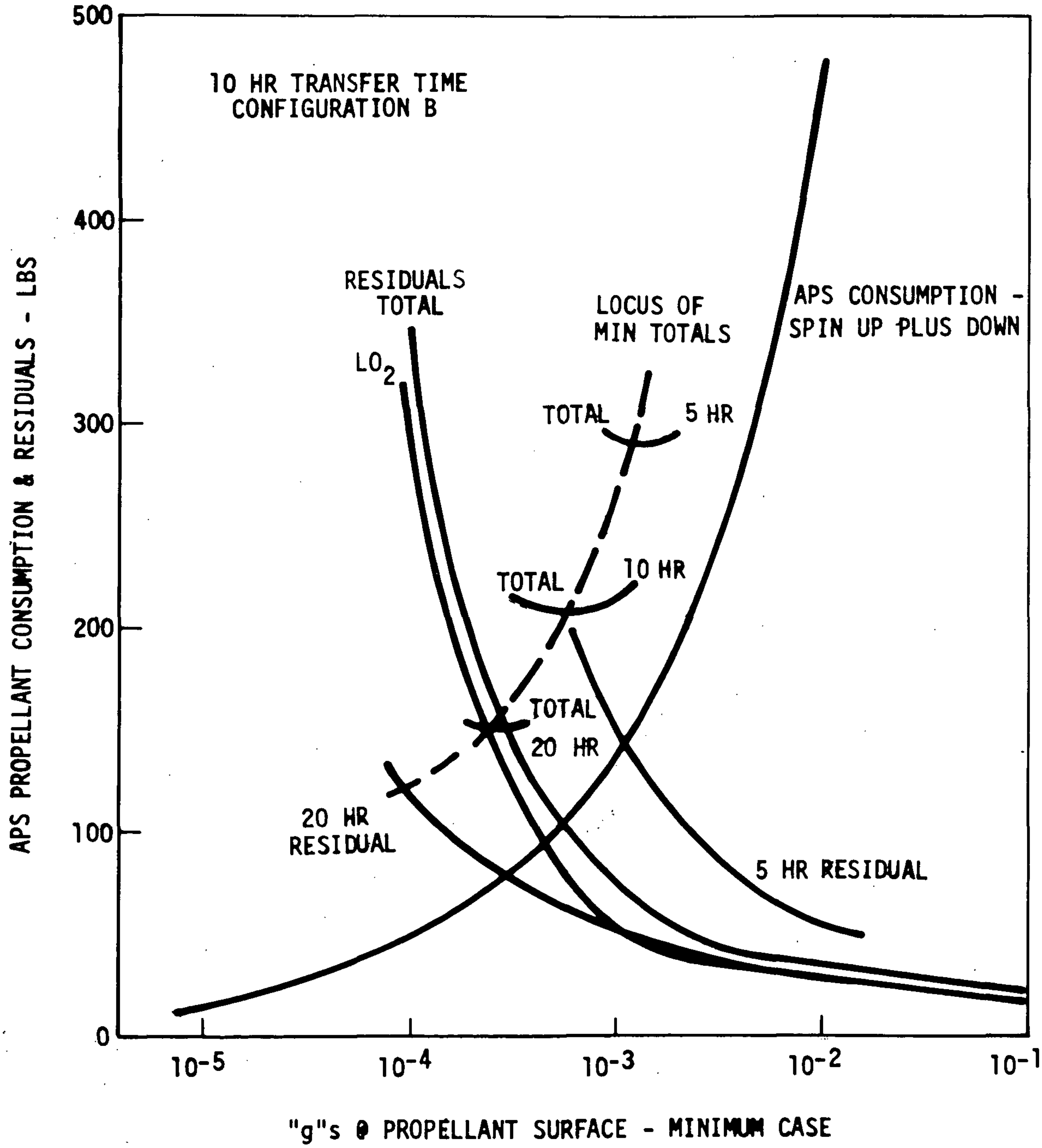


Figure 6.3.2-25 Propellant Consumption - Tug - Rotational Acceleration - Configuration B

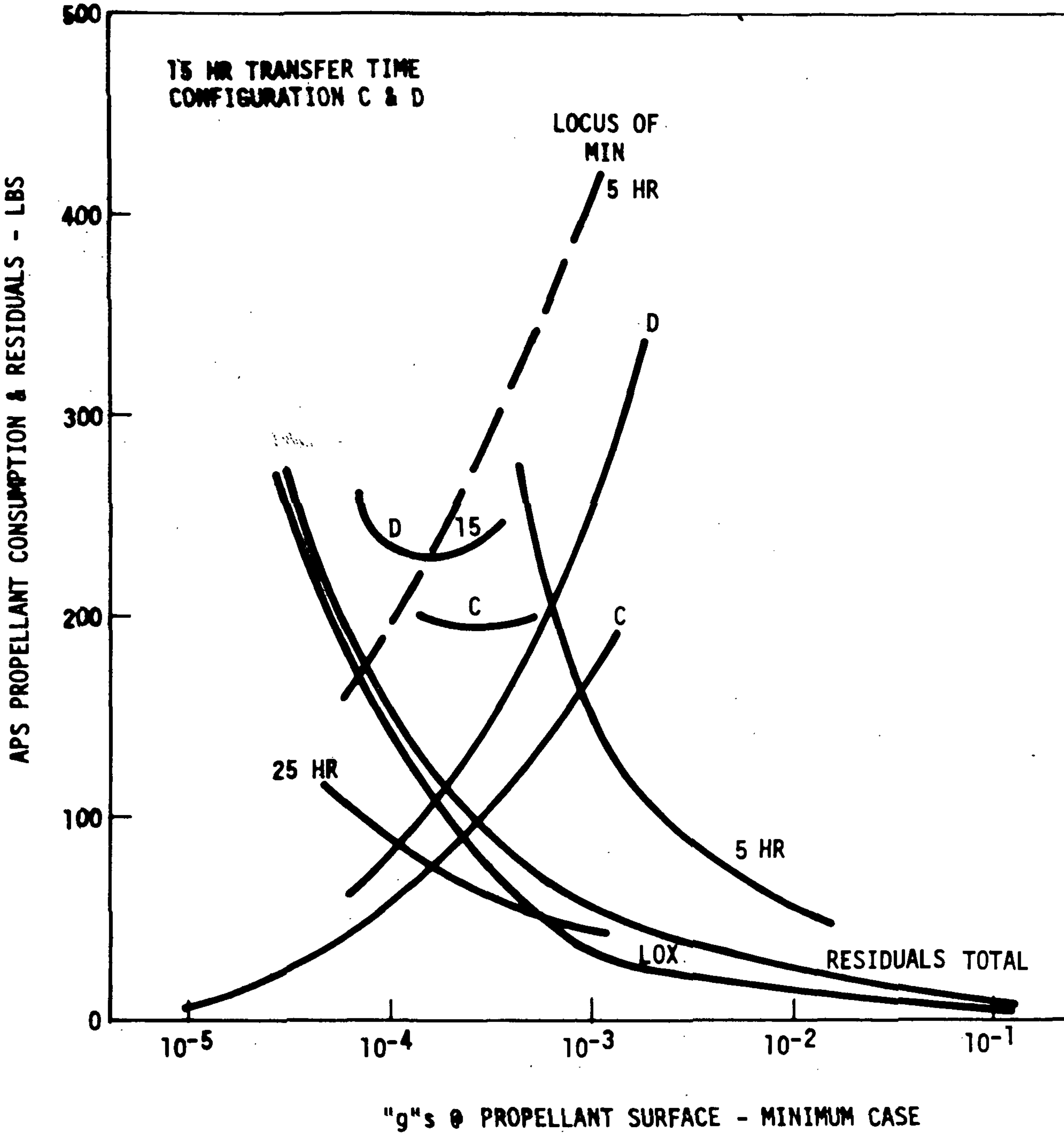


Figure 6.3.2-26 Propellant Consumption - CIS -
Rotational Acceleration - Configuration C&D

Configurations E and F assumptions are similar to the above for Configurations C and D. RNS-mounted thrusters of 24 pounds were assumed positioned five feet from the trailing edge with balancing thrusters at the extremity of the logistic module for Configuration E and at orbiter station-Z eight feet for Configuration F. Configuration E thrusters were assumed to utilize GH_2, heated and expanded with an I_{sp} of 200. Configuration F uses the same thruster on the RNS (orbiter +Z 227 feet) and orbiter mounted N_2H_4 engines at -Z eight feet (I_{sp} 230). The minimum losses for Configurations E and F were 125 pounds and 280 pounds, respectively. These are shown on Figure 6.3.2-27.

6.3.2.4 Effect of Rotation on Orbiter Crew

In conjunction with the rotational acceleration propellant loss analysis, a parallel investigation was conducted relative to the effect of these rotational rates on the orbiter crew. For this study the crew compartment acceleration levels encountered during the radial mode of propellant transfer to the tug were calculated for Configurations A and B, defined in Section 6.3.2.3.

The curves of Figure 6.3.2-28 depict the crew compartment acceleration at shuttle orbiter Station 519 and the propellant settling acceleration for the propellant liquid surface level closest to the combined vehicle center of gravity (c.g.) for various pitch rotation rates. The Configuration A design point of 2×10^{-3} g propellant settling results in 2.8×10^{-2} g in the crew compartment. For Configuration B, these values are 8×10^{-4} and 1.3×10^{-3} g, respectively. From a crew comfort viewpoint, Configuration B is preferred.

The possibilities of any adverse effects on the crew resulting from spacecraft pitch rotation on the order of 10 to 80 revolutions per hour (RPH) were investigated. It was determined that the rotation rates are within personnel tolerance limitations even for the extended periods of 10 hours or more. However, certain minor equipment accommodations may be necessary. Various tasks and activities normally done in zero g may require slight alteration because of the low acceleration forces. It is believed that use of personnel straps may be necessary to avoid fatigue in maintaining an upright seated position although the torso forces are only a few ounces. The procedures utilized for activities such as eating and waste elimination may need to be changed in some minor detail. Additional constraint provisions of strings, straps, or velcro may be necessary to prevent objects from floating out of normal position during certain operations.

In summary it is believed that the spacecraft rotation mode for refueling will have no adverse physiological affects on the crew and minor, if any, impact on crew systems design.

6.3.2.5 Propellant Orientation During Rotational Transfer

An important consideration in the evaluation of rotational propellant transfer is the center of gravity (c.g.) location and migration during the transfer operation. Successful transfer of propellants using centrifugal forces to orient and settle the propellant depends on maintaining a favorable location of propellants relative to the axis of rotation which is determined by the c.g. From the standpoint of ullage venting, the most desirable location for the c.g. is

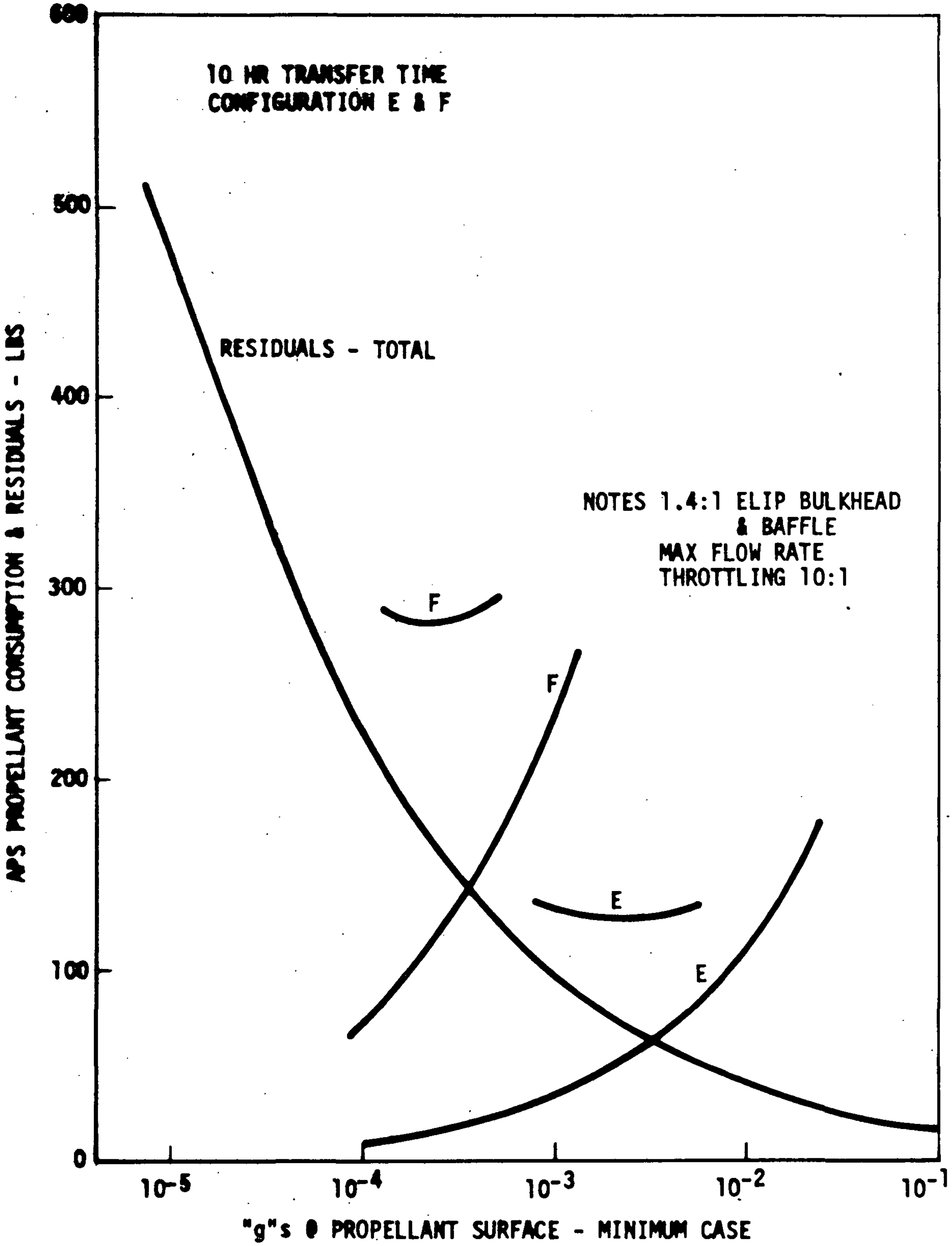


Figure 6.3.2-27 Propellant Consumption - RNS - Rotational Acceleration - Configuration E & F

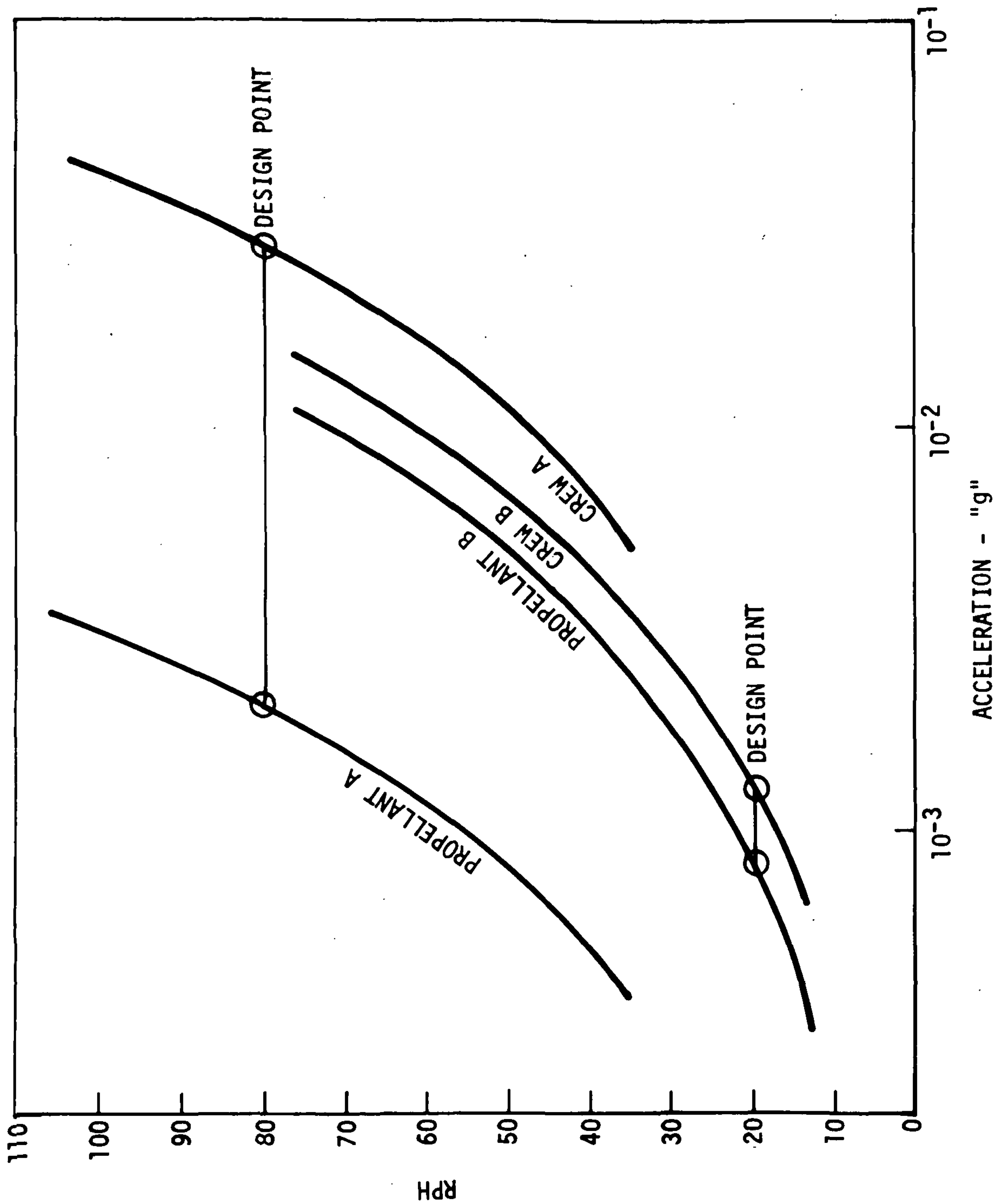


Figure 6.3.2-28 Rotation Rate - Configuration A & B

outside the propellant tanks so that the relative position of the liquid and ullage location remains essentially fixed. This requires separating the logistic module and the receiver vehicle by a rigid boom to distribute the vehicle masses so that, whether loaded or empty, the c.g. of the configuration will always fall between the two vehicles. However, when the weight differential between the logistic module and receiver vehicle is large (e.g., a loaded CIS and logistic module), the length of the boom becomes prohibitively long.

Transfer of propellants in the rotational mode without the use of a transfer boom was investigated for CIS. Two configurations were studied. Configuration C, consisting of only the CIS and logistic module (shuttle detached), and Configuration D, consisting of the CIS and logistic module with the shuttle attached as a counterweight. Configurations C and D refer to the configuration sketches presented in Section 6.3.2.3. For each of the configurations, calculations were made of c.g. locations and liquid levels as functions of the number of propellant loads transferred to the CIS. The c.g. locations and liquid levels were determined for conditions just prior to initiating propellant transfer and after completion of transfer for each load transferred.

The CIS and logistic module weight characteristics assumed for the study are shown in Table 6.3.2-4. The CIS propellant loading requirements were obtained from Reference 6.0-9.

Table 6.3.2-4. CIS/Logistic Module Weight Characteristics

CIS	
	pounds
Usable LO_2	783,500
Usable LH_2	131,000
Residual LO_2	1,900
Residual LH_2	4,800
LOGISTIC TANK	
LO_2 losses per load	3,200
LH_2 losses per load	200
LO_2 transferred	46,000
LH_2 transferred	10,600
Filled logistic module	65,000
Empty logistic module	8,400
CIS BOILOFF RATES	
	pounds/hour
LO_2	0
LH_2	10

The c.g. location and liquid level as a function of the number of loads transferred to the receiver tanks are shown in Figures 6.3.2-29 and 6.3.2-30 for the CIS/module and CIS/module/shuttle configurations, respectively. For both configurations, the LO_2 is filled from the bottom on the tank (same as ground fill). The LH_2 tank is filled from the tank bottom and/or top depending on the position providing the most favorable location of the liquid and ullage with respect to the c.g.

In the separate rotational mode, the LH_2 tank is filled from the top so that the relative position of the LH_2 ullage remains fixed (always located adjacent to the tank bottom) throughout the entire propellant transfer operation. This simplifies ullage venting during transfer. The LO_2 ullage remains above the liquid until the last few logistic module propellant loads are transferred. At this time, the location of the c.g. causes the LO_2 to shift to the tank wall and the ullage moves within the bulk liquid as illustrated in Figure 6.3.2-29. Venting of the LO_2 tank under these circumstances may be satisfied by a single vent outlet as long as the ullage shift is small and the ullage location stable after detaching from the wall.

The c.g. locations and liquid levels as functions of the number of logistic module propellant loads transferred to the CIS stage for a deployed rotational mode are shown in Figure 6.3.2-30. The LH_2 tank is filled at the tank bottom for the first five propellant tank loads transferred, and from the top of the tank for the remaining. The LH_2 ullage shifts from the forward end of the tank to a position within the bulk liquid. This requires an additional vent outlet besides that provided for flight operational requirements. The LO_2 tank ullage remains located at the forward end of the tank throughout the entire propellant transfer operation.

The advantages of both modes of transfer, besides lower propellant losses mentioned previously, is that perturbations to the parking orbit which may require corrective maneuvers are reduced since thrusting is required only during spin up and despin maneuvers and not continuously as required for the linear propellant transfer mode. Separate rotational transfer offers an advantage in that the shuttle can be used for other purposes during the propellant transfer period. An advantage of the deployed rotational mode is that one docking maneuver is eliminated since the logistic module remains attached to the shuttle.

The requirement for providing additional CIS propellant tank fill inlets and vent outlets to satisfy all phases of operations (ground fill, inorbit transfer, and flight) adds to the complexity of the propellant feed and ullage vent systems. The position stability of the ullage once it detaches from the tank wall (near full conditions) may be poor so that the placement of vent outlets in order to satisfy adequate ullage venting can become a major design problem. Another disadvantage of the rotational propellant transfer mode is that if discrete liquid level sensing gauging systems are used (i.e., capacitance probe and point sensors), gauging measurements are non-linear.

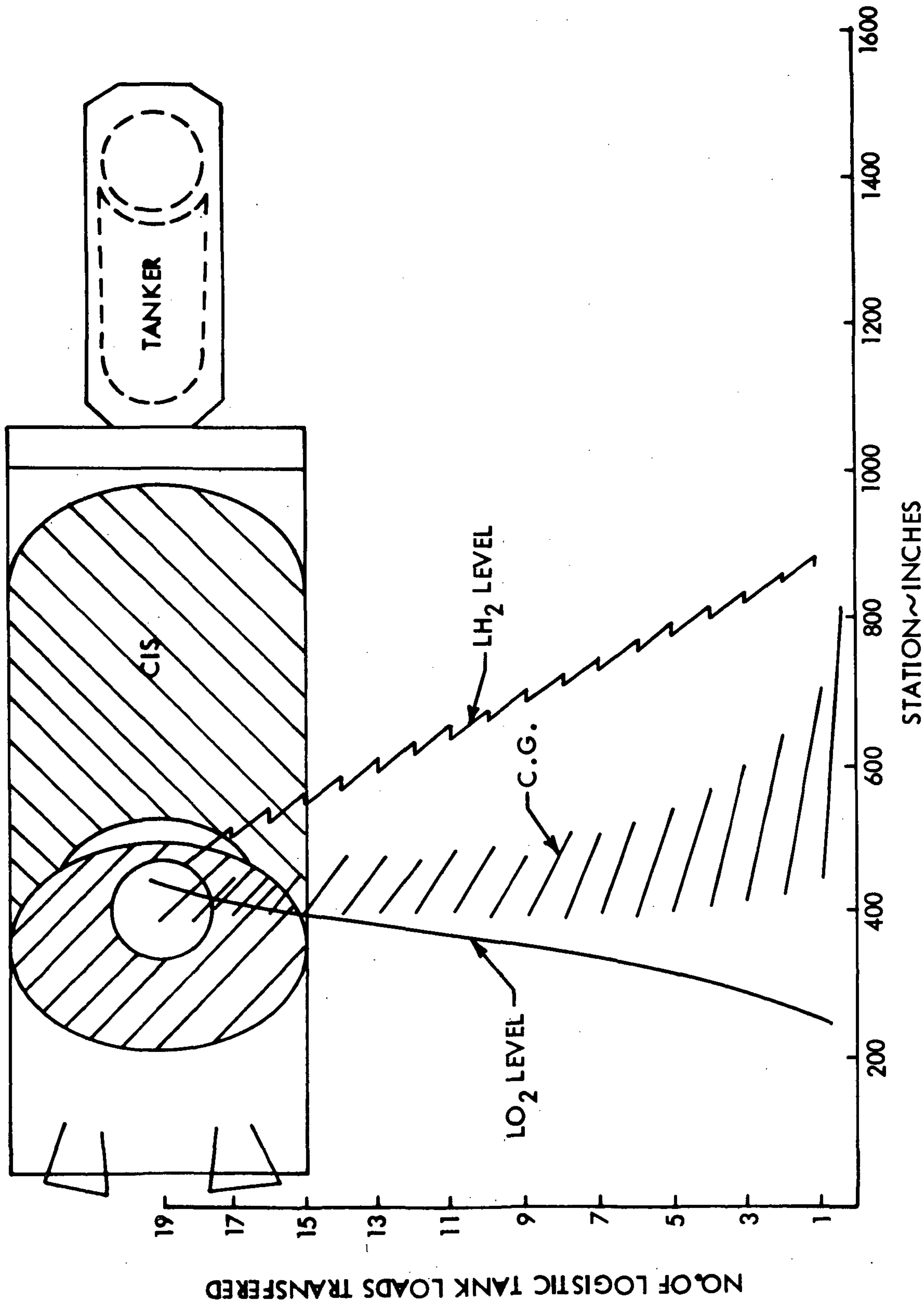


Figure 6.3.2-29 Separate Rotation C.G. Location

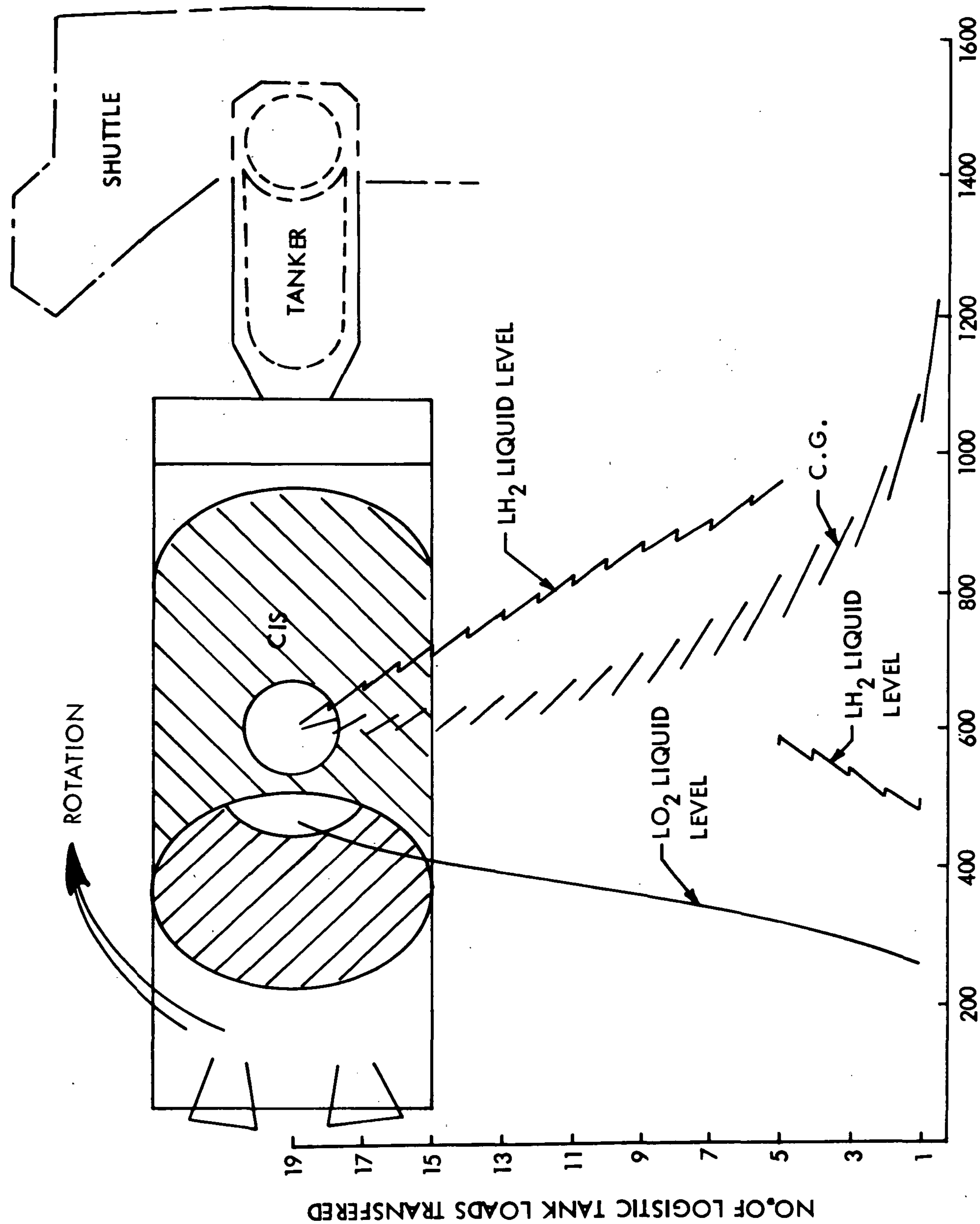


Figure 6.3.2-30 Deployed Rotation C.G. Location

Another problem associated with rotational propellant transfer is that the location of the propellant in the receiver tank is unpredictable in a zero g environment. When the logistic module is not attached to the CIS the propellants can migrate to other parts of the tank. After spin up from periods of zero g, the c.g. location can change from that which prevailed at completion of the last logistic module load. As a result, the orientation of propellant and ullage becomes unpredictable and further complicates the problem of providing adequate tank venting.

6.3.2.6 Orbital Mechanics for Linear Acceleration

The orbital mechanics of linear acceleration transfer techniques were analyzed to establish a viable means of operation. Thrust vector orientation was analyzed for two modes, in-plane, and cross-plane (normal to the shuttle or originating orbit plane) using a NR precision trajectory computer program. The computer program is based on stepwise integration and includes all appreciable earth oblateness effects. The data were obtained by first establishing, as a standard, the chronology of orbital position of a body orbiting the earth at 180 nautical miles. Additional computer runs were made with a small thrust equivalent to an acceleration level of 10^{-4} g. The thrust was oriented in the plane of orbit and then normal to the plane of the standard orbit. The data were then compared to establish line-of-sight separation with the tug/logistics tank and quiescent shuttle as a function of time.

The results of this analysis are shown graphically on Figure 6.3.2-31. This figure presents the separation distance (line of sight) and the altitude locus for one orbit for both the in-plane and cross-plane techniques. The ten-orbit separation distance for the cross-plane technique is also indicated. As shown on the figure in-plane thrusting at a constant inertial attitude results in an orbital path divergent to the initial orbit. This maneuver would result in de-orbit and earth impact within a few orbits. When a cross-plane thrust is applied to an orbiting body the net result is a plane change. If the thrust is applied continuously, the orbit path is constantly changing direction. The geometry of the orbit is such that the path returns to its point of origin once each revolution.

The oblateness of the earth introduces orbital perturbations such that the two orbits do not return to the exact same point of cross so long as the low level thrust is maintained. For the case examined, with 10^{-4} g acceleration, the point of nearest approach of the two orbiting bodies was approximately 0.0079 n mi at the end of one orbit. At the end of five orbits this distance was 0.2350 n mi. Extrapolating the nearest approach distance to 10 orbits ($\sim$15 hours) indicates a separation of about 7.1 miles with the separation being mainly normal to the orbit plane of the shuttle. If the thrust of the tug/logistics module were terminated at this point, an orbit tilted to shuttle orbit would result. The orbits would cross at a point 90 degrees (and 270 degrees) from the point of thrust cessation. A delta velocity of approximately 50 ft/sec applied at the nodal point (90 degrees or 270 degrees) would return the tug logistic module to orbital plane of the shuttle.

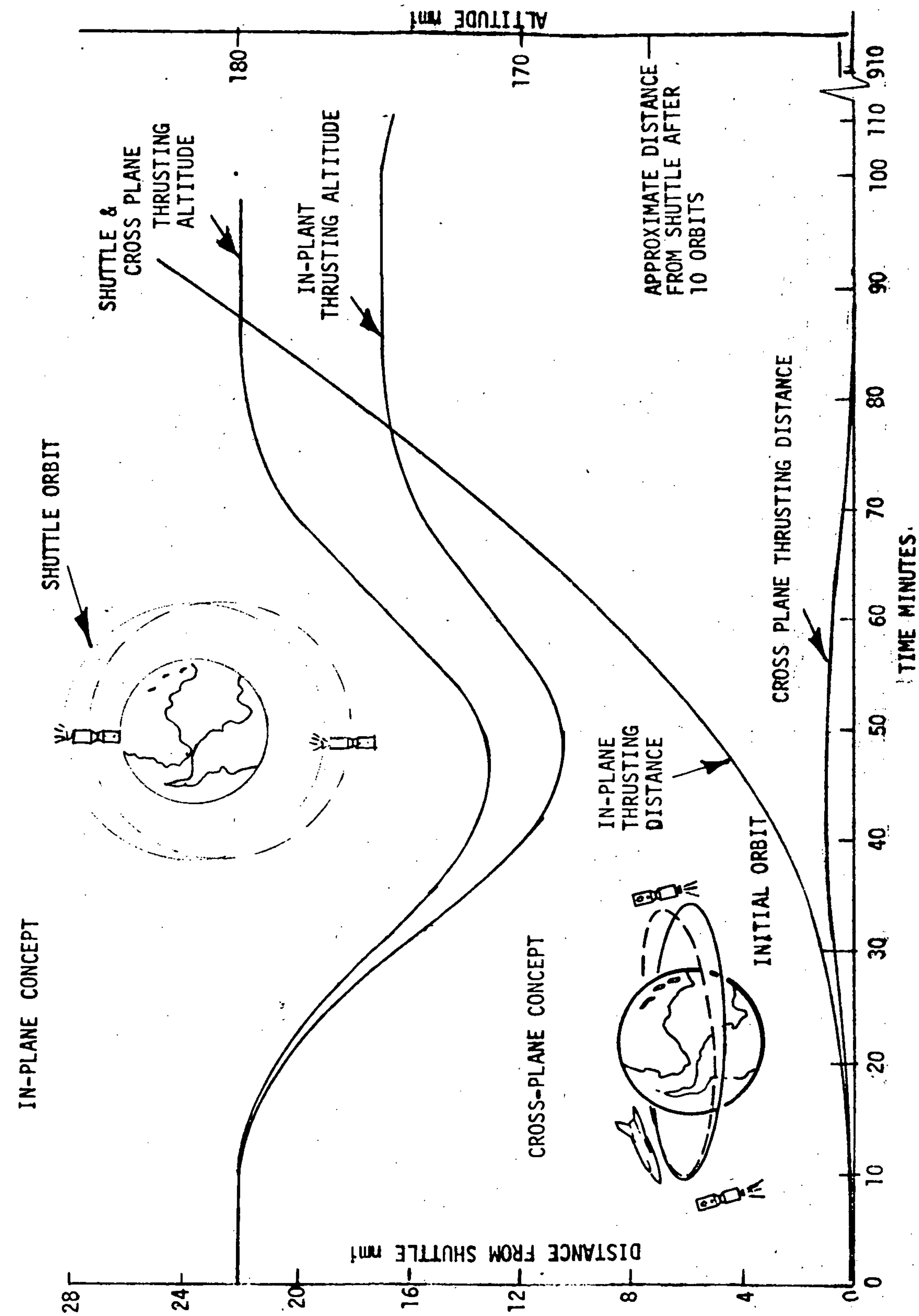


Figure 6.3.2-31 Linear Acceleration/Propellant Settling Orbital Mechanics

6.3.2.7 Characteristics of Capillary Propellant Transfer

The previous analyses have shown the range of operations, transfer times, flow rates, acceleration levels, and flow throttling ratio by which efficient transfer of propellant to various users can be achieved. Propellant transfer by linear acceleration to the tug and, to a lesser extent, to the RNS, can be achieved with low propellant losses. Propellant transfer to the CIS involved moderate propellant losses. Furthermore, transfer to the CIS involved changing acceleration levels and transfer times for successive transfers, very low acceleration levels for the last few transfers, and appreciable maneuvering requirements for all transfers. Thus, although the linear acceleration technique will provide adequate performance, the complexity of this technique makes it desirable that alternatives be considered. One such alternative which appears particularly promising is a zero-g transfer mode which utilizes capillary devices for efficient feedout and vapor/liquid interface control.

A schematic of a capillary system to effect transfer of propellant from the logistic module to the CIS is shown in Figure 6.3.2-32. This capillary system consists of capillary collector tubes in the logistic module tanks for propellant acquisition from any region of the tanks. The CIS requires fill and vapor/liquid interface control baffles to permit orderly filling and vapor return to the source tanks. These hardware provisions permit reduction of thrusting requirements; however, thrusting is still required for gauging, for initial settling, and at times during transfer to reposition dislocated propellant. Thrusting may also be required continuously during the last few transfers because of the difficulty of obtaining propellant-free vapor return from the nearly full receiver tank.

This approach reduces the thrusting and maneuvering requirements and decreased jet propellant consumption. It may be possible to utilize the existing auxiliary propulsion system on the CIS. Because of the network of collector tubes along the perimeter of the logistic tank, residuals are less sensitive to sloshing. Another advantage of using this quasi-passive method of transfer is the compatibility of this approach with a wide range of receiver configurations; i.e., center of mass location is not as important if only limited maneuvers are involved.

Of course, use of capillary devices introduces additional hardware complexity and development risk. Problems associated with such devices are identified in the Supporting Research and Technology section of Volume IV.

6.3.3 Comparison of Candidate Liquid/Vapor Interface Control Concepts for Tug

Four liquid vapor interface control concepts for direct transfer to the space-based tug are shown in Figure 6.3.3-1. The concepts are (1) in-bay rotation, (2) deployed rotation, (3) deployed linear, and (4) separate linear. The first two concepts utilize rotational acceleration for liquid/vapor interface control during transfer; the remaining two concepts use linear acceleration. For in-bay rotation the logistic module remains in the orbiter cargo bay. For deployed rotation, the logistic module is deployed into a position where the center of mass at any time during the transfer lies along the axis of symmetry of the logistic module and between the module and the orbiter. The orbiter

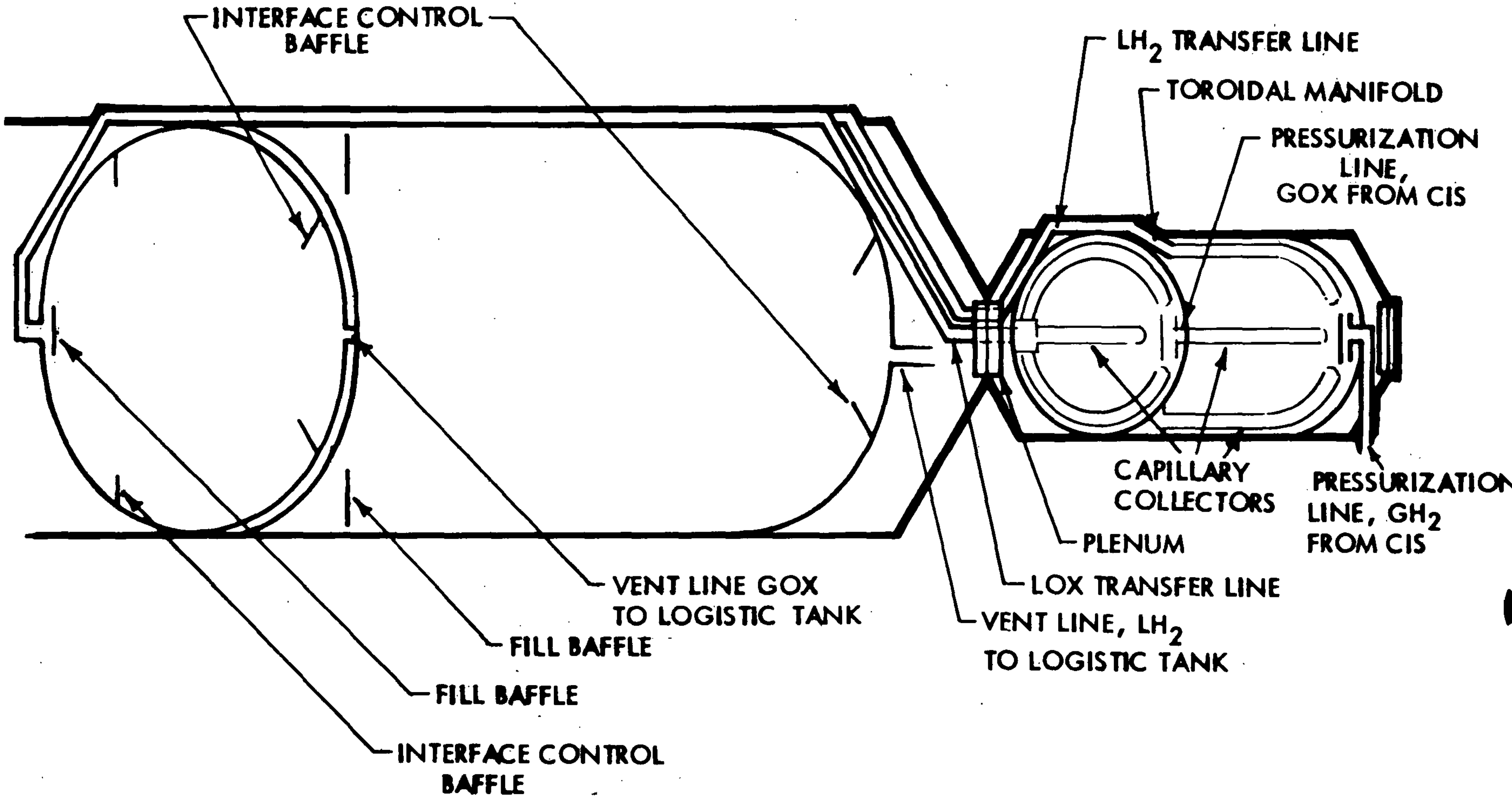


Figure 6.3.2-32 Capillary Zero g Transfer with Vapor Return

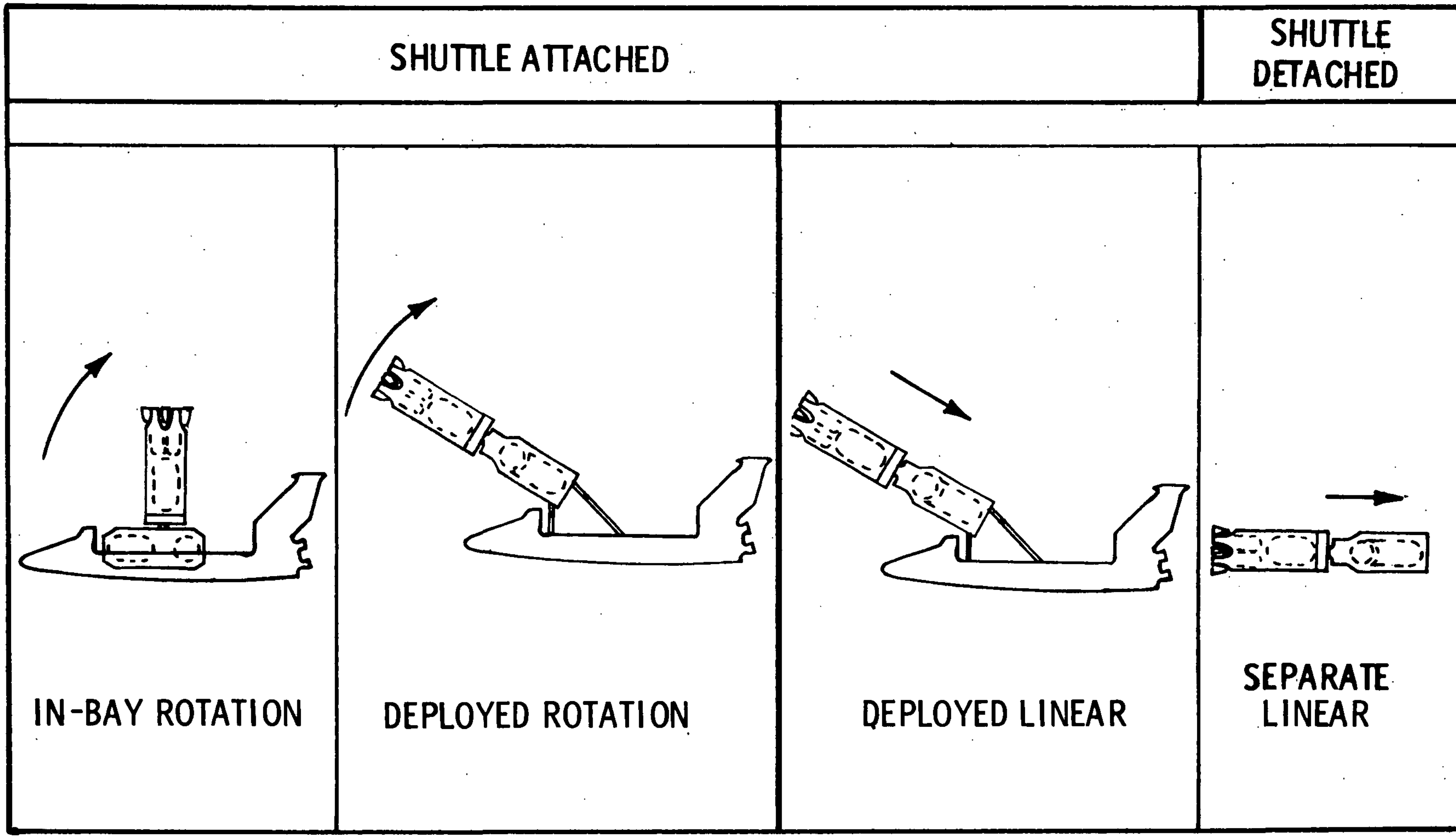


Figure 6.3.3-1 Alternate Concepts-Direct Transfer to Tug.

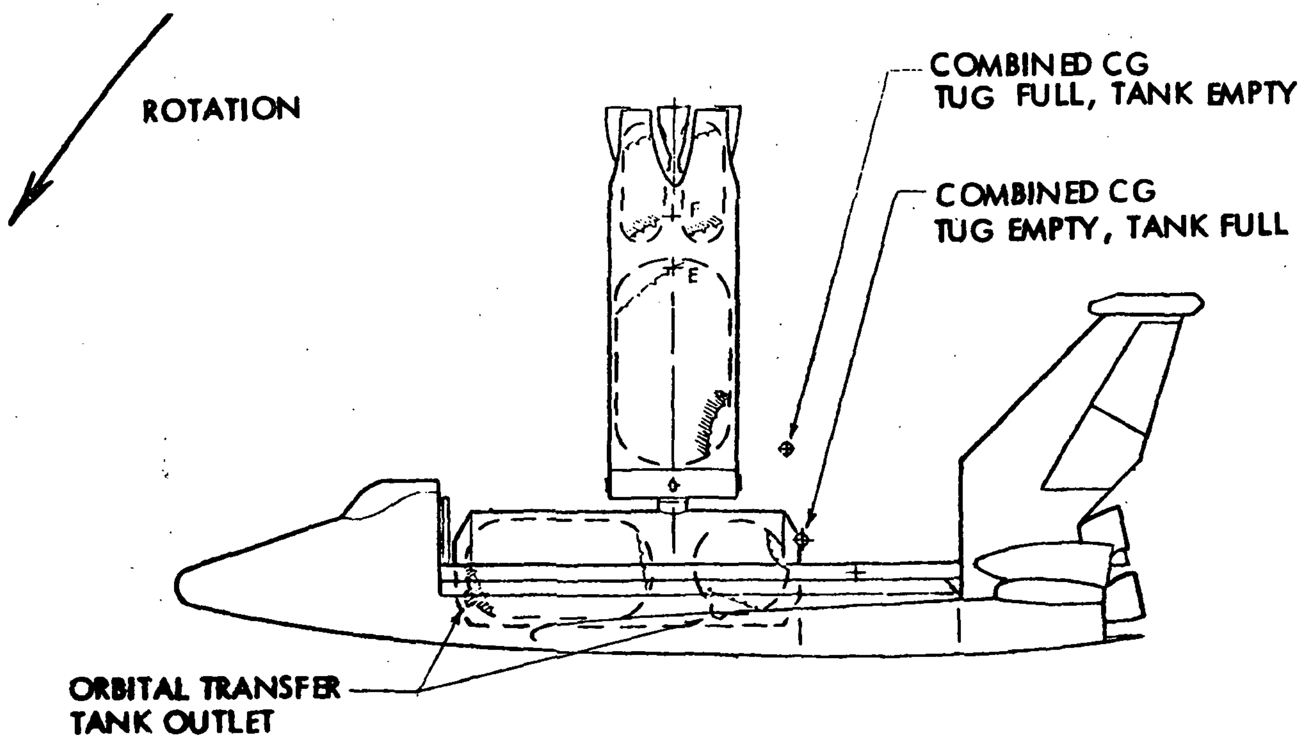
functions as a counterweight. The configuration for the deployed linear concept is identical to that for the deployed rotational concept, but employs linear rather than radial acceleration for interface control. The separate linear concept is the only one for which the orbiter is detached. Linear acceleration is also used for interface control for this case.

Figures 6.3.3-2 through 6.3.3-5 present additional details for each concept. For the in-bay rotation concept, the logistic module remains in the cargo bay and the orbiter functions as a tanker. All in-space interfacing between the orbiter and logistic module is eliminated. One rendezvous and docking operation is required for each transfer operation. Cargo bay c.g. constraints dictate an inverted tank mounting and the c.g. migration is along the side of the receiver tanks. As a result of these constraints, ground fill and drain, ground and space venting, and propellant gauging are more complex. The orbiter and crew are necessarily retained for the full duration of the transfer.

The deployment angle for the deployed rotation concept is selected to align the c.g.'s of the vehicles and provide c.g. migration along the longitudinal axes of the tug and tank. This technique requires one rendezvous and docking for each transfer operation. The manipulator reach requirements are extended to facilitate soft docking of the tug and tank. Deployment, docking, and support of the tank are considered feasible but complex. The flexible lines between the orbiter and tank eliminate in-space connections for this interface but introduce additional design problems. The attitude of the logistic module in the cargo bay simplifies the ground fill, vent, and gauging configuration. The orbiter and crew are retained for the full duration of the propellant transfer operation.

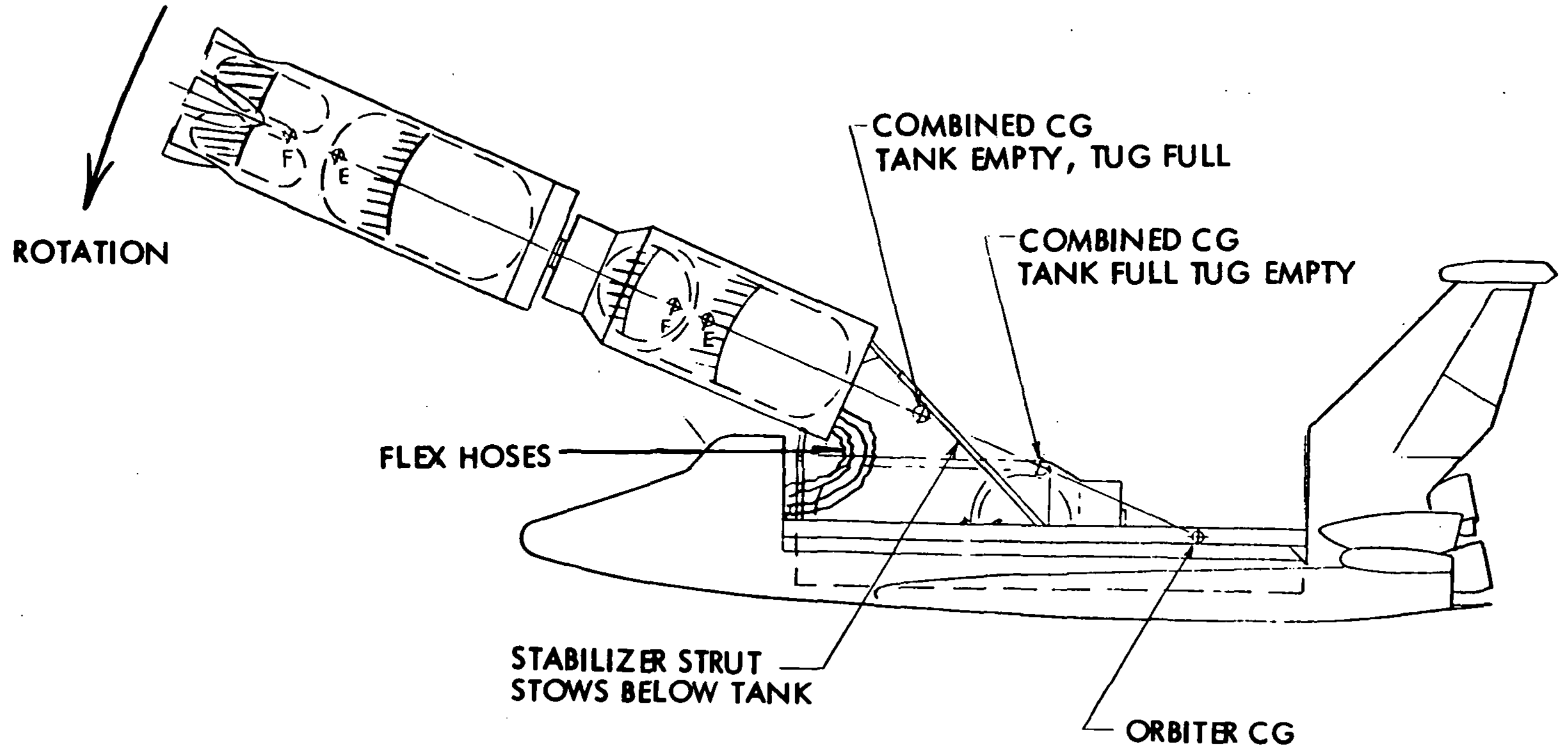
For the deployed linear concept the logistic module is deployed and fixed at the proper angle to align the c.g.'s of the tug, tank, and orbiter. As propellant is transferred the total c.g. will migrate along the tug and tank longitudinal axes. This technique requires one rendezvous and docking for each transfer operation. The manipulator reach requirements are extended to facilitate soft docking of the elements. Deployment docking, and support of the tank are considered feasible but complex. The flexible lines between the orbiter and the tank eliminate in-space connections for this interface but introduce additional design and configuration problems. The attitude of the logistic tank in the cargo bay is normal which simplifies the fill, vent, and gauging configuration. The orbiter and crew are retained for the full duration of the propellant transfer operation.

During the separate linear concept after the initial rendezvous, the logistic tank is deployed from the cargo bay and soft docked with the tug using an orbiter manipulator arm. The tug and tank are then separated from the orbiter and the propellant transferred using cross plane linear acceleration to control the liquid vapor interface. After completion of the propellant transfer, the tug and tank are again docked with the orbiter and the tank returned to the orbiter cargo bay. This concept requires two rendezvous and docking operations and two orbiter and tank interface (fluid and mechanical) operations per transfer. The logistic module attitude in the cargo bay is such that propellant is settled to the aft end for both orbital and ground operations which simplifies the fill, vent and gauging configurations. During the propellant transfer period,



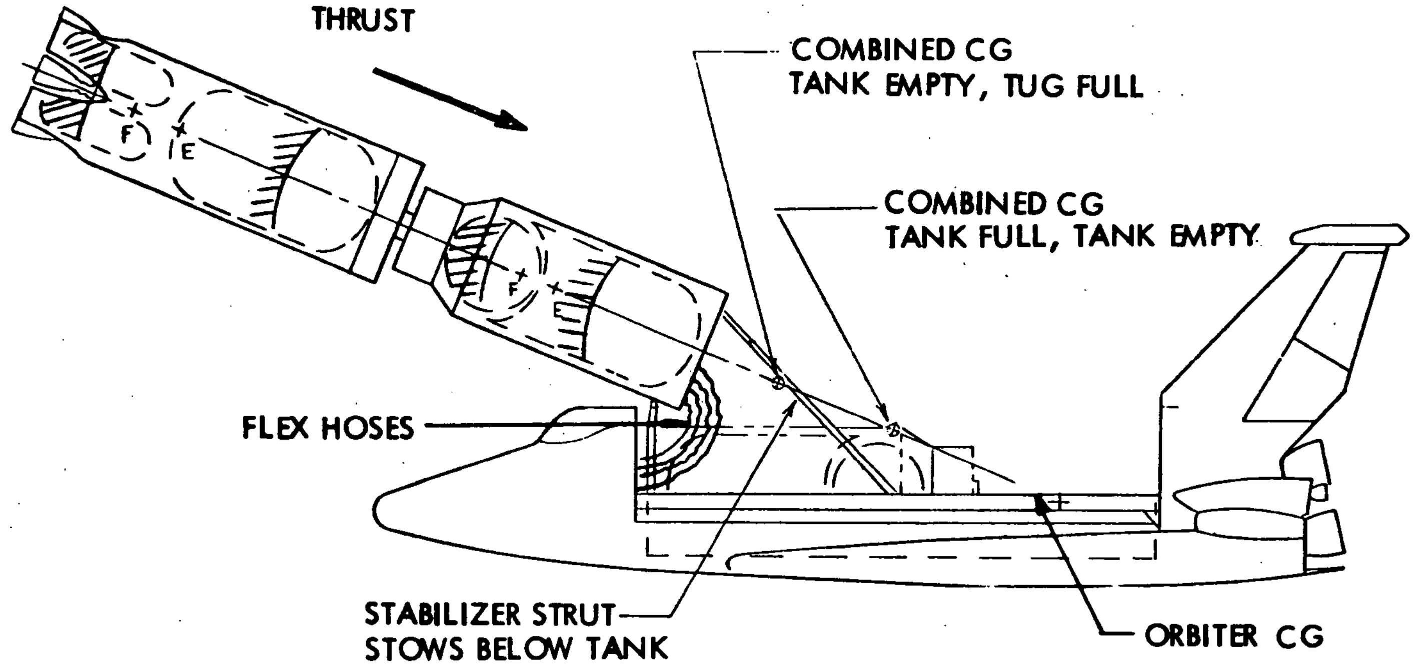
- INVERTED GROUND FILL & DRAIN
- ONE RENDEZVOUS & DOCKING PER TRANSFER
- COMPATIBLE WITH CURRENT MANIPULATOR ARMS
- NO DEPLOYMENT OF TANK
- SHUTTLE & CREW RETAINED DURING TRANSFER
- SIDE TANK VENTING OF TUG
- NON LINEAR GAUGING

Figure 6.3.3-2 In-Bay Rotation Direct Transfer to Tug.



- ONE RENDEZVOUS AND DOCKING PER TRANSFER
- LONGER MANIPULATOR ARMS REQUIRED
- COMPLEX DEPLOYMENT AND SUPPORT
- FLEXIBLE PROPELLANT LINE INTERFACE
- SHUTTLE AND CREW RETAINED DURING TRANSFER
- STANDARD TUG FILL AND VENT LOCATION
- LINEAR GAUGING

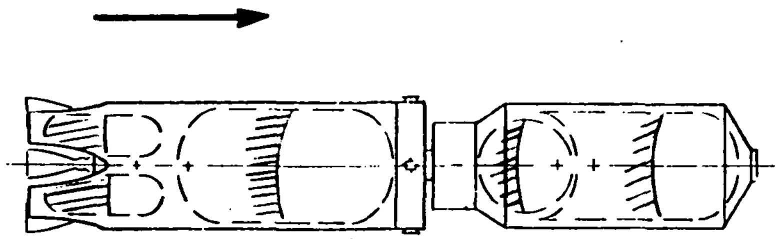
Figure 6.3.3-3 Deployed Rotation Direct Transfer to Tug



- ONE RENDEZVOUS AND DOCKING PER TRANSFER
- LONGER MANIPULATOR ARMS REQUIRED
- COMPLEX DEPLOYMENT AND SUPPORT
- FLEXIBLE PROPELLANT LINE INTERFACE
- SHUTTLE AND CREW RETAINED DURING TRANSFER
- STANDARD TUG FILL AND VENT LOCATION
- LINEAR GAUGING

Figure 6.3.3-4 Deployed Linear Direct Transfer to Tug

THRUST



- TWO RENDEZVOUS AND DOCKING PER TRANSFER
- SHUTTLE DEPLOYMENT & PROPELLANT TRANSFER CONCEPTS ARE INDEPENDENT
- FLUID INTERFACE DISCONNECTS REQUIRED (SHUTTLE/TANK)
- SHUTTLE AND CREW FREE DURING TRANSFER
- STANDARD TUG FILL AND VENT LOCATIONS
- LINEAR GAUGING

Figure 6.3.3-5 Separate Linear Direct Transfer to Tug the orbiter and crew are free to monitor the operation or perform other tasks. Propellant transfer losses based on a 10-hour transfer time and computed in percentage of the 60,000-pound propellant load are summarized in Table 6.3.3-1. This data show the major losses to the APS propellant and liquid residuals, the propellant required for net positive suction pressure (NPSP) control, and other purposes being small and insensitive to transfer method. For the rotational methods the acceleration level was chosen to minimize propellant losses. For the linear acceleration methods 10^{-4} g proved the most practical acceleration level. The rotational methods entailed lower propellant losses than the linear methods because of the small APS propellant requirements and reduced residuals.

APS propellant requirements are substantially lower for the separate linear method than for the deployed linear method, because of the space shuttle mass which must be accelerated for the deployed linear method.

A breakdown of logistic program costs for the tug supportive missions are presented in Figure 6.3.3-6. Program costs are based on the Program C level, Concept 2, employing the separate linear acceleration transfer configuration. Shuttle flight and hardware cost elements and the relative magnitude of the costs to each other are indicated.

A comparison of the cost of propellant transfer losses and the percentage of the total program cost that the losses represent is made for the various propellant transfer configurations. As shown, the cost of the transfer losses for all the configurations are small when compared to total program costs. Therefore, it is concluded that the program costs are relatively insensitive to the transfer configuration and factors other than cost are the major drivers which influence the transfer configuration selection.

The principal factors considered in the analysis of each of the concepts evaluated have been assessed as advantages and disadvantages and are shown in the trade table, Table 6.3.3-2. This table presents the results of this assessment and indicates that no single driver identified one concept as superior. Also, no single factor was identified which could be used to eliminate any of the concepts. Therefore, as indicated in the table, the separate linear concept was selected as the study baseline. This selection was based primarily upon the considerations indicated by the asterisk on Table 6.3.3-2 showing that the separate linear concept will minimize impact on the shuttle design and crew requirements, reduce overall complexity of the configuration, and prevent excessive operational cost.

6.3.4 Comparison of Candidate Liquid/Vapor Interface Control Concepts for CIS

Four options were evaluated for the transfer of propellant from the logistic module to the CIS. These options were separate rotation, deployed rotation, separate linear and capillary. Two configurations employ rotation for liquid vapor interface control; one employs linear acceleration for interface control; and one uses capillary devices for logistic module propellant acquisition. For this last configuration, linear thrusting is used to augment the transfer by providing settling at the beginning of transfer and thereafter, as required, to maintain interface control. These approaches are shown in Figure 6.3.4-1.

Table 6.3.3-1 Summary of Propellant Losses

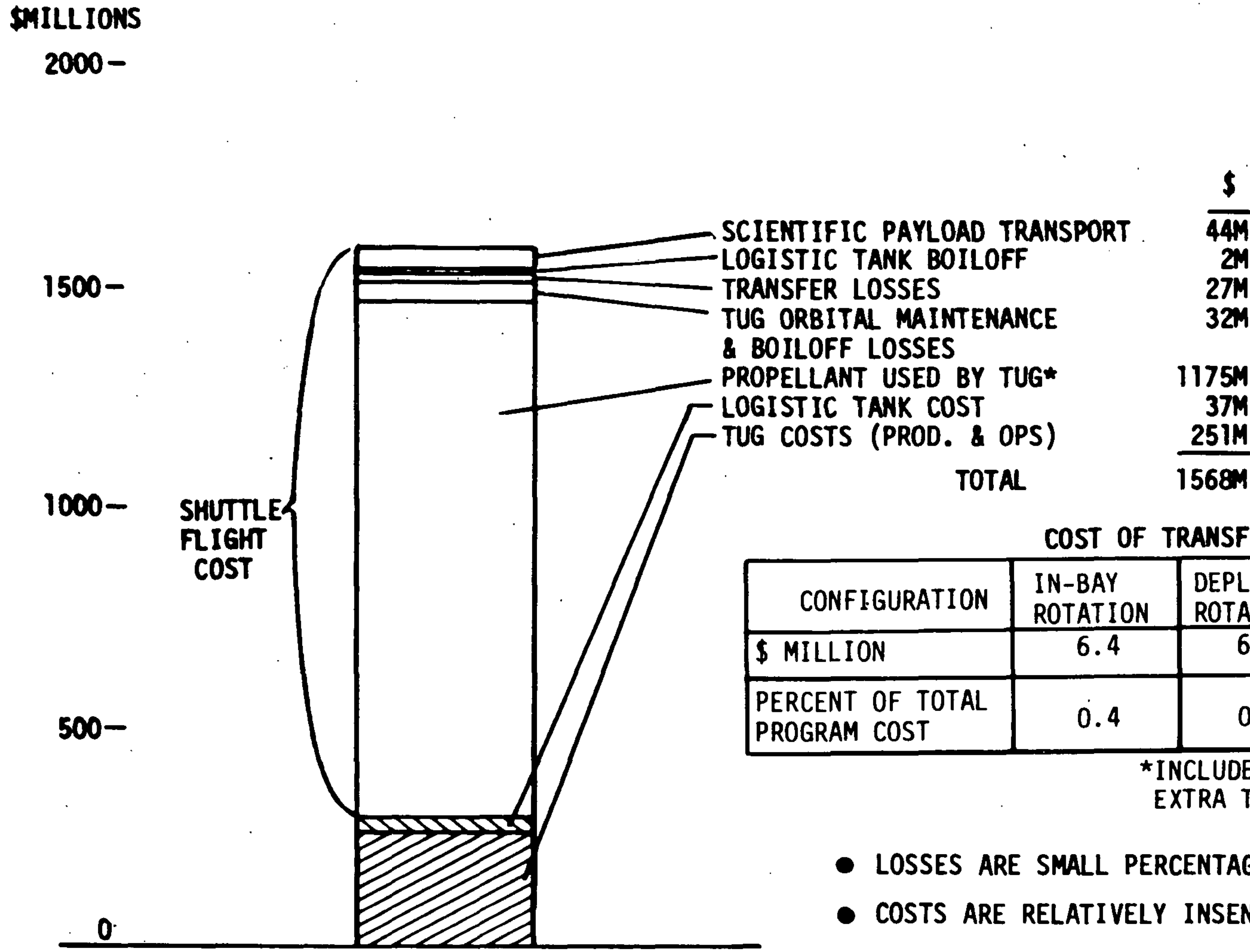
CONFIGURATION	IN-BAY ROTATION	DEPLOYED ROTATION	DEPLOYED LINEAR	SEPARATE LINEAR
ACCELERATION, G'S	2×10^{-3}	8×10^{-4}	10^{-4}	10^{-4}
APS PROPELLANT	0.2	0.2	4.2	1.3
LIQUID RESIDUALS	0.2	0.2	0.6	0.6
NPSP CONTROL	0.1	0.1	0.1	0.1
OTHER*	0.1	0.1	0.1	0.1
TOTAL	0.6	0.6	5.0	2.1

*PUMPING POWER, LINE RESIDUALS, LINE CHILLDOWN, AND HEAT LEAK

BASED ON 10 HRS TRANSFER TIME

LOSSES IN % OF 60 KLBS

- PROGRAM LEVEL C, LOGISTIC CONCEPT 2
- 1985-1990
- 0-30 DEGREE INCLINATIONS



COST OF TRANSFER LOSSES

CONFIGURATION	IN-BAY ROTATION	DEPLOYED ROTATION	DEPLOYED LINEAR	SEPARATE LINEAR
$ MILLION	6.4	6.4	32.2	27.0
PERCENT OF TOTAL PROGRAM COST	0.4	0.4	2.1	1.7

*INCLUDES TANK TRANSPORT - $96M
EXTRA TUG TRANSPORT - $16M

- LOSSES ARE SMALL PERCENTAGE OF PROGRAM COST
- COSTS ARE RELATIVELY INSENSITIVE TO TRANSFER CONCEPT

Figure 6.3.3-6 Logistic Program Costs for Direct Transfer to Tug

Table 6.3.3-2 Liquid/Vapor Interface Control Trade Table - Tug

REQUIREMENT	ALTERNATE APPROACHES				SELECTION
	RADIAL ACCELERATION		LINEAR ACCELERATION		
	IN BAY ①	DEPLOYED ②	DEPLOYED ③	SEPARATE ④	RANKING
PROBLEM DURING IN-ORBIT PROPELLANT TRANSFER IT IS NECESSARY TO INSURE THE ULLAGE AND LIQUID IS PROPERLY LOCATED WITHIN THE DELIVERY AND RECEIVER TANKS TO ALLOW ACCEPTABLE SUPPLIER TANK OUT-FLOW AND RECEIVER TANK IN FLOW	**PROS** • NO DEPLOYMENT OF TANK • ONE RENDEZVOUS & DOCKING PER TRANSFER • LOW PROPELLANT LOSS • COMPATIBLE WITH CURRENT MANIPULATOR ARMS	**PROS** • STANDARD TUG FILL & VENT LOCATION • LINEAR GAUGING • ONE RENDEZVOUS & DOCKING PER TRANSFER • LOW PROPELLANT LOSS	**PROS** • STANDARD TUG FILL & VENT LOCATIONS • LINEAR GAUGING • ONE RENDEZVOUS & DOCKING PER TRANSFER	**PROS** • STANDARD TUG FILL & VENT LOCATIONS • LINEAR GAUGING * INDEPENDENT DEPLOYMENT & TRANSFER CONCEPTS *SHUTTLE & CREW FREE DURING TRANSFER	PROPELLANT TRANSFER LOSS ① ② ④ ③ COMPATIBILITY ④ ① ② ③ DEVELOPMENT RISK ④ ① ② ③ SAFETY ④ ③ ② ①
EVALUATION CRITERIA PROPELLANT TRANSFER LOSS COMPATIBILITY OF DELIVERY AND RECEIVER VEHICLE DEVELOPMENT RISK	**CONS** *SIDE TANK VENTING OF TUG *NON-LINEAR GAUGING • INVERTED GROVED FILL & VENT *SHUTTLE & CREW RETAINED DURING TRANSFER	**CONS** • FLEXIBLE PROPELLANT LINE INTERFACE • COMPLEX DEPLOYMENT & SUPPORT *SHUTTLE & CREW RETAINED DURING TRANSFER • LONGER MANIPULATION ARMS	**CONS** *HIGH PROPELLANT LOSS • FLEXIBLE PROPELLANT LINE INTERFACE • COMPLEX DEPLOYMENT & SUPPORT *SHUTTLE & CREW RETAINED DURING TRANSFER LONGER MANIPULATION ARMS	**CONS** • MODERATE PROPELLANT LOSSES • FLUID INTERFACE DISCONNECTS • TWO RENDEZVOUS & DOCKING PER TRANSFER	**SELECTED APPROACH** LINEAR ACCELERATION SEPARATE ④ NO MAJOR DRIVER PRIMARILY BASED ON * ITEMS

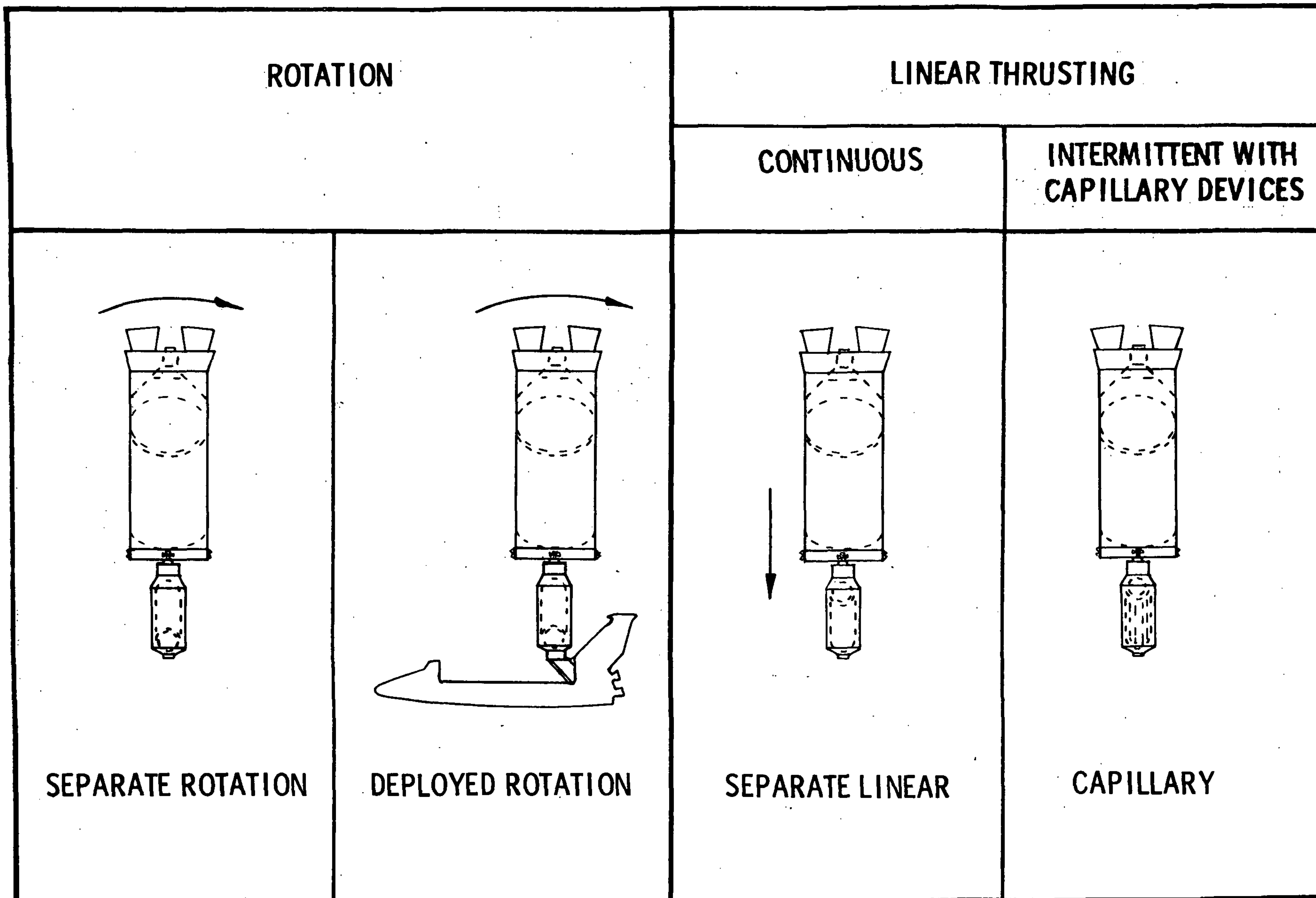


Figure 6.3.4-1 Alternatives-Direct Transfer to CIS

The size of the CIS and the need for multiple fillings to accomplish refueling make the overall transfer considerably more complicated than was the case with the tug. Problems involved include weight and moment of inertia change, storage between transfers, multiple docking and interconnects, and variation in refueling time with successive transfers.

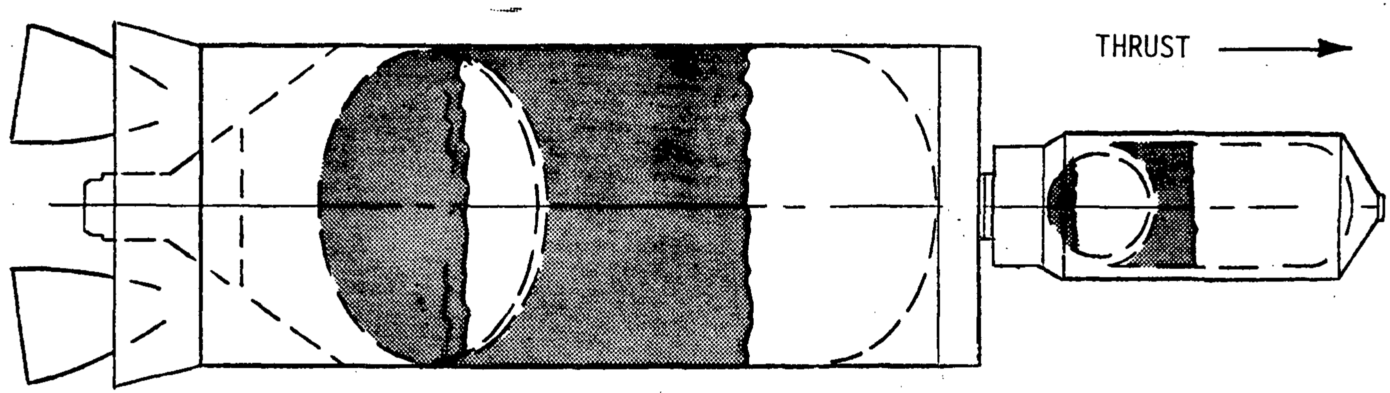
Rotational propellant transfer is attractive because total propellant losses attributed to transfer are low, primarily because propulsive thrusting is required only for spin-up and spin-down maneuvers. Successful propellant transfer using centrifugal forces to orientate and settle propellant depends on maintaining a favorable location of propellants with respect to the axis of rotation, defined by the location of the center of gravity. Center of gravity locations and liquid levels as functions of the CIS propellant loading are shown in Figure 6.3.2-29 for the separate rotation configuration (shuttle detached). In order to provide the most favorable c.g. location the LH_2 is filled from the forward end of the tank.

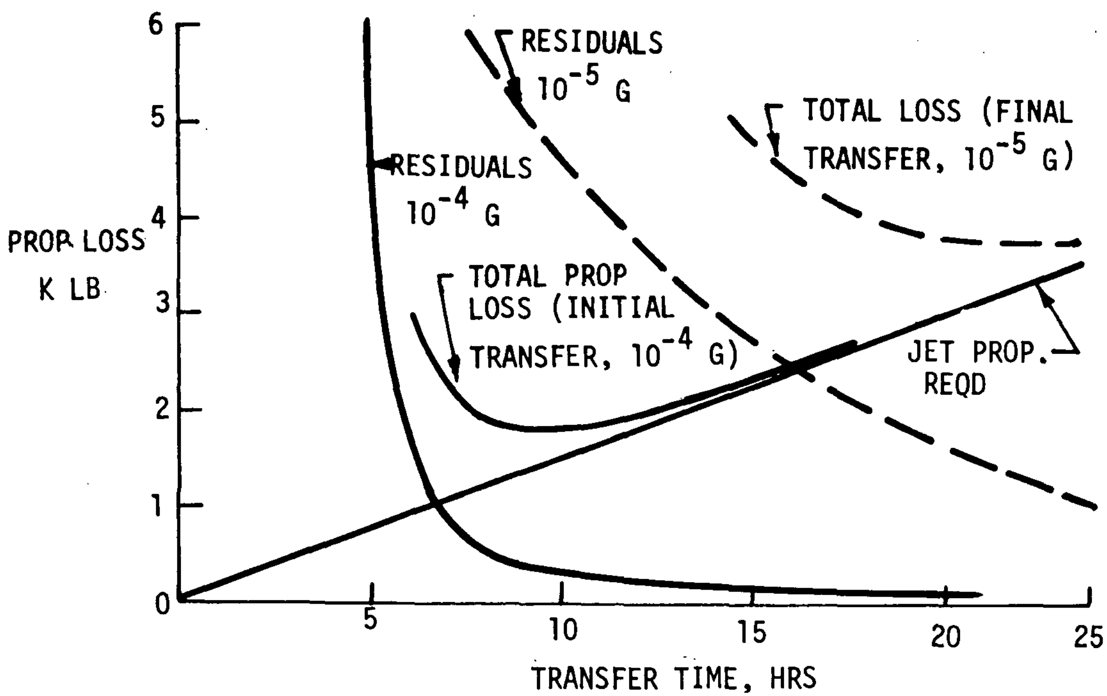
The primary disadvantages of this concept are: (1) reversed LH_2 fill and vent locations relative to normal propulsive operation, (2) large CIS ullage volumes are desirable to assure venting of gas, and (3) non-linear propellant gauging due to centrally located ullage. Additionally, liquid orientation after spin up from periods of zero g is unpredictable.

Locations of the center of gravity and liquid levels as functions of the CIS propellant loading for the deployed rotation transfer configuration are shown in Figure 6.3.2-30. As compared to separate rotation, the c.g. is shifted forward so that the LO_2 tank ullage always remains at the top of the tank. The LH_2 tank is initially filled from the bottom of the tank and as the c.g. location moves aft, filling is switched to the forward end. The same disadvantages listed for the separate rotation transfer configuration apply to the deployed rotation transfer concept.

For the separate linear transfer option, the logistic module is removed from the orbiter and mated with the CIS. To simplify propulsion requirements, a single engine system, providing a single thrust level, is proposed for each transfer. As the weight of propellant in the CIS increases with each transfer, the acceleration developed decreases with each transfer, encompassing approximately a six-fold range over the entire refueling. A thrust level which provides an acceleration range from approximately 10^{-4} g for the initial transfer to 10^{-5} g for the final transfer appears to represent the best compromise between propellant losses on the one hand and degree of interface control and transfer time on the other. On Figure 6.3.4-2 a propellant residual summarized for the initial and final transfer and jet propellant requirements are plotted versus transfer time. By summing residual propellant and jet propellant, both of which constitute losses, transfer times resulting in minimum losses are obtained. These preferred transfer times are about eight hours for the initial transfer and 20 hours for the final transfer.

Characteristics of this mode of transfer, in addition to those given above, include; two rendezvous and docking per transfer, the orbiter and crew are free during the transfer, and standard fill and vent locations can be utilized.





- TWO RENDEZVOUS & DOCKING PER TRANSFER
- SHUTTLE & CREW FREE DURING TRANSFER
- STANDARD FILL & VENT LOCATION
- CONSTANT THRUST FOR A EACH TRANSFER
- G LEVEL VARIES WITH PROPELLANT LOAD
- OPTIMUM TRANSFER TIME VARIES WITH LOAD

Figure 6.3.4-2 Separate Linear-Direct Transfer to CIS

Utilization of capillary devices to reduce thrusting requirements is an attractive option for propellant transfer to the CIS. The capillary system consists of capillary collector tubes in the logistic module tanks for propellant acquisition from any region of the tanks. The CIS requires fill and vapor/liquid interface control baffles to permit orderly filling and vapor return to the logistics tanks. These hardware provisions permit reduction of thrusting requirements. Thrusting is required for initial settling, intermittently during transfer to reposition dislocated propellant and for gauging. Thrusting may also be required continuously during the last few transfers because of the difficulty of obtaining propellant-free vapor return from a nearly full receiver tank. This sporadic thrusting approach reduces the total thrusting and maneuvering requirements and decreases jet propellant consumption. Another advantage of using this quasi-passive method of transfer is the compatibility of this approach with a wide range of receiver configurations (i.e., c.g. location is not as important if only limited maneuvers are involved).

Use of capillary devices introduces additional hardware complexity and development risk. Propellant dislocation or sloshing in the receiver tank could impair vapor return. An example of one of the development problems is the integration of capillary devices with thermal control requirements.

The four alternate methods for propellant transfer are compared on the basis of propellant transfer losses on Table 6.3.4-1. A transfer time of 15 hours was used for each case, as representing an overall average. Losses associated with pressure control were as in the case of the tug, insensitive to the transfer method, but, unlike the tug, these losses were not insignificant. However, APS propellant and liquid residuals were still the key determinants of transfer efficiency.

For the rotational methods the acceleration level was chosen to minimize APS propellant and liquid residual losses. For both separate and deployed rotation the losses were about the same. For the separate linear method, as thrust was constant, the acceleration level varied with propellant loaded. Losses are averaged over the different transfers. These losses were appreciably greater than those for the rotational methods. The large APS propellant requirements associated with the separate linear method may be substantially reduced by utilization of capillary devices. The capillary method reduced liquid residuals by about 30 percent, compared to the separate linear method. These improvements placed the use of capillary devices between the rotational and the separate linear methods in respect to propellant losses.

A breakdown of logistic program costs for the CIS supportive missions for program level D using separate linear propellant transfer is shown in Figure 6.3.4-3. Hardware and shuttle flight costs and their relative magnitudes are indicated. Of particular interest is the cost of the propellant transfer losses as a function of the propellant transfer configuration. A comparison of the cost of transfer losses and the percentage of total program cost that the losses represent is made for various transfer configurations. As indicated, the costs of the transfer losses for all configurations are small when compared with the total program costs. It is concluded that program costs are relatively insensitive to the propellant transfer configuration and that factors other than costs are the drivers influencing configuration selection.

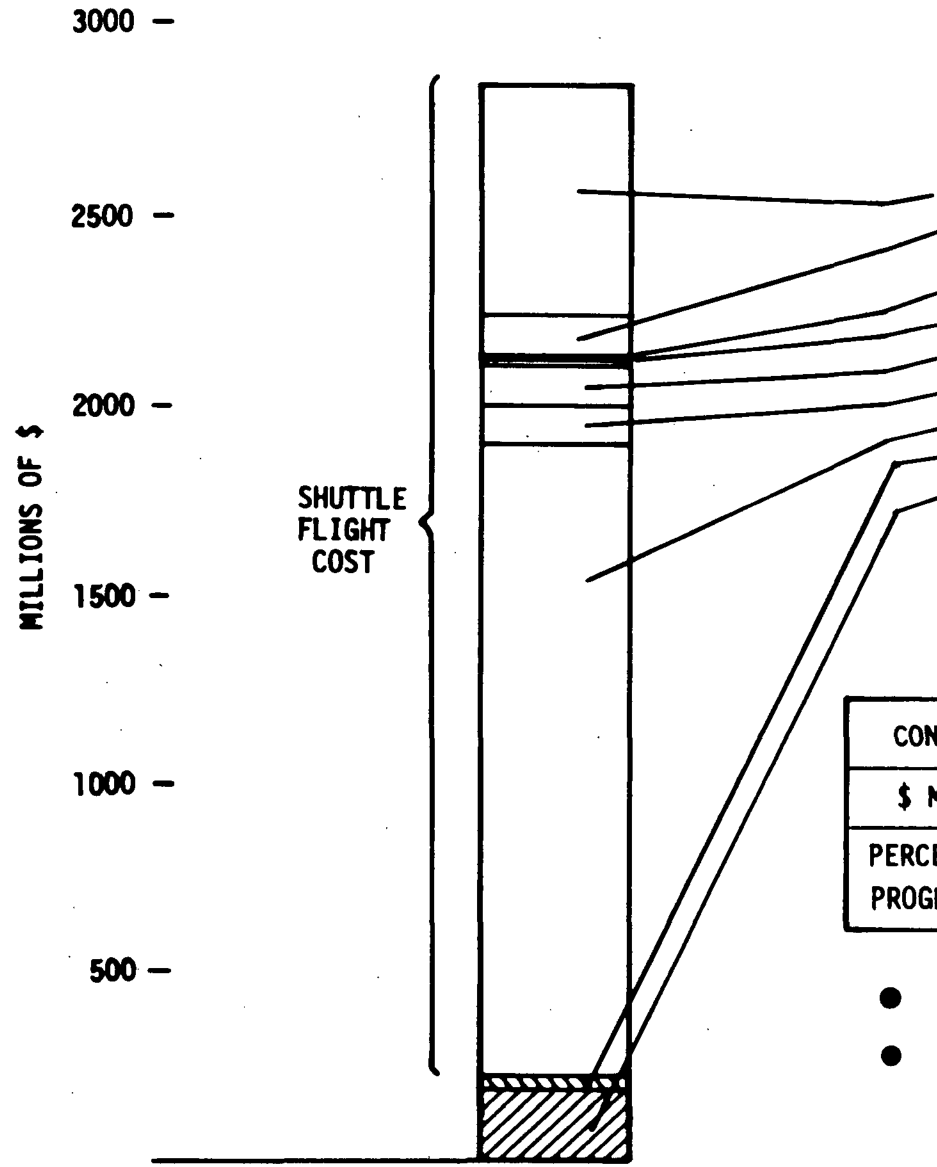
Table 6.3.4-1 Composite Propellant Transfer Losses for CIS

- BASED ON 15 HR AVERAGE TRANSFER TIME (8 TO 20 HRS)
- LOSSES IN % OF 60 K LBS
- LOSSES FOR AVERAGE TRANSFER

CONFIGURATION	SEPARATE ROTATION	DEPLOYED ROTATION	SEPARATE LINEAR	CAPILLARY
ACCELERATION, G'S	4×10^{-4}	2×10^{-4}	$10^{-4}/10^{-5}$	$0/10^{-3}$
APS PROPELLANT	.2	.2	2.9	0.3
LIQUID RESIDUALS	.2	.2	1.7	1.2
NPSP CONTROL	1.0	1.0	1.0	1.0
OTHER*	0.1	0.1	0.1	0.1
TOTAL	1.5	1.5	5.7	2.6

*PUMPING POWER, LINE RESIDUALS, LINE CHILLDOWN, AND HEAT LEAK

- PROGRAM LEVEL D
- 1985 - 1990



COST OF TRANSFER LOSSES

CONFIGURATION	SEPARATE ROTATION	DEPLOYED ROTATION	SEPARATE LINEAR	CAPILLARY
$ MILLIONS	28	28	106	49
PERCENT OF TOTAL PROGRAM COST	1.0	1.0	3.7	1.7

- LOSSES ARE SMALL PERCENTAGE OF PROGRAM COSTS
- COSTS ARE RELATIVELY INSENSITIVE TO TRANSFER CONCEPT

Figure 6.3.4-3. Logistic Program Costs For Direct Transfer to CIS

A significant drawnback common to both rotational and the capillary concepts defined is position orientation of the receiver tank ullage to assure liquid free venting. This drawback could be alleviated by providing a receiver tank thermodynamic control concept which could perform efficiently by the venting of liquid or gas. Use of the user vehicle's currently proposed thermodynamic vent system provides just this advantage. The operation of the system and its potential application to the propellant transfer operation is discussed in Section 6.4, Receiver Tank Thermodynamic Control.

Use of a slightly modified version (mixers and increased flow rate) of this currently proposed user vehicle system would significantly enhance the feasibility and practicability of the proposed rotational and capillary liquid/vapor interface control concepts. For this reason, the following trade study comparisons are based on the assumption that thermodynamic venting would be the receiver tank thermodynamic control mode. These concept combinations are then compared with linear acceleration using gas return to source tank since this mode provides positive liquid/vapor interface control in both the source and receiver tanks.

Table 6.3.4-2 presents advantages and disadvantages of the four propellant transfer liquid/vapor interface control methods for CIS. Those considerations which were key drivers in the choice of the baseline are indicated by an asterisk. Both rotational methods had the disadvantage of non-linear gauging (because of the shape and location of the ullage bubble) and the requirement for bulk liquid mixers. The latter impacts the CIS design and operational complexity. Although the capillary method has definite propellant loss and operational advantages, it suffers from several disadvantages; **development risk and integration of the capillary devices with thermal** control provisions.

The technique of separate linear acceleration with gas return to the source tank was chosen as the study baseline. Advantages such as simplified propellant gauging, no need for mixers, and no impact on the CIS thermodynamic vent system are believed to appreciably outweigh the disadvantages of moderately high propellant losses, an additional rendezvous and docking per transfer, and variable acceleration levels and transfer times with successive transfers.

The method of propellant transfer illustrated as separate linear on Figure 6.3.4-1 was chosen as the study baseline, and the subject for further investigation. This method employed continuous linear acceleration in the range 10^{-4} to 10^{-5} g for vapor/liquid interface control. Vapor return to the source tank provides efficient and, what is believed to be, reliable thermodynamic control during transfer.

The primary bases for this choice were simplified propellant gauging, no impact on the user thermodynamic vent system, and the minimum development risk.

Work done on the reusable nuclear stage revealed similar considerations and resulted in the same baseline choice.

Table 6.3.4-2 Liquid/Vapor Interface Control Trade Table - CIS

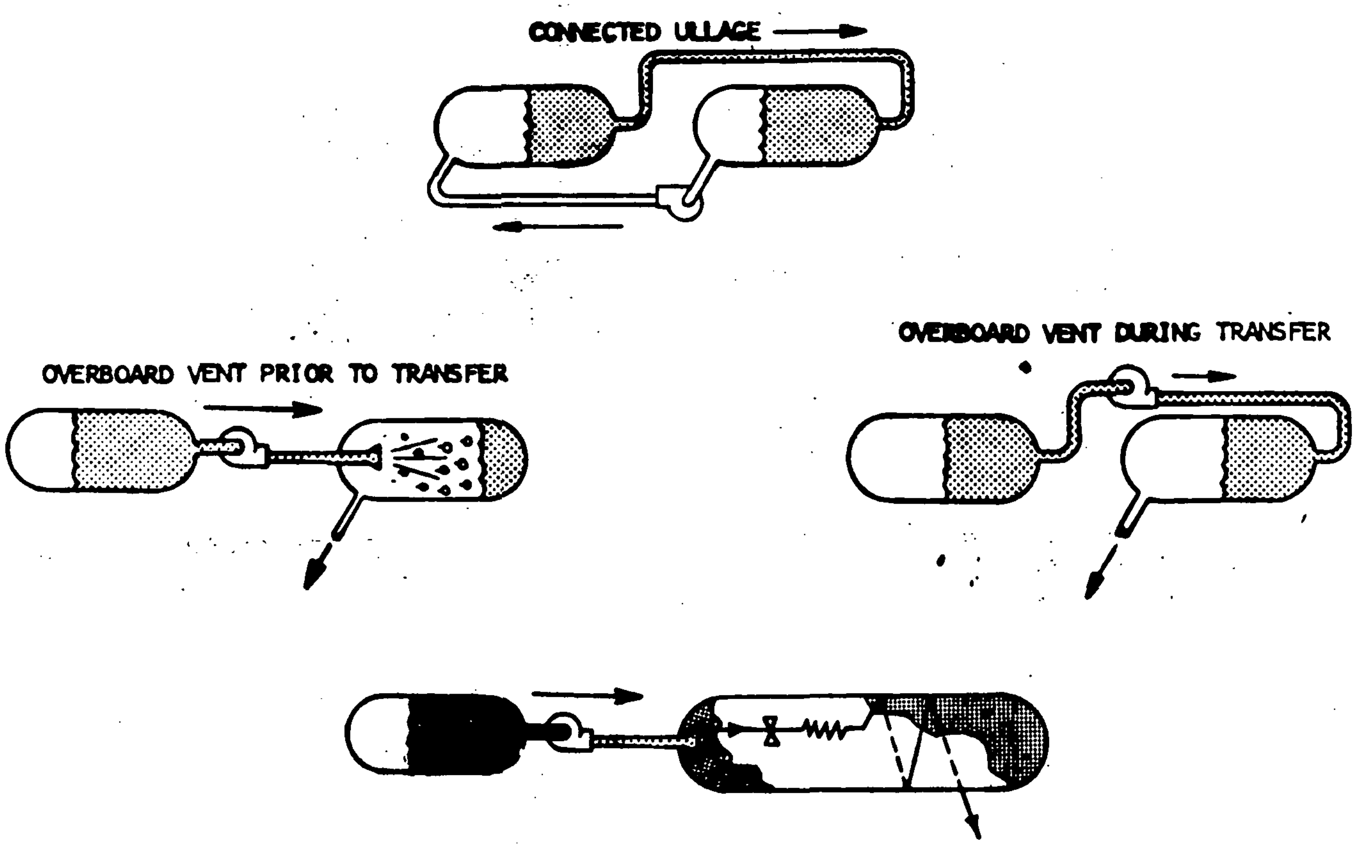
	ALTERNATE APPROACHES				
REQUIREMENT	SEPARATE ROTATION 1	DEPLOYED ROTATION 2	SEPARATE LINEAR 3	CAPILLARY 4	RANKING
PROBLEM DURING IN-ORBIT PROPELLANT TRANSFER IT IS NECESSARY TO INSURE THE ULLAGE AND LIQUID IS PROPERLY LOCATED WITHIN THE DELIVERY AND RECEIVER TANKS TO ALLOW ACCEPTABLE SUPPLIER TANK OUTFLOW AND RECEIVER TANK INFLOW	PROS • LOW PROPELLANT LOSSES • SHUTTLE & CREW FREE DURING TRANSFER	PROS • ONE RENDEZVOUS & DOCKING PER TRANSFER • LOW PROPELLANT LOSSES	PROS *LINEAR PROPELLANT GAUGING *NO IMPACT ON USER THERMODYNAMIC VENT SYSTEM • BULK LIQUID MIXERS NOT REQUIRED • MINIMUM DEVELOPMENT RISK	PROS • REDUCED PROPELLANT LOSSES • APS NOT REQUIRED ON LOGISTICS TANK • SHUTTLE MAX OR MAY NOT REMAIN	PROPELLANT TRANSFER LOSS 1 2 4 3 COMPATIBILITY 3 2 1 4 DEVELOPMENT RISK 3 1 2 4 SAFETY 3 1 4 2
EVALUATION CRITERIA PROPELLANT TRANSFER LOSS COMPATIBILITY OF DELIVERY AND RECEIVER VEHICLE DEVELOPMENT RISK SAFETY	CONS *NON-LINEAR GAUGING *BULK LIQUID MIXERS • IMPACT ON THERMODYNAMIC VENT SYSTEM • INCREASED DEVELOPMENT RISK	CONS *NON-LINEAR GAUGING • SHUTTLE & CREW RETAINED DURING TRANSFER • INCREASED DEVELOPMENT RISK	CONS • MODERATELY HIGH PROPELLANT LOSSES • TWO RENDEZVOUS & DOCKINGS PER TRANSFER • VARIABLE ACCELERATION LEVELS & TRANSFER TIMES	CONS • INTERNAL TANK HARDWARE COMPLEXITY *INTEGRATION WITH THERMAL CONTROL • IMPACT ON USER THERMO-VENT SYSTEM	SELECTED APPROACH SEPARATE LINEAR 3 NO MAJOR DRIVER PRIMARILY BASED ON * ITEMS

6.4 RECEIVER TANK THERMODYNAMIC CONTROL

6.4.1 Candidate Concepts

During cryogenic propellant transfer some form of thermodynamic control of the receiver tank is required. This control is necessary to avoid exceeding receiver tank pressure limits and to control propellant temperature within the band required by the using vehicle. The receiver tank will normally contain some propellant residual in a liquid or gaseous state. In addition, gas will be generated in the receiver tank when the relatively warm tank walls give up heat to the incoming cryogenic propellant. This residual and generated gas must be displaced, compressed, or condensed as propellant continues to enter the tank. The resultant temperature of the propellant transferred will depend on the initial conditions and the performance of the thermodynamic control concept employed.

Thermodynamic control concepts evaluated during the study include (1) overboard vent during transfer, (2) connected ullage, (3) overboard vent prior to transfer, and (4) use of a thermodynamic vent system. These concepts are shown schematically as follows:



THERMODYNAMIC VENT

6.4.2 Discussion of Candidate Concepts

Direct overboard venting to space is perhaps the simplest method available for receiver tank thermodynamic control. The method provides receiver tank pressure relief by venting the displaced gas directly overboard. The venting is accomplished through a vent valve automatically modulated to maintain the correct receiver tank pressure. Some form of liquid/vapor interface control is required with this method to preclude the loss of liquid propellant during the vent process.

An analysis was conducted to determine the weight of propellant lost by overboard vent during fill for tug and RNS. Results of the analysis are presented in Figure 6.4.2-1. It can be seen from the data that vented mass losses can be decreased considerably in most cases by an increase in the ullage temperature. Therefore, the losses can be reduced with a slow laminate fill allowing ullage gas temperature stratification. The gas vented would then be at a maximum temperature above the liquid. Conversely, a relative fast fill with appreciable liquid/vapor interface distortion and ullage gas mixing would increase the propellant weight loss.

Additional losses would be incurred by the generation of the vapor that must be supplied to the source tank to displace the effluxing liquid. One way of conserving propellant during a transfer is by returning the vapor displaced in the receiver tank back to the ullage of the supply tank. This not only avoids venting the receiver tank gas to space, but reduces the need for vapor generation to maintain pressure in the supply tank. The vapor return concept also allows the liquid vaporized during the transfer line and tank chilldown operation to be used to supplement ullage pressurization. This will reduce the losses associated with the pressurization system. Analyses conducted during the Orbital Propellant Storage System Feasibility Study (Reference 6.0-2) showed that a significant contribution to the mass required for ullage pressurization can be made by utilizing the propellant vaporized by receiver tank chilldown. The magnitude of the contribution is, of course, dependent on the initial temperature of the receiver tank. In some cases, the using vehicle will have some liquid residual from the last mission, and thus require a minimal chill. The savings of chill gas can be fully realized only if the vapor return concept is used for receiver tank thermodynamic control.

The overboard vent prior to transfer concept was evaluated relative to propellant losses. This process eliminates the need for liquid/vapor interface control in the receiver tank. If a receiver tank can be filled without venting, the need for propellant positioning in the receiver tank is eliminated and thus no propellant is needed for thrust; however, propellant positioning in the delivery tank (bladder or capillary device) is required. Consequently, a thermodynamic analysis was performed for a receiver tank fill process to determine the possibility of receiver tank pressures remaining within acceptable limits without venting. Receiver tank pressure was plotted as a function of percentage liquid by volume in the receiver tank during the fill process without reference to the absolute volume of the tank. It was assumed that the receiver tank residuals were dumped and/or vented to space prior to the initiation of transfer thus resulting in the tank contents being initially near the triple point pressure. Propellant from the delivery tank saturated at 14.7 psia was

PROPELLANT LOST BY OVERBOARD VENT DURING FILL

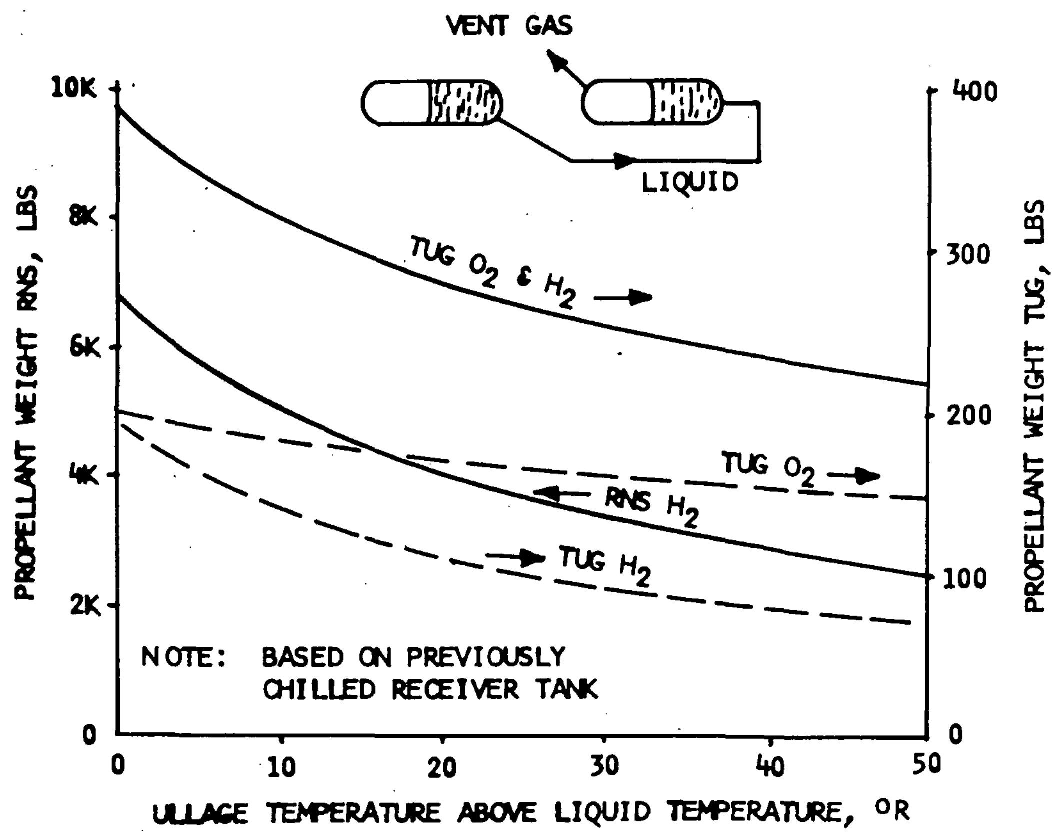


Figure 6.4.2-1 Overboard Vent During Transfer

also assumed. The results, presented on Figure 6.4.2-2 show that the receiver tank pressures will not exceed 16 psia for a tank (LO_2 or LH_2) filled to 95 percent capacity. For the case analyzed, pressurization expulsion or pump expulsion is accomplished at a pressure of 22 psia. Spray fill nozzles at either end of the receiver tank or mixing devices within the receiver tank are required to ensure good heat transfer between the tank contents and incoming fluid. As the first particles of liquid enter the receiver, a part of the liquid will flash off into vapor, thus cooling the remainder of the entering liquid to some common stabilization pressure. As the fill operation proceeds, the receiver tank pressure will continue to increase gradually. The thermodynamic principle utilized in the analysis is that the internal energy of the final fluids in the tank is equal to the internal energy of the initial fluids of the tank plus the enthalpy of the incoming liquid.

In summary, it is theoretically feasible to transfer a cryogenic propellant to an almost empty tank in a zero-g environment without venting if a slight increase in propellant vapor pressure (approximately 1 psi) is acceptable. However, multiple transfers may be required for tug, CIS, and RNS to obtain sufficient propellant to conduct the mission, and tank venting between loads for thermodynamic control would create substantial propellant losses.

Use of the receiver vehicle's existing thermodynamic control was analyzed during the S-II Orbital Propellant Storage System Feasibility Study (Reference 6.0-2) for the case of a complete user vehicle filling during a single transfer from a depot. For this case, the flow capacity of the user vehicle thermodynamic vent system was too low to allow propellant transfer in a reasonable length of time. For the case of propellant transfer directly from a logistic module to the CIS, 19 transfers are required over a period of 80 days. This longer fill time will allow propellant temperature and pressure control at a reduced vent flow rate. For this reason, a re-assessment has been made of the possibility of using the user vehicle's existing thermodynamic vent system for transfer operations.

The pressure traces shown on Figure 6.4.2-3 were based on propellant mixing, with the energy of compression and condensation of the displaced ullage going into the bulk of the liquid and raising the vapor pressure. The cooling rates were those required to condense exactly the displaced ullage over a period of 96 hours. When the tanks are nearly empty, mixing is not essential because the overall pressure rise due to compression is not great; but when the tanks are nearly fully, periodic mixing is necessary to condense the displaced ullage or the overall pressure rise becomes prohibitively large.

The most likely application of filling with a closed vent would be for the LO_2 system where the receiver tank ullage gas could be completely conserved through condensation by cooling with the hydrogen vent gas. This application would be of especial interest for a zero-g fill operation, or for a rotational acceleration method where the center of gravity falls within the LO_2 tank. For a hybrid system, which involves the LO_2 tank only for closed vent filling, there would be no penalty for the LH_2 thermodynamic vent system (TVS). Approximately 300 pounds of LO_2 could be saved at the expense of only 150 pounds for a helium expulsion system on the logistics tanks, yielding a net savings of

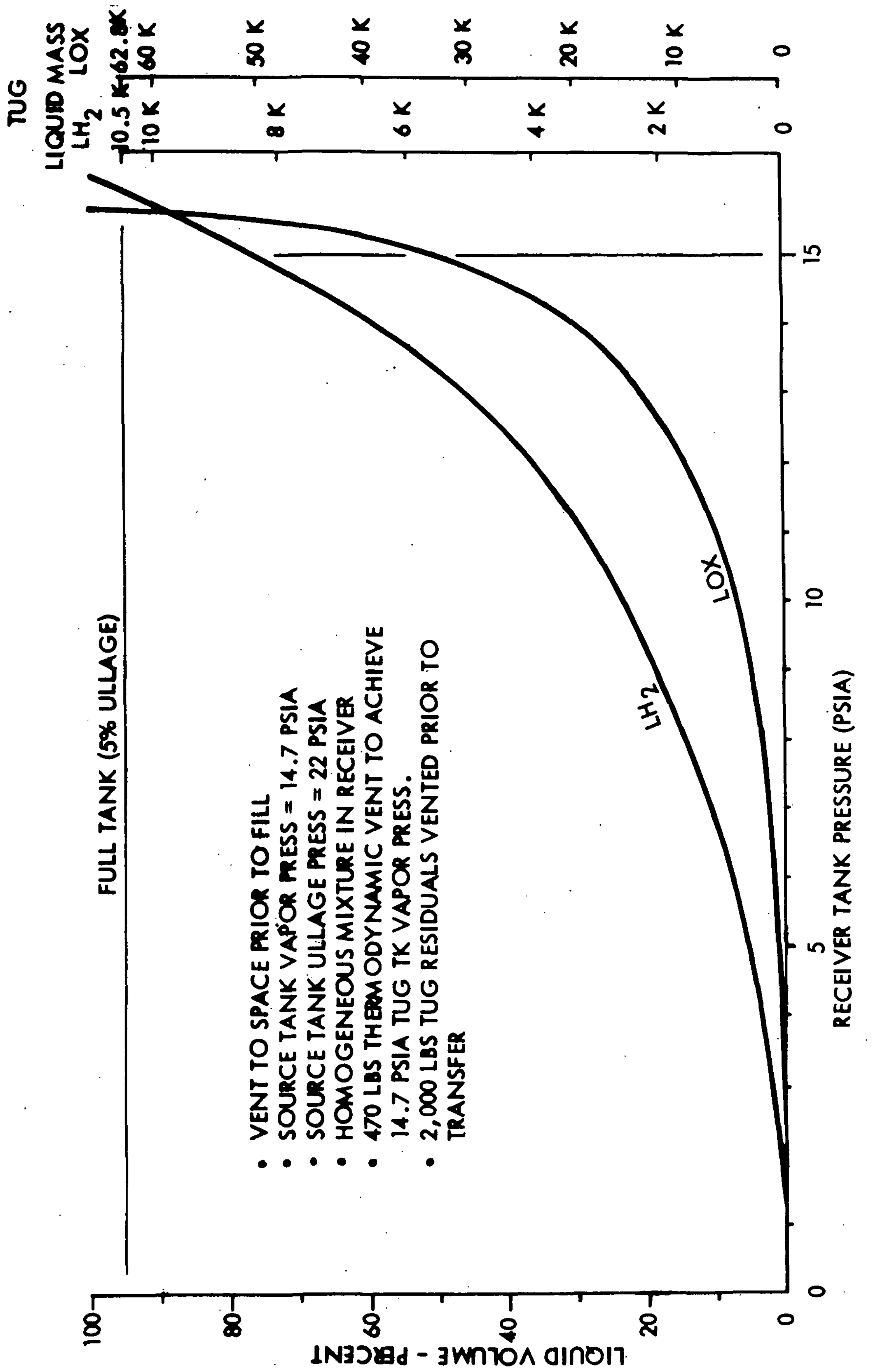
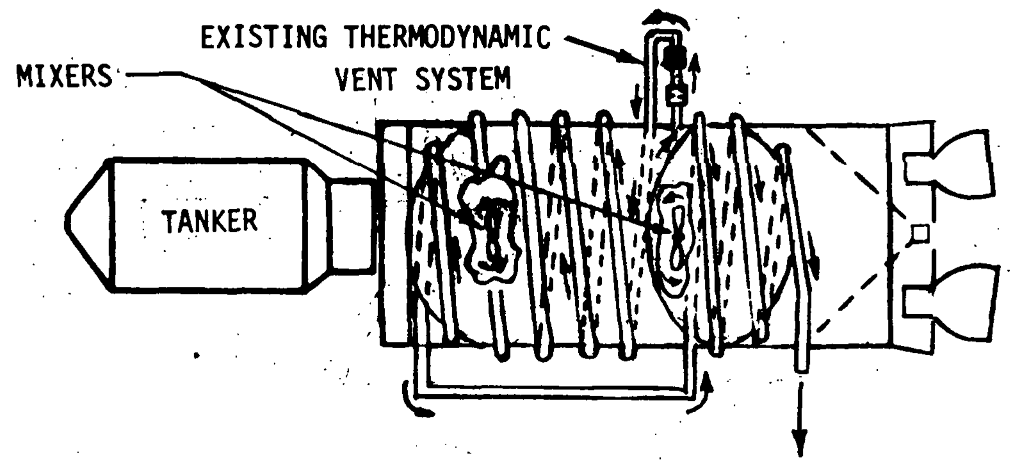
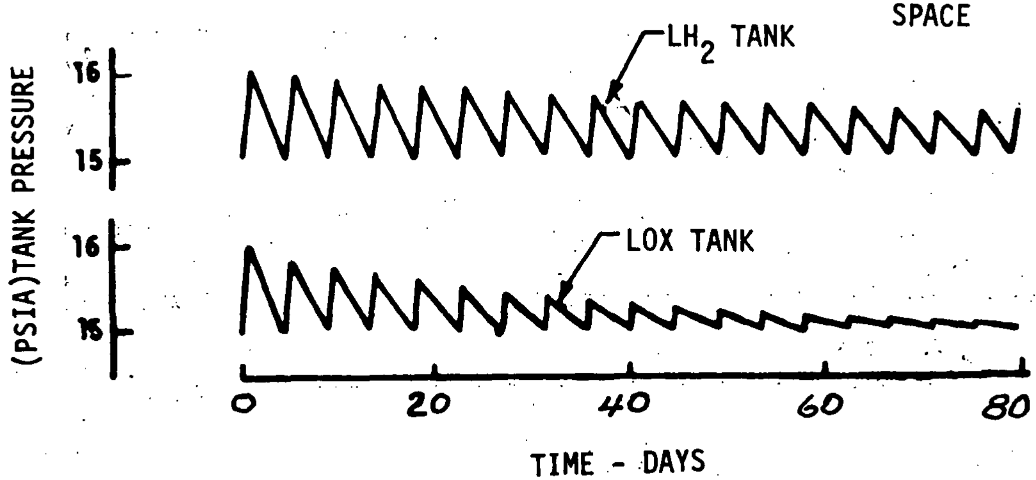


Figure 6.4.2-2 Overboard Vent Prior to Transfer





- EXISTING USER VEHICLE THERMODYNAMIC VENT SYSTEM USED
- EFFICIENT VENTING OF LIQUID OR GAS
- FEASIBLE DUE TO LONG PERIODS BETWEEN TRANSFERS
- MIXERS REQUIRED TO AVOID EXCESSIVE TANK PRESSURE
- NO VENTING OF OXYGEN

Figure 6.4.2-3 Thermodynamic Venting During Transfer

150 pounds per transfer. For application to the hydrogen tank, there would be approximately a 20-percent increase in flow requirements for the TVS, or a 200-pound net loss of hydrogen per orbiter load.

The results of this analysis show that the use of the existing user vehicle thermodynamic vent system during the transfer operation is feasible for the case where a complete fill requires several partial transfers over an extended period of time. Use of this concept improves the feasibility of those rotational and capillary liquid/vapor interface control concepts which do not provide positive or predictable propellant orientation in the receiver tank. However, use of this concept would introduce some user vehicle impact due to the additional requirement for bulk liquid mixers, increased thermodynamic vent system flow capability, and increased vent system control complexity.

6.4.3 Comparison of Candidate Concepts

The four thermodynamic control concepts for direct transfer of propellants have been evaluated on the basis of propellant losses, compatibility, development risk and safety. The evaluation is summarized on the Trade Study, Table 6.4.3-1. The connected ullage and thermodynamic vent provide the least propellant losses for the multiple delivery loads required. If multiple loads were not required the losses for overboard vent prior to transfer would be competitive. Overboard vent during transfer is simplest and has been used extensively, therefore rates best in development risk; however, the connected ullage is very close. All concepts are compatible with the selected liquid/vapor interface control concept; however, only the connected ullage is compatible with the selected expulsion concept. The connected ullage is also considered safest since no additional gas system (storage bottle or gas generator system) is required to pressurize the delivery tank during transfer.

The connected ullage was selected as the baseline primarily on the basis of minimum propellant loss, system simplicity, and compatibility with the expulsion subsystem selected.

6.5 EXPULSION

6.5.1 Candidate Concepts

Expulsion is the propellant transfer subsystem for forcing the propellant out of the source tank. Concepts for propellant expulsion that were evaluated included:

- Liquid pump expulsion
- Gas pump expulsion
- Liquid-to-gas conversion pressure expulsion
- Stored gas pressure expulsion
- Positive displacement

These concepts are presented schematically in Figure 6.5.1-1.

Table 6.4.3-1 Thermodynamic Control Concept Trade Table

REQUIREMENTS	ALTERNATE APPROACHES				SELECTION
	OVERBOARD VENT DURING TRANSFER (1)	CONNECTED ULLAGE (2)	OVERBOARD VENT PRIOR TO TRANSFER (3)	THERMODYNAMIC VENT (4)	
THERMODYNAMIC CONTROL OF THE RECEIVER TANK IS REQUIRED TO AVOID EXCEEDING RECEIVER TANK PRESSURE LIMITS AND CONTROL PROPELLANT TEMPERATURE	PROS NO GAS RETURN LINE REQUIRED	PROS MINIMUM PROPELLANT VENT LOSS PROVIDES SOURCE TANK LIQUID DISPLACEMENT LIQUID/VAPOR INTERFACE CONTROL NOT CRITICAL	PROS NO RECEIVER TANK LIQUID/VAPOR INTERFACE CONTROL REQUIRED	PROS NO RECEIVER TANK LIQUID/VAPOR IN SURFACE CONTROL REQUIRED	PROPELLANT TRANSFER LOSS (2) (4) (3) (1) COMPATIBILITY (1) (2) (4) (3) DEVELOPMENT RISK (1) (2) (3) (4) SAFETY (2) (3) (1) (4)
EVALUATION CRITERIA PROPELLANT TRANSFER LOSS COMPATIBILITY OF DELIVERY AND RECEIVER VEHICLE DEVELOPMENT RISK SAFETY	CON SOURCE TANK GAS SUPPLY REQUIRED FOR LIQUID DISPLACEMENT LIQUID/VAPOR INTERFACE CONTROL CRITICAL	CON LIQUID/VAPOR INTERFACE CONTROL REQUIRED FOR RECEIVER ADDITIONAL LINE INTERFACES	CON LOSS OF USER VEHICLE INITIAL PROPELLANT RESIDUALS GOOD FLUID MIXING REQUIRED SOURCE	CON BULK LIQUID MASSES INCREASED FLOW INCREASE SYSTEM CONTROL COMPLEXITY	SELECTED APPROACH CONCEPT SELECTED (2) MAINLY ON BASIS OF MINIMUM PROPELLANT LOSS

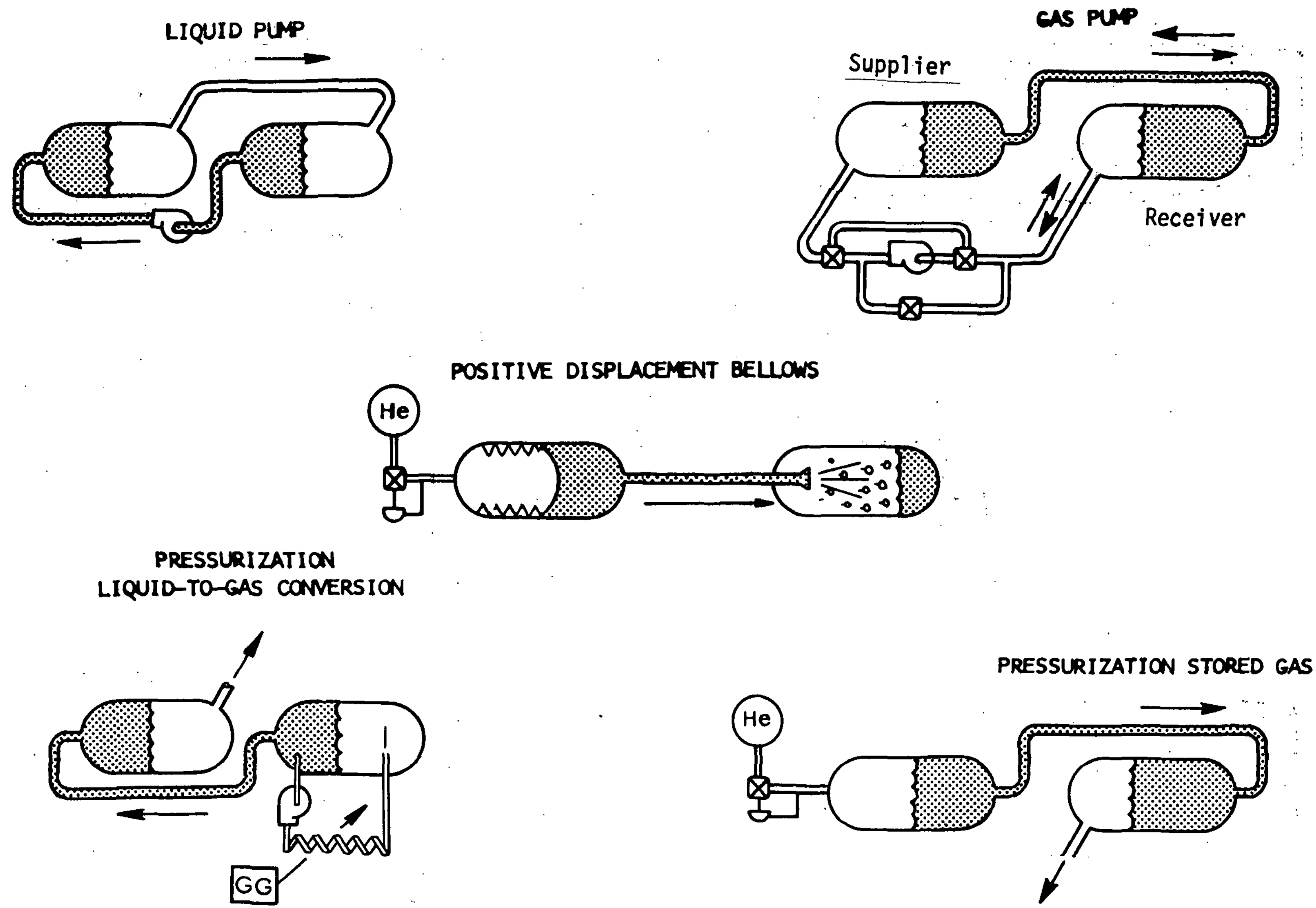


Figure 6.5.1-1 Liquid Expulsion Concepts

The liquid pump expulsion system consists of a pump located in the extreme exit portion of the propellant tank or in the propellant transfer line between the tanker and receiver vehicles. The pump is driven by some power system such as a gas generator and turbine or fuel cell and electric motor. The gas pump expulsion system has a blower in the ullage connecting line that provides the differential pressure between the two tanks. Advantages of this system are the ease of reversing the operation by valving and exterior location of pump for maintenance. The pressure expulsion system uses ullage pressure for the primary propellant transfer force. A pump to supply secondary propellant flow through a gas generator and heat exchanger unit is necessary to provide the required ullage pressure for forcing the propellant into the receiver vehicle for the liquid-to-gas conversion system. Independent stored gas bottles and control valves are used in the stored gas system. Positive displacement using a bladder or metallic bellows offers the advantages of physical separation of vapor and liquid and low residuals. The main disadvantages of bladders for cryogenic applications are materials development, cycle life, and permeability. Metallic bellows are superior to bladders with regard to the above problems. However, they are subject to such problems as vapor formation due to heat leakage, high weight, difficulty of purging convolutions and incompatibility with topping during ground fill.

6.5.2 Discussion of Candidate Concepts

Liquid and gas pump expulsion systems were evaluated for propellant losses and power requirements. Propellant losses considered included propellant required for pump power, pressurization, depressurization, thrust development, and gas generation. Figure 6.5.2-1 shows the parametric hydrodynamic characteristics for a 15-hour LH_2 logistic tank to tug liquid pump transfer concept. The significant penalty factors, pumping power, line residual, line weight, and line boil-off, are presented as a function of transfer line diameter. As shown on Figure 6.5.2-1, the penalty is only 0.02 percent for a line size of approximately 0.8 inch diameter. Attention is called to the pumping power curve which indicates that the pumping power loss is insignificant for larger size lines; however, boil-off weight and residual are significant. These data indicate that line sizes must be optimized to minimize losses.

The power requirement for propellant transfer is a function primarily of propellant density, propellant flow rate and transfer line size. The following equation was used for obtaining power.

$$Hp = \frac{0.95\ f\ L\ \dot{W}^3}{w^2\ D^5 \eta}$$

Hp = Horse power

P = .02 line friction factor

L = Line equivalent length (inch)

$\dot{W}$ = Liquid propellant flow rate (lb/sec)

w = Liquid propellant density (lb/ft^3)

D = Line diameter (inch)

η = 50 percent pump and motor combined efficiency

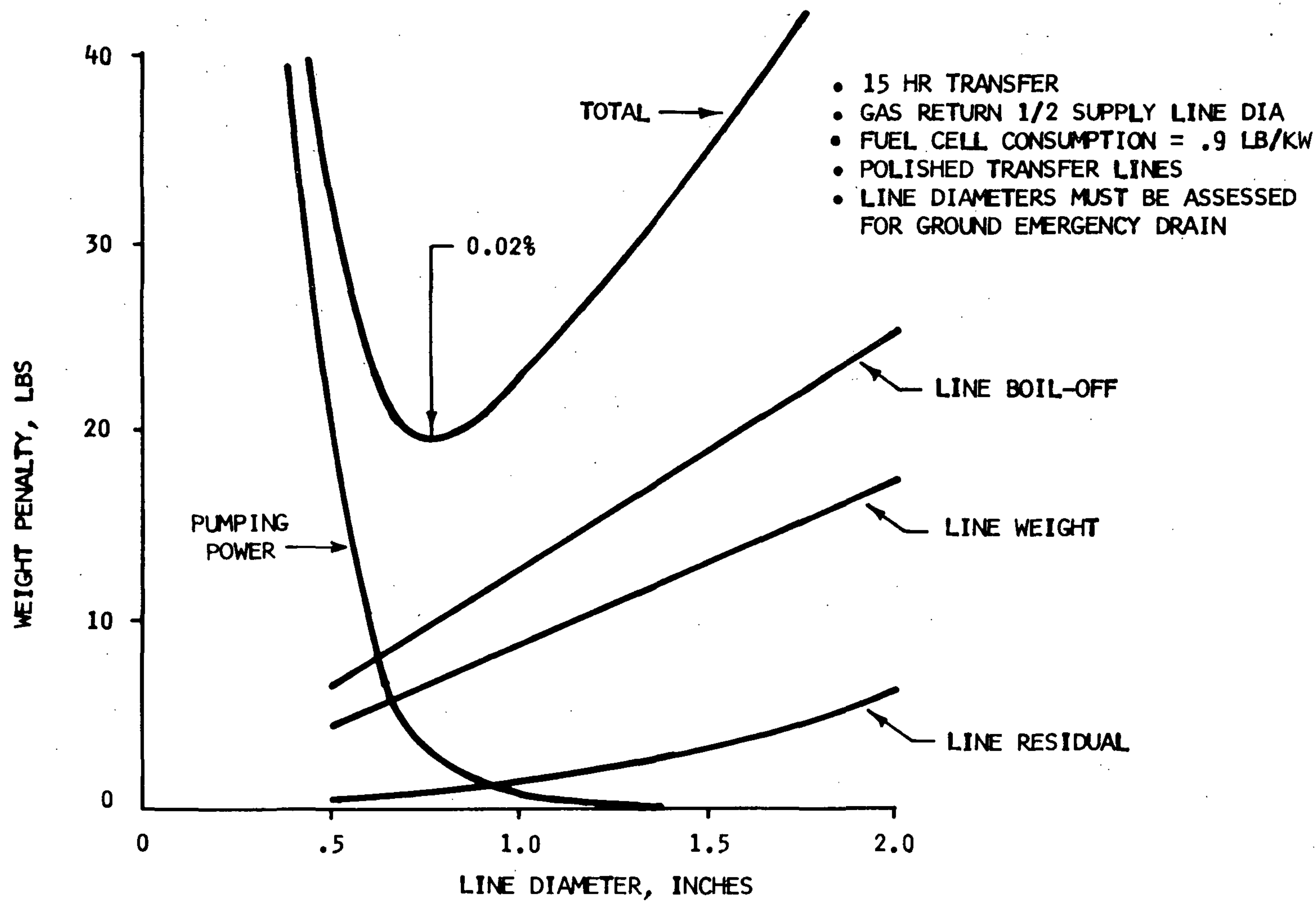


FIGURE 6.5.2-1 Logistic Tank-to-Tug LH_2 Transfer Line Diameter

Table 6.5.2-1 was constructed from baseline propellant transfer times lengths, propellant loading and transfer times, as noted for the various vehicles. Since the transfer pump flow must be throttled at the start of transfer for chilldown and at the end of transfer for minimizing residuals, the following assumptions were made:

1. Eighty percent of the full propellant load is transferred at fast fill rate

2. Twenty percent of the full propellant load is transferred at the throttled fill rate

3. Throttled fill rate is 10 percent of fast fill rate

The horsepower calculations were made from Table 6.5.2-1 as indicated. The horsepower values, tabulated as a function of line diameter for the different vehicles, were plotted on Figures 6.5.2-2 and 6.5.2-3 for LH_2 and LO_2 transfers, respectively. The baseline diameters were selected as indicated.

A weight penalty comparison of oxygen pressurization expulsion, helium pressurization expulsion, and pump expulsion for a 15-hour logistic tank to tug transfer is presented in Figure 6.5.2-4. The data indicate that the pump concept for LO_2 transfer has the lowest weight penalty and oxygen (gas) pressurization transfer has the highest penalty. Although not presented here, similar data for the LH_2 system show the pump with the lowest and the helium pressurization with the highest penalty.

The most attractive features of a positive displacement concept are no liquid/vapor interface control is required and it has the potential of reducing the liquid residuals to a minimum. However, the number of fabrication and operational disadvantages associated with its use make this concept unacceptable. Some of these disadvantages are: large sizes are difficult to manufacture (equipment and technique are available for sizes over 40 inches in diameter), hardware weight is high and, as listed on Figure 6.5.2-5, an unusually high number of compatibility and operational problems are anticipated.

6.5.3 Comparison of Candidate Concepts

Data were evaluated and trades made relative to the basic evaluation criteria of transfer loss, compatibility of delivery and receiver tanks, development risk and safety. The results of the evaluation are presented in Table 6.5.3-1. As indicated on the table the gas pump concept has been selected as the baseline expulsion concept. This concept was selected primarily because of its reversing capability, ease of maintenance, and transfer loss characteristics. Some development work will be necessary to develop the blower so it will stall at the receiver tank over-pressure limit. However, this is considered well within the current state of the art.

Table 6.5.2-1 Propellant Transfer Data

LH$_2$ TRANSFER

	TUG	CIS	RNS
TRANSFER TIME (T)	10 HRS	15 HRS	15 HRS
PROPELLANT LOAD (W)	10 K-LBS	10 K-LBS	34 K-LBS
LINE EQUIV. LENGTH $\left(\frac{L}{T2}\right)$	150 FT	200 FT	275 FT
PROPELLANT FAST FLOW RATE ($\dot{W}$)	.78 LB/SEC	.52 LB/SEC	1.76 LB/SEC
PROPELLANT FAST FLOW RATE (3600 $\dot{W}$)	2800 LB/HR	1870 LB/HR	6350 LB/HR
($\dot{W}^3$)	.47	.14	5.5
$\text{h.p.} = \frac{1.95 \times 10^{-3} L\dot{W}^3}{D^5}$	$\frac{1.65}{D^5}$	$\frac{.655}{D^5}$	$\frac{35.4}{D^5}$

LOX TRANSFER

	TUG	CIS
TRANSFER TIME (T)	10 HR	15 HRS
PROPELLANT LOAD (W)	50 K-LB	50 K-LB
LINE EQUIV. LENGTH $\frac{L}{T2}$	150 FT	200 FT
PROPELLANT FLOWRATE ($\dot{W}$)	3.9 LB/SEC	2.6 LB/SEC
(3600 $\dot{W}$)	14,000 LB/HR	9350 LB/HR
$\dot{W}^3$	59	17.5
$\text{h.p.} = \frac{7.55 \times 10^{-6} L \dot{W}^3}{D^5} =$	$\frac{.8}{D^5}$	$\frac{.32}{D^5}$

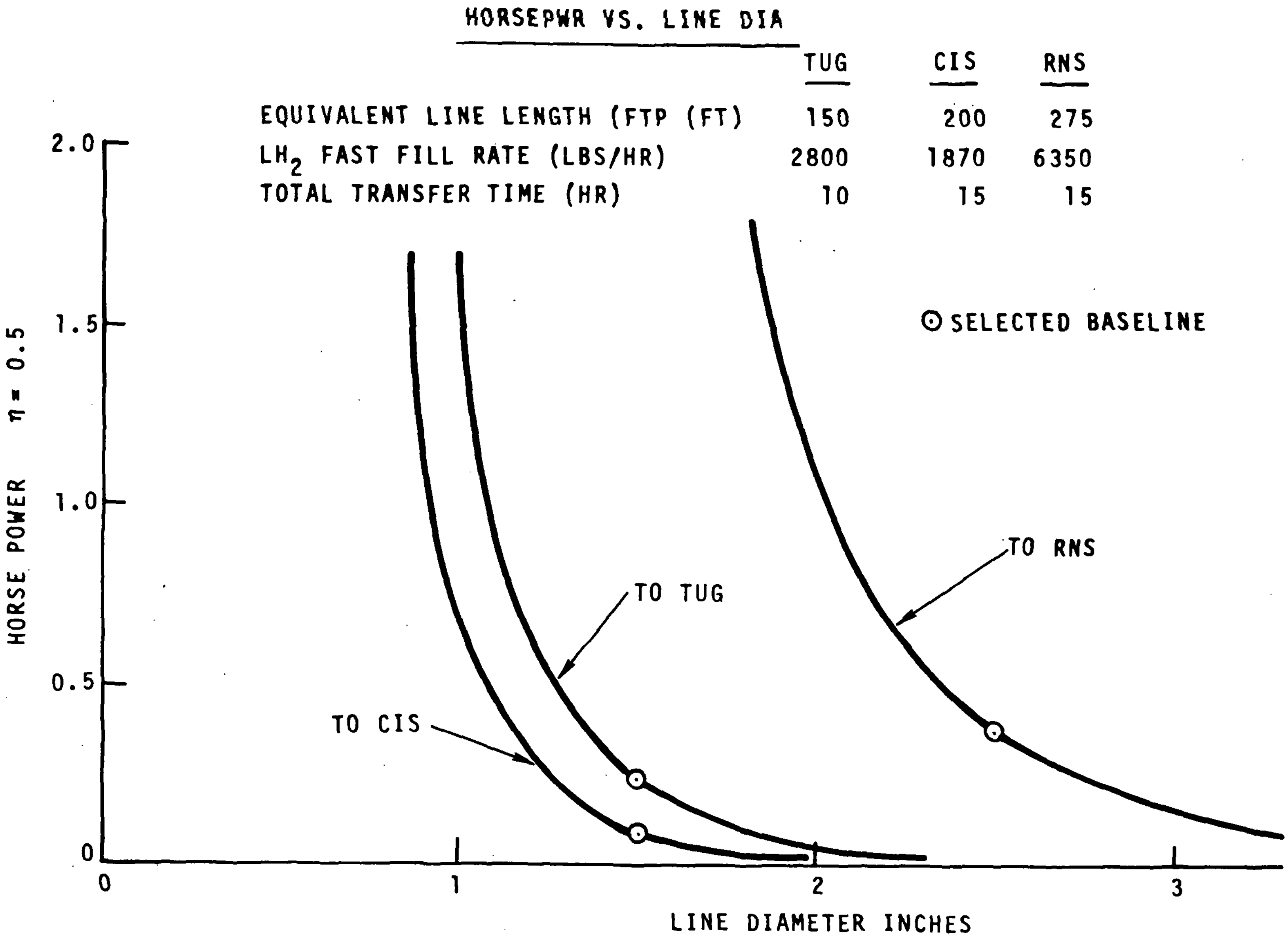


Figure 6.5.2-2 LH_2 Transfer from Logistics Tank Power

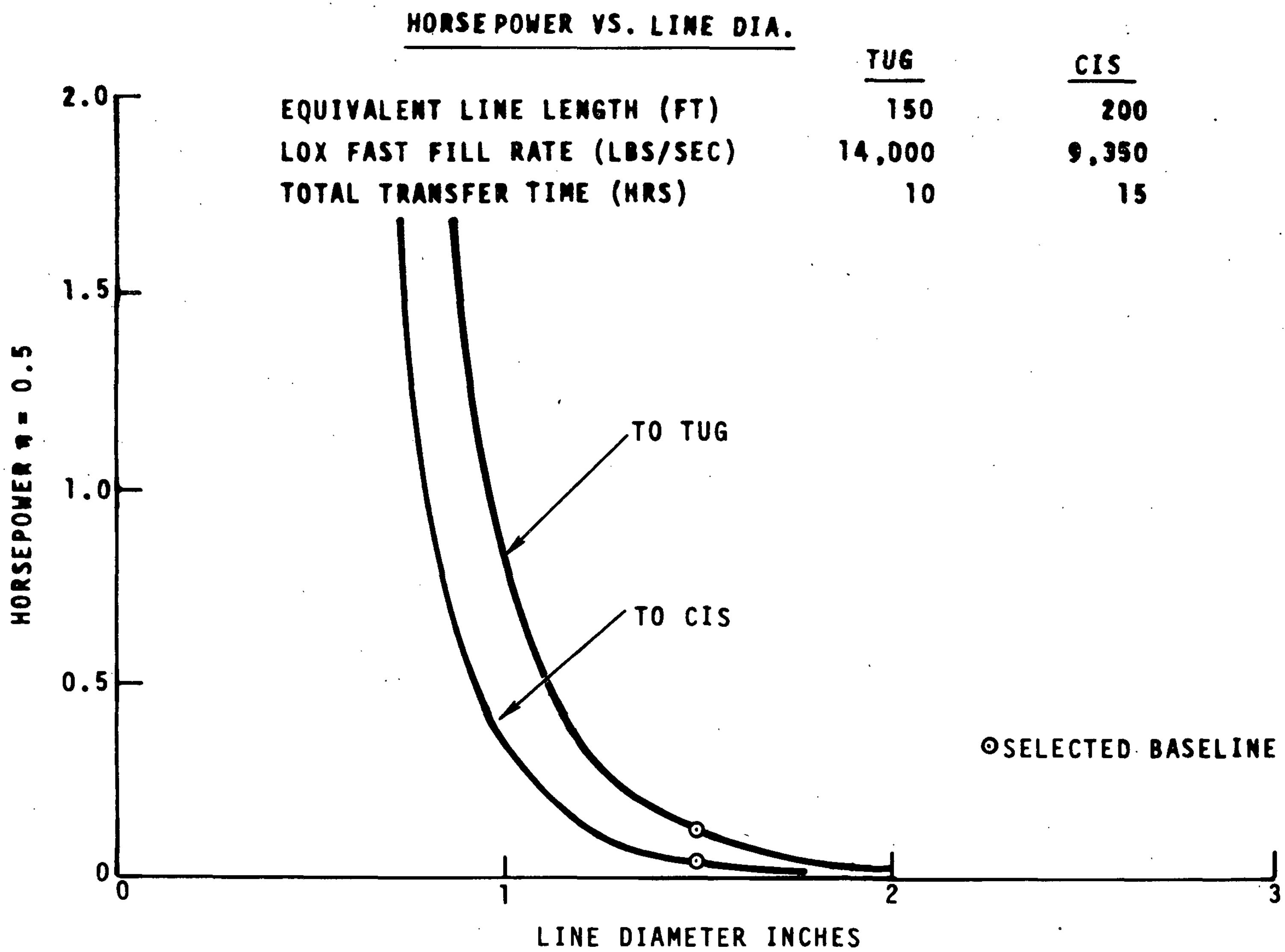


Figure 6.5.2-3 LO_2 Transfer from Logistics Tank Power

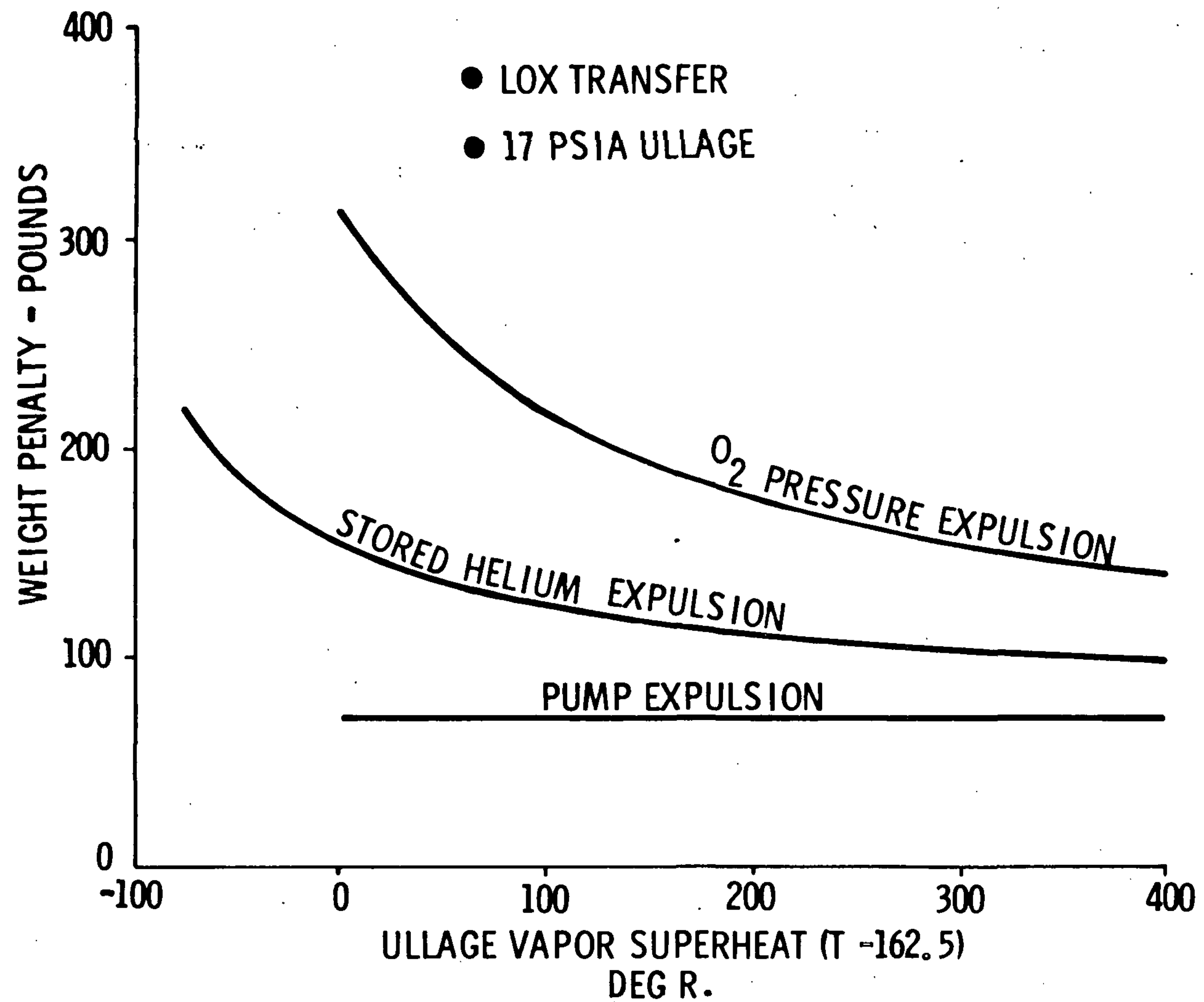
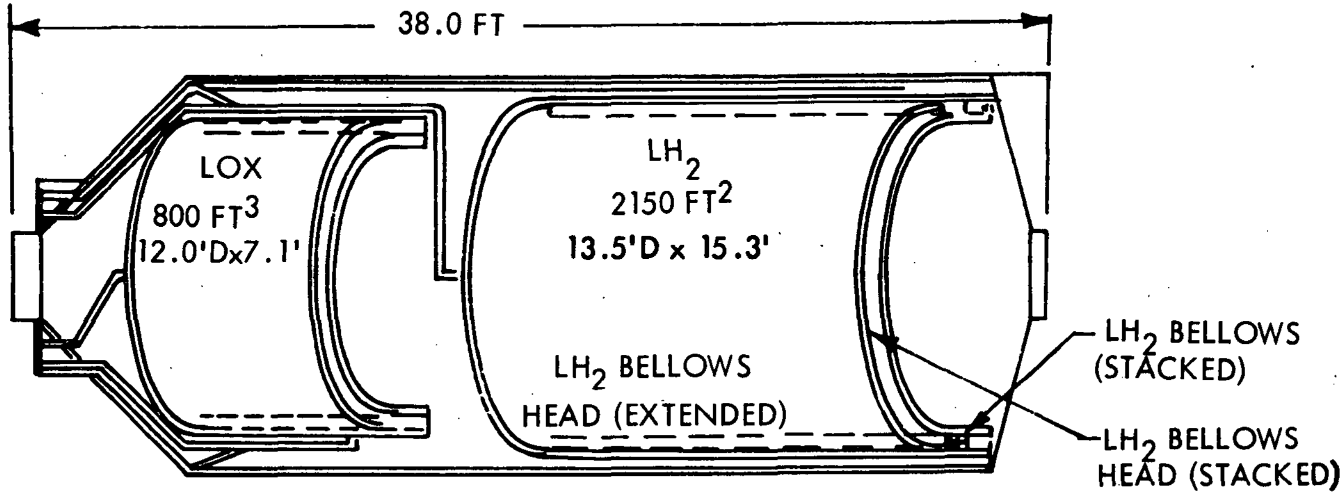


Figure 6.5.2-4. Comparison of Expulsion Concepts (Logistics Tank-to-Tug Transfer)



ADVANTAGES

1. NO PROPELLANT SETTLING REQUIRED
2. SHUTTLE CAN REMAIN ATTACHED DURING TRANSFER OPERATIONS
3. RESIDUALS MAY BE LOWER THAN FOR OTHER CONCEPTS
4. NO TRANSFER PUMPS REQUIRED
5. NO GAS RETURN LINE REQUIRED
6. SMALL VOLUME TO INERT OR PURGE

DISADVANTAGES

1. PROPER VENTING OF BELLOWS DIFFICULT
2. ΔP CRITICAL TO PREVENT BUCKLING
3. SUPPORT OF EXPANDED BELLOWS FOR ACCELERATION LOADING DIFFICULT
4. REQUIRES ACTUATION SYSTEM
5. PURGING OR INERTING OF CONVOLUTIONS DIFFICULT
6. TWO PHASE FLOW WHEN USED WITH VENT PRIOR TO FILL
7. INCOMPATIBLE WITH "TOPPING-OFF"
8. INCOMPATIBLE WITH MOST STATE-OF-THE-ART LIQUID GAUGING SYSTEMS
9. FLOW DIRECTION NOT REVERSIBLE

Figure 6.5.2-5 Positive Displacement Expulsion Bellows Concept

Table 6.5.3-1 Expulsion System Selection Trade Table

REQUIREMENT	ALTERNATIVE APPROACHES			SELECTION
THIS SUBSYSTEM MUST PROVIDE THE ENERGY AND/OR MEANS OF EXPELLING THE PROPELLANT FROM THE SUPPLIER VEHICLE INTO THE RECEIVER VEHICLE	(1) LIQUID PUMP PROS LOW WEIGHT PENALTY CONS • IN-TANK PUMPS • RECEIVER PUMP REQD FOR REVERSAL • POOR ACCESS FOR MAINTENANCE	(2) GAS PUMP PROS • TRANSFER REVERSIBLE • LOW WEIGHT PENALTY GOOD MAINTENANCE • CAPABILITY--EXTERNAL ACCESS CONS • HIGH SPEED • SHORTER LIFE	(3) POSITIVE DISPLACEMENT PROS NO PROPELLANT SETTLING REQUIRED CONS INCOMPATIBLE WITH USER CONFIGURATION	RANKING TRANSFER LOSS (2) (1) (3) (4) (5) COMPATIBILITY (2) (4) (1) (5) (3) DEVELOPMENT RISK (2) (1) (3) (5) (4)
• PROPELLANT TRANSFER LOSSES • DEVELOPMENT RISK • SAFETY	(4) LIQ/GAS CONVERSION PRESSURIZATION PROS • LIGHTER THAN STORED GAS FOR LOX SYSTEM CONS WEIGHT PENALTY FOR GAS GENERATION EQUIPMENT	(5) STORED GAS PRESSURIZATION PROS • TANK PURGING CAPABILITY IF REQUIRED LIGHTER THAN GAS CONVERSION FOR LH_2 SYSTEM CONS • WEIGHT PENALTY FOR GAS SUPPLY • INCOMPATIBLE WITH CONNECTED ULLAGE		SELECTED APPROACH GAS PUMPS (2) SELECTED PRIMARILY ON THE BASIS OF REVERSIBILITY

6.6 NET POSITIVE SUCTION PRESSURE CONTROL

6.6.1 Candidate Concepts

Some type of net positive suction pressure (NPSP) control by pressurization is generally required for expulsion or transfer of cryogenic propellants. Pressurization provides and maintains the liquid in a subcooled state during transfer to avoid cavitation or boiling. Most pumps have a specific NPSP requirement which dictates the minimum level of pressurization required.

Active and passive pressurization systems have been considered during this study. Active systems include:

- Liquid-to-gas conversion using a gas generator and pump
- Liquid-to-gas conversion using a solar heat exchanger and pump
- Stored gas

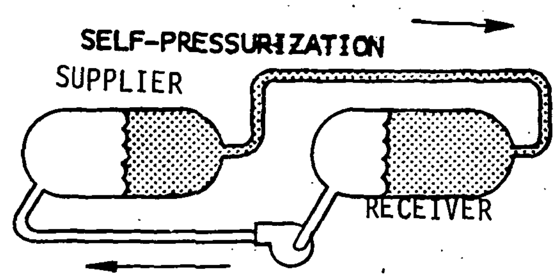
The passive system is self pressurization. Schematics of these concepts are shown in Figure 6.6.1-1.

6.6.2 Discussion of Candidate Concepts

The liquid-to-gas conversion concept, using a gas generator and pump, operates by the gas generator driven supply pump providing LH_2 to the heat exchanger cold side and to the fuel side of the gas generator. LO_2 is provided by a similar pump to the oxidizer side of the gas generator and heat exchanger. The gas generator exhaust is utilized for the hot side of the heat exchanger. The LH_2 provided to the cold side of the heat exchanger is vaporized to pressurize the vehicle ullage. The liquid-to-gas conversion concept using a solar heat exchanger is similar to the gas generator system except the solar heat exchanger replaces the gas generator heat exchanger. Solar energy would provide the heat of vaporization for the liquid-to-gas conversion of the pressurant. The stored gas concept provides the pressurant (normally helium) from high-pressure storage bottles that is expanded under controlled conditions, when pressurization is required.

Self-pressurization utilizes no active pressurization system to control NPSP. The ullage pressure is provided by the vapor pressure of the liquid itself because it is in a state of saturation (boiling) at the vapor/liquid interface.

Figure 6.6.2-1 data generated during Study 8, Cryogenic Acquisition and Transfer, which was conducted by NR as part of the Saturn S-II Advanced Technology Studies (Reference SD 71-768), shows that bubble collapse times resulting from an increase in ullage pressure can be relatively long even though the fluid is instantly subcooled by the pressure level change. These data assume that the bubbles are collapsed by convection heat transfer in the liquid and mass transfer between the liquid and the bubble. An analysis of this type will tend to produce conservative data or maximum collapse times with factors such as propellant agitation and the existence of a firm liquid/vapor interface reducing the collapse time. Although collapse times in an operating system would tend to be shorter than those shown, it is concluded that it is undesirable for the bulk propellant to become saturated during transfer operations, such as may be the case if a self-pressurization concept were to be employed.



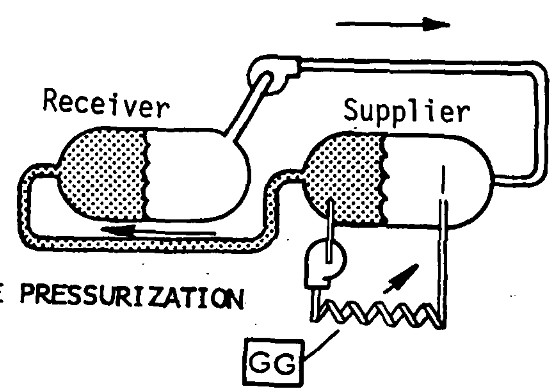


Figure 6.6.1-1 NPSP Control Concepts

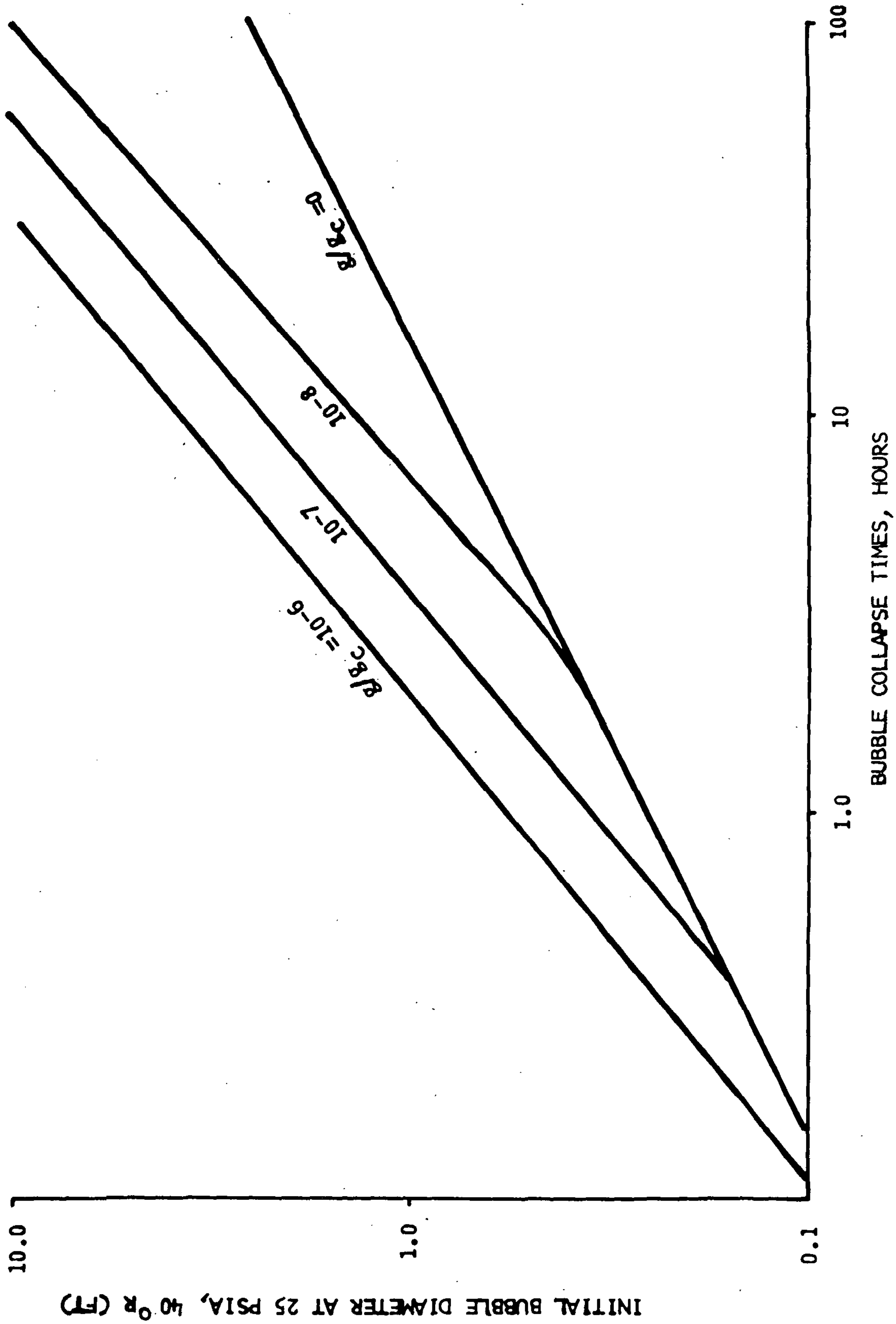


Figure 6.6.2-1 Bubble Collapse Times for Hydrogen Vapor

Figure 6.6.2-2 shows the weight penalty including system hardware for stored helium and liquid-to-gas conversion active NPSP control systems for both LH_2 and LO_2. The weight of the propellant required to provide a 2 psi NPSP for the logistic tank, tug, and CIS is shown on Figure 2.6.2-3.

6.6.3 Comparison of Candidate NPSP Control Concepts

Two alternate approaches for NPSP control were evaluated on the basis of propellant losses, compatibility, development risk and safety. The evaluation is summarized in the trade table, Table 6.6.3-1. As noted, the active pressurization system was selected for the study baseline on the basis of development risk. Self-pressurization, although feasible, is not proven and is not expected to be fully developed within the study time frame.

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6.0-9 Design and System Analysis of a Chemical Interorbital Shuttle, Phase II Results, Contract NAS8-27670, North American Rockwell, SD 71-586, December 15, 1971.

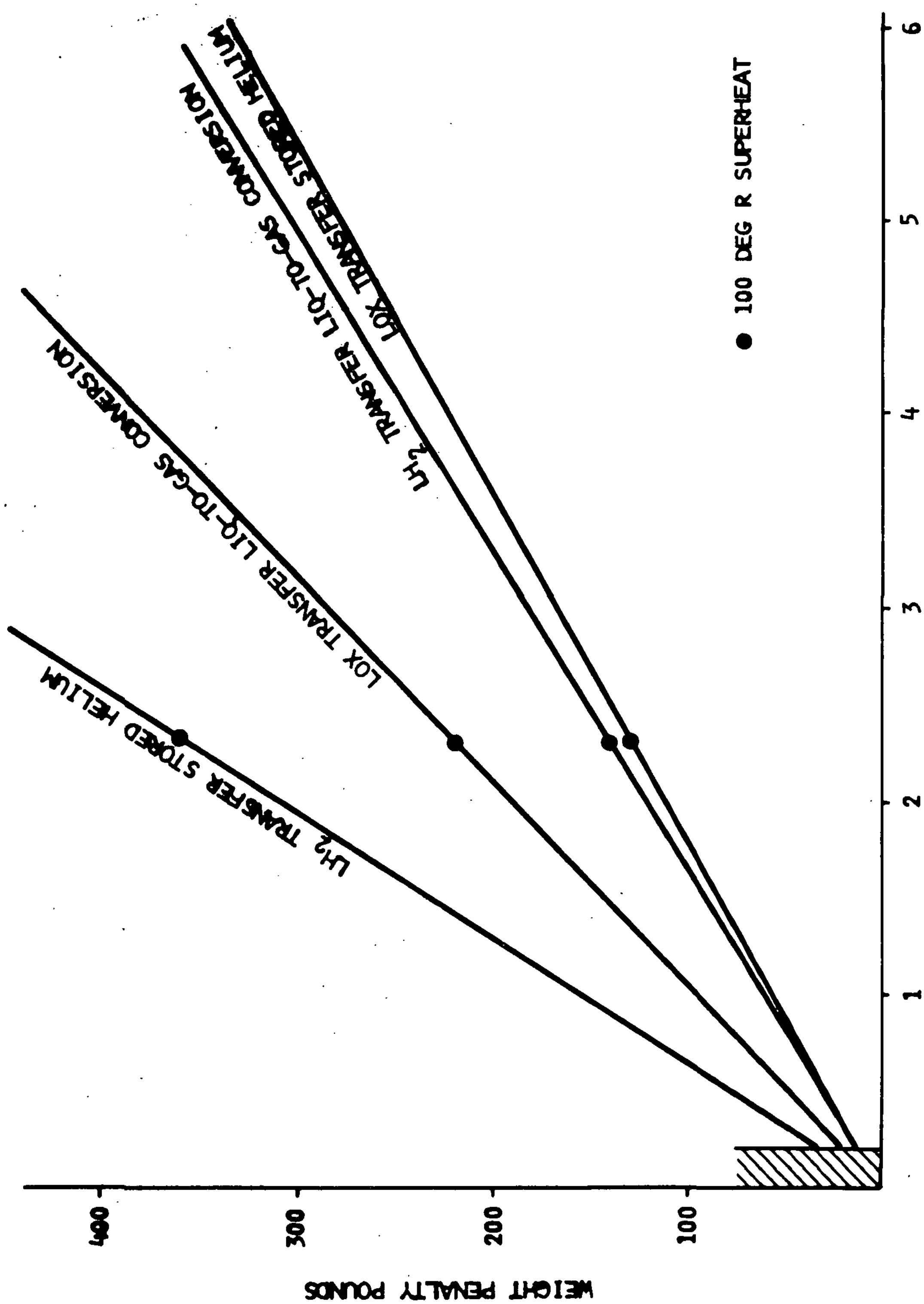

Figure 6.6.2-2 Logistic Tank to Tug Transfer NPSP Control

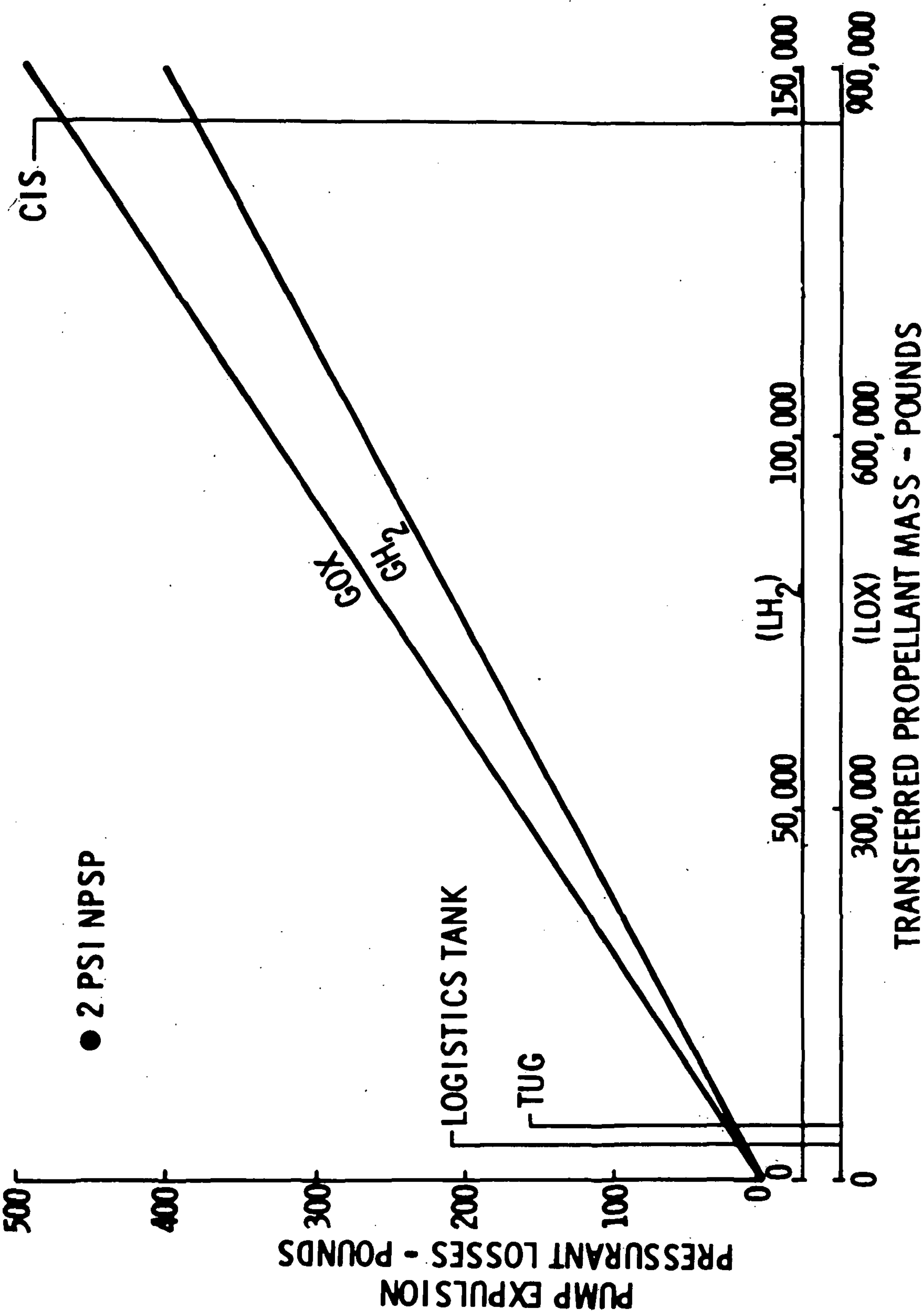

Figure 6.6.2-3 Pressurant Losses for NPSP Control

Table 6.6.3-1 Net Positive Suction Pressure Control Trade Table

REQUIREMENT	ALTERNATE APPROACHES		SELECTION
THIS SUBSYSTEM PROVIDES VAPOR PRESSURE CONTROL TO ASSURE ACCEPTABLE QUALITY PROPELLANTS.	SELF PRESSURIZATION 1	ACTIVE PRESSURIZATION 2	PROPELLANT TRANSFER LOSS 1 2 COMPATIBILITY 2 1 DEVELOPMENT RISK 2 1 SAFETY 2 1 SELECTED APPROACH ACTIVE PRESSURIZATION 2 SELECTED ON BASIS OF DEVELOPMENT RISK
EVALUATION CRITERIA PROPELLANT TRANSFER LOSS COMPATIBILITY OF DELIVERY AND RECEIVER VEHICLE DEVELOPMENT RISK SAFETY	PROS PASSIVE SYSTEM NO PROPELLANT LOSS CONS BOILING (TWO PHASE FLOW) GAS ENTRAPMENT IN USER CAPILLARY DEVICES POOR PERFORMANCE PREDICTABILITY	PROS ALL LIQUID TRANSFER FLOW GOOD PERFORMANCE PREDICTABILITY CONS SIGNIFICANT PROPELLANT LOSSES ACTIVE SYSTEM REQUIREMENT	

7.0 PROPULSION SYSTEMS FOR PROPELLANT TRANSFER

7.1 INTRODUCTION

Analyses conducted (in Section 6) evaluated linear and rotational acceleration as the method of providing liquid/vapor interface control for low g, in-space propellant transfer operations. The analyses resulted in the selection of the continuous linear acceleration mode as baseline. Other analyses optimized propellant transfer durations and concluded that approximately 8 to 10 hours were required for the tug and 10 to 20 hours were required for the Chemical Interorbital Shuttle (CIS) or Reusable Nuclear Shuttle (RNS). In view of these conclusions, a survey of potential propulsion systems for use during the propellant transfer operation was conducted. The objective of this trade study was to select a baseline propulsion system concept that would best satisfy the requirements of the propellant logistic system considering the factors of existing and projected thruster technology, development risk, compatibility with current baseline spacecraft concepts, operational efficiency, hardware commonality, maintenance, and cost.

7.2 SUMMARY OF RESULTS

Based on the operational requirements for in-space propellant transfer, a trade study was conducted to examine the available propulsion concepts and to select a baseline propulsion system design. It was determined that a LO_2/LH_2 system installed in the logistics module represented the best compromise. Storable monopropellant, storable bi-propellant, electrolysis and cold gas systems were also studied. A summary of candidates is presented in Table 7.2-1.

It was established that the endurance requirements for the logistics propulsion system over several missions is so high that mounting of the system in the logistics module is virtually mandatory to permit periodic ground maintenance of the thrusters. This conclusion was based on a survey of existing and projected thrust chamber technology.

The selection of a LO_2/LH_2 type system insures compatibility with the logistic module systems and opens the possibility of commonality of hardware with the user vehicle propulsion systems. The baseline propulsion system for the logistic module consists of liquid oxygen and hydrogen start tanks integral with, and inside, the main logistic tanks; conditioning unit pumps driven by gas generators; high pressure gas storage accumulators; and valves and controls. This propulsion system concept is attractive primarily because of high performance, use of propellants available in the module, and commonality of hardware with the user vehicles.

It was determined that new engines must be developed for the propellant logistics application. More than one size thruster may be necessary.

Table 7.2-1 Propulsion Systems Trade Table

FUNCTIONAL & TECHNICAL REQUIREMENTS	ALTERNATIVE APPROACHES	

PROVIDE CONTINUOUS ACCELERATION TO THE TUG, CIS, RNS DURING PROPELLANT TRANSFER OPERATIONS TRANSFER TIMES VARY BETWEEN 10 & 20 HOURS WITH A LIFE REQUIREMENT FOR UP TO 20 MISSIONS & 19 PROPELLANT TRANSFERS PER MISSION
CRITERIA
● PERFORMANCE
● COMPLEXITY
● DEVELOPMENT RISK
● INTEGRATION CAPABILITY
THRUST LEVEL (TOTAL LBS)
TUG 7.5
CIS 11.0
RNS 3.8
	1. $LO_2 - LH_2$ (LIQUID OXYGEN LIQUID HYDROGEN)
	PRO
	● HIGH PERFORMANCE I_s - 400
	● LOW WEIGHT & VOLUME TAP OFF MAIN TANKS
	● SIMPLIFIES LOGISTICS
	● EARLY INTEGRATED
	● COMMONALITY WITH USER VEHICLES
	CON
	● COMPLEX - PUMPS, GAS GENERATORS
	● SMALL THRUSTERS IN DEVEL. STAGE
	● NOT DEVELOPED - HIGH DEVELOPMENT COSTS
		2. N_2O_4-MMH (NITROGEN TETROXIDE & MONOMETHYL HYDRAZINE)
		PRO
		● GOOD PERFORMANCE - I_s - 290
		● INDEPENDENT SYSTEM
		● GOOD RELIABILITY - PROVEN
		● SYSTEMS FLYING
		  ● SMALL THRUSTERS QUALIFIED
		  ● 2-6 HOUR CAPABILITY EXISTS
		● SIMPLIFIED MAINTENANCE
		● LOW DEVELOPMENT COSTS
		CON
		● WEIGHT & VOLUME IMPACT
		● NO TANK OR PROPELLANT SHARING
		● NOT COMPATIBLE WITH LOGISTICS MODULE SYSTEMS

3. N_2H_4 (HYDRAZINE MONOPROPELLANT)
PRO
● INDEPENDENT SYSTEM
● GOOD RELIABILITY - PROVEN
● SYSTEMS FLYING
  ● ONE PROPELLANT
  ● THRUSTERS QUALIFIED
  ● UP TO 16 HRS DEMONSTRATED
  ● Δ QUAL REQUIRED
● LOW DEVELOPMENT COSTS
● COMMONALITY WITH ORBITER
CON
● LOW PERFORMANCE
● REQUIRES MORE TANKAGE THAN N_2O_4 - MMH
● POTENTIAL CATALYST PROBLEMS
● MAY IMPACT DEVELOPMENT
	4. GH_2 (GASEOUS HYDROGEN)
	PRO
	● SIMPLE
	● LOW PRESSURE SYSTEM
	● SHARE MAIN PROPELLANT TANK & PROPELLANT
	● HIGHLY RELIABLE
	● FEW PARTS
	CON
	● GOOD PERFORMANCE ATTAINABLE ONLY BY ADDING LARGE HEAT EXCHANGERS
	● POTENTIAL THERMAL & STRUCTURAL INTERFACES REQUIRES FURTHER STUDY
		SELECTION
		PERFORMANCE
		1 2 3 4
		COMPLEXITY
		2 3 4 1
		DEVELOPMENT RISK
		3 2 4 1
		INTEGRATION
		1 4 3 2
		SELECTED SYSTEM
		LO_2-LH_2 1
		BASICALLY BECAUSE IT IS COMPATIBLE WITH OTHER LOGISTICS MODULE SYSTEMS

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If a propulsion system concept independent of the main logistic tanks proves desirable in later evaluations, a logical alternate to the above first choice recommendation would be the earth storable, bipropellant system. Such a system and its components are in a later stage of development from past programs which reduces development costs, is lighter in weight and smaller in volume for the self-contained module design, and would benefit from commonality with shuttle orbiter hardware.

7.3 REQUIREMENTS

The selection of linear acceleration as the propellant transfer liquid/vapor interface control concept imposes a continuous thrust requirement of 20 hours on the propulsion system for CIS logistic operations under maximum gross weight conditions (final transfer to fill). In addition, it has been determined that 19 shuttle orbiter flights and propellant transfer cycles are required to fill the CIS for its baseline mission. The CIS design life is ten missions. Therefore, the propellant liquid/vapor interface control propulsion system ideally should achieve a firing life of 285 hours without maintenance, using an average 15 hour fill time, to support one CIS mission and a total life capability of 2850 hours to be compatible with the CIS longevity. Similar, but less stringent, requirements exist for the tug and RNS. Since the CIS requires more deliveries and transfers, it will exert the major influence in the selection of the propellant logistics propulsion system concepts. The total settling thrust required as provided by redundant thruster pairs for the tug, CIS, and RNS is 7.5, 11.0 and 3.8 pounds, respectively.

Further transfer propulsion system requirements are concerned with achieving compatibility with the existing baseline designs and flight operation plans of the shuttle orbiter and in-space user vehicles, and obtaining a cost effective configuration.

7.4 CANDIDATE CONCEPTS

Selection of the baseline design for the propellant/vapor interface control propulsion system for logistic operations requires evaluation of several possible design options. The major alternatives considered are summarized as follows: user vehicle or logistic tank mounted thrusters, existing auxiliary propulsion systems (APS) or new propulsion systems and thrusters, single or multiple thrusters, and propellant selection among cold gas, storable monopropellant, storable bipropellant, and LO_2/LH_2. The significant considerations involving these options are presented in the following paragraphs:

7.4.1 Summary of Candidate Concepts

The location of the propulsion system utilized for propellant/vapor interface control must be determined before further trade studies are meaningful. The propulsion system and thrusters can be mounted in the logistic tank assembly or in the space based user vehicles. If the latter is selected, use of the existing or new systems and thrusters provides further options.

Since the logistic tank module returns to earth with the orbiter after each in-orbit transfer operation, the opportunity exists for frequent maintenance of any systems installed in the module assembly if required. If the propulsion system used for propellant transfer operations is placed on the user vehicle, the maintenance free life requirement is extremely high. In the case of the CIS, approximately 2850 hours firing life would be required for a separate propulsion system and longer system capability if combined with the APS. The tug and RNS total life requirements for the propulsion systems is about 500 and 1000 hours, respectively. Refer to Table 7.4-1.

Table 7.4-1 Engine Operational Requirements for Propellant Settling

	TUG	CIS	RNS
Continuous Thrust (hrs) average	10	15	10
Design Life (missions)	50	10	10
Total Life Requirements (hrs)	500	2850	1000
Minimum Acceleration (g)	10^{-4}	10^{-5}	10^{-5}
Total Thrust Level (lbs)	7.5	11.0	3.8

At the present time, most operational thrusters have a demonstrated firing life of less than one hour and in most cases only a few minutes. Some exceptions exist such as a 16 hour demonstration test conducted on a small hydrazine monopropellant thruster using test facility tankage and plumbing. Other low thrust (millipound level) systems have the potential of unlimited life, but in the thrust range under consideration here, highly qualified systems are not yet available. Apollo N_2O_4/MMH and N_2O_4/A-50 bipropellant reaction control systems have been run continuously for up to six hours; and on a cumulative life basis to 12 hours. The projected life capability of advanced LO_2/LH_2 systems has been predicted to be about 1000 hours based on tests conducted under a NASA-LeRC Contract (NAS3-14352). This is insufficient to fill the 2850 hour (unattended) requirements noted above.

It is concluded that insufficient extended life data exist on propulsion components to risk installation of the propulsion systems for propellant/vapor interface control on the user vehicles. It also follows that this propulsion system function should not be combined with the existing user vehicles APS for the same reason.

In this connection, consideration was given to the possibility of utilizing the components or complete systems to be developed for the tug, CIS and RNS attitude propulsion systems in the logistics tank assembly for the propellant transfer impulse requirement. It is believed that tanks, valves, pumps, regulators, gas generators, and couplings are prime candidates for common usage.

The thrust level requirement for the user vehicles stationkeeping is in the 20 to 250-pound range with pulse mode and short steady-state firings. This function is handled by the APS and requires up to ten hours total life, the exact value depending on the receiver vehicle and mission considered.

The requirement for propellant transfer propulsion system is for lower thrust to provide continuous burns of up to 20 hours with a total life objective of about 3,000 hours. Since the projected life, operational mode, and thrust level of current baseline design APS thrusters are incompatible with the propellant logistic requirements, new thrusters will be necessary.

The optimum thrust level necessary for each vehicle being served will be different. Therefore, it may be necessary to develop more than one thruster size. The possibility exists that the smaller thrusters developed for the tug can be utilized in the heavier vehicles by a multiplicity of thruster units. If such is the case, development and component logistics costs would be reduced which must be traded against the added installation expense.

Selection of the propellant logistic propulsion system generic types was given consideration in the study. The systems analyzed were cryogenic bipropellant, monopropellant, earth storable bipropellant, electrolysis, and cold gas. Descriptions with advantages and disadvantages are presented in the following paragraphs.

7.4.1.1 Cryogenic Bipropellant System

The propellants utilized in this system are stored cryogenics (i.e., LO_2 and LH_2), which are stored at low pressure. Figure 7.4-1 presents a schematic of a typical system. The engines operate from two accumulators, one of which stores GH_2, the other GO_2. When the system is activated, GH_2 and GO_2 flow to the engines and the gas generators where they are ignited. The gas generators drive the turbopump and provide heat to vaporize the pumped cryogenics which are stored in the accumulators. The system "bootstraps" and is self-propagating. The system shuts down when the gas supply is shut off.

<u>System Characteristics</u>

<u>Components</u>

- Cryogenic storage tanks
- Regulators, Check Valves
- Shutoff valves
- Gas generators
- Turbopumps
- Accumulators (storage tanks for gas)
- Engines, Valves
- Spark ignition

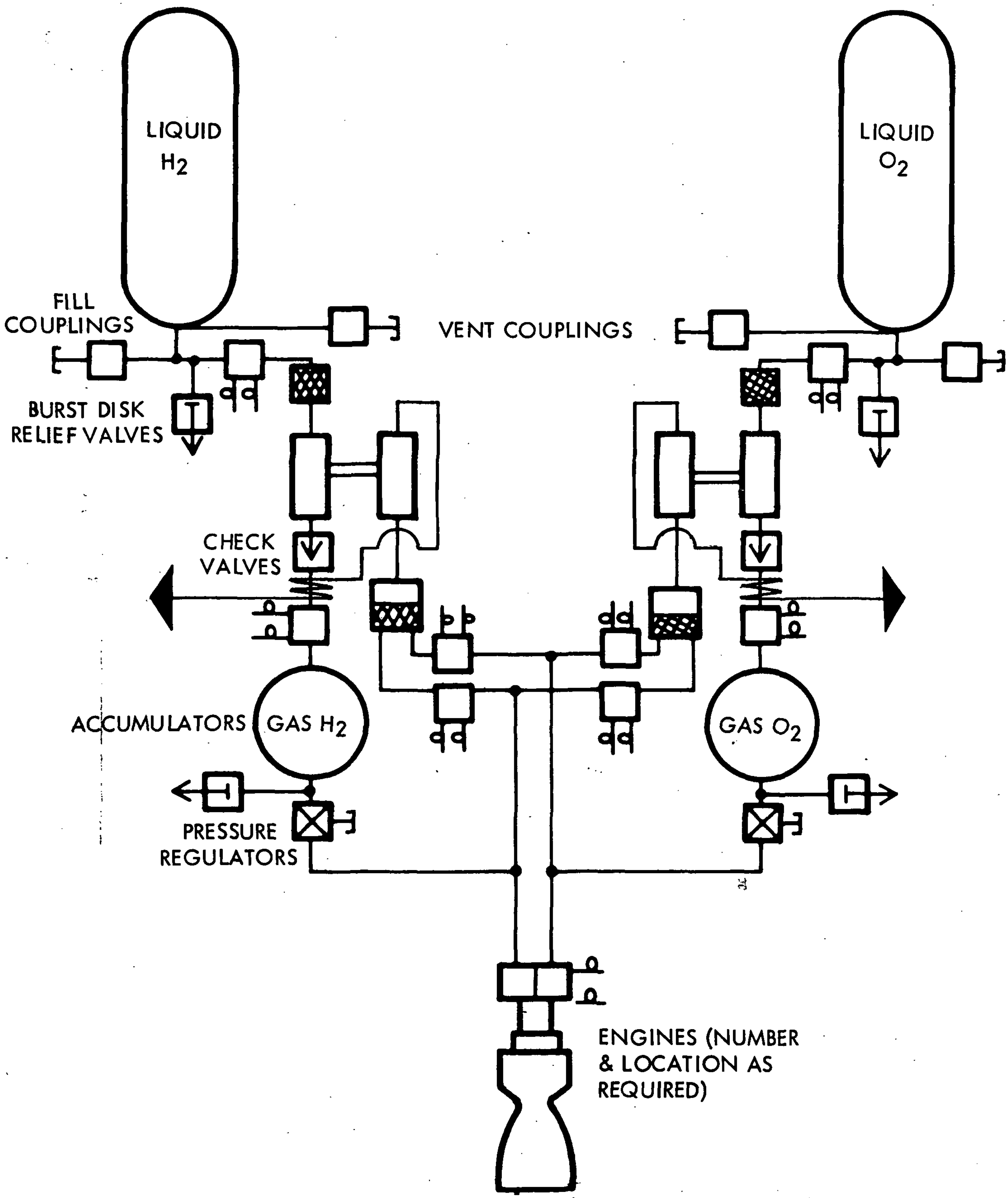


Figure 7.4-1 Cryogenic Oxygen-Hydrogen System Schematic

Advantages	Disadvantages
1. High I_{sp} = 400 sec.	1. Thermal leaks
2. Clean plume	2. Complex
3. Low pressure propellant tanks	3. Major development issue: pumps and low thrust systems
4. Minimum logistics	4. Small thrusters not yet developed

7.4.1.2 Hydrazine Monopropellant System

A single propellant is used to provide the impulse of the entire system. Figure 7.4-2 presents a schematic of a typical system. The propellant is decomposed thermally or catalytically. Hydrazine which has a good performance is used for this description. The hydrazine (N_2H_4) decomposes to NH_3 and N_2. The highest performance is obtained when disassociation of the NH_3 is minimized. In a hydrazine system, the catalyst is usually contained in the thrusters where the N_2H_4 flows through the catalyst bed. This is an exothermal reaction providing low molecular weight gases. Some of the NH_3 is further decomposed to N_2 and H2; this is an endothermic reaction which degrades performance.

N_2H_4 (Hydrazine) System Characteristics

1. Fixed thrust - regulated system
2. Catalyst bed - (Shell 405)
3. Pressurant tank/GN_2
4. Isolation Valve
5. Regulator
6. Positive expulsion propellant tank(s) w/bladder or diaphragm
7. Engines w/filters and valves
8. Heaters - maintain 35 F
9. I_{sp} = 230-240 max.

Capabilities

F = 0.1 lbf tested
1600 lbf tested

Steady state burn 16 hours (small 0.1 lb thrusters)

Advantages	Disadvantages
1. Single propellant	1. Low performance
2. "Clean" plume	2. Catalyst limitation and cost
3. Pressure fed	3. Freezes at +35 F
4. Simple	

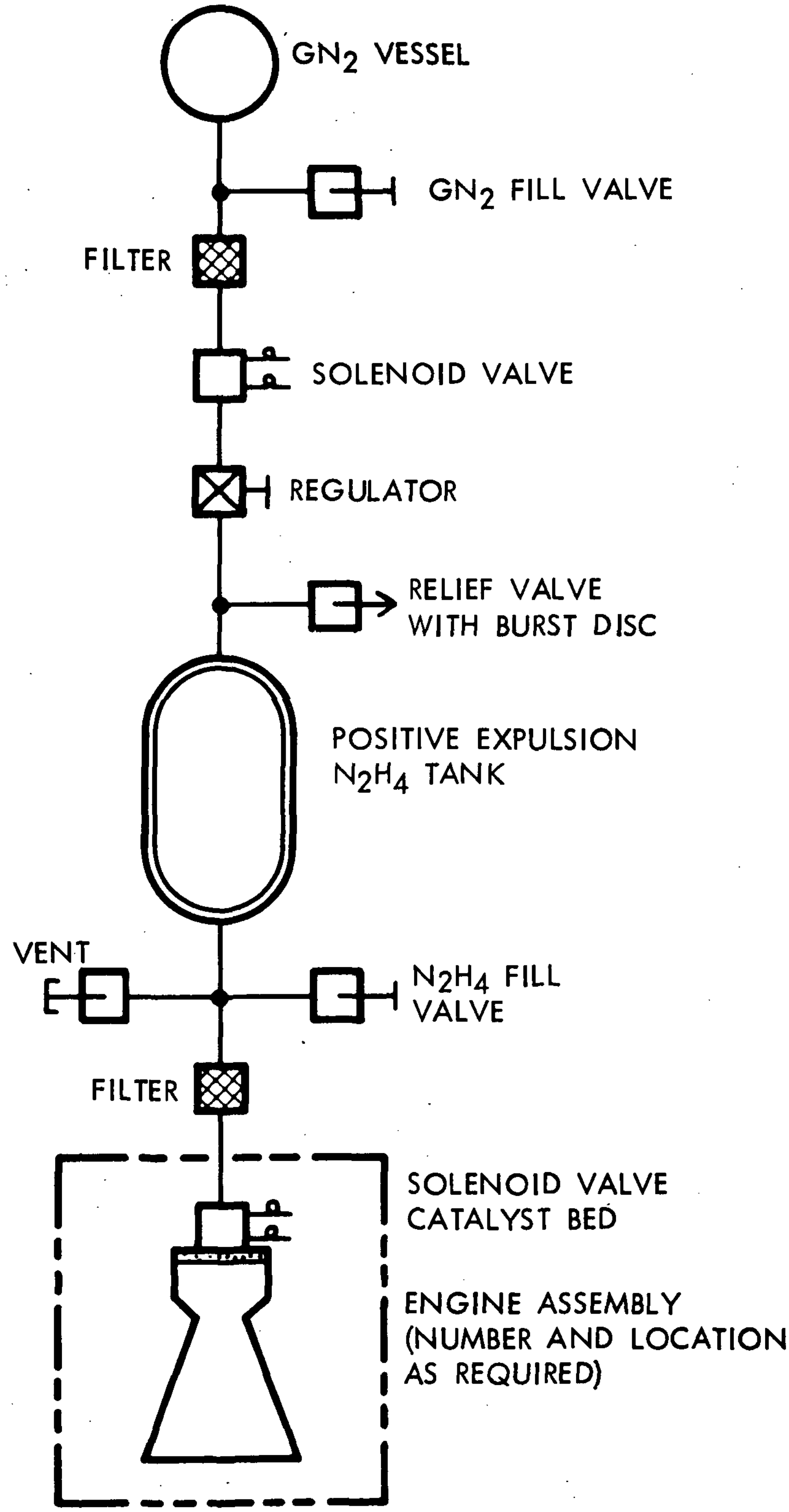


Figure 7.4-2 Typical Monopropellant System Schematic

7.4.1.3 Storable Bipropellant System

Two liquids, an oxidizer and a fuel are pressure fed (or pump fed) through regulators, and orifices into an engine to provide thrusting impulse. Figure 7.4-3 represents a typical system. The propellants are ignited in the engine by an external source or hypergolically, such as is the case with N_2O_4 and MMH. The propellants are kept isolated until mixed in the thrust chamber. The earth-storable bipropellant system characteristics follow:

<u>N_2O_4 - MMH System Characteristics</u>

1. Fixed thrust
2. Regulated system
3. High pressure gas storage tank(s)
4. Separate propellant manifolds
   a. Oxidizer
   b. Fuel
5. Isolation valves
6. Regulators
7. Positive expulsion propellant tanks with bladder; diaphragm, bellows
8. Engine - Bipropellant valves
9. Heater for N_2O_4

<u>Capabilities</u>

1. Wide thrust ranges tested and qualified
2. Components with 3-5 years experience available
3. Several systems exist with several hours demonstrated life

<u>Advantages</u>

1. I_{sp} = 290 sec.
2. Pressure fed
3. Minimal development risk

<u>Disadvantages</u>

1. Thermal control required
2. Relatively large number of components
3. Complex
4. Potential plume problems

7.4.1.4 Electrolysis System

Gaseous propellants are produced by the electrolysis of some suitable liquid such as water. Refer to Figure 7.4-4. The system contains an electrolytic cell. On the application of voltage to the electrodes in this cell, gas is generated and regulated to provide constant pressures at the thrusters. The gases are ignited to provide propulsive forces.

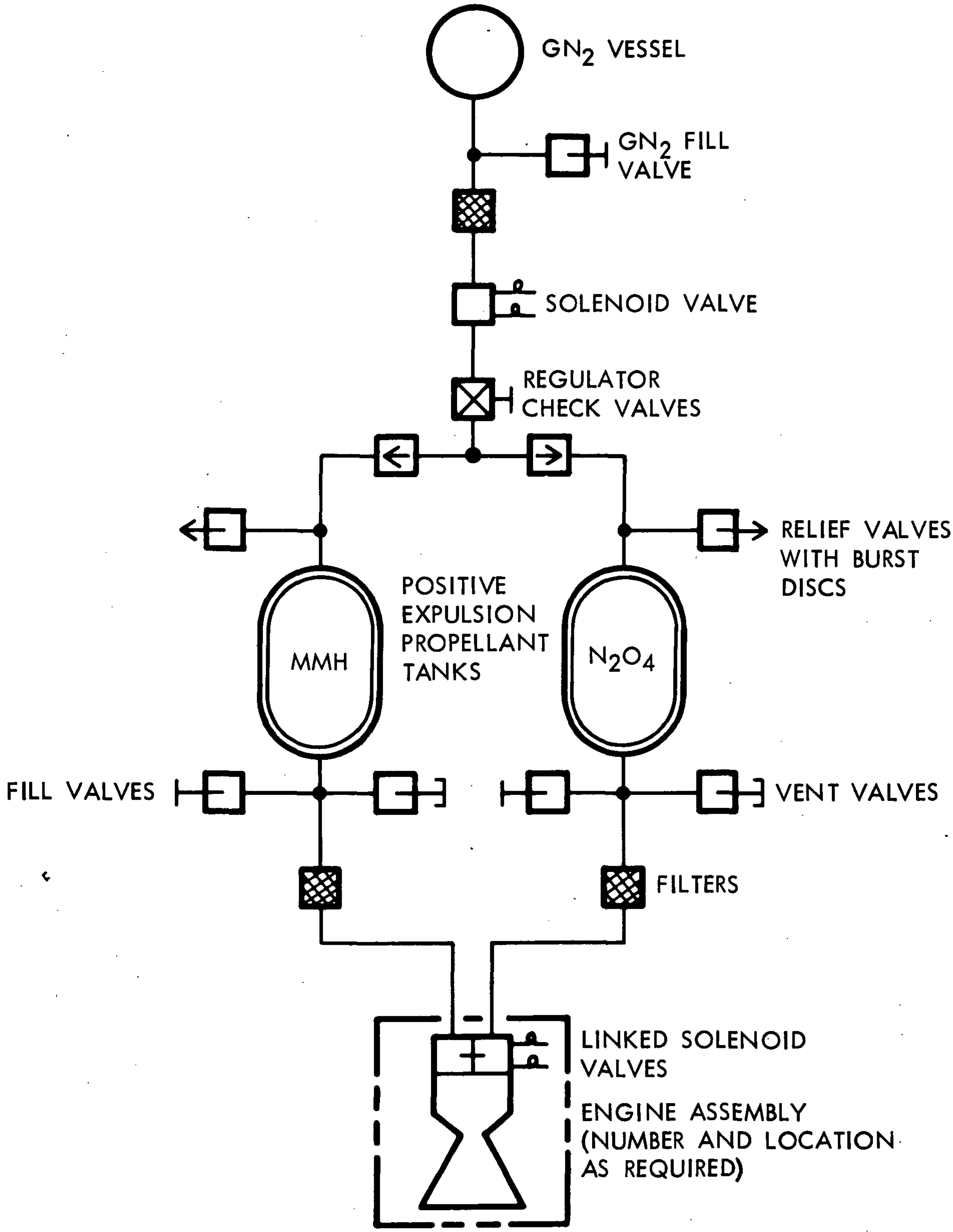


Figure 7.4-3 Typical Earth Storable Bipropellant System Schematic

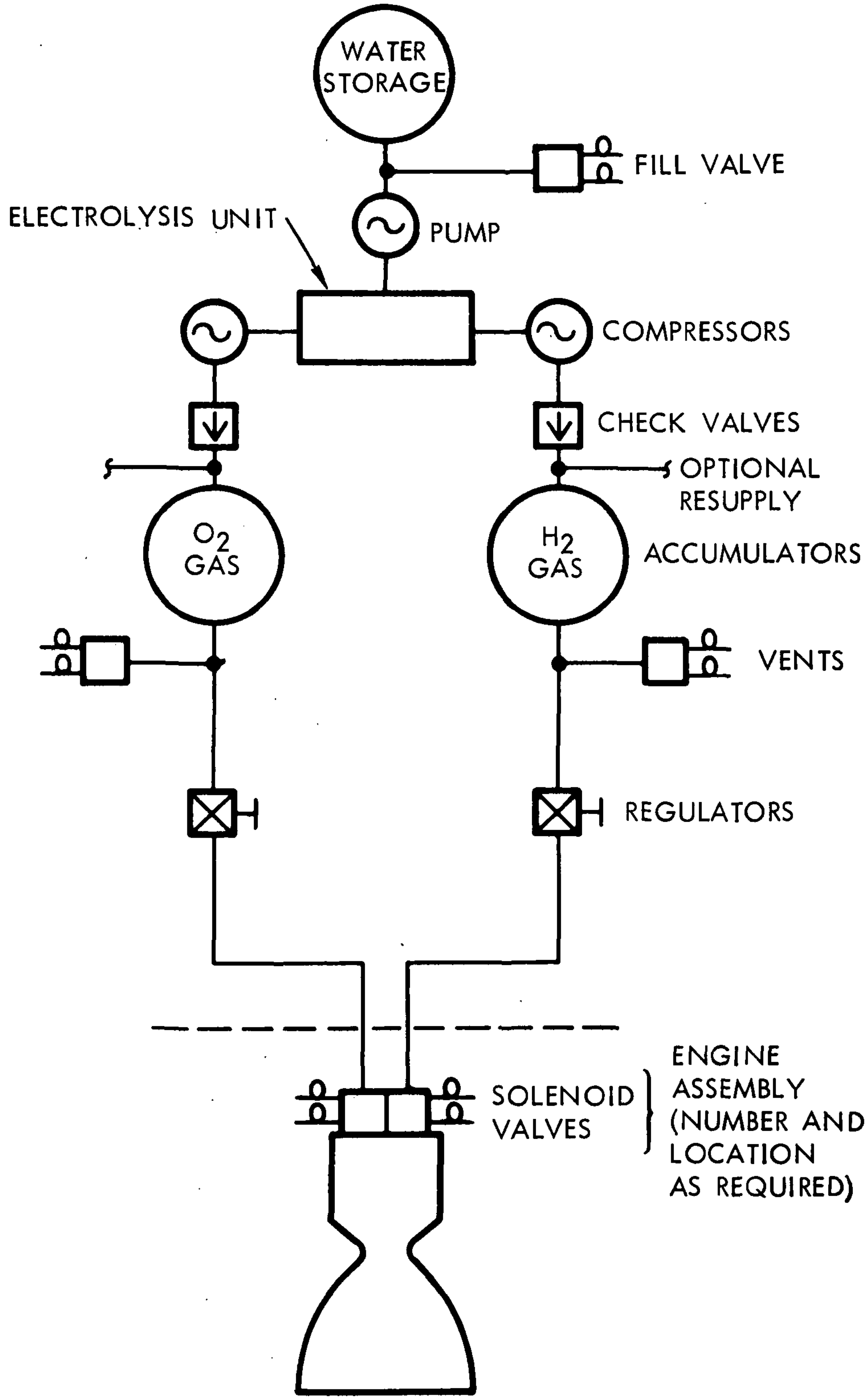


Figure 7.4-4 Typical Electrolysis System Schematic

Characteristics

Components

Storage tank
Electrodes
Power source
Regulator
Spark ignition
Thrusters

Advantages

1. If power and water available, can be useful
2. I_{sp} = 350 sec.
3. Simple system

Disadvantages

1. High power requirements 24 watts/millipound thrust
2. Explosion hazard
3. Because of (1), required thrust levels impractical unless surplus of power and water exist
4. Electrolysis unit not developed for zero and low g

7.4.1.5 Cold Gas System

A stored gas is used as the propellant for the mass expulsion system. The gas is usually stored at very high pressure for maximum quantity in as small a volume as possible due to the low performance. A typical system is shown in Figure 7.4-5. The lower molecular weight gas has the higher performance, but because of the low density, the tanks become very large and the system gets very heavy. The effective I_{sp} is the deliverable performance based on overall system efficiency. Cold gas systems are usually only considered for low total impulse requirements.

System Characteristics

Components

Storage tank (high pressure)
Regulators
Valves
Heat source
Engines
Pressure transducers
Relief valves
Fill valves

Advantages

1. Simple
2. High reliability
3. No performance degradation in pulse mode
4. No ignition or mixing problems

Disadvantages

1. Usually very heavy
2. Low I_{sp}. Usually good for low I_T
3. Requires heating to get the higher performance

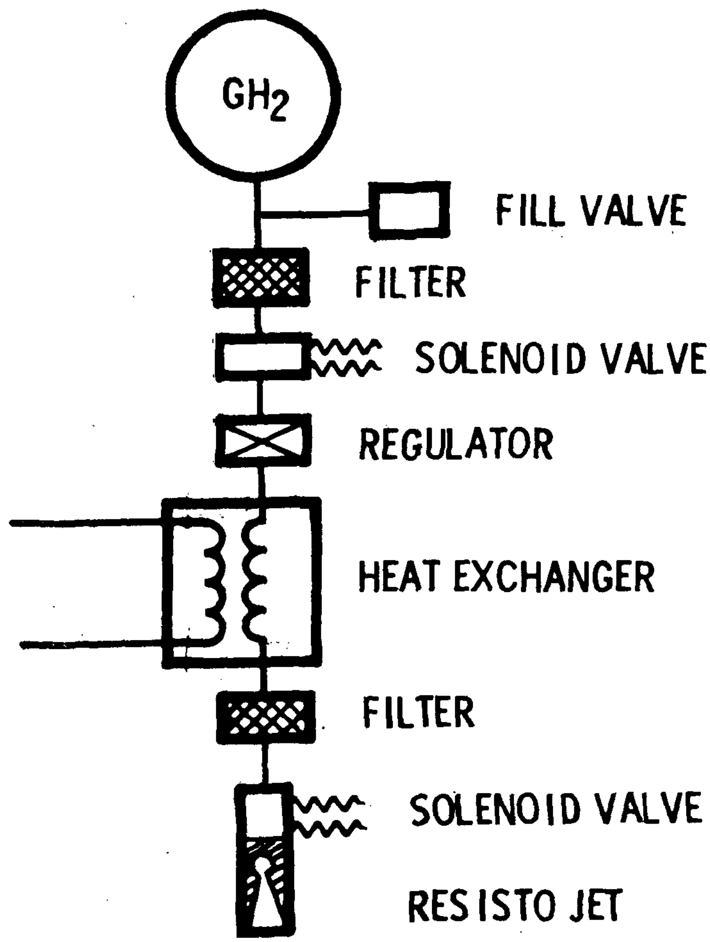


Figure 7.4-5 Typical Cold Gas System Schematic

7.4.2 Discussion of Candidate Concepts

The foregoing shows the general categories of propulsion systems considered. These systems were used in a trade study to determine the best system for the transfer operations, and are discussed in the following paragraphs. Electrolysis was eliminated immediately due to the excessive power requirement and was not studied further.

7.4.2.1 Earth Storable Bipropellant

The candidate propellant is N_2O_4 - MMH since it is one of the higher performance systems with several qualified, space-rated, man-rated systems currently operational; Apollo, Mariner, Minute Man Post Boost Propulsion system, etc. The thrust levels vary from one pound to several hundred thousand pounds.

7.4.2.2 Cryogenics Bipropellant

The LOX-LH_2 concept was selected as the baseline because of its high performance using the primary propellants of the logistics module. Also, since LOX-LH_2 systems are presently baselined for attitude control propulsion system applications on the space-based tug, the CIS, and the RNS, many of the system components can be common. Although no small GOX-GH_2 thrusters are in present use, the main propulsion systems of the upper stages of the Saturn V vehicle have adequately demonstrated the concept. Recent development effort has been initiated on low (20-100 pounds) thrust level units.

7.4.2.3 Monopropellant

The hydrazine system was considered for this trade due to the high performance and the fact that several systems are flight qualified and are operational on programs such is Intelestat, Transtage, Viking, Comsat and Mariner. Thrust levels vary from 0.1 pounds to about 1500 pounds on development articles.

7.4.2.4 Cold Gas

Due to the availability, GH_2 is considered as this propellant. Since the boiloff hydrogen is a waste product and is stored in the propellant tank, it can be considered a candidate. Normally a cold gas system would not be considered for this application due to the volume and weight required and total impulse. But since this GH_2 is on board the tug, CIS and RNS, it is given special treatment for this report. Cold gas systems are very reliable and have been used on several successful space vehicles. Several GN_2 systems are operating on Vela, Mariner, etc. Thrust levels are normally low due to the restricted applications. Thrust levels of 0.01 pounds to 10 pounds are normal.

Figure 7.4-6 is a plot of propellant requirements vs. total impulse for the various concepts. The GH_2 and the N_2O_4-MMH systems have similar performance. However, it should be noted that the GH_2 performance is stated for + 40 F. The problem of providing +40 F hydrogen is one of heating.
This cold gas system appears attractive because a certain amount of hydrogen is lost due to boiloff during normal operations. A brief analysis was conducted to determine the applicability of this approach of utilizing wasted

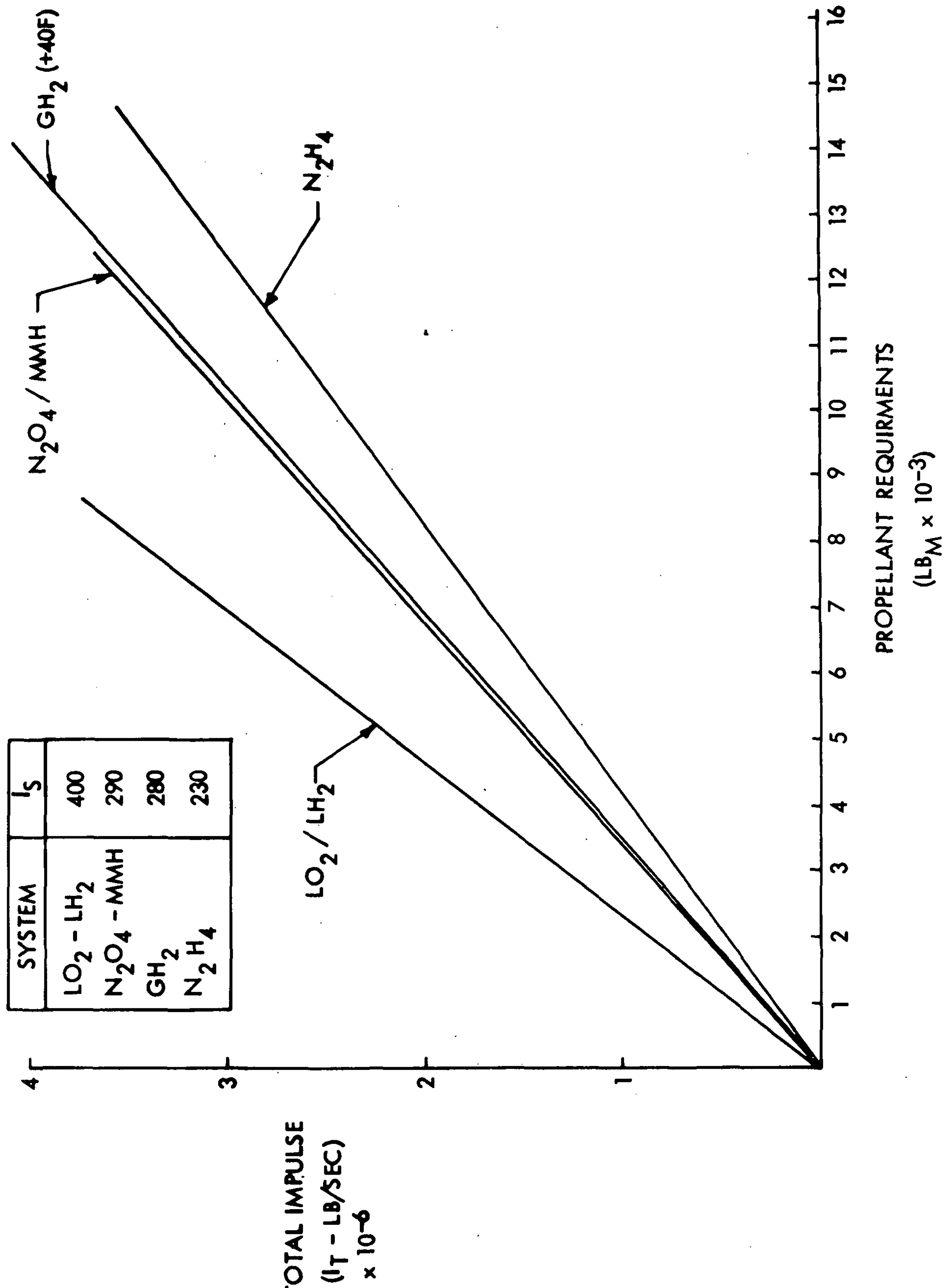


Figure 7.4-6 Propellant Requirements Vs Total Impulse

propellant for the cold gas system concept for transfer propulsion. Since the GH_2 is readily available and is eventually dumped overboard, it should be considered to improve the overall system efficiency. A discussion of the boiloff hydrogen appears below.

7.4.2.5 Use of Boiloff GH_2 for Propellant Transfer

The cold gas (GH_2) system is attractive because a certain amount of GH2 is lost due to boiloff during normal operations. The hydrogen boiloff is estimated to be about 3-1/2 lb/hr. This is a continuous loss which is substantial over long periods of time. The hydrogen is circulated around the LH_2 and LO_2 tanks and structure for cooling purposes and is then dumped overboard. At this point, the gas has been heated to about 200 - 300 R. If some means is available to accumulate this propellant, it may be possible to make use of it for the linear propulsion system propellant. This boiloff would be accumulated as sensible heat in the liquid between propellant transfer operations and would be used in a blowdown mode during transfer.

The requirement for the tug to provide the 10^{-4} g acceleration for propellant transfer is 7.5 pounds thrust. The total impulse for the ten hour transfer time is:

$$I_T = Ft$$

$$= (7.5 \text{ pounds}) \quad (3.6 \times 10^4 \text{ seconds})$$

$$I_T = 27 \times 10^4 \text{ pound-seconds}$$

The boiloff rate of 3-1/2 pounds/hour would accumulate about 350 pounds between transfers (about 100 hours). Depending on the temperature at which the H_2 is provided to the thrusters, the total impulse can vary from 52,500 lb-sec at 200 R to 97,250 lb-sec at 500 R, which is significantly less than that required for even the minimum transfer for the tug. The additional impulse can be provided by using LH_2 from the main propellant supply.

The following is a discussion of the performance and problems associated with the use of GH_2.

Figure 7.4-7 presents a plot of performance (I_{sp}) versus propellant requirements for the 27×10^4 lb-sec total impulse requirement. To minimize propellant utilization, the higher I_{sp} should be used which requires heating the hydrogen to about 500 R. This means a differential temperature of greater than 200 R. This heat requirement can be provided in at least two ways.

a. Electrical Heaters - The power to provide this heat is calculated below:

$$C_p \, (H_2) = 3.8 \text{ Btu/lb/°F}$$

$$h = 3.8 \times 200 = 760 \text{ Btu/lb}$$

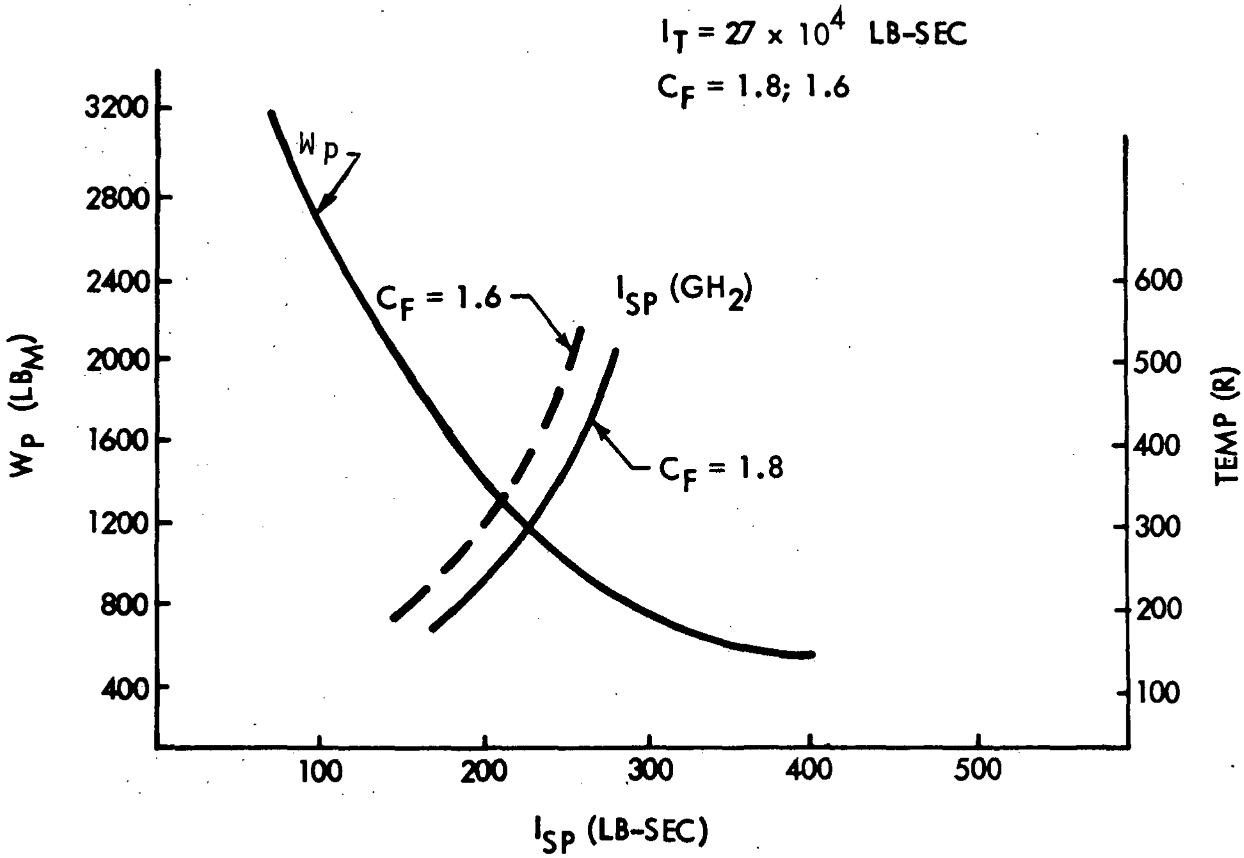


Figure 7.4-7 Propellant Requirements Vs Performance

$$\text{Power} = (2.928 \times 10^{-4})\ (760)$$

$$\text{Power} = 223\ \text{whr/lb}$$

$$P = (223\ \text{whr/lb})\ (92\ \text{lb/hr})$$

$$P = 20516\ \text{watts} = 20.5\ \text{kw}$$

The present tug configuration has one fuel cell with a total output of 3 kw. The CIS has three fuel cells with two on-line at one time. The total output of these two cells is 15 kw. A third standby fuel cell is on board and if on-line at the same time, could produce a total of 22.5 kw. This is essentially 100 percent duty cycle for the CIS to provide the high temperature H_2 for the Propulsion System. The tug would be unable to perform other necessary operations.

b. Use of solar energy to heat the hydrogen. This can be accomplished by adding a heat exchanger to the outside of the receiver vehicle. This heat exchanger for the CIS would be a coil of 1 in. x 1/2 in. aluminum tubing spaced at 35 in. around the CIS for its entire length. This is added weight, added complexity, and added costs.

With the above analysis, it appears that the use of hydrogen as a cold gas system is not the most efficient way of providing the impulse for continuous linear acceleration.

7.5 COMPARISON OF CANDIDATES

The candidate propulsion systems were evaluated and a baseline system selected as summarized in the trade table, Table 7.2-1. The cryogenic bipropellant (LO_2/LH_2) system was selected primarily on the basis of compatibility with the logistic module propellant and subsystems, thereby minimizing design complexity, and for the expected advantages of propulsion component commonality with the user vehicles.

8.0 LOGISTIC TANK THERMAL INSULATION

8.1 INTRODUCTION

The purpose of this section of the report is to present thermal performance and related information on several insulation candidates developed during the course of the study to select the logistic propellant tank baseline insulation system. The insulation system for the logistic tanks must be thermally effective to minimize boil-off losses and light in weight to maximize the amount of usable propellants that can be delivered to the in-space users. The insulation candidates evaluated were: foam, multilayer insulation (MLI), MLI with foam substrate, dewar, and dewar/MLI integrated systems.

The criteria used in the selection of the logistic tank insulation system were: thermal performance, insulation system weight, and system operational complexity. Other criteria such as system reliability, manufacturing ease, inspection, initial and operational costs of the insulation system, etc., were also considered, but were not used as the selecting criteria for the preliminary evaluation.

8.2 SUMMARY OF RESULTS

The results of the trade study are summarized in Table 8.2-1. The advantages and disadvantages of the candidate insulation concepts and their ranking on the basis of thermal performance, weight, and operational complexity are presented.

Although no candidate ranked first in all three categories, MLI, Concept 2, was selected for the logistic propellant tank insulation because of its low operational weight penalty. For a two-day average logistic timeline, the weight penalty for the MLI is about 800 pounds, whereas for the foam insulation, the penalty is over 5000 pounds. Only for a mission time of less than two hours, would the weight penalty for foam insulation be less than for MLI. Thus, MLI which ranked first on thermal performance and second on system weight is the best selection on a combined basis of thermal performance and system weight. These out weigh the lowest-ranking for MLI on the basis of operational complexity since it requires helium purge during ground operation and a means for purging during reentry and extensive manufacturing care. Current developments and experiences with MLI indicate that these requirements can be readily satisfied.

On the basis of a two-day average propellant logistic timeline, one inch MLI is selected. For a mission time longer than six days, a thickness of 1-1/2 inches for MLI would be selected.

With the current development efforts by the NASA and others, MLI would be readily available to meet the schedule for the propellant logistic program.

8.3 CANDIDATE INSULATION SYSTEMS

The insulation system candidates were: foam, multilayer insulation, dewar, and integrated systems. The integrated systems were a MLI with 1/2 inch foam substrate and a dewar with a MLI blanket on the tank walls.

Table 8.2-1 Logistic Propellant Tank Thermal Insulation Trade Table

FUNCTIONAL AND TECHNICAL REQUIREMENTS	MATRIX OF ALTERNATIVE INSULATION CONCEPTS			SELECTION
HIGHLY EFFECTIVE THERMAL INSULATION SYSTEM IS REQUIRED ON THE LOGISTIC PROPELLANT TANKS TO DELIVER MAXIMUM USEABLE PROPELLANTS TO THE USERS IN SPACE TRADE CONSIDERATIONS THERMAL PERFORMANCE SYSTEM WEIGHT OPERATION COMPLEXITY	1 FOAM PRO 1. LOW HEAT LEAK DURING GROUND OPERATION 2. MANUFACTURING EASE 3. LIGHTEST HARDWARE WEIGHT CON 1. HIGH HEAT LEAK IN SPACE	2 MLI PRO 1. LOWEST INSULATION OPERATIONAL WEIGHT 2. LOW HEAT LEAK IN SPACE 3. LIGHT WEIGHT CON 1. HIGH HEAT LEAK DURING GROUND OPERATION 2. REQUIRES HELIUM PURGE DURING GROUND OPERATION 3. REQUIRES RE-ENTRY PURGE SYSTEM 4. REQUIRES EXTENSIVE MANUFACTURING CARE	3 DEWAR PRO 1. FAIRLY LOW HEAT LEAT AT ALL PHASES CON 1. HEAVY WEIGHT 2. MUST MAINTAIN HIGH VACUUM	• THERMAL PERFORMANCE 2-4-5-3-1 SYSTEM WEIGHT 1-2-4-3-5 OPERATIONAL COMPLEXITY 1-3-5-4-2 • BASED ON LOGISTIC MODULE 2 DAY AVERAGE PROPELLANT LOGISTIC TIME LINE SELECTED CONCEPT CONCEPT 2 MLI
		4 MLI + FOAM SUBTRATE FOAM SUBSTRATE PRO 1. LOW HEAT LEAK AT ALL PHASES 2. ALLOWS USE OF GN_2 PURGE CON 1. REQUIRES RE-ENTRY PURGE SYSTEM 2. REQUIRES EXTENSIVE MANUFACTURING CARE	5 DEWAR + MLI PRO 1. LOW HEAT LEAK AT ALL PHASES CON 1. HEAVY WEIGHT 2. MUST MAINTAIN HIGH VACUUM	

Installation schemes of the insulation systems are illustrated in Figure 8.3-1. The left half of the figure shows the foam, MLI, and the MLI/foam integrated system, while the right half of the figure shows the dewar and the dewar/MLI integrated system.

8.3.1 Foam

Foams have been used to insulate cryogenic tanks for commercial and space vehicle applications for many years. Among many types of foam available, polyurethanes are by far the most widely used for the cryogenic insulation purposes. The advantages of this insulation include: low thermal conductivity; very good physical properties at cryogenic temperatures; light weight; easy to handle, fabricate, and maintain, self-supporting structure; and relatively low costs. The disadvantage is the divots caused by cryo pumping. To prevent an occurrence of divots caused by cryo pumping, foam layers on cryogenic tanks are sealed against condensible gases. The close cell polyurethane foam, when it reaches the vacuum of space, expands out and reduces its thermal conductivity, although it never reaches the low thermal conductivity of the other candidates.

8.3.2 Multilayer Insulation (With and Without Foam Substrate)

Multilayer insulation has become a strong contender for use in space vehicle applications due to its lower thermal conductivity in space. MLI consists of many layers of radiation-reflecting shields separated by either low-conductivity spacers or embossments (or crinkles) on the shields. This assembly is placed perpendicular to the flow of heat. Each layer contains a thin, low emissivity radiation shield enabling the layer to reflect a large percentage of the radiation it receivers from a warmer surface. The radiation shields are separated from each other to reduce the heat transferred from shield to shield by solid conduction. The gas in the space between the shields is removed to decrease the conduction by gas molecules.

Various MLI concepts and methods of installation have been investigated by NR and other companies including Goodyear, McDonnel Douglas, Lockheed General Dynamics-Convair, National Research Corp. The MLI can be categorized in two classes depending on the material and method used between the shields; i.e., semi-rigid and spacerless. A semi-rigid MLI employs low-conductivity material as a spacer between the shields, consequently, is relatively rigid as compared to those of the spacerless class. A spacerless MLI eliminates the spacers and instead uses embossments or crinkles in the shields to separate them. The insulations in this class are lighter in weight and many of them have lower thermal conductivities as compared to the semi-rigid types. NR has done extensive research on the spacerless types of insulation using single aluminized mylar (SAM) and has insulated a 105-inch diameter tank with SAM for the NASA evaluation. SAM uses a post support in which dacron straps and small pins are used to support the insulation at a fixed layer density. In this way, the heat transfer rate can be controlled more closely than most other available MLI types.

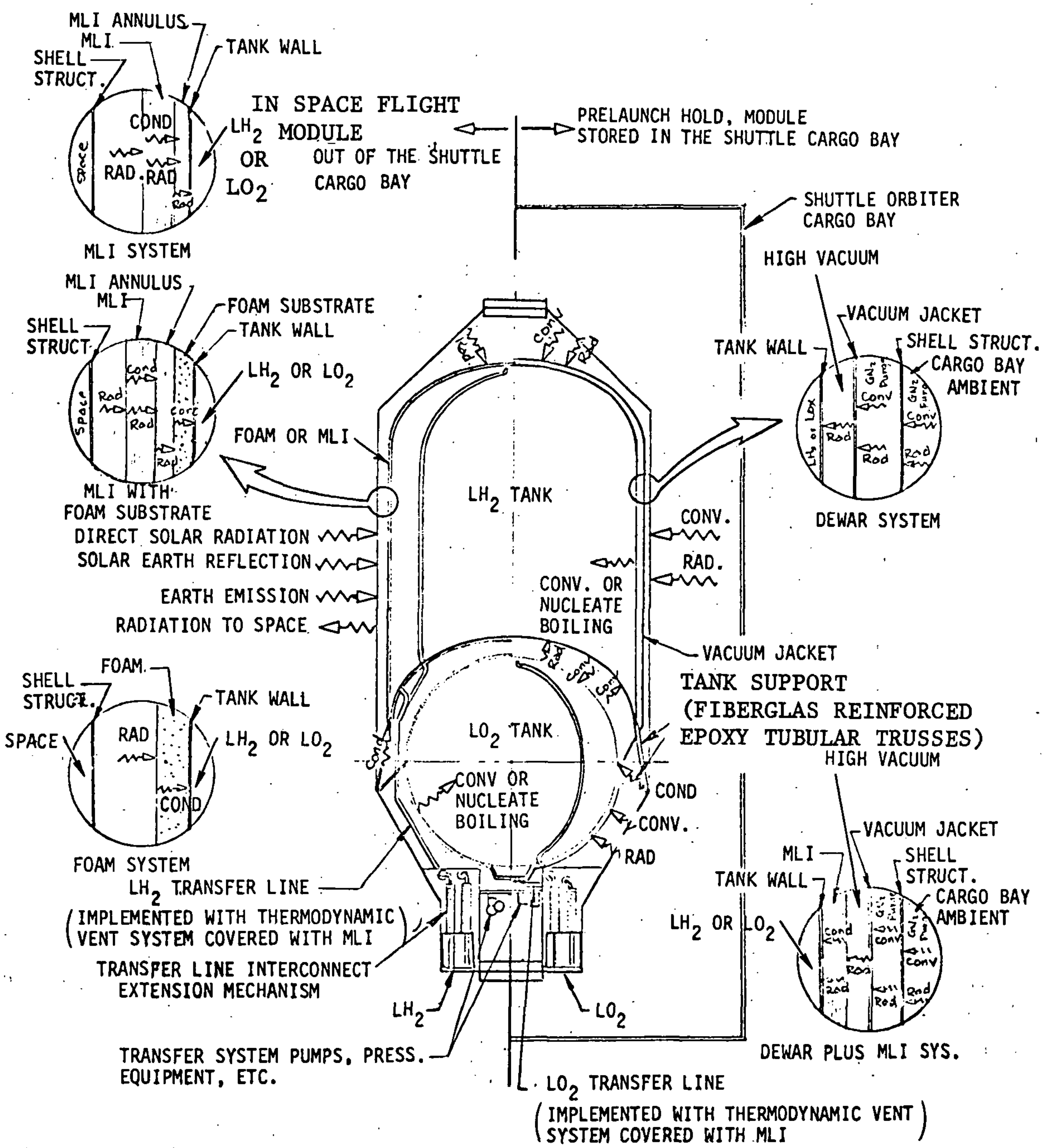


Figure 8.3-1 Modes of Heat Transfer

8.3.3 Dewar (With and Without MLI Blanket)

Dewars have been used in commercial applications to store cryogenic fluids for many years. A dewar is a double-walled container with the gas evacuated from the annulus between the two walls. The evacuation of gas from the annulus minimizes or eliminates heat transfer by conduction of the gas and makes radiation between the two walls the dominating heat transfer mode. Improvements to the basic dewar concept is mainly in the insulating method and material used between the two walls. In small laboratory dewars, the improvement consists of silvered walls and maintaining higher vacuum levels for longer periods. In larger vessels, insulations such as powders, fibrous materials, foam, or MLI are used to reduce the radiation heat transfer. In this study, a layer of MLI on the inner vessels, the LH_2 and LO_2 tank walls, was considered. The outer vessel or vacuum jacket was assumed to house both the LH_2 and LO_2 tanks as illustrated in Figure 8.3-1.

An important attribute of the dewar system is that once it is manufactured and evacuated, it should continue to maintain the vacuum for many years with only the need for a re-evacuation of the annulus at specific intervals. When integrated with an MLI blanket, the thermal conductivity of the MLI in space can be attained for all mission phases. The vacuum jacket, being a hard shell, is more difficult to damage and offers to withstand high aerodynamic loads. The rugged vacuum jacket should keep the MLI insulation from being damaged by operating and manufacturing personnel, and it is sealed which should protect the MLI from atmospheric degradation.

Any damage causing leakage into the vacuum space can be a serious problem. Even small pin hole leaks, either from the atmosphere or from the contained cryogen, can raise the thermal conductivity by orders of magnitude in a matter of minutes or hours. An incorporation of vacuum-ion pumps on the vacuum jacket and using it as a vacuum gage will give a continuous verification of operational readiness and immediate sensitivity to any internal leakage of fluid. Another disadvantage to the dewar design is the great weight associated with the vacuum jacket; 2100 pounds was estimated for the vacuum jacket and associated hardware.

8.4 INSULATION HEAT TRANSFER ANALYSIS

Determination of the heat transfer rate into the propellant tanks for the candidate insulation systems was made to establish their thermal performance. The heat transfer paths or heat leak paths to the on-board propellants were divided in two basic areas: the tank wall insulation and tank penetration. The tank penetration includes the tank supports, propellant logistic transfer lines, fill and drain lines, vent lines, pressurization lines, instrumentation bosses, etc. Detail discussion on the estimates of the propellant heat leaks through the tank wall insulations and through the tank penetrations are presented in later paragraphs.

Modes of heat transfer considered in the analysis to develop the thermal models are shown in Figure 8.3-1. The right half of the figure shows the heat transfer modes for the prelaunch hold condition while the logistic module is stored in the shuttle orbiter cargo bay. Heat transfer to the module outer shell structure was by convection of the cargo bay purge gas and by radiation of the cargo bay structure. The cargo bay structure was assumed to be 120 F. The 120 F cargo bay was also a ground rule specified by the NASA for the space tug point design study. Heat transfer to the propellant tanks were radiation from the module shell structure, convection of the module purge gas, radiation and/or conduction of the insulation system, and conduction through the tank supports and other tank penetrations.

Modes of heat transfer of the logistic module in space flight out of the cargo bay are shown on the left half of the figure. The orbital heat loads consisting of direct solar radiation, reflected solar radiation, and earth emitted radiation were applied directly to the module shell structure. Heat transfer to the propellant tanks were radiation from the module shell structure, radiation and/or conduction of the insulation system, and conduction through the tank supports and other tank penetrations. The logistic module in space receives heat from the sun plus reflected heat from nearby planetary bodies as stated. Simultaneously, heat is radiated from the vehicle to space and also to nearby planetary bodies. Similarly, radiative heat is exchanged between the tank walls and the module shell and surrounding structures. Under equilibrium condition, the difference between the heat absorbed and the heat emitted from the tank outer surfaces results in heat leak to the stored cryogen.

Two thermal models for each insulation configuration were developed; i.e., one model with the logistic module stored in the shuttle orbiter cargo bay and another model with the module in orbit out of the cargo bay. The models that were used in calculating heat transfer rates to the module propellants were used with an NR developed general thermal analyzer digital computer program. The IBM 360 computer program used for calculation of propellant heat transfer rates and transient temperatures is a compiler-type general heat transfer program which solves any transient or steady-state thermal problem whose finite difference equation can be represented by a simple electrical network. It is an n-dimensional program which includes heat transfer by conduction, radiation and convection. Since the heat transfer problem is represented by the analogy of an electrical network, the problem will consist of node points and conductors between the node points. The node points may have finite capacitance. Boundary node points have no capacitance and their potential value may be a constant or some function of other problem variables.

8.4.1 Heat Leak Through Tank Wall Insulation

The heat leak calculations on the LH_2 and LO_2 tank sidewalls and the intertank bulkhead were made parametrically for various insulation thicknesses and vacuum levels. The heat leak rates were determined for ground hold, boost (MLI only), and space flight.

8.4.1.1 Foam

The calculated heat leak rates to the LH_2 and LO_2 tanks insulated with polyurethan foam are presented in Figure 8.4-1 as a function of the foam thicknesses. The heat leak rates are given for ground hold and space flight. The ground heat leak rates are for the logistic module being stored in the GN_2 purge cargo bay environment of the space shuttle. The cargo bay structure temperature was

NOTE:

1. GROUND HEAT LEAK ASSUMES THAT LOGISTICS MODULE IS STORED IN THE GN_2 PURGED SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS 120 F
2. SPACE HEAT LEAK ASSUMES THAT MODULE IS OUT OF THE CARGO BAY AND IS ORBITING WITH ITS BROADSIDE TO THE SUN
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED

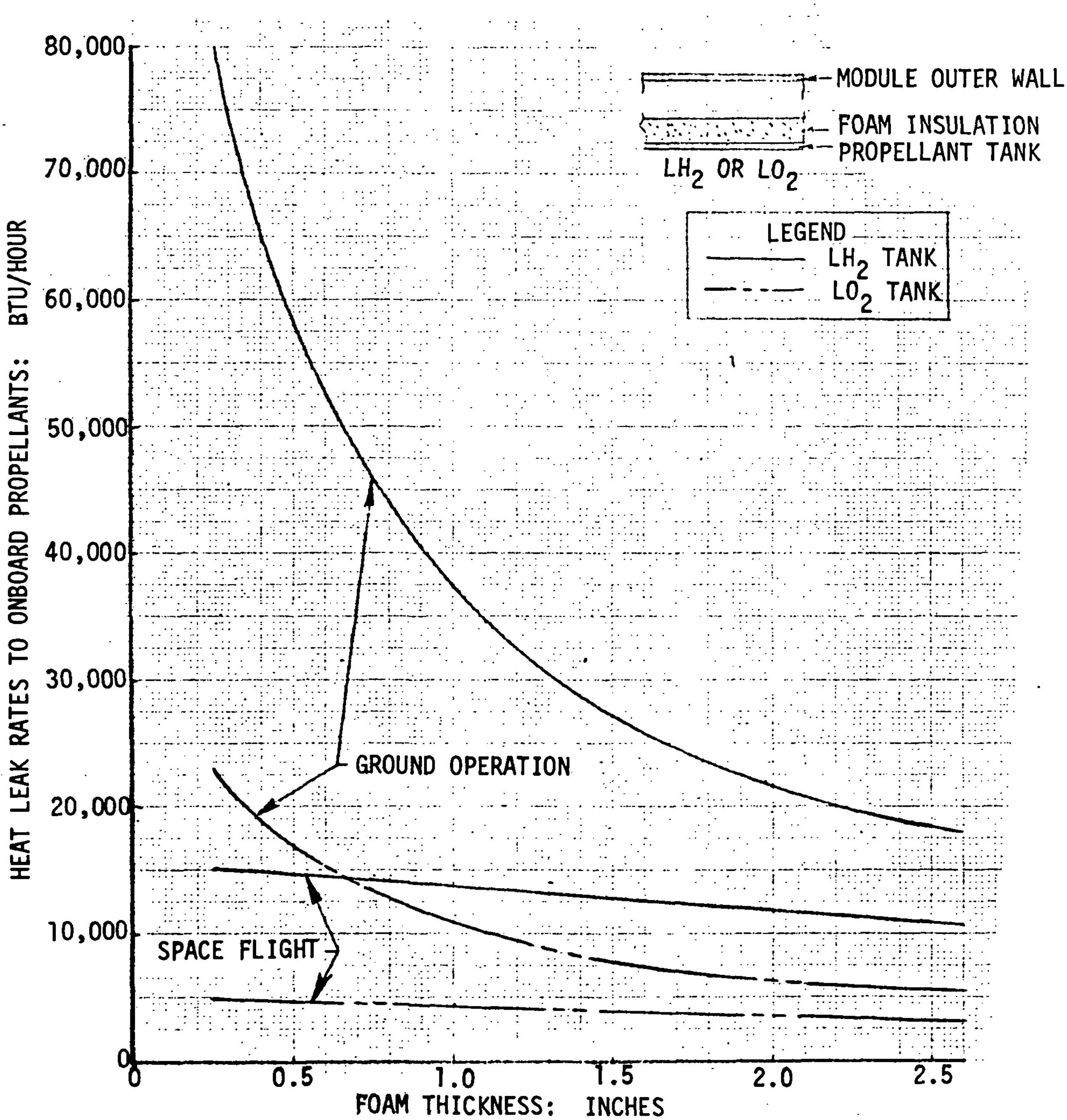


Figure 8.4-1 Heat Leaks to the Onboard Propellants Through the Tank Walls Insulated with Foam

assumed to be 120°F. The space flight heat leak rates are for the module out of the cargo bay with its broadside exposed to the sun.

8.4.1.2 Multilayer Insulation (With and Without Foam Substrate)

The study of heat leak into the propellant tank insulated with MLI was based on using SAM which is embossed single aluminized mylar with perforations for venting. The SAM concept relies on the embossment pattern to minimize conduction between shields. However, the embossment pattern in mylar shields is adversely affected at temperatures above 140°F. For this reason, the insulations in the areas that are subjected to high aerodynamic heating rates may require the use of embossed single aluminized kapton of which upper temperature limit is 350 F. The effective thermal conductivities of single aluminized mylar (SAM) and single aluminized kapton (SAK) will be essentially the same in space, since heat transfer through the insulation in space is by radiation and conduction which are primarily dependent on the surface properties of the reflective coating and embossment pattern respectively. An investigation is being conducted for a high temperature MLI. The use of gold-coated Kapton film will permit temperatures up to 500 F without damage.

The MLI system exhibits extremely low thermal conductivity in space as compared to that of foam insulation. However, the MLI thermal conductivity is higher than foam conductivity when purged with helium during ground operation. The helium purge is required to prevent gas condensation on the reflective shields since moisture will attack the reflective surfaces, and consequently degrade the MLI performance. MLI with foam substrate permits the use of GN_2 purge, resulting in lowering the thermal conductivity during ground operation. MLI in a high vacuum dewar has very low thermal conductivity at all times, but the drawback of this integrated system is its considerably heavy hardware weight. The MLI with no dewar is free to vent during the boost phase, but requires repressurization prior to re-entry to prevent moist air ingressing between the layers. The ground purging and re-entry repressurization add complexity to the MLI operation. The MLI/dewar system eliminates the on-board repressurization system.

The calculation of heat leak through the MLI included heat transfer by gas conduction, MLI conduction, MLI radiation and conduction through the support posts. The gas conduction was based on equations given in "Cryogenic Systems" by Randall Barron and varied as a function of both mean temperature and pressure. For ground heat leak calculations, the MLI was assumed filled with helium gas at a pressure of 760 torrs and for space applications, a pressure of 0.31×10^{-4} torr was used. The MLI conduction and MLI radiation were computed from a universal equation for effective thermal conductivity developed by NR by using standard curve fitting techniques for test data. The equations which were programmed into the general thermal analyzer are shown below:

$$K_{EFF} = K_{COND} + K_{RAD} \tag{8.4-1}$$

$$K_{COND} = AN^{C}(T_H + T_C)/2 \tag{8.4-2}$$

$$K_{RAD} = B\sigma(F_E + \tau)\ (T_H^{\,4} + T_C^{\,4})/12\ N\ (1 - \tau)\ (T_H + T_C) \tag{8.4-3}$$

where

$A = 2.05 \times 10^{-16}$ (Conduction Coefficient)

$N = 60$ Layers/Inch

$C = 4.75$

T_H = Hot Side Temperature

T_C = Cold Side Temperature

$B = 0.84$ (Radiation Coefficient)

σ = Stefan-Boltzman Constant

$F_E = 1/(1/\varepsilon_1 + 1/\varepsilon_2 - 1)$

$\varepsilon_1 = 0.045$ (Aluminized Side Emissivity)

$\varepsilon_2 = 0.35$ (Mylar Side Emissivity)

$\tau = 0.01$ (Percent of Area with Perforations)

The degradation of SAM thermal performance by posts is accounted for by a solid conduction connected between the inner and outer MLI surfaces. The posts are assumed to be made of fiberglass reinforced phenolic and one post every two square feet.

During ground-hold and launch-to-orbit phases of vehicle operation, energy will be transferred through MLI composites by gas conduction in addition to radiation and solid conduction. Ordinary gas conduction occurs during ground-hold because the gas pressure between MLI layers is near atmospheric pressure. The variation in thermal conductivity of atmospheric pressure helium gas with temperature is nearly linear between ambient and cryogenic temperature.

During launch-to-orbit, a typical MLI composite will evacuate itself to the low pressure corresponding to orbital altitude. The gas conduction mechanism at orbital pressure differs from ordinary conduction at atmospheric pressure. Gas molecules rarely collide, thus an individual gas molecule travels across the space between MLI layers without exchanging energy (free molecular regime). An analysis of the energy transferred by molecular conduction is presented in "Cryogenic Systems" by Randall Barron. The following result which applies to parallel plates and concentric cylinders and spheres was obtained:

$$Q/A = F_a \, GP \, (T_h - T_c), \qquad (8.4\text{-}4)$$

where G is an equivalent conductivity coefficient defined by

$$G = \frac{\gamma + 1}{\gamma - 1} \left(\frac{g_c R}{8 \pi MT} \right)^{1/2} \qquad (8.4\text{-}5)$$

and F_a is the accommodation coefficient factor defined by

$$\frac{1}{F_a} = \frac{1}{a_1} + \frac{1}{a_2} - 1 \qquad (8.4\text{-}6)$$

NR has extended this result to include the effects of multiple layers, variable pressure, and temperature dependent properties. When more than two participating surfaces are involved, the geometry laws used in radiation can be applied. Applying the summation technique for adding radiation conductance in series, the following gas conduction result is obtained:

$$Q/A = F_a \frac{GP}{N} (T_2 - T_1). \qquad (8.4\text{-}7)$$

The use of N in the summation instead of N-1 or N+1 as other authors have obtained is a matter of computational convenience. It is expected that this term will be multiplied by an experimental coefficient when evaluating the effective thermal conductivity of MLI composites and that layer density will be an important variable.

Published thermal conductivity data for common gases suggests that K is relatively constant at high pressure where continuum effects predominate and that K varies linearly with pressure at low pressures where free molecule effects predominate. NR uses the following expression to evaluate pressure dependent thermal conductivity values.

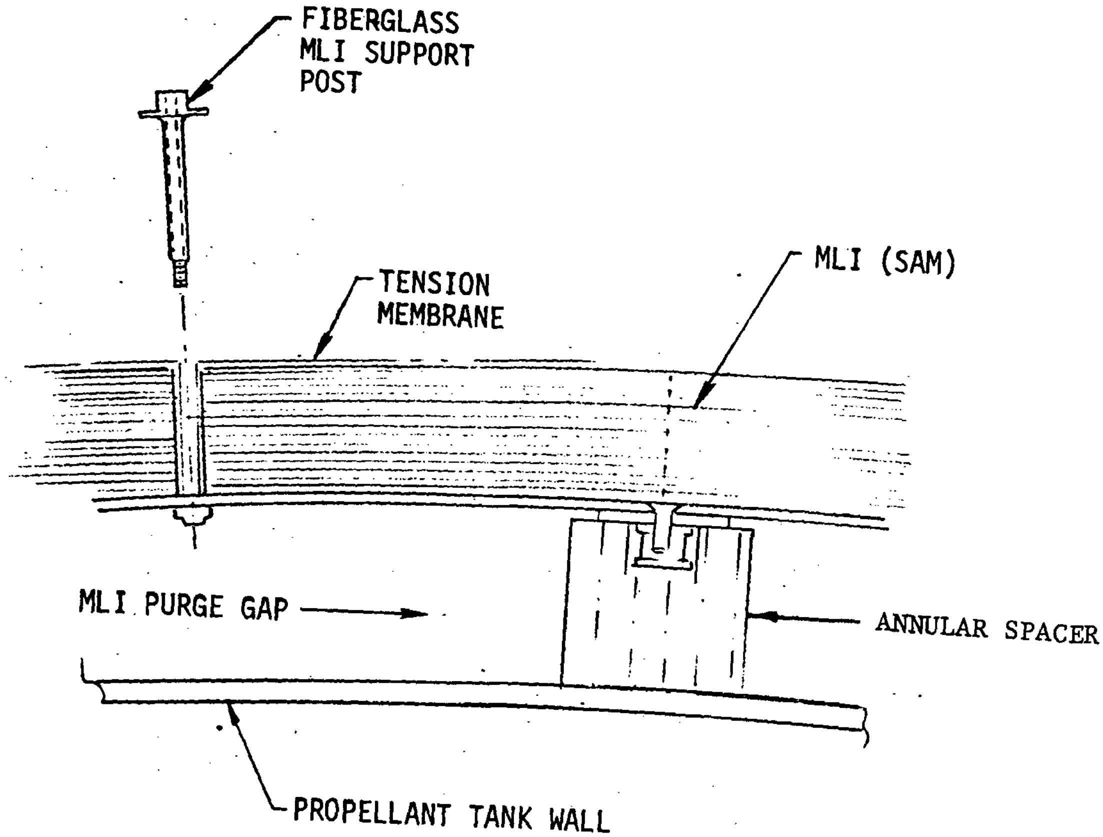
$$\frac{1}{K} = \frac{1}{K\infty} + \frac{N}{F_a\, GP} \qquad (8.4\text{-}8)$$

The expression reduces to the proper limiting values at the pressure extremes, reasonably predicts the effective K in the transition flow regime between the continuum and free molecule flow regimes, and is easy to program for thermal analyzer calculations.

The effect of temperature dependent properties on effective thermal conductivity is accounted for by evaluating Equation 8.4-8 at different temperatures in the cryogenic range. The resulting K values are input to the thermal analyzer as bivariable curves with temperature as the independent variable and pressure as the parameter.

The equivalent network shown on Figure 8.4-2 was prepared to calculate heat leak and temperature distributions. The temperature distribution through SAM was represented by four nodes: one at the LH_2 tank side surface, and interior nodes at the one-third points. The energy transfer between nodes is represented by three parallel conductors; one each for radiation, solid conduction, and gas conductivity. The effects of lateral conduction parallel to laminations of MLI have been neglected. The additional conductor connected between the inner and outer MLI surfaces represents the MLI post.

The heat leak rates of the propellant tank walls insulated with MLI and MLI with 1/2-inch foam substrate are presented in Figures 8.4-3 and 8.4-4, respectively. The heat leak rates are plotted as a function of the MLI thicknesses. Reduction



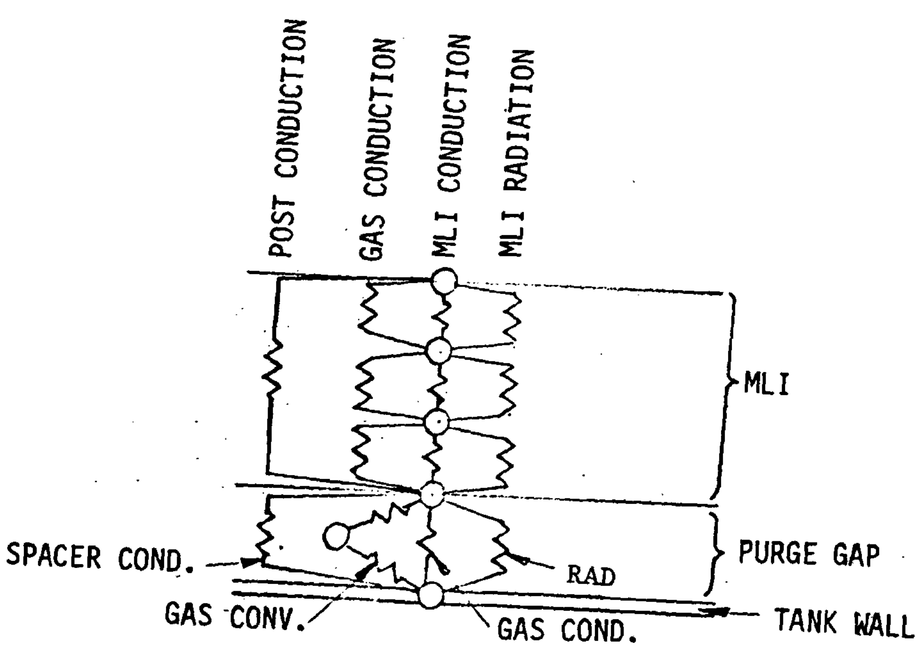


Figure 8.4-2 MLI Installation and Thermal Network

NOTE:

1. GROUND HEAT LEAK ASSUMES THAT LOGISTICS MODULE IS STORED IN THE GN_2 PURGED SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS 120 F. THE MODULE IS PURGED WITH HELIUM.
2. SPACE HEAT LEAK ASSUMES THAT MODULE IS OUT OF THE CARGO BAY AND IS ORBITING WITH ITS BROADSIDE TO THE SUN. MLI INTERSTITIAL PRESSURE $= 3.1 \times 10^{-5}$ TORR.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED .

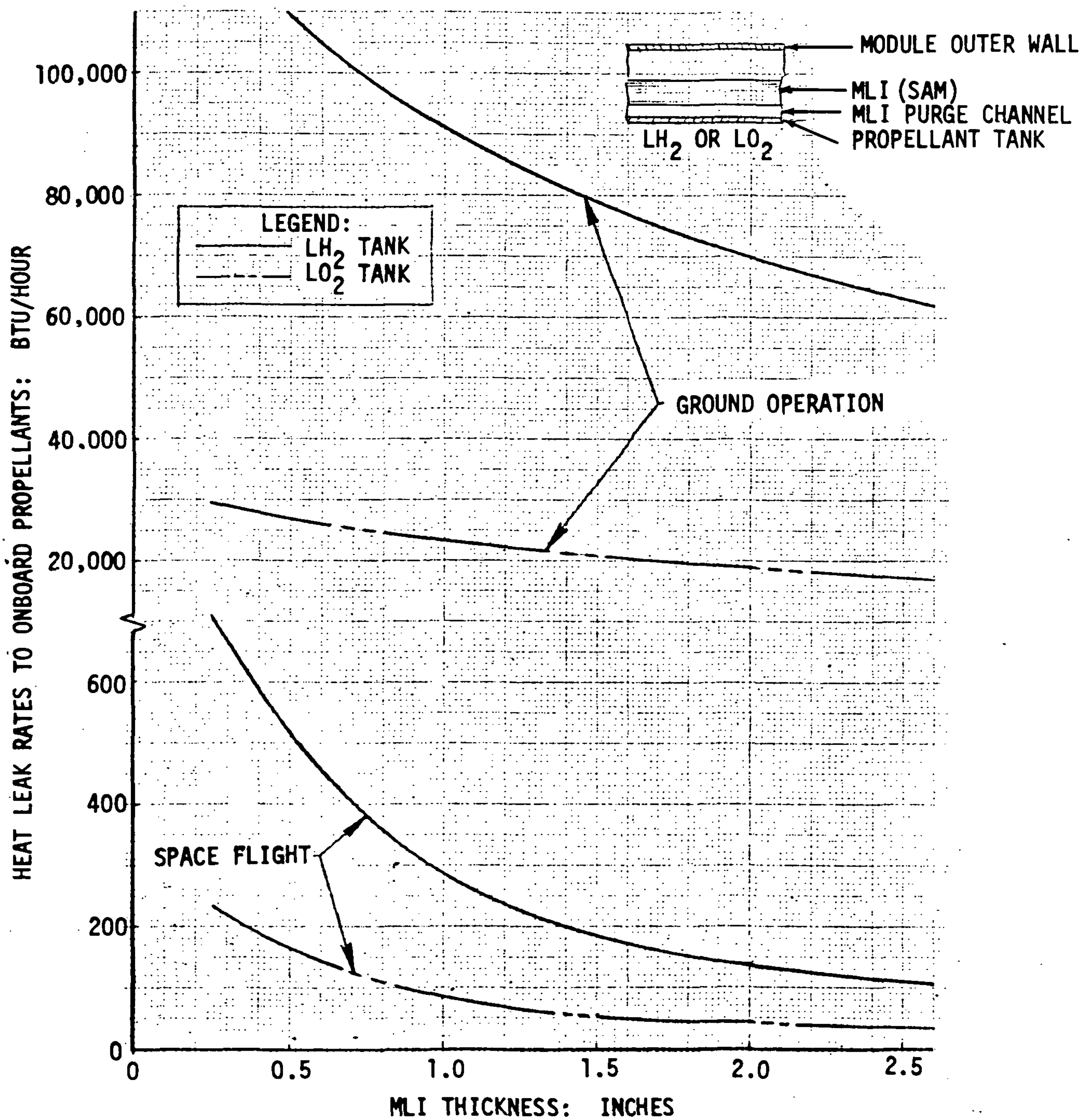


Figure 8.4-3 Heat Leaks to the Onboard Propellants Through the Tank Walls Insulated with MLI

NOTE:

1. GROUND HEAT LEAK ASSUMES THAT LOGISTICS MODULE IS STORED IN THE GN_2 PURGED SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS 120 F. MLI IS PURGED WITH GN_2.
2. SPACE HEAT LEAK ASSUMES THAT MODULE IS OUT OF THE CARGO BAY AND IS ORBITING WITH ITS BROADSIDE TO THE SUN. MLI INTERSTITIAL PRESSURE $= 3.1 \times 10^{-5}$ TORR.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED.

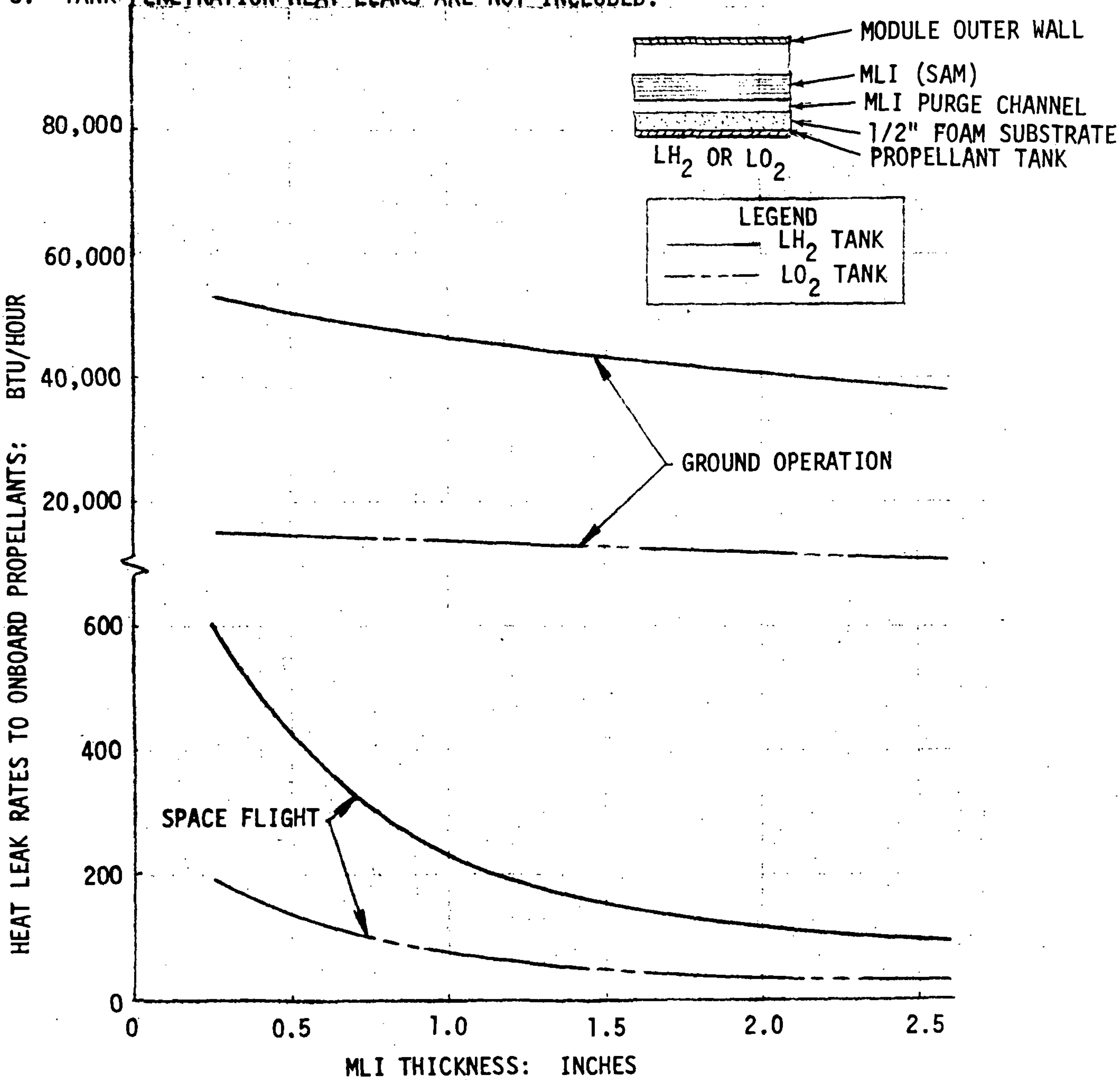


Figure 8.4-4 Heat Leaks to the Onboard Propellants Through the Tank Walls Insulated with MLI and 1/2 Inch Foam Substrate

in the heat leak rates can be realized by an addition of 1/2-inch foam substrate. The foam substrate permits the use of GN_2 for the MLI purge during ground operation instead of helium and reduces the ground heat leak rates by the lower thermal conductivity of GN_2.

Assessments of the integrated heat leaks to the on-board propellants during the launch ascent to orbit sequence are necessary in calculations of the insulation system operational weights. Computer program runs were performed to determine the integrated heat leaks for the MLI insulated tanks since a large variation in the heat leak rates occurs during the ascent phase. The determinations of the integrated heat leaks for the foam and dewar systems were performed by hand calculation since the heat leak rates of these insulation systems are not affected significantly by the change in the atmospheric pressure during ascent phase. It was assumed that the change in the cargo bay structure temperature due to aerodynamic heating would not significantly affect the heat leak rates if the cargo bay were well insulated.

Figures 8.4-5 and 8.4-6 present the integrated heat leaks to the MLI insulated tanks during the launch ascent to orbit sequence. The integrated heat leaks are given for several MLI thicknesses between 1/2 to 2 inches. The interstitial pressure of the MLI blanket in the study was estimated from the predicted ascent pressure profile for a typical launch vehicle compartment shown in NR report SD 71-263, "Cryo Storage Thermal Improvement." The analysis assumed that the pressure is uniformly distributed throughout the MLI blanket at any time. The integrated heat leak curves of the figures show a high heat leak during the first few minutes of boost followed by a sharp reduction in the heat leak resulting from improvement of the MLI performance due to outgassing.

The integrated heat leaks for the MLI with 1/2-inch foam substrate were estimated from the results obtained for the MLI with no substrate. The MLI interstitial gas pressure profile during ascent phase was unable to be determined due to lack of the necessary data and information relative to the outgassing rate of foam in vacuum. The outgassing of the foam substrate would maintain the MLI interstitial gas pressure at a higher level and would affect the MLI thermal performance. It is anticipated that the outgassing rate is high at the beginning of flight and gradually reduced to nil. However, the outgassing effect to the MLI performance is not expected to be significant, especially for the foam substrate that is laid directly on the cryogenic tank surface and is maintained at a cryogenic temperature range. Laboratory tests would be required to investigate the foam outgassing phenomena under various vacuum levels and wide foam temperature range.

8.4.1.3 Dewar (With and Without MLI Blanket)

For a dewar, heat is transferred across the vacuum annular space by radiation from the vacuum jacket to the cold inner wall, by gaseous conduction through the residual gas in the annular space, and by solid conduction through spacers, propellant and pressurization lines, instrumentation bosses, etc. A measure of the heat transfer by gas conduction is the effective thermal conductivity of the residual gas. Gas heat transfer at ordinary pressure is called continuum flow because the fluid may be treated as continuous medium in analyzing the heat

NOTE:

1. LOGISTIC MODULE IS STORED IN THE SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS ASSUMED 120 F.
2. DURING PRELAUNCH HOLD, THE CARGO BAY IS PURGED WITH GN_2 WHILE LOGISTIC MODULE IS PURGED WITH HELIUM.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED.

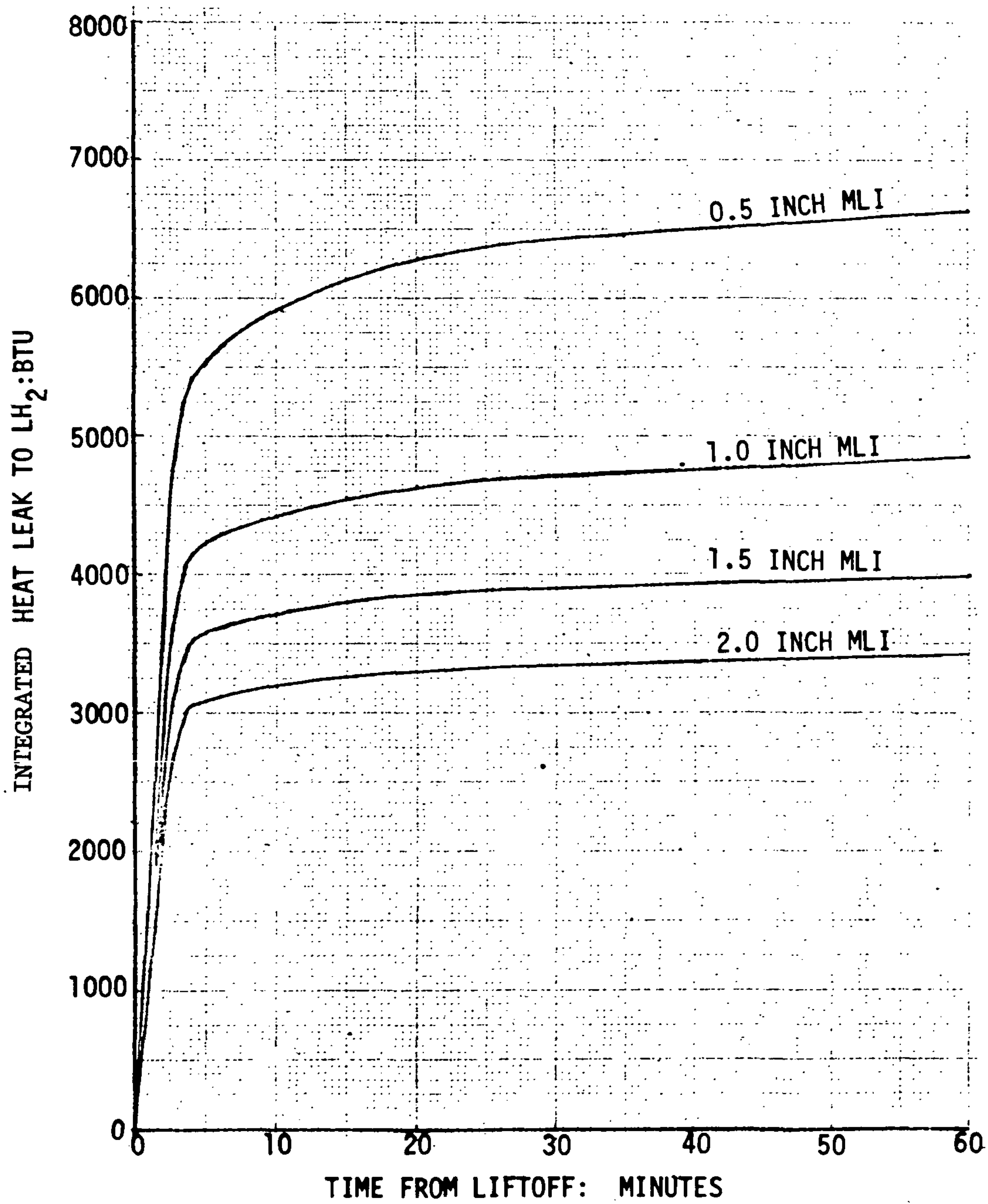


Figure 8.4-5 Integrated Heat Leaks to the MLI Insulated LH_2 Tank During Launch Ascent to Orbit

NOTE:
1. LOGISTIC MODULE IS STORED IN THE SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS ASSUMED 120 F.
2. DURING PRELAUNCH HOLD, THE CARGO BAY IS PURGED WITH GN_2 WHILE LOGISTIC MODULE IS PURGED WITH HELIUM.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED.

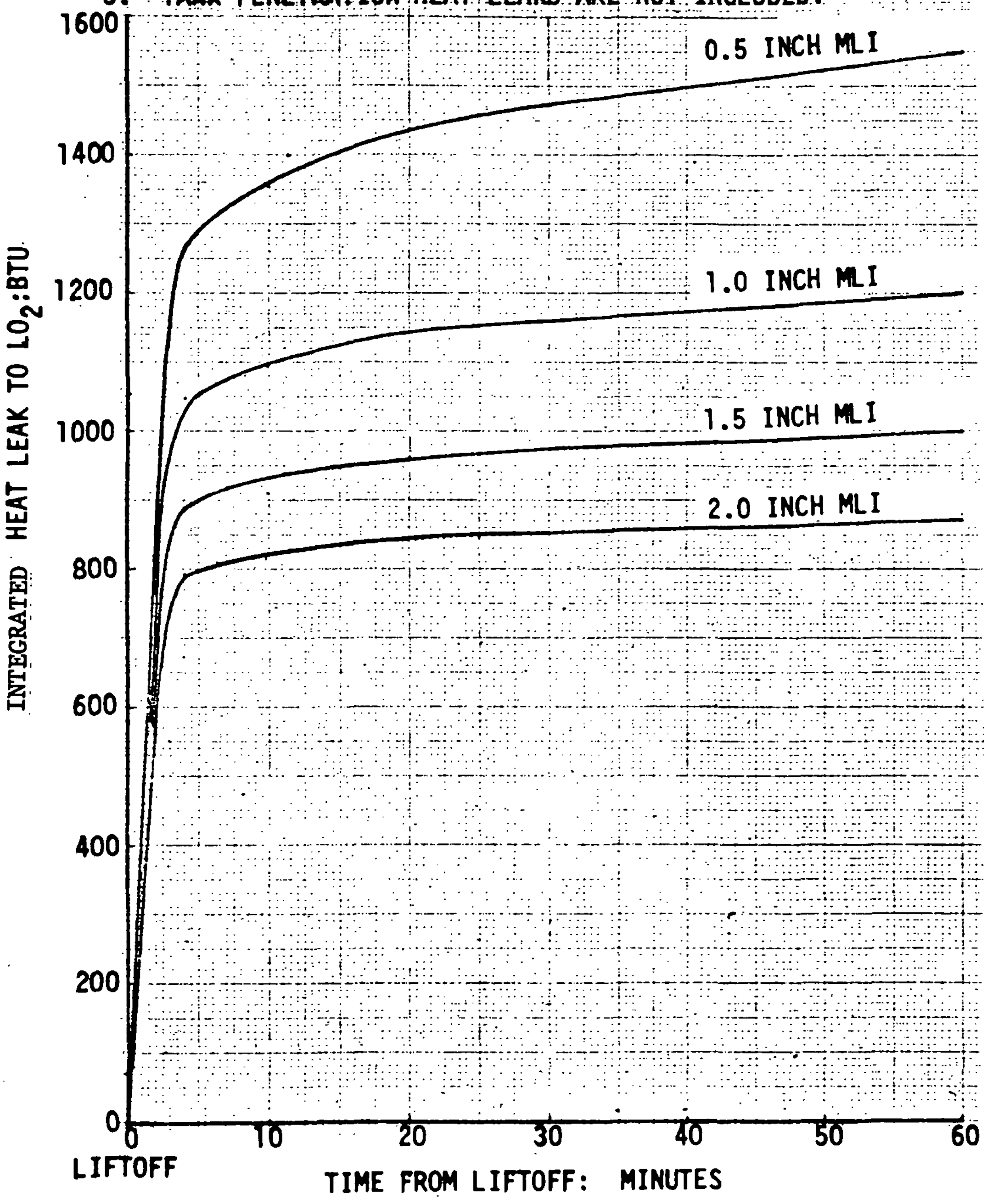


Figure 8.4-6 **Integrated Heat Leaks to the MLI Insulated LO_2 Tank During Launch Ascent to Orbit**

flow situation. Thermal conductivity in this regime is independent of gas pressure. At low gas pressures, however, a gas cannot be treated as a continuous medium, and another flow regime is obtained, free molecular flow. In the free-molecular-flow regime, the mean flow path of the gas molecules strike the sides of the annulus more often than they strike each other. Between the continuum-flow and free-molecular-flow regime is a mixed flow regime which is called a transition-flow regime. The technique and equations for the MLI were used to calculate the gas thermal conductivity. Equation 8.4-8 was modified to account for the large annular spacing considered in the study.

$$\frac{1}{K} = \frac{1}{K_\infty} + \frac{1}{xF_a GP} \qquad (8.4\text{-}9)$$

where x = spacing between the inner and outer walls.

The ground hold and space flight heat leak rates of the LH_2 and LO_2 tanks with a dewar-type construction are presented in Figures 8.4-7 and 8.4-8, respectively. The ground-hold heat leak rates are for the logistic module being stored in the GN_2 purged shuttle cargo bay. The cargo bay structure temperature was assumed to be 120 F. The spaceflight heat leak rates are for the logistic module out of the cargo bay and orbiting with its broadside to the sun. The heat leak rates are given as the functions of the interstitial helium gas vacuum levels and the size of gap between the tank wall and vacuum jacket. The heat leak rates are lower with larger gaps except for the 9 inches of gap where higher rates at the helium pressure above 5 x 10^{-2} torr are obtained. The gap between the stage shell structure and LH_2 tank wall is 9 inches. Therefore, the 9 inches of gap is the situation where the stage shell structure is constituting the vacuum jacket. The higher heat leak of the 9-inch gap is attributed to the elimination of the vacuum jacket which is providing a radiation barrier in addition to the pressure carrier.

Effect of the interstitial helium pressure variation to the propellant tank heat leaks are evident on Figure 8.4-7 and 8.4-8. Until the helium pressure is reduced to about one torr, the heat leak rates remain constant, indicating that the helium conductivity at this pressure range is in the continuum-flow regime. At the pressures below one torr, the helium conductivity is in the transition-flow and free-molecular-flow regimes and resultant sharp reduction in the heat leak rates occur. The heat leak rates reach to the minimum at about 5 x 10^{-4} torr and remain constant below this pressure. At this low pressure level, the heat transfer by the gas conduction becomes nil and the radiation between the annular surfaces begins to dominate the heat transfer. For this reason, the spacing of the vacuum jacket has no effect on the heat leaks. The minimum heat leak rates for the ground hold are 6980 Btu/hr and 2280 Btu/hr for the LH_2 and LO_2 tanks, respectively. For the space flight, the respective minimum values are 3880 Btu/hr and 1240 Btu/hr, referring to the figures.

To improve the thermal performance of the dewar system, several types of insulation have been used in the annular space, as previously discussed. In the study, a layer of MLI was considered since it has shown the most effective means to reduce heat transfer under high vacuum. The heat transfer equations for the MLI

NOTE:
1. GROUND HEAT LEAK ASSUMES THAT LOGISTIC MODULE IS STORED IN THE GN_2 PURGED SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS 120 F.
2. SPACE HEAT LEAK ASSUMES THAT MODULE IS OUT OF THE CARGO BAY AND IS ORBITING WITH ITS BROADSIDE TO THE SUN.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED

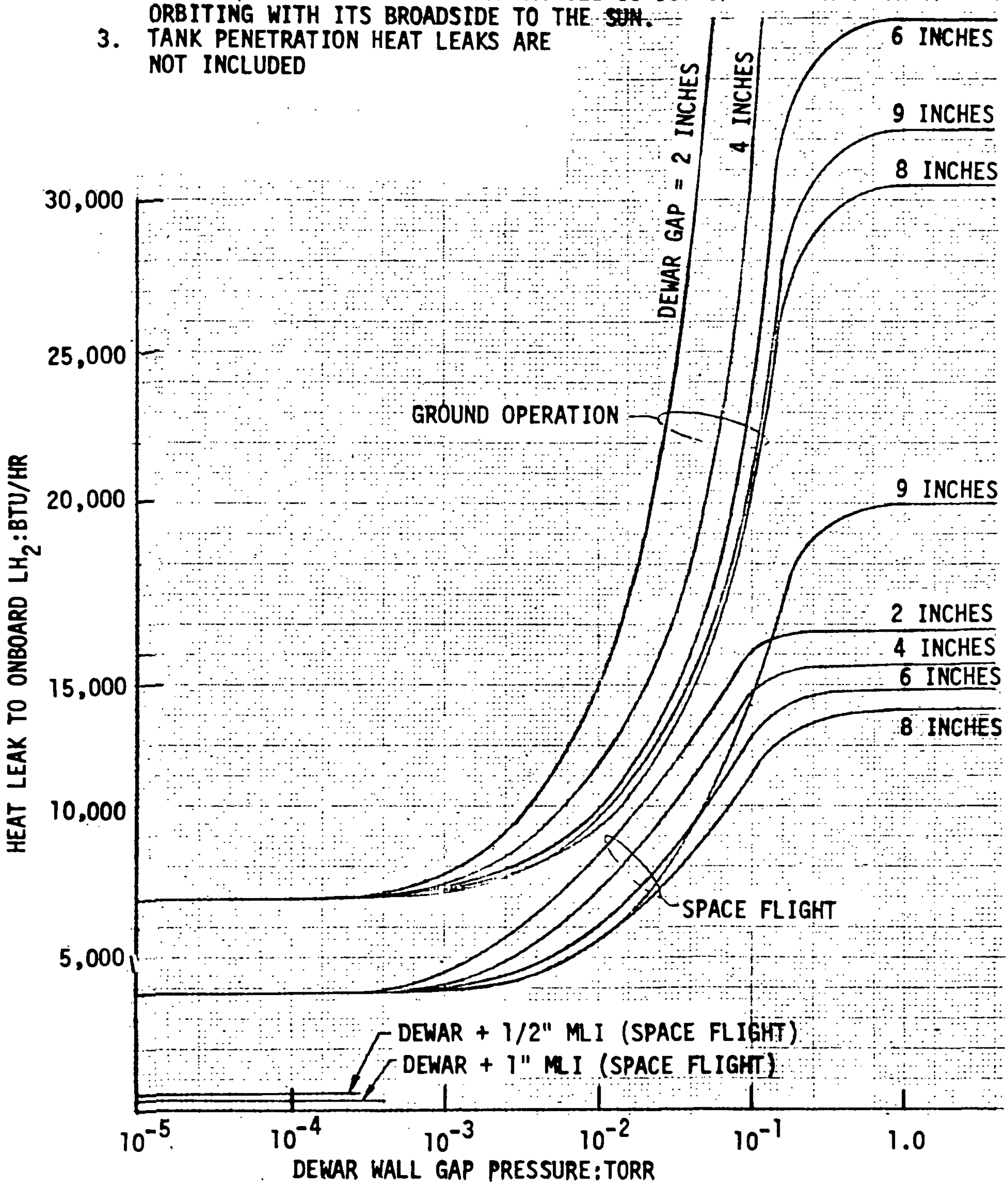


Figure 8.4-7 Heat Leaks to the Onboard LH_2 Through the Dewar Type Tank Wall

NOTE:

1. GROUND HEAT LEAK ASSUMES THAT LOGISTIC MODULE IS STORED IN THE GN_2 PURGED SHUTTLE CARGO BAY OF WHICH STRUCTURE TEMPERATURE IS 120 F.
2. SPACE HEAT LEAK ASSUMES THAT MODULE IS OUT OF THE CARGO BAY AND IS ORBITING WITH ITS BROADSIDE TO THE SUN.
3. TANK PENETRATION HEAT LEAKS ARE NOT INCLUDED.

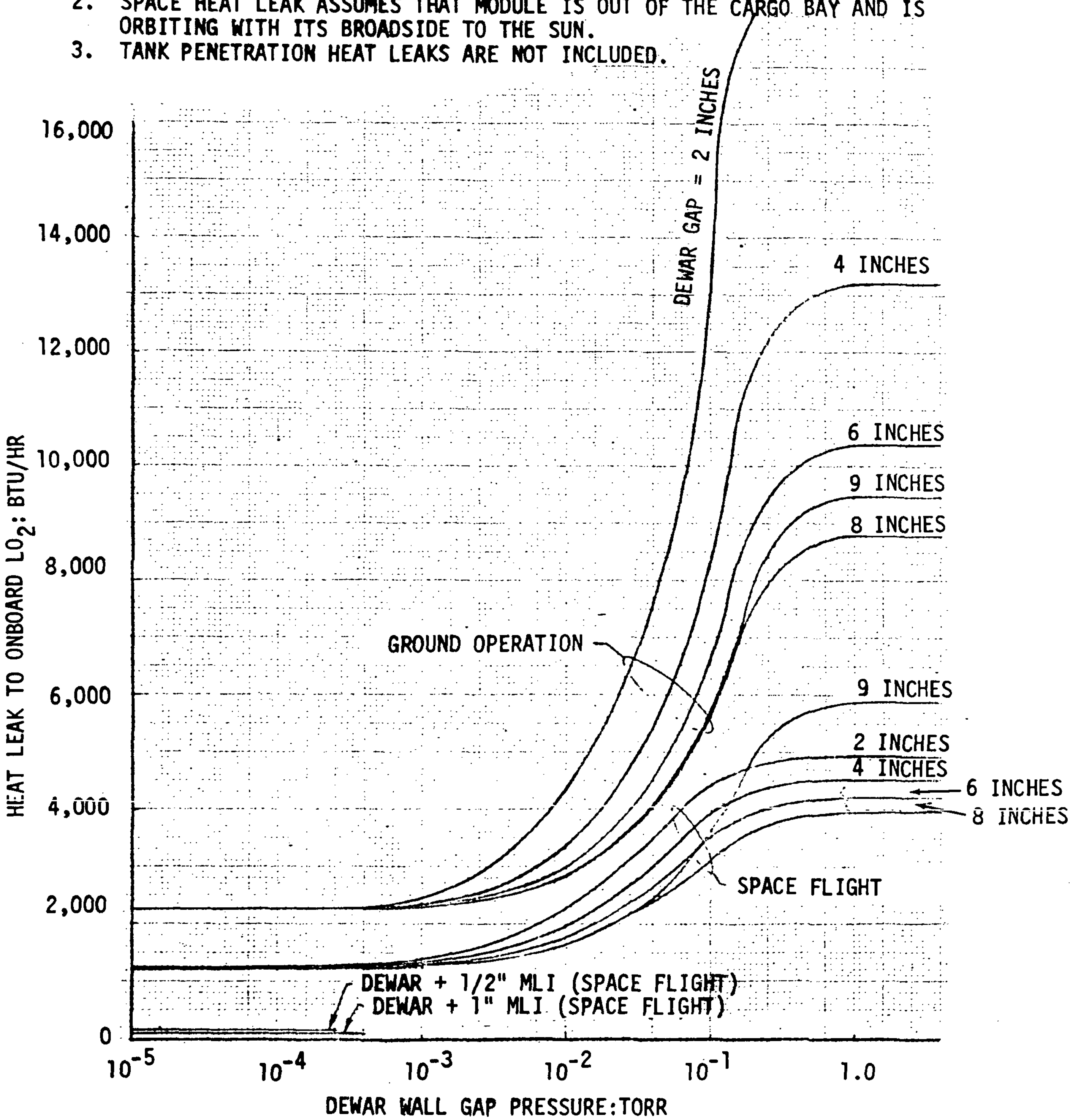


Figure 8.4-8 Heat Leaks to the Onboard LO_2 Through the Dewar Type Tank Wall

design can be adopted to compute the propellant heat leaks. Similar results calculated for the MLI under high vacuum were obtained for the MLI/dewar system. This is because the major difference in the heat leak calculations of the two systems is in the N term. For the MLI/dewar system, the N term should include one additional radiation shield to account for the vacuum jacket. The calculated heat leak rates for the LH_2 and LO_2 tanks are shown in Figures 8.4-7 and 8.4-8, respectively. Significant reduction in the heat leak rates can be realized with the addition of MLI blanket. The heat leak rates are about two orders of magnitude smaller than those for the dewar with no MLI.

8.4.2 Heat Leak Through Tank Penetrations

Heat leak through the tank penetrations does not have a real effect in selection of the propellant tank insulation system since the design of the tank penetrations can be identical for all evaluated insulation systems. Thus, approximately the same penetration heat leak rates can be expected. Nevertheless, the penetration heat leaks must be reduced to a minimum to achieve a highly effective cryogenic tank insulation system. A thermally poor penetration design can nullify the effect of using a high performance insulation system on the tank walls.

The major penetrations of the module propellant tanks are the (1) tank supports, (2) tank fill and drain lines, (3) vent lines, and (4) tank pressurization lines. The propellant transfer lines are designed with thermodynamic vents to cool the lines. Therefore, it was assumed that there would be no heat transfer between the transfer lines and tanks. To minimize the heat leaks by conduction through the tank penetrations, low thermal conductivity materials such as fiberglass epoxy and boron epoxy combined with graphite epoxy are used. The selected composite materials provide not only a low thermal conductivity, but also a relatively high structural strength and yet they are light in weight. The heat leak analyses on the tank fill and drain, vent, and pressurization lines were limited to estimation only since these tank line designs were not clearly defined at the time of the analyses.

8.4.2.1 Tank Supports

The LH_2 and LO_2 tanks of the logistic module are supported from the outer shell structure by tubular truss works. Each truss work is comprised of 48 tubular trusses which are assembled in a w form around the tank circumference as shown on Figure 8.3-1. The trusses are made up of reinforced fiberglass epoxy to reduce the heat leaks to the on-board propellants. The diameters of the trusses are 3/4 inch and 1 inch for the LH_2 and LO_2 tank, respectively. Other dimensions of the trusses are given in Figure 8.4-9. The hollow core of the tubular trusses are filled loosely with aluminized mylar sheets, similar to those used for the MLI, to reduce the heat leak by radiation through the hollow portion of the trusses.

The heat leak rates through the propellant tank support are presented in Figures 8.4-9 and 8.4-10 for the logistic module in a prelaunch hold environment and in a space environment, respectively. The heat leak rates for the prelaunch hold environment are given as a function of the shuttle orbiter cargo bay ambient temperatures. The cargo bay ambient temperature during prelaunch hold will be influenced by the cargo bay structure temperature, condition of GN_2 used for

NOTE
1. THE PROPELLANT TANKS ARE INSULATED WITH A 1/2 INCH OF SAM MULTILAYER INSULATION
2. DURING PRELAUNCH HOLD, THE SHUTTLE ORBITER CARGO BAY IS PURGED WITH GN_2 WHILE THE MODEL IS PURGED WITH HELIUM GAS
3. PREDOMINANT HEAT TRANSFER MODE OF THE PURGED AREAS IS ASSUMED NATURAL CONVECTION
4. THE CARGO BAY STRUCTURE TEMPERATURE IS ASSUMED 120F

	L	ℓ	DIA	t	NUMBER OF TRUSSES
LH_2 TANK SUPPORT	INCH 46	INCH 12	INCH 3/4	INCH .016	48
LO_2 TANK SUPPORT	26	12	1	.024	48

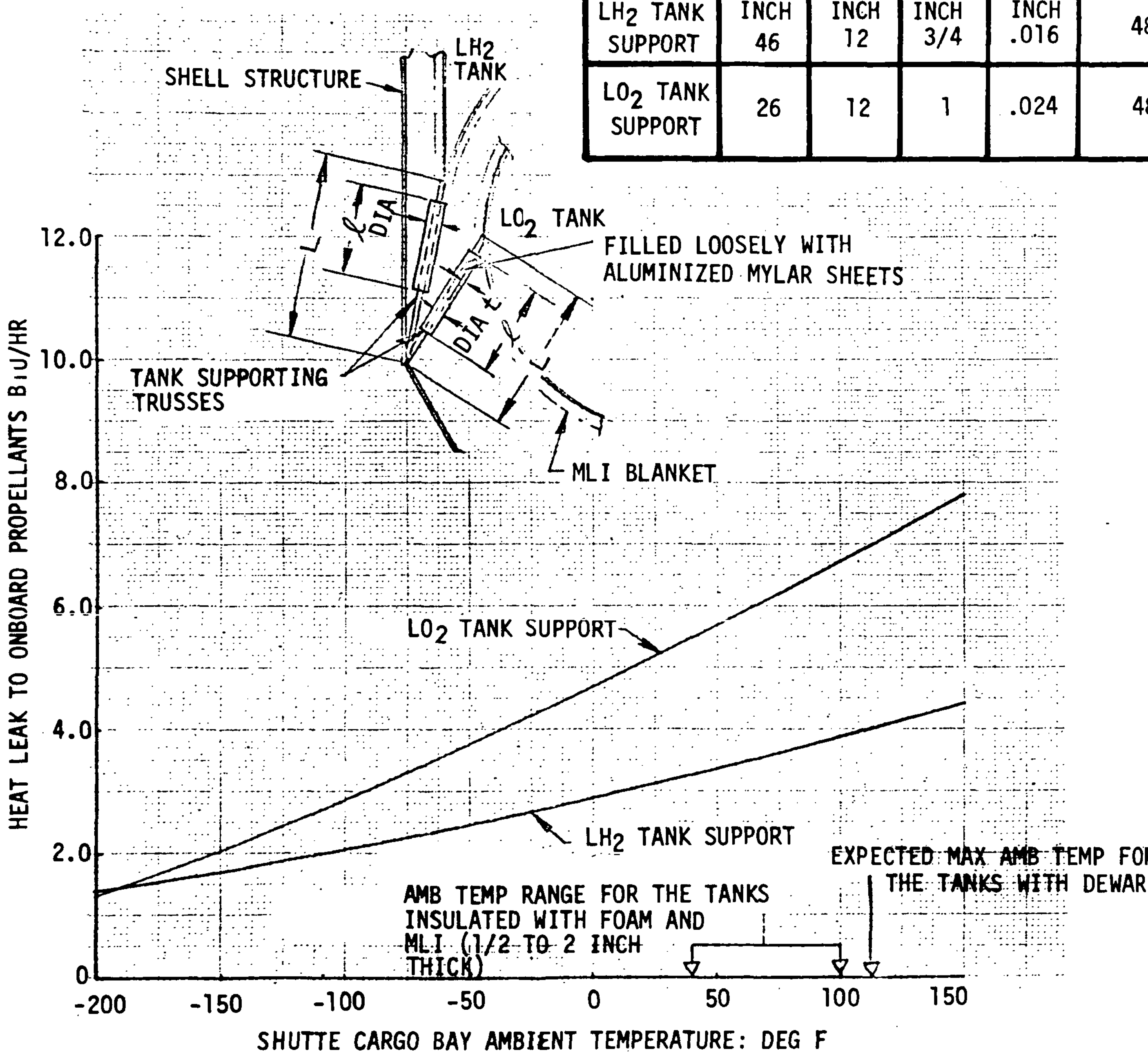


Figure 8.4-9 Heat Leaks to the Onboard Propellants Through the Tank Supports During Prelaunch Operation

NOTE
1. THE PROPELLANT TANKS ARE INSULATED WITH A 1/2 INCH OF SAM MULTILAYER INSULATION
2. THE SHELL STRUCTURE TEMPERATURE IS ASSUMED UNIFORM AROUND THE CIRCUMFERENCE

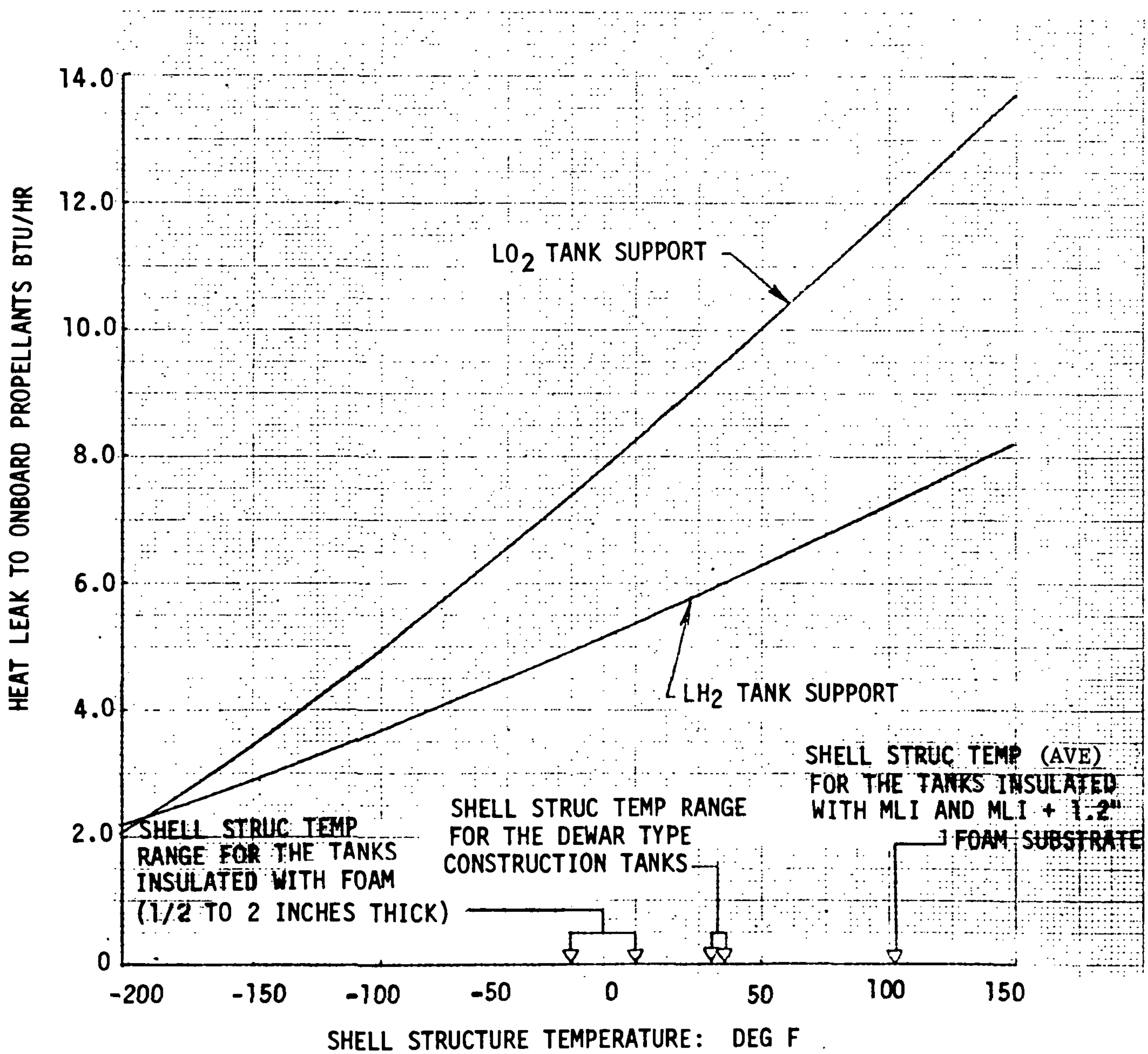


Figure 8.4-10 Heat Leaks to the Onboard Propellants Through the Tank Supports in Space Flight

the cargo bay purge, and condition of purge gas used for the module. A temperature of 120 F is assumed for the cargo bay structure. For the 120 F cargo bay structure, the cargo bay GN_2 ambient temperatures between 40 F and 100 F can be expected for the one-half to two inches of foam or MLI on the propellant tank. For the dewar with high vacuum, an ambient temperature as high as 113 F was calculated. For these ambient temperatures, Figure 8.4-9 shows that the heat leak rates through the LH_2 and LO_2 tank supports during prelaunch hold operations are 3.5 $\pm$0.5 Btu/hr and 6.0 $\pm$1.0 Btu/hr, respectively. The higher heat leak of the LO_2 tank support is attributed to a large cross sectional area of the trusses. It is more than two times greater than that of the LH_2 tank supporting trusses.

The space flight heat leaks are given in Figure 8.4-10 as a function of the shell structure temperatures. The shell structure temperature varies with solar angle. It also varies around the periphery of the shell structure when flying broadside to the sun. The shell structure average temperatures are between -26 F and 0 F for the one-half to two inches of foam, between 30 F and 36 F for the high-vacuum dewar, and about 103 F for the MLI with and without foam substrate. For these shell structure temperatures, the respective heat leak rates for the LH_2 and LO_2 tank supports are approximately 5 Btu/hr and 7.8 Btu/hr for the foam, 6 Btu/hr and 9.4 Btu/hr for the dewar; and 7.3 Btu/hr and 12 Btu/hr for the MLI and MLI-foam substrate systems, referring to Figure 8.4-10.

8.4.2.2 Propellant Tank Fill and Drain, Vent, and Pressurization Lines

The propellant tank fill and drain, vent, and pressurization lines are the principal lines necessary for the tank operation, but they constitute heat short paths to the stored cryogenic propellants. To reduce the heat conduction through these tank penetrations, thermal isolators are incorporated in the lines outboard but near as possible to the points where the lines enter the tank walls. For the fill and drain lines, the thermal isolators are located immediately outboard of the shutoff valves which are attached to the tank walls. The thermal isolators are made of 3 mil CRES foil reinforced with four layers of 0.01 inch glassfiber epoxy as shown in Figure 8.4-11. The lines and thermal isolators are insulated with 1/2-inch of MLI. The length of the thermal isolators was determined to be six inches long based on the thermal effectiveness of the thermal isolator in space and the problem associated with an installation of lengthy duct of such type.

The heat leak rates through the tank fill and drain, vent, and pressurization lines were made on the assumption that the lines are two-inch diameter aluminum alloy duct with six-inch long thermal isolator and the temperatures of the line ends at the tank walls are at or near the respective propellant temperatures. These assumptions were made because the tank service line designs were not clearly defined at the time of the analysis. Based on these assumptions, the calculated heat leak rates for each of the LH_2 and LO_2 tank service lines are approximately 2.5 and 1.5 Btu/hr, respectively, for the prelaunch hold and approximately 1.5 and 1.25 Btu/hr, respectively, for the space flight. Assuming that there are three service lines to each tank, the total respective heat leak rates are 7.5 and 4.5 Btu/hr for the prelaunch hold and 4.5 and 3.75 Btu/hr for the space flight.

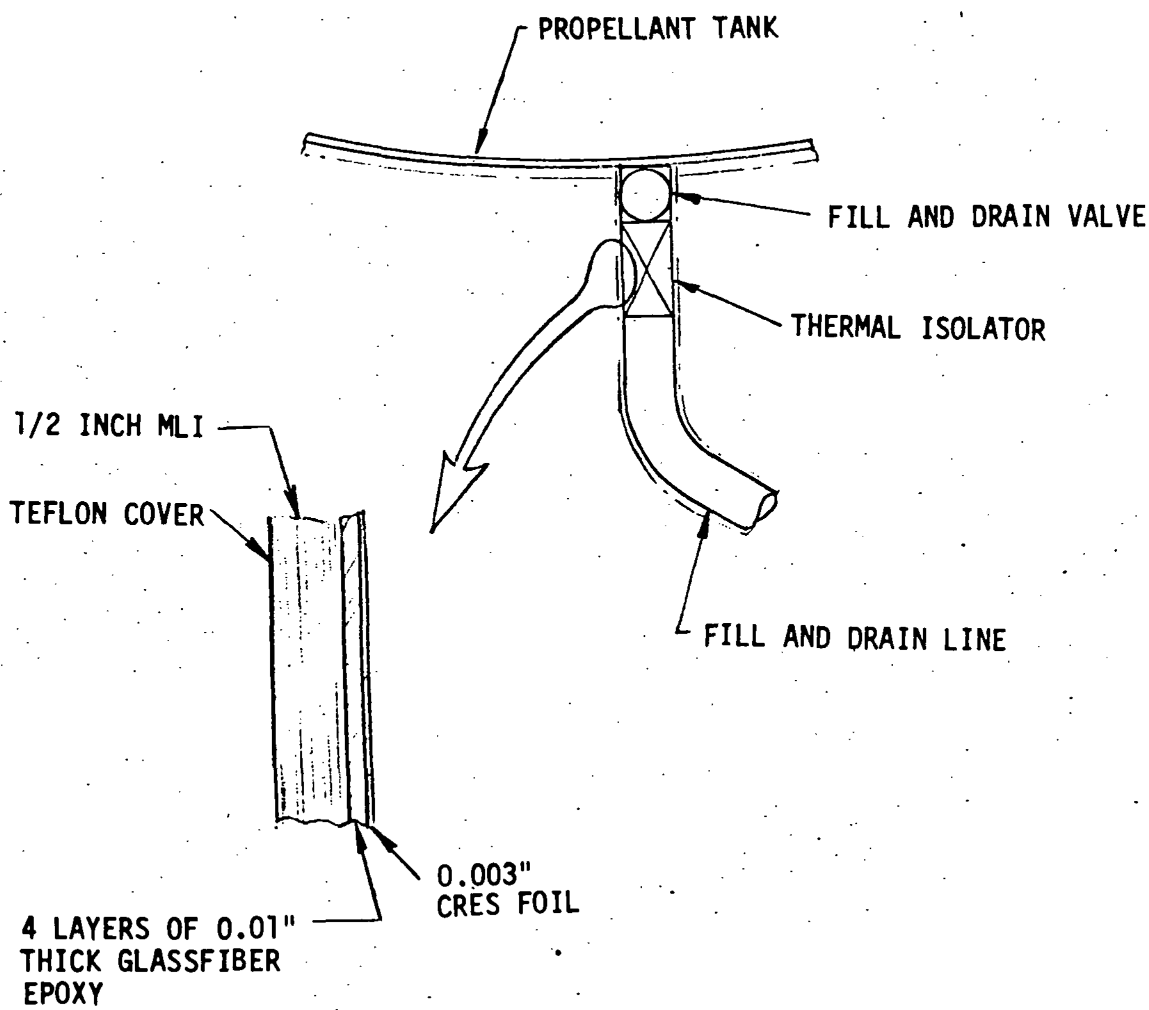


Figure 8.4-11. Typical Thermal Isolator Design of the Propellant Tank Service Lines (Fill and Drain Line Configuration)

8.5 COMPARISON OF THE CANDIDATE INSULATION SYSTEM OPERATIONAL WEIGHTS

A comparison of the insulation system operational weights was made since the operational weights are directly related to the insulation thermal performance and the insulation material requirement. The insulation thermal performance was expressed in terms of the amount of unusable propellant weight in pounds (sometimes referred to as an equivalent propellant boil-off weight) at the completion of the mission. In order to reduce the unusable propellant weight and the heat leak rates, the insulation thickness must increase, thus increasing the insulation system weight. Therefore, there is an optimum insulation thickness that would result in the delivery of maximum usable propellant.

Table 8.5-1 presents the comparison of the candidate insulation system operational weights after two days of logistic module operation. Two days is considered as the average propellant logistic mission time. The mission time in the calculation included three minutes of prelaunch hold. Referring to the table, MLI of one-inch thickness has the lightest operational weight of 805 pounds for two days of operation. This MLI operational weight is about three to eight times smaller than those of the dewar and foam systems.

Figure 8.5-1 presents the operational weight comparison as a function of the mission time. For ground operation and short-term space missions, foam presents the lowest operational weight. However, due to the extremely high heat leak of the foam for space operations, its operational weight becomes higher than those of the MLI and MLI plus 1/2-inch foam substrate systems after about three hours and higher than those of the dewar and dewar/MLI integrated system after about 18 hours of mission time. An operation weight comparison for various MLI thicknesses is also shown on the figure. A 1/2-inch MLI is lighter than the system using one-inch MLI when the mission time is less than 1-1/2 days. If the mission is longer than six days, a 1-1/2 inch MLI system presents less weight penalty. On the basis of two-day average propellant logistic timeline, a one-inch MLI system is the best selection for the propellant logistic module from the standpoint of the insulation system operational weight.

In computing the unusable propellant weights, the heat leak rates of the tank walls and tank penetrations presented in Section 8.4 were used. The unusable propellant weights were computed by dividing the accumulated heat leaks by the heat of vaporization. Heat of vaporization is 192 Btu/lb for LH_2 and 92 Btu/lb for LO_2. The system hardware weights were computed by using the following weight estimates:

Foam System

Foam = 2 lb/ft^3

MLI System

MLI = 1.75 lb/ft^3 including supporting pins and installation hardwares

Reentry purge system = 400 pounds including helium gas

Table 8.5-1 Candidate Insulation System Operational Weights After 2 Days of Operation

INSULATION SYSTEM	THICKNESS INCHES	*EQUIVALENT PROPELLANT BOILOFF RATE								ACCUMULATED UNUSABLE PROPELLANTS (LBS)				INSUL SYSTEM HARDWARE WEIGHT (LBS)	INSUL SYSTEM OPERATIONAL WEIGHT (LBS)	ORDER OF LESS WEIGHT PENALTY
		GROUND				SPACE										
		LH_2 TANK		LO_2 TANK		LH_2 TANK		LO_2 TANK								
		LBS/HR	%/HR	LBS/HR	%/HR	LBS/HR	%/DAY	LBS/HR	%/DAY	LH_2 TANK	LO_2 TANK	PENETRATIONS	TOTAL			
FOAM	1.0	195	2.1	119	0.2	73	18.8	47	2.2	3500	2250	8.6	5758.6	242	6000.6	8
	2.0	114	1.2	70	0.1	61	15.7	38	1.8	2950	1850	8.6	4808.6	484	5292.6	7
MLI (SAM)	0.5	541	5.8	293	0.6	2.9	0.7	1.9	0.09	192	114	11.0	317	506	823	2
	1.0	474	5.1	257	0.5	1.5	0.4	1.0	0.05	115	68	11.0	194	611	805	1
	1.5	411	4.4	228	0.4	1.1	0.3	0.7	0.03	86	51	11.0	148	719	867	3
MLI (SAM) PLUS FOAM SUBSTRATE	1.0 0.5	244	2.6	149	0.3	1.3	0.3	0.9	0.04	88	57	11.0	156	732	888	4
DEWAR	-	36	0.4	25	0.05	20	5.2	14	0.7	960	650	8.6	1618.6	2100	3718.6	6
DEWAR PLUS MLI (SAM)	- 0.5	5	0.05	3	0.006	2.9	0.7	1.9	0.09	135	87	11.0	233	2210	2443	5

NOTE

1. THE OPERATION TIME INCLUDES 3 MINUTES OF PRELAUNCH HOLD
2. THE ASSUMED PROPELLANT LOADING SCHEDULE IS 9,300 POUNDS OF LH_2 AND 50,900 POUNDS OF LO_2
3. * PROPELLANT BOILOFF RATES INCLUDE THE TANK PENETRATION HEAT LEAKS
4. THE MLI FINAL PRESSURE IN SPACE IS ASSUMED 3.1×10^{-5} TORR
5. THE DEWAR INTERSTITUAL PRESSURE IS ASSUMED LESS THAN 5×10^{-4} TORR

NOTE

1. THE MISSION TIME INCLUDES 3 MINUTES OF PRELAUNCH HOLD
2. THE MLI FINAL PRESSURE IN SPACE IS ASSUMED 3.1×10^{-5} TORR
3. THE DEWAR PRESSURE IS LESS THAN 5×10^{-4} TORR

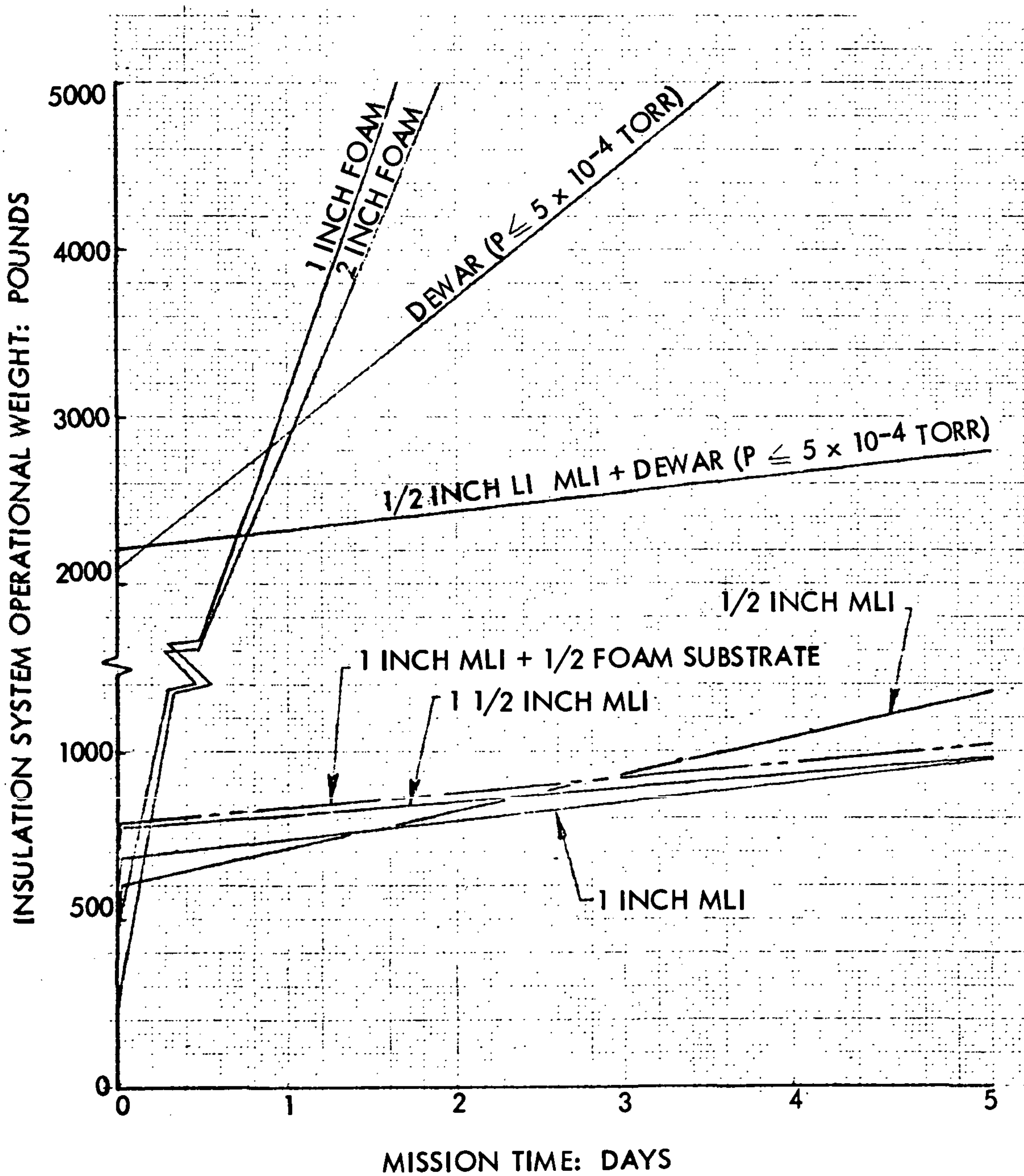


Figure 8.5-1 Comparison of the Candidate Insulation System Operational Weights

Dewar System

Vacuum jacket = 2100 pounds including vacuum lines and valves

The 2100 pounds of dewar hardware weight may be reduced if the logistic module shell structure is used as the pressure carrying vessel. However, the weight saving is not expected to be great since the shell structure must be strengthened.

9.0 PROPELLANT GAUGING

9.1 INTRODUCTION

This section presents the results of a study to select a logistic tank gauging system concept for providing propellant quantity measurements for the baseline logistic tank, and to determine what impact the mode of propellant transfer has with respect to propellant losses as a result of the receiver tank propellant gauging system inaccuracies. The gauging system should provide reasonably accurate and reliable liquid propellant quantity measurements while subjected to acceleration environments ranging from as high as 3 g during the launch abort mode, to as low as 10^{-5} g throughout the course of on-orbit propellant transfer.

This study consisted of developing the functional requirements of the logistic tank gauging system, reviewing current and advanced "state-of-the-art" propellant gauging technology and its application to the logistic tank, and selecting a gauging system concept for the baseline logistic tank design. In addition, propellant losses associated with the receiver vehicle propellant gauging inaccuracies as a result of conditions imposed by rotational and linear acceleration propellant transfer modes was investigated. Current and advanced state-of-the-art propellant gauging concepts applicable to zero and positive g environmental conditions were considered.

The major problem associated with propellant gauging is providing accurate mass measurements during on-orbit operations under low g acceleration conditions. Propellant gauging under positive acceleration conditions greater than 10^{-3} g's are generally simplified because of the presence of an acceleration vector which allows the prediction of the propellant location and the liquid surface shape within the confines of the tank. Tanked propellants under these conditions are capable of being measured by discrete point sensing devices such as capacitance probes, optical probes, and hot wire (resistance measurement) sensors. From these measurements the propellant mass can be determined. For acceleration forces in the range of 10^{-4} to 10^{-5} g the propellant settling forces predominate but large errors can occur using existing liquid level sensing devices because of liquid surface distortions at the sensing elements due to meniscus effects. Additional errors in mass measurement systems on a liquid level/volume relationships can occur at low g due to surface distortion at the tank walls. In very low, or zero g environments, the propellant becomes randomly oriented and the use of liquid level sensing devices becomes impractical.

Several propellant gauging techniques capable of mass quantity measurements in any acceleration environment, including zero g have been under investigation for the past few years. The major advantage of these systems is that continuous mass measurements of tanked propellants are available without a propellant loss penalty associated with liquid settling. Techniques have been developed and demonstrated that are capable of measuring random oriented propellant; however, the currently obtainable accuracies of these systems are low, ranging from two to ten percent of a fully loaded tank. This is far below the gauging accuracies that can be obtained from liquid level sensing devices, even in low acceleration environments (10^{-3} to 10^{-5} g).

9.2 SUMMARY OF RESULTS

A propellant gauging system concept which utilizes vertical resistance wire sensing elements and is capable of providing loading accuracies of approximately 0.4 and 0.7 percent of a full tank load for LO_2 and LH_2 tanks, respectively, has been selected for the logistic tank propellant gauging system. The system provides the capability of continuous gauging over the entire range of loadings required to support all logistic tank missions. The system also provides liquid level measurement during launch and on-orbit propellant transfer operations for satisfying requirements of propellant dumping, initiating, and monitoring transfer flow control, and providing propellant depletion indications. A summary of the trade study is presented in Figure 9.2-1.

The selection of the vertical resistance wire concept over the more conventional gauging systems (i.e., capacitance probe and point sensors) was based on its relative simplicity, self-calibration capability, good gauging accuracy, and dual capability of satisfying propellant gauging requirements for both ground and on-orbit operations. For these reasons, this concept is recommended for propellant gauging in the tug, CIS, and RNS stages as well as the logistic tank. Although the selected system is conceptual at this time, it is felt that the concept provides the most feasible means of satisfying the propellant logistic system propellant gauging requirements and it is recommended that work be undertaken for the development of this system.

In concept, zero g gauging systems are ideal for on-orbit application, but their presistent hardware problems do not permit recommendation of these systems at this time.

A comparison of propellang gauging accuracy in a rotational transfer mode and a linear acceleration transfer mode indicated that better gauging could be obtained under linear acceleration conditions. By equating individual tank (LO_2 and LH_2) gauging accuracy to total tanked propellant gauging accuracy, under linear acceleration the total accuracy was 0.69 percent while under rotational acceleration, the accuracy was 0.81 percent. These percentages represent approximately 6300 and 7400 pounds of propellant above the nominal loading requirements for a CIS supportive mission for linear and rotational transfer modes, respectively.

9.3 REQUIREMENTS

The functional requirements of a propellant gauging system in support of in-space propellant logistics must consider both the receiver tank and the logistic tank requirements and their interrelationship. During the final stages of an orbital propellant transfer operation, the most critical propellant gauging requirements are imposed on the receiver vehicle system since the receiver vehicle flight performance is predicated on an optimized propellant loading. Although the logistic tank gauging system could provide an indication of the receiver tank propellant quantity by measuring the amount of propellant transferred, a higher confidence would be placed in receiver tank gauging since this would account for boil-off vented propellant, or other miscellaneous losses.

The four major functions of the logistic tank gauging system are in support of: (1) ground loading, (2) measuring propellant dumped for emergency reentry, (3) initiating and controlling flow throttling during propellant transfer, and (4) propellant depletion indications.

Table 9.2-1 Propellant Gauging System Trade Table

FUNCTIONAL AND TECHNICAL REQUIREMENTS	ALTERNATE APPROACHES			SELECTION
GROUND LOADING DUMP ZERO g FLOW THROTTLING ZERO g TRANSFER TERMINATION	DISCRETE POINT SENSORS ①	CAPACITANCE PROBE ②	VERTICAL RESISTANCE WIRE ③	ACCURACY ON GROUND ① ③ ② ACCURACY IN ORBIT ③ ② ① SIMPLICITY ③ ② ① USABLE IN ORBIT ③ ② ①
	PROS VERY ACCURATE AT ONE g	PROS ACCURATE AT ONE g CONTINUOUS GAGING AVAILABLE FOR ENTIRE TANK LENGTH	PROS SELF CALIBRATING INSENSITIVE TO ALL SIGNIGICANT SOURCES OF ELECTRICAL SENSING ERRORS	
CRITERIA ACCURACY ON GROUND ACCURACY ON-ORBIT SIMPLICITY BUT SATISFIES TASK USABLE IN ORBIT	CONS NOT ACCURATE AT LOW g SUBJECT TO SLOSH INACCURACIES ONLY ACCURATE AT SPECIFIC POINTS	CONS WILL NOT FUNCTION ACCURATELY AT LOW g DUE TO CAPILLARY ACTION SENSITIVE TO EFFECTS OF STRAY CAPACITANCE AND RESISTANCE NO SELF CALIBRATION METHOD	CONS LOW LEVEL CURRENT CAUSES SMALL RESISTANCE CHANGE WHICH CAN BE LOST IN STRAY RESISTANCE NOT CURRENTLY PROVEN	SELECTED CONCEPT VERTICAL RESISTANCE WIRE PRIMARILY SELECTED BECAUSE OF USE IN ORBIT

The ground loading gauging requirements are the most stringent for the tug supportive mission because it must provide flexibility of loading over an anticipated range of 15 percent to 100 percent of full load. The logistic tank loading for CIS and RNS supportive mission require only fully loaded tanks.

The three remaining requirements are in-flight liquid level sensing requirements. In-flight sensing is required to verify that sufficient propellant is dumped to satisfy maximum allowable shuttle payload for a launch abort, liquid level sensors provide a signal to initiate propellant flow throttling to reduce logistic tank residuals during propellant transfer, and liquid level sensing is required to indicate propellant logistic tank depletion and terminate propellant transfer.

Gauging accuracy is the most critical during ground loading. For the tug supportive missions, the logistic tank shares the shuttle cargo bay with the tug payload and the tank loading becomes critical so shuttle payload limits will not be exceeded. Therefore, the best obtainable gauging system accuracy within the state-of-the-art technology available should be used. Current state-of-the-art discrete point propellant gauging systems can exhibit ground loading accuracies of 0.4 and 0.6 percent of a full tank load for LO_2 and LH_2 tanks, respectively.

Where continuous loading measurements are required to satisfy propellant tank off-loading, a capacitance probe covering the entire length of the tank can provide loading accuracies of 0.5 and 0.7 percent (of full load) for LO_2 and LH_2 tanks, respectively.

Logistic tank propellant liquid level measurements (for propellant dump and transfer) does not require particularly accurate indications. Liquid level measurements of one percent can, in most cases, be accepted without impacting the propellant transfer operations.

9.4 CANDIDATE CONCEPTS

In order to determine the concepts from which the logistic tank propellant gauging system can be developed, a review of the current state-of-the-art liquid gauging technology was performed. A summary of the liquid detection systems specifying accuracy, stability, power requirements, and operational gravity requirements are presented in Table 9.4-1. Two gauging system concepts shown in Table 9.4-1 were chosen for the logistic tank gauging system trade studies; namely, discrete point sensors and capacitance probes. A third concept was also considered and is not shown in the table since it is conceptual and little development has been done on it to date. This system consists of vertical heated wire sensing elements that provide an analog of the liquid level based on the resistance characteristics.

Although consideration was given to zero g propellant gauging systems, in reviewing their "state of the art", two factors are evident. One is that acceptable accuracies are not attainable by any of the present zero g gauging concepts (acoustic compliance, acoustic resonance, RF resonance, and nucleonic) except possibly in the last two cases through the use of an extremely large number of sensors. The other factor is that the zero g hardware has not demonstrated high reliability required for logistic tank or receiver vehicle application. The most promising zero g concepts appears to be a nucleonic type being developed by the Air Force Rocket Propulsion Laboratory (AFRPL) which uses solid state, variable output radiation sources, and solid state, cadmium telluride detectors. The status of this and other zero g systems should be reviewed at design go-ahead.

Table 9.4-1 Liquid Detection Systems
Design and Performance Features

MANUFACTURER	ACCURACY	STABILITY	POWER REQUIREMENTS	GRAVITY REQ'M TS	REMARKS
ACOUSTICA ASSOCIATES, INC.	±0.50 INCH	200 HOURS	10 WATTS	POSITIVE G	ULTRASONIC SENSOR SYSTEM
ACOUSTICA ASSOCIATES, INC.	±0.1 INCH	1000 HOURS	7 WATTS/UNIT LH_2 3.5 W/UNIT LOX	POSITIVE G	DISCRETE LEVEL SENSOR SYSTEM
ACOUSTICA ASSOCIATES, INC.	+ 1 TO + 2% FULL TO EMPTY	200 HOURS BETWEEN CALIBRATION	35 WATTS	POSITIVE OR ZERO G	INFRASONIC SENSOR SYSTEM
BENDIX CORP.	± 10 TO ± 1% FULL TO EMPTY	200 HOURS BETWEEN CALIBRATION	30 WATTS	POSITIVE OR ZERO G	RADIO FREQUENCY SENSOR SYSTEM
CONRAC CORP	+1.5 TO 0.5% FULL TO EMPTY	200 HOURS BETWEEN CALIBRATION	27 WATTS	POSITIVE OR ZERO G	RADIATION SENSOR SYSTEM
INDUSTRIAL NUCLEONICS CORP.	± 2.0%	200 HOURS BETWEEN CALIBRATION	40 WATTS	POSITIVE OR ZERO G	RADIATION SENSOR SYSTEM
SIMMONDS PRECISION CORP.	± 0.5%	200 HOURS	30 WATTS	POSITIVE G	COAXIAL CYLINDER CAPACITANCE SYSTEM
TRANS-SONICS CORP.	± 2.0%	200 HOURS BETWEEN CALIBRATION	50 WATTS	POSITIVE OR ZERO G	PERIPHERAL CAPACITANCE SYSTEM
TRW, INC.	+ 1 TO + 2% FULL TO EMPTY	200 HOURS	7 WATTS	POSITIVE OR ZERO G	INFRASONIC SENSOR SYSTEM
TYCO LABORATORIES	± 2.0%	200 HOURS	7 WATTS	POSITIVE OR ZERO G	RADIATION SENSOR SYSTEM
UNITED CONTROLS	± .06 INCH	1000 HOURS	7 WATTS/UNIT	POSITIVE G	DISCRETE LEVEL SENSOR SYSTEM

9.5 DISCUSSION OF CANDIDATE CONCEPTS

9.5.1 Discrete Level Sensors

The discrete level sensor system uses a resistance type sensing element and an electronic control module. The wire sensing element is powered by a constant current to maintain its temperature at a level above the surrounding environment. Since the resistance of the wire changes as a function of temperature, any change in the medium in contact with the wire causes a relatively large (discrete) change in resistance which is sensed by the control module.

The sensing wire is usually positioned horizontally to the surface of the liquid so that the entire wire is immersed at one time. The sensor system is very accurate in a one g environment being capable of measuring liquid heights to within ± 0.05 inches. At low g a liquid meniscus keeps the wire wet even though the liquid surface is below the sensors. At g levels of 10^{-4}, the meniscus height is estimated at 11 inches and at 10^{-5} g, a height of 20 inches. Even if heating elements were provided to the element housing, wicking action can keep the element wet and give large liquid level errors which are not predictable due to vibration and sloshing in the tank. Also at low g levels sloshing frequency are very slow and damping almost nil. Point sensor outputs would be difficult to interpret under such conditions.

9.5.2 Capacitance Probes

The capacitance probe sensing system consists of two coaxial capacitance tubes extending the length of the tank. The capacitance between the two cylinders changes as a function of the propellant within the tank. An electronic control package inserts the capacitance into a bridge circuit which nulls itself electrically by generating a voltage that represents an amount of propellant.

The capacitance probe is an accurate system for propellant mass measurements in a one g (or greater) environment. Loading accuracies of 0.5 and 0.7 percent of full load are capable for LO_2 and LH_2, respectively. The main advantage of the capacitance probe is that it provides continuous gauging for the entire length of the tank.

In a low g environment the capacitance system will not function properly as a gauging system because the capillary action of the liquid would fill the coaxial tubes and give an erroneous output. As an example, the capillary rise in a two-inch diameter tube is approximately 86 feet at 10^{-5} g. A capacitance system which uses two parallel, small diameter wires to form the electrodes of a probe, providing a continuous analogy output over its entire length, has also been considered. However, since the electrical output of such a probe is several orders of magnitude less than conventional probes, it would be extremely sensitive to effects of stray capacitance and stray resistance of leadwires and connectors. No satisfactory method of self-calibration to eliminate these effects is currently available.

9.5.3 Vertical Resistance Wires

A vertical resistance wire sensing concept has been investigated for low g propellant gauging. The vertical resistance wire concept consists of wire elements (one to three feet long) strung vertically between two end supports. A 200 milliamp current is applied and the resistance of the wire is a direct analog of the liquid level.

A self-calibrating feature is incorporated by alternately applying a low (5 ma) current to the sensing wire, which produces no appreciable self heating, but serves to calibrate the resistance of the entire measurement loop.

It is estimated that an analog heated wire gauging system can be made insensitive to all significant sources of electrical sensing errors. Liquid level accuracy to within two precent of the sensing element length is anticipated. The advantage of the resistance wire system is that in a low g environment, the liquid meniscus height on the wire is greatly reduced. Calculation shows that a vertical wire 0.010 inch in diameter will support a liquid hydrogen meniscus of 3 inches at 10^{-5} g (as compared to 20 inches for a horizontally positioned wire). Corresponding heights of the liquid oxygen meniscus is expected.

Although no meaningful development work has been accomplished, the concept appears to be the more practical method of overcoming the capillary problems associated with low g gauging systems. Therefore, it is recommended that development of this concept be undertaken.

9.6 GAUGING SYSTEM SELECTION

Two primary considerations led to the selection of the logistic tank gauging system. The first consisted of obtaining the best possible gauging accuracy for both ground fill and on-orbit operations that is within the current or near future "state-of-the-art" gauging technology. The second consideration was to keep the gauging system as simple as possible, preferably choosing a single system that would fulfill both ground fill and on-orbit gauging requirements to within acceptable gauging accuracies. Other factors such as hardware and development costs and system reliability were also considered; however, it was felt that on a comparison basis, these factors would not influence the system selection significantly. From the three candidate system concepts evaluated the vertical resistance wire concept was selected for the logistic tank.

Ground loading requirements can be satisfied by any of the three concepts. A capacitance probe running the entire length of the tank can provide mass loading accuracies of 0.55 percent and 0.65 percent of the full load for LO_2 and LH_2 respectively, as demonstrated by the S-II gauging system. The capacitance probe also provides tank off-loading capabilities to within the approximate full load accuracies. Normally discrete point sensors are also used in conjunction with the capacitance probe for calibration purposes. The main disadvantage of this type of system is that in a low g environment, the capillary action of the fluids fills the coaxial tubes which results in erroneous outputs.

The conventional horizontal oriented (with respect to the surface) discrete point sensors provide accurate liquid level measurements in one g or higher environments such as ground loading. Accuracy of measurements using discrete sensors are slightly better than the capacitance probe (0.4 for LO_2, and 0.6 for LH_2) if the location of the sensor corresponds to the tank loading requirements. However, in cases where partial tank loading is required, such as the logistic tank, either many sensing elements have to be installed or provisions made for relocating tank loading sensors to correspond to the specific mission loading requirements. In low g (i.e., 10^{-5} g) environments that can exist during on-orbit operations a conventional sensor can suspend a meniscus as high as 20 inches or more, above the surface of the liquid. Even if heating elements are provided on the housing, wicking action and the effects of sloshing will keep the sensors wet and result in large unpredictable gauging errors.

The vertical sensing wire gauging concept can provide ground loading liquid level measurement accuracies comparable to conventional discrete point sensors, and from liquid level indication, tank mass can be calculated to higher accuracy than can be obtained with the capacitance probe. For on-orbit gauging under acceleration environments ranging as low as 10^{-5} g the estimated accuracy is under one percent of full tank loading using linear acceleration propellant positioning.

The vertical wire concept essentially takes a conventional heat wire, increases its length and strings it vertically between two end supports. The use of longer wires requires more sophisticated circuitry than the conventional sensor, to achieve good level accuracy. Because of the low current (200 ma) required to avoid short circuit arc energies below the ignition levels for LO_2 and O_2/H_2 mixtures, resistance change is small and if not properly designed, stray resistances can overshadow signal changes. To overcome these problems, a self-calibrating technique is used that provides two levels of current, one which produces no appreciable self heating but serves to calibrate the background resistance, and the other a higher current ($\sim$200 ma) which produces self heating and increases the resistance in the wire.

It is assumed that any increase of resistance by the signal conditioner is due entirely to the sensing wire since the resistance and self heating over the rest of the loop is very low. The delta resistance is used as a measure of the liquid level at the wire since the wire exposed to the liquid will heat very little due to the high heat transfer rate of nucleate boiling while the wires exposed to the vapor will heat several degrees. Iron-nickel alloys are available for wires which have both high resistance and appreciable resistance /temperature coefficient, coupled with good mechanical strength.

The installation of the vertical resistance wire would be in three-foot segments, each overlapping and covering the entire tank length. The vertical resistance wire installed in this manner provides the off-loading capabilities of a capacitance probe, and can also be used as either a mass indicator or liquid level sensor for propellant dump and on-orbit transfer flow control.

An analysis was performed to estimate logistic tank loading accuracies during ground and in-flight loading using the vertical resistance wire gauging system. For the analysis, the magnitude of individual sources of error resulting from a low g environment were estimated for those errors that contribute to total system error. Other source errors (density, location, etc.) were obtained from typical S-II gauging system accuracies. The error source and the value assigned are presented in Table 9.6-1. Total system accuracy is based on the root-sum-square of the individual source errors. The electronics error source is in reality a heat transfer calibration error. A value of two percent of the sensor length (3 feet) was used for this error. The errors for a capacitance probe system under similar ground loading conditions are shown for comparison.

Loading accuracies for on-orbit propellant transfer using the vertical resistance wire system for mass measurements was analyzed for conditions associated with gauging under rotational and linear acceleration transfer modes. For study purposes, a separate rotational transfer mode and separate linear acceleration mode for a CIS loading was assumed. The estimated gauging error sources and predicted gauging accuracy under each transfer mode are shown in Table 9.6-2.

During the low g environment of on-orbit operations, two source of gauging errors develop which are insignificant under a one g environment. These errors are liquid level indication errors resulting from the wire fluid meniscus and computation errors in propellant mass due to the uncertainty of the sidewall fluid meniscus. It was assumed that in a rotational mode of transfer, the hydrogen tank liquid propellant would be divided into two masses, located at opposite ends of the tank. Therefore, error sources become higher by a factor of two, when compared to corresponding errors for linear acceleration gauging. Meniscus errors are reduced in rotational transfer since the g level exerted on the liquid surfaces are two orders of magnitude greater than linear acceleration transfer.

The uncertainty of the interface curvature, a function of the location of gravity and rate of rotation, results in an additional error in predicting the total mass of the tanked propellants. This uncertainty would undoubtedly become quite large in receiver tanks that are fully loaded due to errors in predicting the ullage bubble volume, shape, and location.

A representative installation design for the vertical resistance wire gauging system element is presented in Figure 9.6-1. The configuration of the upper support arm helps prevent the meniscus from wetting the wire during a low g environment; an additional anti-wetting provision is incorporated by providing a copper lead wire to the upper position of the resistance wire. The copper wire offers little resistance and if wetted will not affect the output of the sensing element.

Table 9.6-1 Propellant Ground Loading Accuracies

ERROR SOURCE	VERTICAL RESISTANCE WIRE (%)		CAPACITANCE PROBE (%)	
	LO_2	LH_2	LO_2	LH_2
[1]Electronics	0.3	0.3	.4	.4
Sensor Location	0.02	0.02	.02	.02
Tank Calibration	0.25	0.25	.25	.25
Density Variation	0.15	0.35	0	0
Bubble Effects	0.20	0.4	.2	.4
GSE Control Band			.2	.2
Slosh Averaging	0.10	0.05	0	0
RSS Total Accuracy	.47	.66	.55	.65

1 Vertical resistance wire electronic effect are the result of heat transfer calibration errors.

Table 9.6-2 On-Orbit Propellant Gauging Accuracy
(Vertical Resistance Wire)

ERROR SOURCE	LINEAR ACCELERATION (10^{-5}g's)		ROTATIONAL ACCELERATION (10^{-3}g's)	
	LO_2	LH_2	LO_2	LH_2*
Heat Trans. Calib.	0.3%	0.3%	0.3%	0.6%
Sensor Location	0.02	0.02	0.02	0.04
Tank Calibration	0.25	0.25	0.25	0.5
Density Variation	0.1	0.35	0.1	0.7
Bubble Effects	0.05	0.1	0.05	0.2
Slosh Averaging	0.5	0.5	0.5	0.7
Wire Meniscus	0.2	0.2	0.09	0.22
Sidewall Meniscus	0.1	0.1		
Ullage Bubble and Surface Configuration Uncertainty	0	0	.3	.6
RSS Total Accuracy	.682%	.765%	.716%	1.43%

*Assumes that the LH_2 ullage bubble is located in the bulk liquid.

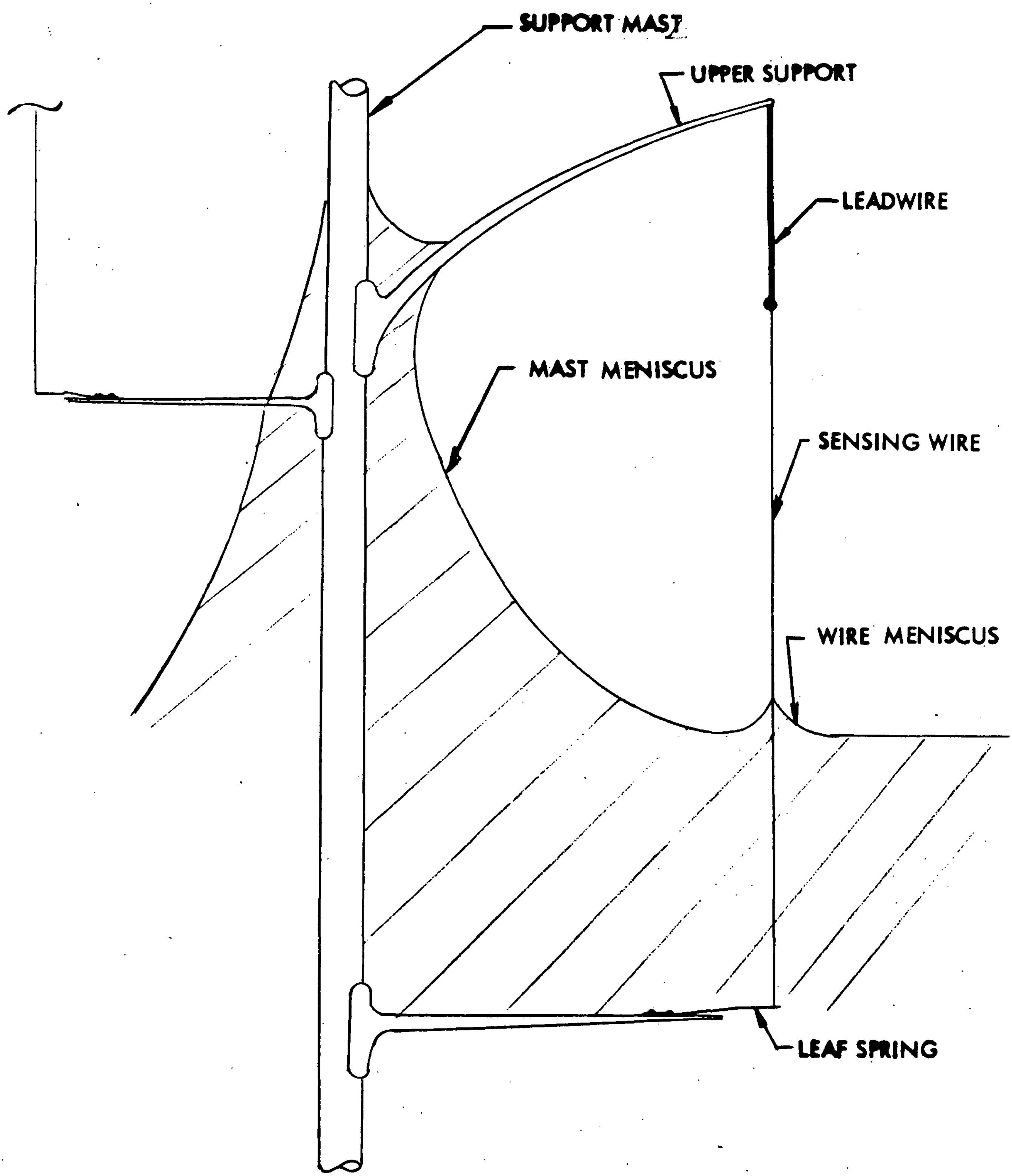


Figure 9.6-1 **Vertical Resistance Wire Gauging System Configuration**

10.0 SLUSH HYDROGEN

10.1 INTRODUCTION

One of the considerations during the study was the possibility of delivering slush hydrogen instead of liquid hydrogen to the space-based vehicles. Since the hydrogen boiloff loss constitutes one of the major propellant losses during space storage, the use of slush hydrogen can reduce this loss by increasing the amount of heat absorbed by the bulk propellant before it reaches the saturation temperature. Slush hydrogen (SH_2) is a mixture of small solid hydrogen particles and liquid hydrogen which can be transferred much like a liquid. The mixture offers the advantage of substantially increasing the bulk propellant heat capacity which results in a potential reduction in boiloff losses. Additional advantages of slush hydrogen are an increase in bulk density, which suggests a smaller storage tank, and lower tank storage pressures resulting in reduced pressurization gas requirements. Consequently, the literature on slush hydrogen was reviewed and evaluated for potential application to the propellant logistic program.

10.2 SUMMARY OF RESULTS

The primary advantage in using slush hydrogen is to reduce liquid hydrogen losses due to boiloff during in-orbit storage. This advantage is most significant where a large quantity of liquid hydrogen has to be stored for extended periods prior to its use, such as is the case for missions in which the CIS or RNS stages are used. Where relatively small quantities of liquid hydrogen are used, such as with the tug missions, slush hydrogen has less effect on hydrogen losses. This is because the major portion of the hydrogen is used shortly after completion of tanking operations. Since the quantity remaining for orbit maintenance is small, the additional heat capacity gained by use of slush hydrogen has little effect on boiloff losses.

In the case of the CIS missions, an elapsed time of six months is required from the start of tanking operations to mission completion. During this time period the hydrogen boiloff losses are approximately 65,700 pounds if the liquid hydrogen is stored at its saturated temperature. The use of 50 percent slush hydrogen can provide a potential reduction in these boiloff losses of 31,500 pounds. In terms of the total program (Level D), the hydrogen boiloff losses can be reduced 315,000 pounds which equates to a program cost savings of 56 million dollars.

The hydrogen boiloff losses for the RNS missions can be virtually eliminated by using slush hydrogen because the additional heat capacity gained more than offsets the total predicted heating loads. In terms of Program Level D, this means that 265,000 pounds of hydrogen boiloff losses are eliminated for a program cost saving of 62 million dollars.

For the CIS or the RNS stages, the potential cost savings are significant. However, the feasibility as applied to these program depends on the cost of developing slush hydrogen technology. It is not considered likely that hydrogen slush technology and the equipment and vehicle configurations required for its use could be developed for either the 56M or the 62M potential cost savings indicated above. Therefore, the use of hydrogen slush has not been adopted as a recommended concept in the ISPLS study. Continued research in hydrogen slush technology is recommended because of special applications which may exist for its use.

Slush hydrogen technology has not advanced far enough to permit application to the space logistics current baseline selections. Production, storage, and handling has not been extended beyond small-scale laboratory testing. However, current research literature states definitely that none of the experimental results in the investigation of slush hydrogen excludes its potential as a propellant for space vehicles. Additionally, small scale experimental results show that 50 percent slush hydrogen can be handled, transferred, and stored much like liquid hydrogen in many respects, making it very attractive as a space propellant. Fifty percent slush hydrogen offers an 18 percent maximum potential reduction in boiloff losses. Slush hydrogen must be stirred to prevent agglomeration, which is a disadvantage compared to liquid hydrogen.

The one basic difference between normal boiling point liquid hydrogen and slush hydrogen is the vapor pressure. Slush hydrogen can be stored at triple point pressure (approximately 1 psia). Helium is required in the ullage for storage pressures greater than triple point pressure on the ground. Atmosphere leakage into the ullage and tank collapse forces can be a problem for storage pressures lower than atmospheric pressure. However, the low pressure storage capability in orbit may be a structural advantage. Helium can be used to maintain any desired storage pressure above the triple point pressure.

Some of the indeterminants, which should be investigated in detail prior to the application of slush hydrogen to space vehicles are:

a. Compatibility with capillary devices

b. Instrumentation and measurements

c. Critical flow velocity as a function of pipe diameter

d. Temperature stratification in deep tanks

e. Percolation in dead-ended lines

f. Heat flux for pipe flow

g. Aging process

h. Cost of production

10.3 MANUFACTURING

Of the several processes that have been tried for the production of slush hydrogen, the freeze-thaw technique applied to the straight (direct) vacuum method has been most successful. As explained in Reference 10.0-1, this process appears to be satisfactory and efficient. The procedure starts with a tank of liquid hydrogen at normal boiling point temperature. The ullage is evacuated, causing an increase in liquid boiloff rate and a reduction in liquid hydrogen temperature until a soft layer of solid hydrogen is formed at the liquid surface. Then the ullage vacuum is released allowing the solid layer to melt into slush, which sinks to the tank bottom because it is more dense than the liquid. Successive cycles of ullage vacuum control produces successive layers of slush until the mixture in the tank eventually reaches the surface at an approximate 0.5 solid fraction. Some kind of stirring apparatus at the tank bottom imparts a gentle circulation to the solid-liquid mixture thus preventing an agglomeration.

The ullage evacuation flow rate must be controlled in the vicinity of one cubic meter per second per square meter of liquid surface for an efficient operation. If the rate is too fast, violent boiling causes liquid droplets to be carried along with the saturated vapor thus increasing the hydrogen mass loss. If the rate is too slow, the layer of solid hydrogen formed at the surface is too hard to be transformed into slush by ullage pressure control alone.

A schematic drawing of a slush generating system is presented on Figure 10.3-1. A vacuum jacket will be needed to minimize the boiloff. High pressure helium (or a pump) can be used for expulsion.

The evacuation manufacturing of slush hydrogen is a subcooling process during which some of the hydrogen is removed from the tank as vapor. In such a subcooling process, the ratio of the initial mass of liquid to the final, or remaining, mass of liquid and/or solid is shown as the Specific Mass Requirement (SMR). The SMR analyses for several methods of slush hydrogen production can be obtained from Reference 10.0-2. The direct vacuum pump SMR for the subcooling and solidification of liquid hydrogen is presented in Table 10.3-1, which shows that 1.195 pounds of normal boiling point liquid hydrogen can be converted to one pound of 50 percent hydrogen slush at the triple point temperature. Obviously, the mass of vapor removed is 0.195 pounds of the original liquid.

Freshly manufactured slush hydrogen undergoes an aging (stabilization) process, a phenomenom which is not well understood. The small solid particles of hydrogen change in shape, size, and density. Most of the change occurs during the first five hours after which the texture remains essentially constant. A stable texture of slush hydrogen is desirable especially from the standpoint of handling and transfer.

Table 10.3-1. Specific Mass Requirement-Slush Hydrogen Production-Direct Vacuum Pump Method

$$\text{Specific Mass Requirement} = \frac{\text{Initial Mass}}{\text{Final Mass}}$$

	SMR
NBP LH_2 to TP LH_2	1.123
TP LH_2 to 50% SH_2	1.0645
NBP LH_2 to 50% SH_2	1.195
TP LH_2 to 100% SH_2	1.129
NBP LH_2 to 100% SH_2	1.1268

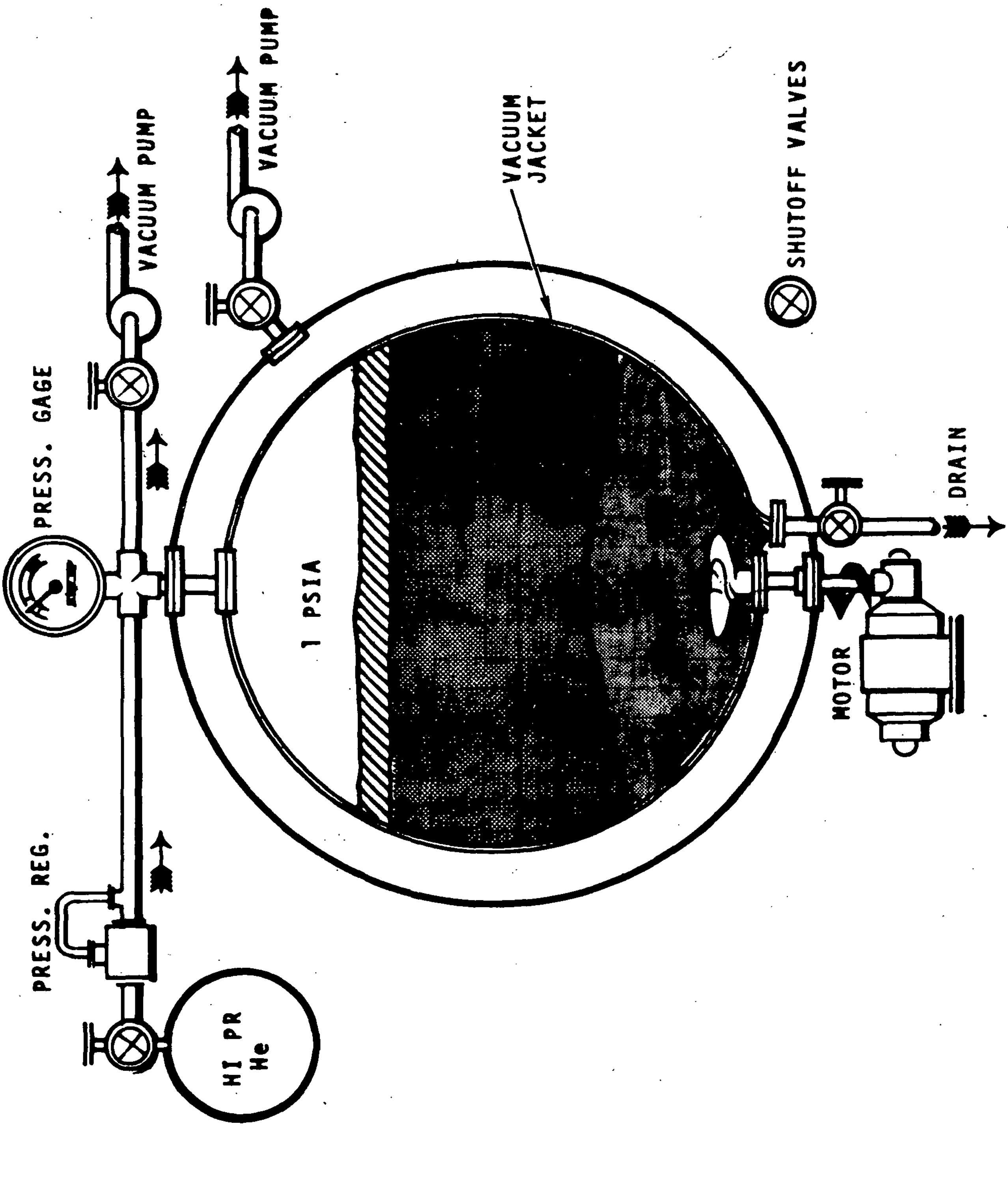


Figure 10.3-1 Slush Hydrogen Generator

10.4 HANDLING CHARACTERISTICS

Slush hydrogen must be aged for at least five hours so that handling characteristics will be consistent and accurately predictable. Small scale manufacturing and testing shows that 50 percent slush hydrogen can be handled much like liquid hydrogen except that continuous stirring may be required to prevent settling and an agglomeration of solid particles. The fluid flow friction factors and pressure losses for slush hardware including lines, orifices, venturies, and globe valve are available in Reference 10.0-3. The fluid flow characteristics of slush hydrogen in other than small size hardware (i.e., 0.65-inch diameter lines and 14-cubic foot tank) are yet to be determined.

10.5 THERMODYNAMIC ADVANTAGES

The densities of liquid, solid and slush hydrogen are presented for comparison in Table 10.5-1. The tabulated values show that density decreases as hydrogen changes from the solid state to normal boiling point liquid. This explains why the solid particles of slush tend to sink to the storage vessel bottom. The density ratio (0.86) of normal boiling point liquid to 50 percent slush hydrogen shows a 14 percent maximum potential savings in storage space. Thus, by substituting 50 percent slush hydrogen for normal boiling point liquid hydrogen, a related vehicle weight saving and mass fraction improvement can be achieved.

The heat capacities of cryogenic hydrogen are presented in Table 10.5-2. These values are also shown on the temperature-entrophy diagram for cryogenic hydrogen in Figure 10.5-1. Figure 10.5-1 shows that 12.5 Btu are required to melt one pound of 50 percent slush hydrogen to triple point liquid hydrogen along a constant temperature and pressure line. This is the heat of fusion for 50 percent slush hydrogen. One pound of triple point liquid hydrogen is increased along the saturated liquid line to normal boiling point liquid by the addition of 22.6 Btu of sensible heat. The latent heat of vaporization, the heat required to completely vaporize one pound of normal boiling point liquid to the saturated vapor line is 192 Btu.

The numerical value of slush hydrogen as a heat sink can be obtained from the cryogenic heat capacities. If slush hydrogen is substituted for a normal boiling point liquid hydrogen delivery load mass (m) requirement, the heat (H) that can be absorbed before the boiling point is reached is:

$$H = (12.5 + 22.6)\ m$$

$$H = 35.1\ m$$

If liquid hydrogen were used for a similar mission with the same heat leak (H), the additional mass (Δm) of liquid hydrogen that must be carried to allow for boiloff is:

$$H = 192\ \Delta m$$

Table 10.5-1. Hydrogen Density

	Density (lb/ft^3)	Pressure (psia)	Temperature (degrees R)
NBP LH_2	4.4	14.7	36.5
TP LH_2	4.8		
50% SH_2	5.1	1.02	24.8
100% SH_2	5.4		

$$\frac{\text{NBP } LH_2 \text{ Density}}{\text{50\% } SH_2 \text{ Density}} = 0.86 \text{ density ratio}$$

Table 10.5-2. Hydrogen Heat Capacity

Heat of vaporization LH_2 @ NBP	h_v = 192 Btu/lb
Sensible heat LH_2 from NBP to TP	h = 22.6 Btu/lb
Heat of fusion	h_f = 25 Btu/lb

TEMPERATURE - ENTROPY DIAGRAM FOR HYDROGEN

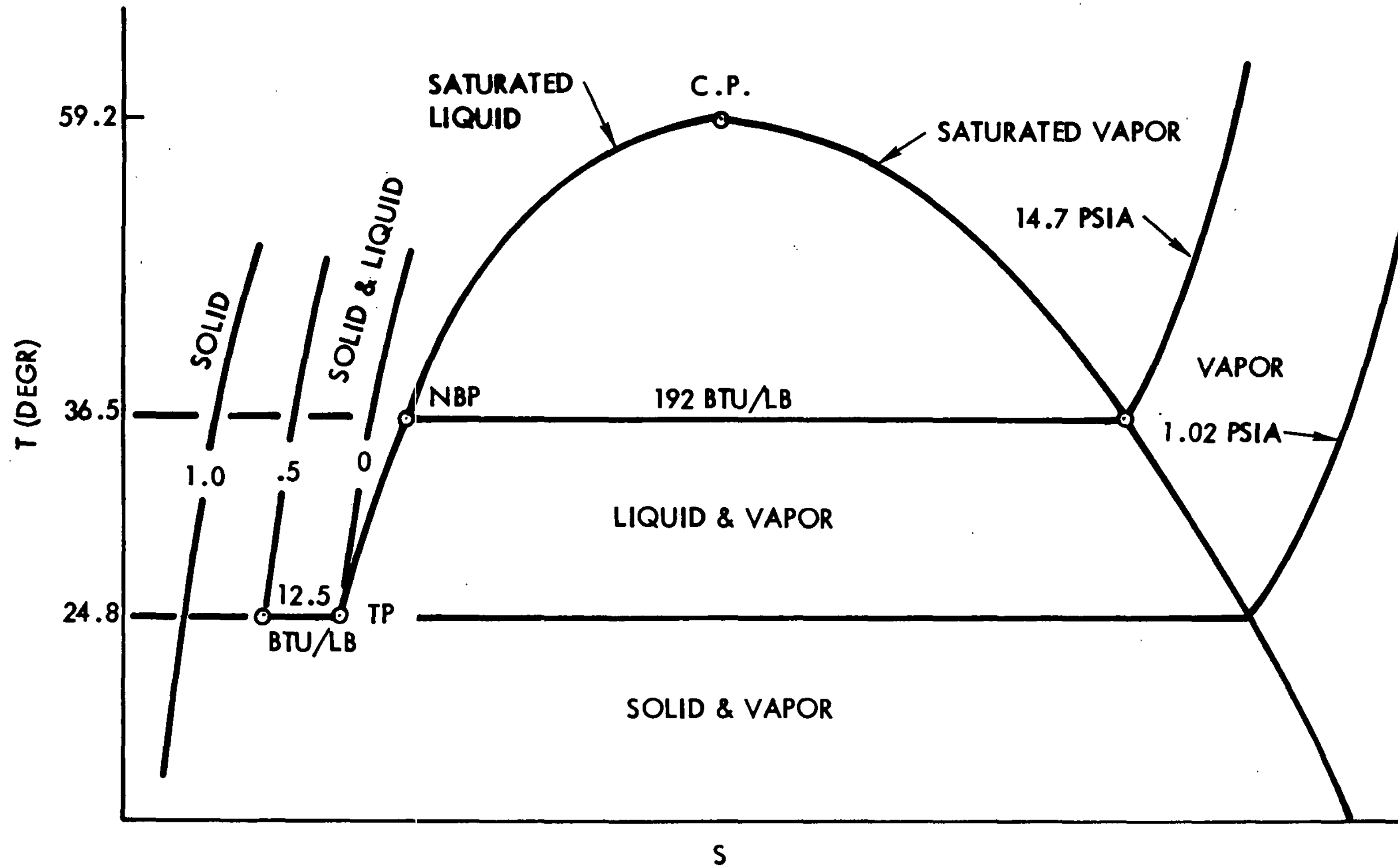


Figure 10.5-1 Temperature - Entropy Diagram for Hydrogen

Since the two heat leaks were made equal,

$$\frac{\Delta m}{m} = \frac{35.1}{192} \sim 0.18 = 18 \text{ percent}$$

Consequently, 18 percent of the delivery load is the maximum potential savings with respect to heat leak.

However, the advantages of density and heat capacity of slush hydrogen are not numerically additive. Figure 10.5-2 is a volumetric schematic comparison of 50 percent slush hydrogen and normal boiling point liquid hydrogen. The comparison is based on a 100-pound normal boiling point liquid hydrogen delivery mass equal to 100 percent volume. The schematic drawing illustrates the maximum volume savings (ΔV_H) that can be realized from heat capacity differences. Tank I is loaded with 118 pounds of normal boiling point liquid hydrogen of which 18 pounds is lost by boiloff resulting in 100 pounds at 100 percent volume of delivery propellant. Tank II is loaded with 100 pounds of slush hydrogen (0.5 solid fraction) at 86 percent of delivery volume which expands to 100 percent volume without a loss of mass.

In the theoretical event of a zero heat leak mission, slush hydrogen has no heat capacity advantage. However, Tank II could be made smaller because no allowance is necessary for expansion. The maximum savings of ΔV_D = 14 percent results from the difference in densities of normal boiling point liquid hydrogen and 0.5 solid fraction slush hydrogen.

When slush hydrogen is used directly in a liquid hydrogen engine, additional heat is required to melt the slush and bring it to normal boiling point temperature. The specific energy of the LH_2 used for the Saturn S-II booster is estimated to be 20,000 Btu per pound. The 35.1 Btu per pound additional heat required for the direct use of slush hydrogen is only 0.175 percent of the 20,000 Btu per pound of hydrogen converted to thrust. Consequently, the penalty for the direct use of slush hydrogen in a liquid hydrogen engine is not significant.

10.6 LOGISTICS TANK OPERATIONS

The current boiloff rate for ground storage and transport LH_2 tanks is 0.5 percent per day. If this minimum rate can be equalled for space application, the 18 percent maximum potential boiloff savings by the utilization of SH_2 can be realized in about 36 days. Since seven days is the longest space storage period that is anticipated for the logistics tank, the potential advantage for the remaining 29 days can be attained only from the larger space vehicles, such as the RNS or the CIS, that are used as self-storage tanks. However, the heat capacity advantage of slush hydrogen over liquid hydrogen appears to be very realistic in the form of potential boiloff savings and improvement in logistics efficiency.

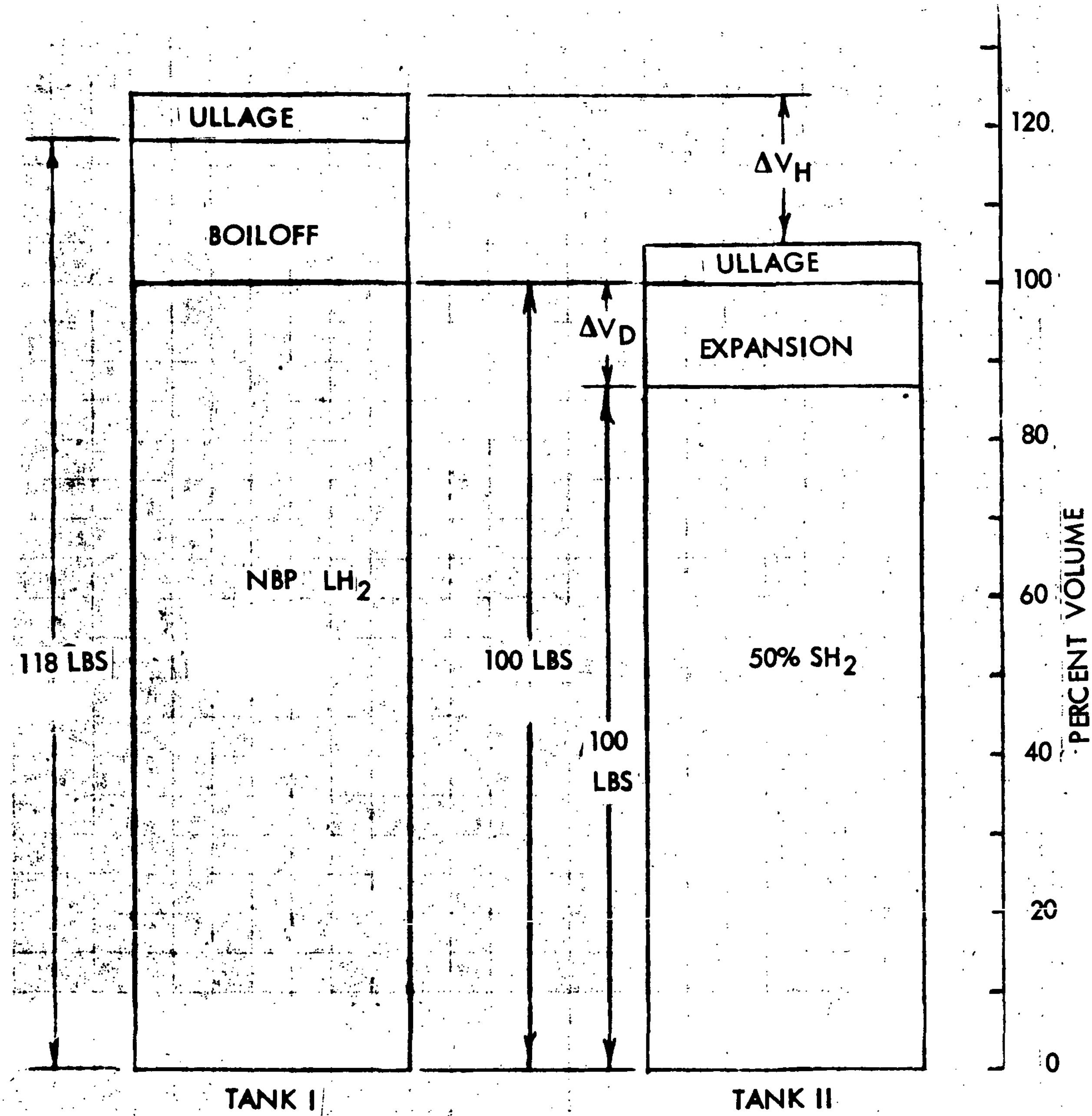


Figure 10.5-2 Volumetric Comparison of 50% Slush Hydrogen and Normal Boiling Point Liquid Hydrogen

The density advantage of SH_2 may be partially realized by appropriate loading or design of the logistics tank. The volume-limited logistics tank, used for servicing the RNS, can be loaded with a greater mass of SH_2 than LH_2. The mass-limited logistics tank for space vehicles other than the RNS can have a lesser volume for the SH_2 mass, equivalent to an LH_2 mass. However, some ullage volume must be allowed for anticipated expansion from melting SH_2. Emergency dumping or venting capability must be available for unanticipated excessive expansion. Any significant actual advantage resulting from the higher density of slush hydrogen appears doubtful.

Slush hydrogen will require stirring for storing and transfer, which could be a considerable disadvantage for space application. The necessity for upgrading the solid fraction of the logistic tank slush hydrogen during the launch countdown is a minor disadvantage. Solid fraction upgrading is accomplished readily by draining the tank mixture through a screen strainer.

10.7 COST ANALYSIS

An analysis was performed to determine the impact on the total program costs by reducing the liquid hydrogen boiloff losses as a result of using slush hydrogen for the CIS and RNS stages. Program Level D for the 1985-1990 time period was assumed as a model for the study. In each case an elapse time of six months was assumed from the start of in-orbit tanking operations to completion of the mission. It was further assumed that hydrogen was delivered and transferred into the receiver vehicles as 50 percent slush hydrogen. Although this approach is non-conservative, it establishes a maximum potential reduction in hydrogen boiloff losses.

The total liquid hydrogen boiloff losses for hydrogen stored in the CIS at saturation temperature for a tank pressure of 15 psia is 65,700 pounds over a six-month period, based on a predicted boiloff rate of 15 pounds per hour. If slush hydrogen is substituted for the saturated hydrogen, the decrease in boiloff losses can be calculated as follows.

The total heat that can be absorbed by 50 percent slush hydrogen before saturation is reached, for a nominal CIS hydrogen tank loading, is 35.1 Btu per pound. The total heat capacity for the 138,000 pounds of propellant is then:

$$138{,}000\ (35.1) = 4{,}843{,}000 \text{ Btu's}$$

The total heat input is:

$$65{,}700 \times 192 = 12{,}614{,}000 \text{ Btu's}$$

Subtracting the heat absorbed by the slush hydrogen from the total heat input and dividing by the latent heat of vaporization, the total boiloff is:

$$\frac{12{,}614{,}000 - 4{,}843{,}000}{192} = 40{,}471 \text{ pounds}$$

By adding the 40,471 pounds to the 138,000 pounds and reiterating, the final boiloff losses amount to approximately 34,200 pounds. This represents a decrease in hydrogen boiloff of:

65,700 - 34,200 = 31,500 pounds

In terms of total program cost savings based on a propellant delivery cost per pound of 178 dollars,* the net cost savings (based on ten payload placements per program) is:

31,500 x 10 x 178 = 56,070,000 dollars

The predicted hydrogen boiloff rate for the RNS stage is 6 pounds per hour when stored at saturation conditions. This results in a hydrogen boiloff loss of 26,500 pounds per mission. By substituting slush hydrogen, the boiloff losses can be eliminated entirely. In terms of total program, a savings of 265,000 pounds of liquid hydrogen can be realized. Based on a delivery cost of 234 dollars per pound* for RNS missions, a total program cost savings of approximately 62 million dollars can be achieved. The reduction in boiloff losses and cost savings are summarized in Table 10.7-1.

Table 10.7-1. Slush Hydrogen Impact on Program Costs

Vehicle	Boiloff for Saturated Storage (Per Mission) pounds	Boiloff Using 50% Slush (Per Mission) pounds	Hydrogen Boiloff Reduction (Per Program) pounds	Cost Savings (Per Program) Million $
CIS	65,700	34,200	315,000	56.07
Remarks - Program Level D del. cost $178/lb				
RNS	26,500	0	265,000	62.0
Remarks - Program Level D del. cost $234/lb				

* See Propellant Delivery Mode, Section 2

10.8 REFERENCES

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