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ORBITAL REFILL OF PROPULSION VEHICLE TANKAGE

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ORBITAL REFILL OF PROPULSION VEHICLE TANKAGE

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FOREWORD

The following final report summarizes the technical effort conducted under Contract NAS3-21360 by the General Dynamics Convair Division from May, 1978 to September 1979. The contract was administered by the National Aeronautics and Space Administration, Lewis Research Center, Cleveland, Ohio.

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Assisting J. Risberg, Modification of HYPRS Computer Program
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All data are presented with the International System of Units as the primary system and English Units as the secondary system. The English system was used for the basic calculations.

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SUMMARY

This study was concerned with three main areas: 1) identification of representative receiver-fluid combinations for propulsion vehicle tankage, 2) on-orbit refill analysis of each of three candidate receivers selected for further evaluation. and 3) modelling analysis to determine experimental conditions necessary for verifying the filling characteristics of each receiver vehicle configuration.

Initially, an evaluation was made of candidate propulsion vehicle system tankage for on-orbit resupply. Various NASA, Convair, and industry studies were reviewed to identify alternative concepts for orbital transfer vehicles (OTV), space platforms/space stations, and spacecrafts. STAR and International Aerospace Abstracts were consulted to insure adequate coverage of representative concepts. All document sources consulted in this review are referenced in Appendix A. After reviewing applicable documentation, vehicles were listed and categorized (Appendix A).

Following the literature review and vehicle documentation process, one vehicle was selected, in accordance with the statement of work, from each of the following categories: an earth-storable vehicle with partial screen acquisition device, a cryogenic vehicle with partial screen-acquisition device, and a cryogenic vehicle without a screen device. The selected vehicle configurations, given in Figure 2-1, were:

- Earth Storable Vehicle - A low thrust (LTL) concept that employs MMH and N_2O_4 propellants. This vehicle is representative of a configuration that can be constructed from existing hardware.

- Cryogenic Vehicle Without Acquisition Device - The Personnel Orbital Transfer Vehicle (POTV) consists of two cryogenic stage for delivering payloads (20,000 kg) to geostationary orbit (GEO) and back to low-earth-orbit (LEO). This vehicle uses LH_2 and LO_2 propellants and will be available in the near term (1980's).

- Cryogenic Vehicle With Partial Acquisition Device - The Cargo Orbital Transfer Vehicle (COTV) is capable of delivering 250,000 kg to GEO and and returning to LEO. This vehicle is contemplated for the 1990's and beyond in the era of space-basing. Again, main tank propellants are LH_2 and LO_2 .

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Vehicle missions were defined for each candidate receiver to encompass the key issues of orbital-refueling operations. These operations included major activities from post-mission "storage" in the LEO parking orbit through resupply. A re-fueling analysis of each mission was then performed to determine such variables as propellant losses, propellant tank pressures, fill levels and total refill time.

Analysis of POTV and COTV orbital refill indicated that key issues (or concerns) were the same for each vehicle. These were: a) how to avoid excessive tank pressures, b) how to avoid liquid venting, and c) how to perform refill with the limited resources available in space. It was also found that the same refill technique and procedures were applicable to each vehicle. This was a surprising result in light of the limited orbiter-tanker resources available for POTV refueling, which contrasts with the substantially greater orbital depot resources assumed for COTV refill operations.

The refueling analysis showed that problems could be circumvented by introducing the processes of initial tank vent, prechill and fill.

Initial tank vent is required only if helium is present. Tank vent (or blowdown) to a low pressure will expel sufficient helium that concern for excessive tank pressure (due to helium partial pressure) or helium trapped within a screen device is eliminated.

The prechill process is required to reduce tank temperature to an acceptably low level prior to initiating tank fill. Prechill will consist of a series of charge and vent cycles, where either liquid or vapor is introduced during the charge cycle. Vapor only will be expelled during each vent cycle because the elevated tank temperatures will preclude the possibility that liquid is present at vent initiation. Significant analysis results are listed below:

1. Liquid oxygen tank prechill is not required because under no circumstance will excessive tank pressures occur during refueling operations. Thus emphasis was directed at the liquid hydrogen tank.

2. Rapid prechill of the hydrogen tank does not appear to be an important consideration. Figure 3-4 indicates that up to 64 hours of activity is required to support a single orbiter/POTV rendezvous and transfer operation, five percent of which may be required for propellant transfer. It seems evident that propellant transfer operations could be increased to 10 percent of the total timeline without significant impact. This is nearly two orders of magnitude more time than the approximate 200 second prechill time indicated by Figure 3-20.

3. Liquid hydrogen consumed for the tank prechill process will have an insignificant influence upon overall efficiency and cost of transporting propellants into space for POTV refueling. As a result, propellant transfer efficiency should not be an important consideration in the prechill process selection.

It is concluded that the prechill process described and analyzed in Section 3.3.2.5 will satisfy the requirements of simplicity, reliability and safety.

At the completion of prechill, the tank is locked up and liquid introduced through one or more spray nozzles to accomplish tank fill. A fill condition of 90 percent or greater will be achieved without the need for venting if near-thermal equilibrium conditions are present. It was determined that sufficient bulk fluid agitation will be created by the entering liquid to provide near-thermal equilibrium during fill. Together, tank prechill and bulk fluid agitation should provide a no-vent fill or refill.

Propellant transfer timelines were developed for a POTV refueled by an orbiter-tanker. Tables 3-13 and 3-15 show that this transfer operation can be performed in three hours by over-lapping LH_2 and LO_2 transfer.

The primary requirements for LTL refueling operations are:

1. Minimize propellant tank venting in the vicinity of the orbiter because N_2O_4 and MMH are corrosive. Liquid venting must be avoided.

2. Prevent helium entry to the screen galleries because vapor-free liquid flow from each propellant tank must be assured.

Refueling will include the initial vent and tank fill processes, but not prechill, because tank and propellant temperatures will be approximately the same. Propellant tank fill pressures will remain below the vent pressure levels if the initial vent (or blowdown) process reduces tank pressure to approximatdy one to two atmospheres.

A procedure was identified that would satisfy the above requirements during initial vent. Basically the approach is to rely upon procedures and added propellant plumbing to transfer propellant between tanks. In this way a tank may be drained of excess propellant prior to the initial vent process that expels helium.

The single potential concern of the selected refueling procedure is that propellant contained within the screen devices might boil during tank vent. Boiling will be avoided if sufficient liquid residual is maintained in contact with the screen to replenish liquid lost through evaporation. Orbital experiments were not recommended because such tests would be configuration sensitive and have limited applicability.

An important result of this study is that zero-g mass gauging devices will be required for on-orbit refueling operations of earth storable and cryogenically fueled vehicles. A survey was conducted of existing radiation and RF type devices to identify the state of the art.

An analysis was also conducted which indicates that propellant mass gauging is feasible through thermodynamic means of measuring tank pressure increases resulting from a fixed helium mass addition.

The processes selected for further evaluation (i.e. modeling) were prechill and fill. The initial tank vent process was judged to be sufficiently well defined to preclude experimentation. Prechill and fill are similar in one important aspect; it is intended that heat and mass transfer be dominated by forced convection in order that these processes remain independent of acceleration environment. Consequently, a modelling analysis was performed to identify conditions under which these processes can be simulated with a 45.7 cm (18 inch) diameter test tank (the largest size that can be contained within a spacelab doublerack). Per NASA/LeRC directive, the Spacelab was groundruled as the orbital experimental test facility.

It was concluded from computer simulations that results could not be directly extrapolated to a full scale OTV, even for tests conducted in a zero-g environment with LH_2. This discrepancy between model and prototype behavior is influenced by the substantial difference between prototype and model tank volume-to-mass ratio, which is an important test parameter. It is expected, however, that the heat transfer phenomena involved in the prechill and fill processes can be evaluated. Empirical coefficients obtained from such tests could be applied to an analytical model such as HYPRES, which would then be employed for full scale vehicle predictions.

Assessments were also made of the influence of a fluid substitute (LN_2) and a one-g test environment on test results. It was concluded that one-g test results would not be applicable to prototype vehicle predictions but that tests with LN_2 would provide useful data.

Finally, discussions with NASA/MSFC safety personnel yielded the following comments regarding the proposed receiver tank experiments to be conducted in the Spacelab environment:

1. A waiver would be required by the experiment integration safety review board to allow the anticipated LN_2 quantities for refueling tests.

2. Liquid hydrogen is unacceptable under any condition.

1

INTRODUCTION

The United States is on the threshold of a space industrialized era. Some of the ambitious space programs conceived by the NASA and industry include the construction of large antenna structures, solar powered satellites, and propellant depots. A common element of these programs is the requirement of effectively transferring propellants in space.

This area of orbital propellant transfer, or propellant management, has long been identified as a technology area by the NASA-LeRC and Convair. A previous study, "Orbital Refill of Fluid Management Systems", Reference 1-1, dealt with the problems of refilling small cryogenic and earth-storable systems. Convair has performed independent studies in the area for several years, Reference 1-2 and 1-3. Experience gained in the previous studies has served as a starting point for this study on orbital refuelling of vehicle tankage.

The objectives of this study were to 1) develop techniques for such necessary orbital propellant transfer and, 2) to identify experimental programs to verify these techniques.

1.1 SCOPE

A number of future missions have been defined which require orbital propellant transfer capability. In near-term, space programs such as the manned-geosynchronous-sortie, and very high energy probes to other planets will require the transfer of propellant quantities in the order of 30 to 300 metric tons per year. Earlier studies have shown that performance and life of operational spacecraft can be increased by resupplying attitude control propellants, fuel cell reactants, sensor coolants, or chemical laser fluids. Beyond the year 2000, large space industrialization programs may require propellant quantities that are several orders of magnitude greater than for the near-term. The most ambitious program now being considered is the Solar Power Satellite Program requiring Heavy Lift Launch Vehicle (HLLV), space construction bases, and both electric and chemical Orbital Transfer Vehicle (OTV).

Although there are many potential orbital refuelling applications, the scope of this study was limited to analysis of and experimental modeling techniques for propellant transfer between supply tanks and receiver OTVs. The three OTV configurations selected for orbital refill analysis were identified using the selection procedure described in Section 2.

1.2 GROUNDRULES

Guidelines were established for selecting three vehicle configurations representative of those contemplated for various types of future space programs. It was required that one vehicle would be selected from each of the following categories: an earth storable vehicle with partial-screen acquisition device; a cryogenic vehicle with a partial-screen acquisition device; and a cryogenic vehicle without a screen-device. Representative vehicle selections were to be made following a literature review of previously conducted NASA and industry studies, as well as current Convair studies on future space programs.

1.2.1 EARTH STORABLE VEHICLE. An additional requirement was imposed upon this selection process; that of identifying hardware either from existing programs, or from previous study efforts. Because the data base for earth storable vehicles and missions was considerably smaller than for cryogenic OTVs, no attempt was made to optimize the vehicle configuration. Rather, the intent was to select a configuration which would be representative of its vehicle class.

1.2.2 CRYOGENIC VEHICLE. Of the two cryogenic vehicle classes selected, one was assumed to be available in the near-term (1980's) and the other was selected for application in the late 1990's and beyond. The vehicle for near-term application was assumed to have subsystems consistent with its early development period. Consequently, this vehicle will not have a screen acquisition device nor any subsystem requiring considerable technology. The more advanced OTV will be comprised of more sophisticated subsystems, such as a partial screen acquisition and an advanced engine system requiring no pre-pressurization.

Different methods of propellant supply will also be available to each OTV. Space programs for the 1980's will rely upon propellant resupply from an orbiter tanker. Programs contemplated for the 1990's and beyond were assumed to have orbital propellant depots available for OTV resupply.

1.2.3 EXPERIMENT MODELING. Experimental modeling techniques of receiver-tank-resupply were developed during the study. These techniques were employed to determine the usefulness of simulant fluids and scale model testing. Both ground based facilities, including drop towers, and the Spacelab on-board the shuttle were assumed to be available for conducting the experiments. Scale model size was limited to the largest experiment test package that could be installed within a Spacelab double rack enclosure. This restriction confined analysis to that of a relatively small scale test tank (approximately one-tenth scale or less).

2

IDENTIFICATION OF CANDIDATE RECEIVERS

The purpose of task I was to select three potential vehicle concepts for subsequent orbital-refueling analysis. These concepts were to be representative of those vehicles expected to be designed for the 1980's and 1990's; that is, representative in terms of equivalent subsystems, orbital staytimes, thermal requirements, legistics and refurbishment requirements.

Following the literature review, one vehicle was selected, in accordance with the statement of work, from each of the following categories:

1. An earth-storable vehicle with partial acquisition device

2. A cryogenic vehicle with partial acquisition device

3. A cryogenic vehicle without an acquisition device.

The three vehicle concepts selected are shown in Figure 2-1.

Convair's approach for reviewing these conceptual designs and for determining representative configurations to be further analyzed in Task II, is shown in Figure 2-2. First, a literature review was accomplished. In order for it to be complete, all potential propulsive vehicle receiver tanks were included to show the wide variety of uses for propellant transfer technology. The types of receiver tanks identified include the following:

Orbital Transfer Vehicles

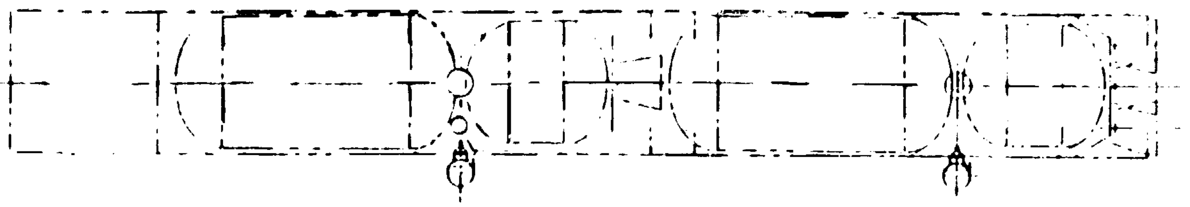
- Mini-maneuvering (e.g., teleoperator)
- High and low thrust chemical
- Nuclear and solar electric (OMS, RCS)

Orbital Maintenance and RCS Tankage

- Space station
- Propellant depots
- Large space structures
- Automated satellites (include cooling propellant)

After reviewing applicable documentation, these vehicles were listed and categorized according to fluids used, flow rates, tank geometry and pressure, acceleration-

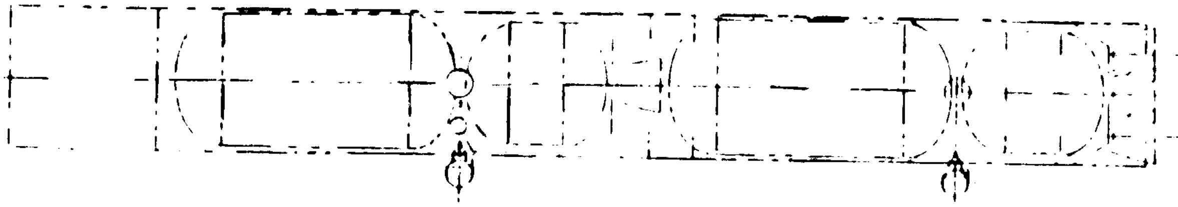
POTV



PROPELLANT CAPACITY, KGM (LBM)

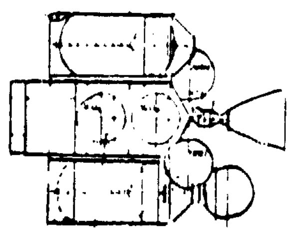
LO_2 = 44,407 (97,900) EA.
LH_2 = 8,256 (18,200) EA.

COTV



LO_2 = 198,817 (438,038) EA.
LH_2 = 37,055 (81,692) EA.

LTL



N_2O_4 = 3,524 (7,768)
MMH = 2,129 (4,693)

Figure 2-1. Three Vehicle Configurations Were Selected With Concurrence From NASA/LeRC

environment, and total quantities of fluids consumed as directed by the statement of work. Baseline vehicle characteristics were derived and candidates which had appropriate requirements for orbital resupply were selected.

2.1 LITERATURE REVIEW

Various NASA, Convair, Aerospace Corp. and industry studies have been reviewed to identify alternative concepts for orbital transfer vehicles (OTV), space platforms/ space stations, and automated and manned spacecraft. The Convair space data banks

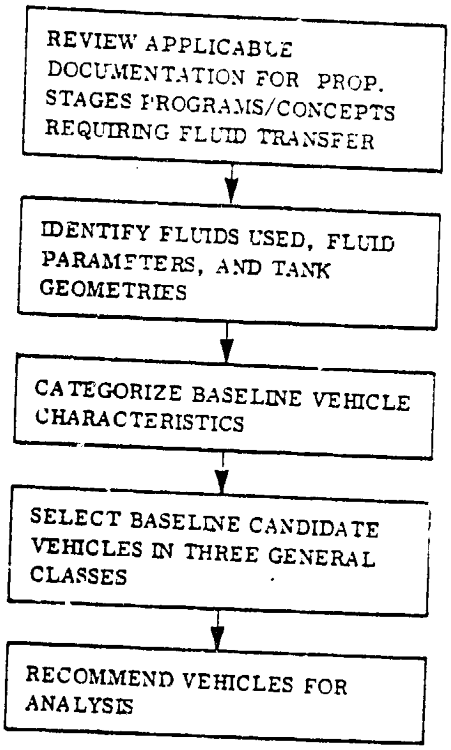


Figure 2-2. A Straightforward Review of all Upper Stage Concepts was Employed to Yield Representative Study-Candidates

and personnel were also reviewed to obtain available information for identification and description of orbital transfer, space platforms and spacecraft expected to have tanks refilled in space. STAR and International Aerospace Abstracts were consulted to insure adequate coverage of representative concepts. All document sources consulted in this review are referenced in Appendix A.

All available tanks that could conceivably be involved in fluid transfer were also considered. This included the STS Reaction Control System (RCS), the Orbital Maneuvering System (OMS) propellant tanks, and the Shuttle External Tanks (ET) as depot or OTV configuration. As information was extracted for each item, the source document identify number and pertinent page numbers were referenced.

Seventy-nine candidates were identified as a result of this literature review, and are tabulated in Tables A-1 through A-4, Appendix A.

The first 19 items (Table A-1) include the STS orbiter, space tugs, and orbital-transfer type vehicles and stages derived from previous upper-stage programs such as reusable Agena, Centaur, and the transtage. Early large-scale orbital-transfer vehicles based on the Saturn V S-II stage tanks and engines are also described. The first nineteen items also include early space stations, LEO observatories, and a space taxi. Items 20 through 39 (Table A-2) include the various OTV concepts identified to support and move large space structures as solar power satellites, large radar platforms, propellant depots, space stations, and manufacturing facilities. Some duplication may exist due to inclusion of competitive concepts and designs. Items 40 through 69 (Table A-3) include concepts for propellant depots, the various development phase versions of solar power satellites, the supporting space stations, staging depots, construction space bases, radar platforms, earth observation platforms, antenna farms (for communication and power relay), and logistics tanks. The emphasis was on showing space station/platform concepts likely to use refillable tanks. Items 70 through 79 (Table A-4) include automated spacecraft likely to include refillable tanks; items 80-82 (Table A-4) include manned-spacecraft concepts.

2.2 IDENTIFICATION OF FLUID PARAMETERS AND TANK GEOMETRY

Vehicle programs and concepts identified from the literature search were further defined according to propellant fluid, tank geometry, operational and venting pressure and acceleration environment, fluid temperature and propellant expulsion rate, as required by the

Statement of Work. Data not currently existing was derived from conceptual design data. For instance, tank geometry was assumed to be constrained by STS Orbiter cargo-bay dimensions, where STS was the designated delivery vehicle. Similarly, acceleration forces were matched to the mission. For example, delivery of large space structures from LEO to GEO requires low-thrust acceleration less than 1G; consequently, propulsion tanks of associated vehicles were assumed to operate in a less-than 1G acceleration environment.

2.3 CATEGORIZATION

A generalized classification of space vehicle tanks into fluid/acquisition classes was accomplished using the following groupings as stipulated by the proposal.

Class A	Storable with Acquisition Device
Class B	Cryogenic with Acquisition Device
Class C	Cryogenic without Acquisition Device

Additional categorization was accomplished in terms of tank size, tank geometry, operating pressure, vent pressure, temperature and flow rates as defined by limits given in Table 2-1. These categories, together with background in mission analysis, enabled the selection of representatives for each tank size.

Table 2-1. Categorization Limits

SIZE:	WEIGHT:
Large	> 45400 kg (100,000 Lbs)
Medium	45400 kg (100,000 Lbs) - 4540 kg (10,000 Lbs), (45000 kg - 2,000 kg for LH_2)
Small	4540 kg (10,000 Lbs) - 454 kg (1000 Lbs)

SHAPE (Geometry):

- Spheroidal
- Cylindrical
- Ellipsoidal
- Toroidal

ACQUISITION SYSTEM/METHOD:

- Acceleration
- Capillary
- Bladder
- Other:
  - Pressure
  - Pump

ACCELERATION LEVEL: < or > 1g

TEMPERATURE:

Cryogenic	<200K (Boils Off)
Avg	200 - 320K (Storable)
High	320 - 400F (Cools Off)
Hot	> 400K (Rapid Cooling)

OPERATING PRESSURE, kN/m^2(psia):

Low	< 200 (29)
Medium	200-2000 (29 - 290)
High	> 2000 (290)

VENT PRESSURE, kN/m^2(psia):

Low	< 200 (29)
Medium	200-2000 (29 - 290)
High	> 2000 (290)

FLOW RATE, $kg/m^2/sec$

Low	< 10 (2)
Medium	10 - 100 (2 - 20)
High	> 100 (20)

2.4 BASELINE CANDIDATE VEHICLES

Baseline candidate vehicles were selected from the total listing in Tables A (Appendix A). This list of candidate vehicles is comprised of representative and realistic designs that are most likely to require propellant transfer in the next two decades.

Initially, likely candidates were screened based on those thought to be applicable for missions projected in this time period. On-orbit propellant transfer will operate in two general mission arenas. One is the transportation of payloads from LEO to a high energy orbit, e.g., LEO to GEO, and LEO to lunar orbit. The second is moving systems within an orbit. The former requires high impulse cryogenic propellants lifting heavy payloads. The latter operates near a base location for servicing and maneuvering payloads. These vehicles may require long orbit stay times between refill and are better suited for earth storable propellants. Generally these vehicles operate within a few hundred miles and in a range of orbit inclinations from the propellant base.

In this timeframe propellant transfer technology will first be used for topping off cryogenic vehicles which cannot be carried full to orbit due to STS payload limitations. Toward the latter part of the 1990s both cryogenic and earth storable vehicles are expected to be space based.

In the process of vehicle selection, those stages based on existing expendable vehicles were eliminated quickly. These vehicles if used in the Shuttle/Orbiter will be flown in one flight; therefore, not requiring propellant transfer. Also eliminated were receivers for RCS propellant. These generally require small amounts of propellant; a better solution might be total receiver tank changeout versus propellant transfer.

In the past two years, much effort has been concentrated defining the Solar Power Satellite (SPS) and its transportation system. Vehicles have been defined in those studies which depend on propellant transfer either at LEO or GEO. The SPS vehicles appear representative of those required for future space needs. From these-defined vehicles, Table 2-2 details the vehicle selected based on propellant type and acquisition device, and tank shape and size.

To enable a more thorough screening, each of the three major categories were broken down into three sub-categories (large, medium, and small tank). An attempt was made to obtain a representative candidate for each major category and sub-category as shown in Table 2-3. This comparative matrix was used to determine the final three vehicles selected.

2.5 VEHICLE RECOMMENDED FOR ANALYSIS

The determination of which three vehicles should be selected from Table 2-3 was primarily based on usage and configurations most likely to be produced in the 1980s

Table 2-2. Initial Selection of Representative Vehicles

Category	Representative Vehicle	Justification
Propellant Type/Acquisition Device		
Storable with partial acquisition device.	Low thrust liquid orbital vehicle (Item 37, 39) (Two options shown)	Maximum number of refills expected
Cryogenic with/without partial acquisition device	Common stage OTV (117K/117K), (Item 22) (With acq. device) POTV, (Item 34) (Same without acq. device	Required for SPS program, or most manned GEO operations
	Pilot plant SPS (Item 44) Liquid—gaseous argon for orbit maintenance + RCS (Periodically refilled) during assembly at LEO	Requires least propellant delivery to LEO during assembly

Category	Representative Vehicle
Tank Shape	
• Cylindrical with curved or ellipsoidal ends	LH_2 tank for common stage OTV (Item 22)
• Ellipsoidal	LO_2 tank for common stage OTV (Item 22)
• Spheroidal	1/2 stage oxidizer & fuel tanks for OTV-475 ST (Item 36) or satellite control section (Item 8)
• Toroidal	On orbit assembly OTV LO_2 tank (Item 29)
Tank Size	
• Large	OTV - 475T (520K/520K) Item 26) LH_2 Cylindrical, LO_2 Ellipsoidal
• Medium	Common stage OTV (117K/117K) Item 22 LH_2 Cylindrical, LO_2 Ellipsoidal
• Small	1/2 stage oxidizer (LO_2) & fuel (LH_2) tanks for OTV 475T (Item 26) or Teleoperator retrieval system hydrazine tanks (N_2H_4 cylindrical)

and 1990s. With NASA/LeRC concurrence, three vehicles shown in Figure 2-1 were selected. Table 2-4 contains a summary tabulation of the vehicle characteristics.

2.5.1 CLASS A - EARTH STORABLE VEHICLE WITH PARTIAL ACQUISITION DEVICE. A low thrust liquid (LTL) concept was selected as representative of this class. The usage of this vehicle would primarily occur near one altitude location, e.g., LEO, GEO, lunar orbit. However, during early years operation it could be considered for moving large space demonstration structures between LEO and GEO. Its primary function would be to service, inspect, and retrieve objects near its operating altitude base. At LEO its prime function would be to enhance the Shuttle capability by placing the Shuttle payload at altitudes and inclinations beyond the STS capability. At other altitudes payload propellant servicing or module replacement are uses which may enhance cost effectiveness. Orbital debris removal is another application for the LTL vehicle.

2.5.2 CLASS B - CRYOGENIC VEHICLE WITH PARTIAL ACQUISITION DEVICE. The Personnel Orbital Transfer Vehicle (POTV) consists of two cryogenic 117,000 pound propellant stages capable of delivering three or four men to geostationary orbit and back to LEO without GEO refueling. The vehicle with GEO refueling is capable of delivering a 75 man passive module plus two-man crew module plus 20,000 Kg (44,000 lb)

Table 2-3. Representative Candidate Receivers

Fluid/Acquisition Class	Tank/Size	Representative Candidate Receiver Vehicle	Tank Propellant Load, kg (lb)	Fluid Temp	Tank Size	Tank Shape	Acq. Method	Design Access	Uses (1980-1995)	Comments	Reference Document
A Storable with Acquisition Device	-1 Large										
	-2 Medium	Low thrust liquid orbital vehicle (Item 37)	N_2O_4 = 13,951 (30,763) MMH = 11,62? (25,6??)	Ambient	Medium	Cyl.	Capillary	Yes	LEO transfer & assembly, section translation, early transfer to GEO	Multipurpose LEO/GEO transfer and manipulation	CASD-AS 77-[illegible], 18 [illegible] 1978, pp. [illegible] - 272
	-3 Small	LTL (with OMS Tank) (Item 39)	N_2O_4 = 3,524 (7,768) MMH = 2,129 (4,693)		Small	Sphere	Capillary	Yes	Orbit Maint + attitude	Military + NASA [illegible] payloads	Table A-1, NASA CR [illegible]; GDC [illegible] 78-[illegible]
		Teleoperator (Item 20)	N_2H_4 = 680 (1,500)		Small	Cyl.	Bladder		LEO transfers, service inspection, retrieval	Frequent refill	
B Cryogenic with Acquisition Device	-1 Large	OTV 475T (520K/520K) Payload = 250 Tons (Item 26)	LO_2 = 198,6?0 (438,000) LH_2 = 37,0?0 (81,600)	Cryo Cryo	Large Large	Ellips. Cyl.	Capillary Capillary	Yes Yes	500,000 lb LEO payload sections transfer from LEO to GEO	If electric OTV infeasible or too slow, potentially largest user of propellant	JSC 11568, Vol. [illegible] p. VI [illegible]
	-2 Medium	Common Stage OTV (117K/117K), 75 men + 44K (Item 22)	LO_2 = 44,41? (97,900) LH_2 = 8,2?0 (18,200)	Cryo Cryo	Medium Medium	Ellips. Cyl.	Capillary Capillary	Yes	LEO to GEO primarily cargo and personnel trans. for	Largest number of refills expected, most common usage	CASD-[illegible]-78-[illegible], JSC-13?57, Vol. II pp. 7-[illegible], 7-1?
	-3 Small	1/2 Stage for OTV 475T (520K/520K) (Item 26)		Cryo Cryo	Small Small	Sphere Sphere	Capillary Capillary		Various uses for orbit circularization and adjustment (LEO to GEO transfer)	Frequent usage, allows flexibility in GEO incremental maneuvers	JSC 11568 Vol. III, p. VI [illegible]-5, VI [illegible]
C Cryogenic without Acquisition Device	-1 Large		LO_2 = 603,700 (1,331,000) LH_2 = 100,700 (222,000)	Cryo Cryo	Large Large	Ogive/Ellips. Cyl.	Accel Accel	Yes Yes	Existing tank for track transfer of LO_2 + LH_2	Many places for delivery of fluids to [illegible] locations in LEO and GEO	TRI A-2, [illegible], [illegible]
	-2 Medium	Personal Orbit Transfer Vehicle (POTV) (Item 34)	LO_2 = 49,500 (89,300) LH_2 = 7,5?9 (16,600)	Cryo Cryo	Medium Medium	Ellips. Cyl.	Accel Accel	Yes Yes	Alternate for LEO to GEO cargo + personnel transfer	Potentially largest number of refills expected	JSC 11568, Vol. [illegible], 1976 p. [illegible]
	-3 Small	Distributed Electric POS Units for SPS + (Item 44)	L_{Ar} = 3,6?? (8,000)	Cryo	Small	Cyl.		Yes	L_{Ar}, G_{Ar} electric accel. for orbit altitude	During construction of LEO SPS orbit maintenance & altitude change	Boeing D180-20[illegible], p. 342; [illegible]

Table 2-4. Additional Characteristics for Selected Representative Vehicle Tanks

Class Reference No.	Selected Vehicle Name	Fluid Type	Fluid Mass Per Tank KGM (LBM)	Tank Surface Area m^2 (Ft^2)	Tank Volume m^3 (Ft^3)	Tank Mass KGM (LBM)	Tank Material	Pressure kN/m^2 (psi) Operating	Pressure kN/m^2 (psi) Maximum
A-3.1	LTL (With OMS Tanks[1]), Item 39	N_2O_4	3524 (7768)	3.6 (39.1)	2.5 (90)	253 (558)	Titanium	1724 (250)	( )
		MMH	2129 (4693)	3.6 (39.1)	2.5 (90)	253 (558)	Titanium	1724 (250)	( )
B-1	OTV (520K/520K) With Acquisition Device (Item 26)	LO_2	198,817 (438,308)	159 (1710)	183 (6460)	1150 (2536)	Welded AL	172 (25)	365 (53)
		LH_2	37,055 (81,692)	339 (3648)	548 (19,363)	2228 (4911)	Welded AL	172 (25)	200 (29)
B-2	Common Stage OTV (117K/117K), Item 22 (With Acquisition Device)	LO_2	44,407 (97,900)	59 (634)	41.3 (1457)	260 (573)	2219-T87 AL	172 (25)	325 (47.1)
		LH_2	8256 (18,200)	128 (1378)	116.1 (4100)	447 (986)	2219-T8 AL, ALY	172 (25)	192 (27.8)

[1] 1.25m (49.1-in.) diameter × 2.4m (94.3-in.) long.

of payload to GEO and returning both manned modules. This is 48,500 Kg (107,000 lb) to GEO and returning 28,500 Kg (63,000 lb).

This vehicle is a very effective system for the SPS era when large manned requirements are required at GEO for repair and construction of these satellites. Further, this vehicle is not limited to the era of heavy lift launch vehicles (HLLVs), the vehicles can be carried in separate Shuttle flights and topped with propellants by an Orbiter/Tanker or by a propellant depot. Early uses of the POTV would be required for GEO payload servicing and repair. The function of man in space is to augment unmanned servicing tasks. Man would be used to diagnose and repair space structures; do the out-of-the-ordinary space functions.

2.5.3 CLASS C - CRYOGENIC VEHICLE WITHOUT ACQUISITION DEVICE. The Cargo Orbital Transfer Vehicle (COTV) is capable of delivering 250,000 Kg (550,000 lb) to GEO and returning to LEO. This vehicle would operate in an era of space basing. Present concepts consider use of an electric OTV. However, these vehicles require nearly a year to transfer payloads from LEO to GEO. Should mission duration requirements of electric OTV technology prove infeasible the COTV would be developed and represent the largest vehicle category.

3

POTV ORBITAL RESUPPLY

In this section a mission scenario will be developed for the POTV concept selected in Section 2. A realistic mission will be defined which encompasses the key issues of orbital-refueling operations. These operations will include all major activities from post-mission "storage" in the LEO parking orbit through resupply. Vehicle and orbiter-tanker subsystem requirements needed for orbital refueling will be identified. Operational procedures and techniques for orbital propellant transfer will then be developed.

3.1 MISSION SCENARIO

In the early 1990's, with propellant depots not yet available, OTV orbital resupply could be provided by dedicated Orbiter tankers. Figure 3-1 illustrates an orbital refueling operation in which propellants are transferred from a tanker kit to an OTV which is docked to the Orbiter. The tanker kit, (consisting of an LH_2 and an LO_2 tank, transfer system, and pressurization system), is contained within the Orbiter payload bay.

TECHNICAL CONSIDERATIONS

- PROPELLANT TRANSFER EFFICIENCY
  - SUPPLY TANK WEIGHTS
  - SUPPLY TANK RESIDUALS
  - OTV FILL LOSSES
  - ORBIT STAY-TIME LOSSES
- OPERATIONS
  - INSULATION
  - PROPELLANT TRANSFER
  - PROPELLANT ACQUISITION

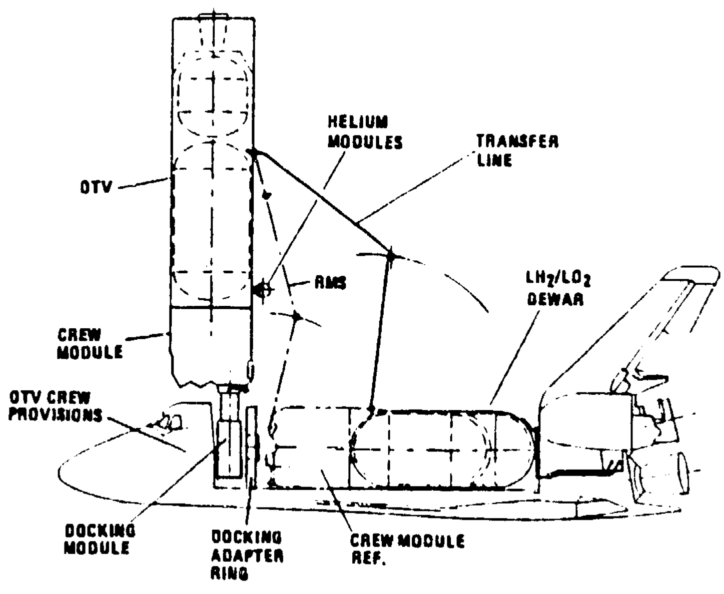


Figure 3-1. Orbiter tanker configuration

3.1.1 SELECTED POTV MISSION. Our current study of orbital propellant handling and storage systems (Contract NAS9-15640, Reference 3-1) has defined the mission operations sequence for a manned five day sortie mission to GEO employing a space-based, two-stage OTV. Figure 3-2 illustrates the major orbital operations required, including rendezvous and docking, propellant transfer, mating of OTV stages and crew module, orbit transfer to GEO, staging, operation at GEO, orbit transfer to LEO, and crew-module retrieval. This mission has been selected as a representative scenario for POTV refueling operations.

3.1.1.1 Timelines. Timelines were developed to determine the impact of various operations. The timeline for the total five-day manned GEO sortie mission is presented in Figure 3-3. This timeline is based on one Orbiter vehicle, two shifts (no weekends) for Orbiter processing, and three shifts for launch processing at the pad. The Orbiter will be committed for 47 days of which the major contributor (73 percent) is ground turnaround time.

The operations timeline for Orbiter flights 1 and 2 is presented in Figure 3-4. The total flight operations time is seen to take less than three days. The first working day of operations is the launch, rendezvous, docking, and IVA inspection and checkout of the POTV. (This also allows time for the crew to adjust to zero-g conditions before EVA is attempted.) The second day is dedicated to performing POTV inspection and maintenance tasks. A space-based POTV would be designed for conditioned monitored maintenance whereby any subsystem degradation or failure would be recognized beforehand and the appropriate module would be aboard the Orbiter for replacement by EVA or RMS. A nominal allowance of 6 hours EVA activity plus 4 1/2 hours pre- and post-EVA operations is considered appropriate for nominal inspection and maintenance tasks.

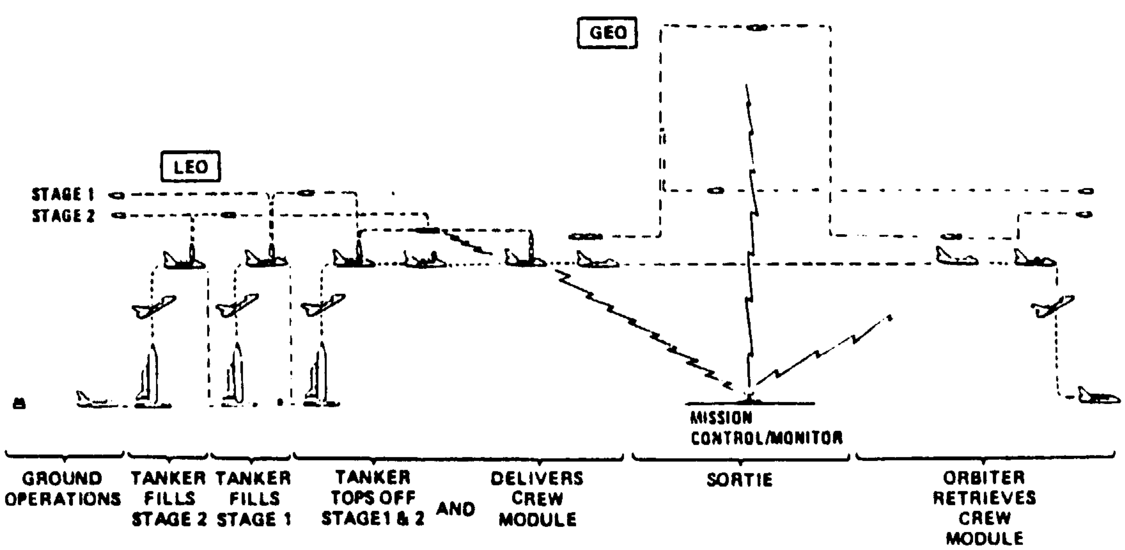
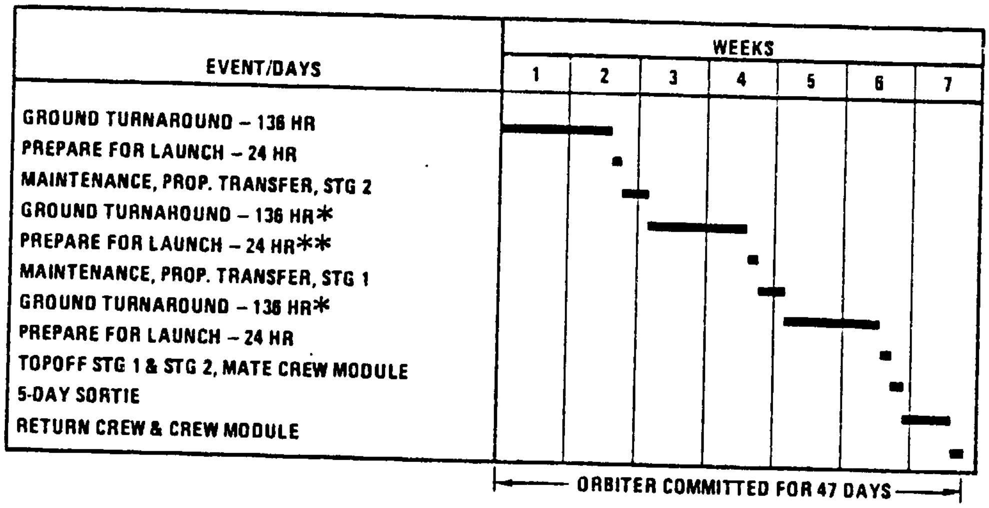
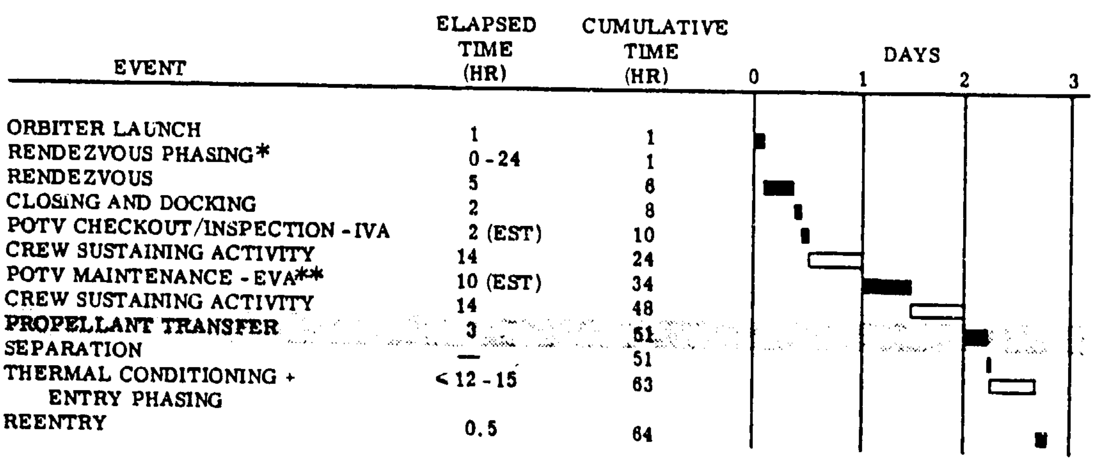


Figure 3-2. Operations for 5-day manned GEO sortie mission.



*2 SHIFTS; NO WEEKEND OPERATION
**3 SHIFTS FOR PAD PROCESSING

Figure 3-3. Timeline for 5-day manned GEO sortie mission (one orbiter tanker)



*ASSUMED ORBIT PHASED WITH LAUNCH SITE (31° INCLINATION, 478 km (258 N.MI.) ALTITUDE), OTHERWISE RENDEZVOUS PHASING IS 0 - 24 HR.

**NOMINAL ALLOWANCE FOR ROUTINE INSPECTION OF OTV WITH MMU AND REPLACEMENT OF MODULES REQUIRING MAINTENANCE (CONDITION MONITORED MAINTENANCE).

Figure 3-4. Tanker flight 1 and 2 operations timeline

It was estimated in Reference 3-1 that only three hours will be required for propellant transfer operations. Although this duration may not be correct, it is significant that propellant transfer may represent only 5 percent of the total flight operations timeline. It appears from Figure 3-4 that doubling this time will have virtually no impact upon the total flight operations. Thus, the capability for rapid propellant transfer became a minor element in this study.

3.1.2 ORBITER TANKER CONFIGURATION. The orbiter tanker selected for this scenario is the configuration defined in Reference 3-1.

The selected dewar pictured in Figure 3-5 features two separate propellant tanks equipped with hemispherical bulkheads. The LO_2 tank has a reversed bulkhead, so that the two tanks can be nested to reduce the overall length. The LH_2 tank is located forward and the LO_2 tank is positioned aft in the Orbiter payload bay. A single vacuum shell equipped with three girth rings and five intermediate stiffener rings encases the two tanks. The forward and aft girth rings serve as structural ties to the Shuttle, and all three girth rings provide support for the tanks. The primary structure for the vacuum shell will be aluminum alloy isogrid, semimonocoque, or a combination of both. Both tanks are suspended from the vacuum-shell girth rings, using low-conductive struts arranged in "V" patterns and oriented such that the load paths are directed tangentially into the aft bulkheads. For the LH_2 tank, these support struts are augmented with low conductive drag links located at the forward bulkhead. Multilayer insulation (MLI) blankets are applied over all surfaces of each tank.

FEATURES:

- GIRTH RINGS SERVE AS STRUCTURAL TIE BETWEEN TANK AND SHUTTLE.
- VACUUM JACKETED.
- LOW CONDUCTIVE TANK SUPPORT STRUTS PROVIDE THERMAL ISOLATION & PERMIT DIMENSIONAL CHANGES
- 10-PLY MLI -1.5 kg/HR (3 LB/HR) BOILOFF
- ACQUISITION SYSTEM DRY OUT PREVENTED DUE TO LOW HEAT LEAK.
- LENGTH = 10.4 m (410 INCHES)
- DEVELOPMENT COST ESTIMATE = \$73 M
- UNIT COST ESTIMATE = \$14 M

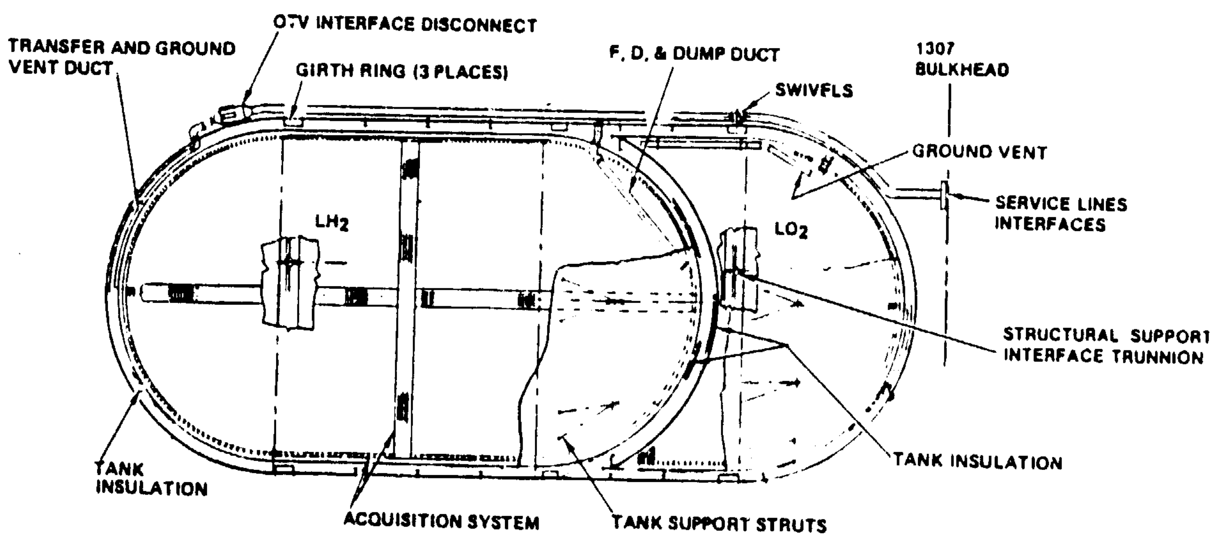
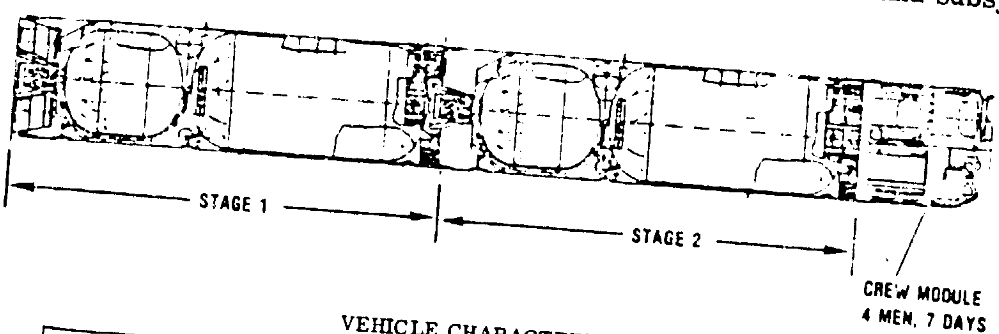


Figure 3-5. Features of an Orbiter tanker kit

Propellant acquisition is accomplished using a full-screen channel system, which maintains communication with propellants located anywhere in the tank without any special settling thrust. This concept was selected because it imposes no restraints upon Orbiter operations, since propellant transfer can be successfully performed for any orientation, even during maneuvers. There is a family of Orbiter interface lines including propellant ground fill and vent, nearly identical to a ground-based OTV. On top of the propellant supply tank are folded lines which can be extended to connect with OTV fill and vent ports.

3.1.3 POTV CONFIGURATION. The basic POTV configuration for performing the five day sortie-mission to GEO is given in Figure 3-6. Vehicle subsystems will be selected on the basis of mission and space-based requirements. A list of subsystems influenced by mission requirements will include tank size, pressurization system, propellant acquisition system, insulation system and vent system. Subsystems influenced by space-basing requirements will include insulation system and vent system. Rationale for subsystems selection influenced by mission requirements will be discussed first. This selection process will be followed by a description of a "typical" OTV mission for which potential problems can be identified and subsystems selected.



VEHICLE CHARACTERISTICS

PARAMETER	STAGE 1	STAGE 2
PHYSICAL		
Length m (ft)	16.0 (52.5)	16.3 (53.5)
Diameter m (ft)	4.5 (14.6)	4.5 (14.6)
Weight kg (lb)	5,992 (13,210) (dry) 59,662 (131,530) (wet)	6,899 (15,210) (dry) 63,227 (139,390) (wet)
MAIN PROPULSION		
Thrust kN (lb)	356 (80,000)	178 (40,000)
No. & Type Engines	(4) RL-10 CAT IV B	(2) RL-10 CAT IV B
I_{SP} (sec)	460	460
Mass Flow-LH_2 kg/sec (lb/sec)	11.2 (24.7)	5.6 (12.35)
Mass Flow-LO_2 kg/sec (lb-sec)	67.6 (149.0)	33.8 (74.5)
Total Impulse kg-sec (lb-sec)	24×10^6 (53×10^6)	24×10^6 (53×10^6)
TANKAGE		
LH_2 Capacity kg (lb)	7,484 (16,500)	7,484 (16,500)
LO_2 Capacity kg (lb)	44,906 (99,000)	44,906 (99,000)
Material	2219 Al	2219 Al
Insulation	MLI	MLI

Figure 3-6. A representative POTV was selected for this study

3.1.3.1 Subsystems Influenced by Mission Requirements.

Pressurization System - The selected pressurization system will require helium for propellant tank pre-pressurization for each main engine start. Pressurization requirements during main engine firing will be autogenous for the liquid hydrogen tank and helium for the liquid oxygen tank. Main engine-start helium usages will not be excessive because engine-start NPSP requirement will be approximately 3.45 kN/m^2 (0.5 psid) (LH_2 tank) and 6.9 kN/m^2 (1.0 psid) (LO_2 tank). Total mission helium usages will be relatively small for the LO_2 tank because helium will be bubbled through the liquid bulk. The tank pressure increase will be primarily due to propellant evaporation into the helium bubbles. Autogeneous pressurization was selected for the hydrogen tank because a) it is a simple and proven approach, and b) the alternative helium pressurization approach will be considerably heavier. This type of pressurization system was analyzed in contract NAS3-20092, Reference 3-2.

Helium within a propellant tank can complicate an orbital tanking procedure because of the need to expel most of the inert gas before propellant transfer can be initiated. Unfortunately, in the near-term, there is no viable alternative to helium pressurization for main engine start since main engine NPSP requirements must be satisfied. An advanced engine with "boot-strap" capability, i.e., with no NSPS requirements, may be developed in the future. A major benefit from this development will be a simplified refueling procedure. Until then, refueling operations must be capable of dealing with helium inside the propellant tanks.

Propellant Acquisition System - Analyses were performed in Contract NAS3-20092 to assess the benefits of a partial propellant acquisition system for OTV. The acquisition system combined with a thermal subcooler was analyzed to determine if these subsystems could replace helium pressurization and RCS subsystems. Although a final assessment has not been reported, it is likely that an acquisition system is not performance effective for a number of OTV missions. At this time, it is judged that a partial screen acquisition device will not be included as part of an OTV configuration. However, an exception to subsystem selection is made in this case and with the pressurization system, as explained below.

To provide a more thorough discussion and analysis of POTV orbital refueling operations, it was decided to include the helium pressurization and screen acquisition (start basket) subsystems. In this way the influence of each upon refueling techniques or procedures could be assessed.

Insulation System - A multilayer insulation (MLI) system was selected as being representative of thermal protection systems which may be employed for OTV. A single blanket consisting of twenty MLI layers was selected on the basis of a previous analysis conducted for Contract NAS3-20092. Radiative properties of the organically-coated aluminized Kapton Superfloc MLI will result in a maximum equilibrium temperature of 289K (~520R) (Figure 3-7) for the estimated α/ϵ of 0.3. This maximum temperature is based upon the worst case assumption of a tank surface continuously exposed to the sun. The lower temperature curve is for the assumed condition of a vehicle rotating at a rate sufficient to maintain uniform skin temperatures throughout.

The time required for the propellant tanks to attain equilibrium temperatures will be dependent upon the propellant mass remaining at POTV final MECO. These residuals must first boiloff and be vented overboard before the propellant tank temperatures can increase. Figure 3-8 gives the time required for residual liquids to boiloff as a function of initial liquid residual. Studies have not yet been conducted from which an acceptable liquid residual range can be determined. On the basis of Centaur vehicle flight experience, however, a residual of about 200 kg (441 lb) per tank appears reasonable. These quantities will boiloff in about 6 to 8 days. Beyond this time, the propellant tanks and remaining vapor will begin to increase in temperature as shown in Figures 3-9 and 3-10. The rate of temperature increase will depend upon the external shield temperature, which can be as high as 289K (520R), as indicated by Figure 3-7. Propellant tank transient time to steady state is given for three external temperatures to show how this transient will be influenced by vehicle roll-rate and α/ϵ. It is seen that liquid-oxygen tank equilibrium can be attained in a minimum of 8 to 12 days, and hydrogen tank equilibrium can occur in a minimum of 16 to 22 days.

Vent System - A thermodynamic vent-system will be required to provide vent capability for the proposed OTV mission profiles. This type of vent system can maintain vehicle tank pressure control in a zero-g environment regardless of fluid quality at the heat exchanger inlet. Vent system sizing will be keyed to the MLI thermal protection capability. Additional vent capability may be required to satisfy refueling procedures. Further discussion will be postponed until after space-basing requirements have been evaluated.

3.1.3.2 Subsystems Influenced by Space-Basing Requirements - Space-basing conditions are defined as those conditions affecting the OTV from post-mission storage of each stage until after resupply. The period where OTV and orbiter are docked is exempted, since it is part of the tanking duration. Any subsystem capability needed to maintain the OTV in a "safed" condition for subsequent refueling operations is considered to be a space-basing vehicle requirement. The insulation and vent systems selection will be influenced by space-basing considerations.

Insulation System -In addition to the mission requirements previously identified, the insulation system must provide thermal protection for propellants where multiple orbiter flights are needed to support a single OTV mission. For this scenario, it is likely that the OTV stages will reside in orbit for several weeks before tanking is complete. Too little insulation will result in excessive propellant boiloff prior to a mission. Ideally, a trade analysis should be conducted to balance mission versus space-based requirements. This study assumed twenty MLI layers was acceptable for both requirements.

NOTES:

1. α/ϵ = shield absorbitivity to emissivity ratio
   = 0.3 (expected value)
2. Curve 1 assumes vehicle roll rate that maintains uniform shield temperature
3. Curve 2 assumes that shield is continuously exposed to solar radiation

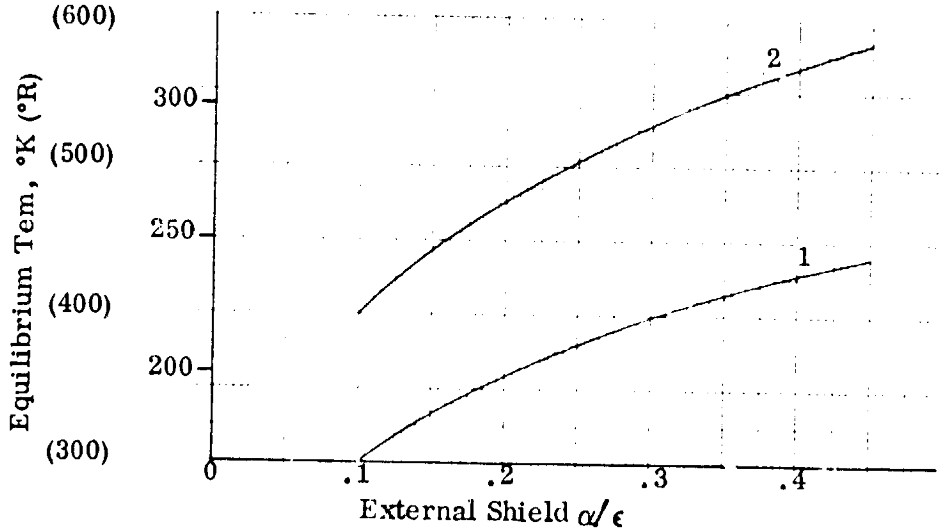


Figure 3-7. Influence of MLI external shield radiative properties and orientation upon propellant tank equilibrium temperature

NOTES:

1. Liquid and vapor initially saturated @ 103 kN/m^2 (15 psia)
2. 20 layers MLI
3. Propellant tank heating rates are: LH_2 tank = 0.18 kW (607 Btu/hr)
   LO_2 tank = 0.07 kW (232 Btu/hr)
4. Time includes ≈ 0.4 days for tank pressure to increase to 138 kN/m^2 (20) vent pressure

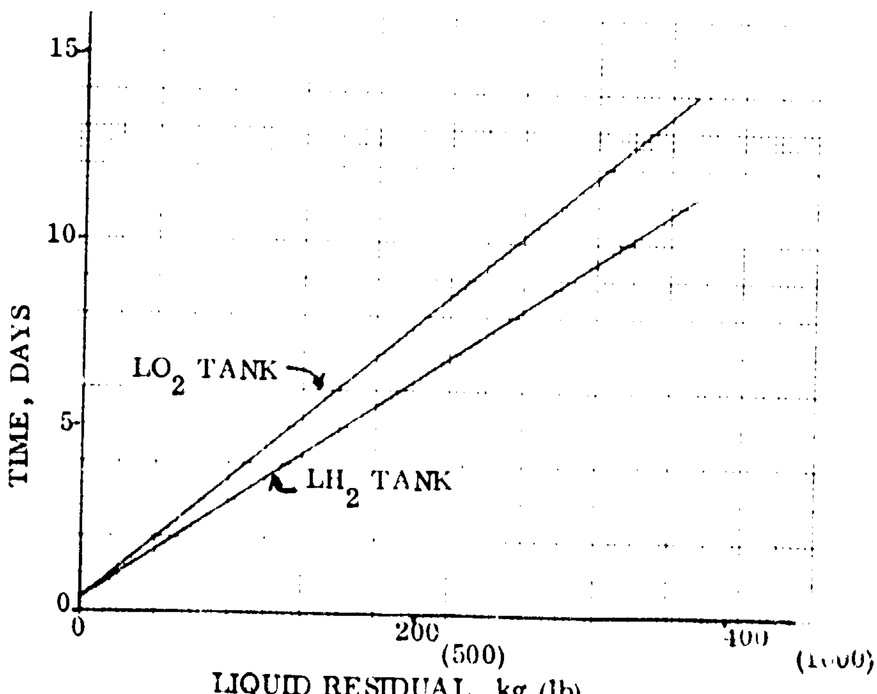


Figure 3-8. Time for POTV Liquid Residuals to Boiloff in LEO

NOTES:

1. 20 layer MLI
2. Propellant tank initially filled with vapor at: $P = 138\ kN/m^2$ (20 psia)
   $T = 111K$ (200 R)
3. Constant tank pressure maintained in crbit
4. Tank skin and ullage reside at same temperature
5. Expected external shield temperature = 278K (520R)

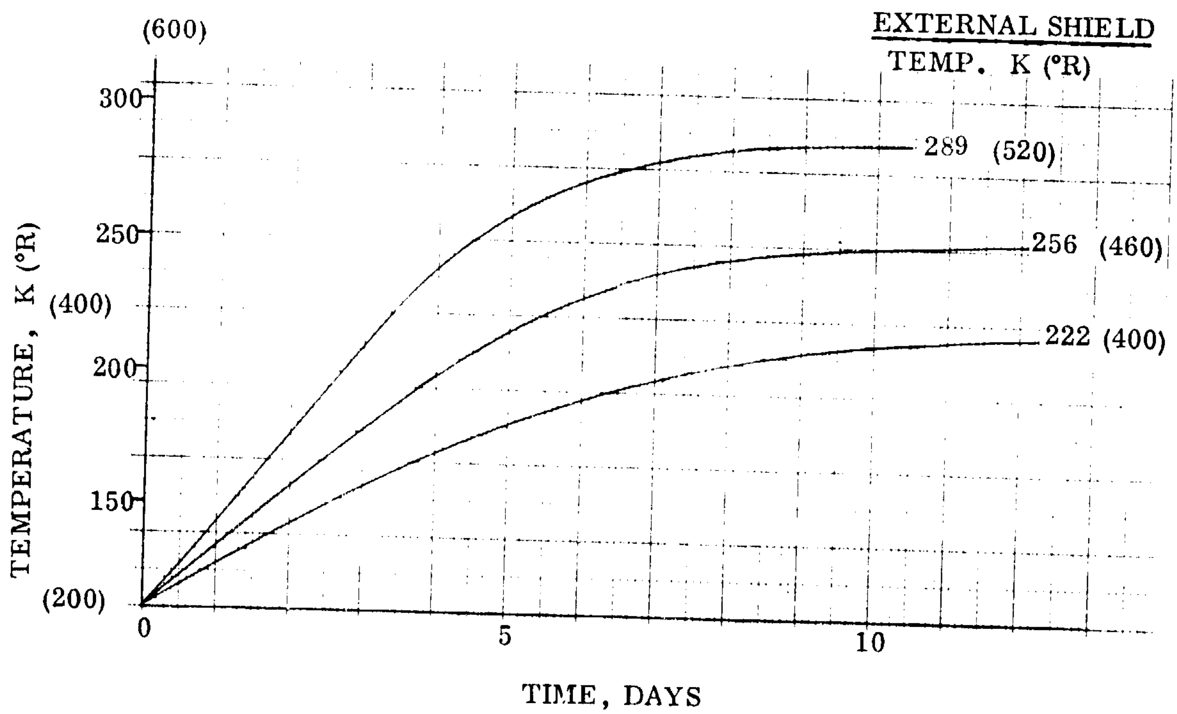


Figure 3-9. Transient time for POTV LO_2 tank to attain temperature equilibrium in LEO

NOTES:

1. 20 layers MLI
2. Propellant tank initially filled with vapor at: $P = 138\ kN/m^2$ (20 psia)
   $T = 26.7K$ (48 R)
3. Constant tank pressure maintained
4. Tank skin and ullage reside at same temperature
5. Expected external shield temperature = 289K (520R)

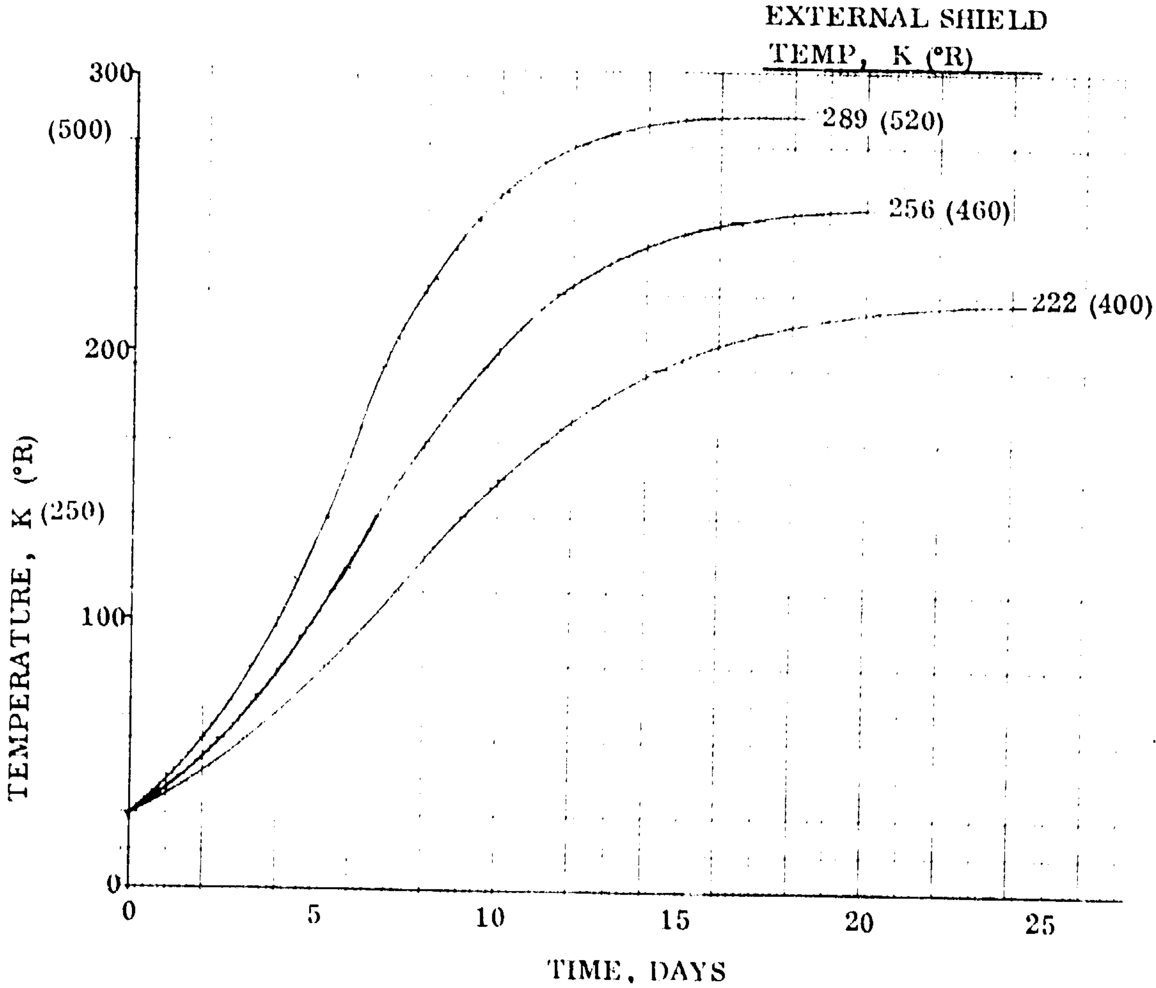


Figure 3-10. Transient time for POTV LH_2 tank to attain temperature equilibrium in LEO

Vent System - There may be an advantage in performing propellant tank blowdown to a low pressure prior to initiating orbital refill. This operation could be performed with the thermodynamic vent system except that its limited vent capability would result in an extremely long blowdown mode. The propellant mass vented during this orbital period is likely to be at least an order of magnitude greater than the mass vented during the mission. Furthermore, the long blowdown duration could delay rendezvous by several days. It appears, therefore, that a requirement exists for another system capable of vent rates at least an order of magnitude greater than those of the thermodynamic vent system. A non-propulsive vent system will be included in the vehicle configuration if required.

3.2 ORBITAL PROPELLANT RESUPPLY TECHNIQUES

Refueling operations of space-based OTVs, whether from an Orbiter tanker or space depot, will present obstacles not heretofore experienced by operational vehicles. The obstacles are: (1) the hostile space environment and (2) the limited resources available for space-based operations.

The space environment introduces three variables (zero-g, vacuum, and changing orbital conditions), all of which will complicate space-based operations. Liquid vapor distribution is not sufficiently well defined in space to enable pure vapor venting during the filling operation. Also, the cost of transporting propellants into space for refueling will be sufficiently high (>400$/kg) that two-phase venting is undesirable. The combination of zero-g and vacuum environments will greatly limit freedom of movement during operations. In addition, the variations of heating and lighting environments in low earth orbit will further complicate refueling operations.

The limited personnel, ancillary equipment, and instrumentation resources available in space serve as a major constraint for OTV refueling operations. The flight crew will perform operations in space that require numerous personnel on the ground and the operations will be more complicated. Personnel could be the most severe constraint because the equipment and instrumentation made available for space-based operations must be limited to that which can be monitored and performed by the flight crew. These constraints of limited resources plus the space environment lead to the minimum set of on-orbit refueling criteria given in Table 3-1.

3.2.1 ON-ORBIT RESUPPLY CONCEPTS.

Filling a vehicle on the ground can be easily accomplished because vapor is readily expelled as liquid is introduced. Even cooling a vehicle on the ground prior to cryogenic fill is a routine operation. However, low-g propellant fill cannot be satisfactorily accomplished with standard one-g techniques. This is because liquid-vapor distribution will not be well defined during the filling operation. Consequently, a routine ground operation such as venting can become a process requiring considerable care in orbit.

Table 3-1. A minimum set of on-orbit refueling criteria is required for acceptable operation

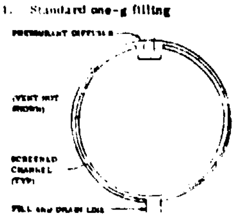
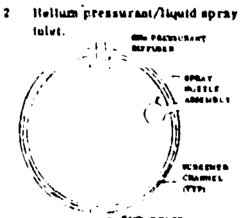
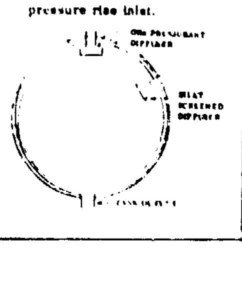
Flexibility	- This is an important ingredient required of refueling procedures because initial conditions may vary from one refueling operation to another. For example, the initial OTV temperature will be a function of time in orbit between refueling operations and liquid residuals at the end of the previous mission. It is conceivable that propellants may reside within the tanks as one extreme condition; the other extreme would be that of vapor-only inside warm propellant tanks. Tank temperature could have a profound influence on the transfer process.
Simplicity	- Limited resources demand that a simple and straightforward procedure be devised. The few personnel available must be capable of connecting and disconnecting transfer lines and monitoring systems to avoid supply tank propellant depletion, receiver tank overfill, or overpressure. We must be able to rely on a limited number of measurements to describe propellant transfer conditions adequately.
Safety	- Operations must be selected to eliminate any concern for tank over-pressure, and mixing of hydrogen and oxygen in a confined area must be avoided.
Precision	- Some degree of precision will be required in this operation to support vehicle missions. Propellant tank pressure, temperatures and tank masses must be known with reasonable accuracy. However, propellant temperatures and pressures can be heavily influenced by the initial OTV thermal condition.
Efficiency	- An efficient propellant-management system is necessary because of the high transporation costs of propellant delivery to orbit.

Propellant fill is further complicated by one of two conditions that may exist: (1) in empty storage tank residing at a substantially higher initial temperature than the cryogen prior to initial fill or (2) a partially full tank requiring that helium pressurant be vented before refill can be initiated. Two questions arise. First, how do we fill the storage tank without expending excessive propellants in the process and without exceeding structural allowable storage tank pressure? Second, how can we vent helium without losing liquid overboard in the process? A resupply concept must satisfactorily handle these conditions in addition to satisfying the requirements and constraints previously identified.

A comprehensive screening of potential resupply concepts was conducted on a previous study, Filling of Orbital Fluid Management Systems, NASA CR159404, Reference 3-3, to identify methods of refilling small-scale propellant-management systems in space.

Systems considered include those using pressurant to condense vapor, valving arrangements, pumping, capillary pumping, use of thermodynamic venting, vacuum refilling, inflow baffling, high pressure manifolding, shaping of channels, and propellant depot refilling stations. These concepts were developed into the most reasonable, or most likely, candidates for orbital refilling. The descriptions and comparisons made in that study are applicable to a wide range of receiver applications, including OTVs. Table 3-2 is an example of the data developed showing candidates, their operation, advantages and disadvantages, and other comments.

Table 3-2. Typical example of a resupply concept screening procedure

CANDIDATE CONCEPTS	METHOD OF OPERATION	ADVANTAGES	DISADVANTAGES
1. Standard one-g filling 	Fill using normal ground filling inlet.	No additional hardware.	Liquid vapor interface unknown during filling. Vapor trapped in device.
2. Helium pressurant/liquid spray inlet. 	Fill tank with liquid using spray nozzle to achieve thermodynamic equilibrium during chilldown and fill. Use helium to condense any vapor trapped in the capillary device during filling. Helium could be added just before transfer to avoid any possibility of forming vapor in the capillary device after pressurization.	Fairly simple hardware. Does not depend upon position of the low-g interface and therefore can be tested in normal gravity with assurance that it will operate in low gravity. Thermal equilibrium results in low pressure rise during fill. No capillary device hardware modification required. Could be retrofitted to existing systems.	High pressure rise will occur during chilldown which may require venting to occur. Use of helium makes filling a partially full tank more complicated because the helium will probably have to be vented off to keep final pressures within limits. Blowdown and purge required to replace GHe with GH_2 before fill of a tank containing only vapor.
3. Helium pressurant/low pressure rise inlet. 	Fill tank with diffuser to achieve low pressure rise during chilldown. Use helium to condense any vapor trapped in the screen device.	Fairly simple hardware. Low pressure rise and therefore best chance of not venting during chilldown. No capillary device hardware mods required. Could be "retrofitted" to existing systems.	Nonequilibrium conditions during fill, therefore high pressure rise. Possibility of vapor formation after helium addition if mixing and therefore pressure decay occurs. Same disadvantages as Concept 2 with respect to GHe removal.

Each of the candidate concepts from the study was compared on the basis of the ten criteria shown in the comparison chart (Table 3-3). The evaluation was performed using hydrogen as the baseline fluid; however, the results are generally applicable to other cryogenic fluids with both condensible and noncondensible pressurant. A more detailed discussion of the concept evaluation can be found in Reference 3-3.

The selected concept (#2) for refilling small scale systems includes filling the tank through a spray nozzle to maintain pressure control, and employing helium pressurant to re-condense vapor trapped within a screen acquisition device. This was one of two concepts evaluated for POTV. The second concept assumed that propellant was introduced through a large diffuser in an effort to achieve extremely low entering velocities. An assessment of the first concept is given in Section 3.3.2. The second concept, which was found to be inadequate, is discussed in Section 3.4.

Table 3-3. Propellant tank refill with liquid spray was previously identified as a viable concept

Key: Highest - 10, Lowest - 0 / Concept and Number (From Table 1) / Criteria	Liquid Spray Inlet/Helium Pressurant	Low Pressure Rise Inlet/Helium Pressurant	Low Pressure Rise and High Chilldown Efficiency Inlets/Helium Pressurant	Autogenous Pressurization to Condense Vapor	Piping Inside Channels to Force Vapor Out of the Channels	Propellant Depot Approach	Thermodynamic Vent or Vapor Cooled Shield Inlet to Remove Fluid From Channels	Open Loop Refrigeration to Condense Vapor	Internal Vanes to Direct Flow	Shaping Capillary Devices to Position and Eject Vapor	Standard One G Filling	Propellant Control Surfaces Vent Tube
	2	3	4	9	6	8	7	12	10	11	1	5
Technical Probability of Successful Operation	10	10	10	7	7	10	7	8	7	7	0	0
Adaptability to Existing Configurations	10	10	10	10	2	10	7	4	4	2		
Operational Simplicity	8	8	6	6	10	1	8	6	9	6		
Versatility	10	10	10	10	10	10	7	7	5	4		
Gravity Insensitivity	9	7	7	6	10	7	3	6	4	1		
Ground Testability	10	7	7	6	10	7	3	6	4	1		
State of Development	9	8	7	5	6	3	5	6	5	6		
Cost Effectiveness/Recurring	10	9	9	8	5	9	8	6	7	6		
Weight/Hardware	9	8	7	9	2	1	7	4	5	5		
Weight/Vented Fluid	10	9	10	10	4	10	7	3	3	4		
Reliability/Safety	10	10	9	9	10	7	9	8	10	10		
Total	105	96	92	86	76	75	71	64	63	52	N/A	N/A

3.3 SELECTED ORBITAL RESUPPLY METHOD

For initial filling, the POTV is expected to reside at a temperature of about 289K (520R). This will be the approximate equilibrium temperature dictated by the multilayer insulation (MLI) radiation properties (Figure 3-7). Vehicle equilibrium conditions are also expected to occur during the approximate six- to eight-week period between the end of one mission and the beginning of refueling operations for another mission. Thus, propellant boiloff could complicate propellant transfer operations. The key issues to be resolved are: 1) how to fill the tanks without expending excessive propellants in the process and without exceeding structural allowable receiver tank pressure, and 2) how to assure that the start basket (if one is present) will be free of vapor at end of propellant tank refill. The approach selected for the POTV appears to resolve these issues. The following steps serve as the primary elements of an acceptable propellant transfer procedure: initial vent, prechill, fill. Each element will be analyzed in detail.

A recommended refueling procedure will be developed for a POTV having each of the following pressurization system-start basket combinations:

a) no helium pressurization - no start basket

b) no helium pressurization - start basket

c) helium pressurization - no start basket

d) helium pressurization - start basket

3.3.1 <u>INITIAL VENT.</u> Propellant tank venting is required whenever it will simplify the prechill and tank-fill operations. There are two occasions when venting is either desirable or mandatory; when helium is to be expelled prior to the refueling operation, and to reduce peak pressures that occur during prechill.

3.3.1.1 <u>Propellant Tank Helium Dilution</u>. Refueling of the empty or near-empty vehicle propellant tanks will be performed prior to a scheduled mission. It is expected that the propellant tanks will be refueled to the 95-97 percent level. Because helium is non-condensible, that quantity in the tank at the start of fill will also be present at the end of tanking. Consequently, the large ullage volume reduction experienced during refill can substantially increase helium partial pressure.

This fact is illustrated by Figure 3-11 which shows that as little as 0.5 kg (1.1 lb) helium in the LO_2 tank can provide a helium partial pressure in excess of 69 kN/m^2 (10 psia). A partial pressure of this magnitude is considered to be unacceptably high. It is estimated that a helium pressure no greater than 20.7 kN/m^2 (3 psia) would be acceptable. This assessment is based upon the need to know liquid-vapor-pressure conditions at the end of propellant tanking. Since only tank pressure can be measured, an uncertainty in the residual helium quantity directly affects our knowledge of propellant vapor pressure.

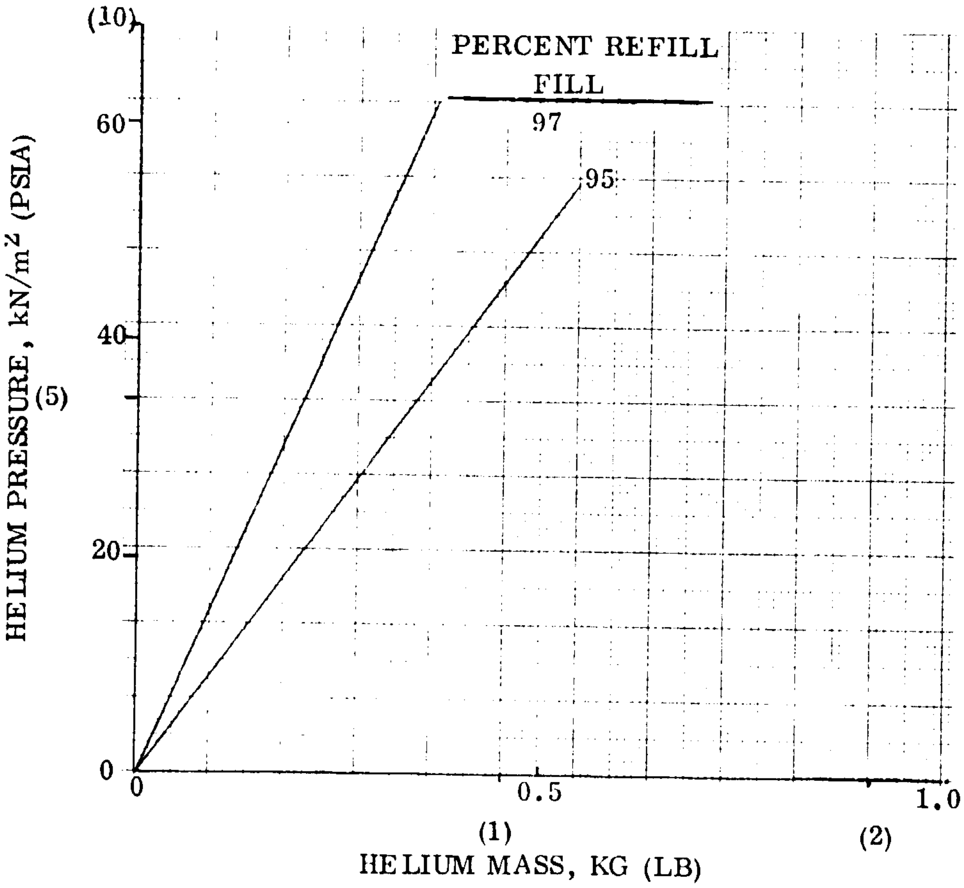


Figure 3-11. Liquid oxygen tank helium partial pressure following refueling operation.

Another important reason for expelling helium from the propellant tanks is the need to maintain relatively low tank pressures during the refueling operation. As a guide, propellant vapor pressures could be maintained between 103 kN/m^2 (15 psia) and 138 kN/m^2 (20 psia), with a maximum allowable tank pressure of about 172 kN/m^2 (25 psia). Should pressure variations of about 13.8 kN/m^2 (2 psi) be selected as a contingency during fill, then helium partial pressure should not exceed 20.7 kN/m^2 (3 psia). This partial pressure will convert to a maximum allowable helium mass of 0.13 kg (0.28 lb) at initiation of oxygen tank fill (Figure 3-11). Considerably more helium will be acceptable in the liquid hydrogen tank; in excess of 1.36 kg (3 lb) according to Figure 3-12.

Helium residuals at MECO were estimated from work performed on Contract NAS3-20092, which are given in Table 3-4. It is clear that the hydrogen tank does not have to be vented to satisfy the previously expressed propellant tank pressure criteria, whereas considerable oxygen tank venting is required. Two tank blowdowns are needed (Figure 3-13) to reduce the residual helium quantity to an acceptable level. Several hours may be required between vent periods to allow an oxygen tank pressure increase to the level indicated in Figure 3-13.

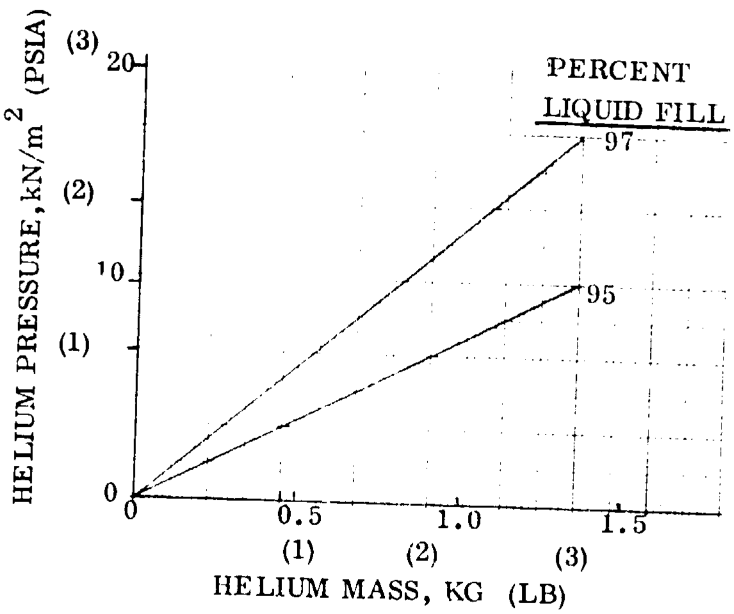


Figure 3-12. Liquid hydrogen tank helium partial pressure following refueling operation

Table 3-4. Helium expended for pressurization of dual stage POTV

	Total Helium To LO_2 Tank, kg (lb)		Total Helium To LH_2 Tank, kg (lb)	
1st Stage	1.75	(3.86)	0.89	(1.97)
2nd Stage	1.66	(3.65)	1.44	(3.18)

Note: Helium quantities extracted from Tables 5 and 6, Reference 5.

3.3.1.2 <u>Peak Pressure Reduction</u>. Receiver tank prechill which follows the initial vent period must be conducted with care in order to avoid an excessive tank pressure buildup. Tank pressure during prechill will be influenced by several factors, including initial pressure at prechill initiation. Thus peak pressures can be controlled, in part, by first venting the tank before initiating prechill. The advantages of this vent process will be quantified in the next section.

3.3.2 <u>RECEIVER TANK PRECHILL</u>. Prechill is required whenever initial tank temperature is such that the stored energy will result in excessive pressure during the tank fill mode. Prechill is accomplished by introducing liquid into the propellant tank at a velocity that provides good heat exchange between the high temperature walls and the cooling fluid. This procedure has the advantage of requiring little mass to effect tank cooling.

NOTES:

1. Initial Blowdown Conditions

   $P = 103\ kN/m^2$ (15 psia)
   $T = 111K$ (200 R)
   Helium Mass = 2.75 kg (3.86 lb)
   Vent Area = 12.9 cm^2 (2 in^2)

2. Helium Mass, kg (lb):

   at end 1st blowdown = .244 (.56)
   at end 2nd blowdown = .089 (.204)

3. It was assumed that ullage temperature increased to 222K (400R) between blowdowns. This could require several hours of heat exchange betwee[illegible] ık wall and ullage.

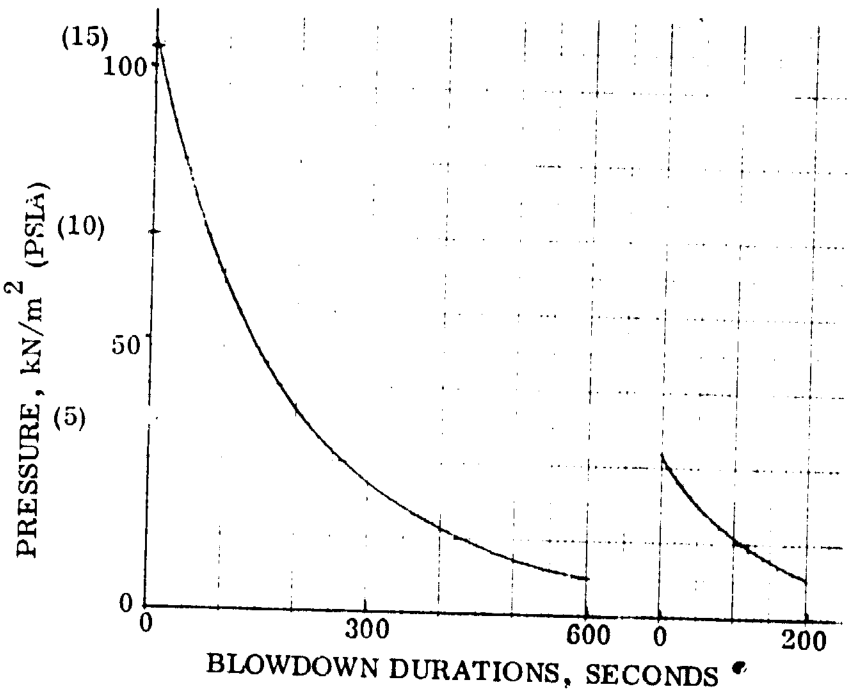


Figure 3-13. Oxygen tank blowdown for helium expulsion

The primary requirement for system prechill is to reduce tank temperatures sufficiently that the fill process will be accomplished with a locked up tank. Venting is unacceptable during the fill mode because of the possiblity that an unknown quantity of liquid will be lost overboard, since propellant control cannot be maintained during this process. Venting is acceptable during prechill, however, because the elevated tank temperatures will quickly evaporate all liquid propellant during this phase.

The key factor in determining prechill requirements is the theoretical maximum pressure that can occur during tank fill as a function of initial tank temperature. Maximum tank pressure, for a given mass addition, occurs when the tank vapor and tank wall reside at the same temperature. The analytical development for theoretical maximum pressure is given below.

The First Law expression for introducing liquid into a container is

$$dE_g + dE_w = h_L dm_L \tag{3-1}$$

$$dE_g = (u_g m_g)_2 - (u_g m_g)_1 \quad = \text{change in ullage energy}$$

If one assumes that the tank is initially evacuated

$$m_{g_1} = 0 \text{ and } dE_g = u_{g_2} m_{g_2} \tag{3-2}$$

Also for an initially evacuated container, $m_{g_2} = dm_L$ (3-3)

$$dE_w = (u_w m_w)_2 - (u_w m_w)_1 \quad = \text{change in tank wall energy}$$

Since tank mass is constant, $dE_w = (u_{w_2} - u_{w_1})\, m_w$ (3-4)

Combining Equations 3-1 through 3-4

$$u_{g_2} m_{g_2} + (u_{w_2} \quad u_{w_1})\, m_w = h_L m_{g2} \tag{3-5}$$

$$(u_{g_2} - h_L)\, m_{g_2} = (u_{w_1} - u_{w_2})\, m_w \tag{3-6}$$

Finally

$$m_{g_2} = (u_{w_1} - u_{w_2})\, m_w/(u_{g_2} - h_L) \tag{3-7}$$

where

u_{w_2} and u_{g_2} are evaluated at T_2

dEg = change in tank wall internal energy

h_L = enthalpy of liquid entering tank

dm_L = differential liquid mass addition to tank

u_g = internal energy of vapor in tank

m_g = mass of vapor in tank

u_w = internal energy of tank wall

m_w = tank wall mass

T = temperature

subscript

1 = conditions at beginning of interval

2 = conditions at end of interval

From the equation of state, gas pressure is

$$P_{g_2} = \left(\frac{m\ ZRT}{V_t}\right)_{g_2} \tag{3-8}$$

where

Z = compressibility factor

R = gas constant

V_t = tank volume

P_{g_2} = gas pressure

The theoretical maximum tank pressures during prechill, as described by equations 3-7 and 3-8 are plotted in Figure 3-14 as a function of initial tank temperature for the POTV. Of particular significance is the conclusion that LO_2 tank overpressure will not occur at any time during tank chill. LO_2 MLI radiative properties should maintain tank equilibrium temperatures below 289 K (520°R). This condition will result in a maximum tank pressure of 138 kN/m^2 (20 psia) which is well below the maximum allowable of about 345 kN/m^2 (5 · psia). For the hydrogen tank, however, the maximum allowable pressure of about 193 kN/m^2 (28 psia) dictates that the propellant tank be prechilled to a temperature less than 236 K (425°R). It is believed that a maximum pressure less than 138 kN/m^2 (20 psia) is acceptable for this phase of the operation. Therefore, the LH_2 tank will be prechilled to a temperature of about 200 K (360°R).

The following charge and vent procedure was selected for LH_2 tank prechill:

a. Meter LH_2 into the tank at a high velocity to provide good heat exchange with the walls.

b. Allow time for a tank pressure increase to 69 kN/m^2 (10 psia) (vapor temperature equals tank temperature at this time)

c. Vent the tank to near zero pressure and repeat steps a and b as required to reduce tank temperature below 200K (360 °R)

There are several questions that can be asked about the selected prechill procedure. These are:

1. How can we be certain that tank over-pressure will not occur during prechill?
2. How can we analytically model a complicated process that includes liquid boiling at hot tank walls as a result of jet or spray impingement?
3. How can we be certain that liquid will not be present when tank venting is initiated?
4. How will we know when the propellant tank has been prechilled below 200K (360R)

Acceptable procedures or processes are described in the following discussion which satisfactorily answers these questions.

3.3.2.1 <u>Tank Over-Pressure.</u> First, tank over-pressure will be prevented by controlling propellant flow into the tank. Accuracy does not appear to be a critical item as Figure 3-15 indicates. For example, the initial LH_2 charge will require about 9.1 kg (20 lbm); this will create a peak pressure of about 69 kN/m^2 (10 psia). If 18.2 kg (40 lbm) of LH_2 is inadvertently introduced, peak tank pressure will be about 124 kN/m^2 (18 psia), which is well below the tank allowable of about 172 kN/m^2 (25 psia). The data of Figure 3-15 was obtained by solving equations 3-7 and 3-8.

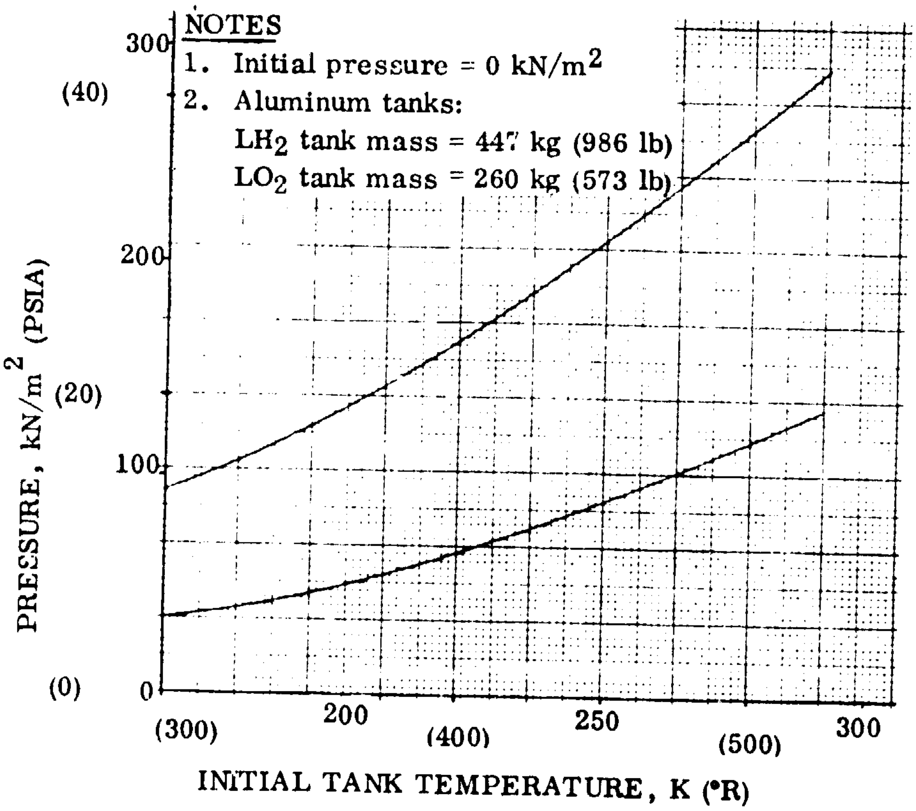


Figure 3-14. POTV LH_2 tank pressures could exceed tank allowables during prechill

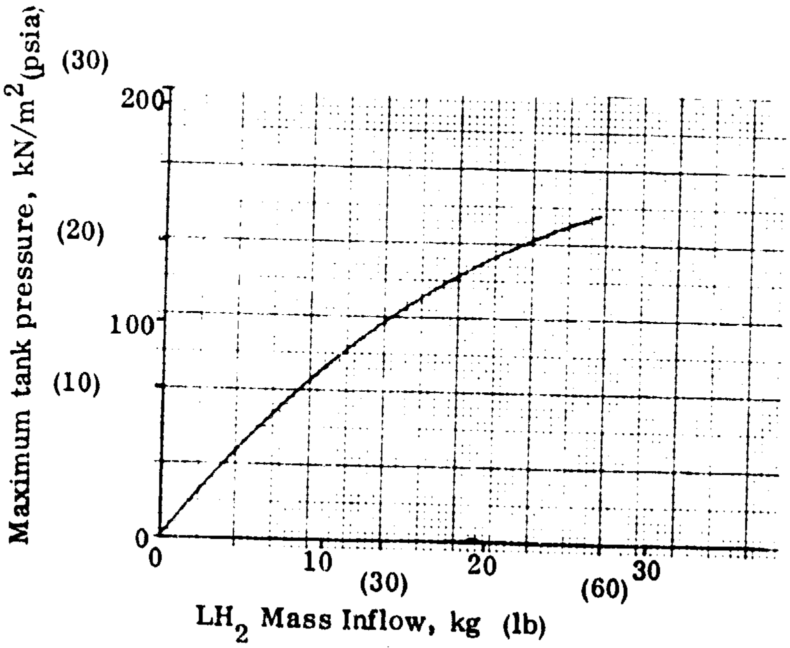


Figure 3-15. Precision metering of LH_2 is not needed to avoid over-pressure during prechill

3.3.2.2 Modelling. An important consideration in selecting a refueling technique is whether it can be subjected to model-scaling. A concern is the early prechill period that will be characterized by a complex thermodynamic and fluid mechanic process due to liquid impingement on the hot tank walls. The resulting forced-convection-nucleate and film-boiling-phenomena are extremely difficult to analytically model or scale. Fortunately, the complicated wall-boiling process can be resolved by sidestepping the issue. The wall-boiling-phenomenon need not be a critical part of prechill because peak pressures will not occur during this period. This is illustrated by Figure 3-16, which indicates that peak pressures should occur long after the LH_2 has evaporated and the vapor temperature increased to wall temperature. Peak pressures will occur only at maximum gas temperatures. This knowledge, plus the fact that tank temperature will be at about 200K (360 R) at prechill temperature, should greatly diminish the possibility that liquid will be present at vent initiation. Another factor to consider is that the heat exchange process during the limited boiling period represents only about ten percent of the total energy removed during prechill. This is additional support for the belief that the initial transient boiling period is not as important to the understanding of prechill as are the latter stages of this process.

3.3.2.3 Liquid Venting. Liquid venting will not occur during prechill-vent period because only vapor will be in the tank at vent initiation. This point is illustrated with a review of the selected prechill procedure (and referring to Figure 3-16):

1. About 9.1 kg (20 lb) LH_2 will be metered into the tank at a high velocity.
2. The peak pressure resulting from this mass addition will be about 69 kN/m^2 (10 psia).

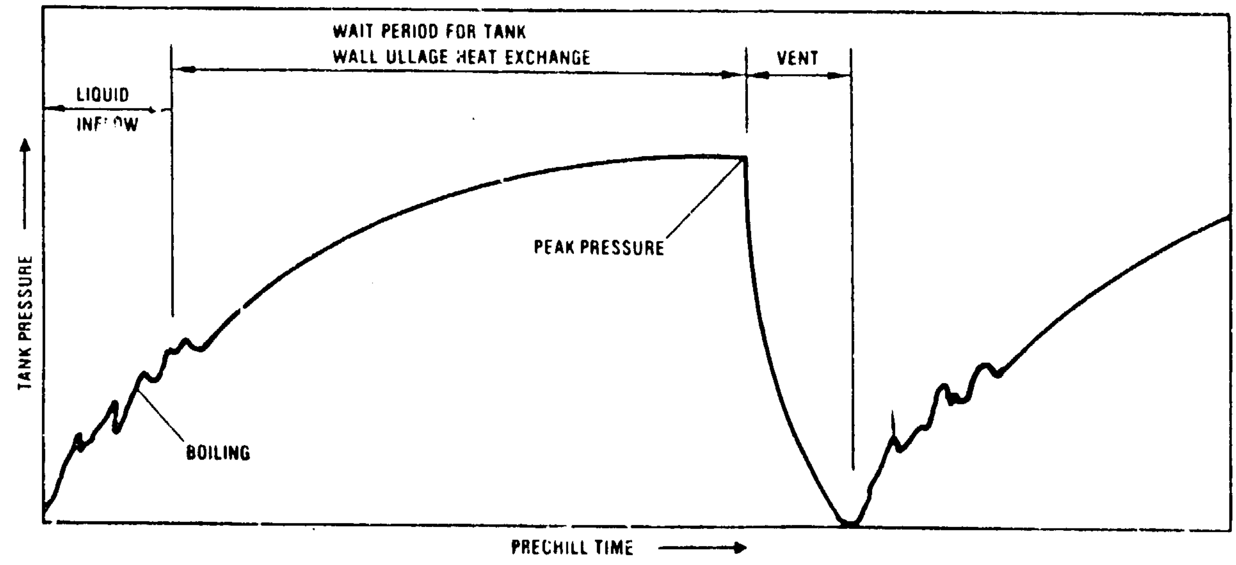


Figure 3-16. A prechill procedure can be identified to eliminate excessive tank pressures due to wall boiling

3. Venting will be initiated as tank pressure peaks out (i.e., pressure rise-rate approaches zero). This should coincide with ullage and tank wall temperatures approaching the same value.

4. If liquid had been present in the tank prior to vent initiation, it is likely that

   a. boiling would then occur due to the average wall temperature being greater than 200 K (360 R), and

   b. tank pressure would not be leveling off due to boiloff.

5. It is highly unlikely that liquid hydrogen can reside in the tank for more than a short time interval, if the average tank temperature is 178K (320 R) greater than liquid temperature.

3.3.2.4 Terminating Prechill. One solution to the problem of determining propellant tank temperature at prechill termination is to monitor a large number of temperatures during this process. Many measurements will be needed because the tanks will not be prechilled at a uniform rate. It is likely that the various measurements could be integrated, with the aid of a computer, to arrive at an average propellant tank temperature. This approach may be unacceptable if telemetry requirements become excessive, or if the transient conditions yield an inaccurate average vehicle temperature.

A potentially useful method is to use the accumulated tank pressure increase during the charge and vent process for determining how much energy has been removed from the propellant tank.

The tank wall energy transferred to the propellant during prechill is approximately proportional to the tank pressure increase. Rearranging equation 3-7 provides the following relationship:

$$(uw_1 - uw_2)\, mw = mg_2\, (ug_2 - h_L) \tag{3-9}$$

and

$$mg_2 = \rho g_2\, V_T \tag{3-10}$$

ρ_{g_2} = propellant tank vapor density

NOTE: This derivation assumes that the tank is initially evacuated.

Equation (3-9) shows that tank-wall energy-removal is a function of final vapor mass and vapor internal energy. However, both mass and internal energy are a function of final temperature and pressure. Consequently, tank energy removal can be related to these variables. This relationship is shown by Figure 3-17. There are two significant points to be made from this figure. First, energy removal is approximately independent of vapor temperature. Second, the energy removal is directly proportional to final tank pressure. It is concluded that the energy extracted during prechill can be approximated from the tank pressure increase experienced during the process, even if gas temperature is not known. The gas temperature uncertainty will result in an uncertainty in propellant vapor generated during prechill.

More propellant will be evaporated than the theoretical minimum required for tank prechill. This is true because gas temperature will be less than tank temperature when venting occurs. Fortunately, the additional vapor that may be vented does not appear to be excessive, as indicated by Figure 3-18. This figure indicates that even if the tank-to-gas temperature difference is as great as 50 K (90°R), hydrogen losses will be increased by only 5.4 kg (12 lb) per POTV stage, a small quantity compared to other losses that will be experienced during refueling operations.

Propellant transporation inefficiencies to be experienced during a POTV orbital refueling operation were determined during the Reference 1-1 study, and are given in Table 3-5. These inefficiencies were calculated for the orbiter-tanker configuration (Figure 3-1) in support of the five-day mission sortie. The listed prechill losses are two times greater than the theoretical minimum values. Even so, these quantities are insignificant and, as a result, it is concluded that a prechill procedure should not be selected on the basis of minimum fluid losses.

3.3.2.5 Prechill Analysis. A prechill subroutine, developed with IRAD funds, was employed to evaluate the details of a POTV prechill process. This subroutine which details are documented in Reference 3-3, was used to analyze fluid management system prechill for that study. The computer program is capable of evaluating prechill for the condition of vapor entry to the propellant tank, but not liquid entry. This limitation does not represent a handicap because, as explained in Section 3.3.2.2, liquid entry will have only a minimal influence upon the process.

The key to this analysis is in identifying the appropriate heat transfer coefficient between incoming vapor and the tank walls. Conventional forced convection expressions for flow over a flat plate or for jet impingement upon a surface, represent configurations that are too dissimilar to be applicable. Instead, a correlation developed for industrial mixing processes was selected as being representative of the heat exchange mechanism that will occur when vapor is continuously introduced into the propellant tank. The heat transfer correlation and required modifications are given as

$$\frac{h}{\rho c_p}\left(N_{PR}\right)^{2/3} = 0.13\left(\frac{(P_i/V)\,\mu}{\rho^2}\right)^{1/4} \quad \text{(Reference 3-3)} \qquad (3\text{-}11)$$

1. POTV Tank Vol. = 116.1 m^3 (4100 ft^2)

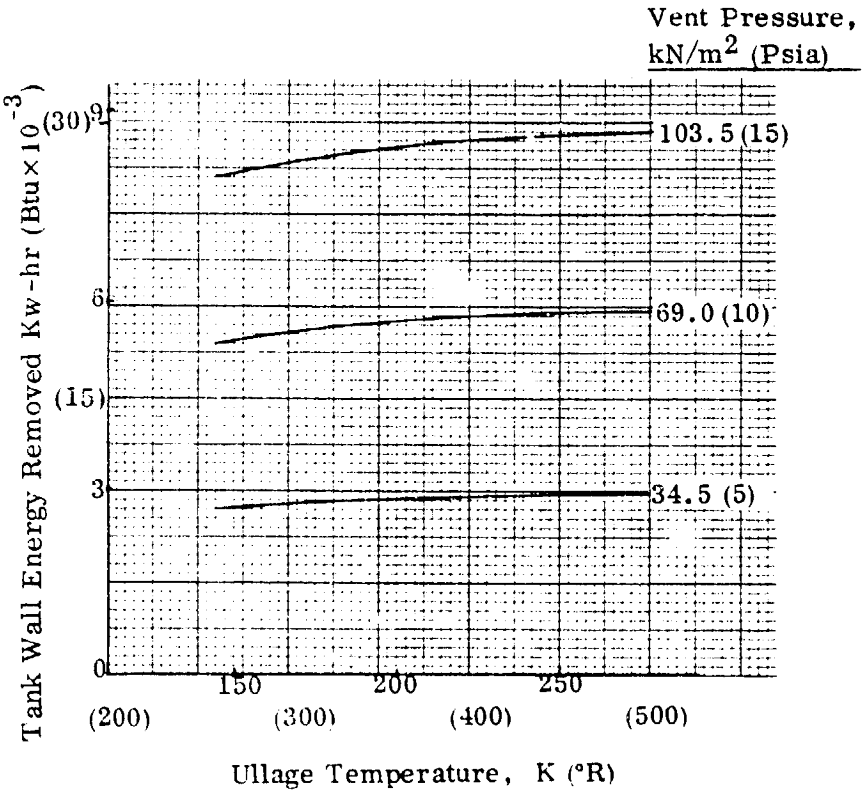
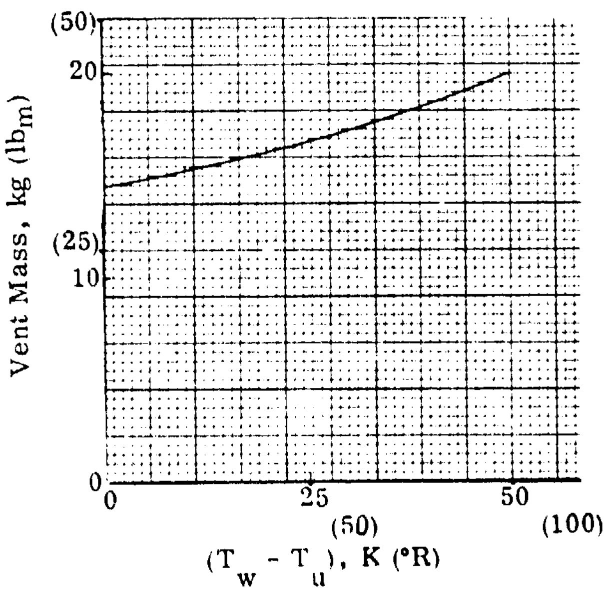


Figure 3-17. Tank Pressure Increases Will Yield Total Energy Removal During Prechill



1. Vent mass is based upon POTV tank prechill to 200 K (360°R).
2. T_w = tank wall temperature at vent initiation.
3. T_u = ullage temperature at vent initiation.

Figure 3-18. Hydrogen Tank Prechill Vent Mass Is Not Excessive Even at Large Tank Wall to Ullage Temperature Differences

Table 3-5. Propellant transportation efficiency for on-orbit resupply of dual stage POTV.

	Inefficiency*, percent per Shuttle payload	
Tanker Kit		
Orbiter support	4.36	
Supply tanks	6.29	
Pressurization and transfer	2.63	
Liquid residuals	1.90	*Shuttle P/L capability = 45,454 kg
Vapor residuals	0.50	
Subtotal	15.68	**Dual stage POTV total boiloff losses are amortized over three Shuttle flights.
Transfer line chill	0.10	
POTV tanks prechill (tanks initially warm)	0.28	
POTV boiloff** assume 29 days wait	0.74	
TOTAL	16.80	
Overall efficiency	83%	

where

h = heat transfer coefficient

ρ = fluid density

c_p = constant pressure heat capacity

N_{PR} = Prandtl number

P_i = mixer input power

V = tank volume

μ = fluid viscosity

Equation 3-11 was developed for liquids contained in cylinders. These liquids were continuously agitated with a mixing unit. Mixer input power was responsible for fluid agitation and is one of the variables of Equation 3-11. It is believed that fluid agitation during prechill will be the same (for equivalent power conditions) whether a mixer or fluid inflow is responsible. Since power output rather than power input will influence fluid agitation, equivalence will be between fluid power input and mixer power output.

This results in

$$P_i(EFF) = P_o = \dot{m}v^2 \tag{3-12}$$

where

P_o = mixer power output

EFF = mixer efficiency (conservatively assumed as 40 percent for this study)

m = entering mass flow rate

v = entering fluid velocity

$\dot{m}v^2$ = fluid power input

Substituting Equation 3-12 into 3-11 results in

$$\frac{h}{c_p} (N_{PR})^{2/3} = 0.163 \left(\frac{(m\, v^2/V)\, \mu}{\rho^2} \right)^{1/4} \tag{3-13}$$

Equation 3-13 indicates that heat transfer to the tank walls can be controlled by varying entering flowrate and velocity.

The following charge and vent procedure was selected for this POTV prechill analysis:

1. Charge the tank at a known vapor flowrate until the difference between wall and gas temperature has reached a specified value; 10 K (18°F) was the selected ΔT.
2. Vent the tank to a pre-determined low pressure. A reasonable level was selected as 6.89 kN/m^2 (1.0 psia).
3. Charge and vent the tank as required to reduce tank temperature to the predetermined level.

Figures 3-19 and 3-20 give tank ullage pressure and wall temperature histories during the prechill period. These curves are based upon adding 9.08 kg (20 lb) hydrogen at 0.91 kg/sec (2.0 lb/sec) during the charge period. Note that the prechill, which includes two charge periods and one vent period, will be about 206 seconds in duration. Peak pressure for this procedure will not exceed 78.6 kN/m^2 (11.4 psia).

The influence of key prechill variables was assessed to determine if precisely known flow conditions would be required during this process. Figure 3-21 shows that velocity and flowrate variations of about 100% will alter prechill durations by about five to

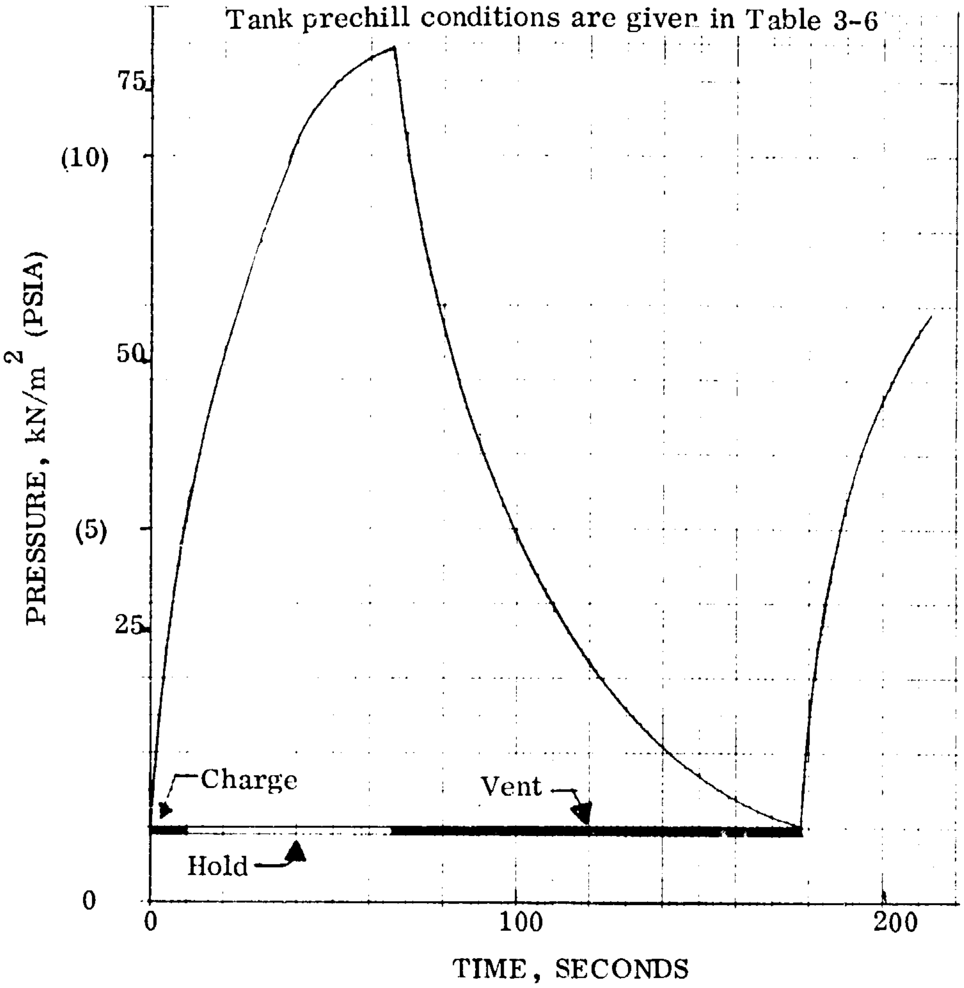


Figure 3-19. POTV Liquid Hydrogen Tank Pressure History During Prechill

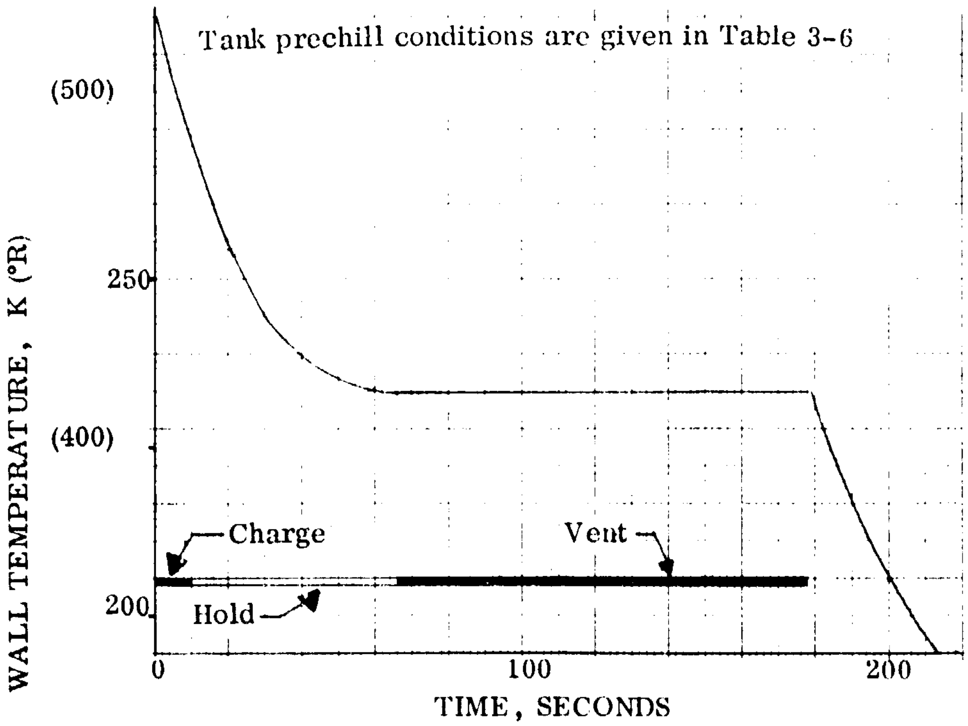


Figure 3-20. POTV Liquid Hydrogen Tank Temperature History During Prechill

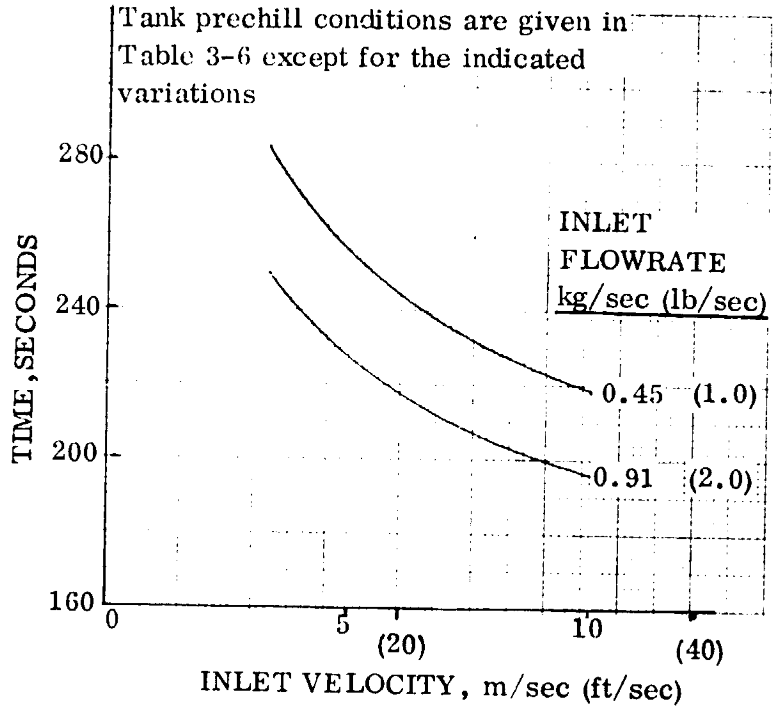


Figure 3-21. Mass flow rate and velocity influence upon liquid hydrogen tank prechill duration

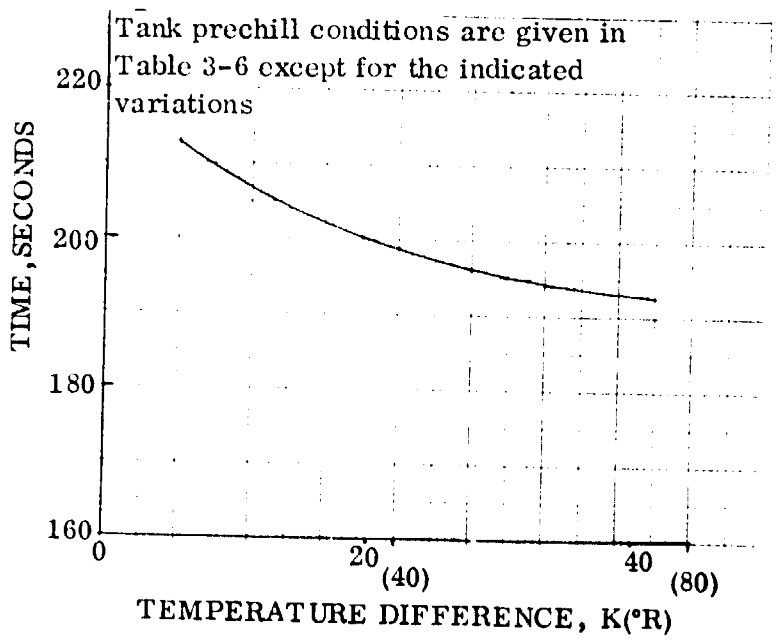


Figure 3-22. Influence of tank wall-to-vapor temperature difference upon prechill duration

fifteen percent. Figure 3-22 indicates that large differences in tank wall-to-gas temperatures will have a minimal influence upon prechill duration. Finally Figure 3-23 shows that if the hydrogen charge mass were to be inadvertently increased from 9.1 kg (20 lb) to 27.3 kg (60 lb), peak pressures will still remain below the maximum allowable of about 172 kN/m^2 (25 psia).

3.2.3.6 Summary. Analysis of the propellant tank prechill process, in this and related studies, has led to several surprises, which are listed below:

1. Liquid oxygen tank prechill is not required because under no circumstance will excessive tank pressures occur during refueling operations. Thus emphasis was directed at the liquid hydrogen tank.
2. Rapid prechill of the hydrogen tank does not appear to be an important consideration. Figure 3-4 indicates that up to 64 hours of activity is required to support a single orbiter/POTV rendezvous and transfer operation, five percent of which may be required for propellant transfer. It seems evident that propellant transfer operations could be increased to 10 percent of the total timeline without significant impact. This is nearly two orders of magnitude more time than the approximate 200 second prechill time indicated by Figure 3-20.
3. Liquid hydrogen consumed for the tank prechill process will have an insignificant influence upon overall efficiency and cost of transporting propellants into space for POTV refuelling. As a result, propellant transfer efficiency should not be an important consideration in the prechill process selection.

It is concluded that the prechill process described and analyzed in Section 3.3.2.5 will satisfy the requirements of simplicity, reliability and safety.

3.3.3 RECEIVER TANK FILL. Tank fill will be initiated after the prechill requirements have been satisfied. The single requirement for tank fill is to maintain acceptably low pressure during the process. Tank pressures will be at a minimum if thermal equilibrium conditions are maintained during fill. Thermal equilibrium will be approached as heat and mass exchange between the phases is increased, which can be achieved by creating a highly agitated fluid condition. Given the assumption of a thermal equilibrium tank fill a simple relationship can be obtained between initial tank temperature at the start of no-vent chill and fill, incoming liquid vapor pressure, and final liquid vapor pressure. This relationship is derived below from the First Law of Thermodynamics for liquid flow into a closed container.

$$dE_g + dE_L + dE_w = h_L \, dm_L \qquad (3\text{-}14)$$

$$dEg = (u_g m_g)_2 - (u_g m_g)_1 = \text{change in ullage energy} \qquad (3\text{-}15)$$

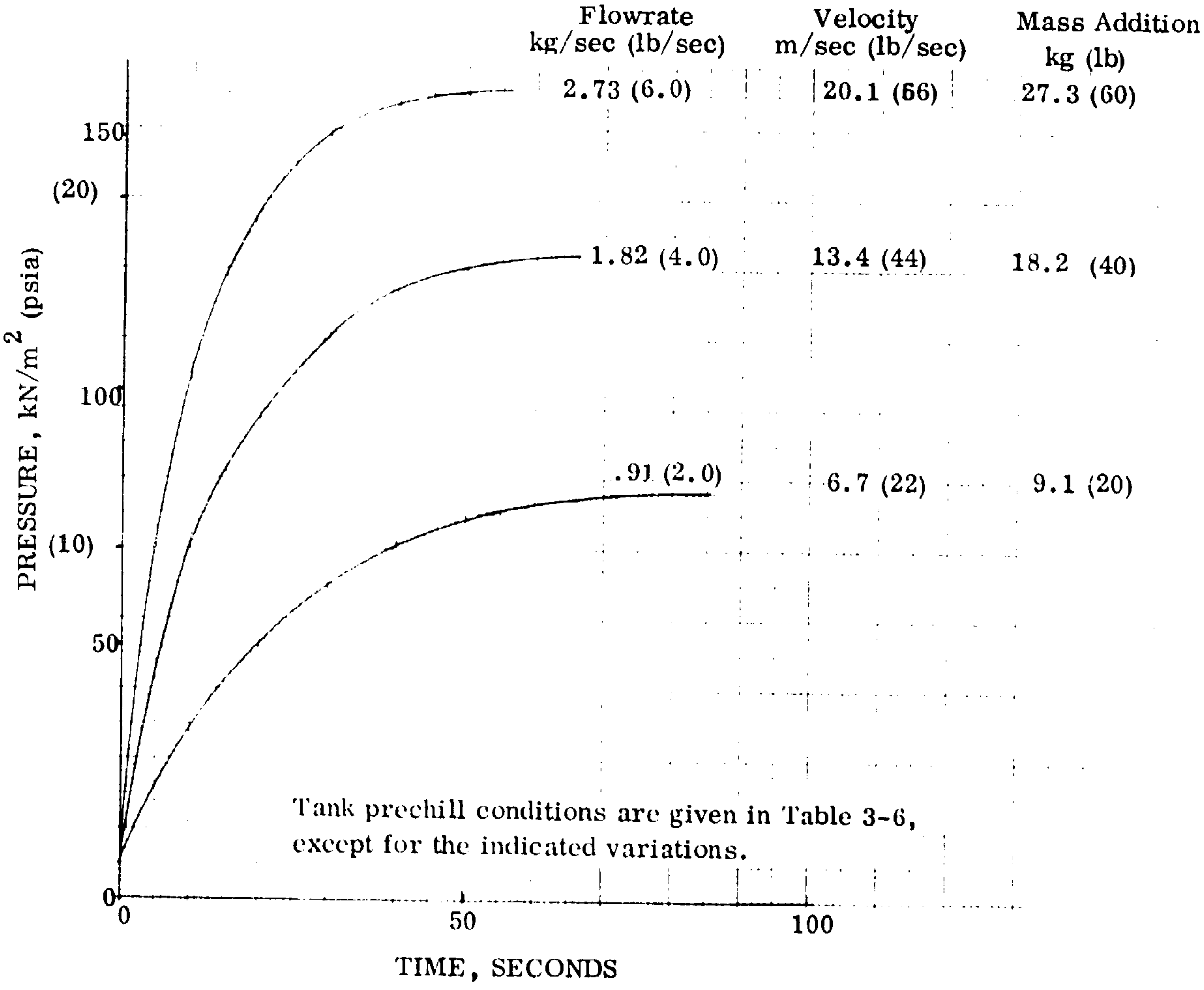


Figure 3-23. Charge Mass Influence Upon Liquid Hydrogen Tank Peak Pressures

Table 3-6. Baseline conditions selected for liquid hydrogen tank prechill procedure.

1. Initial Propellant Tank Temperature = 289 K (520R)

2. Initial Pressure = 6.9 kN/m^2 (1.0 psia)

3. Hydrogen vapor saturated at 103.4 kN/m^2 (15 psia) enters propellant tank

4. Entering flowrate = .91 kg/sec (2 lb/sec)

5. Entering velocity = 6.7 m/sec (22 ft/sec)

6. Hydrogen charge terminated after 9.1 kg (20 lb) enters tank

7. Tank vent initiated when tank-to-ullage temperature difference becomes 5.6 K (10R)

8. Propellant tank vented to 6.9 kN/m^2 (1.0 psia)

9. Vent area = 37.2 cm^2 (5.76 $in.^2$)

10. Tank mass = 447.2 kg (986 lb)

11. Tank volume = 116 m^3 (4100 ft^3)

$$dE_L = (u_L m_L)_2 - (u_L m_L)_1 = \text{change in liquid energy}$$

$$h_L dm_L = h_L(m_{L_2} - m_{L_1}) = \text{enthalpy change due to entering liquid} \quad (3\text{-}16)$$

$$dE_w = (u_w m_w)_2 - (u_w m_w)_1 = m_w (u_2 - u_1)_w \quad (3\text{-}17)$$

$$= (m_w \, \Delta u_w) = \text{change in tank wall energy} \quad (3\text{-}18)$$

For the assumption of an initially evacuated tank

$$m_{g_1} = m_{L_1} = 0$$

and

$$dE_g = u_{g_2} m_{g_2}$$

$$dE_L = u_{L_2} m_{L_2} \quad (3\text{-}19)$$

$$h_L \, dm_L = h_L \, m_{L_2} \quad (3\text{-}20)$$

Combining Equations 3-14 through 3-21 we have (3-21)

$$u_{g_2} m_{g_2} + u_{L_2} m_{L_2} + m_w \Delta u_w = h_L \, m_{L_2} \quad (3\text{-}22)$$

Solving equation 3-22 for h_L, results in

$$h_L = u_{g_2} m_{g_2}/m_{L_2} + u_{L_2} + \Delta u_w m_w / m_{L_2} \quad (3\text{-}23)$$

At thermal equilibrium h_L, u_{L_2} and u_{g_2} can be related to liquid vapor pressure and temperature. Consequently, u_{L_2} and u_{g_2} are known once final vapor pressure is specified and Δu_w is known since initial and final temperatures are given. Finally, h_L (which is a function of entering-liquid-vapor-pressure) can be determined for a desired liquid fill condition.

Equation 3-23 is summarized in Figures 3-24 and 3-25 which give entering liquid vapor pressure as a function of initial tank temperature and final tanked liquid vapor pressure for a 95 percent liquid fill condition. Note that final vapor pressure will be greater than entering liquid vapor pressure. This difference is due to the combination of initial tank wall energy and the heat of compression, which are released to the tank fluid during chill and fill. Figure 3-25 shows that final LO_2 vapor pressure will be about 6.9 kN/m^2 (1.0 psia) greater than entering vapor pressure. This small difference is due to the high propellant thermal mass. Final LH_2 vapor pressure will be

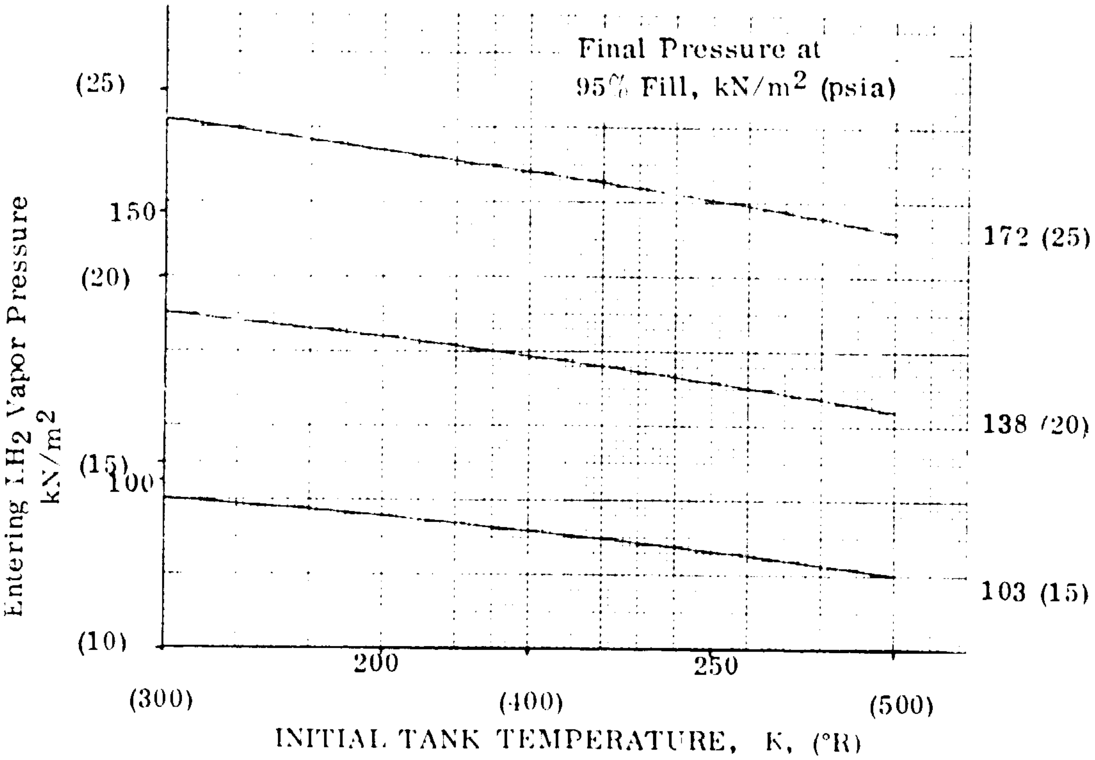


Figure 3-24. Final LH_2 Tank Pressures for Thermodynamic Equilibrium Fill Process (POTV and COTV Tanks).

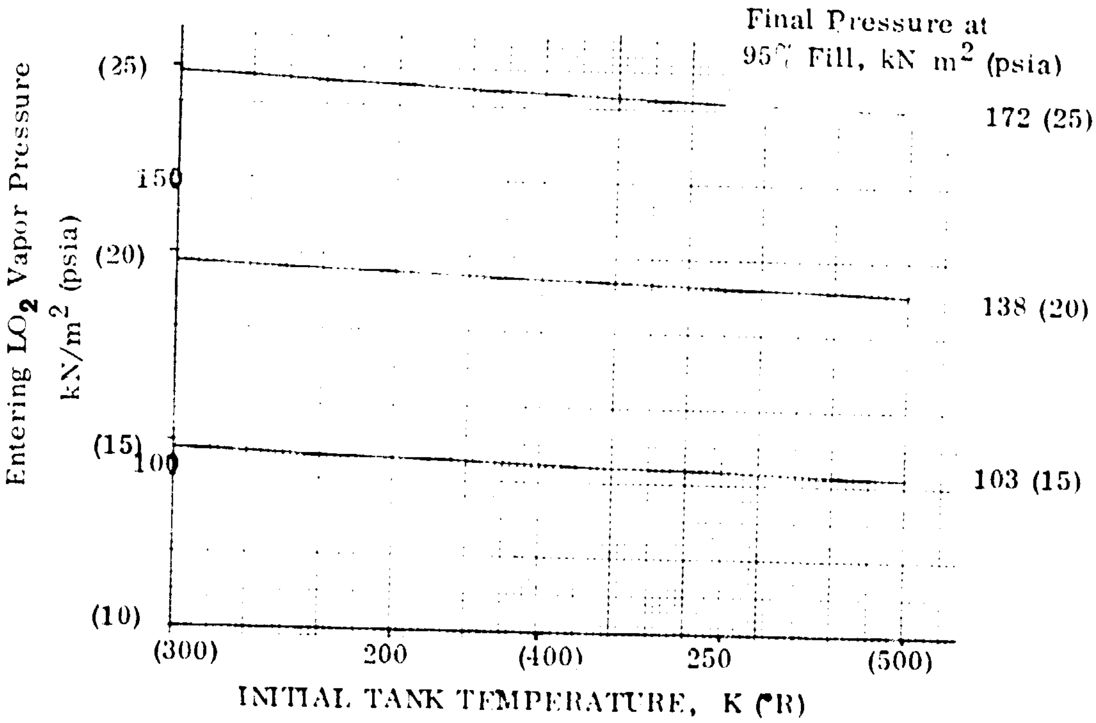


Figure 3-25. Final LO_2 Tank Pressures for Thermodynamic Equilibrium Fill Process (POTV and COTV Tanks)

about 20.7 kN/m^2 (3.0 psia) greater than entering vapor pressure, as indicated by Figure 3-24. All of the above is based upon the assumption that near-thermal equilibrium conditions will exist during tank fill, as will be discussed below.

3.3.3.1 Tank refill (autogenous). Refill of a propellant tank containing liquid and its own vapor as a pressurant is a straightforward operation. Liquid must be introduced at the correct vapor pressure, and fluid inflow conditions must be sufficiently high to assure near-thermodynamic equilibrium conditions during fill. An evaluation will be made of the relationship between initial and final tank fluid conditions, and entering liquid vapor pressure for a thermal equilibrium process. The inlet vapor pressure is determined on the basis of the following First Law analysis

$$dE_L + dE_g = hdm \tag{3-24}$$

$$dE_L = (u_L m_L)_2 - (u_L m_L)_1 = \text{change in liquid energy} \tag{3-25}$$

$$dE_g = (u_g m_g)_2 - (u_g m_g)_1 = \text{change in vapor energy} \tag{3-26}$$

$$hdm = h\,(m_{T2} - m_{T1}) = \text{total energy of entering liquid} \tag{3-27}$$

$$m_T = m_L + m_g$$

The following equalities result from the assumptions of phase equilibrium and initial liquid temperature equals final liquid temperature: $u_{L2} = u_{L1} = u_L$ and $u_{g2} = u_{g1} = u_g$. Combining these conditions with Equations 3-24 through 3-27 results in

$$u_L\,(m_{L2} - m_{L1}) + u_g\,(m_{g2} - m_{g1}) = h\,(m_{T2} - m_{T1}) \tag{3-28}$$

Now, total mass within the tank can be expressed as

$$m_T = V_T\,\rho_L - (\alpha - 1)\,m_g \tag{3-29}$$

and

$$M_{T2} - m_{T1} = (\alpha - 1)\,(m_{g1} - m_{g2}) \tag{3-30}$$

where

m_T = $m_L + m_g$ = total mass of fluid in tank

ρ_L = liquid density

ρ_g = vapor density

α = ρ_L / ρ_g

Combining Equations 3-28 through 3-30 we find that

$$u_L (m_{L2} - m_{L1}) + u_g (m_{g2} - m_{h1}) = h (\alpha - 1) (m_{g1} - m_{g2}) \tag{3-31}$$

Also, by adding and subtracting $u_L m_{g1}$ and $u_L m_{g2}$ to the left side of Equation 3-31

$$u_L (m_{T2} - m_{T1}) + u_{ev} (m_{g2} - m_{g1}) = h (\alpha - 1) (m_{g1} - m_{g2}) \tag{3-32}$$

where

u_{ev} $= (u_g - u_L)$ = internal energy of evaporation

Combining Equations 3-30 and 3-32

$$u_L (\alpha - 1) (m_{g1} - m_{g2}) + u_{ev} (m_{g2} - m_{g1}) = h (\alpha - 1) (m_{g2} - m_{g1}) \tag{3-33}$$

Solving for the entering liquid enthalpy,

$$h = u_L - uev/(\alpha - 1) \tag{3-34}$$

Results are plotted in Figure 3-26 showing liquid hydrogen tank final vapor pressure as a function of incoming liquid vapor pressure. It is interesting to note that this vapor pressure relationship is independent of initial propellant load at the start of refill. Furthermore, although Equation 3-34 was developed for an initial thermal equilibrium condition, results are also applicable to an initially superheated ullage.

The above results indicate that refill can be simplified because the same entering liquid vapor pressure will be required regardless of initial fill condition and ullage temperature. Figure 3-26 indicates that final tank pressure will be approximately 27.6 kN/m^2 (4 psia) higher than entering liquid vapor pressure.

The thermal equilibrium processes described above have demonstrated that tank fill (and refill) can be performed without having to vent. This is an important conclusion because selective vapor venting will not be possible once tank fill is initiated. Liquid venting must be avoided because the propellant loss and resulting disturbing torques could be unacceptably high. Consequently, it is mandatory that the refueling process be performed with a closed tank. The next step is to analytically describe the tank fill process in order to assess the influence of such key variables as entering flowrate and velocity upon thermal equilibrium. If thermal equilibrium conditions can be readily achieved for a reasonable range propellant flowrates and velocities, then orbital refueling, without venting, will be possible.

3.3.3.2 Tank fill analysis. The intent of the tank fill process will be to create turbulent conditions within the tank. These conditions will be achieved by introducing

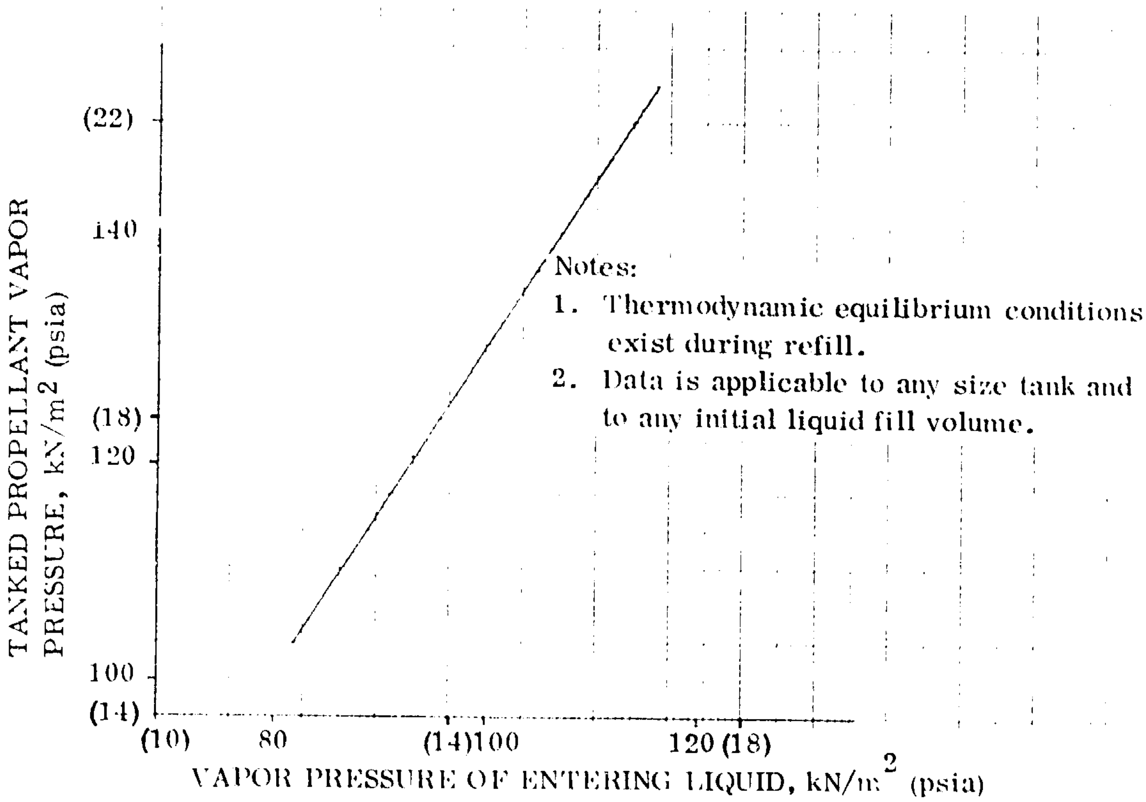


Figure 3-26. Entering Liquid Hydrogen Vapor Pressure Required to Maintain a Constant Vapor Pressure in Tank During Fill

liquid into the tank at high velocities (and perhaps through a spray nozzle) to provide the high heat-transfer rates needed to attain near-thermal equilibrium. As tank-fill continues, the internal tank fluid environment changes from liquid droplets in the ullage volume to vapor bubbles entrained within a liquid bulk. It is expected that the transition from heat transfer dominated by liquid droplets to heat transfer dominated by vapor bubbles will occur in the range of 40% to 60% liquid fill. The mechanism of liquid spray in a vapor environment will change to that of vapor bubble entrainment and dispersal, within the liquid volume. This mechanism will be the dominant mode of heat and mass exchange throughout much of the tank fill process, and is the only mechanism that will influence tank pressures toward the completion of tank-fill.

The basis for any fill process is that sufficient fluid motion created within a propellant tank will maintain near thermal equilibrium. As equilibrium is approached, the pressure difference (ΔP) between tank pressure and liquid vapor pressure will approach zero, and absolute tank pressures during fill will approach a minimum. To aid in describing the phenomena that are expected to occur, tank fill experiment results are hypothesized and given in Figure 3-27. The figure illustrates two important points: First, tank ΔP becomes smaller as flowrate for a given tank configuration is increased. This is expected, because heat transfer coefficients will increase as flowrate

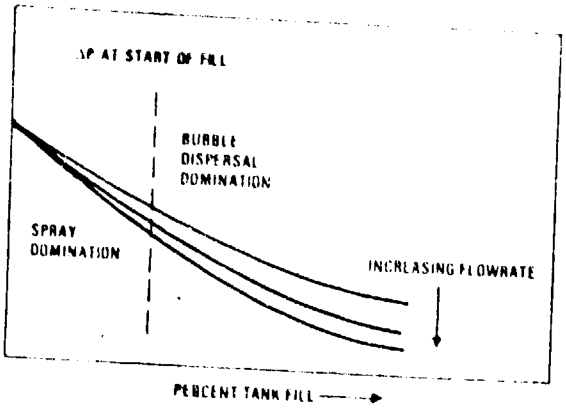


Figure 3-27. Flowrate influences tank pressure during fill.

increases. Second, there is a fill flowrate above which variations in ΔP become insignificant because fluid agitation has been increased to a level where near thermodynamic equilibrium already exists. The objective of this analysis will be to identify fluid inflow parameters that will provide near-equilibrium conditions during fill.

Fill Model for Liquid Spray Dominance.

During the early phases of tank fill, near-thermal equilibrium conditions will be achieved by introducing liquid into the propellant tanks through spray nozzles. The resulting spray will create a large liquid-vapor surface. The combination of large surface area and high droplet velocity will provide the high heat transfer rates needed to attain near-equilibrium conditions.

General Dynamics developed a computer program (HYPRES) on IRAD funds that describes the thermodynamic and fluid mechanic phenomena occurring during the early phase of a receiver tank fill process. This tank fill program was used to predict fluid management system fill pressures during a previous study, Filling of Orbital Fluid Management Systems, Reference 3-3. Subroutine DROP, of program HYPRES, which describes the ullage-spray droplet interaction is given in Appendix C of that Reference. The equations contained in this subroutine indicate that droplet diameter, spray velocity and mass flowrate may have a major influence upon propellant tank pressures during the early stages of the refuelling operation. Consequently, a series of computer runs were made with HYPRES to evaluate these variables and to assess whether acceptably low liquid hydrogen tank pressures would be maintained. These results are summarized in Figure 3-28 and 3-29.

Figure 3-28 shows the influence of droplet diameter upon tank pressure during the liquid spray dominance period. The range of droplet sizes considered include sizes that should be present during an actual refuelling operation. However, the figure indicates that droplet diameter will not have a major influence on tank pressure. Thus, it appears that droplet diameters outside the indicated range will have a minimal impact upon refuelling operations.

Figure 3-29 gives the influence of liquid spray volume upon hydrogen tank fill pressures. An input to HYPRES includes a variable (called PACK) which identifies the volume of liquid in droplet form that will exchange energy with the ullage. Specifically, PACK is the volume ratio of liquid to ullage, and this term is multiplied by ullage volume to obtained total liquid droplet spray volume. Figure 3-29 indicates that this

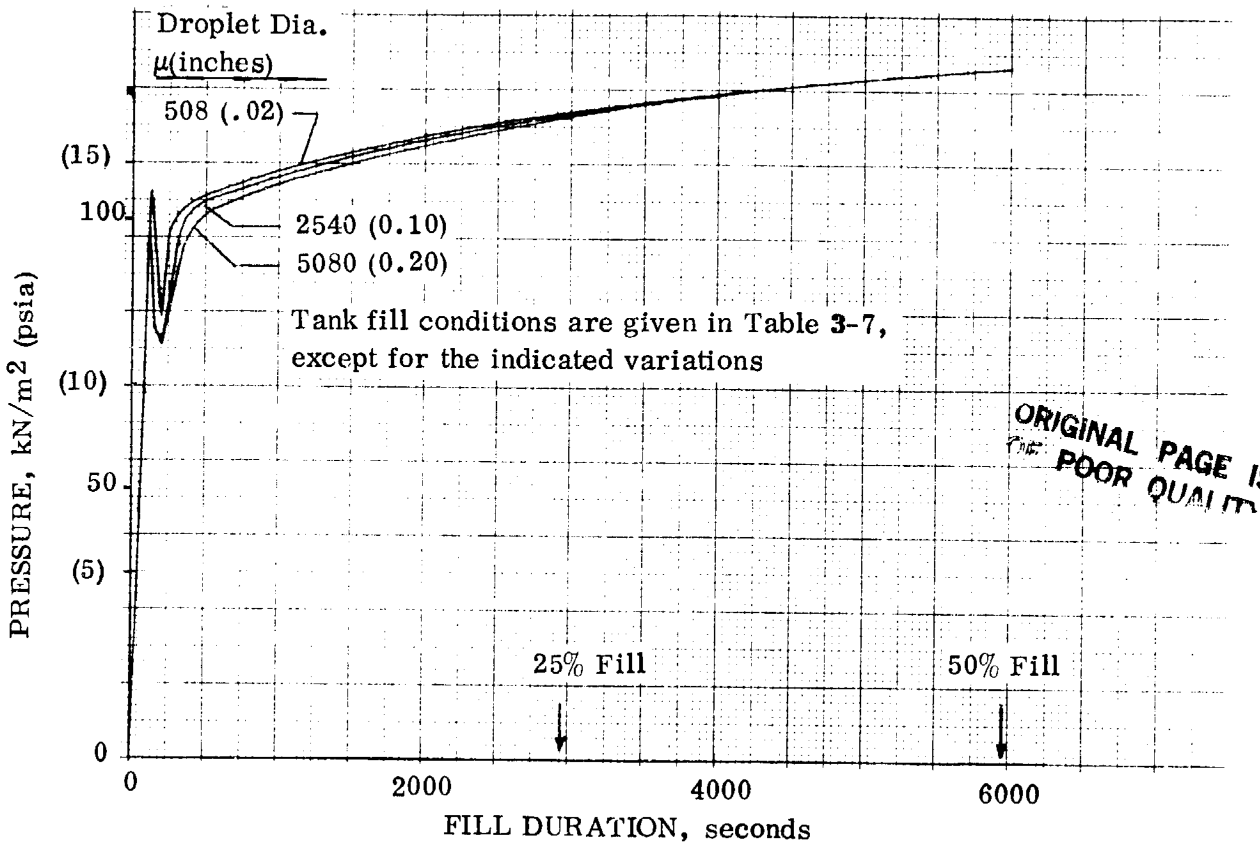


Figure 3-28. Influence of Liquid Spray Droplet Diameter Upon Liquid Hydrogen Tank During Fill

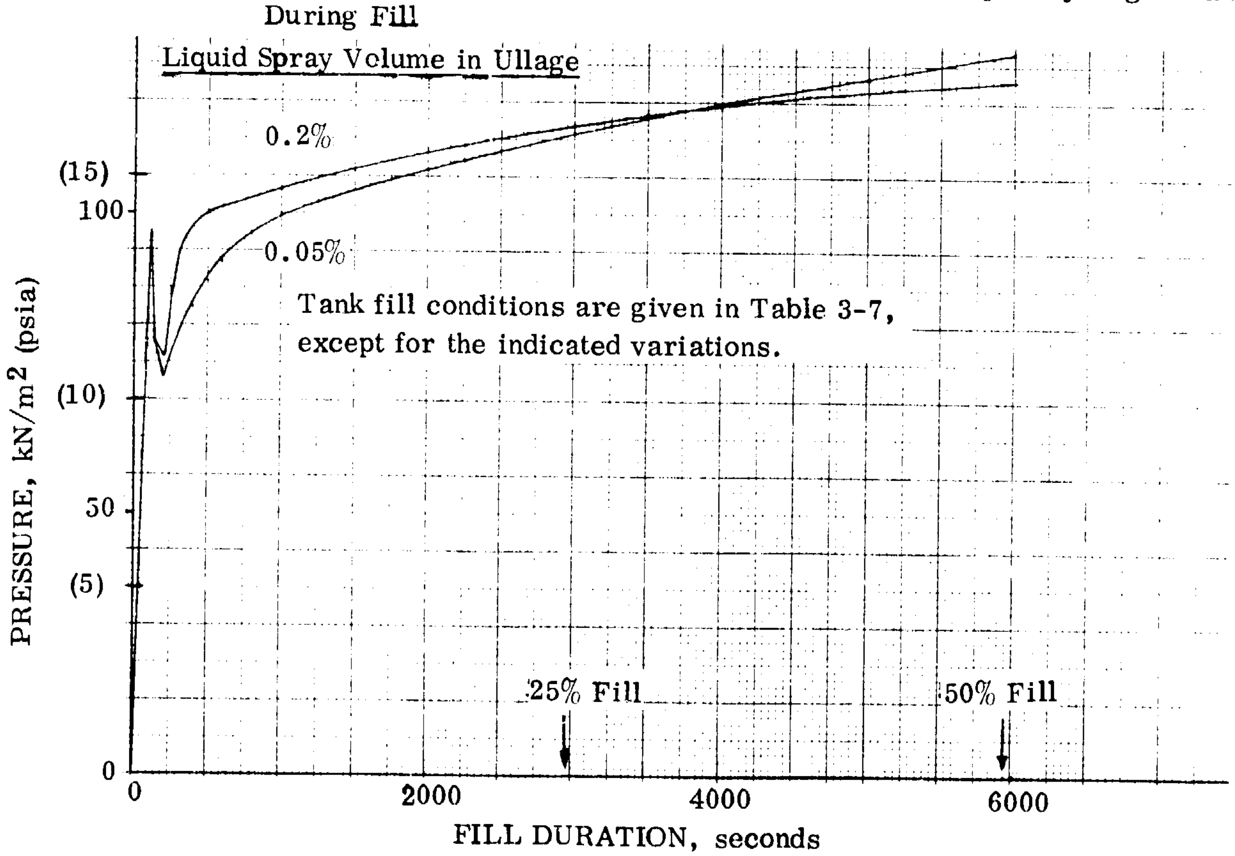


Figure 3-29. Influence of Liquid Spray Volume in Ullage Upon Liquid Hydrogen Tank Pressure During Fill

spray volume will have a greater influence upon pressures than droplet diameter. This influence, however, does not appear to be significant. Note that the two curves cross-over at about 4000 seconds (36% fill level. Beyond this time the case for PACK=0.2% predicts a lower pressure than the case for PACK=0.05%. This cross-over is probably due to the greater liquid-ullage heat exchange (for PACK=0.2%) which tends to maintain near-thermal equilibrium conditions.

Figure 3-30 shows the influence of propellant flowrate upon tank pressure. For this case pressures are plotted against propellant fill levels in order to normalize the influence of different tanking flowrates. It is seen that higher flowrates will result in slightly lower tank pressures during the early stages of fill, followed by slightly higher tank pressures later in the fill process. The lower initial pressures during the early part of tanking are caused by the quenching influence of the higher mass flow condition. The higher pressure rise rates that occur later in the tanking operation are due to an inability to transfer the higher heat of compression rates from ullage to propellant.

Table 3-8 and Figure 3-31 serve to illustrate this point. The energy exchange required to achieve thermal equilibrium between ullage and propellant bulk is summarized in Table 3-8. This quantity for the POTV hydrogen tank is approximately 13.48 kW-hr (46010 Btu), and remains independent of refueling duration. The rate of heat exchange,

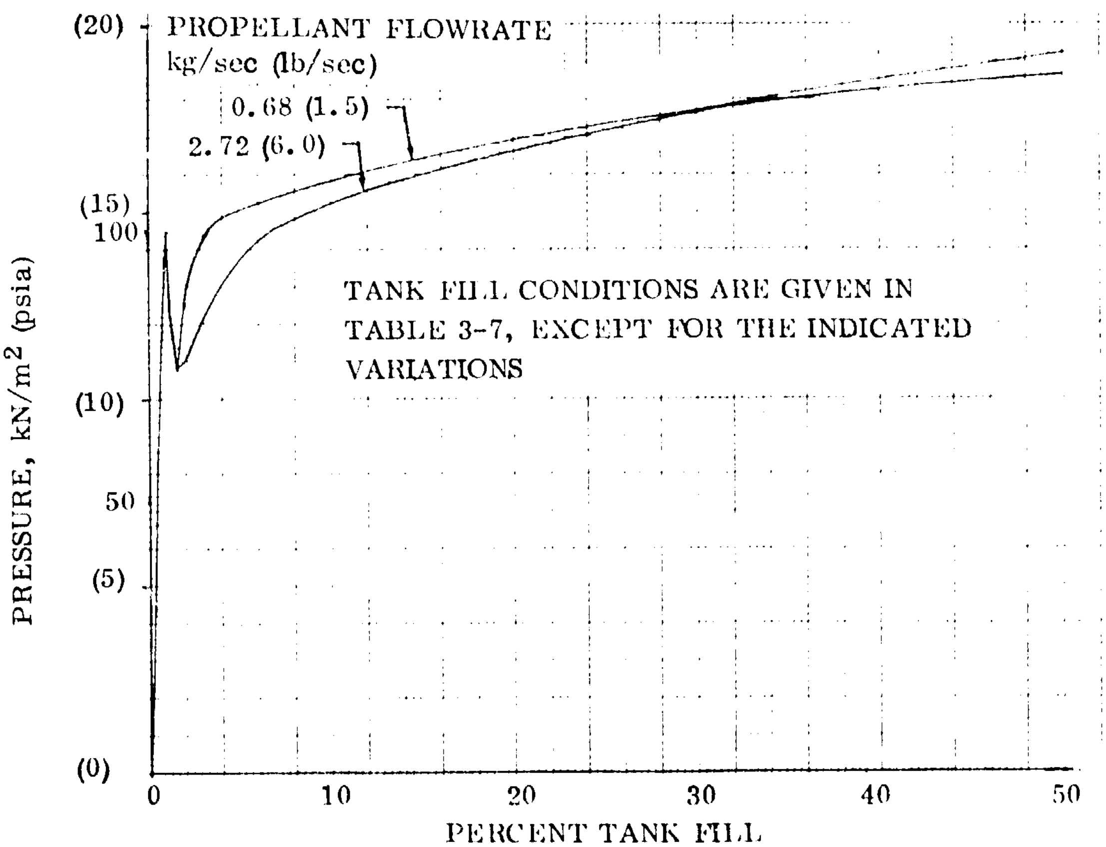


Figure 3-30. Mass Flow Rate Influence Upon Liquid Hydrogen Tank Pressure During Fill

Table 3-7. Baseline conditions selected for liquid hydrogen propellant tank fill analysis.

1. Initial tank temperature = 200 K (360 R)
2. Tank mass = 447.2 kg (986 lb)
3. Tank volume = 116 m^3 (4100 ft^3)
4. Initial tank pressure = 0 kN/m^2 (0 psia)
5. Liquid hydrogen, saturated at 103.4 kN/m^2 (15 psia) enters propellant tank
6. Entering liquid flowrate = 0.68 kg/sec (1.5 lb/sec)
7. Entering liquid velocity = 3.05 m/sec (10 ft/sec)
8. Spray droplet diameter = 2540 μ (0.1 inch)
9. (Liquid spray volume in ullage/ullage volume) = .002

however, will be inversely proportional to tank fill duration, as illustrated by Figure 3-31. This figure indicates that the average heat exchange rate will be 13.48 kW (12.78 Btu/sec) and 6.74 kW (6.39 Btu/sec) respectively, for one hour and two hour tank fill durations.

Computer runs were also conducted to determine the influence of inlet velocity upon fill pressures. No plots are given because it was found that tank pressure variations will be insignificant over a velocity range of 3.05 m/sec (10 ft/sec) to 15.25 m/sec (50 ft/sec).

Table 3-8. Ullage cooling required to attain thermal equilibrium during POTV LH_2 tank fill.

	Energy KW-hr(Btu)	Comments
Initial Ullage Energy	10.4 (35500)	Based upon tank wall temperature of 194.4 K (350R) at end of prechill
Heat of compression	3.33 (11380)	Compression heating due to fill process
Final ullage energy	0.25 (840)	Assumes 5% ullage volume at saturated conditions
Ullage energy removal requirements	13.48 (46010)	This energy must be transferred to liquid in order to attain thermal equilibrium by the end of tank fill.

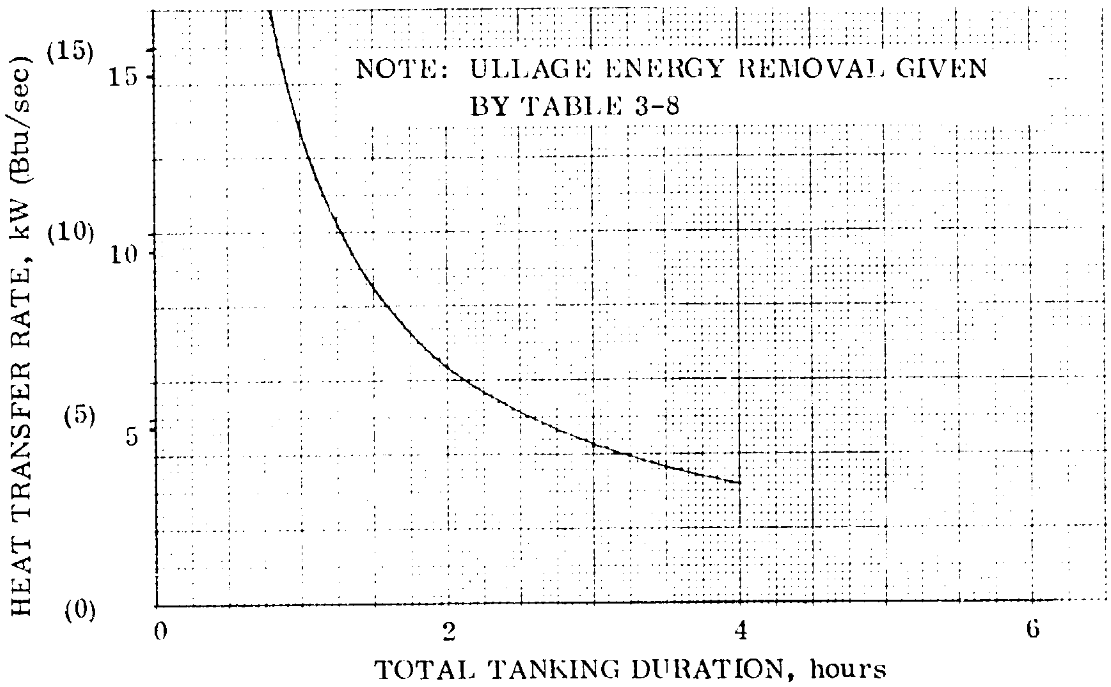


Figure 3-31. Required Average Ullage-to-Liquid Heat Transfer Rate for Liquid Hydrogen Tank Refuelling Operation

Fill Model for Vapor Bubble Dominance. When the propellant tanks are approximately 50% full, the dominant heat exchange mode will be that of convection and condensation between the liquid bulk and entrained vapor bubbles.

In a previous study (Reference 3-3) equations were obtained which predict this heat exchange during tank fill. These equations (given below) include the influence of inlet fluid conditions, fluid properties and tank geometry.

$$\left(\frac{h}{Cp\ \rho_L}\right)\left(N_{Pr}\right)^{2/3} = .163\left[\frac{(\dot{m}v^2/V_L)\ \mu_L}{\rho_L^2}\right]^{1/4} \tag{3-35}$$

where:

h = heat transfer coefficient

C_p = constant pressure heat capacity

ρ_L = liquid density

N_{Pr} = Prandtl number

$\dot{m}$ = entering mass flow rate

v = entering liquid velocity

V_L = liquid volume in tank

μ_L = liquid viscosity

$$d = \frac{1.134\,\sigma^{0.6}\ \epsilon^{0.5}}{\gamma^{.2}\ (mv^2/V_L)^{0.4}} + .09 \qquad (3\text{-}36)$$

where:

d = bubble diameter, cm

σ = surface tension, dyne/cm

ϵ = entrained vapor volume to total vapor plus liquid volume

γ = liquid specific gravity

Note: These equations are from Reference 3-3 equations 3-53 and 3-33, respectively.

Equations 3-35 and 3-36 are modifications of empirically derived expressions from industrial applications for which vapor dispersal in liquid is essential to achieving a high rate of heat and mass transfer. This subject has been extensively studied and much of the work has been collected and summarized in Reference 3-4. The empirically derived expressions were applied to the refueling process by replacing the mechanical mixer power term, employed for industrial applications, with an equivalent fluid power expression, $\dot{m}v^2$. This conversion was accomplished with the following relationships:

Fluid Power = Mixer Power Output

Mixer Power Output = Efficiency × Mixer Power Input

$\therefore$ $\dot{m}v^2$ = 40 Percent × Mixer Power Input

where 40 percent efficiency represents a conservative value.

The heat transfer rate between an individual bubble and the liquid bulk can be expressed as

$$\frac{Q_B}{V_B} = \frac{hA_B(T_B - T_L)}{V_B} = \frac{\pi d^2}{(\pi/6)\,d^3} h\,(T_B - T_L) = \frac{6h\,(T_B - T_L)}{d} \qquad (3\text{-}37)$$

where

$\dot{Q}_B$ = heat transfer rate from each bubble

V_B = bubble volume

T_B = bubble vapor temperature

T_L = liquid bulk temperature

Total heat transfer rate from the total dispersed vapor volume can be determined from Equation 3-37 by introducing the total number of vapor bubbles, n,

$$\frac{\dot{Q}_T}{V_g} = \frac{n\dot{Q}_B}{nV_B} = \frac{\dot{Q}_B}{V_B} = \frac{6h\,(T_B - T_L)}{d} \qquad (3\text{-}38)$$

where h and d are determined from Equations 3-35 and 3-36, respectively, and

$\dot{Q}_T$ = total heat transfer rate from the entrained vapor

V_g = total vapor volume entrained in liquid

Equations 3-35, 3-36 and 3-38 were added to the HYPRES program to provide capability for evaluating tank fill conditions during the vapor-bubble dominance mode. A series of computer runs were conducted to evaluate this mechanism. Results are given in Figures 3-32 through 3-34 which are, respectively, a continuation of Figures 3-28 through 3-30. A transition from liquid spray heat exchange to vapor-bubble heat exchange was imposed at the 70 percent propellant fill level. A sudden pressure drop occurred coincident with this heat exchange transition. This pressure drop was obviously due to the increased heat transfer rate that created near-thermal equilibrium conditions.

The curves of Figures 3-32 through 3-34 require clarification, especially following the transition to vapor-bubble heat exchange mechanism. First, there is no indicated tank pressure change due to variations in spray droplet diameter or PACK factor. This occurs because heat exchange will be influenced only by entering flowrate, velocity, and entrained vapor volume, E, and these quantities are identical for each case plotted in Figures 3-32 and 3-33.

Figure 3-34 also shows a negligible difference in pressures, due to flowrate variations, following the transition in heat exchange mechanism. For these conditions, however, compensating factors may influence tank pressures: a) the high flowrate condition requires a high liquid-ullage heat exchange rate in order to maintain low pressures,

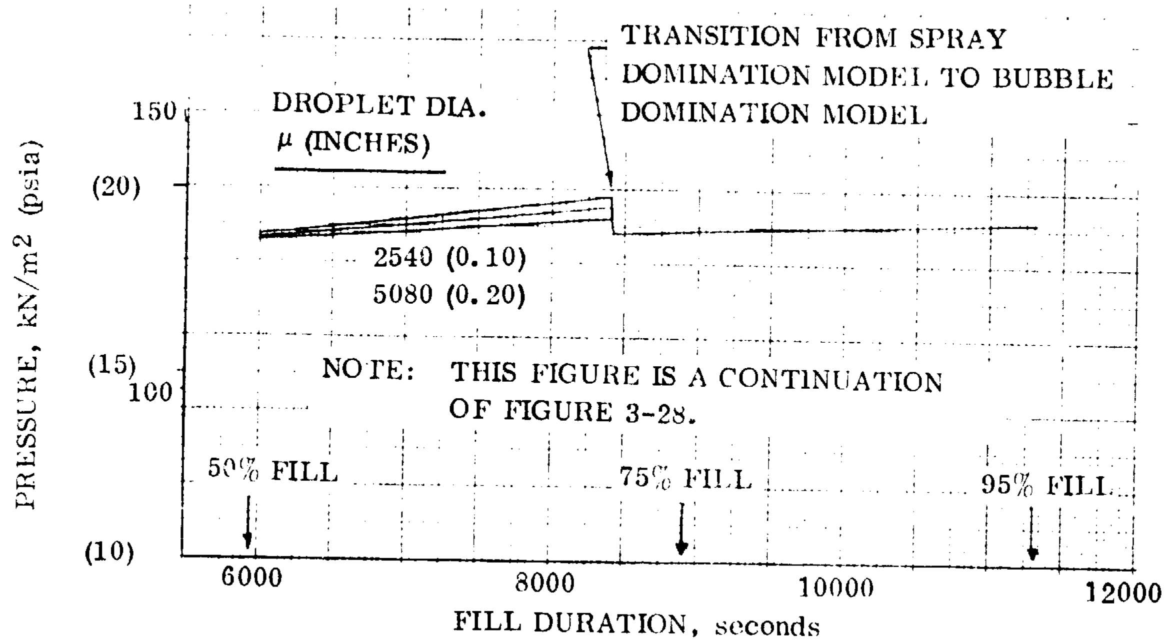


Figure 3-32. Influence of Vapor Bubble Heat Exchange Mechanism Upon Hydrogen Tank Fill Pressures for Range of Liquid Spray Droplet Diameter

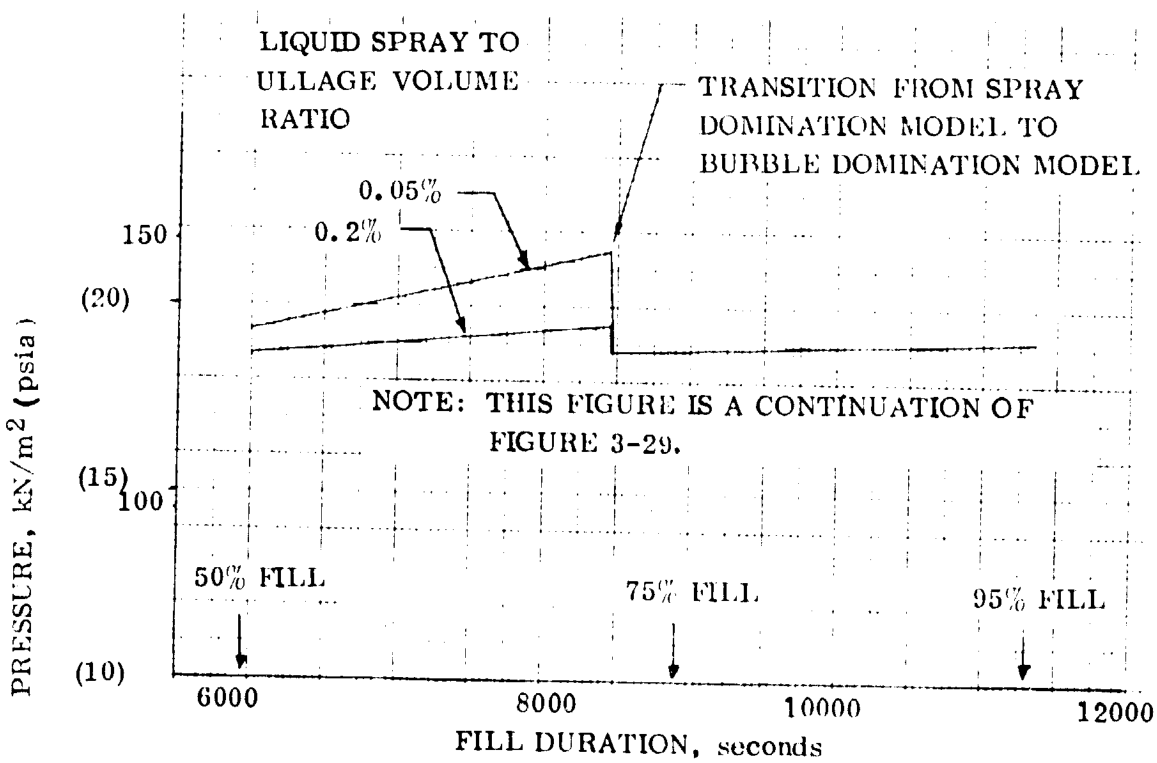


Figure 3-33. Influence of Vapor-Bubble Heat Exchange Mechanism Upon Hydrogen Tank Fill Pressures for Range of Liquid Spray Volume

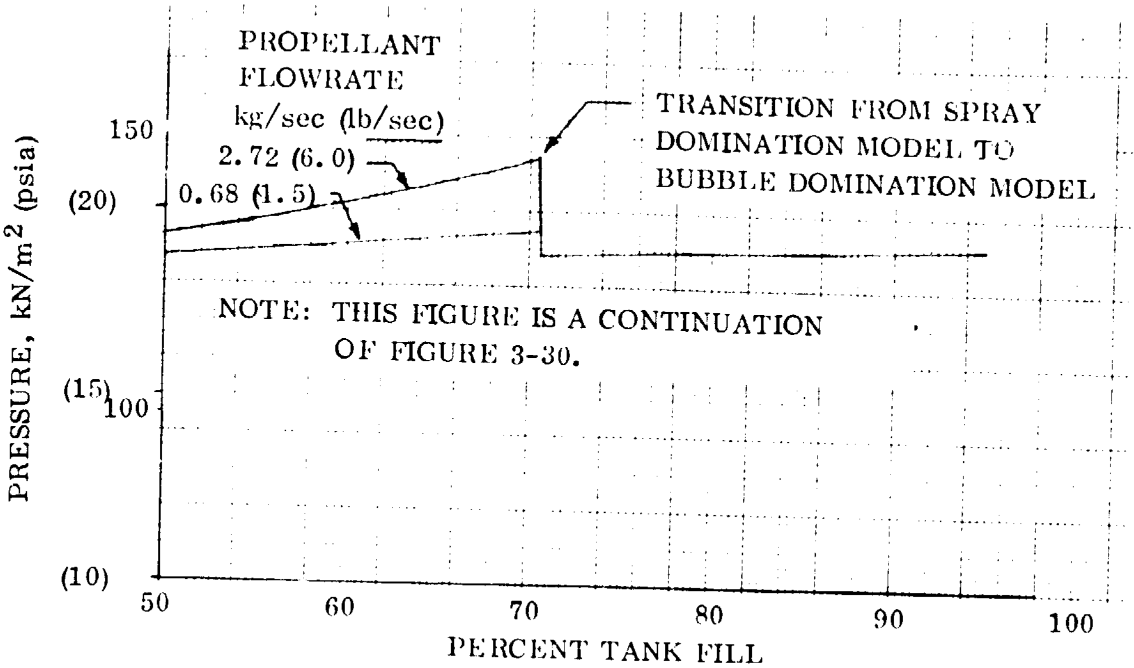


Figure 3-34. Influence of Vapor Bubble Heat Exchange Mechanism Upon Hydrogen Tank Fill Pressures for Range of Tanking Flow Rates

and b) the high flowrate condition creates a high heat transfer coefficient, which serves to maintain low pressures. The negligible difference in tanking pr ssure indicates that the above factors were, in fact, compensating, or that the heat exchange mechanism is sufficient to assure near-thermal equilibrium conditions. The latter condition is a more likely possibility.

Results of this analysis supports the assertion, stated earlier, that heat and mass exchange due to vapor bubble entrainment and dispersal within the liquid volume will dominate. More important is the likelihood that this is the only mechanism that will influence tank pressure towards the completion of tank fill. Furthermore, it appears that heat and mass exchange rates will be of such a high magnitude that near-thermal equilibrium conditions will be maintained over a broad range of entering liquid flowrates and velocities. This is a significant conclusion because the thermal equilibrium parametric data of Figures 3-24 through 3-26 can be employed to predict tank pressure conditions at fill (or refill) completion.

3.3.3.3 Supply tank influence. All analyses to date have been based on the assumption that propellant enters the receiver tank at a constant temperature. This temperature corresponds to a vapor pressure of 103.4 kN/m^2 (15 psia). In fact, entering liquid vapor pressure (and temperature) will vary throughout the refueling process because of the supply tank pressurization method. Helium pressurization of the supply tank during propellant transfer was selected because it appears to have fewer complications than other approaches.

Helium will be bubbled into the propellant to effect boiloff, which aids in pressurization. This boiloff cools the liquid and reduced its vapor pressure during propellant transfer. Figures 3-35 and 3-36 show the resulting supply tank propellant temperatures and vapor pressure histories during outflow.

Figure 3-37 compares receiver tank pressure histories for the constant temperature and varying temperature conditions. Note that for the variable supply tank conditions the tank pressure rise-rates are not as great as for the constant temperature case. Furthermore tank pressure begins to decay after the receiver tank is 40 percent filled, whereas pressure continues rising for the other case. This behavior can be explained by referring to Figure 3-24, which shows that final vapor pressure is a function of entering liquid vapor pressure. Since the average entering vapor pressure will be lower for the variable supply tank conditions, final vapor pressure must also be lower.

There are two factors that control the increasing and decreasing pressures indicated by Figure 3-37; the heat of compression caused by propellant fill, and the steadily decreasing enthalpy of entering propellant. The heat of compression causes tank pressure to rise continuously, as indicated by the constant incoming temperature case. The second factor is responsible for pressure decay, as the lower temperature incoming liquid mixes with the propellant bulk. Evidently, the heat of compression is the greater influence during the early stages of fill, and the lower temperature liquid is dominant beyond the 40 percent fill condition.

It should also be emphasized that a single orbiter-tanker mission will not refuel the POTV propellant tanks. The Reference 3-1 study was based upon an orbiter payload capability of 45,360 kg (100,000 lb), which translates to a liquid hydrogen supply tank volume of 77 m^3 (2720 ft^3). This system is capable of refueling the POTV fuel tank to the 66.4 percent level, as indicated by Table 3-9. The liquid oxygen tank would also be filled to the same level. As Figure 3-2 indicated, a second tanker flight would refuel the second POTV stage, followed by a third flight which would top off both stages and deliver the crew module.

3.3.3.4 Alternative refill concept. An alternative concept of on-orbit refuelling was evaluated during the study. This concept, which requires introducing propellants through a diffuser to achieve extremely low entering velocities, is a major departure from the selected technique of a high velocity liquid spray. This alternative was based upon the premise that large quantities of liquid would enter the propellant tank before striking the opposite tank wall. The resulting boiloff would not create a high tank pressure because the propellant bulk would mix with vapor and maintain a low pressure. In assessing the spray nozzle versus inlet diffuser configurations the following minimum set of requirements were considered:

1. Avoid excessive tank pressures throughout the prechill and fill processes

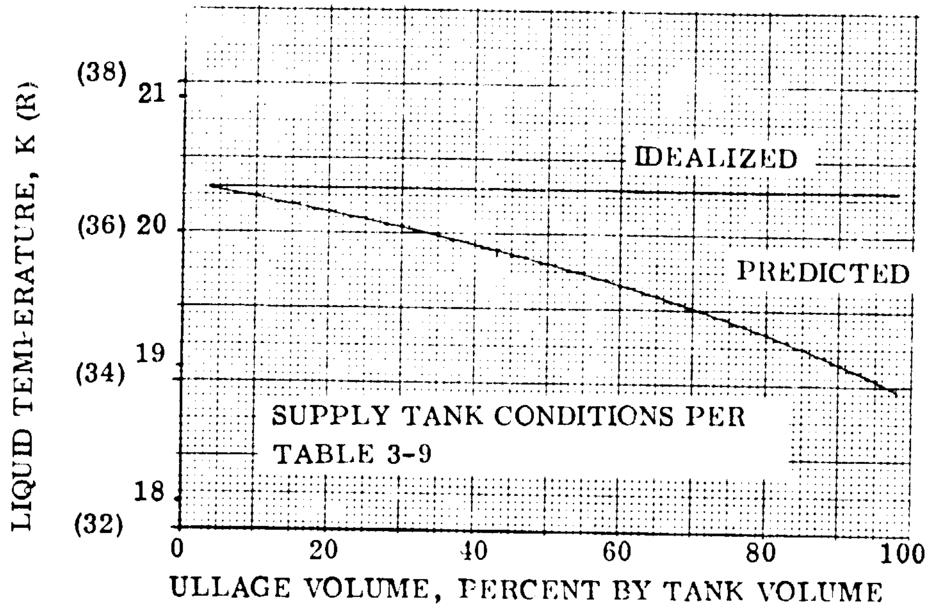


Figure 3-35. Supply Tank Liquid Temperature During POTV Refill

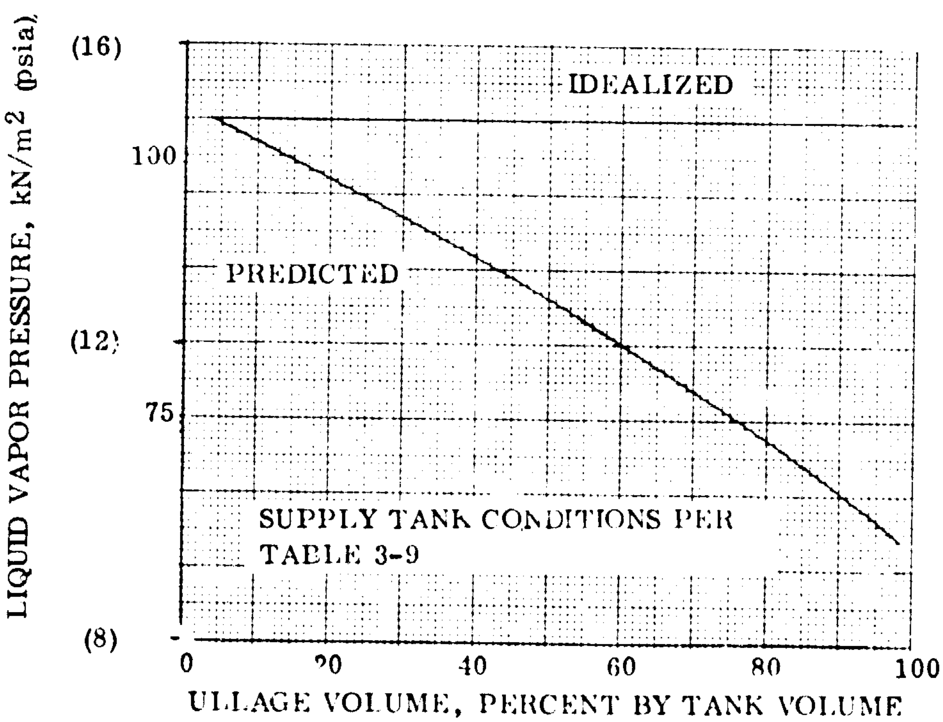


Figure 3-36. Supply Tank Liquid Vapor Pressure During POTV Refill

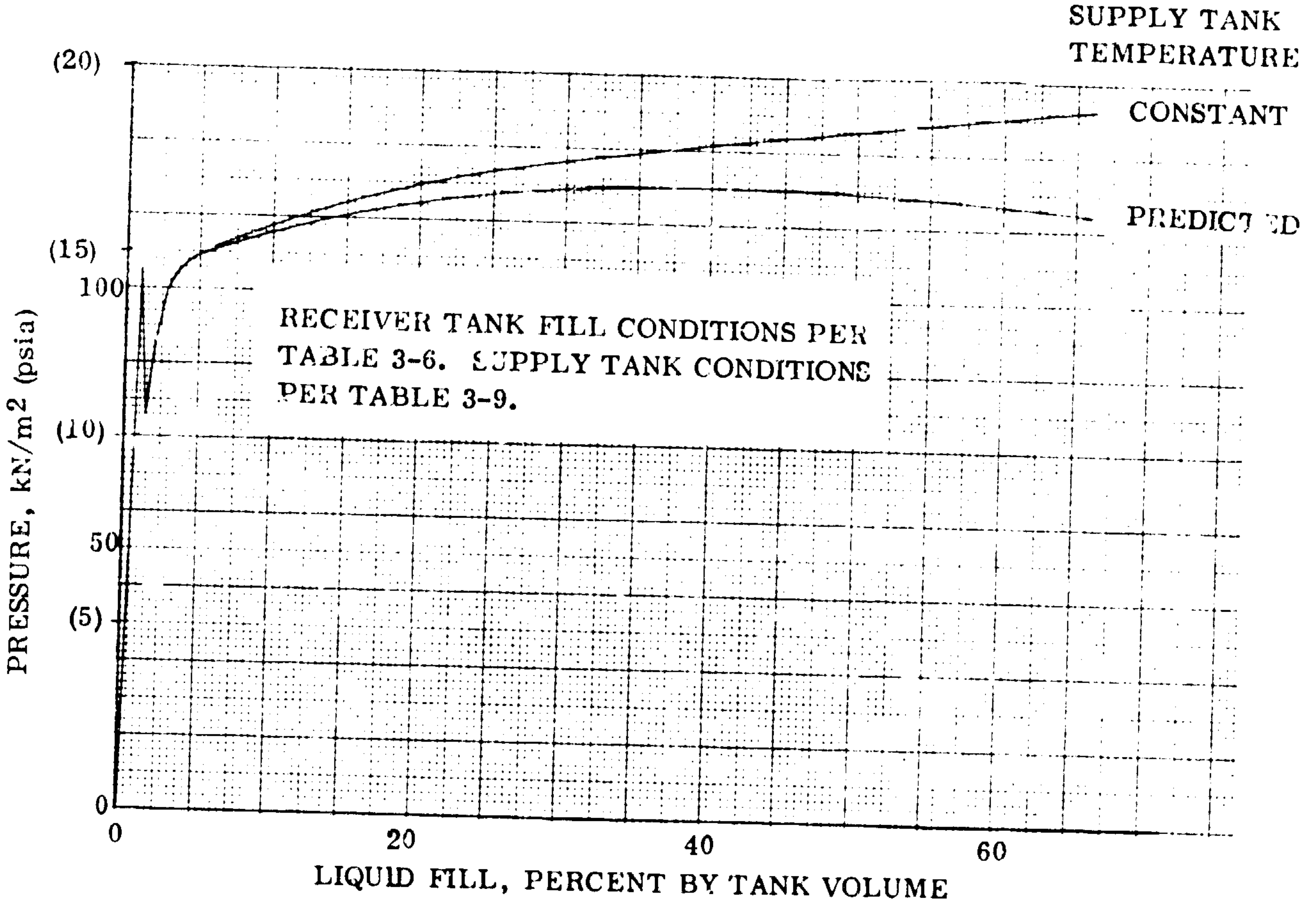


Figure 3-37. Supply tank LH_2 temperature influence upon POTV tank pressure during refill.

Table 3-9. Supply tank conditions during LH_2 tank POTV refill.

Initial liquid vapor pressure	= 103.4 kN/m^2 (15 psia)
Initial liquid temperature	= 20.3 K (36.6°R)
Supply tank volume	= 77 m^3 (2720 ft^3)
Initial ullage volume	= 3.5 percent
Unusable liquid residual	= 2 percent
Usable liquid residual*	= 72.7 m^3 (2570 ft^3)

* This is 66.4% of POTV LH_2 tank volume.

2. Minimize vent mass requirements

3. If venting is necessary, avoid two-phase venting

With regard to the LO_2 tank, any tank inlet configuration will be satisfactory. Venting will not be required during prechill; therefore, two phase venting is not a possibility. The only question that may arise is that of excessive tank pressure during the fill process. It is obvious from Figure 3-25 that moderately low pressure will exist for near-thermal equilibrium conditions. But these conditions will occur only if sufficient fluid turbulence is present to enhance heat and mass exchange between the phases. Fluid turbulence is more likely to occur with a spray nozzle than with an inlet diffuser, whose primary function is to introduce liquid into the tank at extremely low velocities. Thus, a spray nozzle inlet configuration would appear to best satisfy the LO_2 tank fill requirements.

A spray nozzle inlet configuration for the LH_2 tank will result in venting during the prechill process, as indicated by Figure 3-19. This is so because the spray velocity conditions will create a high heat and mass exchange environment between hydrogen vapor or liquid and the tank walls. However, two-phase venting will be avoided because liquid will not be present in the tank when venting is initiated. Once tank fill is initiated, the spray nozzles will create the turbulent fluid conditions that are beneficial to maintaining low pressures during fill.

The affect of an inlet diffuser upon hydrogen tank pressure during prechill is not quite so clear. Ideally, a low velocity diffuser will allow large quantities of propellant to be introduced into the tank before the leading edge of a large-diameter jet impinges at the opposite end of the tank. It has been hypothesized that a large propellant mass will serve to quench the pressure-rise that occurs due to liquid impingement upon the hot walls. However, the propellant mass can only be effective if sufficient heat exchange occurs with the vapor. Such conditions may not exist because of the deliberately low velocity of entering liquid. Thus, it is possible that an inlet diffuser configuration may not be able to satisfy the conflicting requirements of both a low velocity, (to assure a large propellant mass in the tank), and a high velocity (to provide effective quenching by that liquid mass). Should venting be required because of insufficient heat exchange between liquid and vapor, liquid may be lost overboard in the process.

Once tank fill is initiated, the spray nozzle will be preferable to an inlet diffuser because of the turbulent conditions that will serve to maintain low pressure.

The choice is between a spray nozzle, which will result in venting during prechill, or an inlet diffuser, which may not require venting, but should it occur, could result in liquid lost overboard. The spray nozzle configuration was selected because prechill losses with this concept will be insignificant compared to other losses (Table 3-5).

3.3.5 MECHANICAL MIXERS TO ASSIST PROPELLANT REFILL. The tank fill analysis of Section 3.3.3.2 described how the key to a successful refill operation would be the turbulent environment created by the entering fluid. Work done on the ullage mass during propellant transfer will be convected to the liquid via the heat exchange mechanism set up by fluid turbulence. If, however, the turbulent heat transfer rates should be inadequate, propellant transfer would have to be interrupted, in order to prevent the continuation of a high pressure rise rate. It is possible that a long time duration would then be required to effect a pressure reduction and to approch near-equilibrium conditions. This delay would occur because flow termination would also reduce the ullage-to-liquid heat transfer process. The consequence of an inadequate fluid turbulent environment could be an undesirably long tank fill process, caused by numerous flow interruptions.

Analyses to date indicate that near-thermal equilibrium conditions can be achieved for a wide range of tanking flowrates. Nevertheless, a backup position should be available in the event that of an excessive tank pressure increase. The solution would be to use a mechanical mixer to provide the additional fluid turbulence needed to achieve near-equilibrium conditions. Mixers will already exist since they are an integral part of the zero-g vent system required for each propellant tank.

3.3.5.1 Mixer power relationship to liquid-ullage heat exchange. The heat exchange mechanism that will exist during periods of mixer operation is that of vapor-bubble dominance described in Section 3.3.3.2. Equations 3-35 and 3-36 apply, except that the original expressions from Reference 3-6 were employed. That is, the equations included an input power term rather the fluid power term.

Figures 3-38 and 3-39 give ullage-to-liquid heating rates for hydrogen and oxygen as a function of mixer power, percent liquid fill, and vapor hold-up (i.e., the percent vapor entrained in liquid). Referring to Figure 3-31, it is seen that the required heating rate can be achieved for a power input of less than 4 watts. Note that input power requirements to achieve a given heating rate are a strong function of vapor hold-up and a very weak function of the percent liquid fill condition. According to Figures 3-40 and 3-41, vapor bubbles generated within the OTV propellant tanks by a mixer (or its equivalent in fluid power) are a function of fluid power for power levels less than about 12 watts.

The mixer power requirements identified by Figures 3-38 and 3-39 are within the range currently being considered for zero-g vent system mixers. Consequently this propellant transfer assist will be available without expending additional resources.

3.3.5.2 Mixer power/fluid power equivalence. Analyses to date include an implied assumption that tank fill durations will be selected on the basis of heat and mass transfer considerations. That is, spray nozzle, flowrate, and velocity solutions will be made to assure thermal equilibrium throughout tank fill. It is possible, however, that flowrate constraints may be imposed by other factors. For example, fluid

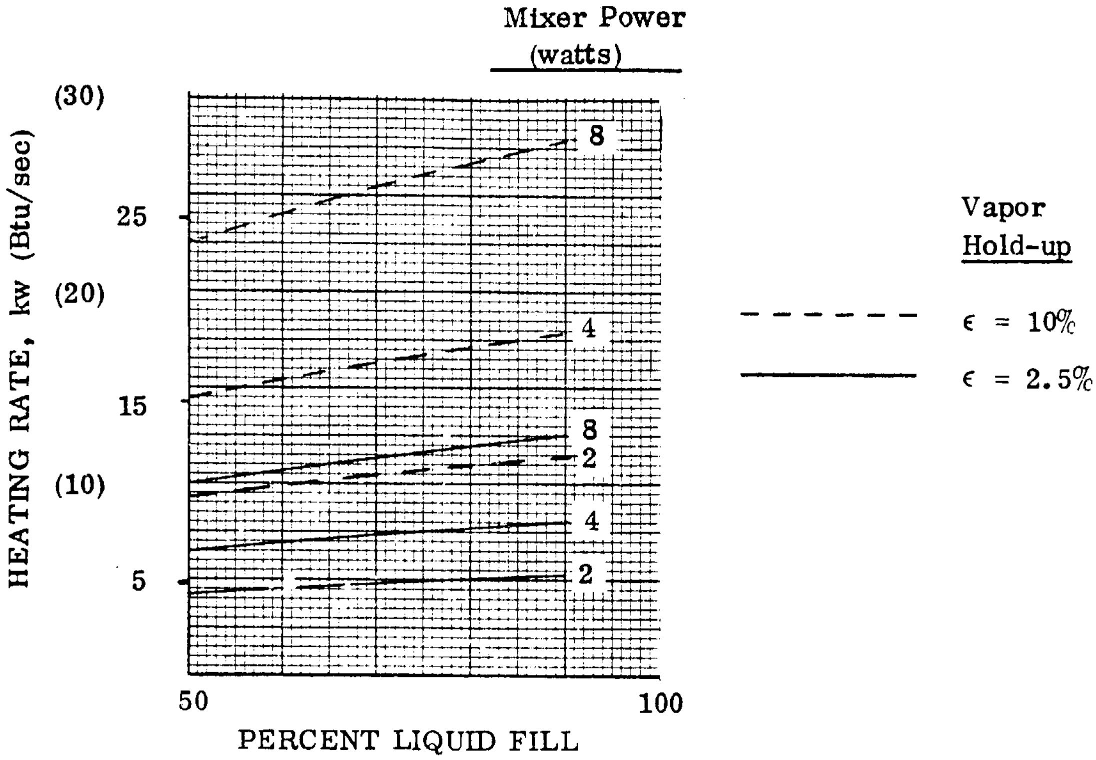


Figure 3-38. Mixer power influence upon entrained vapor-to-liquid hydrogen heat transfer rate.

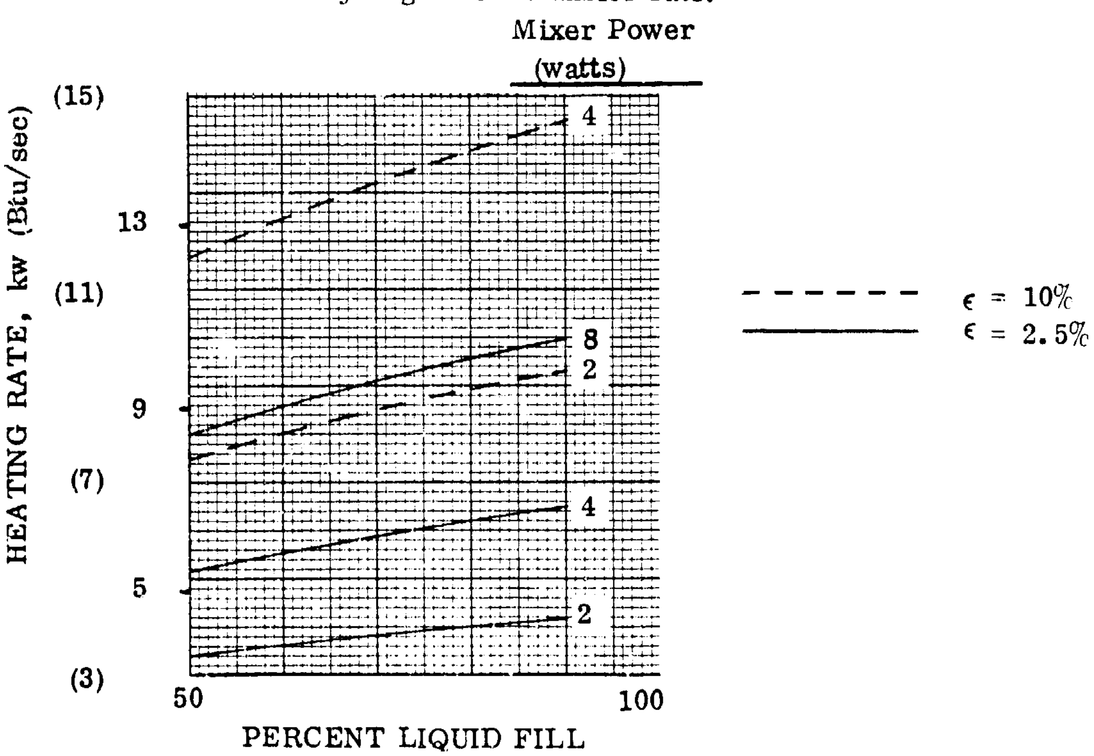


Figure 3-39. Mixer power influence upon entrained vapor-to-liquid oxygen heat transfer rate.

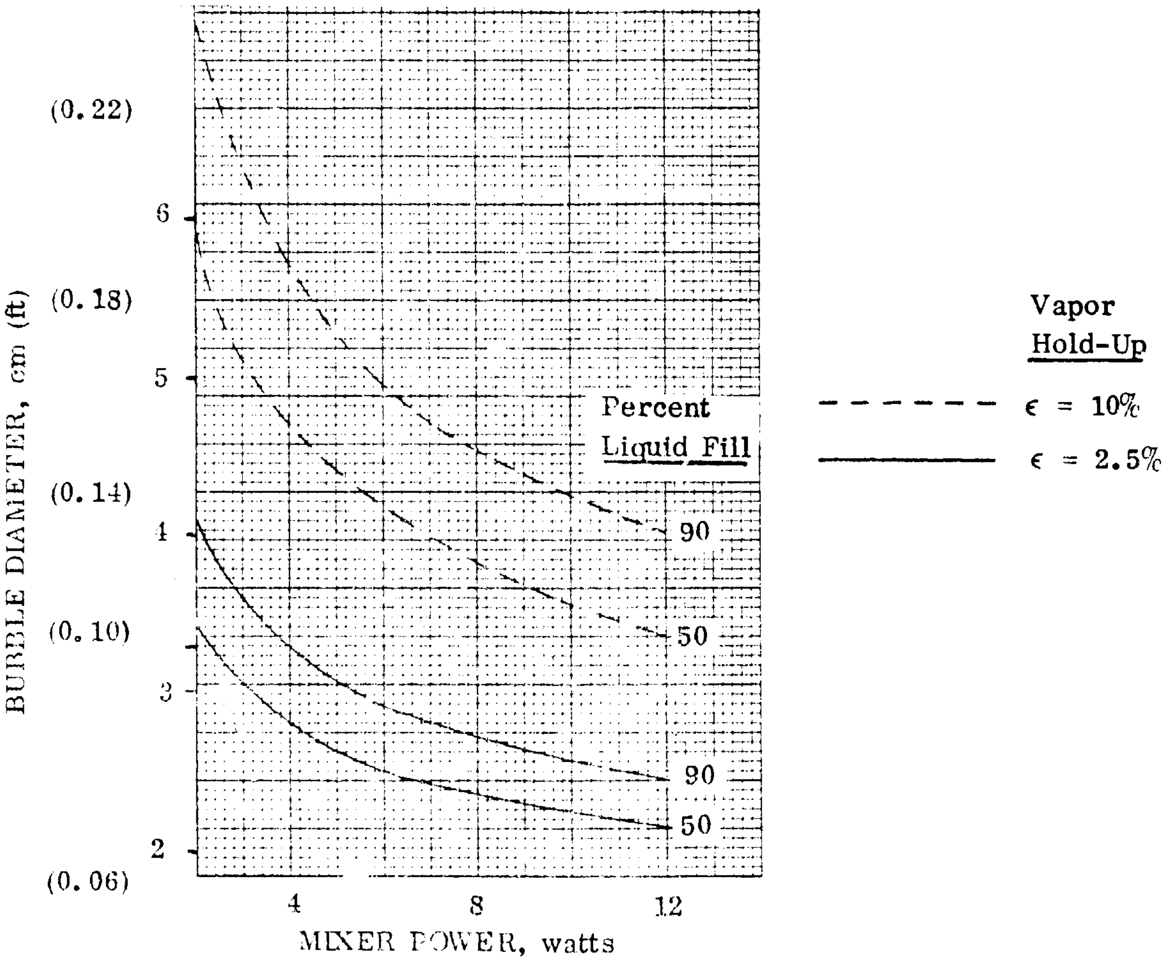


Figure 3-40. Mixer power influence upon hydrogen bubble diameter during tank fill.

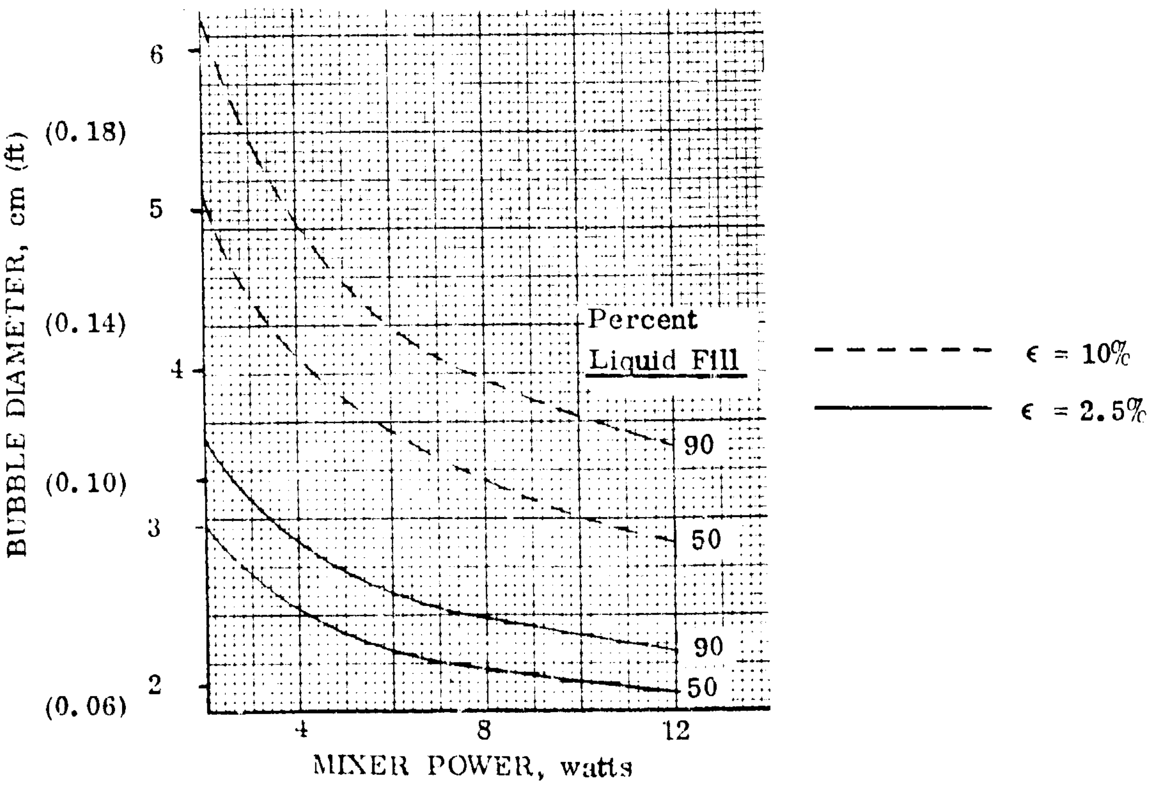


Figure 3-41. Mixer power influence upon oxygen bubble diameter during tank fill.

loads acting on the transfer lines and/or propellant tanks may preclude tank fill durations of less than five or six hours. It might not be possible to maintain low tanking pressures at these lower flowrate without contin... us mixer power assist. For this circumstance the required fluid turbulence would be provided with a combination of fluid power and mixer power. This combination could be selected from curves similar to those given in Figures 3-42 and 3-43. An example of how these figures can be applied is given below:

1. Assumptions
   - The fluid power equivalence of 15 watts mixer power is required during tanking to maintain near-equilibrium conditions.
     - Other considerations require that tank fill be performed in six hours using a 6.35 cm (2.5 inch) diameter equivalent nozzle inlet.
2. LH_2 Tank Solution
   - Fluid power = 2.75 watts
   - Mixer Power = 15.00 - 2.75 = 12.25 watts required
3. LO_2 Tank Solution
   - Fluid power = 1.75 watts
   - Mixer power = 15.00 - 1.75 = 13.25 watts required.

3.3.6 START BASKET REFILL. If propellant screen acquisition devices (start baskets) are included as POTV subsystems, an additional complication must be addressed in selecting an on-orbit refill concept, that of completely refilling the start baskets without trapping vapor. Vapor entrapment is unacceptable because pure liquid flow from the start basket is normally required, and cannot be guaranteed unless the screen device is free of vapor. Vapor will be present at some time within the start basket regardless of the method of propellant fill. Any trapped vapor bubbles must subsequently be condensed by using pressurant. The type of pressurant can either be helium (non-condensible) or vaporized propellant (condensible). The question of trapped helium bubbles will be discussed in Section 3.3.6.3.

Two methods of vapor bubble collapse were evaluated; passive and active. In each case the propellant tank will be pressurized to sub-cool the liquid surrounding the entrapped vapor, so that vapor condensation will occur. The passive method assumes that conduction heat transfer is the only mechanism that exists to cool and subsequently condense the vapor. The active method relies upon forced convection heat transfer to condense the entrapped vapor. An evaluation of each method follows.

3.3.6.1 Passive method of bubble collapse. Start basket designs, generated for POTV during the Reference 3-2 study, were analyzed to determine if vapor-free refill would be possible. A sketch and start-basket-dimensions are given in Figure 3-44 and Table 3-10. A very conservative assumption was made for this analysis. The start basket would be filled with a single large bubble at the end of the on-orbit

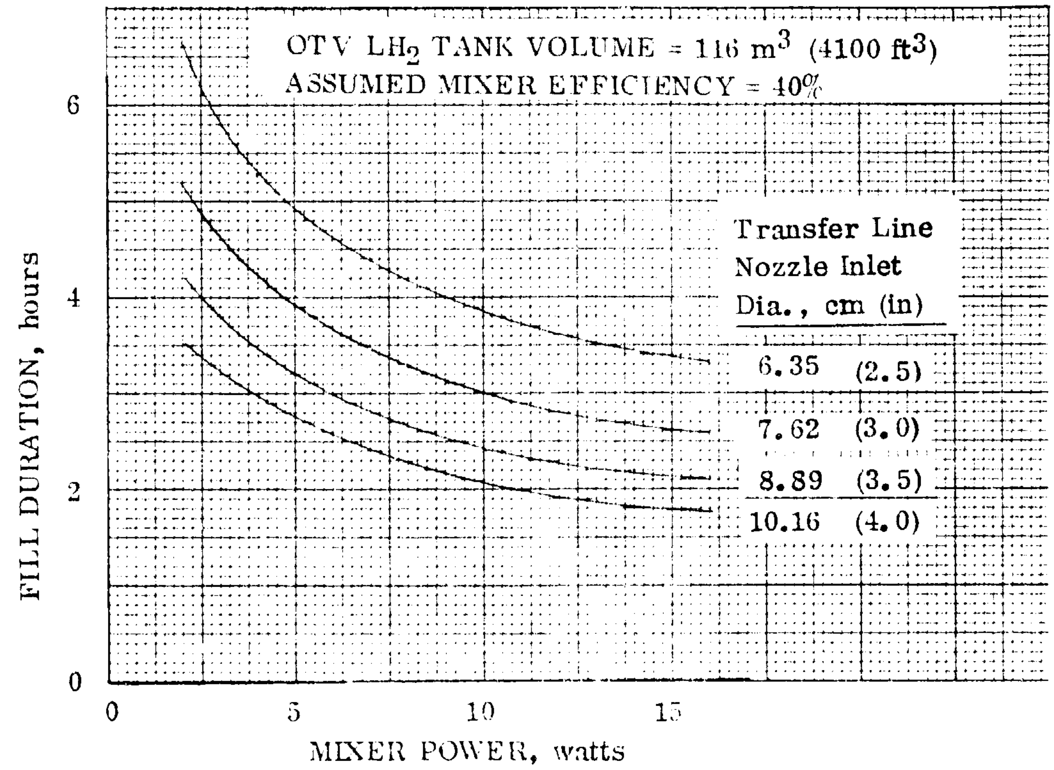


Figure 3-42. Fluid Power Input Equivalence to Mixer Power During OTV LH_2 Tank Fill

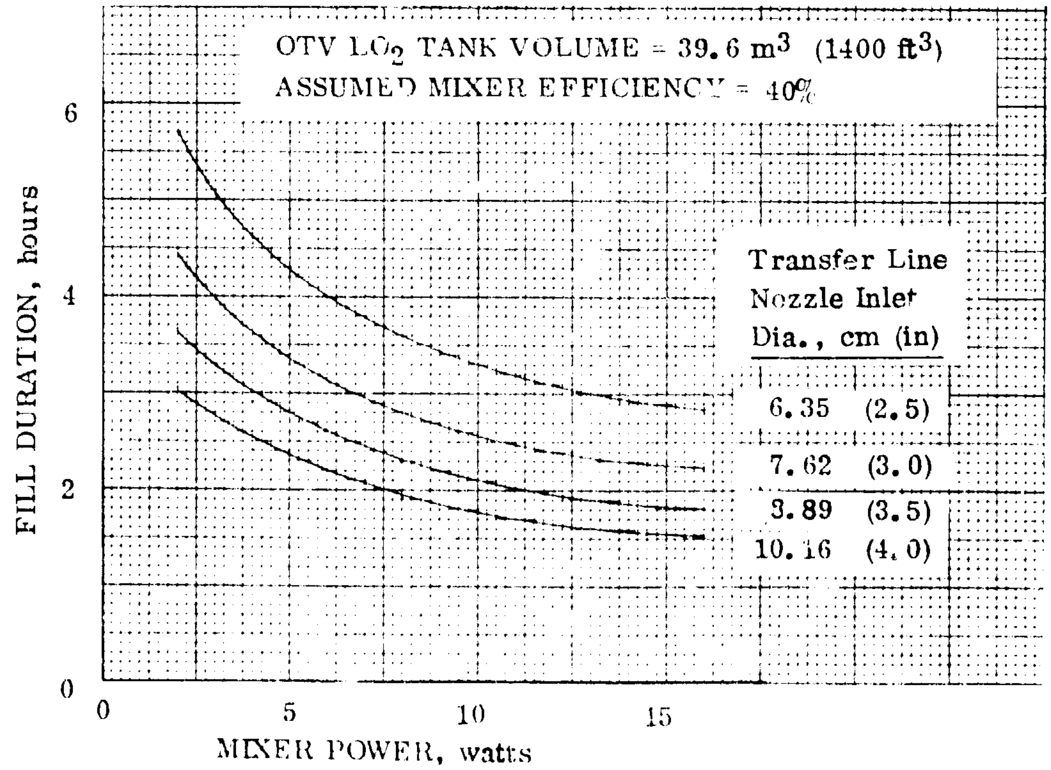


Figure 3-43. Fluid Power Input Equivalence to Mixer Power During OTV LO_2 Tank Fill

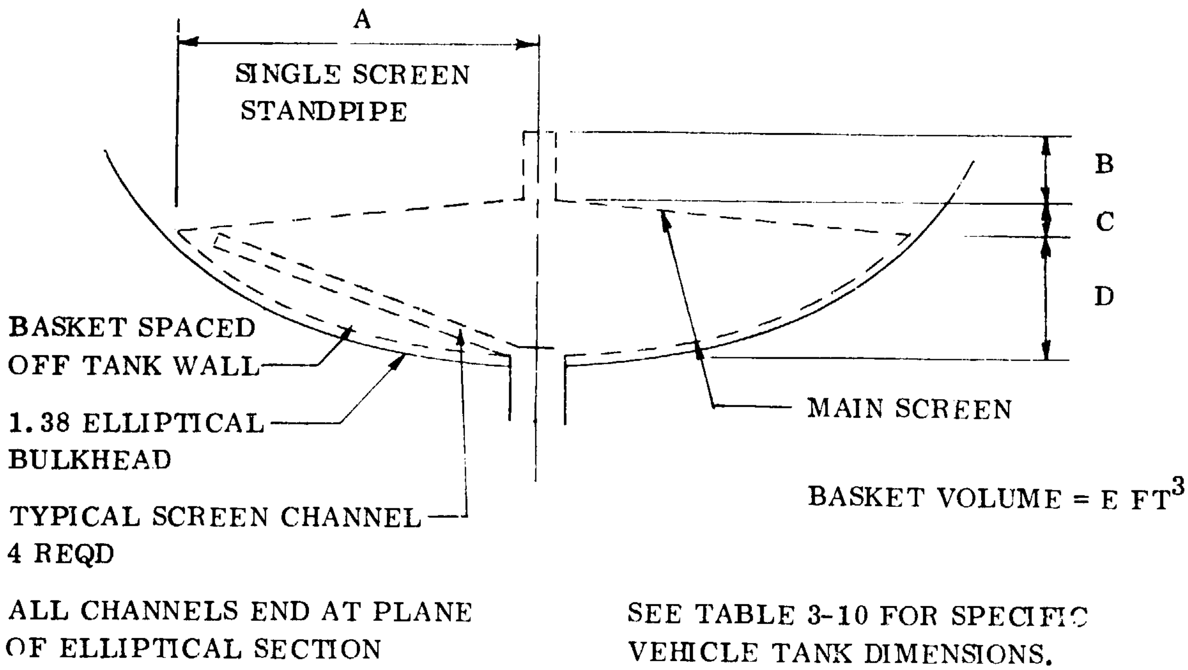


Figure 3-44. Start basket schematic.

Table 3-10. POTV start basket characteristics.

	LH_2 Tank	LO_2 Tank
A. Radius, cm (in)	142.2 (56.0)	85.1 (33.5)
B. Standpipe height, cm (in.)	11.9 (4.7)	4.6 (1.8)
C. Cone height, cm (in.)	16.5 (6.5)	9.9 (3.9)
D. Ellipsoidal height, cm (in.)	42.2 (16.6)	17.5 (6.9)
E. Basket volume, m^3 (ft^2)	1.92 (67.8)	0.34 (11.9)

refill operation. The largest spherical bubble that can be trapped within each screen device was computed to be about 0.61 m (2 ft) and 0.3 m (1 ft). An analysis was conducted to determine the time for each bubble to collapse.

Bubble condensation times were determined using bubble collapse equations described in Reference 3-3. Computer output results are given in Figure 3-45 and 3-46 for hydrogen and oxygen, respectively. The oxygen bubble will collapse in about 5 minutes if it is subcooled by about 20.7 kN/m^2 (3 psid), which is an acceptable maximum helium partial pressure during refill. The hydrogen bubble will require about three to four hours to collapse at the same degree of subcooling, which may be an unacceptably long duration. These calculated collapse times should be greater than actual collapse times because of the conservatively large bubble sizes selected. It should also be mentioned that the bubble collapse predictions are subject to added uncertainty because the analytical model was developed for small diameter bubbles.

3.3.6.2 Active method of bubble collapse. If vapor-bubble collapse times which use the passive method are excessive, an alternative is to use an active method to greatly decrease bubble collapse times. This method requires that propellant be sprayed into the start basket during tank fill. The fluid agitation induced by entering propellant will create a high heat exchange mechanism equivalent to that occurring outside the start basket during propellant fill. Furthermore, this turbulent fluid condition will serve to create small vapor bubbles which will greatly enhance the condensation process.

A model has been developed to determine the conditions under which the condensation of all propellant vapor within a start basket will occur. The model is based upon the following assumptions:

1. Liquid inflow velocity and flowrate are known.

2. Only liquid will exit the start basket; vapor is removed only by condensation.

3. Liquid enters the basket at a constant temperature.

4. Vapor trapped within the basket is in spherical bubble form (Figure 3-47). Thus, this model is valid only for vapor volume conditions of about 40 percent or less by basket volume.

5. Bubble diameter is obtained from equation 3-36,

$$d = \frac{1.134\, \sigma^{0.6}\, \epsilon^{0.5}}{\gamma^{0.2} (mv^2/V_L)^{0.4}} + .09$$

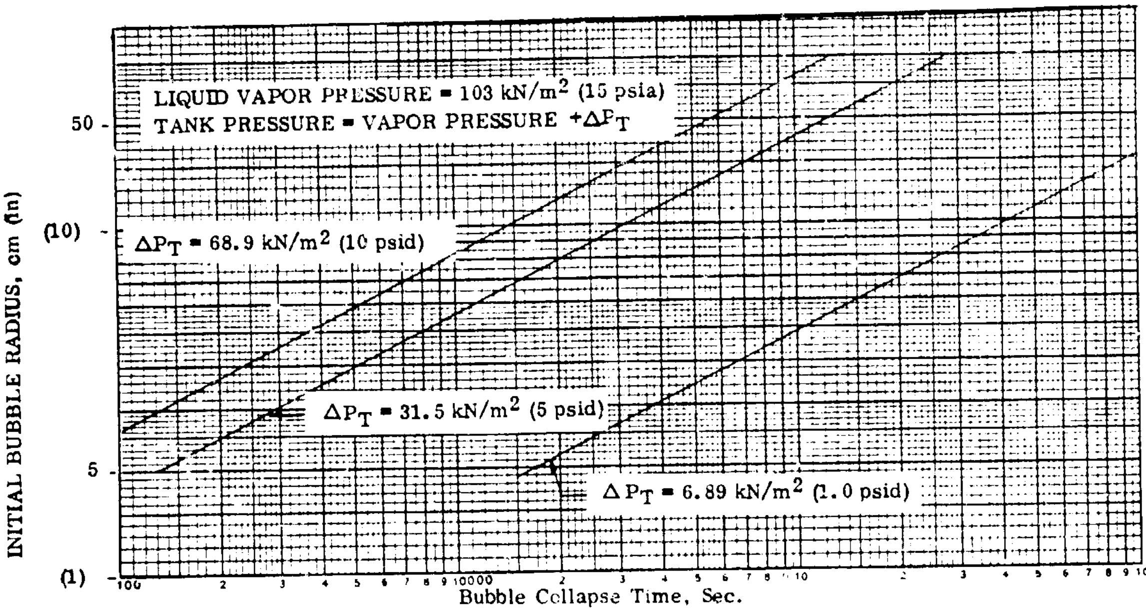


Figure 3-45. Collapse Times for Spherical Bubbles in Liquid Hydrogen

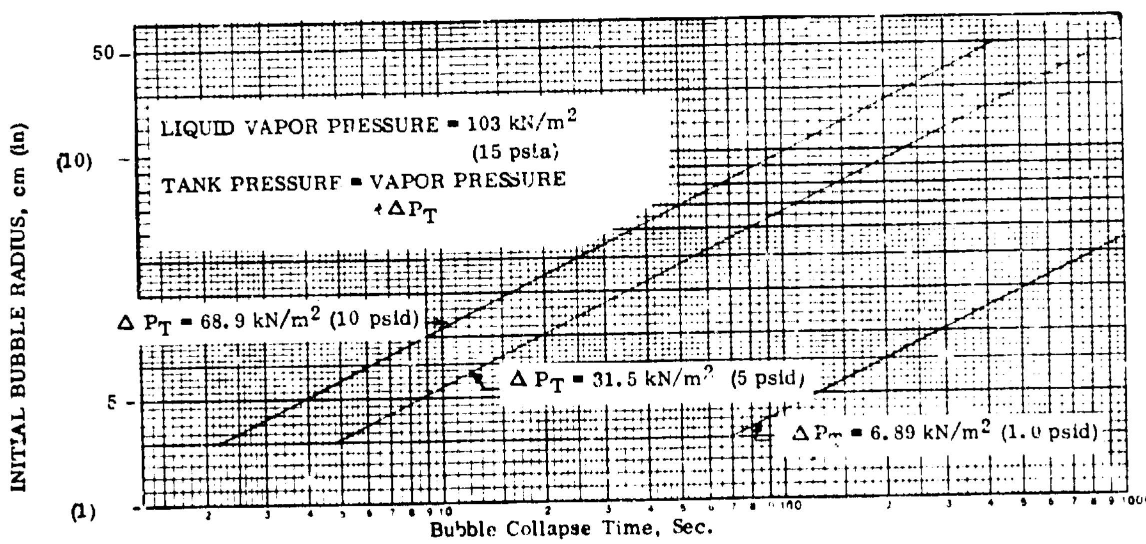


Figure 3-46. Collapse Times for Spherical Bubbles in Liquid Oxygen

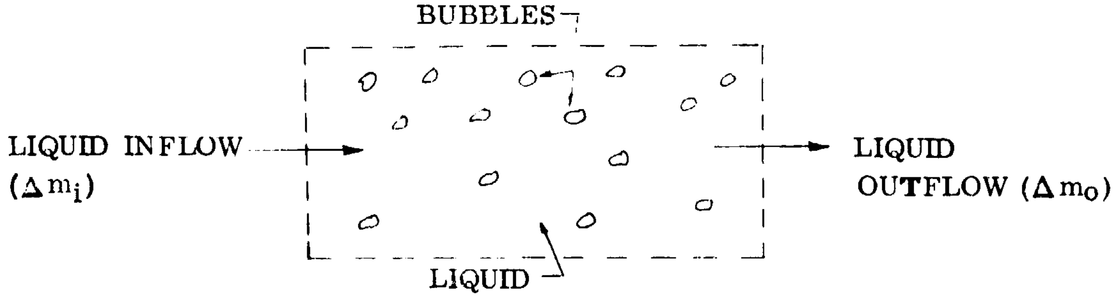


Figure 3-47. Active Method for Start Basket Bubble Collapse

where

d = bubble diameter, cm

σ = surface tension, dyne/cm

ϵ = entrained vapor volume to basket volume ratio

γ = liquid specific gravity

V_L = $(1 - \epsilon)\, V_T$ = liquid volume within basket

V_T = basket volume

Substituting $(1 - \epsilon)\, V_T$ for V_L gives,

$$d = \frac{1.134\, \sigma^{.6} (1 - \epsilon)^{.4}\, \epsilon^{.5}}{\gamma^{.2} (\dot{m}v^2/V_T)^{.4}} + .09 \qquad (3\text{-}39)$$

Figures 3-48 and 3-49 give plots of hydrogen and oxygen bubble diameter as as a function of $\dot{m}v^2/V_T$ and ϵ.

6. Bubble condensation rates are extremely high for the small bubble diameters anticipated (see Figures 3-50 and 3-51). Consequently, it is expected that liquid within the basket will rapidly saturate at tank pressure due to absorbing the heat of condensation.

7. All liquid leaving the basket will exit saturated at tank pressure.

8. Vapor bubble pressure = tank pressure throughout tank fill.

An analytical model that includes the above list of assumptions is given below:

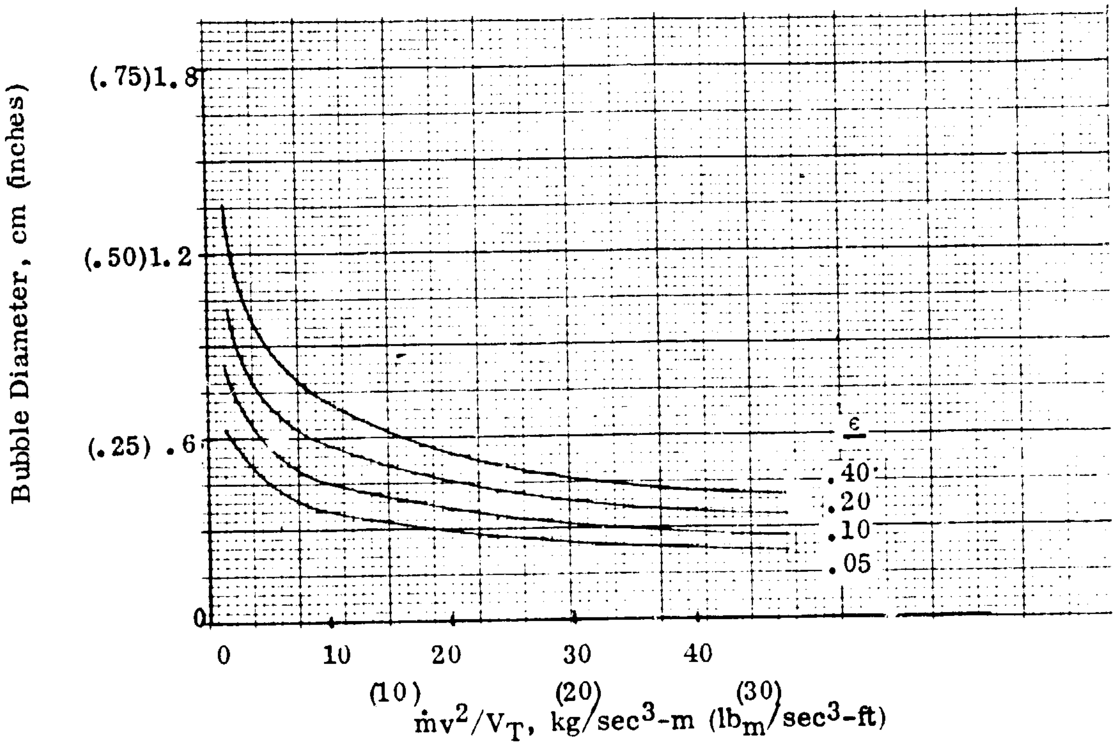


Figure 3-48. Hydrogen Vapor Bubble Diameter During Start Basket Refill.

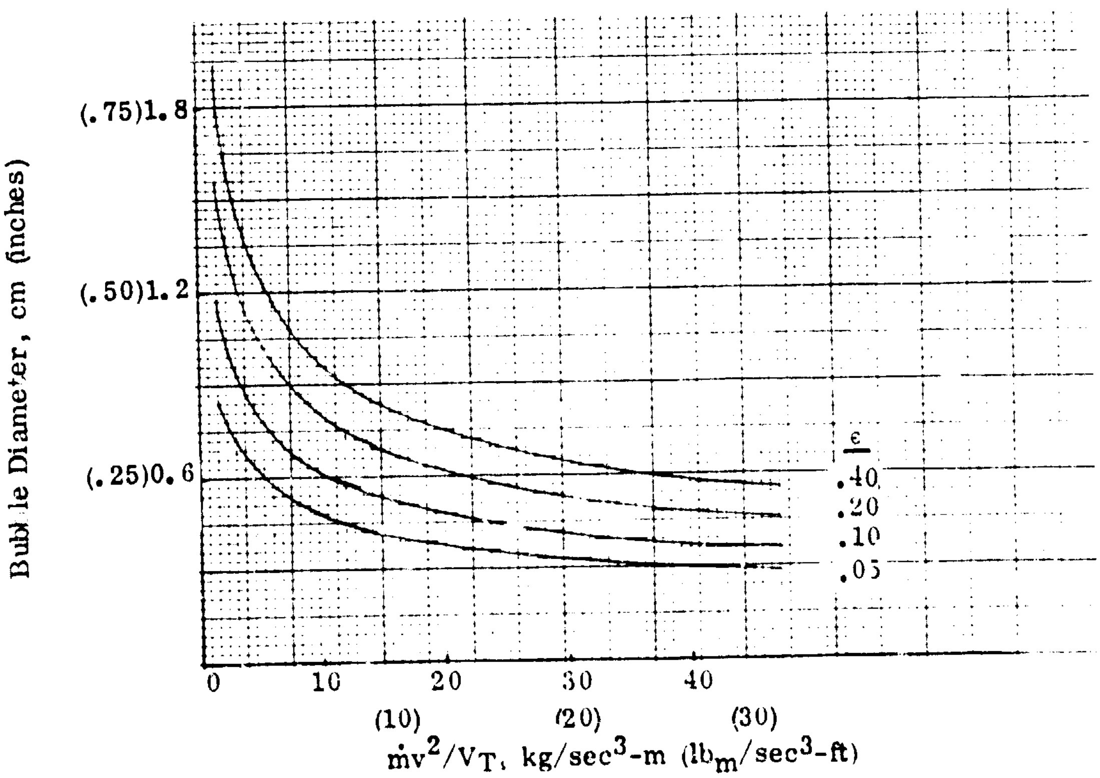


Figure 3-49. Oxygen Vapor Bubble Diameter During Start Basket Refill.

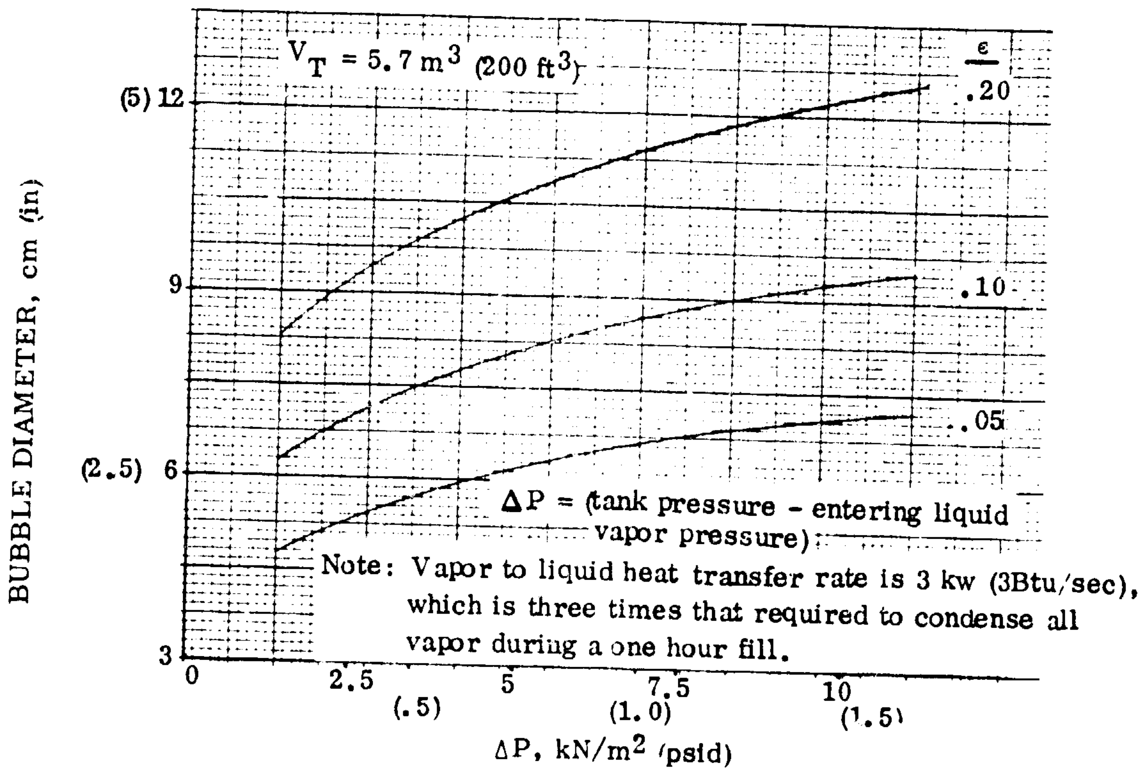


Figure 3-50. Maximum Allowable Bubble Diameter for Condensing Hydrogen Vapor in Start Basket During POTV Propellant Tank Fill

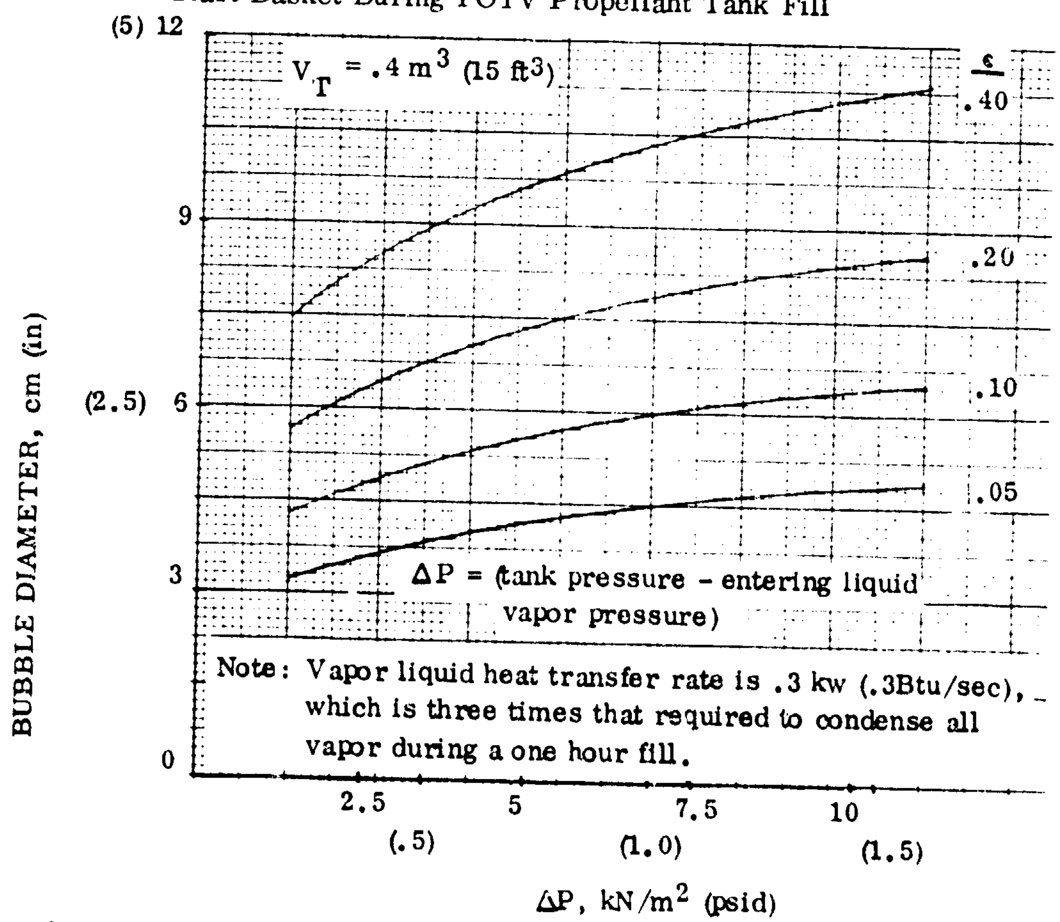


Figure 3-51. Maximum Allowable Bubble Diameter for Condensing Oxygen Vapor in Start Basket During POTV Propellant Tank Fill

First Law

$$(mu)_{2L} + (mu)_{2v} - (mu)_{1L} - (mu)_{1v} = h_i \Delta m_i - h_o \Delta m_o \quad (3\text{-}40)$$

where

m	=	mass
u	=	internal energy
h_i	=	enthalpy of entering liquid mass increment
h_o	=	enthalpy of exiting liquid mass increment
Δm_i	=	liquid mass increment entering tank in time,
Δm_o	=	liquid mass increment leaving tank in time,

subscript

2	=	conditions at end of interval
1	=	conditions at start of interval
L	=	liquid
v	=	vapor

Now, $u_{2v} = u_{1v} = u_v$ (vapor is saturated at tank pressure, assumption 8)

Also, $u_{2L} = u_{1L} = u_L$ (assumption 6)

Therefore,

$$(m_2 - m_1)_L u_L + (m_2 - m_1)_v u_v = h_i \Delta m_i - h_o \Delta m_o \quad (3\text{-}41)$$

From continuity, $m_{1L} + m_{1v} + \Delta m_i = m_{2L} + m_{2v} + \Delta m_o$ (3-42)

or, $m_{2T} - m_{1T} = \Delta m_{BASKET} = \Delta m_i - \Delta m_o$ (3-43)

where $m_T = m_L + m_v$ (3-44)

Δm_{BASKET} = propellant mass increment accumulated in start basket

Adding and subtracting $m_{2v} u_L$ and $m_{1v} u_L$ to equation 3-41 results in,

$$(m_{2T} - m_{1T}) u_L + (m_{2v} - m_{1v}) u_{ev} = h_i \Delta m_i - h_o \Delta m_o \quad (3\text{-}45)$$

where

$$u_{ev} = (u_g - u_L)$$

But,

$$m_{2v} - m_{1v} = - \frac{(m_{2T} - m_{1T})}{(\rho_L/\rho_g) - 1} \qquad (3\text{-}46)$$

Therefore, equation (3-45) becomes

$$(m_{2T} - m_{1T})\left(u_L - \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right) = h_i \, \Delta m_i - h_o \, \Delta m_o \qquad (3\text{-}47)$$

Substituting equation (3-43) into (3-47) gives,

$$(\Delta m_i - \Delta m_o)\left(u_L - \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right) = h_i \, \Delta m_i - h_o \, \Delta m_o \qquad (3\text{-}48)$$

Rearranging terms and solving for $\Delta m_o / \Delta m_i$ we have,

$$\frac{\Delta m_o}{\Delta m_i} = \left[h_i - u_L + \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right] \Big/ \left[h_o - u_L + \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right] \qquad (3\text{-}49)$$

However, from equation (3-43) we can show that

$$\frac{\Delta m_{BASKET}}{\Delta m_i} = 1 - \frac{\Delta m_o}{\Delta m_i} \qquad \text{or,}$$

$$\frac{\Delta m_{BASKET}}{\Delta m_i} = 1 - \left[h_i - u_L + \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right] \Big/ \left[h_o - u_L + \frac{u_{ev}}{(\rho_L/\rho_g) - 1}\right] \qquad (3\text{-}50)$$

Equation (3-50) gives the maximum fraction of entering propellant that will accumulate within the start basket as a result of vapor condensation. This fraction is shown in Figure 3-52 for hydrogen and oxygen as a function of the differential pressure term, ΔP (tank pressure minus entering liquid vapor pressure). These curves indicate that an increasing fraction of entering liquid remains in the basket as ΔP increases. An increase in ΔP is equivalent to liquid entering the basket at cooler temperatures, which increases its heat absorbing capability. This in turn will increase the rate of vapor condensation, and subsequent liquid accumulation within the basket.

According to Figure 3-52, 100% of LO_2 entering the basket will remain within the basket, if the entering liquid is subcooled by about 6.8 kN/m^2 (1.0 psid), or greater. For this condition one could select a flowrate (m_o) that would just fill the basket at tank fill completion (Figure 3-53). A conservative, and preferred, approach would be to provide

$\frac{\Delta m_B}{\Delta m_i}$ = mass fraction of entering liquid which remains in start basket

$\dot{m}$ = $\dot{m}_o/(\Delta m_B/\Delta m_i)$

$\dot{m}_c$ = $\frac{\text{maximum basket liquid mass}}{\text{tank fill duration}}$

$\dot{m}$ = selected start basket flowrate

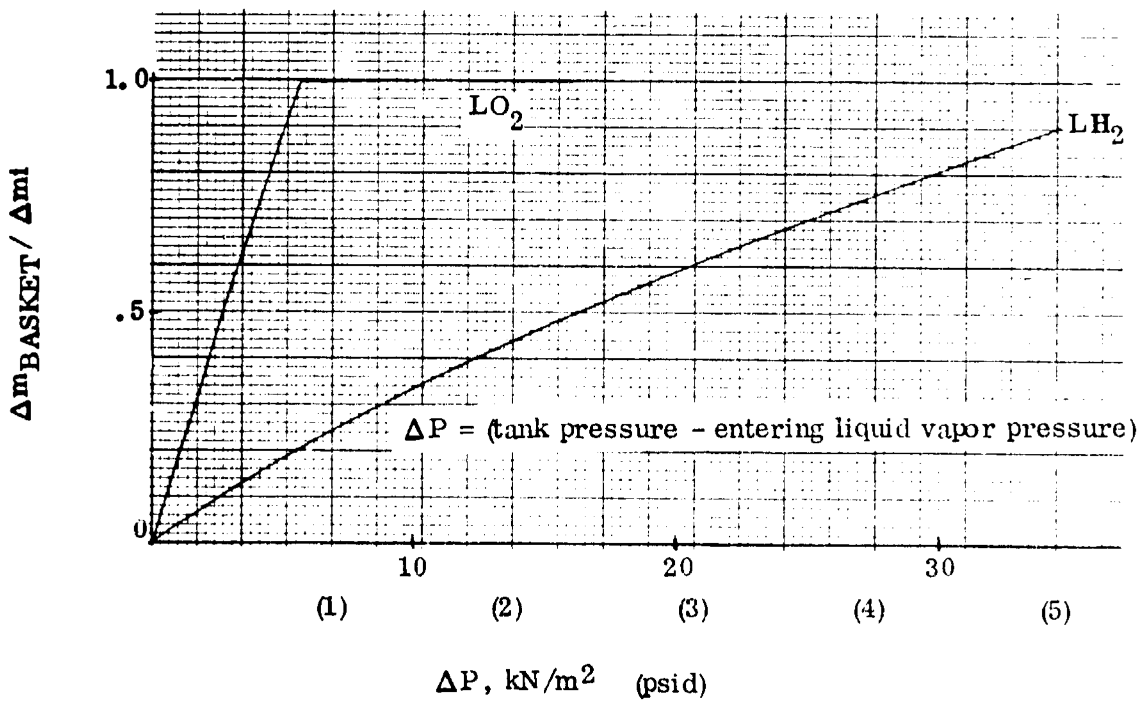


Figure 3-52. Influence of liquid subcooling upon start basket refill flow parameters.

a greater basket flowrate ($\dot{m}$) to assure basket refill even at ΔP's less than 6.9 kN/m^2 (1.0 psid). As a first approximation it is suggested that $\dot{m} = 3\,\dot{m}_o$. It is seen from the curve that the basket will completely refill for this flow condition at ΔP's greater than 1.7 kN/m^2 (0.25 psid).

Hydrogen properties are such that a considerably greater ΔP is required to accumulate the same fraction of LH_2 entering a basket than if LO_2 was entering a basket. This is illustrated by the hydrogen curve of Figure 3-52, which shows that about 40 kN/m^2 (5.7 psid) is needed for all entering liquid to remain within the basket. If, as above, we select $\dot{m} = 3\,\dot{m}_o$, then basket refill will occur for a ΔP greater than 10 kN/m^2

(1.4 psid). This condition should be satisfied since Figure 3-24 shows that final LH_2 equilibrium tank pressure for POTV will be about 21 kN/m^2 (3 psid) greater than entering LH_2 for initial tank wall temperatures less than 200K (360°R). The degree of liquid subcooling will be even greater if non-thermal equilibrium conditions occur during tank fill.

The above analysis has indicated that liquid inflow to a start basket is a feasible means of providing basket refill during the propellant tank fill process. A conservative thermodynamic mechanism has been analyzed which indicates that a reasonable range of propellant inflow rates will provide basket refill for POTV. In fact, a flowrate of $\dot{m} = 3\,\dot{m}_o$ should result in a completely filled start basket at the end of propellant tank fill.

3.3.6.3 Summary. The liquid-oxygen tank start basket will be vapor free within one-half hour after being pressurized by 20.7 kN/m^2 (3 psid). This same passive approach may require a considerably longer time for the liquid hydrogen tank. An alternative active-method was considered for the hydrogen tank in order to reduce the time required to condense all vapor. This condition will be satisfied at the end of propellant tank fill if hydrogen flows into the start basket at three times the rate indicated by Figure 3-53. This method requires the additional complication of a small diameter line plumbed between the main fill duct and the start basket. A preliminary design sketch of this set-up is given in Section 3.4.

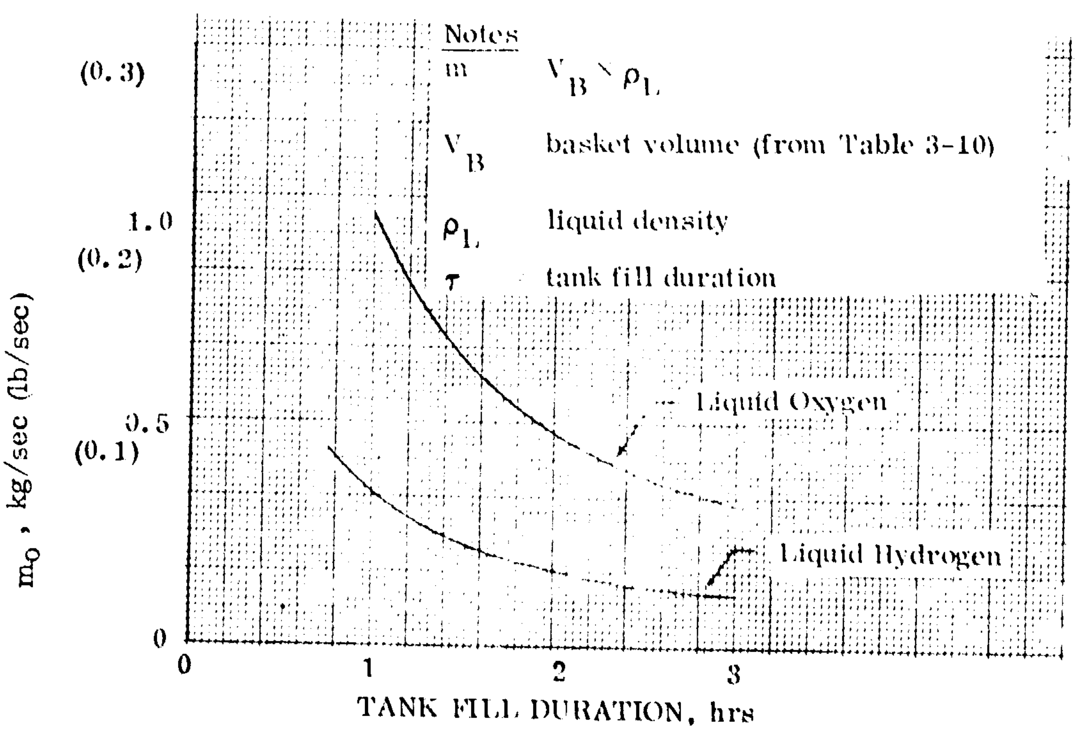


Figure 3-53. Propellant Flowrate Required to Fill Start Basket During Tank Fill Operations

3.4 ORBITAL PROPELLANT TANKING OPERATIONS

The impact of the on-orbit refill upon vehicle design is described in this section. Conceptual design modifications are presented which depict propellant tank, transfer line and helium system modifications. A detailed discussion of an on-orbit refill procedures is given, including how the procedures are influenced by the presence of helium and/or start baskets.

3.4.1 CONCEPTUAL DESIGN MODIFICATIONS FOR ON-ORBIT REFILL.

The analyses of Section 3.3 have established requirements for achieving POTV on-orbit refill from an orbiter-tanker. Requirements which affect the vehicle design include:

a. A fluid spray circuit through which propellants are introduced into the propellant tanks.

b. A bleed line to route LH_2 to the start basket during refill.

c. A non-propulsive vent system through which propellant vapor is vented during tank blowdown.

These and other plumbing modifications are depicted in Section 3.4.1.1. Section 3.4.1.2 and 3.4.1.3 includes sketches of recommended transfer line designs and a modularized helium pressurization system.

3.4.1.1 Propellant tank modifications. Figures 3-54 and 3-55 show some tankage system arrangements for a two stage POTV. Propellant refill is accomplished from the shuttle, therefore, the reaching capability of the transfer lines is limited to the length of the Remote Manipulating System (RMS). We have the choice of 1) loading each stage separately, followed by interstage attachment maneuvers, 2) having the stages initially interconnected and equipped with external plumbing kits or 3) adding a motor driven hinge system at the POTV separation plane so that the vehicle can be "jack knifed" to within reach of the RMS. It is assumed that each stage will be tanked separately, thus avoiding the crossing of the separation plane with propellant ducts. The systems shown are common to each stage and include propellant fill circuits vent systems and acquisition devices for both the fuel and oxidizer tanks.

The fuel tank fill circuit starts at the skin line of the POTV body structure with disconnect assembly containing an internal poppet closure, a static seal interface, and a cone section which serves as an alignment tool and a structural attachment with the mating half. A flexible duct section is routed from the disconnect to a fuel tank inlet fitting. The duct material is Cres and incorporates three axially restrained flex joints which permits length and angular changes between the duct ends.

The tank inlet fitting is equipped with a flange (inside the tank) for receiving a distribution duct, and a boss for attaching a 3/4-inch bleed line. The distribution duct runs along the cylindrical section of tank and is equipped with two outlets containing spray

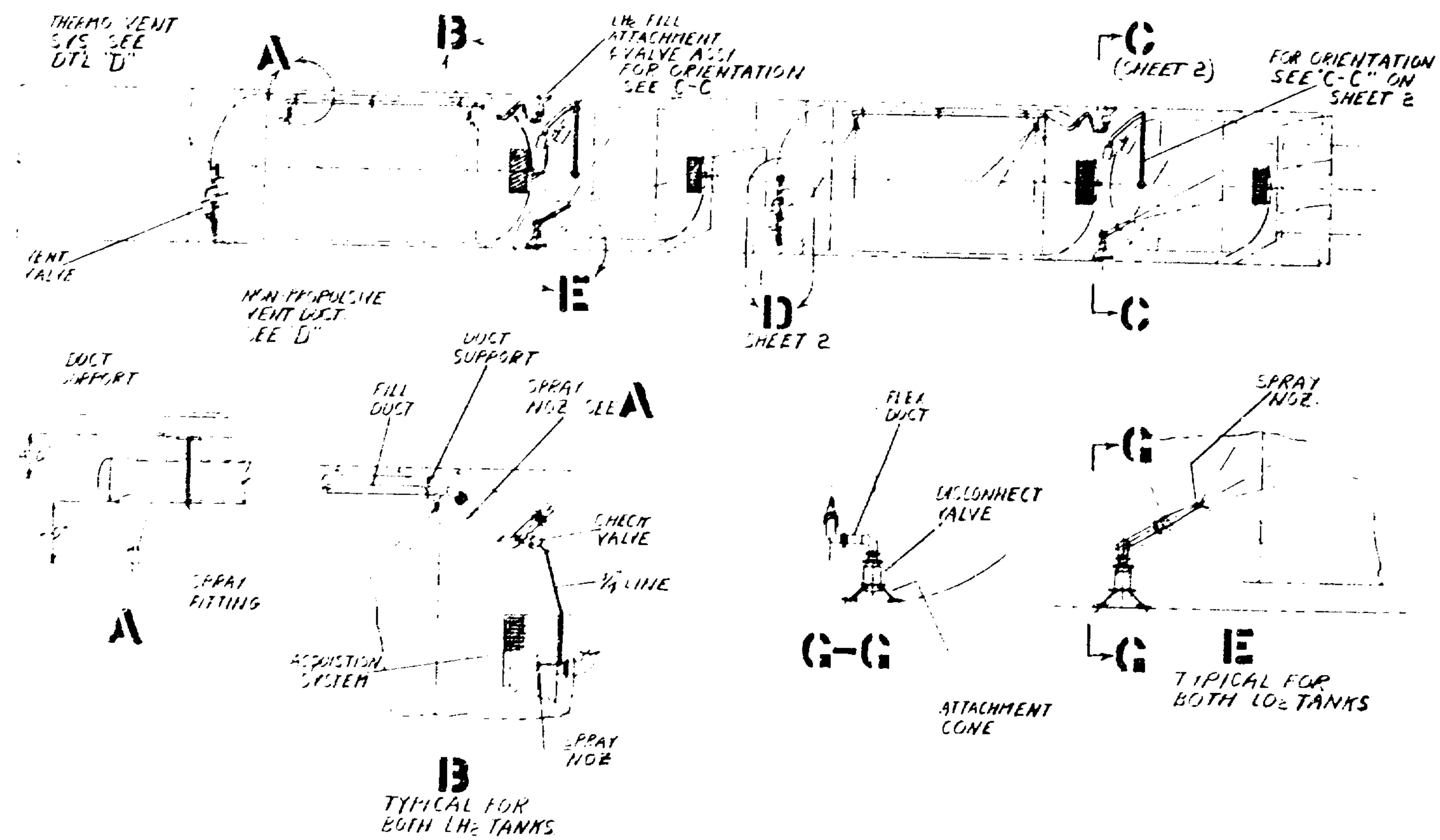


Figure 3-54. POTV Tankage Systems

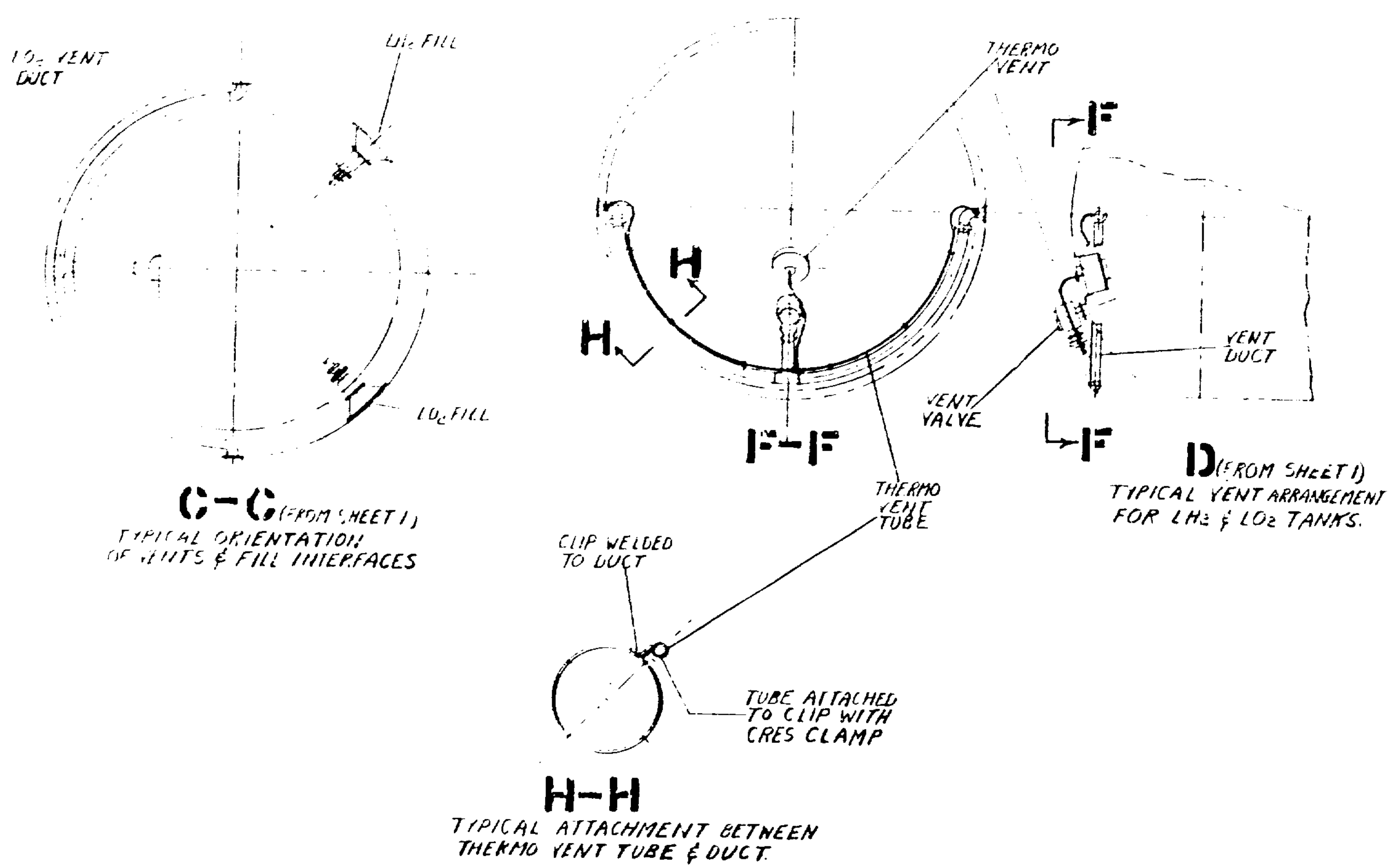


Figure 3-55. POTV Tankage Systems

nozzles. Two nozzles are shown in Figure 3-54. Although it is not possible to analytically determine how many nozzles will be required to assure adequate fluid mixing, it is believed that two or three nozzles will be satisfactory.

The 3/4-inch bleed line will direct LH_2 into the start basket to assure that all initially trapped vapor will be condensed by the completion of propellant fill.

The fill system for the oxidizer tank is basically the same as that for the fuel tank except the internal distribution duct has been replaced with a short tapered duct section equipped with a spray nozzle at the end. A bleed line to the acquisition device is not shown because calculations indicate that all vapor initially trapped within the basket will be condensed without resorting to liquid spray inflow.

Vent systems for both the fuel and oxidizer tanks consist of one vent valve located outside the tank wall, a non-propulsive vent duct, a thermodynamic vent system mounted off the forward bulkhead inside the tank, and a small vent tube for the thermodynamic vent system. The non-propulsive vent duct is routed from the vent valve to two points at the skin line and spaced 180° apart. The thermodynamic vent tube is supported from the vent duct and terminates near the duct ends.

3.4.1.2 Transfer line design. The transfer system is basically two separate circuits (one for LO_2 and one for LH_2) running from the POTV loading point back through to the orbiter supply tanks. Each circuit has a movable transfer line section connected to a fixed tank outlet line. The basic plumbing material is 304L cres.

Each movable transfer line consists of two sections interconnected with a tangential rotary joint. The forward section is equipped with a disconnect system for connecting to the POTV. The lines are deployed by the Shuttle RMS which engages with the disconnect valve. The transfer lines are supported at the base through tangential rotary joints attached to a support bridge forward of the fuel tank. In the stowed position the lines have additional support from the tank girth rings. Each line is also equipped with pneumatic tubes and electrical cables for actuating and monitoring the disconnect valve.

A disconnect arrangement for the POTV side only is included in Figure 3-56. The disconnect is designed so that the task of the RMS is to position the mating half approximately within the alignment cone. This mating half (attached to the RMS) has latch systems for gross capture followed by "draw down" and final alignment. The RMS, therefore, does not react disconnect loads. The arrangement shown is passive and consists of a cone and a disconnect assembly equipped with an internal spring loaded poppet and a flat external land for sealing with the mating half. The disconnect is attached to the cone using a ball-socket type fitting which permits angular misalignments. The mating disconnect assembly has an electromechanical drive which positions a pressure sensitive static seal against the flat land. The seal arrangement is shown in detail "C" of the figure. The electromechanically driven assembly is basically a short

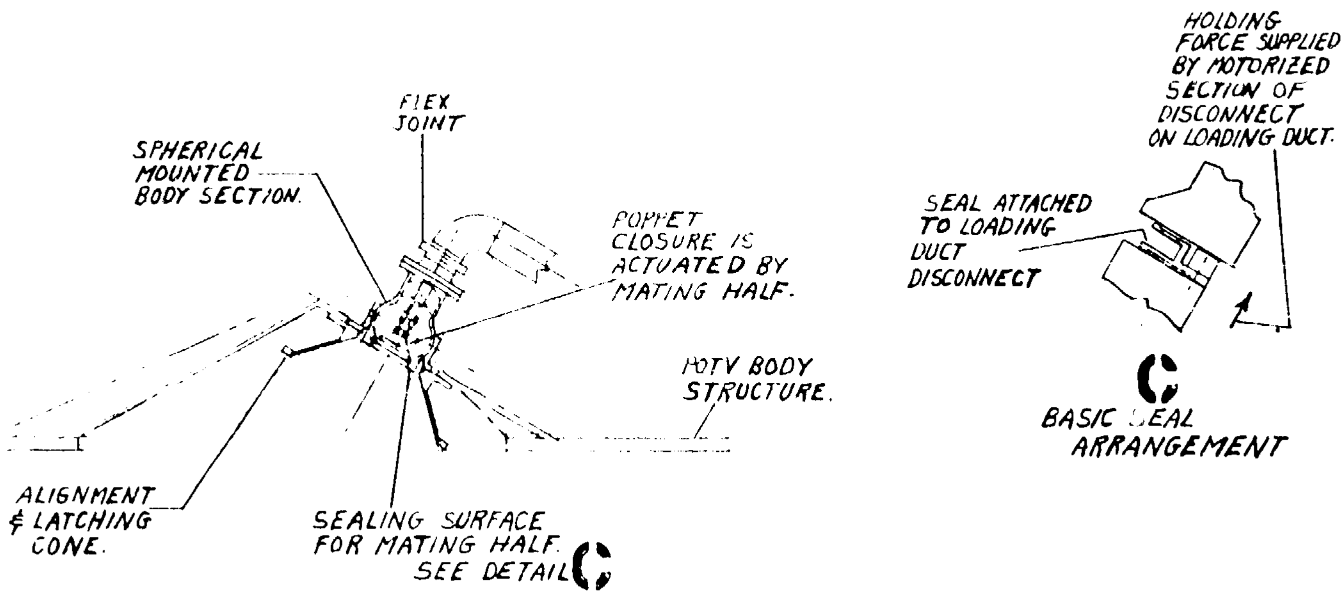


Figure 3-56. Typical vehicle propellant disconnect arrangement.

telescopic tube sealed externally with a bellows and actuated externally by three screw jack actuators or by a single threaded collar with a worm drive. GD/C has designed a liquid flourine tandem butterfly valve for zero spillage at separation using such techniques. For this application, the telescopic section would have an internal poppet closure similar to that shown in Figure 3-56.

In summary, a typical connection starts with the RMS positioning the transfer line disconnect inside the cone. Angular, axial and lateral misalignments are permitted at this time. The second step is to actuate the locking pawls on the transfer line disconnect to an inboard position which captures the flange on the cone. This is a loose engagement only between cones. On the third step the pawls are actuated parallel to the disconnect center line which pulls the two cones together thus completing the structural connection. Fourth, the motorized disconnect carrying the external seal is actuated pressing the seal against the flat land. Upon further actuation, the poppet on the transfer line disconnect forces the poppet shown in the layout to an open position. Fluid transfer is now permitted. For disengagement, the above steps are reversed. After fluid transfer both poppets are closed before the external seal is disengaged from the flat land. Also the poppet interfaces are designed for near zero spillage.

A detailed discussion of the design and operation of a transfer line disconnect valve is given in Section 5.4.1.

3.4.1.3 Helium system design. Space-based vehicles will require gaseous helium resupply during refueling operations. One method of resupply is by flowing helium from a storage tank (located in the Shuttle payload bay) to the vehicle through a long transfer line. However, analyses indicate that helium bottle charge operations would be lengthy if excessively high bottle temperatures are to be avoided.

The transfer line will have an electrical power cable attached to the line and permanently connected to the disconnect system located at the end of the line. The transfer line must have mobility which in turn requires flex joints either in the form of braided hoses, swivel type joints with dynamic seals, or loop bends in the tubing. Conventional bellows joints would not be applicable due to high pressure requirements. The use of hoses, swivel joints and loop bends presents packaging and weight penalties. Also, the safety of this transfer operation is of concern because an approximate 20 m (60 ft) line, pressurized to 34500 kN/m^2 (5000 psi), will be essentially unsupported except at the orbiter and at the vehicle. Consequently, an alternative method of resupply was selected.

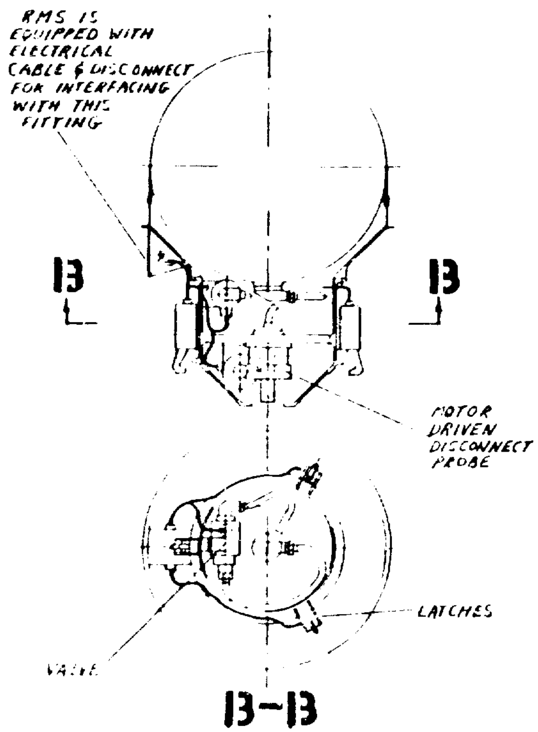
An alternative would be to employ separate pre-loaded modular helium bottles that would be externally attached to the vehicle. A disconnect system capable of mating a structural attachment and a fluid connection with the vehicle is required with either approach. Several disconnects are required of the modular bottles, however, as compared to only one for a transfer line.

The orbiter remote manipulating system (RMS) will be employed to connect transfer line or modular bottles to the vehicle. The RMS will place the disconnects reasonably close to the mating target. Pressure-area loads will be reacted only by the disconnect structure, and not by the RMS.

The helium bottle modules shown in Figure 3-57 are intended for use on POTV's, COTV's and LTL's serviced from the shuttle. Only the module station locations would be changed. Basically, the pre-loaded helium modules are picked up with an RMS and plugged into external OTV interfaces which in turn have interconnecting plumbing to the OTV systems.

Detail "B" of Figure 3-57 shows a typical helium module which is a high pressure bottle equipped with a tangential support skirt, a docking cone, a latching system, a shutoff valve and a motor driven disconnect. The tangential skirt section has a fitting which interfaces with the RMS. The only requirement in the case of a Shuttle RMS would be the addition of an end fitting. This fitting would include a power cable from the Shuttle for actuating the latch and disconnect systems.

A typical transfer sequence consists of connecting an RMS to the helium module; placing the module loosely (wide tolerances) into a mating cone on the OTV; actuating the latches to an inboard position to insure a gross capture and finally moving the latches in an axial direction which completes the structural connection. With the structural connection completed, the disconnect system is energized making a seal between module and vehicle. The socket portion of the disconnect contains the seals and the probe section on the OTV is float mounted to compensate for misalignments. Since temperatures are basically ambient, the seal system consists of "O" rings equipped with backup rings to prevent "blow out." The seal design includes provisions for each replacement.



ITEM (NO SCALE)	WT. LBS
SHUT OFF VALVE	10.0 INCL BRACKETS
PLUMBING	2.0 INCL FITTINGS
MOTORIZED DISCONNECT ASSY	15.0
WIRING	6.0
TOTAL	147.0

ITEM (NO SCALE)	WT. LBS.
BOTTLE	78.0
SUPPORT SKIRT	6.0
SUPPORT CONE	8.0
DOCKING CONE ASSY	12.0
MOTORIZED LATCHES	9.0
ATTACHMENT FITTING FOR RMS	1.0

Figure 3-57. Modular Pressurization System. (Applicable to POTV, COTV and LTL Vehicle)

3.4.2 ORBITAL PROPELLANT TANKING OPERATIONS. On-orbit refueling of a dual stage POTV will require three Shuttle flights, each having a payload capability of 45,360 kg(100,000 lb). The first Shuttle flight will transfer propellant to the second POTV stage. The second Shuttle flight will transfer propellant to the second stage. The third flight will carry the POTV payload and sufficient propellants for topping the two stages prior to mating ana launch.

Refill operations will be influenced by type of POTV subsystems and by whether it is the first or subsequent shuttle flight for the mission. These influences will first be evaluated before a detailed refueling procedure is presented.

3.4.2.1 Subsystem influence upon refill procedures. Two subsystems that will have a direct effect upon refill operations are the pressurization and propellant acquisition (start basket) systems. If helium is required for vehicle tanks pressurization, the refill process will become complicated because of the need to first expel the pressurant. The presence of propellant tank start baskets require procedural changes to accommodate the requirement that the screened volumes be vapor-free at the end of tanking. Table 3-11 indicates what elements of an orbital refill process will be affected by these variables. It is evident from this table that a vehicle having no helium pressurant and no start basket will be the simplest to refuel in space. The most difficult combination would be the inclusion of the two subsystems. This latter combination presents a potentially serious concern that is addressed below.

Start basket/helium combination. This subsystems combination can result in helium entrapment within the start basket at refill completion. Helium entrapment is a serious concern because 1) the start basket must be vapor-free in order function properly during the OTV mission and, 2) the entrapped helium, unlike propellant vapor, cannot be removed by condensation. The best solution is to perform a series of propellant tank blowdown modes, similar that described by Figure 3-13, until a negligible helium quantity remains. This would be a time-consuming process and is best performed prior to orbiter-rendezvous.

3.4.2.2 Shuttle flight influence upon refill procedure. Three shuttle flights will be used to refill the two POTV stages. Refill procedures for the first two shuttle flights will be identical because each stage will be empty. Thus the decision on how to handle liquid residuals, tank vent, and tank prechill will be the same. The third shuttle flight will provide the remaining propellants to complete tank fill. Obviously, the above tasks will not be performed. It will only be necessary to chill the transfer line prior to resuming the tank fill process. Transfer line childown may be performed differently for the third shuttle flight than for the first two flights. With the first two shuttle flights it is likely that vapor generated during childown will flow into the empty propellant tanks. Vapor generated during transfer line childown of the third shuttle flight will likely be vented overboard rather than into the propellant tanks.

Table 3-11. Subsystem Influence Upon Refill Procedures

Item	No Start Basket		Start Basket	
	No Helium	Helium	No Helium	Helium
Liquid Residual				
• Dump liquid prior to rendezvous	No	Yes	No	Yes*
Tank Vent				
• Tank Temperature > 200K (360R)	Yes	Yes	Yes	Yes*
• Tank Temperature < 200K (360R)	No	Yes	No	Yes*
Tank Prechill				
• Tank Temperature > 200K (360R)	Yes	Yes	Yes	Yes
• Tank Temperature < 200K (360R)	No	No	No	No
Tank Fill				
• Provide start basket bleed line	N.A.	N.A.	Yes	Yes
• Pressurize tanks for start basket vapor collapse	N.A.	N.A.	Yes	Yes

*A potential concern is that helium may be trapped within start basket. Solution is discussed in Section 3.4.2.1.

3.4.2.3 Tank fill procedures. On-orbit tank fill/refill procedures will be influenced by many variables including subsystem selection (see Table 3-11) and propellant (the oxygen tank requires no refill). It is not practical to prepare a table itemizing the individual procedural steps for each type of operations. Rather, a fill procedure has been developed for the most complicated case, which includes:

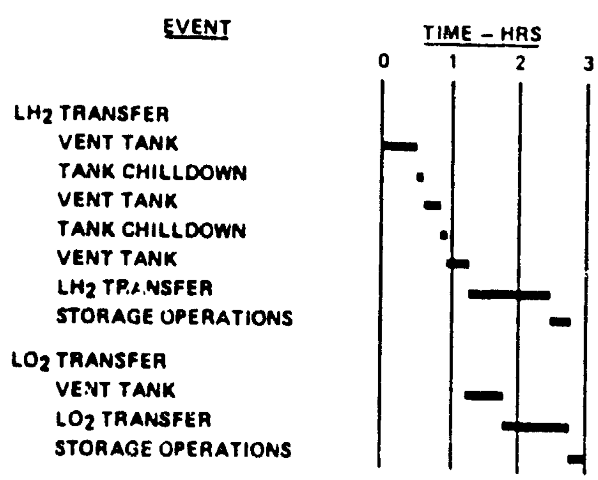
a) Propellant tanks, helium pressurization

b) Start baskets

c) Initial tank temperatures at 289 K (520R).

A total propellant transfer operation of three hours was selected. Table 3-12 shows that this transfer duration can be accomplished by over-lapping LH_2 and LO_2 transfer operations.

Table 3-12. Propellant Transfer Can be Accomplished Within Three Hours



POTV propellant transfer timelines for orbiter flights 1 and 2 are given in Table 3-13. In addition to the prechill and tank fill events, these timelines include the transfer line operations of attaching, venting, purging, disconnecting and returning to orbiter. LO_2 transfer operations are initiated 49 minutes into the LH_2 transfer operations. This overlapping is required because a single RMS is employed for both propellant tanks. LH_2 transfer timelines are determined for the tank fill conditions of Table 3-14, using the analysis of section 3.3. LO_2 transfer timelines are estimates extrapolated from the LH_2 analysis. A total of 176 minutes is required to complete propellant transfer.

POTV propellant transfer timelines for orbiter flight 3 are given in Table 3-15. The primary difference between this flight and the earlier flights is that 1) the propellant tanks are about 75 percent full, and 2) each propellant tank of each stage will be filled, which requires increased RMS activity. A total of 122 minutes is required to complete transfer operations.

The procedures listed in Tables 3-13 and 3-15 do not include all fluid transfer operations required of on-orbit refill. It is likely that helium and earth storable propellants will also be transferred. Transfer operations for these fluids were identified during the "Orbital Propellant Handling and Storage Systems Definition Study", Reference 3-1. An evaluation of these operations indicated that transfer times of 162 minutes and 222 minutes respectively, will be required for the modular helium bottles and N_2H_4 storage system refill. Additional details and assumptions related to the transfer of helium and N_2H_4 are given in Reference 3-1.

3.4.3 ZERO-G MASS GAUGING. One of the more serious problems with any propellant transfer process is to determine when to terminate tank fill. For the geosynchronous mission, each receiver tank must be filled to about the 97% level. Tank overfill must be avoided because of the potential for tank overpressure, but tank underfill could jeopardize mission success. Estimating propellant mass loaded with flow-meter measurements will not be sufficiently accurate. It is expected that a zero-g mass gauging device will be required. Liquid-level sensors will be useless in a low-g environment, because liquid and vapor will be distributed throughout the tank. Yet some technique for accurately measuring propellant mass quantities in this environment is essential to on-orbit propellant transfer.

3.4.3.1 Current Mass Gauging Devices. Technology studies and development engineering has previously been performed on the following mass gauging devices:

1. The TRW developed system based on absorption of gamma radiation.
2. The General Nucleonics radiation absorption system using Krypton 80 as the source.
3. The Bendix system, which uses a radio frequency (RF) technique.

Table 3-13. Propellant transfer timeline (Flights 1 & 2).

Event	Elapsed Time (min)	Cumulative Time (min)
TRANSFER LH_2 (1)		0
Vent OTV LH_2 tank	15	15
Attach LH_2 transfer line	8	23
Vent LH_2 transfer line	1	24
Close LH_2 transfer line	—	24
Close LH_2 tank	—	24
LH_2 transfer and thermal hold	5 (2)	29
Vent LH_2 tank	5 (2)	34
Close tank	—	34
LH_2 transfer and thermal hold	5 (2)	39
Vent LH_2 tank		44
Close vent	—	44
Transfer LH_2	90	134
Switch to topping rate	10	144
Sense vapor flow in supply tank	—	144
Close valves	—	144
Vent LH_2 transfer line	3	147
Purge LH_2 transfer line with He	5	152
Close LH_2 transfer line vent	—	152
Disconnect LH_2 transfer line	1	153
Purge LH_2 transfer line with He	2	155
Return LH_2 transfer line to Orbiter	8	163
TRANSFER LO_2 (2)	—	49
Attach LO_2 transfer line	7	56
Vent LO_2 transfer line	1	57
Close LO_2 transfer line vent	—	57
Hold	2	59
Vent OTV LO_2 tank	20	79
Close LO_2 tank	—	79
Transfer LO_2	60	139
Switch to topping rate	10	149
Sense vapor flow in supply tank	—	149
Close valves	—	149
Hold	8	157
Vent LO_2 transfer line	3	160
Purge LO_2 transfer line with He	5	165
Close LO_2 transfer line vent	—	165
Disconnect LO_2 transfer line	1	166
Purge LO_2 transfer line with He	2	168
Return LO_2 transfer line to Orbiter	8	176

(1) Timelines are based upon flow conditions given in Table 3-14.
(2) Timelines are estimates.

Table 3-14. LH_2 Propellant Transfer Flow Conditions Selected for POTV Refill Operations

Prechill(1)

LH_2 flowrate	=	.45 kg/sec (1.0 lb/sec)
LH_2 velocity	=	3.05 m/sec (10 ft/sec)

Tank Fill(2)

LH_2 flowrate	=	.91 kg/sec (2.0 lb/sec)
LH_2 velocity	=	6.10 m/sec (20 ft/sec)

Topping(3)

LH_2 flowrate	=	.45 kg/sec (1.0 lb/sec)
LH_2 velocity	=	3.05 m/sec (10 ft/sec)

(1) Prechill durations of Tables 3-13 and 3-14 are acceptable per Figure 3-21.
(2) Tank fill durations of Tables 3-13 and 3-14 are acceptable per Figures 3-30 and 3-34.
(3) Topping durations of Tables 3-13 and 3-14 are acceptable per Figures 3-30 and 3-34.

The RF type, whose development was also undertaken by the National Bureau of Standards, has an accuracy of about ±3 percent but tends to be geometry and configuration sensitive. The nucleonics type, developed by TRW and General Nucleonics, uses a radiation/detector device and has an accuracy of about ±2%. Resistance to the operational use of this system has been reported because of the potential radiation hazard. If the radiation were not a problem the system could be a very viable approach.

Although much progress has been made with these devices, system verification remains to be demonstrated on large scale systems in a zero-gravity environment. Until such time that one of the above devices has been proven for orbital propellant transfer, a search for alternative methods should continue.

Table 3-15. Propellant transfer timeline (Flight 3).

Event	Elapsed Time (min)	Cumulative Time (min)
TRANSFER LH_2 (1)		0
Attach LH_2 transfer line to 2nd stage	8	8
Vent transfer line	1	9
Close transfer line vent	—	9
Slow flow into transfer line	1	10
Transfer LH_2	25	35
Sense tank LH_2 mass	—	35
Switch to topping rate	10	45
Sense tank LH_2 mass	—	45
Close LH_2 valves	—	45
Attach LH_2 transfer line to 1st stage	8	53
Transfer LH_2	25	78
Sense tank LH_2 mass	—	78
Switch to topping rate	10	88
Sense tank LH_2 mass	—	88
Close LH_2 valves	—	88
Vent LH_2 transfer line	3	91
Purge LH_2 transfer line with He	5	96
Close LH_2 transfer line vent	—	96
Disconnect LH_2 transfer line	1	97
Purge LH_2 transfer line with He	2	99
Return LH_2 transfer line to Orbiter	8	107
TRANSFER LO_2 (2)	—	10
Attach LO_2 transfer line to 2nd stage	7	17
Vent transfer line	1	18
Close transfer line vent	—	18
Slow flow into transfer line	1	19
Transfer LO_2	15	34
Sense tank LO_2 mass	—	34
Switch to topping rate	10	44
Sense tank LO_2 mass	—	44
Close LO_2 valves	—	44
Hold	11	55
Attach LO_2 transfer line to 1st stage	8	63
Transfer LO_2	15	78
Sense tank LO_2 mass	—	78
Switch to topping rate	10	88
Sense tank LO_2 mass	—	88
Close LO_2 valves	—	88
Hold	13	101
Vent LO_2 transfer line	3	104
Purge LO_2 transfer line with He	5	109
Close LO_2 transfer line vent	—	109
Disconnect LO_2 transfer line	3	112
Purge LO_2 transfer line	2	114
Return LO_2 transfer line to Orbiter	8	122

(1) Timelines are based upon flow conditions of Table 3-14.

(2) Timelines are estimates.

3.4.3.1 Thermodynamic Mass Gauging. Thermodynamic mass gauging is an alternative approach that may be useful for orbital refuelling operations. This approach relies upon measuring the tank pressure increase resulting from a known helium mass introduced into a nearly full propellant tank. Ullage volume can be determined from these measurements in combination with a knowledge of propellant temperature. The tanked propellant mass can then be determined from the product of calculated liquid volume and liquid density. The following procedure, and its attendant assumptions, is considered reasonable for zero-g mass gauging operations (liquid hydrogen tanking is assumed, but the approach is also applicable to liquid oxygen):

1. Conduct tank prechill and fill as described in section 3.2 and 3.3.

   - Rely upon a flowmeter (assume ±8% accuracy) to provide gross indications of propellant loading.

   - Terminate propellant transfer at an 88% fill indication. The actual tanked quantity will reside between 80% and 96%. This will prevent the possibility of tank overfill.

2. Turn on thermodynamic vent system (TVS) mixer to achieve and maintain thermodynamic equilibrium.

   - At equilibrium, vapor and liquid reside at the same temperature. Temperature can be accurately determined from tank pressure.

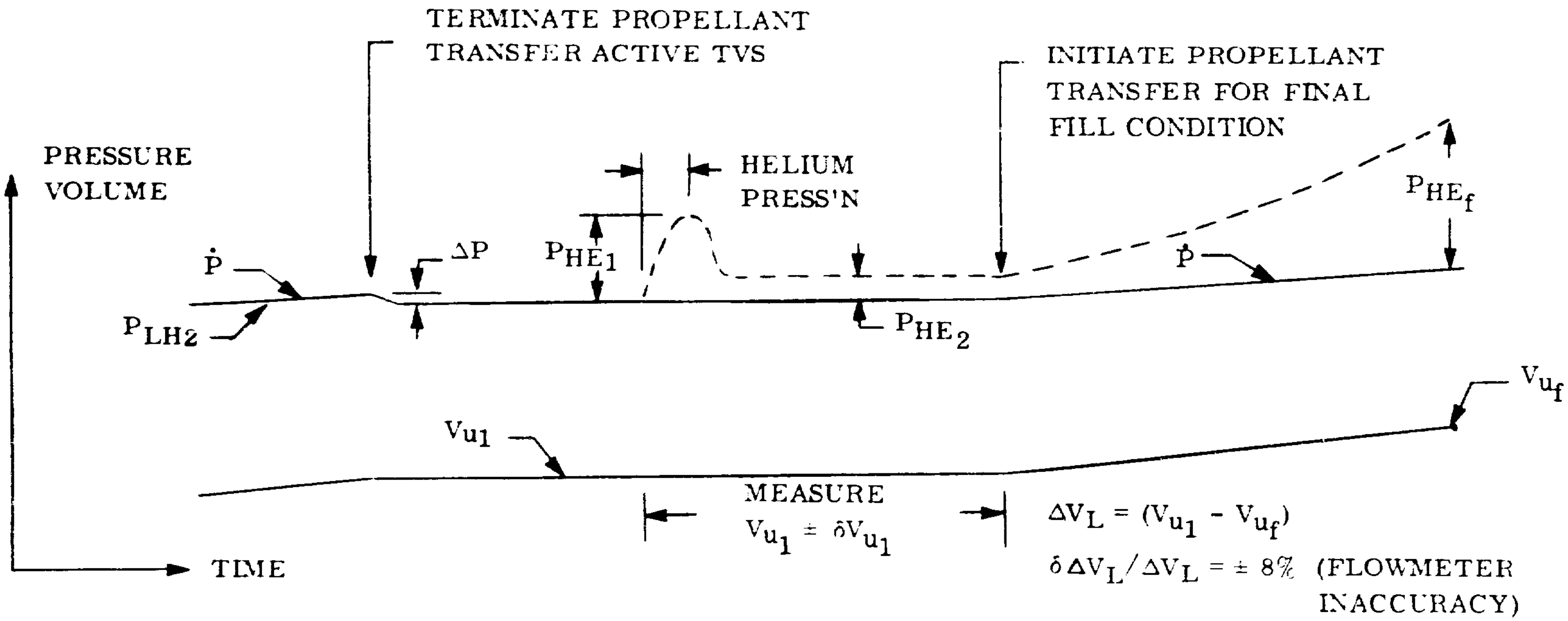
   - Assuming a 2.1 kN/m^2 (0.3 psi) error on absolute pressure (measurement), the temperature error will be about 0.06 K (0.1 R), which is equivalent to a 0.3% error for tank conditions of 138 kN/m^2 (20 psia).

3. Initiate helium pressurization after tank pressure has stabilized (i.e. pressure rise rate ≈ 0). Figure 3-58 depicts the anticipated propellant tank pressure and volume excursions during this period.

   - Measure P_{He_2} after tank pressure has stabilized

4. Ullage volume determined from Equation of State calculations

   - $$V_{u_1} = (\Delta m \times R \times T_L) / P_{He_2} \qquad (3\text{-}51)$$



TANKING ERROR = ROOT-SUM-SQUARE (RSS) OF $\delta V_{u_1}/V_{u_1}$ AND $\delta \Delta V_L/\Delta V_L$

$\dot{P}$ = PRESSURE RISE RATE DURING TANK FILL
P_{LH2} = LIQUID HYDROGEN VAPOR PRESSURE DURING FILL
ΔP = PRESSURE DECAY MAY OCCUR IF NON-EQUILIBRIUM CONDITIONS EXIST WHEN MIXER IS TURNED ON
P_{HE1} = PRESSURE RISE DURING HELIUM PRESSURIZATION
P_{HE2} = HELIUM PRESSURE AT EQUILIBRIUM
P_{HEf} = HELIUM PRESSURE AT END PROPELLANT FILL
V_{u1} = ULLAGE VOLUME MEASURED FROM THERODYNAMIC MASS GAUGING
V_{uf} = DESIRED ULLAGE VOLUME AT END PROPELLANT FILL
ΔV_L = PROPELLANT VOLUME ADDITION TO ATTAIN FINAL LIQUID FILL

Figure 3-58. Propellant Tank Conditions for Thermodynamic Mass Gauging Operations where:

V_{u_1} = ullage volume

Δm = helium mass addition

R = helium gas constant

T_L = liquid temperature

- The inaccuracy of the ullage volume calculation can be determined with the following error analysis on (3-51):

$$\frac{\delta V_{u_1}}{V_{u_1}} = \frac{\delta \Delta m}{\Delta m} + \frac{\delta T_L}{T_L} + \frac{\delta P_{HE_2}}{P_{HE_2}} \tag{3-52}$$

By taking a root-sum-square (RSS) of the above variables (which is the accepted approach), we have

$$\frac{\delta V_{u_1}}{V_{u_1}} = \pm\sqrt{\left(\frac{\delta \Delta m}{\Delta m}\right)^2 + \left(\frac{\delta T_L}{T_L}\right)^2 + \left(\frac{\delta P_{HE_2}}{P_{HE_2}}\right)^2} \tag{3-53}$$

where:

$\frac{\delta \Delta m}{\Delta m}$ = ±7% (assumed helium mass flow accuracy)

$\frac{\delta T_L}{T_L}$ = ±0.3% (from item 2 above)

δP_{HE_2} = ±0.1 psia (estimate based upon Centaur experience with high accuracy transducers)

Calculate the required liquid volume to achieve a final fill of 95%

- $\Delta V_L = (V_{u_1} - V_{u_f})$

where:

V_{u_f} = ullage volume at 95% liquid fill

ΔV_L = LH_2 volume addition to achieve tank fill

- The inaccuracy in providing ΔV_L is,

$$\frac{\delta \Delta V_L}{\Delta V_L} = \pm 8\% \text{ (flowmeter accuracy)}$$

6. Re-initiate tank fill and introduce ΔV_L propellants to achieve a final propellant load.

- Propellant loading error can be defined as

$$\frac{\delta m_L}{m_L} = \frac{\rho_L \delta V_L}{\rho_L V_L} = \frac{\delta V_L}{V_L} \qquad (3\text{-}54)$$

where:

$$\frac{\delta V_L}{V_L} = \pm \left(\frac{\delta V_{u_1}}{V_L}\right)^2 + \left(\frac{\delta \Delta V_L}{V_L}\right) \qquad (3\text{-}55)$$

Propellant Loading Error. Equations 3-51, 3-53 and 3-55 were solved for liquid fill conditions of 80% and 88% at the start of mass gauging operations. Additionally, a P_{He_2} range of 3.4 kN/m^2 (0.5 psi) to 7.6 kN/m^2 (1.1 psi) was assumed during mass gauging. Results are summarized in Figure 3-59. Note that tanking error can vary between 1.5% and 4.4% with this mass gauging method.

It is evident from Figure 3-59 that tanking error can be reduced by increasing P_{He_2}. The only limit to this increase is that the resulting P_{He_3} at tank fill completion cannot be excessive. Figures 3-60 and 3-61 display P_{He_3} as a function of final propellant fill and P_{He_2}. It is seen that P_{He_3} can become as great as 62 kN/m^2 (9 psi).

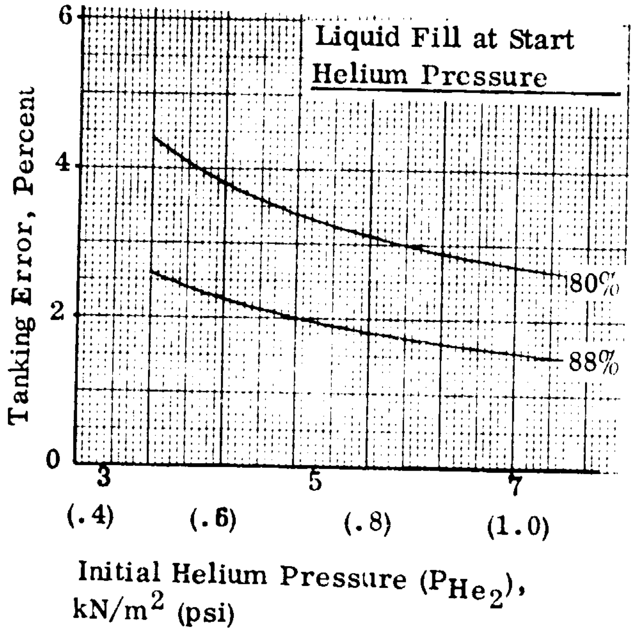


Figure 3-59. Thermodynamic Mass Gauging Tanking Error

A more acceptable pressure level would appear to be about 28 to 34 kN/m^2 (4 to 5 psi). If a propellant fill goal of 95% to 97% is considered acceptable, the Figure 3-60 indicates that the maximum acceptable P_{He_3} is about 5.2 kN/m^2 (0.75 psi). According to Figure 3-59, this will result in a minimum tanking error of 1.9% to 3.2%.

Mass Gauging Concerns. The success of the thermodynamic mass gauge rests upon two key factors; confidence in the liquid mass flowmeter, and the ability to maintain thermodynamic equilibrium with the TVS mixer during the mass gauging operations.

This technique depends upon a mass flowmeter to provide a gross indication of tank fill. Thus, there must be sufficient confidence in this instrument that propellant transfer will not be terminated until a high fill condition is indicated. If such confidence does not exist, the alternative would be to terminate propellant transfer at the 70% or 60% fill indication. An early flow termination could increase tanking error to the point of rendering this technique useless.

The second important factor is that of maintaining near-thermodynamic equilibrium conditions. Without this guarantee, ullage temperature uncertainty would increase the uncertainty in δV_{u_1}. Such as increase would result in a corresponding propellant loading uncertainty. It appears, however, from the analysis of section 3.3 that near-thermal equilibrium conditions will not be difficult to maintain.

In spite of the above concerns, it appears that thermodynamic mass gauging is a viable technique that warrants further consideration.

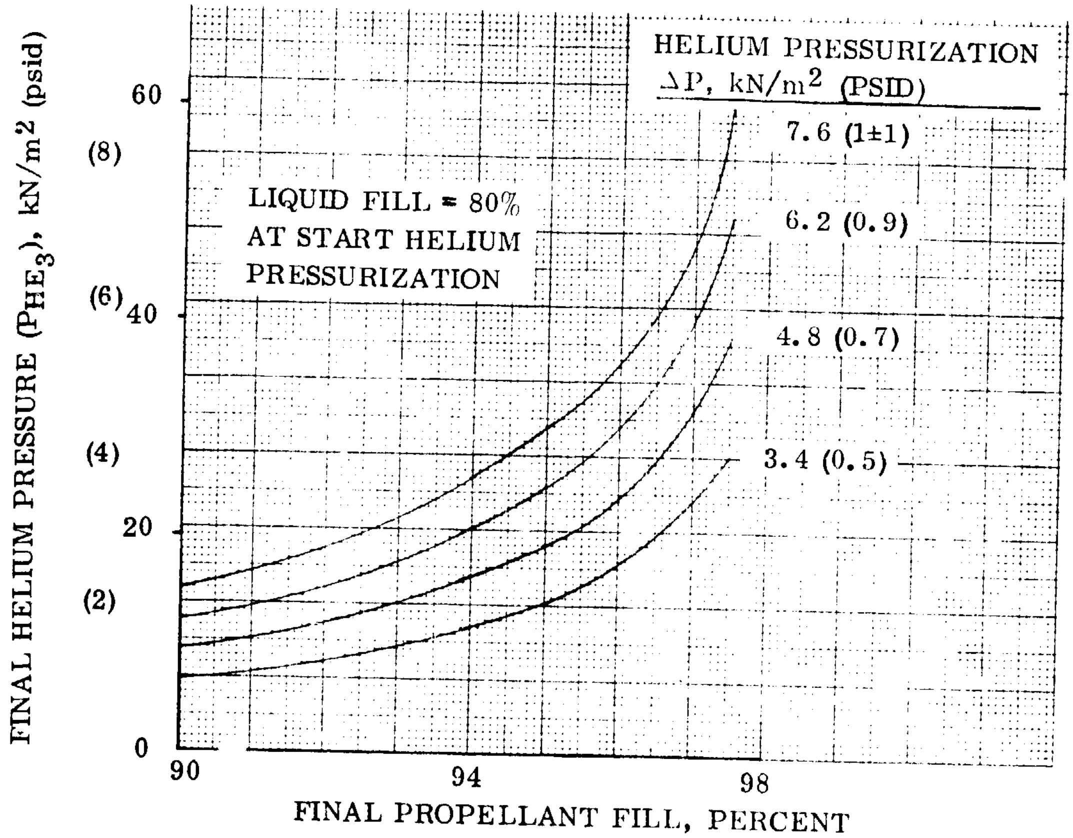


Figure 3-60. Final Helium Partial Pressures Resulting From LH_2 Tank Mass Gauging Operations at 80 Percent Fill

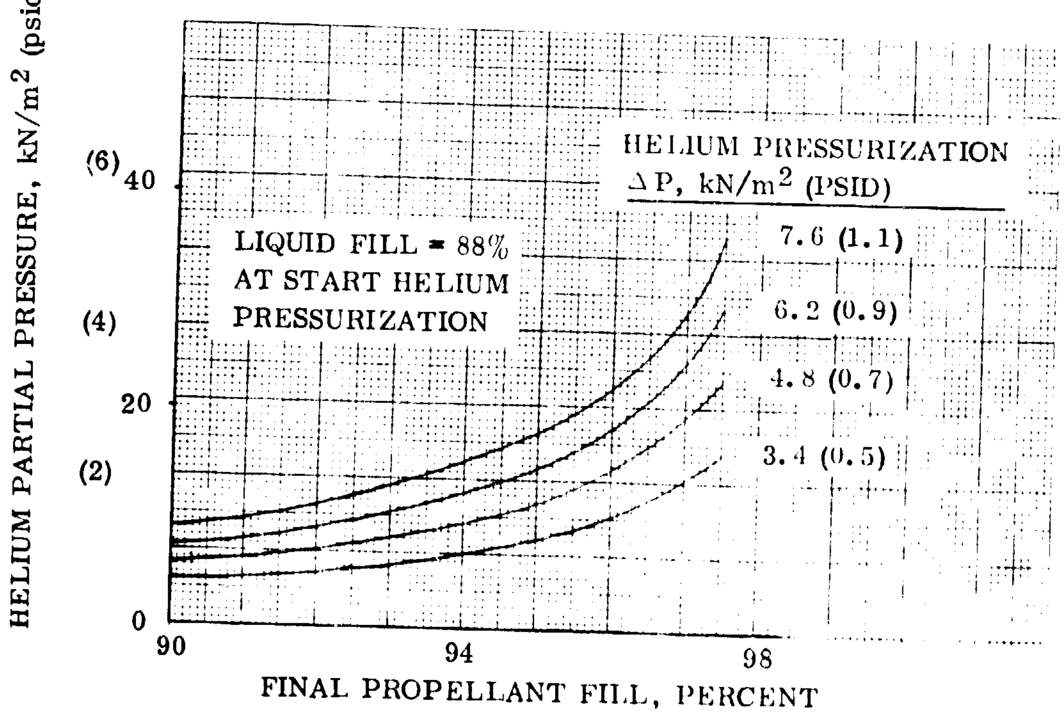


Figure 3-61. Final Helium Partial Pressures Resulting From LH_2 Tank Mass Gauging Operations at 88 Percent Fill

4

COTV ORBITAL RESUPPLY

In this section a mission scenario will be developed for the Cargo Orbital Transfer Vehicle (COTV) concept selected in Section 2. A mission will be defined as an aid in identifying the key issues of orbital refill operations. These operations will include all major activities from post-mission return to LEO through resupply. Vehicle and depot subsystem orbital refill requirements will also be identified. Operational procedures and techniques for propellant transfer will then be developed. Attention will focus only on COTV differences with POTV refill operations, because many aspects of orbital propellant transfer operations will be similar or identical.

4.1 MISSION SCENARIO

Large space industrialization programs have been conceived for the 1990's and beyond. One of the most ambitious is the Solar Powered Satellite (SPS) program that will require propellant quantities several orders of magnitude greater than those identified for the near term. In this SPS project Heavy Lift Launch Vehicles (HLLV) will be developed to transport personnel, material and propellants into low earth orbit (LEO). Liquid hydrogen and liquid oxygen will be transported in large supply tanks containing approximately 436 metric tons (960,000 lb) at a 6 to 1 ratio. These supply tanks will be transferred directly to depots in LEO, and empty tanks will be returned to earth for refuelling. Depot ancillary equipment will include reliquefiers for reclaiming stored propellant boiloff, and large solar arrays for converting solar power to the electrical energy needed to operate the orbital facility.

Personnel and material will be transported from depots to higher orbital altitudes to support industrialization programs. Personnel will be transported in POTVs; material (or cargo) will be dispatched to work sites (perhaps GEO) in a common stage COTV. Conceptual studies have been performed to identify COTV configurations, some of which were discussed in Section 2.

4.1.1 SELECTED COTV MISSIONS. A two stage COTV will fly a round trip mission to dispatch its payload to GEO. Total mission duration will be approximately five days; the first stage will return to LEO a short time after the vehicle is launched. The second stage will return to LEO following the five day round trip mission to GEO. Upon return, each vehicle stage will enter a phasing orbit preparatory to depot rendezvous. Several revolutions of this orbit ($\sim$ 3.1 hours per revolution) may be required before rendezvous is attempted. Rendezvous and docking maneuvers will consume approximately two hours.

Post-docking operations will include cost effective procedures to reclaim all propellant liquid and vapor practicable. This is the primary area where substantial differences between POTV and COTV operations will occur because the depot ancillary equipment can be employed to minimize propellant loss. The impetus for such procedures is derived from the relatively high cost of transporting propellants into space.

4.1.1.1 Timelines. Timelines have not yet been developed for COTV operations. However, a previous study "Orbital Propellant Handling and Storage Systems for Large Space Programs," Reference 4-1, indicated that a launch schedule of ten or more flights per year is conceivable. Therefore, a rapid turnaround time between missions is expected. Subsystems will be inspected and tested to verify flight worthiness once post-mission operations are complete. A safed condition will exist once liquid propellants have been transferred and tank pressures stabilized. Pressure stabilization will be relatively easy to manage once liquids have been transferred.

4.1.2 ORBITAL DEPOT CONFIGURATION. Orbital depot conceptual designs were developed under Contract NAS9-15640 (Reference 4-2) for Johnson Space Center (JSC). That study concluded that large scale space activities would benefit from orbital propellant depots such as shown in Figure 4-1, which illustrates a five-storage-module depot, with optional crew quarters and maintenance hangars, refueling a manned OTV. Basic features include capillary propellant acquisition systems so that no rotation or thrusting is necessary to position propellants. Reliquefaction systems eliminate long term boiloff losses.

Benefits of such a depot are:

a. Reliquefaction eliminates boiloff losses.

b. Operations are more flexible, accommodating launch delays and OTV waiting.

c. Operating costs are reduced for probable 1990's OTV missions.

Investment considerations are:

a. Reliquefaction with power and radiators is a major item.

b. Station keeping uses significant amounts of propellants.

c. Propellant delivery tanks may also be used for storage.

d. Docking provisions, solar power array, and radiators are common to any space base.

4.1.2.1 Ancillary equipment. Reliquefiers will be permanently plumbed to the supply tank farm for continuous reliquation of propellant boiloff. Studies conducted in Reference 4-2 indicate that reliquefiers will be cost-effective in contrast to the alternative of transporting additional propellants from earth. Since they require electrical

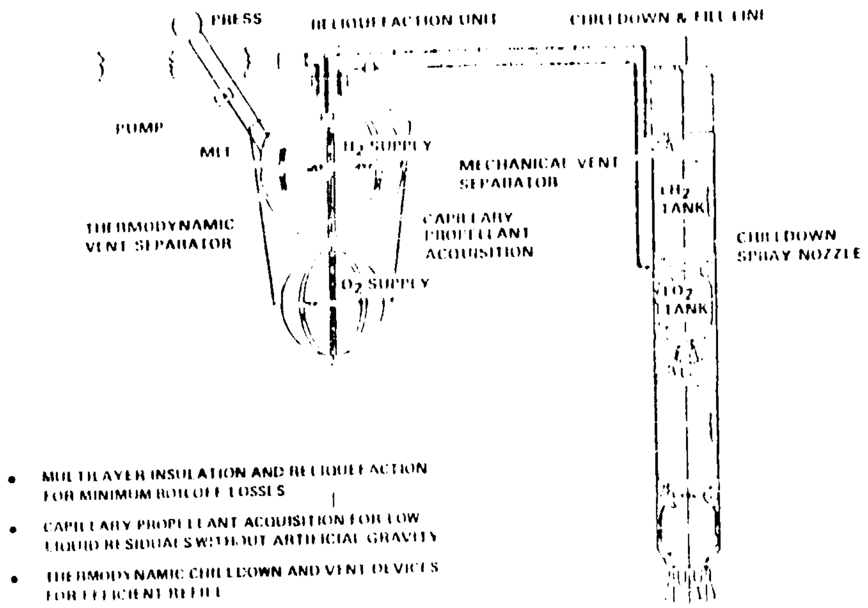


Figure 4-1. Orbital Propellant Depot

power for operation and radiators for heat rejection, electrical power will be provided using solar arrays that will directly convert solar energy. Figure 4-2 provides size and cost data on reliquefiers, solar arrays and radiators obtained from the previously mentioned study. It is apparent that the depot physical configuration will be dominated by the solar array and radiators. These data are for reliquefaction rates of 3.5 kg/hr (7.7 lb/hr) LH_2 and 7 kg/hr (15.6 lb/hr) LO_2.

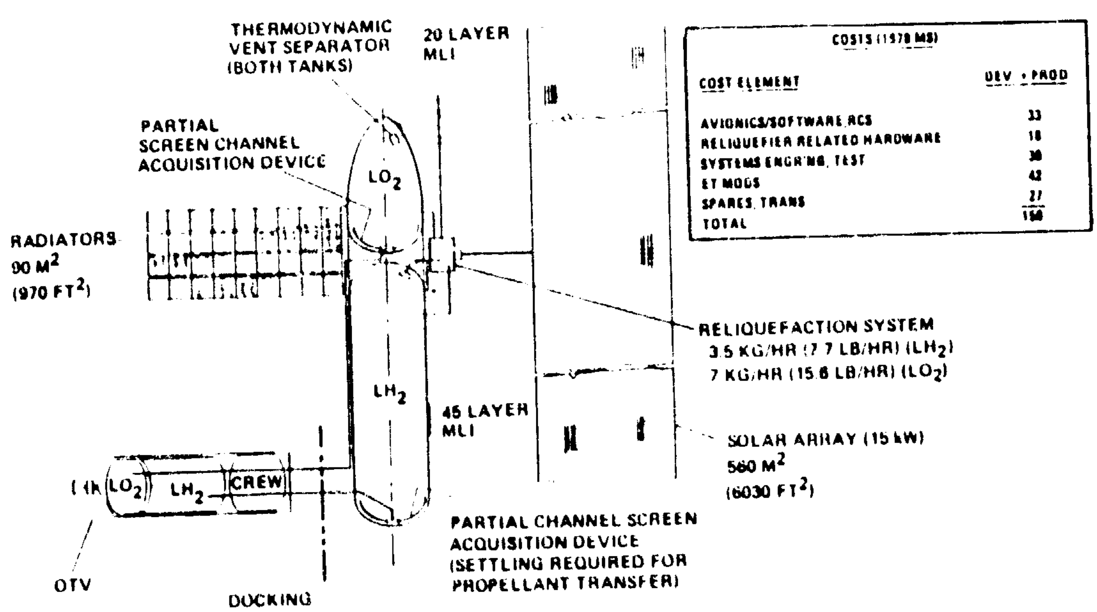
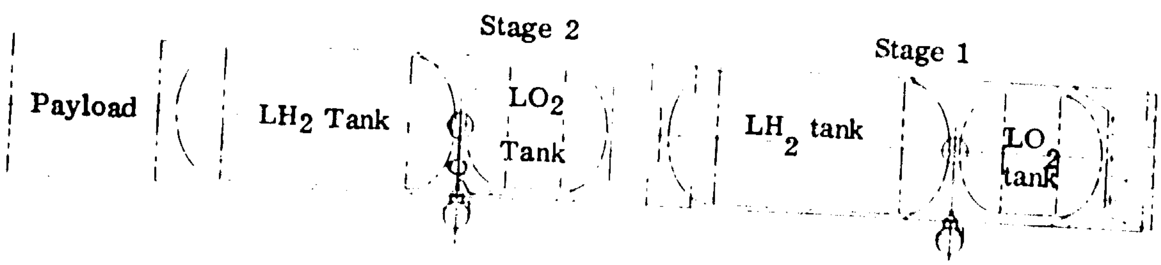


Figure 4-2. Schematic: ET Propellant Depot

Two phase pumps. The ability to readily transfer propellants from supply to receiver tank is a necessary requirement for orbital depot operations. It has generally been assumed that pressure-fed systems will be employed to effect on-orbit transfer of propellants. There are instances, however, when this approach appears impractical. An excellent example of this occurs when transferring residual propellants from COTV to the depot supply tanks. The mass required to pressurize the vehicle tanks will be of the same magnitude as the residual liquid to be transferred. A preferred, and cost effective approach will be to use pumps for transferring residual propellants, rather than a pressure-fed system. Such pumps require two phase pumping capability because bulk boiling occurs within the propellant tanks and saturated propellants are expelled. The absence of buoyancy during on-orbit propellant transfer may allow considerable vapor flow, and pumps will have to be designed for this flow condition.

4.1.3 COTV CONFIGURATION. The basic COTV tank parameters are given in Figure 4-3. In this basic structure are included those subsystems needed to achieve COTV mission and orbital refill requirements. These subsystems are pressurization, insulation, thermodynamic vent system, non-propulsive vent system, propellant



Parameter	Each Stage	
Physical		
Length, m (ft)	48	(157.5)
Diameter, m (ft)	8.4	(27.6)
Weight, kg (lb)	251,750	(555,000) wet
	15,880	(35,000)
Tankage		
LH_2 Capacity, kg (lb)	37,050	(81,690)
LO_2 Capacity, kg (lb)	198,820	(438,300)
Material	2219 - T87 Al	
Insulation	MLI	

Figure 4-3. COTV Characteristics

management (start basket), an advanced attitude control system (AACS), and an advanced main engine system. Only the advanced attitude control and main engine systems will be substantially different from the POTV systems described in Section 3. Neither system will have a major influence upon orbital refill operations.

4.1.3.1 Advanced attitude control system. The AACS will be designed to extract liquid hydrogen and liquid oxygen propellants from the main tanks. This system will be employed for all precision maneuvers required during the COTV mission and the post-mission docking operations. However, the AACS will not be a factor once docking is complete.

4.1.3.2 Advanced main engine. The advanced main engine system will be able to operate at zero net positive suction pressure (NPSP) and have "boot-strap" capability. That is, the vehicle will be able to achieve main engine start without benefit of pre-pressurization or propellant settling. When the engine valves are opened, propellants will flow from the acquisition devices to their respective main engine inlets and generate a low thrust. This low thrust will result in a higher flow rate demand, which in turn will generate a higher thrust level until the engine has "boot-strapped" to steady state operating levels.

Because this advanced engine will preclude pre-pressurization for main engine start (MES), helium will not be present in either liquid hydrogen or liquid oxygen propellant tank. This condition can greatly simplify refill operations, as was discussed for POTV. However, in order to provide a more thorough discussion of orbital depot refill operations, a helium pressurization system will also be assumed for COTV. The following variations to the basic vehicle configuration will be analyzed in this section.

1. No helium pressurization and no start basket.
2. No helium pressurization with start basket.
3. Helium pressurization and no start basket.
4. Helium pressurization with start basket.

4.1.3.3 COTV/POTV procedural differences. Refill operations for all four configurations will be identical to their respective POTV configurations analyzed in Section 3. No further analysis is required for COTV, although curves will be provided depicting the influence of prechill and tank fill flow parameters upon each process. Only the post-mission operations will differ from those previously discussed in Section 3. These operations will focus upon reclaiming residual propellant vapor and liquid using the depot ancillary equipment; there will be no provisions for reclaiming propellant residuals on POTV missions.

4.2 POST MISSION DE-TANKING OPERATIONS

Once vehicle docking is complete and the payload/crew module is secure, post mission de-tanking operations will be initiated. One or more of the following reasons may be justification for initiating propellant de-tanking in a timely manner.

If it is feasible to reclaim propellants —

- Return liquid to the better insulated supply tanks in order to minimize boiloff.
- Reliquefy propellant vapor and return to supply tank, thereby reclaiming vapor for future missions.

If a helium pressurization system is used (i.e., helium is in propellant tanks) —

- Helium must be expelled before refill can be initiated. This requires that tank propellants be expelled.

If an autogenous pressurization system is used (i.e., no helium) —

- Propellants will be expelled if vehicle component replacement requires that propellant tanks be "opened-up". As an example, vent valve replacement can be performed conveniently only if tank pressure resides at or near zero.

It is seen from the above that propellants should almost always be transferred from the COTV shortly after docking. An exception to this would be if a) there was no advantage to reclaiming residual propellants, b) the vehicle stage had autogenous pressurization, and c) component replacement would not be performed between vehicle missions. Under these conditions, propellants would remain aboard, gradually evaporate and be vented. There would be no compelling reason for transferring liquid propellant because all maintenance, except for major component replacement, could be performed with liquid on-board.

The remaining discussion and analysis will deal with the question of when it is cost effective to reclaim residual propellants, and when these propellants should be dumped overboard. Because helium can complicate procedures, operations for an autogenous pressurization system will be evaluated separately from helium system operations.

4.2.1 OPERATIONS FOR AUTOGENOUS PRESSURANT. A totally autogenous pressurization system for COTV will only be realized for an advanced engine system with zero NPSP capability. Propellant vapor will be tapped off from the engine system for tank pressurization during "transient" and steady state operation. Consequently, only liquid and its vapor will reside within the propellant tanks following a mission. This condition reduces the complexity of propellant transfer operations.

4.2.1.1 The cost of propellant dumping - The cost of dumping residual propellants overboard rather than reclaiming them is determined by the cost of transporting the equivalent propellant mass from earth. Residual propellant quantities for each COTV stage are itemized in Table 4-1, which totals for two stages are 5,896 kg (13,000 lb). Propellant transportation costs during the 1990's are expected to be in the range of 22\$/kg (10\$/lb) to 44\$kg (20\$/lb). The total estimated costs of dumping propellant is given in Figure 4-4, and can be as great as 30 million dollars for a ten year period. The cost of producing propellants was not included because these costs will represent less than one percent of propellant transportation costs.

Table 4-1. COTV propellant tanks final MECO residuals.

	Tank Volume, $m^3(ft^3)$	Final MECO Pressure, kN/m^2(psia)	Vapor[(1)] Density $kg/m^3(lb/ft^3)$	Vapor Residual kg(lb)	Liquid[(2)] Residual, kg(lb)	Total Residual kg(lb)
H_2 Tank	548 (19360)	103.4 (15)	6.62 (.085)	747 (1648)	195 (430)	942 (2078)
LO_2 Tank	183 (6460)	103.4 (15)	22.18 (.285)	835 (1840)	1171 (2581)	2006 (4421)

(1) saturated vapor

(2) assumes 0.5% of total propellant load.

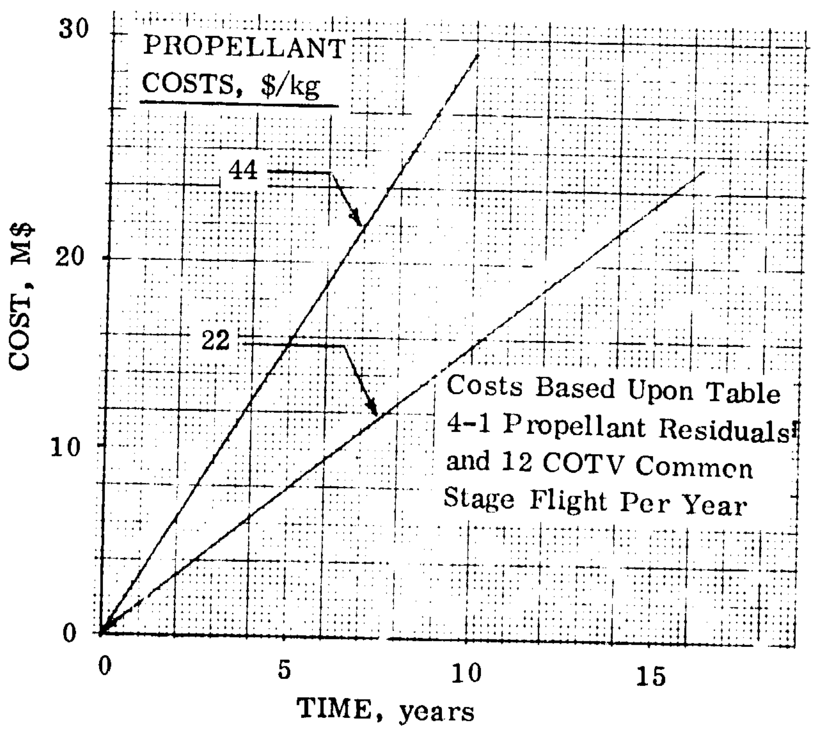


Figure 4-4. Cost of Replacing Dumped COTV Residuals

4.2.1.2 Propellant reclamation — Ideally, it will be desirable to first transfer all liquid from COTV, and then deal with the question of reliquefying the propellant vapor. However, pure liquid transfer from COTV to the depot supply tanks will be extremely difficult to achieve because liquid-vapor distribution, created by the near zero-g environment, may be unknown. Consequently, it is necessary to consider two-phase fluid transfer rather than the separate operations of liquid phase transfer followed by gas phase transfer. There are three alternatives available for transferring residual propellants to depot supply tanks. These are

Case 1 - transfer a two phase fluid directly to the supply tank.

Case 2 - transfer the fluid to a reliquefier; the resultant pure liquid can then be transferred to a supply tank.

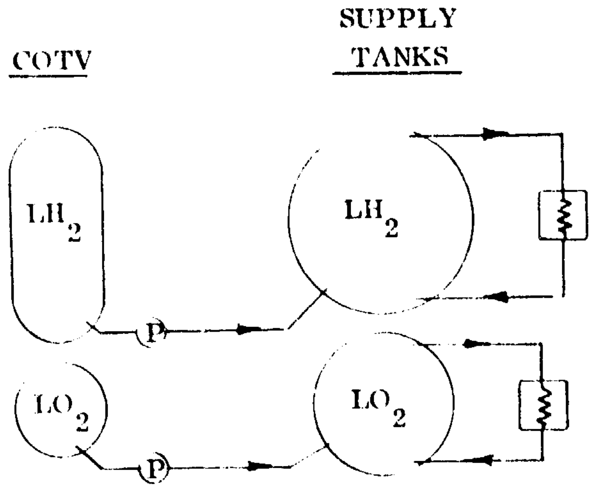
Case 3 - Connect propellant tanks vent plumbing to the supply tank and reliquefy boiloff on a continuous basis.

A schematic of each alternative, with a brief discussion of advantages and disadvantages, is given in Figure 4-5. The most costly approach will be Case 2 because substantially larger reliquefiers are required if the residual vapor is to be recondensed during the transfer process. This higher flow rate requirement translates into larger reliquefiers, radiators and solar array. Case 3 is unacceptable because propellant tank pressures will remain high for the entire period between launches, thereby violating the groundrule to reduce tank pressures to a low level in a timely manner. Case 1 will have the shortest duration for propellant transfer since the only limitation on transfer flow rate is pump size. This case will be analyzed in greater detail.

Case 1 Evaluation. A compressor will be required for propellant transfer to the supply tank because receiver tank pressure will decay to a low level during this process. Since compressors may not operate satisfactorily under mixed phase conditions, a vent tube will be needed to probe the ullage such that a high percentage of vapor by volume is expelled at all times. Once fluid transfer is complete, the vapor is processed through the supply tank reliquefiers over a long time period. The time required to complete reliquefaction will depend upon fluid residual mass and reliquefaction capacity.

It must be recognized that reliquefaction is not cost - free. The supply tank reliquefaction systems will be sized to handle a design boiloff rate. Reliquefactor of propellant residuals represents a load over and above the design conditions. Consequently, separate units or added capability for existing units is required. In either case, the additional cost must be borne as the cost to reclaim residual propellants. These costs are explored in the following paragraphs.

Reclamation Costs. The rate at which propellant quantities are reliquefied will depend, in part, upon the available residual mass and the time available for reliquefaction.



Ⓟ pump/compressor

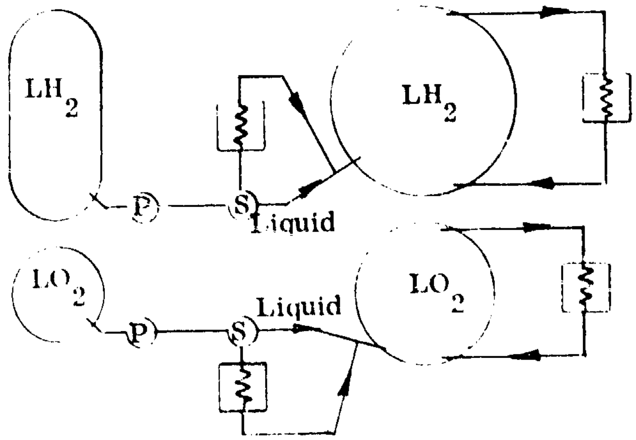
Reliquefiers

Advantages

1. Vapor can be condensed by supply tank reliquefiers over long time period.
2. Fluid transfer occurs in timely manner.

Disadvantage – pumps/compressors required for two-phase fluid transfer

CASE 1



Ⓟ pump/compressor

Reliquefier

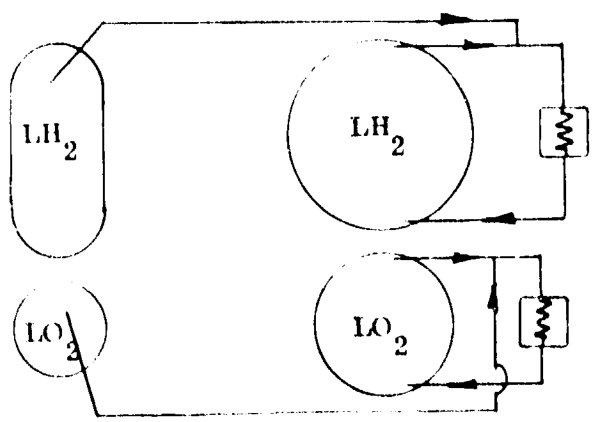
Ⓢ liquid-vapor separator

Advantage – pure liquid returned to supply tanks.

Disadvantages

1. Large reliquefiers required to condense vapor during short transfer period.
2. Liquid-vapor separator and pumps required for two-phase fluid transfer.

CASE 2



Advantage – No new equipment required – Vapor flows directly to supply tanks reliquefiers as receiver tanks pressures increase above vent levels.

Disadvantage – Long time required for boiloff of receiver tank propellants.

CASE 3

Figure 4-5. Alternative methods of reclaiming COTV residual propellants during vehicle post mission operations.

Estimated residual propellant quantities have been given in Table 4-1. These propellants should probably be reliquefied in a time comparable to the interval between COTV launches; which is about 30-36 days for the selected scenario of 10-12 launches per year. Figure 4-6 gives reliquefaction rates as a function of time for the COTV hydrogen residuals. (Note: This discussion will focus on hydrogen because its reliquefaction costs are considerably greater than for oxygen.)

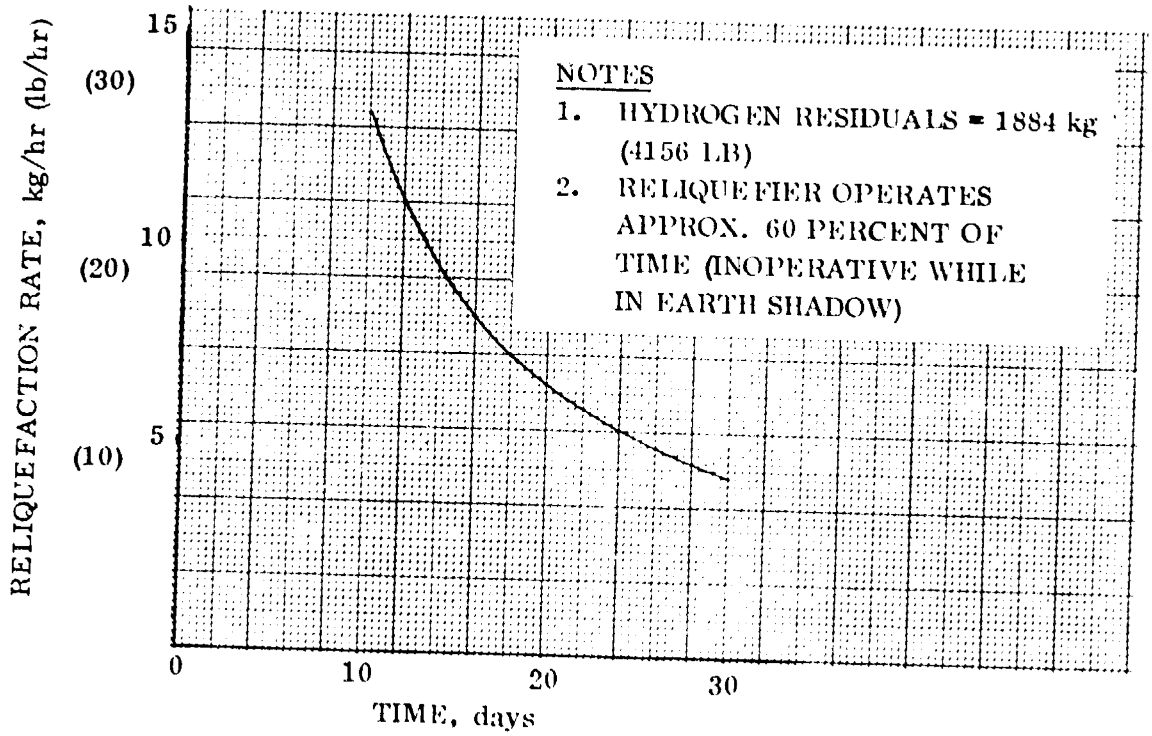


Figure 4-6. Reliquefier Capacity–Time Requirements for Reliquefying COTV Hydrogen Residuals

The cost of hydrogen reliquefaction units and their associated capacities are given in Table 4-2 for two unit sizes. These costs were determined from Reference 4-1 and do not include development costs; only the cost of an additional unit is assessed. The smallest unit has a hydrogen reliquefaction capacity of 9.5 kg/hr (21 lb/hr) which is capable of reliquefying COTV hydrogen residuals in about 14 days. This unit is an adequate size for propellant reclamation, and its 16.6 M$ cost represents the cost of reclaiming propellants. Figure 4-7 indicates that about five to ten years of operation may be required to recover reliquefier costs.

4.2.1.3 Residuals for RCS propellants — The orbital depot will consume substantial quantities of hydrogen and oxygen for drag makeup and attitude control requirements. The annual usage rates were estimated in Reference 4-1 at 22608 kg (49,872 lb)

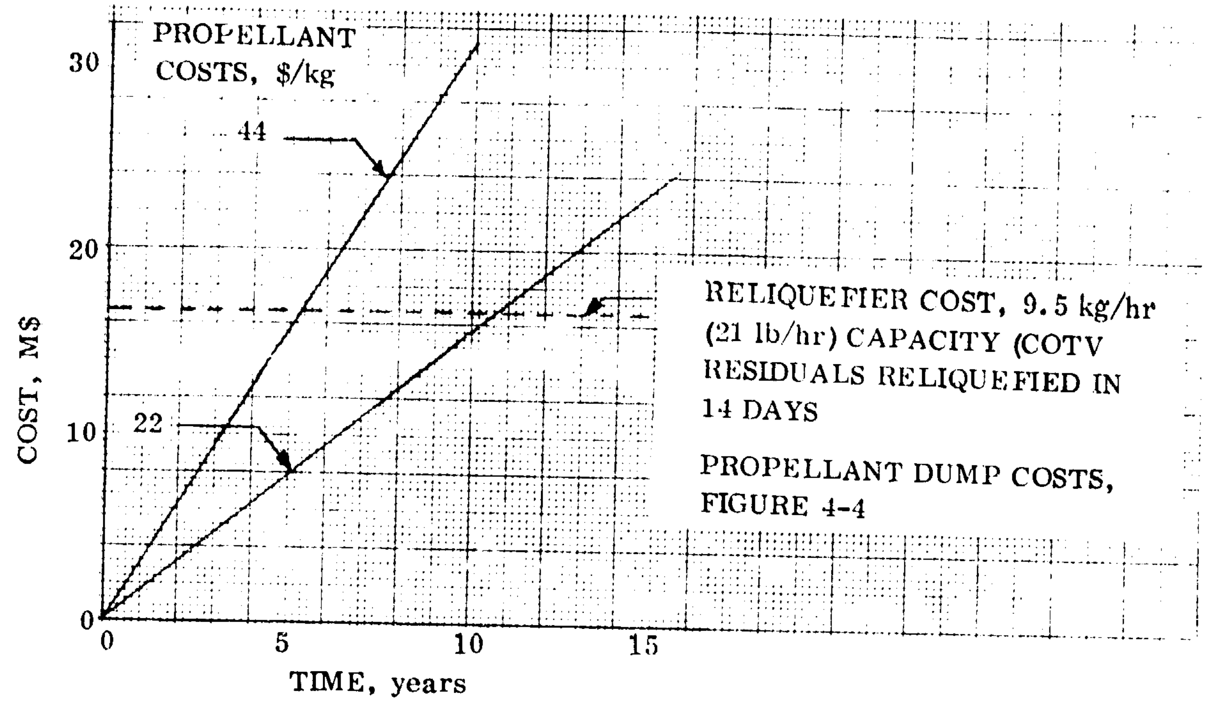


Figure 4-7. Cost of dumping COTV hydrogen residuals compared to cost of reclaiming vapor residual.

hydrogen and 48144 kg (106,104 lb) oxygen. Propellant residuals (from 12 COTV launches/year) could provide up to 40 percent of the annual RCS propellant needs if the residuals can be transferred to suitable storage tanks. That is, the fluids must be transferred and stored in vapor form in order to avoid the high cost of reliquefaction. Performance of an oxygen/hydrogen vapor feed system, is shown in Figure 4-8. A vapor feed system appears to be feasible because performance degradation is somewhat insensitive to chamber pressure (which reflects storage pressure conditions). It is also estimated that storage temperature variations will have minimal effect on RCS engine performance.

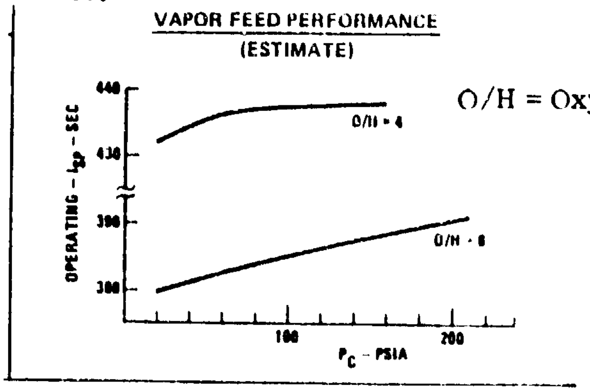


Figure 4-8. Estimated vapor feed performance for Oxygen/Hydrogen RCS engines.

Table 4-2. Propellant Depot Facility Cost Estimate (Table 9-2, Reference 4-1) (Millions of 1979$)

Cost Element	5 M lb Capacity			System Dedicated to Reliquefaction of COTV Propellant Residuals	
	Size	Dev.	Proc.	5 M lb Capacity	10 M lb Capacity
Structure	15 K lb	13.23	0.06	-	-
Avionics/Software	500 lb	24.48	2.61	-	-
Solar Array	33.3 m^2	-	0.01	0.01	0.02
Electrical Power System	1000 lb	6.26	0.95	0.95	1.90
Fluid System/Plumbing	1500 lb	5.23	3.30	3.30	6.60
Reliquifiers*	2200 lb	16.32	6.69	6.69	13.38
Radiators	300 lb	0.54	0.20	0.20	0.40
RCS System	400 lb	6.56	2.62	-	-
Subtotal		72.71	16.44	11.15	22.30
Floating Items		27.64	5.59	3.79	7.58
Initial Spares		—	2.47	1.67	3.34
Initial Transportation			0.21	-	-
		100.35	24.71	16.61	33.22

* 9.5 kg/hr (21 lb/hr) GH_2 reliquefaction and 29.5 kg/hr (65 lb/hr) GO_2 reliquefaction.

(1) Hydrogen reliquefaction requirements represent major portion of costs.

(2) No development costs for solar array. It is assumed that development costs have been borne by the Solar Powered Satellite (SPS).

Two disadvantages have been identified for an oxygen/hydrogen vapor feed system. These are: a) the need for compressors to transfer residual propellants to the RCS storage tanks, and b) the increased storage tank mass and volume required for vapor storage over that of liquid storage.

These disadvantages appear to be minimal as explained below:

1. Compressors and/or considerably more complex equipment will be required for the reliquefier approach.
2. The increased RCS storage tank weight should be small in comparison to the residual mass reclaimed annually.

4.2.2 OPERATIONS FOR HELIUM PRESSURANT. Reclaiming COTV residual propellants can be a more complicated process, than described in Section 4.2.1, if helium is present in the propellant tanks. The helium will have been introduced during the vehicle mission to provide NPSP for each main engine start. As discussed in Section 3, helium must be removed prior to a refill operation because excessive partial pressure may result for a fully loaded tank condition. Steps must also be taken to prevent helium from entering the start baskets during refill because, once present, that helium will remain until expelled during a vehicle mission. Pure liquid flow from the start baskets is a requirement; consequently, helium trapped within a start basket is unacceptable.

These potential helium problems will be avoided with POTV by expelling virtually all propellant residuals and helium from the propellant tanks through a blowdown process. Several blowdown periods are contemplated to provide assurance that helium is diluted to an acceptably low level. This approach is acceptable for POTV because equipment and facilities will not be available in the orbiter to reclaim propellant residuals. But, since it can be assumed that the appropriate facilities and equipment will be an integral part of the orbital depot, and evaluation is required to determine what options are available, and which option is acceptable. Table 4-3 identifies the problems (caused by helium) associated with the COTV post-mission operations; possible solutions are summarized for each problem. The following three solutions (which are identical to those of Section 4.2.1) will be discussed:

1. Propellant dump to space (with no attempt to reclaim residuals).
2. Reliquefy propellant vapor and return to the depot supply tanks.
3. Transfer residuals to depot RCS propellant tanks for subsequent reuse for drag makeup and attitude control requirements.

4.2.2.1 The cost of propellant dumping — The cost of dumping propellants from a COTV will be the same whether the vehicle includes a helium pressurization system or an autogenous pressurization. This is because propellant tank total residuals will

Table 4-3. On-Orbit Refueling of a COTV (Helium Pressurization System)

	VEHICLE HARDWARE REQUIREMENTS	ADVANTAGES	DISADVANTAGES
Requirement No. 1: Refill partially full LH_2 and LO_2 propellant tanks containing helium pressurant in the ullage.			
Problem No. 1: Refill attempted without venting will result in an excessive helium partial pressure.			
Solutions:			
1-A Expel helium by venting the tank to a low pressure prior to initiating refill.	None. But must have capability of maintaining vent valves open down to low pressures.	Simple approach.	Will lose approximately 1383 kg GH_2 and 1606 kg GO_2 plus all propellant residual during process.
1-B Dilute helium concentration by introducing a closed cycle purge procedure	Plumbing mods for closed cycle purge.	No propellant vapor or liquid will be lost.	Complex and costly space depot equipment required. Process takes weeks to complete.
1-C Pump tank contents to depot RCS propellant tanks. Use for depot drag makeup and attitude control requirements.	None	Propellants will provide about 40% of depot RCS requirements. Minimum cost.	Helium may degrade H_2/O_2 engine ISP.
Problem No. 2: COTV tank venting on-orbit will include propellant vapor and possibly liquid propellant. How can liquid be recovered before a venting or purge procedure is initiated?			
Solutions:			
2-A Orient the vehicle such that aerodynamic drag will collect propellants around the start basket. Slowly drain liquid from the tanks until all available liquid is reclaimed.	Plumbing mods to drain liquid from propellant tanks.	Simple modification	It is unlikely that aerodynamic drag at 250 n.mi. is sufficient to collect propellants.
2-B Establish a procedure whereby vehicle propellants are transferred to a small, well-insulated receiver tank while under RCS thrust.	Plumbing mods to support receiver tank and to drain liquid from propellant tanks.	Propellant drain can be conducted in timely manner.	Additional RCS propellants required. Procedures and operations may be complex and time consuming.
2-C Introduce a screen channel capillary device that can scavenge liquid residuals off the tank walls during an extended time period.	Design a screen channel device for each propellant tank.	Simple propellant drain procedure.	Major modification to propellant tank designs. Propellant drain may be time consuming.
* 2-D No problem exists if Solution 1-C for Requirement No. 1 is selected.			

* Recommended solution.

Table 4-3. On-Orbit Refueling of a COTV (Helium Pressurization System) (Contd.)

	VEHICLE HARDWARE REQUIREMENTS	ADVANTAGES	DISADVANTAGES
Problem No. 3: How can liquid and vapor be recovered during a purge or vent process?			
Solutions			
3-A The closed cycle purge process will reclaim liquid, propellant vapor and helium while maintaining a constant propellant tank pressure.	Plumbing mods for closed cycle	No propellant vapor or liquid will be lost.	Complex and costly space depot equipment required. Process takes weeks to complete.
3-B The open cycle vent process described by Figure will reclaim liquid, propellant vapor and helium during the tank pressure decay.	Plumbing mods for open cycle purge.	No propellant vapor or liquid will be lost.	Same as above but less re-liquefaction required.
* 3-C No problem exists if solution 1-C for Requirement No. 1 is selected.			
Requirement No. 2: Guarantee that helium will not penetrate and remain trapped within the start baskets.			
Problem No. 1: How can helium entry to the start baskets be prevented?			
Solutions:			
* 1-A If the propellant tanks are vented down for purposes of dumping fluid overboard or recovering fluids, it will be possible to ultimately evacuate the tank and thereby eliminate all helium.	—	The approach guarantees that helium will not penetrate start baskets.	Substantial propellant lost for the overboard dump process. No disadvantage for the recovery process.
1-B Start baskets will remain filled with liquid during the closed cycle purge process as long as communications is maintained between tank propellants and start basket propellants.			Cannot guarantee that start basket will remain filled with liquid during purge process.

* Recommended solution.

be the same for either vehicle system. Vapor residuals will be about the same because they are only minimally affected by pressurization system. Liquid residuals will be the same because they will be controlled by the same type of propellant utilization system. The cost data of Figure 4-4 is applicable.

4.2.2.2 Propellant reclamation — The possibility of excessive propellant losses was the impetus for considering reliquefiers to reclaim all residual propellants following a COTV mission. A schematic of this method is shown in Figure 4-9 for hydrogen; procedures and assumptions are given below.

1. A hydrogen-helium vapor mixture will be extracted from the propellant tank. Little or no liquid is vented because it remains in contact with the tank surfaces and start basket.
2. Tank pressure will decay to approximately 6.9 kN/m^2 (1.0 psia) during the vapor expulsion process. Much of the liquid will boil during this period.
3. A compressor in the vent line will increase vapor pressure as it flows to the reliquefier. This compressor must be capable of increasing pressures to greater than 103 kN/m^2 (15 psia), the storage tank pressure.

   Note: Compressor costs are not included in this evaluation.
4. The hydrogen-helium mixture enters the reliquefier where propellant vapor is recondensed. Reliquefier electrical power is provided by an existing solar array, and heat rejection is provided by space radiators.

   Note: Only the cost of additional reliquefiers and radiators will be considered in this evaluation. These costs are given in Table 4-2.
5. Liquid hydrogen is separated from helium using a mechanical separator, with the helium being returned to its storage tank, and LH_2 returned to its storage tank.

The cost of reliquefying residual vapor containing helium will be about the same as the costs for no helium. Consequently, the same cost curve (Figure 4-7) is applicable. The same conclusion is also drawn, which is, the cost of reliquefaction is sufficiently high that propellant residual reclamation is not an obvious choice.

4.2.2.3 Residual for RCS propellants — Employing propellant residuals for RCS propellants is the same task whether or not helium is involved. The discussion of Section 4.2.1.3 is, therefore, applicable. The single exception to Section 4.2.1.3 is that RCS engine performance will be degraded by the presence of the inert gas. However, degradation will be small because helium represents a fraction of the total propellant residuals.

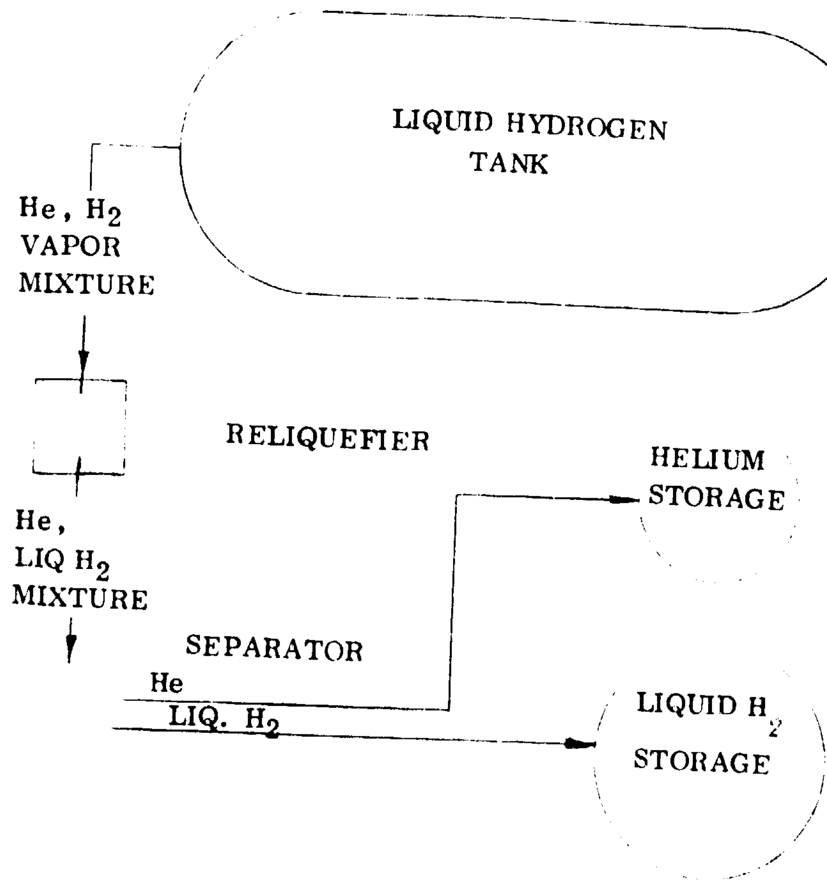


Figure 4-9. Schematic of a Propellant Residual Reclamation Process

4.3 COTV ON-ORBIT RESUPPLY

The technical obstacles associated with COTV refill operations from an orbital depot are similar to those identified for POTV refill from an orbiter-tanker. That is, the hostile space environment (zero-g and vacuum) combined with the limited resources available for space-based operations will complicate refill operations. A major difference between refill from an orbital depot or from an orbiter tanker is that resources available to the former should be substantially greater in terms of personnel and facilities. The additional resources should make it easier to perform COTV refill.

The thermodynamic and fluid mechanic considerations of COTV on-orbit refill are identical to those for the POTV. Consequently the same elements of an acceptable propellant transfer procedure apply: initial vent, prechill and fill. The POTV initial vent is equivalent to the COTV de-tanking operations of Section 4.2 Also, prechill and tank fill criteria for COTV is identical to that for POTV because the physical mechanism will be the same. Consequently, this discussion will focus on the propellant transfer flow conditions required to satisfactorily prechill and fill the propellant tanks.

4.3.1 COTV PRECHILL. The requirement for prechill of the propellant tank is to reduce tank temperatures sufficiently that the fill process will be accomplished without having to vent. The theoretical maximum tank pressures during prechill, as described by equations 3-7 and 3-8 are plotted in Figure 4-10 for the COTV and POTV. Note that peak pressures appear to be virtually independent of tank size. This is because the volume-to-mass ratios of each vehicle propellant tank is nearly the same. It will be shown in Section 6, on scaling, that this ratio is an important prechill scaling parameter. As was determined in Section 3.3.2, the LH_2 tank will be prechilled to a temperature not exceeding 200K (360R). This step will guarantee that the absolute maximum tank pressure will not exceed 138 kN/m^2 (20 psia). The liquid oxygen tank will not require a prechill process (as was previously determined for POTV).

4.3.1.1 Prechill procedures — The following charge and vent procedure was selected for LH_2 tank prechill:

1. Meter LH_2 into the tank at a high velocity to provide good heat exchange with the walls.
2. Allow time for tank pressure to peak out at about 69 kN/m^2 (10 psia). Vapor temperature equals tank temperature at this time.
3. Vent the tank to near zero pressure and repeat steps 1 and 2 as required to reduce tank temperature below 200 K (360 R).

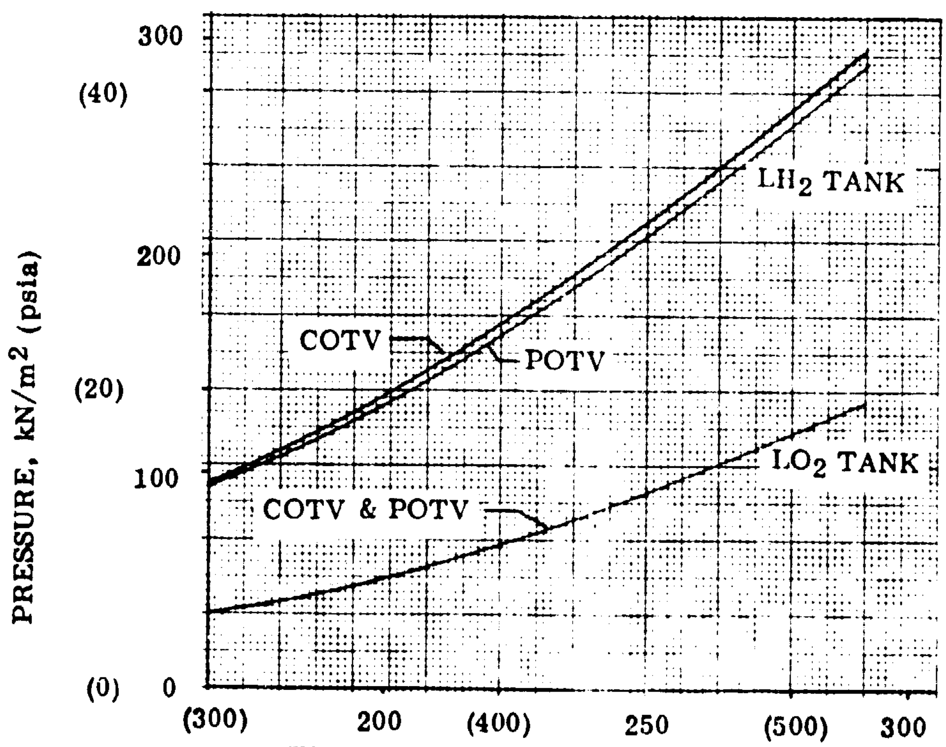


Figure 4-10. COTV and POTV Peak Prechill Pressures Will be the Same

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Figures 4-11 and 4-12 give tank ullage pressure and wall temperature histories during this prechill period. These curves are based upon introducing 42.6 kg (94 lb) hydrogen at 0.91 kg/sec (2 lb/sec) during the charge period. Peak pressures for this procedure will not exceed 79.3 kN/m^2 (11.5 psia). Flow conditions selected for this simulation are given in Table 4-4 and appear to be readily attainable conditions. It is concluded that LH_2 tank prechill will be a rather straightforward process, and of a sufficiently short duration that it can readily be integrated into a vehicle mission prelaunch sequence.

A comparison of the aforementioned ullage pressure and wall temperature histories with those for POTV (Figures 3-19 and 3-20) reveals an obvious similarity between both sets of conditions. The similarity is even more striking if the data is plotted with respect to a normalized time (Figure 4-13). Time is normalized by dividing the actual time by the total time of each charge duration. The excellent data correlation supports the thesis that prechill data on one vehicle configuration and size can be extrapolated to other sizes. This subject will be discussed in Section 6.

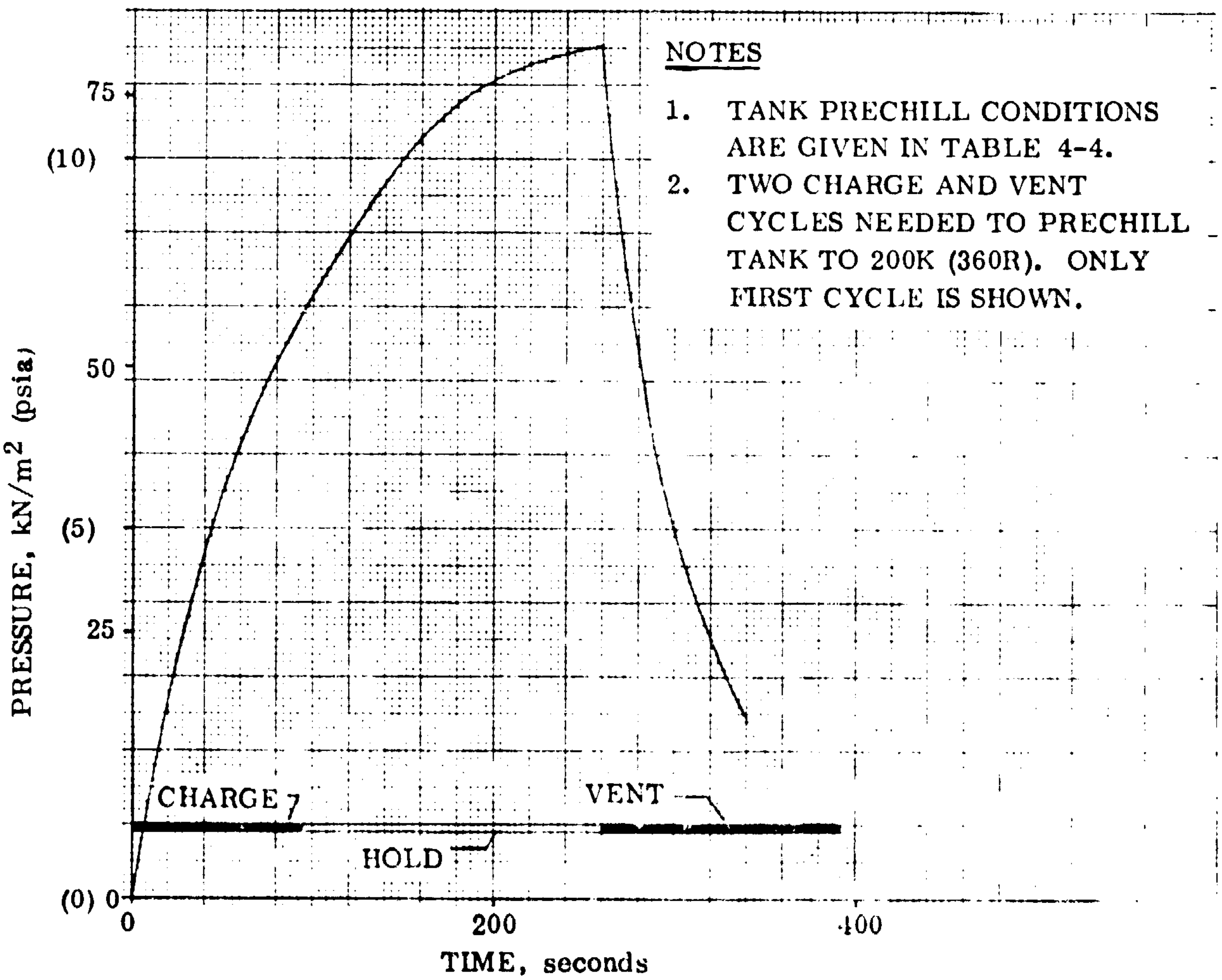

Figure 4-11. COTV Liquid Hydrogen Tank Pressure History During Prechill

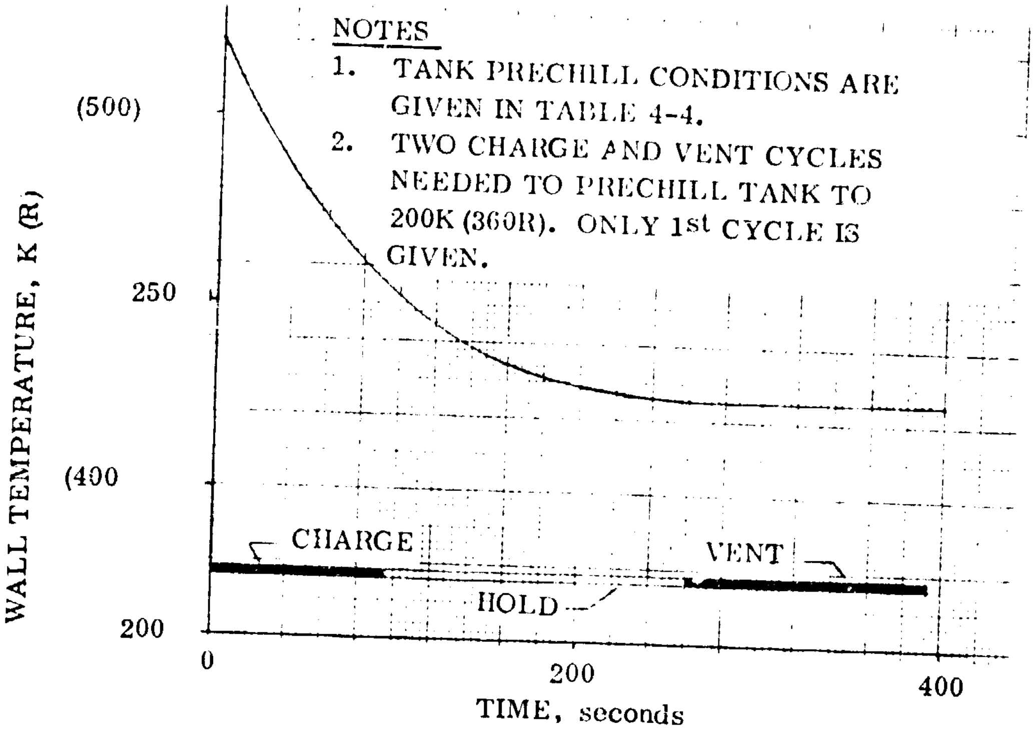

Figure 4-12. COTV Liquid Hydrogen Tank Temperature History During Prechill

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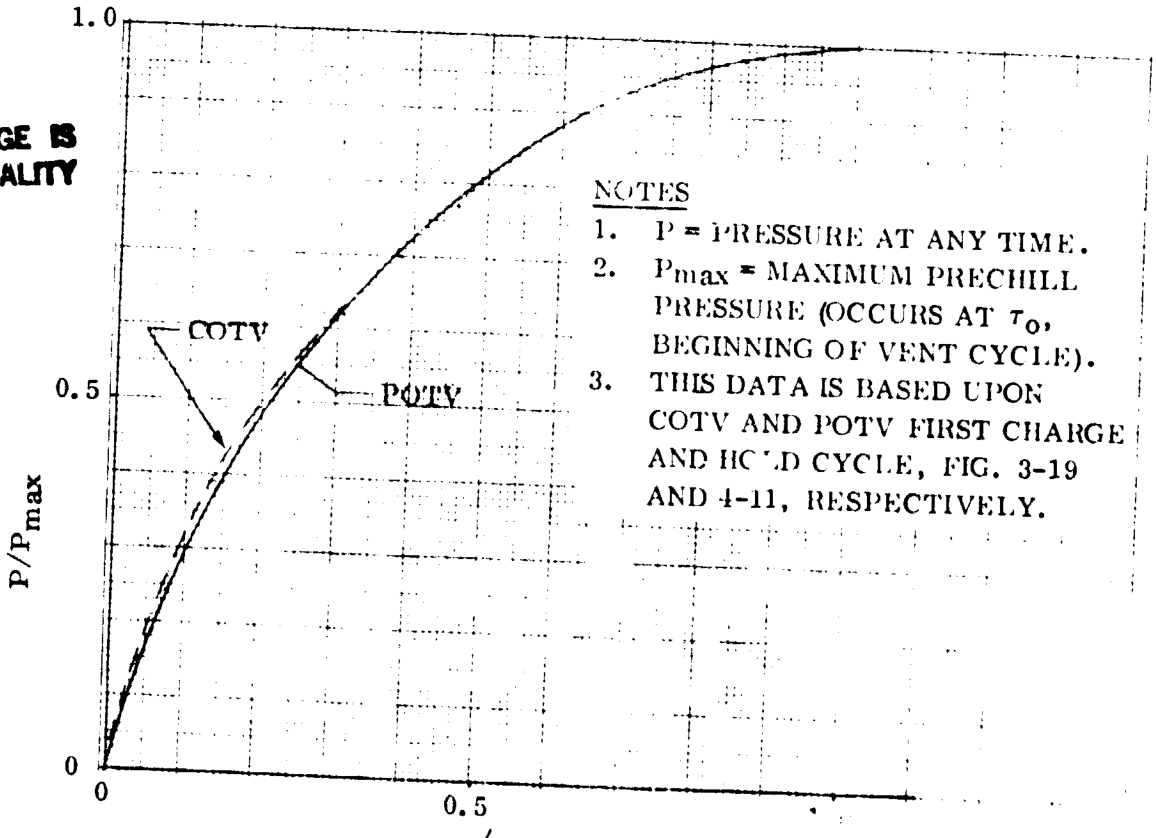


Figure 4-13. Excellent Correlation Exists Between COTV and POTV LH_2 Tank Prechill Predictions

Table 4-4. Conditions Selected for COTV Liquid Hydrogen Tank Prechill Procedure

1. Initial propellant tank temperature = 289°K (520R)
2. Initial pressure = 6.9 kN/m^2 (1.0 psia)
3. Hydrogen vapor saturated at 103.4 kN/m^2 (15 psia) enter propellant tank
4. Entering flowrate = .91 kg/sec (2 lb/sec)
5. Entering velocity = 6.7 m/sec (22 ft/sec)
6. Hydrogen charge terminated after 42.6 kg (94 lb) enters tank
7. Tank vent initiated when tank-to ullage temperature difference becomes 5.6 K (10R)
8. Propellant tank vented to 6.9 kN/m^2 (1.0 psia)
9. Vent area = 148.6 cm^2 (23 in.^{2})
10. Tank mass = 2228 (4911 lb)
11. Tank volume = 548 (19363 ft^3)

4.3.2 COTV TANK FILL. Tank fill will be initiated after prechill is completed. The single requirement for tank fill is to maintain acceptably low pressures during the process. The ideal condition will be to maintain thermal equilibrium during fill, which can be approached as heat and mass exchange between the phases is increased. It was proposed in Section 3.3.3 that near-thermal equilibrium conditions can be attained by introducing propellant to the tanks through spray nozzles. A high energy-exchange rate will be provided during the early part of the fill process (to about a 40 percent fill level) as a result of liquid spray interaction with the ullage. An even greater energy exchange rate will occur during the latter stage of tank fill (about 60 percent to 100 percent liquid fill) due to the interaction of vapor entrained in the liquid bulk.

Figures 3-24 and 3-25 give the relationship between entering hydrogen and oxygen liquid vapor pressure and final tank pressure for thermal equilibrium. This data is applicable both to POTV and COTV propellant tanks. These curves show that tank fill pressures will be maintained within acceptable levels if near-equilibrium conditions are achieved. The HYPRES computer program (which was used for POTV analysis) was employed to determine propellant tank pressure histories for a representative COTV fill condition. Results are plotted in Figure 4-14 for the set of flow conditions identified in Table 4-4. The only difference between the two computer runs is in the entering liquid temperature. The higher receiver tank pressure history is based upon a fixed inlet temperature condition (an idealized assumption). The lower tank pressure

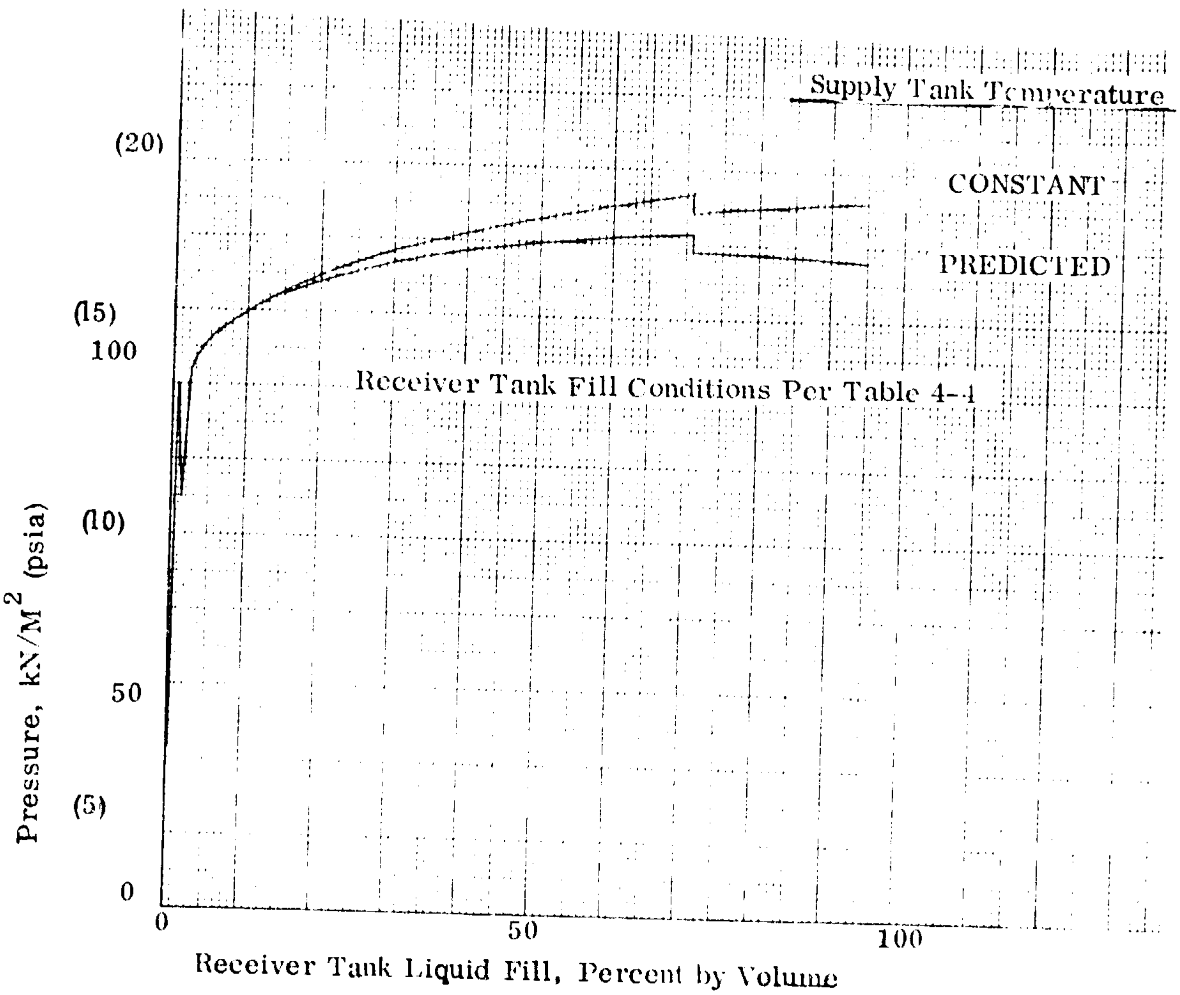


Figure 4-14 Supply Tank Temperature Influence Upon COTV Pressure During Refill.

history reflects the decreasing temperature conditions that occur within the supply tank during propellant expulsion. Figures 4-15 and 4-16 compare inlet hydrogen temperature and pressure histories for the two tank fill conditions.

SUMMARY

Orbital refill operations for a COTV will be very similar to POTV refill. Prechill procedures will be virtually identical, including the requirement to prechill the liquid hydrogen tank below 200K (360R) before tank fill is initiated. The same number of charge and vent cycles can be employed, and the resulting maximum pressures will be the same. Tank fill procedures will also be similar between the two vehicle stages. Propellants will be sprayed into the tanks to enhance mixing between the fluid phases needed to approach near-thermal equilibrium fill. The primary difference between POTV and COTV fill will not be procedural, but thermodynamic. The different receiver-tank-supply tank volume relationships will result in different propellant inlet temperature conditions during fill. The subsequent receiver tank pressure histories

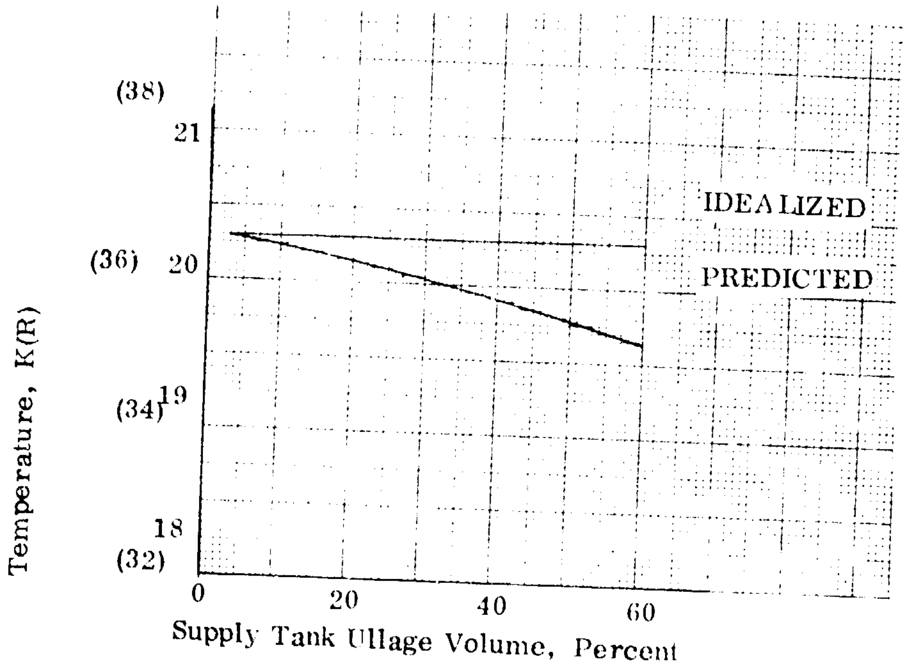


Figure 4-15. Predicted Supply Tank Liquid Temperature During COTV Refill

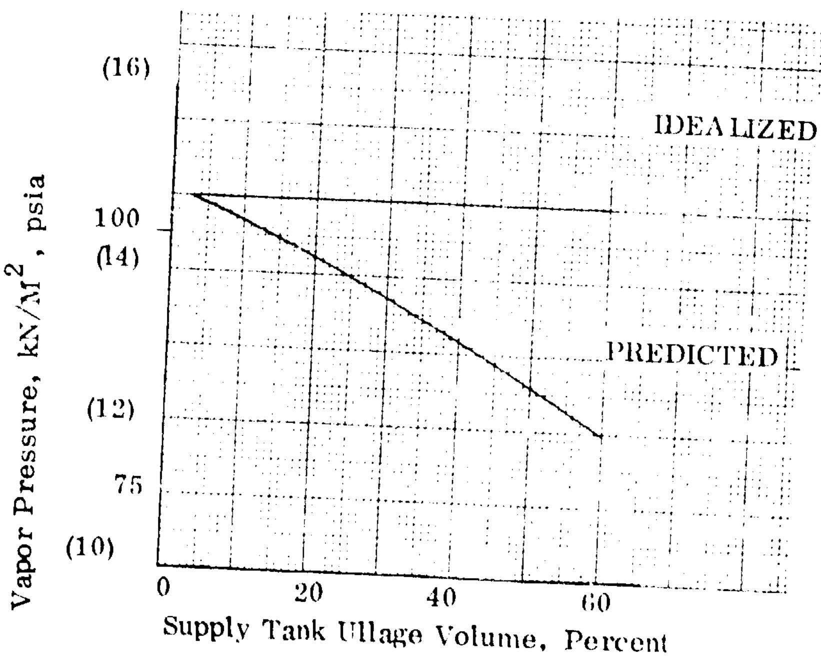


Figure 4-16 Predicted Supply Tank Liquid Vapor Pressure During COTV Refill

will also differ (compare Figure 3-37 to 4-14). This difference, however, is not sufficient to alter the conclusion that POTV and COTV orbital refill operations can be considered identical. Any experimental program devised for one vehicle configuration will also be applicable to the other.

The only new requirements identified by the COTV analysis dealt with alternatives of reclaiming, or employing, the vehicle propellant residuals. It was concluded that reclaiming residuals through a reliquefaction process was marginal at best because of high reliquefier costs. A suprising result was that dumping propellants overboard could be preferrable to reliquefaction. The best alternative, however, could be to use the residuals to supplement the orbital depot RCS system propellant requirements.

5

LTL ORBITAL RESUPPLY

A mission scenario will be developed in this section for the low thrust liquid (LTL) earth storable propellant vehicle concept. A mission will be defined which encompasses the important issues of orbital-refueling operations using earth storable propellants. These operations will include the major activities from post-mission storage in the LEO parking orbit through resupply. Particular emphasis will be placed upon vehicle "safing" operations that can be performed prior to orbiter-tanker rendezvous. Vehicle and orbiter-tanker subsystem requirements needed for orbital refueling will be identified. Operational procedures and techniques for orbital propellant transfer will then be developed.

A groundrule imposed upon this vehicle concept was to utilize hardware from existing/on-going programs, if possible. Because the data base for earth storable vehicles and missions was considerably smaller than for cryogenic OTVs, no attempt was made to optimize the vehicle configuration. Rather, the intent was to select a configuration that would be representative of its vehicle class.

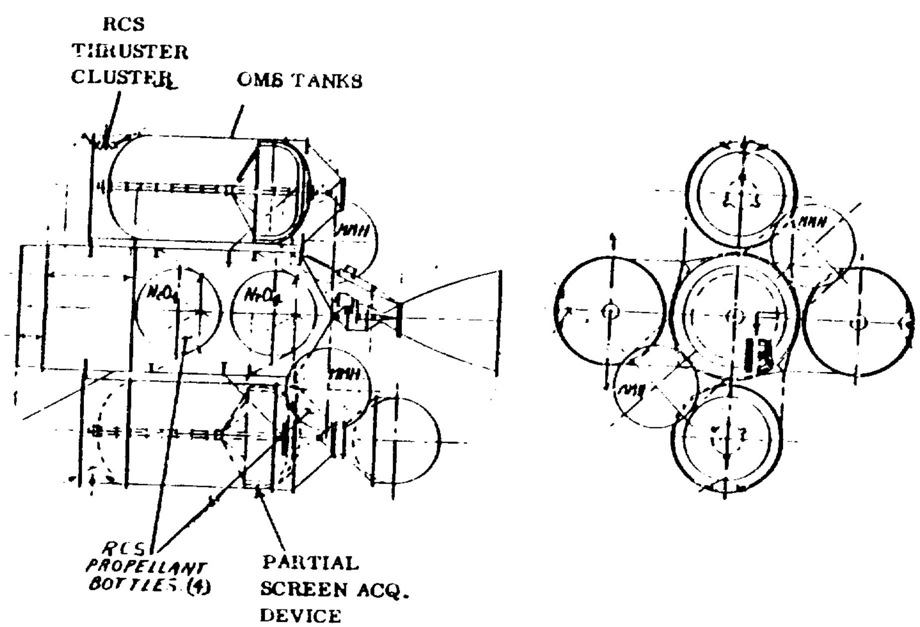
5.1 MISSION SCENARIO

In the 1980's-1990's time frame a requirement may exist for a versatile, low cost vehicle capable of accomplishing small orbital transfers, service, inspection, and retrieval tasks. It will require frequent refill from a dedicated orbiter-tanker. The orbiter-tanker will be equipped to resupply the main propellant tanks and the RCS supply tanks with N_2O_4 and MMH, and the pressurization system bottles with helium.

5.1.1 SELECTED LTL MISSION. As previously stated, there is not the data base available describing potential earth storable vehicle missions that exists for cryogenic POTV's. Fortunately, it appears that a wide range of missions could be selected without impacting orbital refill procedures. For this study, the LTL primary task will be to transfer large space structures (65317 kg (144,000 lb) from LEO (~ 200 n.mi.) to a slightly higher orbit (~ 600 n.mi.) under low accelerations. A vehicle thrust level of approximately 26.7 kN (6000 lbs) will be required.

5.1.2 LTL CONFIGURATION. In keeping with the intent of a low cost vehicle design, the LTL configuration is comprised of existing hardware. Figure 5-1 provides a description of the selected systems, all of which are currently available.

5.1.2.1 <u>Main propulsion tankage</u> — This propulsion tankage system consists of four identical propellant tanks, each containing a screen acquisition device (four galleries)



TANK CONFIGURATION

- MAIN PROPULSION SYSTEM
  - FOUR SHUTTLE OMS TANKS
  - ENGINE SYSTEM 26.7 kN (6000 lbs)
- REACTION CONTROL SYSTEM
  - FOUR RCS BOTTLES
  - RCS MOTORS 98 N (22 lbs)

PROPELLANTS	VAPOR PRESSURE, kN/m^2	(psia)
• N_2O_4	131	(19)
• MMH	6.9	(1)

GROUNDRULE: EMPLOY EXISTING HARDWARE, WHERE POSSIBLE, TO CONSTRUCT VEHICLE

Figure 5-1. LTL Earth Storables Vehicle Configuration and a bulkhead screen, Figure 5-2. These propellant tanks were designed for the Shuttle orbital maneuvering system (OMS) and use N_2O_4 and MMH as propellants. The tanks are designed to operate at approximately 1760 kN/m^2 (255 psia), which is required to satisfy engine inlet conditions. Helium will maintain the operating tank pressures since propellant vapor pressures are relatively low; about 131 kN/m^2 (19 psia) and 6.9 kN/m^2 (1.0 psia) respectively for N_2O_4 and MMH.

The engine system will have a thrust level of 26.7 kN (6000 lbs), which was selected for the mission.

5.1.2.2 Reaction control system — Very little data was obtained on the Reaction Control System (RCS), other than the system (thrusters, supply tank, plumbing) was developed for orbiter and includes a screen acquisition device in each tank. This system is designed for the same operating pressure and uses the same propellants as does the main propulsion system.

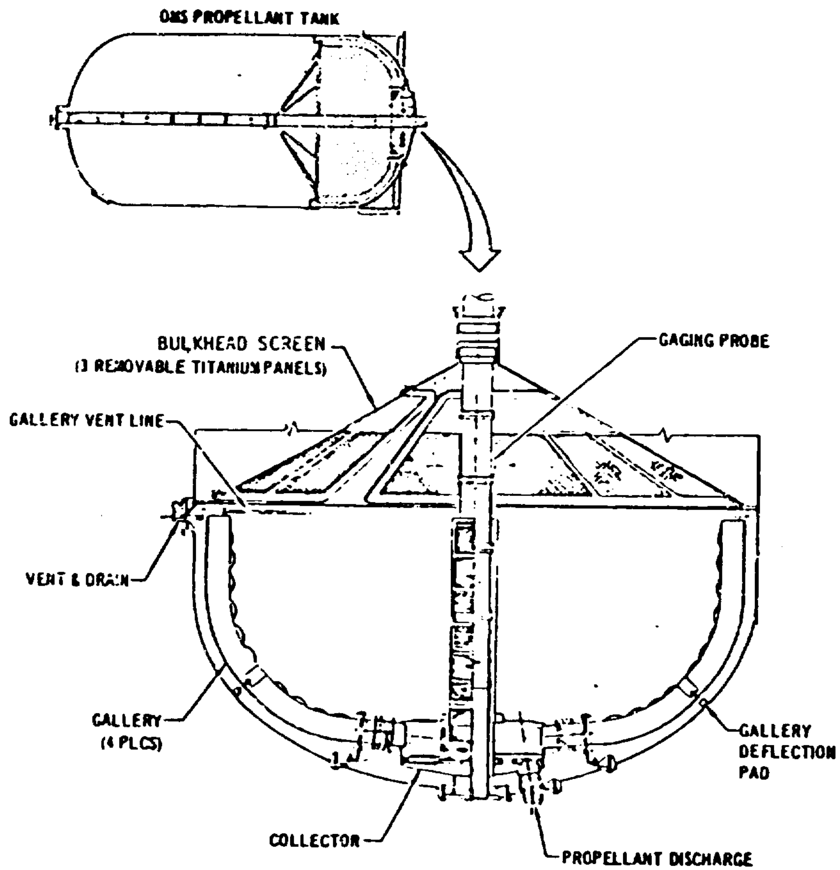
It was assumed for this study that problems associated with orbital refill of the OMS and RCS propulsion systems would be identical. Analysis effort was concentrated on the main propulsion system, and the results would also be applicable to the RCS system.

5.1.3 LTL VEHICLE CONCEPTS. Two vehicle concepts were evaluated to determine which would most closely satisfy the guidelines established for a low cost vehicle. Each concept employs existing hardware.

5.1.3.1 Vehicle concept one — Each of the two Shuttle pod assemblies contains two cylindrical 2.55 m^3 (90 ft^3) propellant tanks for the OMS engines and two spherical tanks for the reaction control systems. Figure 5-3 shows a MMH/N_2O_4 LTL vehicle concept which uses these shuttle tanks. A cylindrical body structure (equipped with a thrust cone), a main engine, four OMS tanks, four RCS tanks, two modular helium bottles, four RCS clusters, an avionics section, and a docking system are the basic vehicle components. The OMS and RCS tanks are supported from the body structure with a system of struts and yoke fittings.

The body structure is a cylinder equipped with kick rings for reacting the loads from the main engine, the tanks, and the payload. The structure also includes provisions at the forward end for mounting astrionics packages. The type of construction is not indicated; however, several are readily adaptable such as the open truss, composite cylinder, skin stringer frame, semi monocque, and the open or closed isogrid.

The main engine is a 26.7 kN (6000 lb) thrust unit with an overall length of 196 cm (77 in.) coupled with an exit diameter of 117 cm (46 in.). The Isp = 310 sec and the expansion ratio = 55. The engine is gimbal mounted for a 7° excursion in any direction using two actuators located 90° apart.



- BULKHEAD SCREEN MAINTAINS PROPELLANTS IN AFT POSITION.

- 100% LIQUID FLOW PROVIDED IF GALLERIES MAINTAIN COMMUNICATION WITH PROPELLANT BULK.

PROPELLANT TANK PROPERTIES

TANK VOLUME $2.55\ m^3$ ($90\ ft^3$)
TANK MATERIAL TITANIUM
OPERATING PRESSURE $1760\ kN/m^2$ (255 psia)

Figure 5-2. OMS Propellant Tank

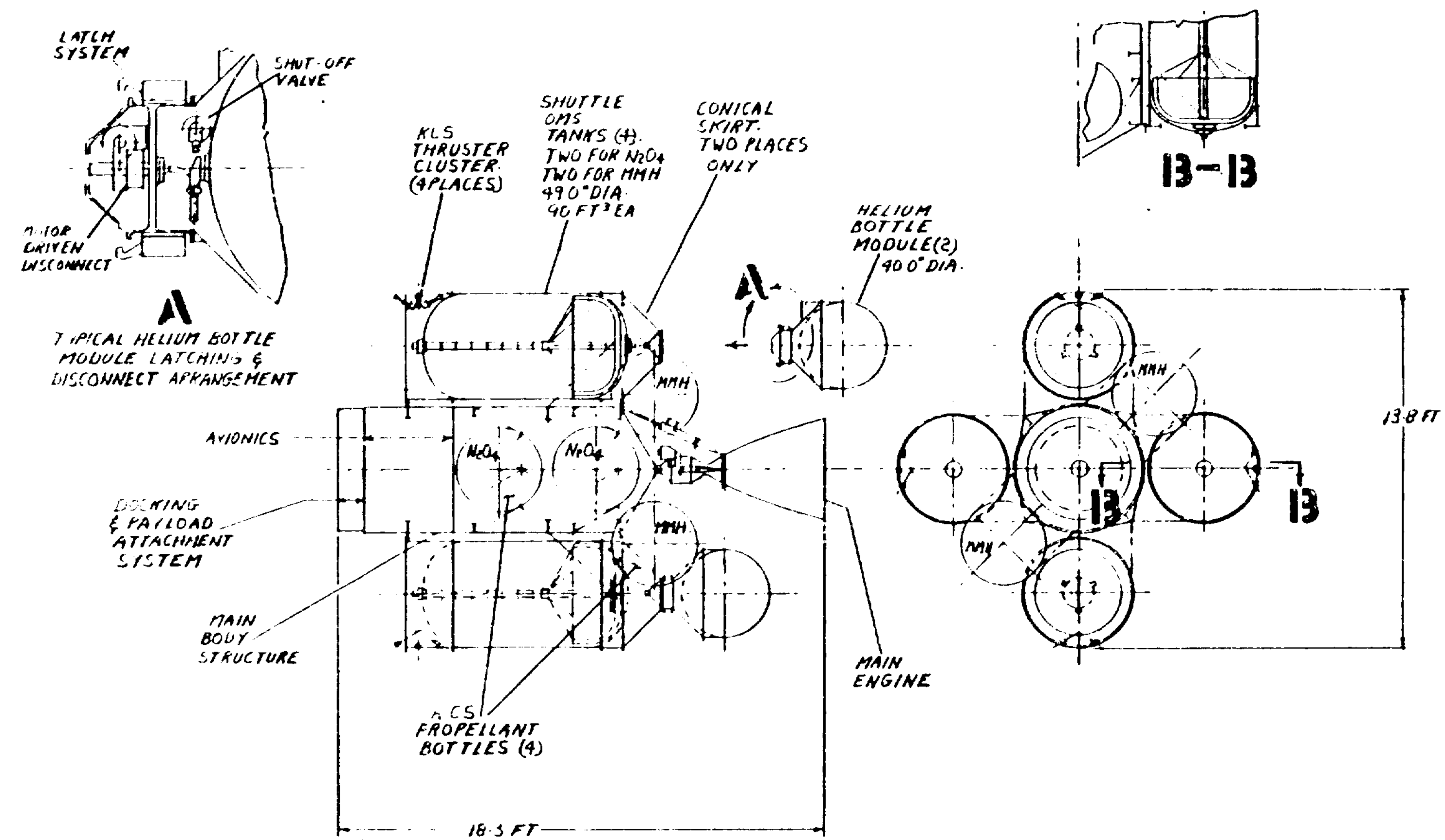


Figure 5-3. Low Thrust Liquid (LTL) Vehicle Concept One

Some minor additions to the OMS tanks will be required such as an aft conical skirt equipped with a disconnect for receiving the helium bottle modules; a forward cylindrical skirt for mounting the ACS clusters and several fittings for structural and plumbing supports. Alterations for plumbing interfaces may also be required to accommodate the vent and fill circuits. The aft conical skirt section previously mentioned will be required for two tanks only; one for each helium bottle module required.

The four RCS tanks are spheres equipped with acquisition devices. Two of the spheres are located inside the body structure and two are located externally near the main engine thrust cone. A strut support system is indicated; however, other methods may be used such as girth flanges or skirts.

The helium supply is two 102 cm (40 in.) spherical modules equipped with a latch system and a motor driven disconnect. Details of this system will be discussed in Section 5.4.2. Additional conceptual information is shown in detail "A" of Figure 5-3.

A typical reaction control system (RCS) cluster consists of four thrusters mounted inside the OMS tank skirt structure ... pointing outboard (see layout). Four clusters are used and are 98 N (22 lb) fo... unit with an Isp = 280 sec and a mixture ratio of 1.65. The approximate overall length of each thruster is 25.4 cm (10 in.) Flange type mounting is provided.

The astrionics section is located at the forward end of the body structure. The four OMS tanks shadow this section therefore heat dissipating systems may be required such as heat pipes and radiators. Possible locations for the radiators would be between the OMS tanks.

The docking and payload attachment system is located forward of the astrionics and interfaces with the shuttle or the payload. Features such as gross capture under wide misalignments, shock absorbing, pull down and final alignment, followed by structural attachment would be included in this system.

5.1.3.2 <u>Vehicle Concept 2</u> — In Concept 1, two types of propellant storage tanks are shown. For the main propulsion system, four OMS tanks are used and for the RCS system, four spherical tanks are used. An alternate approach is shown in Figure 5-4 which uses six OMS tanks. Two of these tanks serve the RCS system and the remaining four supply the main propulsion. Similar to Concept 1, the tanks are attached to a cylindrical body structure equipped with a thrust cone. In this case the thrust cone is reversed to minimize the vehicle length.

Two of the RCS clusters are mounted on the forward ends of two OMS tanks and the remaining two clusters are supported from an open truss structure located between the OMS tanks. The purpose for the truss structure is to permit a 90° spacing between clusters. The truss is shown attached to the main body section, however,

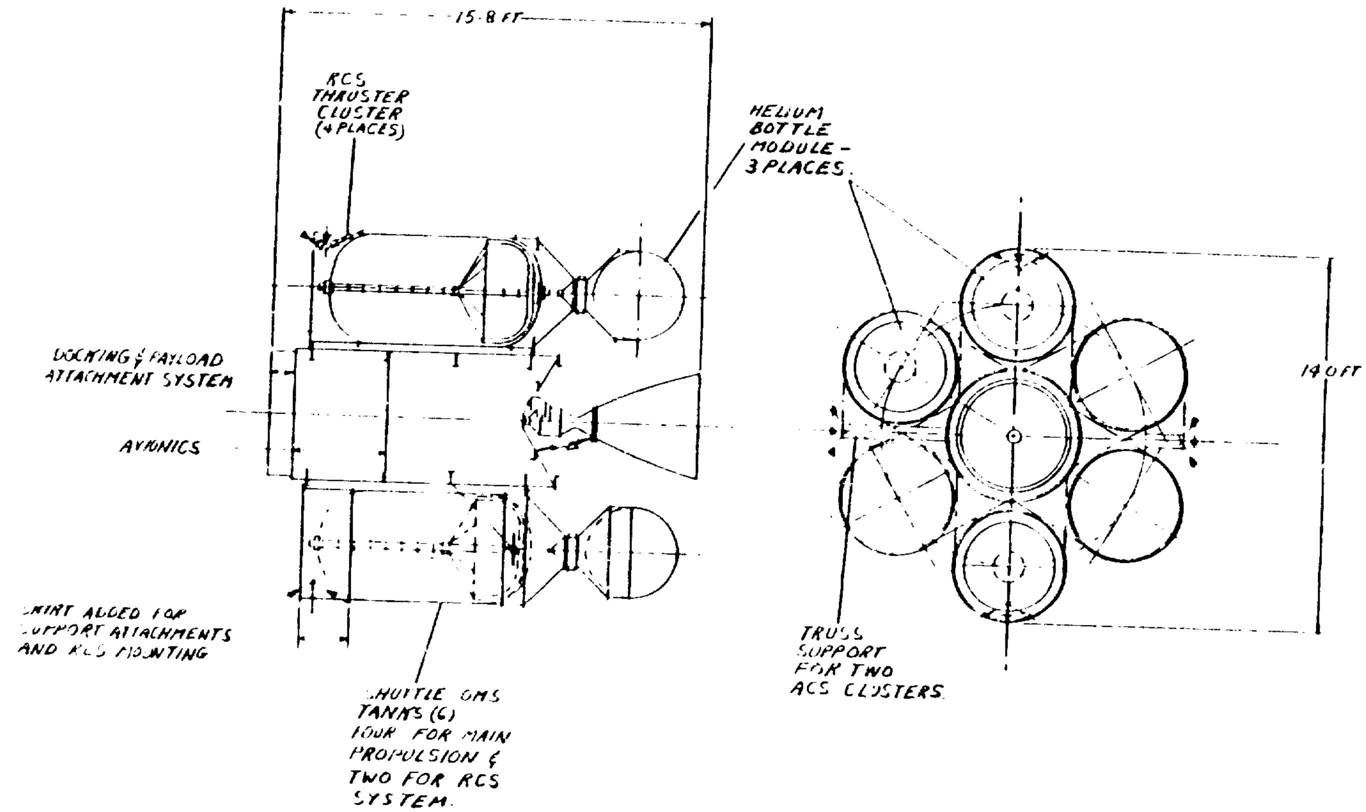


Figure 5-4. Low Thrust Liquid (LTL) Vehicle Concept 2

other methods could be used such as a bridge strucuure between two OMS tanks or a cantilever support fitting attached to one OMS tank.

Similar to Concept 1, modular type helium bottles are used. For this case, three bottles are required.

The primary advantage of concept 2 over concept 1 is that it could be simpler to construct. There is a single type of propellant storage tank (the OMS tank) rather than both the OMS tank and RCS tank. Furthermore, structural attachments would be simpler for the six common tanks. It is possible that concept 2 would be less costly than concept 1. The primary disadvantage of concept 2 is that it is a heavier vehicle. The two OMS tanks weigh more than the four RCS spheres they replace. Also, the increased volume of the OMS tanks will require an additional helium bottle for pressurization.

Problems and/or solutions to on-orbit resupply should be similar between concepts 1 and 2. These are refill procedural differences that may favor concept 1, which is why this concept was selected for evaluating orbital refill. These differences will be discussed in Section 5. 3. 2.

5. 1. 3. 3 Fluid systems for concept 1 — Figure 5-5 shows the basic plumbing for the propellant fill, vent and cross-over circuits. A system schematic is given in Figure 5-6. Plumbing for the main propulsion, RCS, and pneumatic systems are shown. Referring to the schematic, the overall system for each propellant consists of a fill circuit and a vent circuit. On the fill side, each pair of OMS tanks and also each pair of RCS tanks are tied together and fed with a common line which starts at the disconnect. The OMS tanks only are equipped with internal fill tubes incorporating spray nozzles. All tanks are equipped with shut-off valves for controlling, filling and transferring.

For the vent circuits, each pair of tanks are tied together and routed through a non-propulsive overboard vent. Each tank is equipped with a vent valve.

Tube routings can vary considerably depending upon vehicle configuration, supports, etc., therefore the purpose of Figure 5-6 is to show only a general approach as to how a system may be laid out. Referring to the N_2O_4 side, the fill circuit starts at the disconnect which is supported from the aft conical skirt on one of the OMS tanks. The fill line is routed to the nearest OMS tank and across the vehicle to the opposite OMS tank. The loop shown in the cone line is due to support attachments on the thrust cone and on the conical skirts. The same arrangement is used for the MMH. For the N_2O_4 tanks located inside the body structure, the fill tubes are routed aft to the main cross over line. For the MMH side, these tubes are routed forward to the cross-over line.

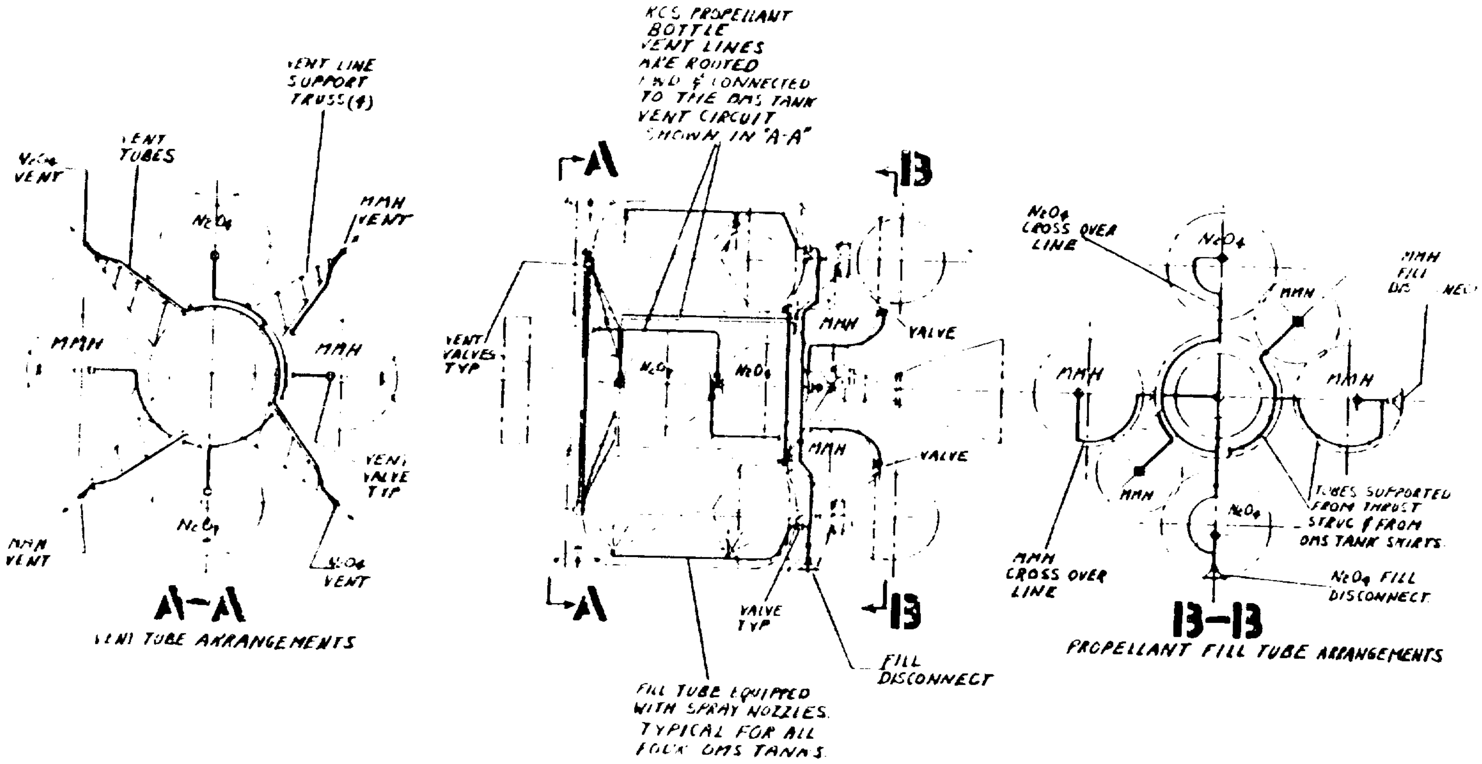


Figure 5-5. LTL Concept 1 Fluid System Plumbing Arrangement

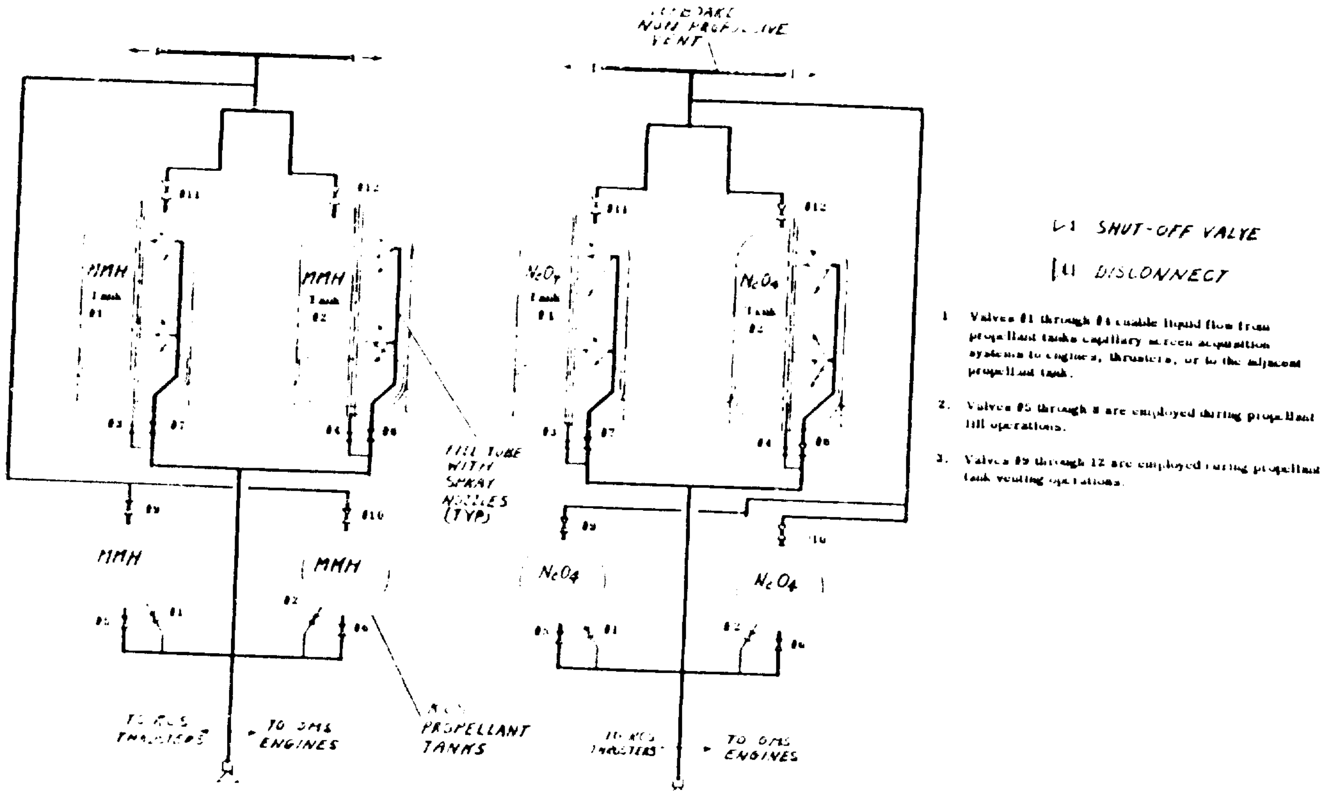


Figure 5-6. LTL Concept 1 Fluid Systems Schematic

A non-propulsive vent system for each propellant is located at the forward end. The exits for each of these circuits are positioned outboard to prevent impingement on the OMS tanks, payload and adjacent structures. Four open truss arms attached to the body structure provide support for the tubing. Basically the non-propulsive vent is a loop of tubing with the ends of 180° apart. This arrangement is frequently referred to as the "steer horn". The vent lines from each tank are connected to this loop of tubing such that there is equal distance from each point of connection to the overboard ends. A typical tube assembly would feature welded joints and the use of bends instead of fittings where possible. Large sections would be bench fabricated to ensure maximum quality control.

The fill and vent circuits were designed for the purpose of enhancing on-orbit propellant refill. Refill must be conducted with caution because of the corrosive nature of the earth storable propellants. Refill can be difficult because the OMS and RCS propellant tanks must first be vented from about 1760 kN/m^2 (255 psia) to 207 kN/m^2 (30 psia) in order to expel sufficient helium to enable refill.

5.2 ORBITAL PROPELLANT RESUPPLY TECHNIQUES

The obstacles of on-orbit refill are the same for the LTL vehicle as for a cryogenic POTV. These obstacles are: 1) the hostile space environment and (2) the limited resources available for space-based operations. The primary variable complicating refill is the zero-g environment.

Propellant tank fill with N_2O_4 or MMH in an orbital environment poses fewer problems than liquid hydrogen because of temperature environment and operating pressure level differences between a earth storable and cryogen system. System temperatures will not vary significantly for these propellants at any time during a fill or refill operation. Consequently, prechill will not be required to precede tank fill.

Propellant tank operating pressure of 1760 kN/m^2 (255 psia) is considerably higher than for liquid hydrogen. This higher pressure level will simplify tank fill because fill pressures will be maintained below the normal operating levels without difficulty.

One problem that cryogenics and earth storables have in common is that an undefined (or poorly defined) liquid-vapor distribution will exist during low-gravity tank fill. This inability to determine vapor location will make it difficult both to assure that no liquid is vented during tank fill, and to assure that vapor is not trapped within the screen channel device.

For the analyses that follow attention will focus upon the problems associated with N_2O_4 and with main propellant tank refill. N_2O_4 is selected because it represents a more difficult propellant to handle than MMH due to its higher vapor pressure. The main propellant tanks were used in these analyses.

5.2.1 PROPELLANT TANK REFILL REQUIREMENTS. Several requirements were identified as being necessary to assure that LTL vehicle refill will be safely and effectively performed. These are general requirements that should apply to a variety of vehicle configurations. There will undoubtedly be configuration-related requirements for any vehicle. However, such requirements cannot be identified without first having considerably more vehicle design detail than was available for this study.

5.2.1.1 <u>Vent propellant tanks prior to orbiter rendezvous</u> — N_2O_4 is a particularly corrosive fluid, in vapor or liquid form. Fluid impingement upon the LTL or orbiter surfaces could have a long-term adverse affect upon vehicle components. The orbiter will be spared this potential hazard if LTL vent procedures are performed well in advance of orbiter rendezvous.

5.2.1.2 Minimize liquid vent potential — This requirement addresses two major concerns; propellant corrosiveness and vehicle control. Liquid venting must be minimized because it is potentially more damaging than vapor venting. First, a liquid vent plume cannot be readily confirmed, i.e., liquid exposed to a vacuum will boil and expand in all directions. Also, the impingement mass flux from a liquid vent is likely to be greater than from a vapor vent plume, which increases the corrosive potential.

The second aspect to the potential problems of liquid venting is vehicle control. Because procedures will be identified for remote venting of the LTL propellant tanks, it is mandatory that vehicle control be maintained. Liquid venting could jeopardize vehicle control because it is unlikely that a net zero thrust would result, even if venting through a non-propulsive (designed for vapor) vent system.

5.2.1.3 <u>Helium must not enter screen galleries</u> — The purpose of the screen acquisition device in each propellant tank is to provide the capability for pure liquid flow. Any helium entering a screen device will be trapped and remain trapped until removed through special procedures, or until it flows from the tank to the engine system. Special procedures for removing trapped helium are undesirable because they are time consuming and may be complicated. Helium expulsion with propellant is unacceptable because it violates the requirement of 100 percent liquid availability to the engine system.

5.2.2 INITIAL FILL. It is unlikely that the vehicle will undergo an initial propellant fill in space because of the ease with which it can be filled on the ground prior to flight. There is a possibility, however, that the propellant tanks could be completely evacuated at a future time for maintenance. Following such an occurrence propellant tank fill would be performed on evacuated tanks residing at ambient temperature.

5.2.2.1 <u>Non-equilibrium fill</u> — The high operating pressure levels will simplify OMS propellant tank fill because fill pressures will be maintained below the normal operating levels without difficulty. This is illustrated in Figure 5-7 which shows maximum tank pressure as a function of percent liquid fill. Isentropic compression of the ullage is assumed, which means that heat exchange with the liquid and tank walls is zero. Note that 90 percent fill can be effected without exceeding 1378 kN/m^2 (200 psia) pressure. Unlike cryogenics this extremely conservative approach can be used because results are acceptably low.

5.2.2.2 Thermal equilibrium tank fill — Figure 5-7 provides evidence that initial tank fill will be readily achieved. It is more reasonable, however, to expect near-thermal equilibrium conditions to exist during fill. This process is described below.

Thermal equilibrium fill represents the minimum propellant tank pressure condition that can exist during fill. The thermal equilibrium tank fill relationship derived for liquid hydrogen (Equation 3-23) applies as well to N_2O_4. This equation is given below.

$$h_L = u_{g_2} m_{g_2}/m_{L_2} + u_{L_2} + \Delta u_w m_w/m_{L_2} \tag{5-1}$$

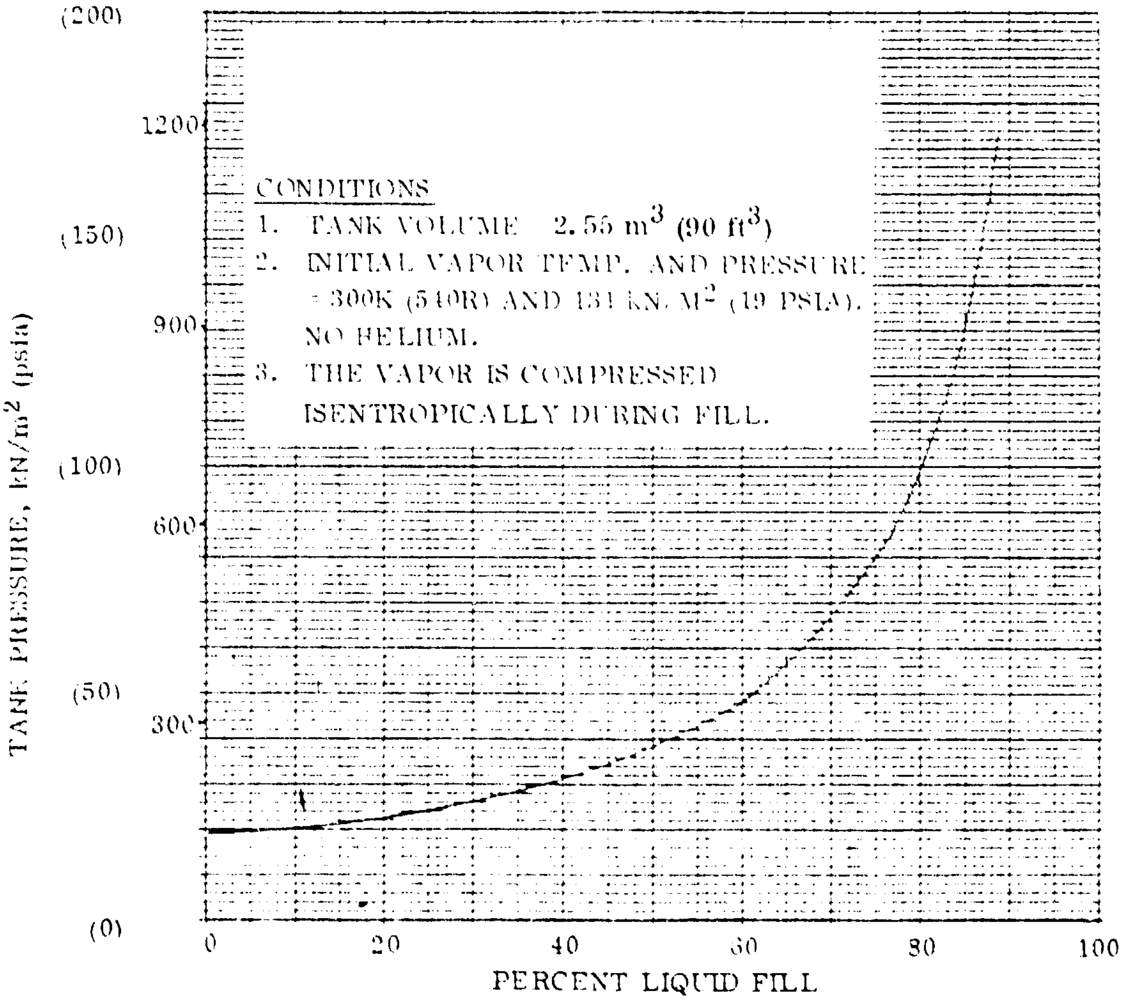


Figure 5-7. Maximum OMS Propellant Tank Pressure During Initial Fill With N_2O_4

where

h_L = enthalpy of entering liquid final vapor

$u_{g_2} m_{g_2}$ = total internal energy

m_{L_2} = final liquid mass

$\Delta u_w m_w$ = storage tank wall energy change during fill

u_{L_2} = final liquid internal energy

It is possible to relate h_L, u_{L_2} and u_{g_2} to liquid vapor pressure and temperature under thermal equilibrium conditions. Thus u_{L_2} and u_{g_2} are known once final vapor pressure is specified. Δu_w can also be determined, when initial and final temperatures are given. Finally, h_L (which is a function of entering liquid vapor pressure) can be determined for a desired liquid fill condition.

Equation 5-1 is summarized for N_2O_4 in Figure 5-8 which gives entering liquid vapor pressure as a function of initial tank temperature and final tanked liquid vapor pressure. It is interesting to note that the effect of initial wall temperature upon final tank pressure is negligible. This data illustrates that the N_2O_4 thermal mass will so dominate tank fill that the tendency will be for final tank pressures to approach that of the incoming liquid vapor pressure.

The same fill technique, that of liquid spray into the tank, will be employed for N_2O_4 as for liquid hydrogen. This approach will assure that N_2O_4 will be at least as close to thermal equilibrium as liquid hydrogen, at the same fluid power inflow conditions. This assessment is made on the basis that the liquid-to-vapor thermal mass ratio is six times greater for N_2O_4 than for hydrogen. Thermal equilibrium is more readily achieved for fluids having high thermal mass ratios.

5.2.3 ON-ORBIT REFILL. Most, if not all, problems associated with LTL vehicle resupply will be associated with the need to vent helium before propellant refill can be initiated. Helium venting must be conducted with care under orbital conditions because liquid may also be vented. In addition, helium may enter the screen channel device (or galleries) during this period, unless precautions are taken.

Helium venting is necessary prior to attempting refill in order to avoid excessively high tank pressures during the refill process. To illustrate this point, helium partial pressure in a N_2O_4 tank will increase from 1758 kN/m^2 (255 psia) to 15820 kN/m^2 (2295 psia) as the tank is filled from 10 percent liquid to 90 percent liquid.

The refill process will not represent a concern once sufficient helium has been vented to avoid high partial pressures at small ullage volume conditions. Thus the key to a

CONDITIONS

1. TANK VOLUME = 2.55 m^3 (90 ft^3)
2. TANK MASS = 253 kg (558 lb_m), TITANIUM
3. TANK PRESSURE IS 0.0 kN/m^2 (0.0 psia) PRIOR TO FILL
4. NO VENTING DURING FILL

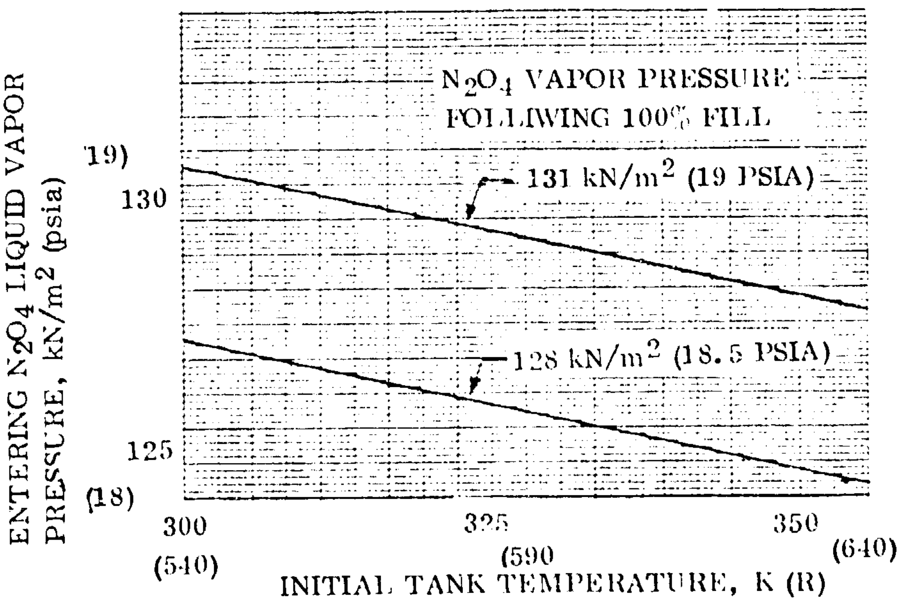


Figure 5-8. Final Storage Tank Pressures for N_2O_4 Thermodynamic Equilibrium Fill Process

successful refill is helium venting which precedes this process.

5.2.3.1 Thermodynamics of propellant tank vent — Tank venting must guarantee that the screened volume will not be contaminated by helium entry. Helium cannot penetrate the device while it remains filled with propellant. However, once vapor resides within the device, helium entry can occur as a result of molecular diffusion or forced convection flow. Thus a vent procedure must be selected that will maintain the screen device filled with liquid.

There are two phenomenon of concern during tank venting: liquid boiling and surface evaporation. Liquid boiling within the galleries must be avoided. Surface evaporation cannot be avoided, but propellant must be available for liquid make-up. The following steps describe a vent process selected to satisfy the requirement of keeping the screen galleries liquid-filled (N_2O_4 properties are assumed because a procedure acceptable to N_2O_4 is also acceptable to MMH):

1. The partial pressure of helium and N_2O_4 vapor will decrease as tank pressure is decreased. N_2O_4 evaporation will begin at the screen surface once liquid

vapor pressure exceeds the partial pressure of the vapor.

2. Liquid will be drawn into the capillary device to replace liquid lost due to surface evaporation at the screen (Figure 5-9). Thus, sufficient liquid volume surrounding the galleries, and in communication with the contained liquid, must be available for liquid replacement. This quantity must be determined.

3. The liquid bulk cannot boil while pressure exceeds liquid vapor pressure. Consequently, tank vent will be terminated at a pressure greater than 131 kN/m^2 (19 psia) to assure that the capillary device remains filled.

   Note: The only evaporation (or boiling) that can occur is at a liquid-vapor interface, and only if vapor pressure at the surface exceeds partial pressure above the surface.

It is estimated that propellant tank pressure must be reduced to about 210 kN/m^2 (30 psia) prior to starting propellant refill. Approximately 12.7 kg (28 lb) of N_2O_4 vapor will be vented in the process. Less than 2 kg (4.4 lb) of MMH vapor will be vented.

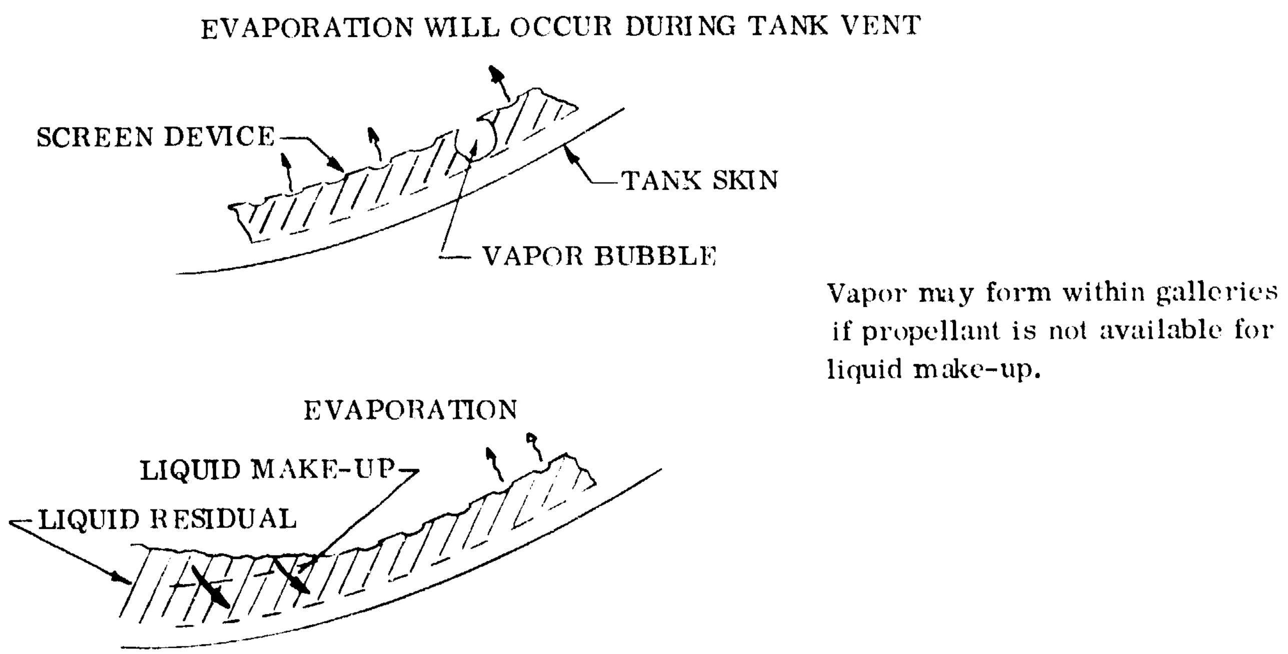


Figure 5-9. Propellant Tank Venting Can Occur Without Losing Liquid From Screen Galleries

5.2.3.2 Propellant tank refill — Propellant tank refill can be performed in a manner similar to that of initial tank fill. As with initial fill, the high operating pressure levels will tend to simplify tank refill. Unlike the initial fill condition, helium in the ullage precludes using the most conservative approach of isentropic compression to verify that refill can be achieved. There is a less conservative method, however, to illustrate that tank refill can be readily accomplished. Figure 5-10 indicates that refill to the 87 percent level will be possible even for isentropic compression. The goal, however, is to achieve a refill of 95 percent, which can readily be attained if near-thermal equilibrium conditions are maintained during refill. According to the discussion of Section 5.2.2.2, the liquid-to-vapor thermal mass of N_2O_4 and helium is such that near thermal equilibrium conditions will be maintained. Figure 5-10 indicates that refill can be achieved even if the ullage is superheated by 55.5 K (100R) above the liquid. Such a temperature differential cannot possibly be sustained within the OMS tanks as propellant enters through spray nozzles. The RCS tanks do not include spray nozzles but, even so, a 55.5°K (100°R) temperature differential will be virtually impossible to sustain. It is concluded that refill will be a straightforward operation.

5.3 HELIUM VENTING

There are two categories of potential problems associated with venting the LTL vehicle OMS tanks; one is liquid venting, and the other is helium entry to the capillary device.

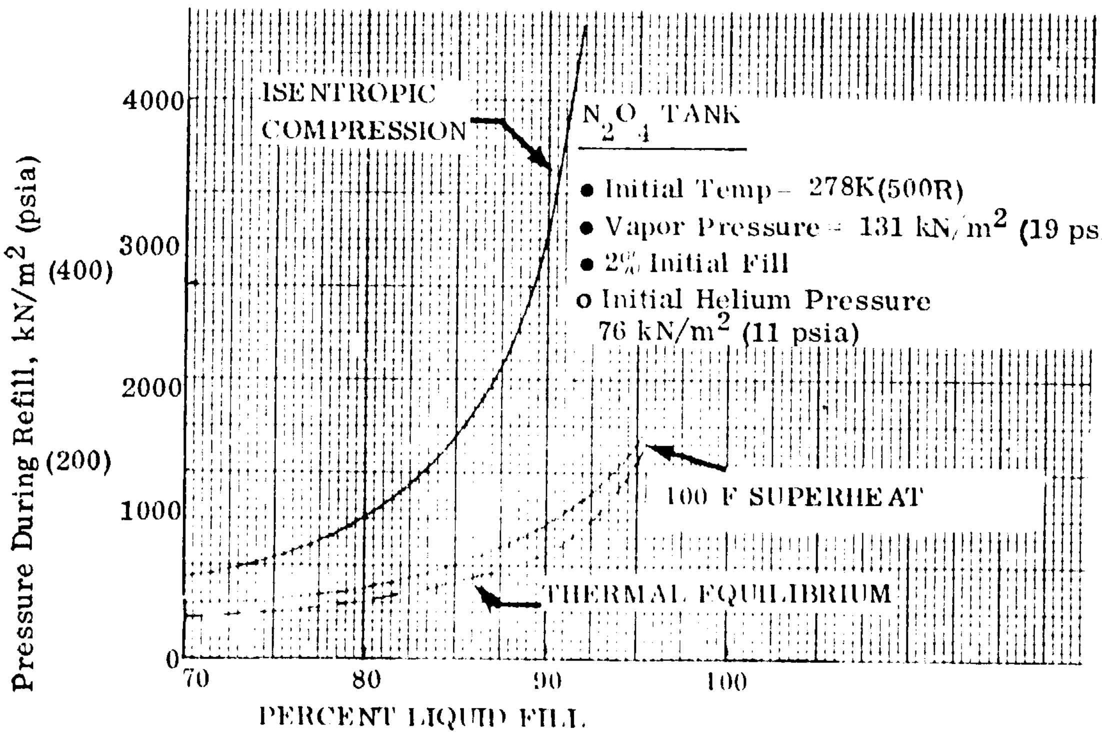


Figure 5-10. OMS Propellant Tank Pressures During Refill

The loss of liquid overboard during the venting process is not only an inefficient operation, it also represents a corrosion hazard if N_2O_4 is vented. Because of its corrosive properties even N_2O_4 vapor venting must be carefully performed to avoid vapor impingement upon LTL vehicle and orbiter surfaces. Liquid venting will represent a more serious concern because the vapor cloud formed when liquid is exposed to the vacuum environment will expand in all directions and be difficult to control.

Regarding the second category of problems, tank venting must guarantee that the screened volume will not be contaminated by helium entry. Helium cannot penetrate the device while it remains filled with propellant. However, once vapor resides within the device, helium entry can occur as a result of molecular diffusion or focused convection flow. Thus a vent procedure must be selected that will prevent the loss of liquid from the screen device.

5.3.1 ALTERNATIVE VENT PROCEDURES. Various alternatives were considered for the tank venting procedure. An overview of each alternative, is presented below.

1. Venting while the propellants are settled by the OMS main engines or the RCS. This concept will eliminate the possibility of venting liquid overboard. A major disadvantage is that the tank cannot be completely vented down. This is because both the main engines and the RCS require a substantial operating pressure.

2. Venting while docked to the orbiter with the propellants settled by the orbiter RCS. This concept also eliminates the possibility of venting liquid overboard. It will also be possible to completely vent the OMS tank down since the orbiter RCS is used to settle propellants. A possible disadvantage is that firing of the orbiter RCS engines after docking has been completed may be an operational complexity. The major disadvantage is that this approach violates the requirement for venting prior to orbiter rendezvous.

3. Venting while docked to the orbiter with the propellant settled by atmospheric drag. This concept eliminates the possibility of venting liquid overboard. One disadvantage is that an orbital altitude < 130 n.mi. is required for aerodynamic drag to overcome propellant surface tension forces, which is substantially below the desired altitude of 200-250 n.mi. Another disadvantage would be the necessity of attaching a vent duct extension to keep the corrosive vapor of the earth storables from contacting the vehicle surface. The requirement for a vent duct extension is based on calculations, plotted in Figure 5-11 of the maximum Prandtl-Meyer expansion angle of an earth storable-helium mixture. The same major disadvantage exists as for Item 2, above.

4. Venting with the propellant unsettled prior to docking with the orbiter. This concept-approach could result in the venting of liquid overboard, which violates a major requirement. It would also require the use of a vent tube to prevent vehicle contact with the vented propellant.

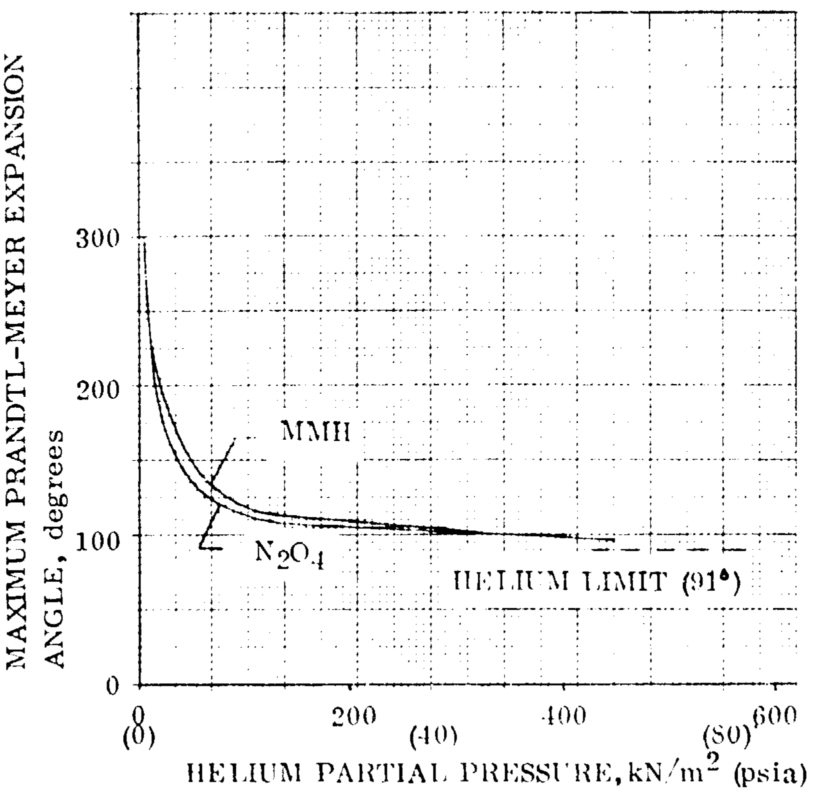


Figure 5-11. Expansion Angle of Ideal N_2O_4 - Helium and MMH-Helium Mixtures as a Function of Helium Partial Pressure

5. Venting through a helium recovery system prior to docking with the orbiter. The objectives of a helium recovery system are a) to eliminate the loss of helium pressurant as a result of venting overboard, and b) to eliminate the hazards of venting corrosive fluids. The recovery system, illustrated in Figure 5-12, is a closed system. Referring to Figure 5-11, a description of the system components is as follows: The molecular sieve is used to absorb any N_2O_4 or MMH vapor contained in the vapor mixture. The multi-stage compressor pumps helium from the OMS tank pressure up to a storage bottle maximum pressure of 33,100 kN/m^2 (4800 psia). However, the helium must be passed through a radiator between each of the compressor stages to reduce vapor temperature to the allowable temperature range of the helium storage supply tank.

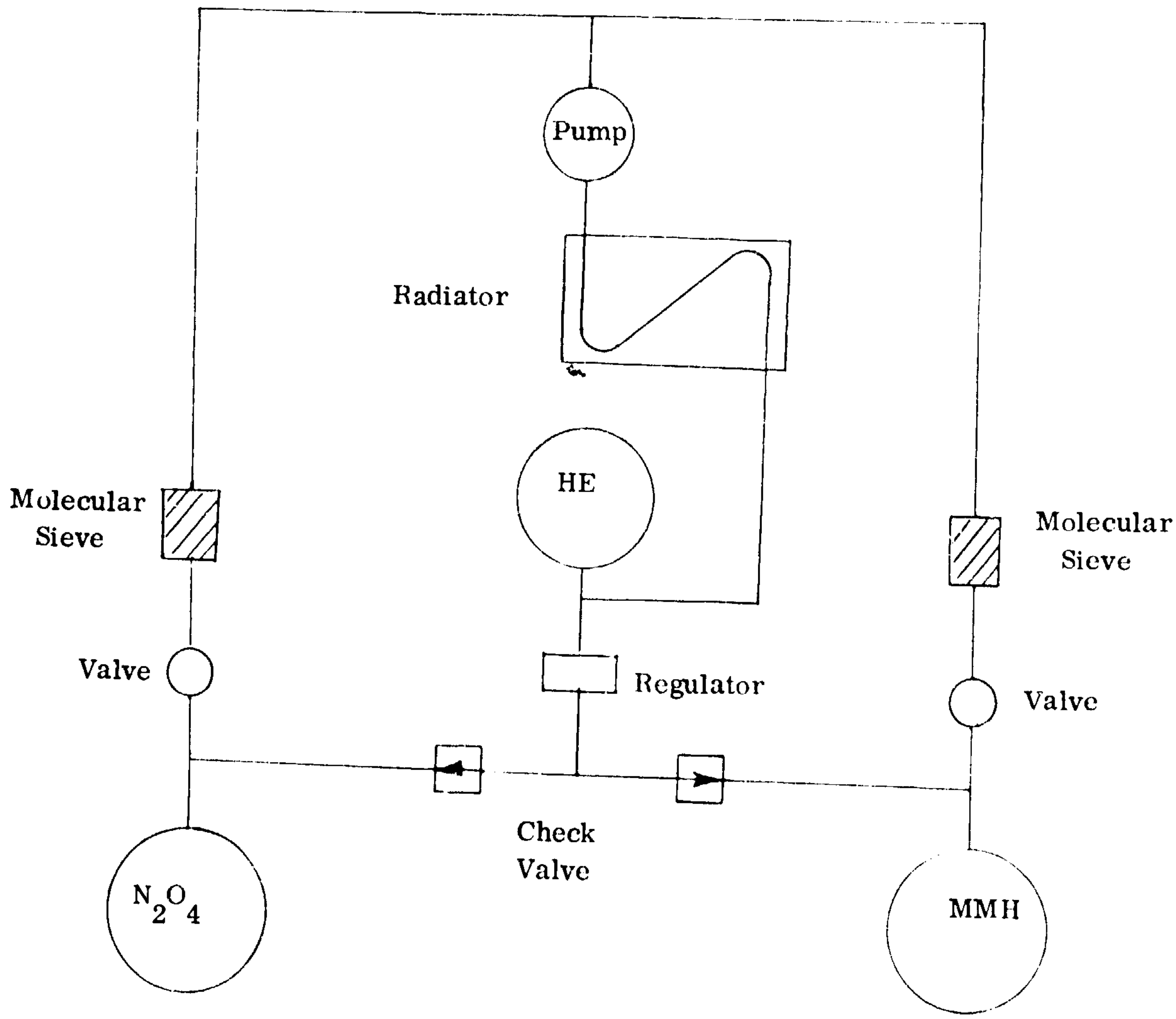


Figure 5-12. Helium Recovery System

This concept requires that the helium recovery system be contained as a package within the LTL vehicle.

6. Venting with the helium recovery system following docking with the orbiter. The helium recovery system used would be identical to that described in method (5). However, the system now would be a unit contained on the orbiter as opposed to a unit contained on the LTL vehicle. Consequently, there would be no need to provide special attention to the attitude control system.

The helium recovery system is a desirable solution to the venting problem since it provides for the continued reuse of the helium pressurant and eliminates any vent hazard. It does represent a major and, perhaps, costly vehicle modification because of the development effort required to integrate a compressor, space radiators and molecular sieves. Furthermore this hardware addition represents a reduction in payload capability and an increased vehicle complexity.

5.3.2 SELECTED VENT PROCEDURE. The preferred procedure for venting helium while minimizing or eliminating the problems of liquid venting is given in Table 5-1. The advantages of this approach are that it is simple and requires no development. Basically the approach is to utilize the two sets of main propellant tanks and RCS propellant tanks for transferring propellants from one tank to another. Table 5-1 describes how any given propellant tank can be drained by transferring propellant to an adjacent tank. Once drained, the tank can be vented with a minimum concern for liquid venting. It is visualized that the procedure can be initiated on a signal from the ground or from the orbiter. Also, portions of the procedure will be automated so that valves can be commanded open and closed on the basis of continuous monitoring of propellant tank pressures and propellant mass gaging output signals. Additional assurance for liquid free venting can be provided, if necessary, by commanding the RCS settling thrusters to fire prior to and during all venting operations.

The Table 5-1 procedure is applicable to N_2O_4 and MMH as well as to the main and RCS propellant tanks. At the completion of this venting operation one set of propellant tanks will be empty (except for a minimum liquid residual volume) and the other set of tanks will contain the bulk of propellant residuals. Tank pressures will be approximately 207 kN/m^2 (30 psia) in all tanks except for a set of RCS propellant tanks. These tanks cannot be vented until after docking with the orbiter is complete because LTL attitude control capability must be maintained until that time, and this capability will be lost once the RCS tanks have been vented. The two tanks will be vented in exactly the same manner as described in Table 5-1.

The following LTL vehicle conditions will exist prior to initiating refueling operations:

1. One each of the OMS and RCS propellant tanks will contain minimum liquid residuals. The other set of tanks will contain propellants in excess of the minimum liquid residuals.
2. Propellant lines are primed with liquid.
3. All screen devices are primed with liquid.
4. All tank pressures are about 207 kN/m^2 (30 psia).

The vent procedure of Table 5-1 is better suited to vehicle concept 1, which has a second set of RCS tanks, than vehicle concept 2. This second tank set provides the capability of first draining propellants from the RCS tanks to be vented while docked to the orbiter. This step greatly minimizes the possibility of losing liquid overboard during the RCS tank vent process. With concept 2, the RCS (OMS) tanks would be vented while containing all propellant residual.

Table 5-1. LTL vehicle tanks venting procedure prior to orbiter rendezvous.

1. Close the shutoff valve between helium supply and OMS MMH tanks. This will enable tank venting without helium resupply through the pressurization system pressure.
2. Monitor the zero-g mass gauging devices of tanks #1 and #2 to determine which contains fewer propellants. Note: The tank with less propellant will be drained first. (Assume for this discussion, that tank #2 has less propellant.)
3. Close valves #3, #4. Valves #7 and #8 are also closed (Refer to Figure 5-6 schematic).
4. Open valve #11. This enables venting through the non-propulsive vent system.
5. Close valve #11 when pressure in tank #1 decays from the initial 1720 kN/m^2 (250 psia) to TBD kN/m^2. This pressure will automatically be selected by computer which continually monitors mass gauging device output.

The procedures will not be performed simultaneously for both propellants because of the desire to avoid simultaneous venting of MMH and N_2O_4 vapors.

6. Open valves #3 and #4. The pressure difference will enable propellant transfer from tank #2 to tank #1.
7. A signal to close valves #3 and #4 will be sent when the mass gauging device indicates that minimum liquid residuals remain in tank #2.
8. Open valve #12 and vent tank #2 from 1720 kN/m^2 (250 psia) to about 207 kN/m^2 (30 psia). The possibility of liquid loss during venting has been minimized because the bulk of propellants was previously transferred to tank #2. Helium penetration of the screen device will not occur during venting because the device is filled with liquid. Furthermore bulk liquid boiling will not occur until tank pressure decays to 131 kN/m^2 (19 psia) for N_2O_4 and 6.9 kN/m^2 (1 psia) for MMH. Tank #2 is now ready to be refilled.
9. Close valve #12 and open valves #3 and #4. The pressure difference will allow propellant transfer from tank #1 to tank #2.
10. A signal to close valves #3 and #4 will be sent when the mass gauging device indicates that minimum liquid residuals remain in Tank #1.
11. Open valve #11 and vent tank #1.
12. Close valve #11 when tank pressure has decayed to about 207 kN/m^2 (30 psia). Tank #1 is now ready to be refilled.

5.4 PROPELLANT REFILL PROCEDURE

The procedure of Table 5-2 has been selected as being applicable to the transfer of MMH and N_2O_4 into the OMS tanks and RCS tank from the orbiter supply tanks. These procedures include the steps needed to assure that propellant will not leak overboard during refill. Two items require particular attention; leakage through an improperly sealed disconnect valve, and residual propellant spilling from the transfer line after disengaging the transfer line disconnect for return to the orbiter cargo bay.

Any propellant spillage must be avoided, or minimized, because of its corrosive nature. The disconnect valve was of sufficient concern that design requirements were established, and a conceptual design was developed. Valve design and operation are further discussed in Section 5.4.1. Residual propellant spill can be readily avoided through a purge procedure which permits helium entry at the transfer valve disconnect and flow toward the supply tank. This low flowrate purge will force the liquid bulk back to the orbiter supply tank.

5.4.1 EARTH STORABLE PROPELLANT DISCONNECT VALVE. A preliminary design of an earth storable disconnect valve is shown in Figure 5-13. The valve is installed on the end of the Shuttle transfer line and makes both structural and fluid seal connections with the LTL prior to the transfer of propellants from the Orbiter-tanker. The Shuttle RMS attaches to the valve housing and deploys the valve and transfer line to the LTL mating interface. The transfer line is equipped with a power cable for operating and monitoring the valve.

As previously stated, Figure 5-13 is a preliminary design effort, and prior to a final selection, it would be required to generate several design options. For example, several actuating methods involving pneumatics, hydraulics, and electro mechanical devices would be included in the tradeoffs. Sealing is also a critical item and will require numerous investigations. The structural attachment system is another area which will require tradeoffs.

It was assumed for this design effort that the valve shall (1) be capable of attaching to the LTL within the positioning tolerance band of the Shuttle RMS, (2) incorporate final alignment provisions before fluid sealing, (3) feature flat sealing surfaces (no probes in holes), and (4) have zero spillage when disconnected. The valve shall also include systems for monitoring the seals before and after transfer. An electro-mechanical method of actuation was assumed.

Referring to Figure 5-13, the main sections of the valve are a flex duct assembly, two actuators, an outer housing, three latch systems, a mating passive assembly which is installed on the LTL, and seal monitoring systems. The overall dimensions shown are larger than initially expected and can be reduced by additional design refinements.

Table 5-2. LTL vehicle propellant tanks refill procedure (applicable to N_2O_4 and MMH).

1. Attach orbiter propellant transfer line to the LTL vehicle. This process includes the following steps as a minimum:

   a. Structurally engage transfer line disconnect to vehicle disconnect.
   b. Perform leak test of disconnect seal cavities to verify that the system is leak-free. This test is performed with GH_e.
   c. Vent GH_e overboard. The transfer line is now engaged and fully evacuated, but the vehicle disconnect valve is still closed.

2. Pressurize the N_2O_4 supply tank to TBD kN/m^2.
3. Open valve at supply tank outlet. N_2O_4 will fill the transfer line.
4. Open vehicle disconnect valve. A fluid path now exists from the supply tank to the vehicle.
5. Open valves #7 and #8. Propellant transfer to Tanks #1 and #2 is in progress. (Refer to Figure 5-6 schematic.)
6. Close valves #7 and #8 when mass gauging devices indicate the tanks are full. Note: The initial tank pressures of 207 kN/m^2 (30 psia) will guarantee the tanks can be filled without exceeding the 1720 kN/m^2 (250 psia) operating pressures.
7. Open valves #5 and #6 to commence RCS tanks refueling.
8. Close valves #5 and #6 when mass gauging devices indicate the tanks are full.
9. Vent supply tank to reduce pressure to TBD kN/m^2.
10. Close vehicle disconnect valve and purge transfer line with GH_e employing the following procedure:

    a. Close supply tank valve.
    b. Open supply tank acquisition device by-pass valve.
    c. Initiate low flowrate GH_e purge through transfer line. GH_e enters at transfer valve disconnect and flows toward supply tank, forcing N_2O_4 into the tank.

11. Disengage transfer line disconnect and return to orbiter cargo bay. The LTL vehicle N_2O_4 refueling operation is now complete.

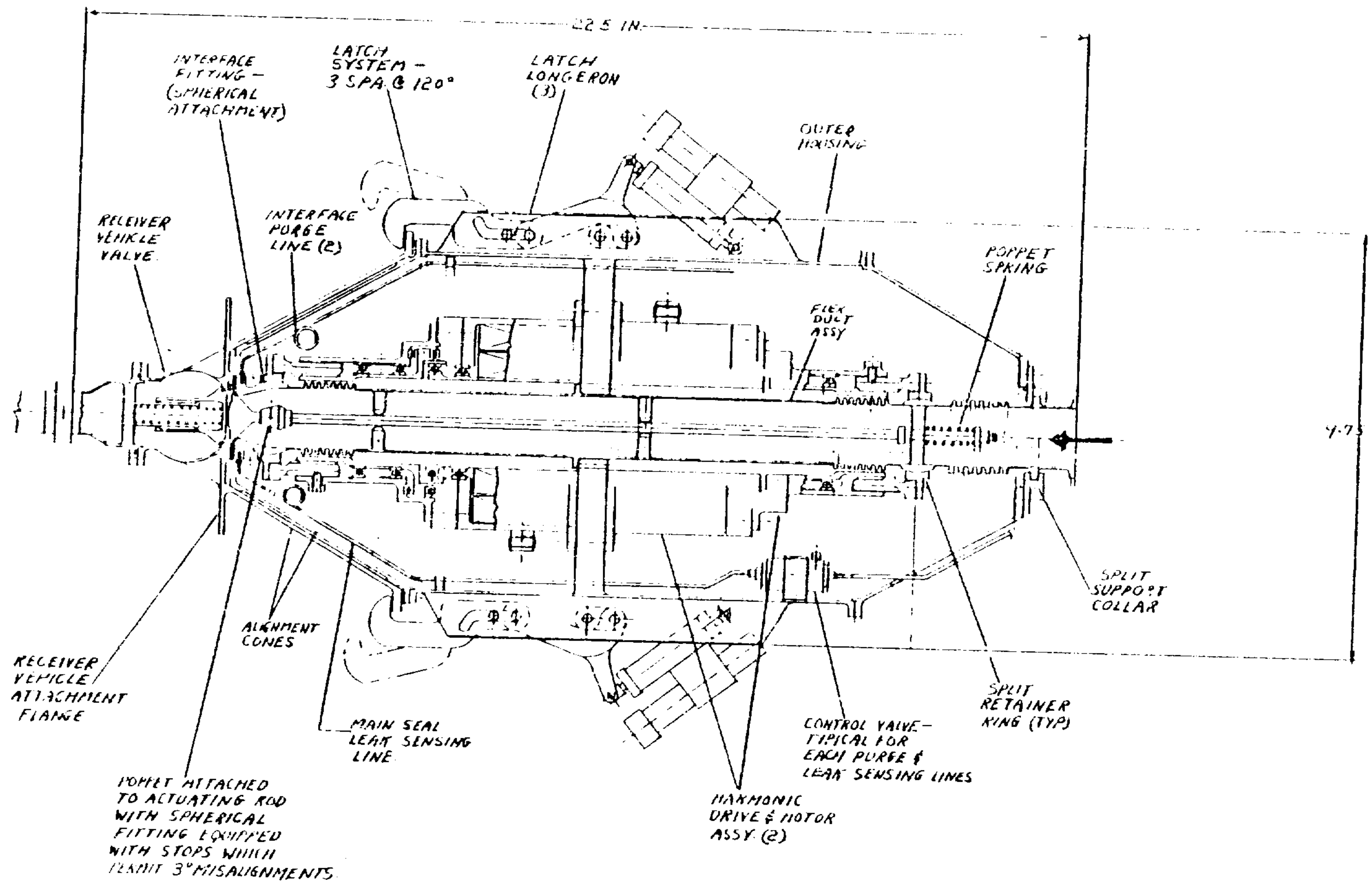


Figure 5-13. Earth Storable Propellant Disconnect Valve

The flex duct assembly runs the full length of the valve and consists of two short tube sections, three bellows, one poppet with an actuating rod assembly, two guide vane fittings, one mounting flange, one interface fitting with seals, one fitting for actuating the poppet and one inlet fitting. Except for the seals and the poppet assembly, all parts are 304L CRES and are welded into one assembly. The seals would probably be a rubber type compound molded or bonded into the interface fitting and the poppet. The poppet assembly will probably be 304L CRES machined parts assembled with threaded fittings. The poppet spring material is 178 Inconel.

The flex duct assembly is the heart of the valve system. Opening or closing is accomplished by compressing or extending the bellows with actuators located outside the flow stream. No dynamic seals are required. The spherical connection between the actuator and the interface fitting plus the ball socket type connection between poppet and actuating rod permits angular misalignments between sealing surfaces.

The actuator system is two harmonic type drives. Each drive is integrated with a hollow shaft, electric motor. The drives and motors are configured to allow the flex duct assembly to pass through at the center line. To permit assembly, each drive has a split collar for attaching to the flex duct assembly. This type of actuator system permits a compact assembly but does require numerous parts. The main parts for each drive are one wave generator, one flex spline, a circular spline and a dynamic spline. Approximately seven bearings will be required (plus retainers), including two for the motor. Two additional screw jack parts are also required on the output end of each drive.

The outer housing consists of a cylindrical section which is equipped with a conical fitting at each end. One conical fitting has two external machined lands plus a flange for mating with the conical section on the LTL. The purpose for the conical mating surfaces is to permit loose engagement even with large misalignments plus accurate alignment when the cones are pulled together.

The second conical fitting has two flanges; one for attaching to the cylindrical section and one for the flex duct attachment. This conical fitting provides a rigid interface for connecting the transfer duct and reacts the pressure area loads from the flex duct bellows. To permit assembly, a split collar is used at the flex duct attachment.

The outer housing cylindrical section has three external longerons for supporting the latching systems and an internal spider beam for supporting the actuators. Each end is also equipped with a flange for attaching the cones. A fitting for attaching to the RMS is located between two of the longerons. This fitting is not shown on the layout.

All outer housing parts are 2219 aluminum alloy. The cones can be either single spun formed parts or weldments consisting of rolled cones and machined ring flanges. The cylindrical section is a weldment. The latching systems are basically an over center device powered by an electromechanical screw jack actuator. The locking pawls are capable of rotating outboard combined with axial displacement away from the housing cone. The pawls can also rotate inboard followed by an axial movement toward the housing cone. This type of kinematics permits gross initial capture with the mating cone on the LTL followed by draw down and final alignment.

The passive section of the valve which attaches to the LTL is a single piece cone and valve body equipped with a spring loaded poppet. The valve body features a flat faced sealing surface equipped with flow passages for interface purging and seal monitoring. Except for the poppet spring, the entire assembly is 2219 aluminum alloy.

Flow passages are provided on both sides of the sealing plane. These passages provide a means for monitoring leakage and for purging the interface cavity. Small tubes with flex loops are routed from these passages to solenoid control valves. The circuits are activated with GH_e. Other additional means for monitoring leaks is the use of sniffers. GH_e supply for the system indicated can be from tubes routed along the transfer duct or from a small bottle (with controls) attached to the valve housing. A second small bottle (attached to the valve housing) can also be used to act as a catch reservoir when purging the interface cavity. The use of bottles attached to the valve simplifies the transfer line assembly since only one electrical cable is required.

An operation sequence is shown on Figure 5-14. Referring to Step 1, the two valve sections are in a gross capture mode. At this stage the latches are rotated inboard which captures the cone flange on the LTL side.

In Step 2 the latches are actuated parallel to the valve axis pulling the two cones together. The structural connection is now completed.

In Step 3 one of the harmonic drives is actuated which engages the primary seals with the mating flat face. Stop lands machined on the flange containing the seals controls the amount of squeeze on the seals. A leak check is performed by pressurizing the cavity between the two primary seals and monitored for pressure decay.

The second harmonic drive is actuated in Step 4 which opens the valve for transfer. During transfer, the main seal is monitored with a sniffer located in the GH_e supply tube.

The valve is closed at the completion of propellant transfer, as indicated in Step 5, and the small interface cavity purged. The sealing arrangement for this purge circuit is shown in Detail "A". Before disengaging, the transfer line is purged and the poppet seal on the LTL side is checked for leaks by activating the sniffer circuit in the tube leading to the interface cavity.

In Step 6, the main seal is disengaged, the latches opened, and the valve separated by the RMS.

5.4.2 HELIUM BOTTLE RESUPPLY. Space-based vehicles will require gaseous helium resupply during refueling operations. One method of resupply is by flowing helium from a storage tank (located in the Shuttle payload bay) to the vehicle through a long transfer line. An alternative would be to employ separate pre-loaded modular helium bottles that would be externally attached to the vehicle. A disconnect system capable of mating a structural attachment and a fluid connection with the vehicle is required with either approach. Several disconnects are required of the modular bottles, however, as compared to only one for a transfer line.

The orbiter remote manipulating system (RMS) will be employed to connect transfer line or modular bottles to the vehicle. The RMS will place the disconnects reasonably close to the mating target. Pressure-area loads will be reacted only by the disconnect structure, and not by the RMS.

5.4.2.1 Helium transfer from orbiter — The transfer line approach will have an electrical power cable attached to the line and permanently connected to the disconnect system located at the end of the line. The transfer line must have mobility which in turn requires flex joints either in the form of braided hoses, swivel type joints with dynamic seals, or loop bends in the tubing. Conventional bellows joints would not be applicable due to high pressure requirements. The use of hoses, swivel joints and loop bends presents packaging and weight penalties. Also, the safety of this transfer operation is of concern because a 21m (70 ft.) line, pressurized to 34500 kN/M^2 (5000 psi), will be essentially unsupported except at the orbiter and at the vehicle.

Aside from the question of safety is the problem of excessive helium temperatures occurring the bottle charging period. This problem exists because the heat of compression generated during the charge period cannot be readily dissipated; and may require the aid of space radiators. Potential solutions are summarized in Table 5-3, only one of which was considered acceptable.

5.4.2.2 Helium Modules — The preferred method for LTL vehicle helium resupply is to use helium bottle modules.

Basically, the pre-loaded helium modules are picked up with an RMS and plugged into external LTL interfaces which in turn have interconnecting plumbing to the LTL systems.

Detail "A" of Figure 5-3 shows a typical helium module which is a high pressure bottle equipped with a tangential support skirt, a docking cone, a latching system, a shutoff valve and a motor driven disconnect. The tangential skirt section has a fitting which interfaces with the RMS. The only requirement in the case of a Shuttle RMS

Table 5-3. LTL Vehicle Helium Re-supply Options

	VEHICLE HARDWARE REQUIREMENTS	ADVANTAGES	DISADVANTAGES
Requirement: Refill LTL helium bottles while in orbit			
Problem: Ambient temperature helium transferred from orbiter to OMS and RCS helium bottles will create excessive helium temperatures during bottle charging period.			
Solutions:			
A. Store helium in orbiter at reduced temperatures	None	Bottle charging can be conducted in a timely manner.	Desired storage temperature not readily attained
B. Cool helium by flowing through radiator before it enters helium bottles.	None	Bottle charging can be conducted in a timely manner.	Desired radiator outlet temperature may not be attained
C. Charge slowly with helium and allow bottles to radiate energy to space.	None	No modifications.	Charge process may be time consuming
D. Combine B and C	None	Bottle charging can be conducted in a timely manner.	Desired radiator outlet temperature may not be attained.
* E. Replace expended helium bottles with full helium bottle.	LTL pressurization systems must be re-designed for a modular concept.	Helium bottles replacement can be conducted in a timely manner.	Modification to LTL.

* Recommended solution.

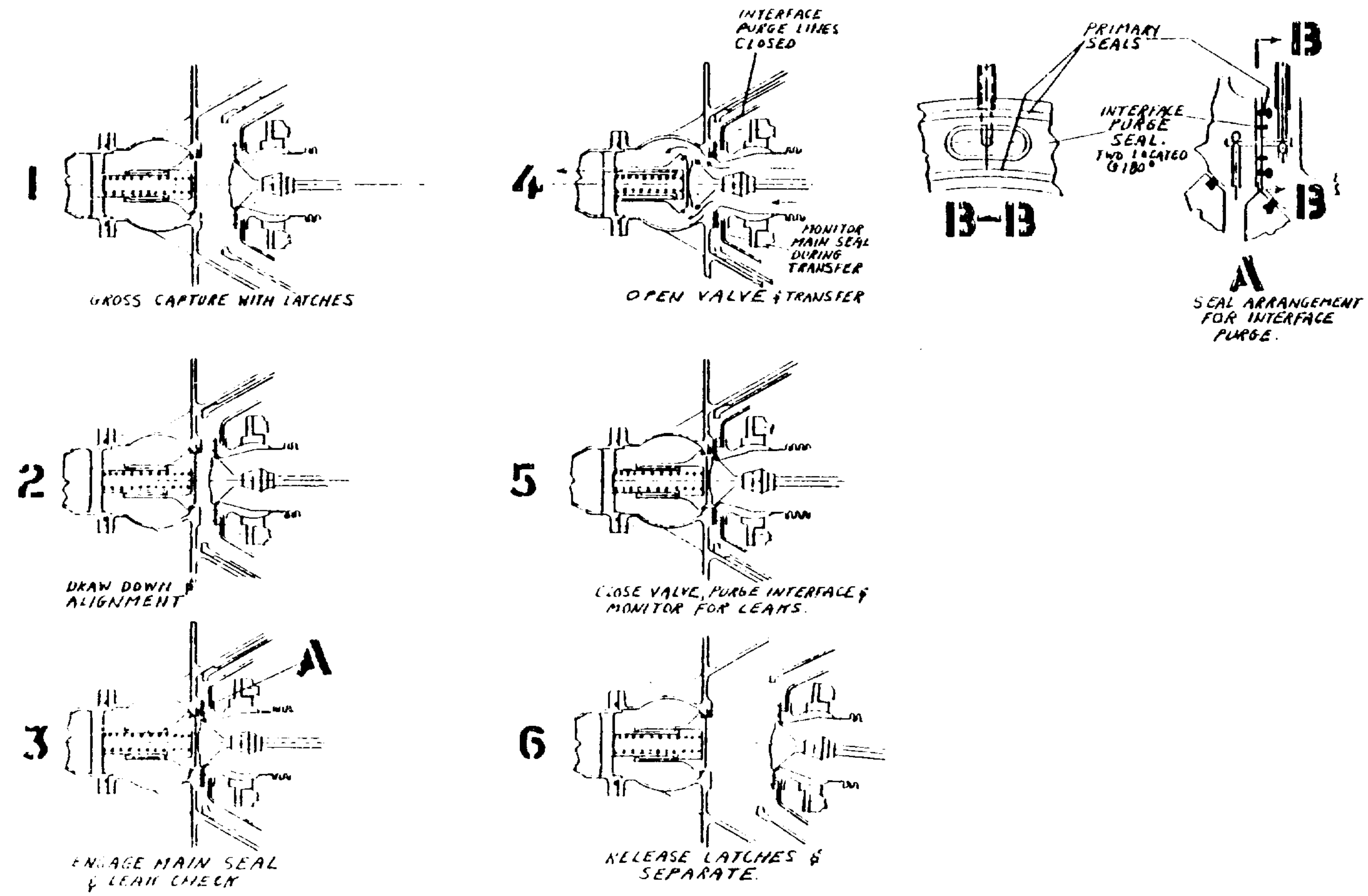


Figure 5-14. Operation Sequence of Earth Storable Propellant Discount Valve

would be the addition of an end fitting. This fitting would include a power cable from the Shuttle for actuating the latch and disconnect systems.

A typical transfer sequence consists of connecting an RMS to the helium module; placing the module loosely (wide tolerances) into a mating cone on the LTL; actuating the latches to an inboard position to insure a gross capture and finally moving the latches in an axial direction which completes the structural connection. With the structural connection completed, the disconnect system is energized making a seal between module and vehicle. The socket portion of the disconnect contains the seals and the probe section on the LTL is float mounted to compensate for misalignments. Since temperatures are basically ambient, the seal system consists of "O" rings equipped with backup rings to prevent "blow out". The seal design includes provisions for easy replacement.

5.4.3 ZERO-G MASS GAUGING. A zero-g mass gauging system will be as important to refill of earth storable vehicles as to POTV and COTV refill. The measurement of propellant mass quantities is critical at two intervals during on-orbit refill; when minimum liquid residuals and when final tanked mass must be measured. Total propellant load must be known to verify that sufficient propellant has been tanked to perform a mission. Propellant quantities in excess of requirements could unnecessarily penalize the vehicle mission. A continuous mass measurement during the latter stages of tank fill would also prevent the protential of tank over-pressure caused by over-fill.

The second occasion where mass measurement will be necessary is during the propellant tank venting procedure described by Table 5-1. Step 5 of this procedure states that a signal will be sent to terminate propellant transfer from one tank to another when the mass gauging device indicates that a minimum liquid residual remains. Screen gallery liquid replenishment (to make-up for surface evaporation) will be provided from this residual liquid volume. Less propellant than this minimum may not be sufficient to maintain communication with the screen device; resulting in vapor penetration due to the loss of liquid replenishment. Substantially more residual than the minimum will increase the probability that propellant can be vented overboard.

6

EXPERIMENTAL MODELING

On-orbit fill and refill of a propulsion tankage system is examined in this section to determine what experimentation is required to demonstrate this capability in a zero or low g environment. Emphasis is placed on identifying the major scaling parameters that must be satisfied in order to model the thermodynamic and fluid mechanic conditions of a refill operation. The influence of fluid properties and model tank scale on the validity of test results was also evaluated. Per the study guidelines, the analysis effort was directed at conducting these experiments in low earth orbit inside the Spacelab.

Modeling or scaling analysis will be used to determine the feasibility, and subsequently the conditions and configurations, of the shuttle experiments for providing data useful in determining procedures for refueling a space-based vehicle. This discussion on scaling will focus on our understanding of the fluid phenomena as well as the complexities involved in experimental modeling.

Based upon the analyses performed in sections 3, 4 and 5, the important areas of cryogenic and earth storable propellant tank on-orbit refill have been identified. These areas, for a cryogenic stage, are tank vent, prechill and fill. The only area of concern for an earth storable stage is tank vent (experimentation is not recommended, as discussed in section 6.). The influence of such factors as helium present within the propellant tanks, and partial acquisition devices was also determined.

Not all of the important elements of a refill process should be subjected to a rigorous experimental program. Rather, only those elements that require verification, or processes which phenomenon is not well understood, should be considered for experimentation. As an example, propellant tanks vent will be an important part of any cryogenic refill operation. There is not a compelling reason, however, for performing such experiments in space. Means can readily be devised for safely venting the propellant tanks without just performing orbital tests. Those areas requiring experimentation are listed below. Specific outputs are identified that will provide design criteria and procedures for refueling operations of space-based OTVs:

1. Tank Prechill - Obtain empirical data to establish relationships for scaling peak tank pressures, prechill times, and vent mass requirements.

2. Tank Fill - Obtain empirical data to establish the influence of propellant tank inflow parameters on thermodynamic equilibrium conditions.

3. Support Experiments Criteria - Identify an empirical relationship between start basket in-flow parameters and successful refill.

Each phase of the propellant-transfer process will require a separate scaling analysis. Three basic scaling methods were considered in this study: (1) dimensional analysis, (2) order of magnitude analysis on the equations of motion, and (3) ratio of phenomenological equations.

Dimensional analysis is the classic method, and it provides a simple and direct procedure for obtaining all parameters that may affect a process. Since exact scaling is not usually possible, however, we must determine which parameters are most important. Dimensional analysis alone does not provide a solution to the problem, and a significant amount of experimentation is required to determine exact relationships between the resulting dimensionless groups. The complex heat and mass exchange mechanism associated with orbital refill did not readily lend itself to dimensional analysis.

The order-of-magnitude and ratio methods require writing the equations that describe the process. With order-of-magnitude analysis, the boundary value problem that describes the process is written and transformed into dimensionless variables. The important variables are determined from an order-of-magnitude analysis and the lower-order terms are neglected in the scaling process. The ratio method is the same approach that is used when an analytical solution is described.

If the differential equation can be solved accurately, the scaling problem is redundant and is replaced by an exact solution. However, it is usually impossible to solve the differential equation with boundary and initial conditions without making some gross simplifying assumptions that may considerably reduce the credibility of the mathematical model.

The ratio method was employed as the scaling technique for the orbital refill process. Empirical equations are identified in Sections 6-2 and 6-3 which describe the prechill and tank fill processes of a cryogenic vehicle refueling operation. The key variables of these processes were readily identified. Model test flow parameters and time scaling relationships were subsequently identified as a function of tank scale.

6.1 RECEIVER TANK SCALE

Normally, when an experimental test program is defined, scaling equations derived from the modeling analyses will serve to identify the tank scale and fluid selection. For this study however, a groundrule to perform tests within the Spacelab facility limited the test tank size to a 1/10th scale maximum. The following relates the specifics of tank size, in addition to tank shape.

6.1.1 RECEIVER TANK SHAPE. The OTV includes liquid hydrogen and liquid oxygen propellant tanks, both of which will be refilled in space. A single receiver tank will be selected for conducting orbital refueling experiments, and the question is, which propellant tank configuration shall be tested?

It was shown in Section 3.3.2.1 (Figure 3-14) that the liquid oxygen propellant tank can be refilled more easily than the liquid hydrogen tank becuase it will not experience excessive pressure during refueling. The conclusion was made, correctly, that liquid hydrogen tank refueling should be subjected to experimentation in order to verify the selected refill technique. However, it will always be more difficult to refill any tank with liquid hydrogen than with liquid oxygen; fluid property differences are responsible for this condition. Consequently, the difficulty in LH_2 tank refill is due to the propellant, not the tank shape. If liquid hydrogen is eliminated as a test fluid, what justification exists for using the LH_2 tank configuration?

It appears that adequate mixing of the liquid and vapor phases will be easier to attain in the LO_2 tank than in the LH_2 tank. This conclusion seems valid if we compare two tanks (having the same volume), where one is cylindrical and has a large length-to-diameter ratio, and the other is a sphere. Intuitively, uniform mixing in a cylinder should be more difficult to achieve than in a sphere. Selection of the LH_2 tank configuration will guarantee that the most difficult configuration for orbital refill will be tested.

6.1.2 TEST SCALE. In general, when an experiment plan is developed it is necessary to compromise between the desire for a full scale test program and the limitations imposed by resources and facilities. Large scale tests are always desirable because the uncertainty of extrapolating test data to a prototype condition can be minimized. Thus, for the Spacelab experiment, the desire is to design the largest tank scale that can be accommodated. In this case, test tank size will be limited to a package that fits within a doublerack structure, Figure 6-1. Design details are provided in the following discussion.

6.1.2.1 <u>Preliminary test tank design.</u> In Figure 6-1, an experimental apparatus is shown positioned in a Spacelab doublerack structure, which will demonstrate on-orbit propellant loading of OTV's. This apparatus consists of a scale model OTV LH_2 tank suspended within a vacuum-jacketed shell. The intent of Figure 6-1 is to determine a basic tank size for the apparatus within the nominal envelopes described by the Spacelab payload accommodation handbook, Reference 6-1. Details for plumbing, wiring and supports are omitted. For this case the areas controlling the tank size are Zones "B" and "C" shown cross hatched on the layout. Zone "B" is reserved for payload cabling and Zone "C" is reserved for subsystem access.

The apparatus is positioned near the right wall of the rack (as viewed by the operator) so that the tank bulkhead clears the corner of Zone "C". Tank length is then limited by Zone "B". Adjustments between diameter and length were made within these confines to arrive at an acceptable L/D ratio. The tank shown has a 457 mm (18 in.)

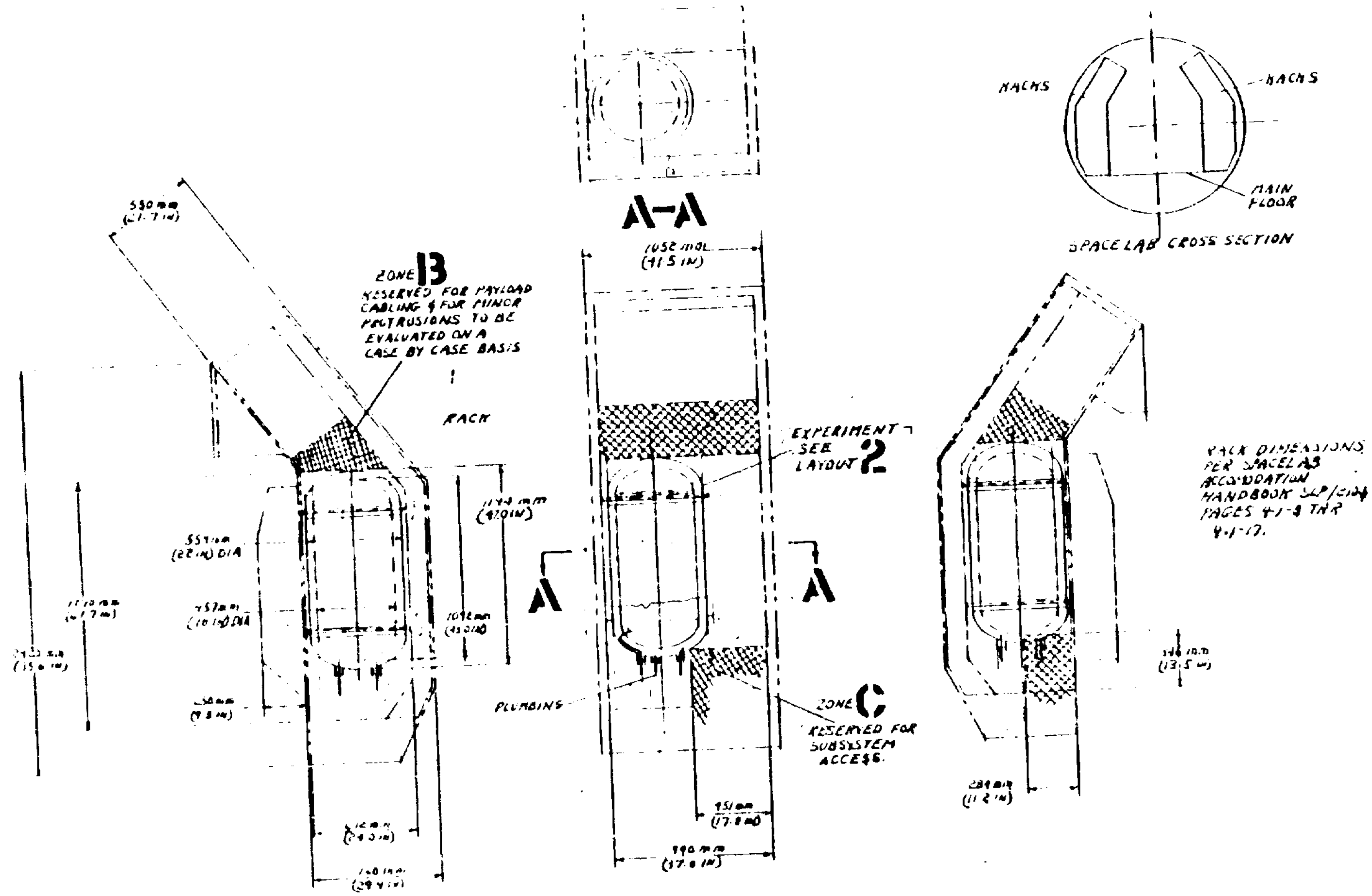


Figure 6-1. Orbital Refill Experiment Installation in Spacelab Standard Double Rack

diameter and a 1092 mm (43 in.) overall length. The tank is equipped with an outer shell having a 559 mm (22 in.) diameter and a 1194 mm (47 in.) overall length. The selected location offers a volume directly below the tank bottom bulkhead which can be used for plumbing and wiring protrusions as shown on the layout. Additional adjustments in tank diameter and length can be made by minor infringements into Zones "B" and "C". For example, it appears that the bottom bulkhead could cut across the corner of Zone "C" without seriously compromising accessibility. For this effort, however, the tank size shown is the maximum within the nominal envelopes.

In Figure 6-2, details are shown for the experiment apparatus using the diameters and lengths established in Figure 6-1. The purpose of Figure 6-2 is to generate a detailed weight breakdown for the tank portion only of the apparatus so that the main drivers can be identified. This weight breakdown in turn was used to determine any revisions to the initial design in an effort to further reduce weight. The outer jacket is included to show the general relationship within the tank and the plumbing.

The tank is a 457.2 mm (18 in.) dia x 762 mm (30 in.) length cylinder equipped with ellipsoidal bulkheads ($a/b = 1.38$) at each end. The material is 2219-T87 aluminum alloy and the minimum gage is 0.51 mm (0.020 in.). The tank is supported from the outer jacket with three pairs of struts at one end and three drag links at the opposite end. Three fill manifolds are installed inside the tank. Provisions for ground fill and drain, vent and electrical, are also included. To permit hardware changes during ground tests, one 132.4 mm (6.0 in.) I. D. access opening is provided at each end. The external surface of the tank is equipped with strip heaters and a multilayer insulation (MLI) blanket.

Referring to the figures, the bulkhead labelled No. 1 has two 1.02 mm (0.040 in.) gage weld zones. The first zone is for the access opening ring and the second zone located at the girth is for the butt weld between the cylinder and bulkhead. This second zone also contains three tangential fittings for the drag link supports (see detail "G"). The bulkhead labelled No. 2 contains a wide weld zone at the girth which contains three pairs of tangential support fittings (see Detail "C"), and the tank wall penetration fittings. This zone also provides the gage increase for the butt weld at the cylinder. A second weld zone near the top of the bulkhead is for the access opening ring. The cylindrical portion of the tank has one weld zone at each end and one running longitudinally. Support lugs for the fill manifolds are welded to the inside surface of one of the end zones.

A typical fill manifold is a length of aluminum alloy tubing equipped with a tee fitting at each end. The side branch of each tee fitting is threaded for attaching spray heads. One of the tee fittings has a protruding rod section (see Detail "J") which engages with the support lug attached to the tank near Bulkhead No. 1. This is a sliding support which provides restraint in any direction normal to the tube while permitting

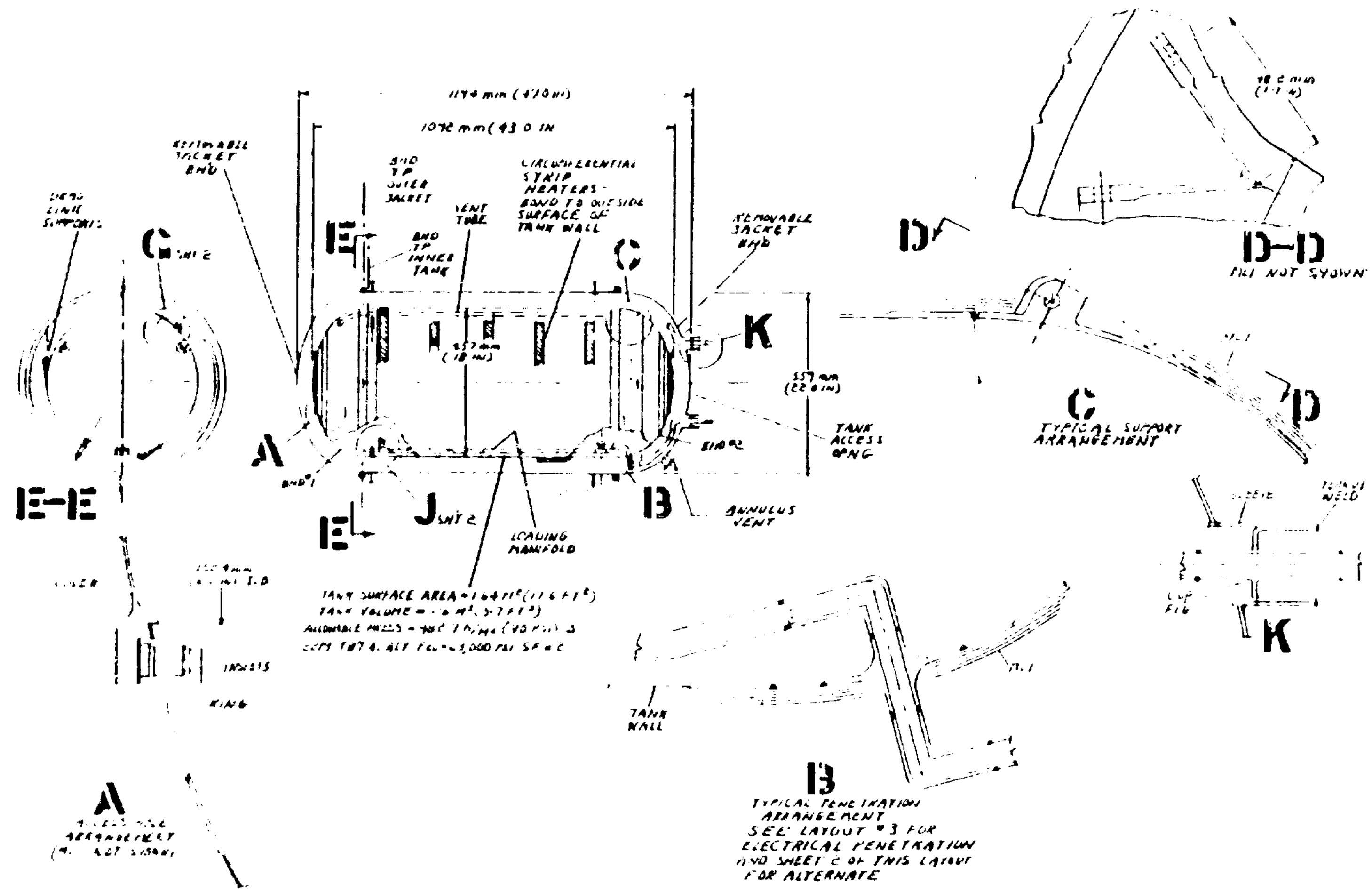


Figure 6-2. Test Tank Design Details

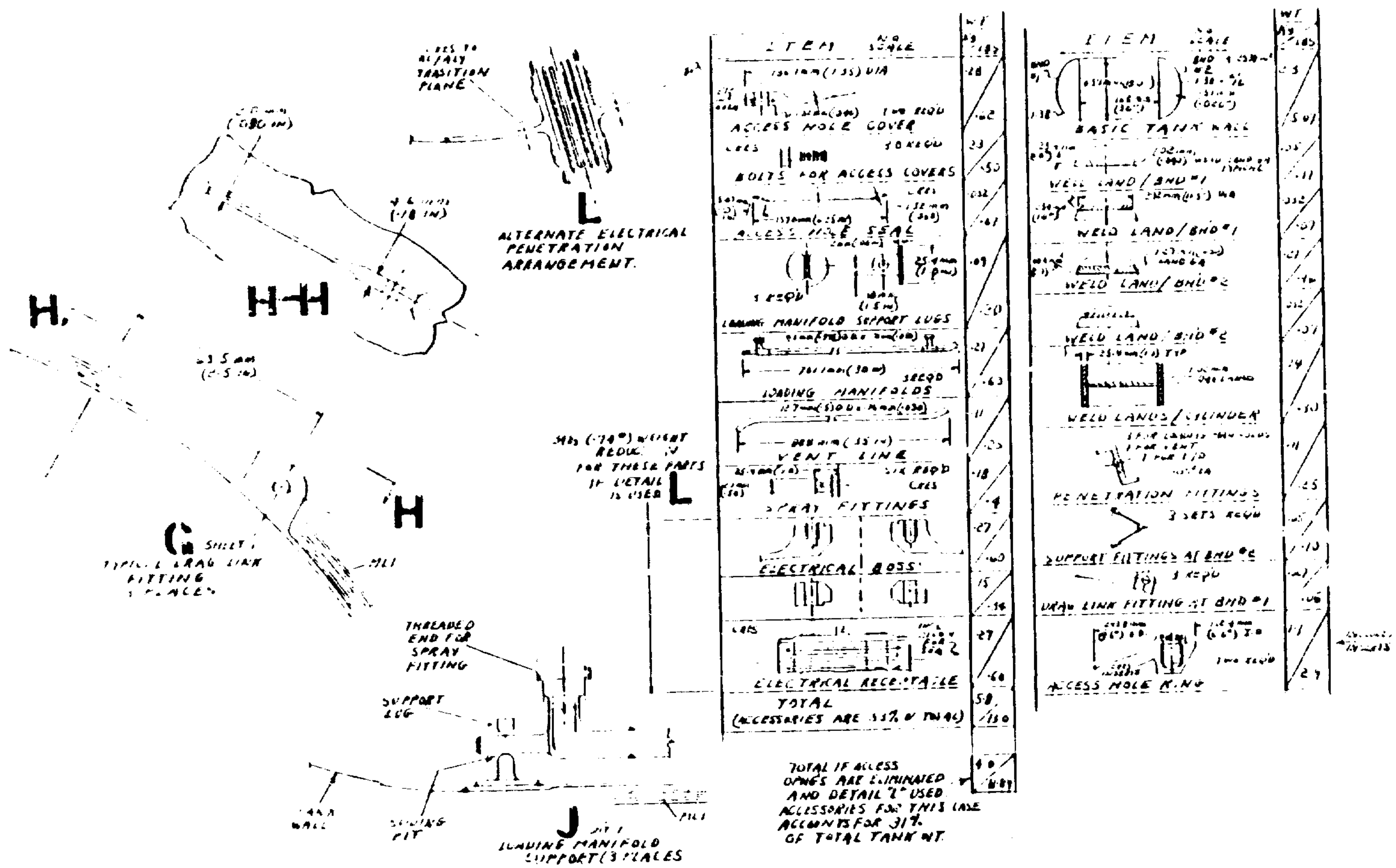


Figure 6-2. Test Tank Design Details (Con't.)

axial movement. The opposite end of the tube (near bulkhead No. 2) is butt welded to the tank wall penetration fitting as shown in Detail "B". The manifold is welded to this penetration fitting prior to installation.

One internal tube extending from bulkhead No. 2 to bulkhead No. 1 is provided for ground venting. Similar to the fill manifolds, this vent tube is welded to a tank wall penetration fitting at bulkhead No. 2. The opposite end of the tube (at bulkhead No. 1) is supported in a manner similar to that described for the fill manifolds.

Ground fill and drain is provided by a penetration fitting located at bulkhead No. 2. This is not shown on the drawing but is similar to that shown for the fill manifolds.

Internal instrumentation is routed through a boss (located through bulkhead No. 2) equipped with a flanged multi-pin connector. A radial seating seal is used for sealing the connector to the boss. An arrangement is shown in detail "B" of Figure 6-3 and an alternate approach is shown in Detail "L". All instrumentation wires inside the tank are bundled into a single cable and supported from the vent tube with clips.

The outside tank surface is equipped with heaters. Several types are available such as coatings directly deposited on the tank wall, blanket types which are fitted with a glove over the tank, and strip or ribbon types which are bonded to the tank wall. Due to versatility and simplicity, the strip type was selected. The heaters are approximately 1.5 in. wide and are arranged in a circumferential pattern along the length of the cylinder. This circumferential pattern may also be extended to the bulkheads, or a longitudinal type pattern can be used similar to gore lines on a bulkhead. All wiring is supported from the tank wall with tape strips which are lapped over the wire and bonded to the tank. The wires are bundled into a single cable at the No. 2 bulkhead and routed through the MLI blanket at a single point.

The entire tank surface, including strip heaters, is covered with a multilayer insulation (MLI) blanket. The blanket is applied in gore sections which interface with two circular cap sections located on the ends of the bulkheads. The sections are attached to the tank wall using "Velcro" tape patches. The pile portions of the tapes are bonded to the tank wall and the hook portions are bonded to the blankets. Cutouts in the blankets provide clearances for the plumbing, wiring and support struts.

The outer jacket is a cylinder equipped with two removable ellipsoidal bulkheads. The material is 2219-T87 aluminum alloy and the wall gage is sized to 207 kN/m^2 (30 psi) external pressure. The cylinder is equipped with two rings (one at each end) which are attached to the tank support struts. These rings also incorporate flanges which interface with the bulkheads and external fittings for attaching to the rack structure. The flanged connections between bulkheads and cylinder use metal "O" rings or radial seating "Cono" seals. All plumbing and wiring circuits penetrate one bulkhead only through cup/sleeve fittings. The cup fittings are welded to the plumbing and the sleeve is welded to the bulkhead. The ends of the cup fittings are sealed to the sleeves with peanut type welds (see detail "K"). When removing the bulkhead, the peanut welds are ground off. When replacing the bulkhead, the cups are resealed with peanut welds. The length of the cup and sleeves are determined for severa bulkhead removables. A vacuum environment is maintained in the annulus volume between tank and jacket during ground test and Spacelab operation. Therefore, the bulkhead is equipped with a flanged penetration fitting for attaching a duct.

Referring to the parts list in Figure 6-2, the total weight for the complete tank assembly is 5.8 kg (13.0 lb). The basic tank shell weighs 2.76 kg (6.08 lb) including the weld lands. The balance is for accessories which accounts for 53 percent of the total weight. Rings and covers for the access openings represent a major portion of the accessory total weight, and these can be eliminated if we are willing to compromise accessibility to the tank interior. For example, access to the tank interior can be accomplished by simply cutting bulkhead No. 1 off at the weld line and rewelding. The width of weld lands would be greater initially than the 25.4 mm (1 in.) shown to permit several cuts, trims and re-welds. This method has risks however, because the procedure amounts to reworking a thin walled aluminum tank with a minimum gage of only 0.55 mm (0.020 in.). Another possible area for weight reduction is the electrical penetration fitting. 0.16 kg (0.34 lb) can be trimmed from this fitting by using the design shown in Detail "L". This alternate design eliminates the separate clamping flange, and integrates the collar (which welds to the tank wall) with the receptacle. The seal is also eliminated. Cooling provisions for the receptacle would be required during welding to prevent damage to the core material which contains the conductor pins. In summary, for both the access holes and the electrical penetration fitting, a 1.93 kg (4.24 lb) weight reduction can be realized which results in a total tank weight of 4 kg (8.84 lb), 31 percent of which is for accessories.

A weight and configuration summary of this tank design is given in Table 6-1. A comparison of test tank to prototype is given in Table 6-2. Note that this tank scale, which is the largest that can be designed into the Spacelab doublerack structure, is 0.108 of the POTV.

Table 6-1. OTV LH_2 Model Test Tank Weight Summary

Layout No.	Tank Volume m^3 (ft^3)	Tank Surface Area m^2 (ft^2)	Conventional Design: Total Tank Weight Inc'l Accessories, kg (lb)	Conventional Design: Accessories % of Total Tank Weight	Non-conventional Design: Total Tank Weight Inc'l Accessories, kg (lb)	Non-conventional Design: Accessories % of Total Tank Weight, kb (lb)
#2	.16 (5.7)	1.64 (17.6)	5.8 (13.0)	53	4.0 (8.84)	31
#3	.93 (32.76)	5.1 (55.0)	10.8 (23.8)	32	9.0 (19.9)	17
#4	3.13 (110.7)	11.5 (123.9)	25.4 (60.2)	15	25.4 (55.9)	9

Table 6-2. Model Test Tank Volume-to-Mass Ratio Comparison With OTV LH_2 Prototype Tank

Tank Scale L*	Tank Diameter cm (in)	Conventional Design V/M m^3/kg (ft^3/lb)	Conventional Design V*/M*(1)	Non-conventional Design V/M m^3/kg (ft^3/lb)	Non-conventional Design V*/M*(1)
.108	45.7(18)	.0278 (.438)	.105	.0410 (.645)	.155
.20	85.9(33.8)	.0874 (1.376)	.331	.1045 (1.646)	.396
.30	128.8(50.7)	.1168 (1.839)	.442	.1257 (1.980)	.476

Prototype Tank (PDTV) Measurements

Volume (V)	=	116 m^3	(4100 ft^3)
Mass (M)	=	447 kg	(986 lb)
Diameter	=	4.29 m	(169 in)
Cylinder length	=	59.5 m	(234.2 in)

(1) (*) refers to ratio of model-to-prototype. An exact (and ideal) scaling of tank properties would result in V*/M* = 1.

6.1.2.2 Larger test tank designs. The inexact scaling of the one-tenth scale model will necessarily create a variance between model and prototype test results. This variance is related to the volume-to-mass ratio differences between tank scales, which is quantitatively evaluated in section 6.2. There was an interest in determining how the volume-to-mass ratio would vary with tank scale. Consequently, a preliminary design was also performed on a two-tenths and a three-tenths scale tank. The two-tenths scale model is basically the same as that described in Figures 6-1 and 6-2. The only differences are minor items such as weld land areas, plumbing lengths, support fitting sizes and quantity of support lugs for the internal plumbing. No outer jacket is shown since this size tank would be transported outside the Spacelab. The tank would probably be suspended from a truss cylinder which in turn interfaces with the shuttle payload support journals.

Referring to the parts list in Figure 6-3, the total tank weight is 10.8 kg (23.8 lb) of which 32 percent is accessories. If the access openings are eliminated and the electrical penetration fitting simplified as described in Layout No. 2, this total tank weight is reduced to 9.0 kg (19.9 kg), of which 17 percent is accessories.

Figure 6-4 is the same as Figure 6-3, except the tank is a three-tenths model. For this case, the basic wall gage was increased from 0.51 mm (0.020 in.) to 0.635 mm (0.025 in.). Minor items such as weld land areas, plumbing lengths, plumbing sizes, and quantity of support fittings have been increased. Results from these two designs are summarized in Tables 6-1 and 6-2.

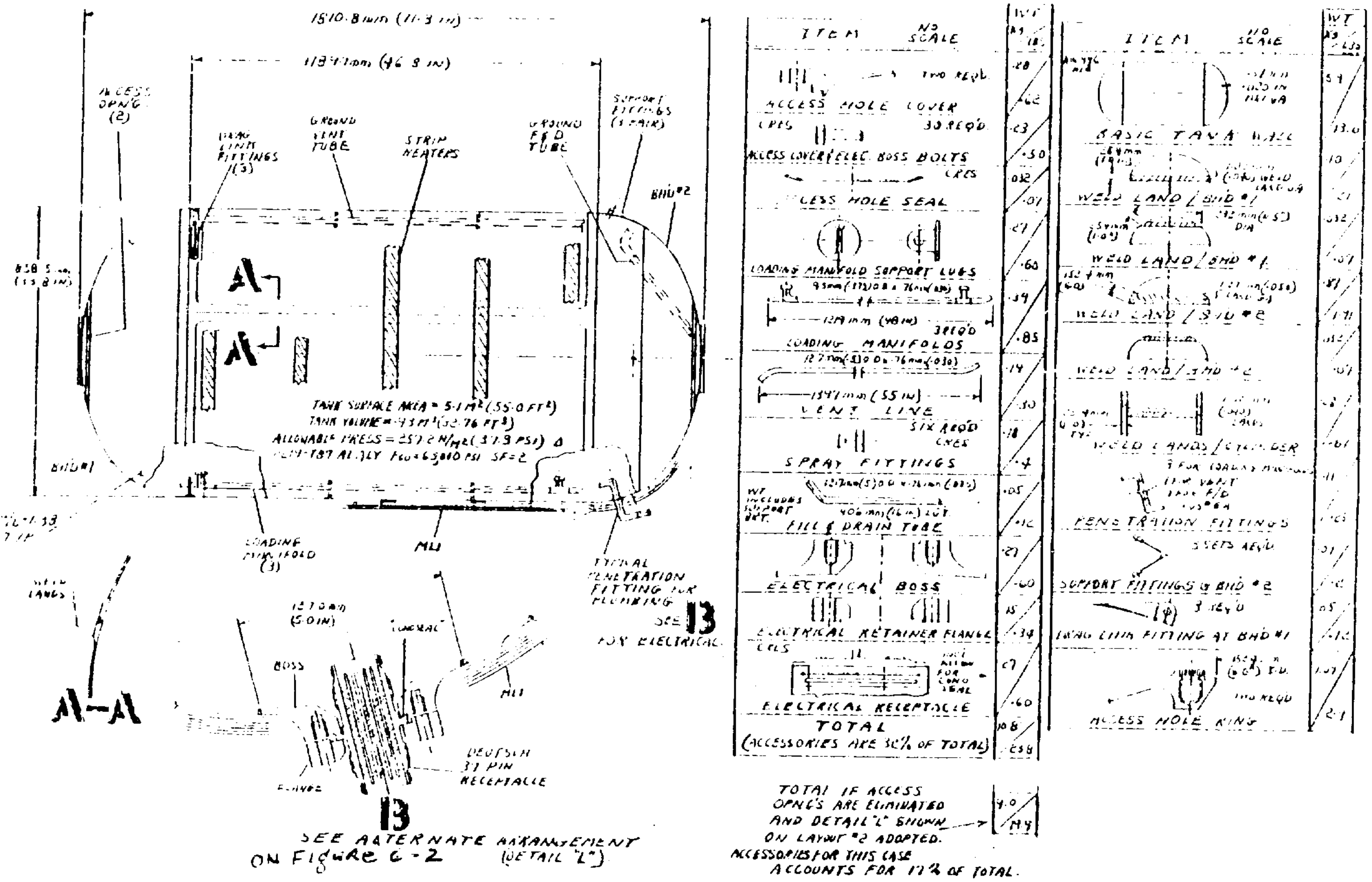


Figure 6-3. Test Tank Design Details (0.2 scale)

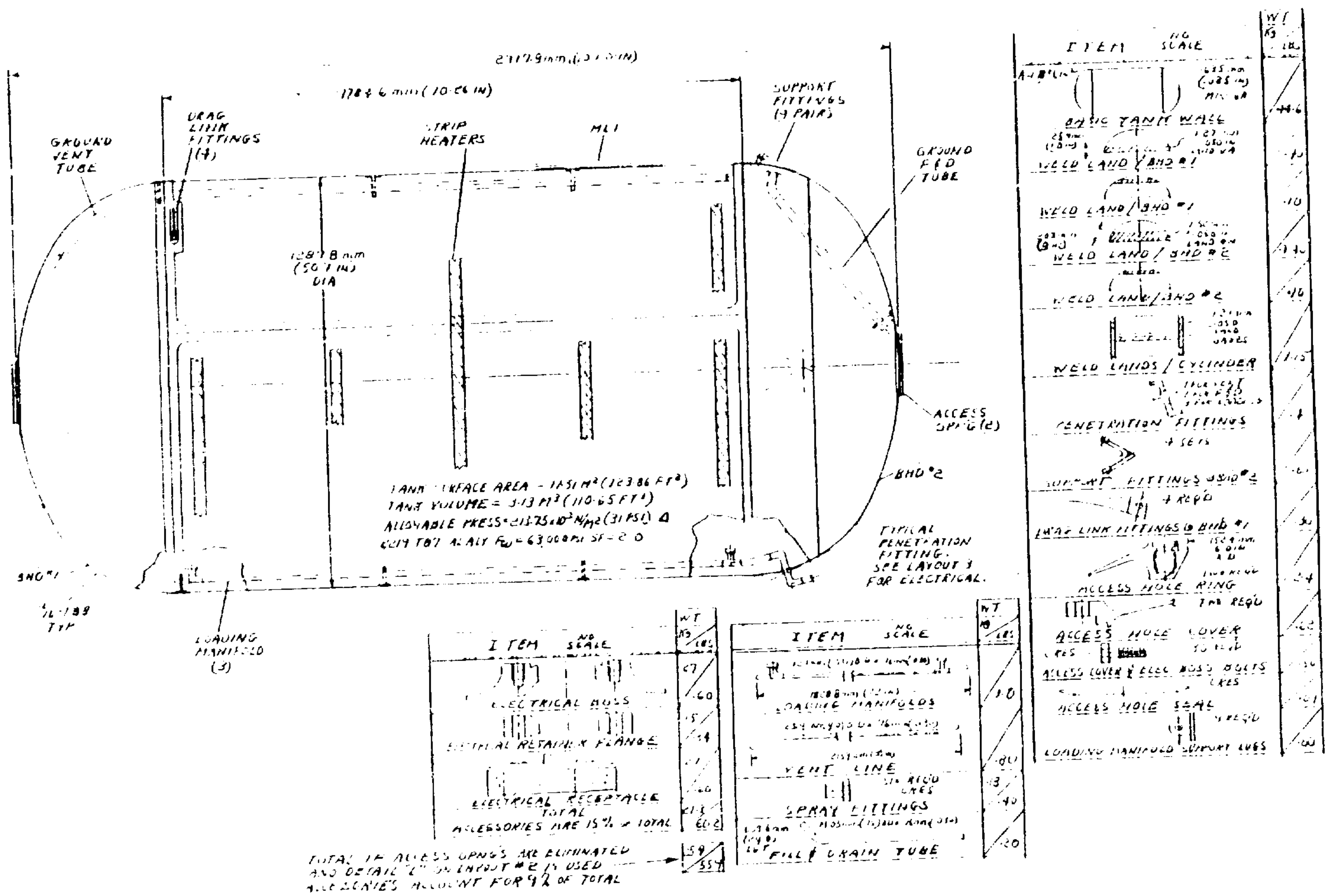


Figure 6-4. Test Tank Design Details (0.3 scale)

6.2 PRECHILL MODELING

The prechill process that has received much attention during this study is illustrated by Figure 6-5. The process will be accomplished by 1) metering liquid at a high velocity into the receiver tank for a fixed duration, 2) allowing heat exchange between fluid and tank walls for an unspecified duration until tank pressure has increased to the vent level, and 3) venting the tanks back down to near zero pressure. The early prechill period is characterized by a complex thermodynamic and fluid mechanic process due to liquid impingement on the hot tank walls. The resulting forced convection nucleate and film boiling phenomena are extremely difficult to analytically model or scale.

Fortunately, it can be shown that this initial period of tank chill may not be significant to the overall process. First, the heat exchange during the limited boiling period represents only about ten percent of the total energy removed during prechill. Second, we should be more concerned with the tank conditions prior to tank vent rather than with the initial transient. The initial transient pressures will be well below the near-steady-state pressures if propellant inflow is adequately controlled. Figure 3-15 indicates that precision metering of LH_2 is not needed to avoid overpressure during prechill. Thus, effort can be concentrated on scaling steady-state conditions of the prechill process.

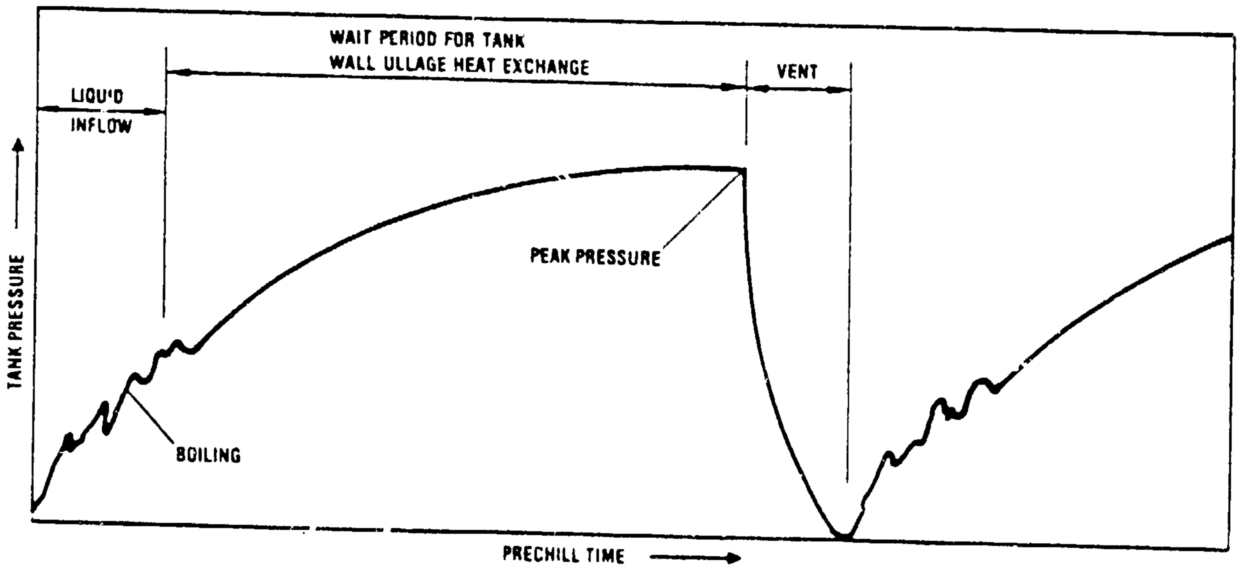


Figure 6-5. A Prechill Procedure Can be Identified to Eliminate Excessive Tank Pressures Due to Wall Boiling

6.2.1 SCALING PEAK PRESSURES. This is an important requirement because excessive pressures must be avoided during POTV on-orbit prechill. Peak pressure will occur as tank and fluid temperatures reach equilibrium. Figure 6-6 shows the theoretical maximum pressure for a POTV liquid hydrogen tank as a function of tank thermo-physical properties and geometry. This figure also shows that tank volume-to-mass ratio (V/M) is an important variable.

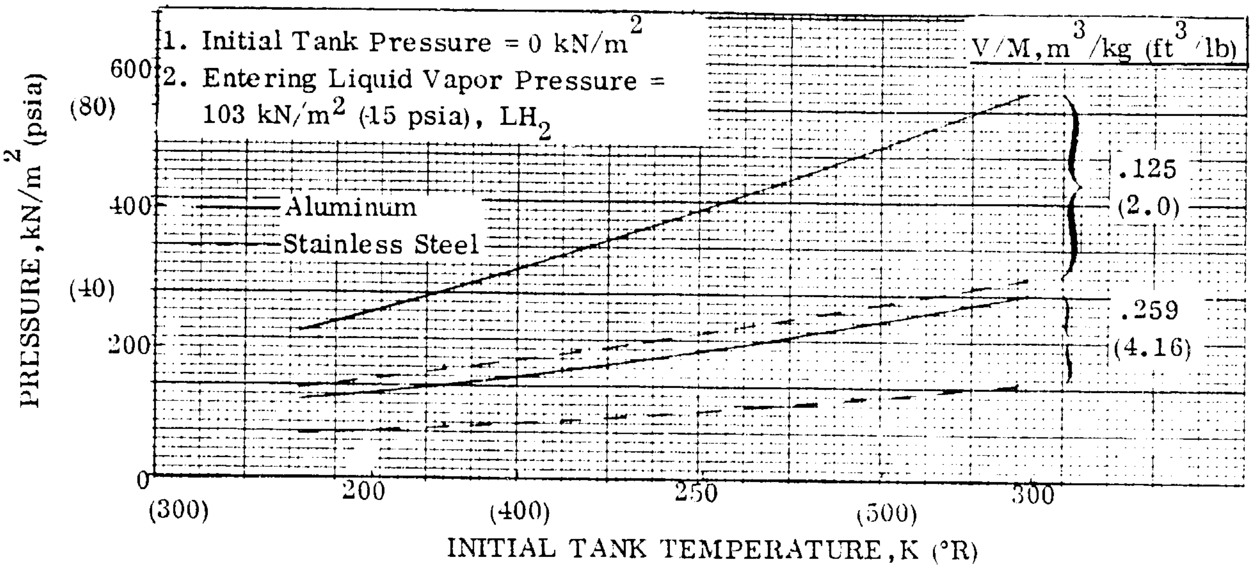


Figure 6-6. Maximum Pressure During OTV Propellant Tank Prechill

Figure 6-7 suggests that the pressure scaling parameter is PV/M. These curves have been generated for a V/M range of .125 m^3/kg (2 ft^3/lb) to .259 m^3/kg (4.16 ft^3/lb) (V/M for OTV is .259 (4.16)). This figure shows that PV/M is a function of initial tank temperature, tank material, and propellant. By selecting these variables to be the same for a model test as for the prototype (full scale) vehicle, the resulting PV/M will also be the same. Expressed mathematically, $(PV/M)_m = (PV/M)_p$, or

$$P^*V^*/M^* = 1 \tag{6-1}$$

where:

	P	=	peak tank pressure during prechill
	V	=	tank volume
	M	=	tank mass
subscript,	m	=	model
	p	=	prototype
superscript, (*)		=	the ratio of model to prototype.

When model test variables are selected such that $P^*V^*/M^* = 1$,

then $\qquad P^* = P_m/P_p = 1 \qquad (6-2)$

Equation 6-2 means that prototype tank peak pressures will be equal to model test tank peak pressures. There are other boundary conditions (discussed later) to be satisfied for the above statement to hold true.

If test constraints are such that $V^*/M^* \neq 1$, then

$$P^* = M^*/V^*, \text{ or } P_p = P_m (V^*/M^*) \qquad (6-3)$$

In this instance, P_p is determined by multiplying the observed P_m by (V^*/M^*).

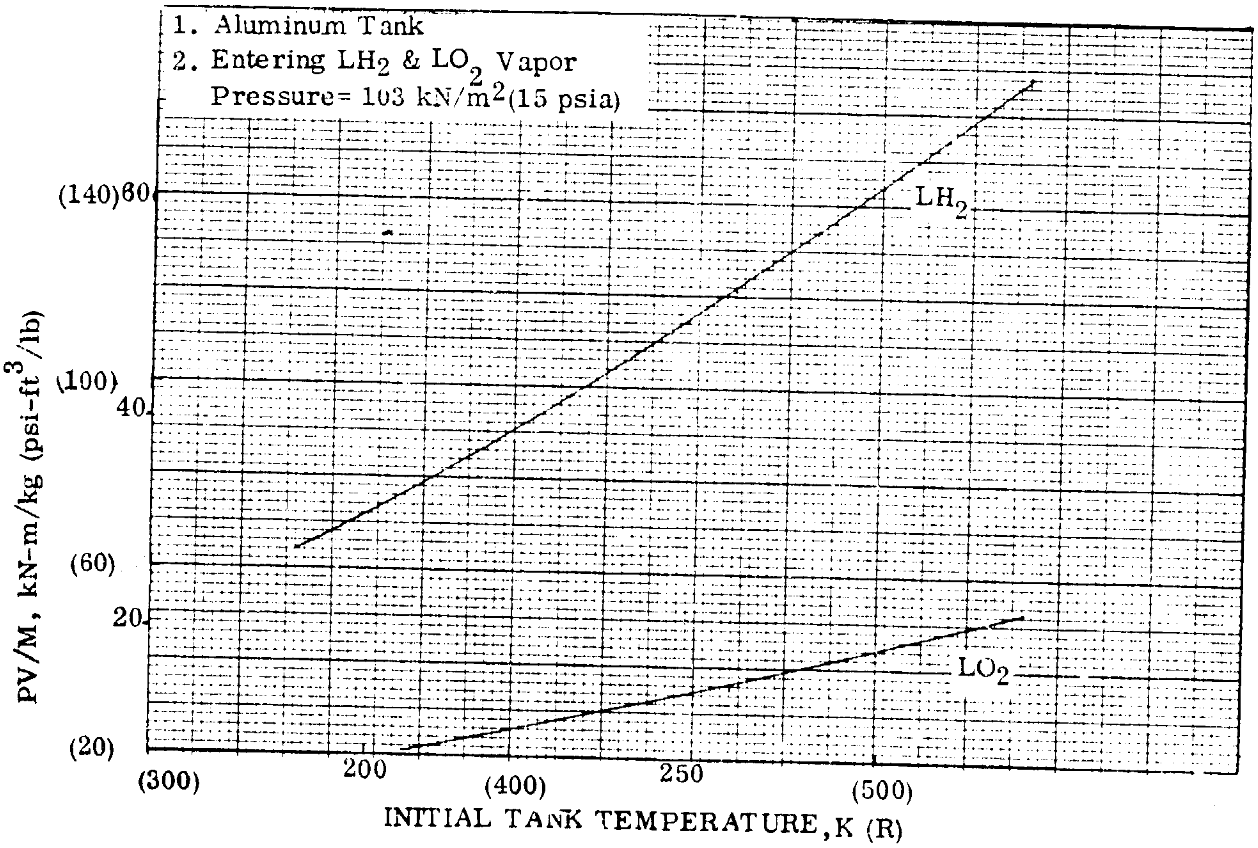


Figure 6-7. PV/M is a Parameter for Scaling Peak Pressures During Prechill

6.2.1.1 <u>Model Tank Size Influence.</u> Test tank scale will have a major affect upon the peak pressures experienced during experimentation, because of the V^*/M^* influence. Figure 6-8 shows the relationship V^*/M^* and tank scale (L^*), where L is a characteristic tank dimension. Figure 6-9 shows the influence of scale upon model tank pressures when modeling of POTV peak pressures is attempted. Referring to section 3.3.2.5 (Figure 3-19), a POTV peak pressure of about 69 kN/m^2 (10 psia) was selected as being acceptable for a prechill procedure. The .108 scale model (for Spacelab) would experience a pressure of nearly 690 kN/m^2 (100 psia) under similar conditions. Thus, it is seen that prechill experiments conducted on Spacelab would produce results substantially different from what would be predicted for a prototype vehicle.

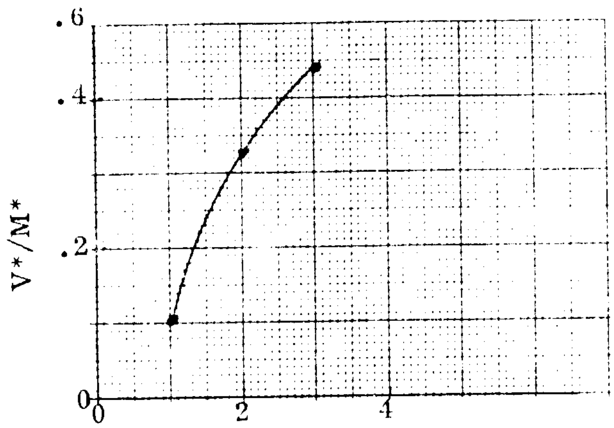


Figure 6-8. Test Tank Scale Influence on V*/M*

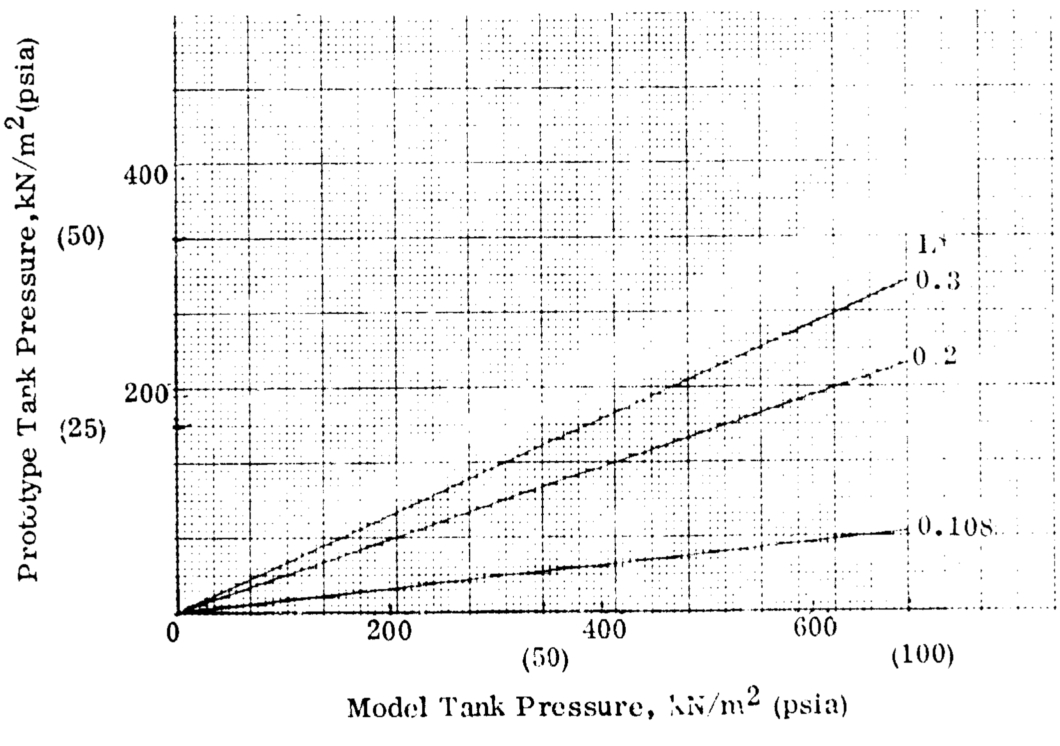


Figure 6-9. Peak Prechill Pressures May Be Excessive for Small Scale Experiments

6.2.2 TIME SCALING. A second major requirement of the prechill experiment will be to develop a time scaling parameter applicable to the tank charging process. This is an important objective because we need to establish propellant transfer timelines. A potentially useful parameter may be developed by assuming the receiver-tank charging process to be equivalent to the transient heat conduction process of a lumped-mass system. Consider the prechill condition described by Figure 6-10:

1. LH_2 flow to OTV is initiated at zero time.

2. A convective heat transfer coefficient exists at the end of the flow period as a result of inflow conditions.

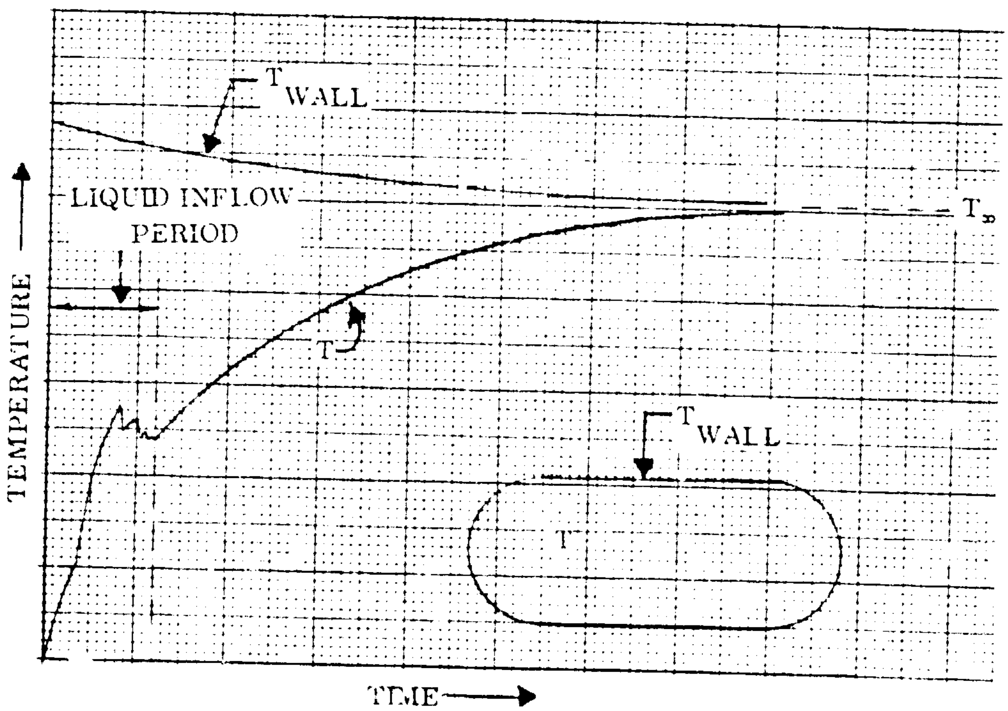


Figure 6-10. $(hA/m\ C_p)t$ is Applicable as Time Scale Parameter for Prechill Process

3. The temperature-time history as indicated by Figure 6-10 can be characterized by

$$\frac{T - T_\infty}{T_o - T_\infty} = e^{-(hA/mCp)\,\tau} \tag{6-4}$$

where

T	=	Fluid temperature at time, t
T_o	=	Initial fluid temperature
T_∞	=	Equilibrium temperature of tank and fluid
h	=	Fluid/tank-wall heat transfer coefficient
A	=	Tank wall surface area
m	=	Fluid mass
Cp	=	Fluid heat capacity
τ	=	Time
θ	=	$(hA/mCp)\,\tau$, dimensionless time

It is obvious from equation 6-4 that the dimensionless temperature parameters will be identical if $\theta^* = 1$. Furthermore, the actual temperature changes will be identical if we impose the additional requirement that

$$m^*/M^* = 1 \tag{6-5}$$

where

m	=	fluid mass
M	=	propellant tank mass

This requirement is derived from the expression for tank-vapor energy exchange that occurs during prechill. The following expression describes energy exchange between tank and vapor.

$$m\,C_v\,\Delta T_v = M C_w\,\Delta T_w \tag{6-6}$$

where

ΔT_v	=	fluid temperature change
ΔT_w	=	tank wall temperature change
C	=	heat capacity

subscript

w	=	tank wall
v	=	fluid vapor

Taking the ratio of model to prototype gives

$$m^* \Delta T_v^* = M^* \Delta T_w^* \tag{6-7}$$

or

$$m^*/M^* = \Delta T_w^* / \Delta T_v^* = 1 \tag{6-8}$$

By imposing a) the requirement of 6-8 and b) the same initial tank temperature for model and prototype, we are assured that tank temperature changes will be identical.

All variables of the dimensionless time parameter, with the exception of h and A, can be described. However, Equation 3-11, shown in Filling of Orbital Fluid Management Systems, NASA CR-159404, relates h to fluid properties and tank inflow conditions. This relationship is given as:

$$\frac{h}{\rho C_p} (N_{PR})^{2/3} = C_1 \left[\frac{(\dot{m}v^2/V)\mu}{\rho^2} \right]^{1/4} \tag{6-9}$$

where:

ρ	=	Fluid density
C_p	=	Constant-pressure heat capacity
N_{PR}	=	Prandtl number
$\dot{m}$	=	Entering mass flow rate
v	=	Entering fluid velocity
V	=	Tank volume
μ	=	Fluid viscosity
C_1	=	Empirical coefficients
$\frac{\dot{m}v^2}{V}$	=	Fluid inflow parameter

It will be now possible to identify the relationship required to satisfy $\theta^* = 1$. This development is given below for the case where tank material are the same for both model and prototype conditions:

Normalizing the dimensionless time parameter, we have

$$\theta^* = h^* A^* \tau^*/m^* \qquad (Cp^* = 1) \tag{6-10}$$

From (6-9) we have

$$h^* = (\dot{m}^* v^{*2}/V^*)^{1/4} (\rho^*)^{1/2} \tag{6-11}$$

But $\rho^* = m^*/V^* = \left(\frac{m^*}{M^*}\right)\left(\frac{M^*}{V^*}\right) = \frac{M^*}{V^*}$ since $\frac{m^*}{M^*} = 1$, from (6-5)

Thus, (6-11) becomes

$$h^* = (\dot{m}^* v^{*2}/V^*)^{1/4} (M^*/V^*)^{1/2} \tag{6-12}$$

Combining 6-10 and 6-12 results in

$$\theta^* = (\dot{m}^* v^{*2}/V^*)^{1/4} (M^*/V^*)^{1/2} (A^* \tau^*/m^*) \tag{6-13}$$

$$\dot{m}^* = m^*/\tau^* \tag{6-14}$$

Combining 6-13 and 6-14 gives

$$\theta^* = (\dot{m}^* v^{*2}/V^*)^{1/4} (M^*/V^*)^{1/2} (A^*/\dot{m}^*) \tag{6-15}$$

now, $$A^* = L^{*2} \tag{6-16}$$

Also the fluid inflow parameter can be modified as follows,

$$\dot{m}^* v^{*2}/V^* = \dot{m}^{*3}/(A_n^{*2} V^*) = \dot{m}^{*3}/L^{*7} \tag{6-17}$$

where:

A_n = nozzle flow area

Combining (6-15), (6-16), and (6-17) gives

$$\theta^* = (\dot{m}^{*3}/L^{*7})^{1/4} (M^*/V^*)^{1/2} \quad (L^{*2}/\dot{m}^*)$$

$$= \left(\frac{L^*}{\dot{m}^*}\right)^{1/4} \quad (M^*/V^*)^{1/2} \tag{6-18}$$

Finally, in order to have similarity between model and prototype dimensionless temperature parameters we require that $\theta^* = 1$, which will be satisfied if

$$(L^*/\dot{m}^*)^{1/4} (M^*/V^*)^{1/2} = 1 \tag{6-19}$$

An experiment model test package design will be heavily influenced by such variables as tank size, test duration, flowrate and velocity requirements, and the system pressures needed to provide these flowrates. Flowrate, $\dot{m}^*$, can be obtained from (6-19).

$$\left(\frac{\dot{m}^*}{L^*}\right)^{1/4} = (M^*/V^*)^{1/2} \text{ or, } \dot{m}^* = L^* (M^*/V^*)^2 \tag{6-20}$$

Velocity, v^*, can be determined from (6-20),

$$\dot{m}^* = \rho^* v^* A_n^* = v^* A_n^* \; (\rho^* = 1). \text{ Therefore,}$$

$$v^* = \frac{L^*}{A_n^*} (M^*/V^*)^2 = \frac{(M^*/V^*)^2}{L^*} \tag{6-21}$$

Time, τ^*, is determined by combining (6-14) and (6-20),

$$\tau^* = \frac{m^*}{\dot{m}^*} = \frac{m^*}{L^* (M^*/V^*)^2} = \left(\frac{m^*}{M^*}\right)\left(\frac{V^*}{L^*}\right) \frac{1}{(M^*/V^*)} = \frac{L^{*2}}{(M^*/V^*)} \tag{6-22}$$

The three flow test variables of equations (6-20) through (6-22) have been determined for the actual model tank configurations of Table 6-2, and are given in Table 6-3. Note that a considerable variation exists in the flow parameters selected for the actual and "ideal" Spacelab experiment test tank. Unquestionably, the flowrates and velocities indicated by Table 6-3 cannot be attained for the actual tank model. Consequently POTV prechill cannot be exactly simulated with 0.108 scale model Spacelab tests, even if liquid hydrogen is used.

Table 6-3. Model tank scale influence upon test variables.

Model Test Variables	Model Tank Scale, L^*			
	0.108(1)	0.2	0.3	0.108(2)
$\dot{m}^*$	9.80	1.825	1.536	.108
v^*	839.8	45.6	17.1	9.26
τ^*	1.28×10^{-4}	.013	.040	.0117

(1) Tank model for Spacelab experiment.

(2) Exactly scaled model for Spacelab experiment ($V^*/M^* = 1$).

6.2.3 FLUID SUBSTITUTE. Liquid hydrogen cannot be used within the Spacelab under any condition. Liquid nitrogen is the only cryogenic alternative that may be acceptable. The liquid nitrogen quantities allowed within the Spacelab will be determined by a payload safety review group which convenes to evaluate experiments planned for the Spacelab. Any experiment which requires large quantities of liquid nitrogen would be carefully reviewed to assure that inadvertent spillage would not create a hazardous environment. It was necessary to assume for this study that an experiment test package could be designed to circumvent potential problems, since a detailed design effort was beyond the scope of this effort.

Before pursuing the influence of LN_2 upon experiment modelling, a point will be made about why non-cryogens may be unsuitable for this experiment. It is believed that tank pressure during much of the fill process will be heavily influenced by heat and mass exchange between the liquid and entrained vapor. Heat and mass exchange is directly proportional to liquid-vapor surface area which, in turn, is dependent upon whether fluid agitation has created individual bubbles or a froth. Now, it has been observed that a frothy condition can readily be created when a non-cryogen, such as Fréon, is mixed with a vapor. Conversely, it has also been observed that LH_2 and LN_2 do not create a froth under similar conditions of agitation. This difference in behavior which may be difficult to quantify, coupled with the complex nature of the tank fill process has led to the conclusion that LN_2 is the only viable fluid substitute for the orbital experiment.

Prechill. The scaling effects of LN_2 upon the prechill process can be determined by employing the following relationships

$$\theta^* = (h^* A^* \tau^*)/(m^* C_p^*) \text{ (from Equation 6-10)} \quad (6\text{-}23)$$

$$h^* = (\dot{m}^* v^{*2}/V^*)^{0.25} \rho^{*0.5} C_p^* \mu^{*0.25}/Pr^{*2/3} \text{ (from Equation 6-9)} \quad (6\text{-}24)$$

where

C_p = fluid heat capacity

ρ = vapor density

μ = vapor viscosity

Pr = Prandtl number

superscript, (*) = the ratio of model to prototype.

In this discussion, model refers to tests conducted with LN_2 and prototype refers to LH_2 tests. Also, for convenience, it is assumed that the model and prototype scale are the same. That is, $L^* = A^* = V^* = 1.0$ where L = characteristic tank dimension and A = tank surface area.

Now, the following fluid property ratios apply for nitrogen and hydrogen:

$$\mu^* = 4.0$$

$$C_p^* = .088$$

$$Pr^* = 1$$

$$\rho_L^* = 11.4$$

Since we are at liberty to select any value for ρ^*, it was decided to select that value which results when model test and prototype peak prechill pressures are the same. This constraint was found to result in $\rho^* = 6.2$, from computer simulations. This is also the same value for m^* since its relationship to density is

$$\rho^* = m^*/V^* = m^* \text{ (since } V^* = 1) \tag{6-25}$$

By working with Equations (6-23) and (6-24) and applying the hydrogen-nitrogen property ratios, it now is possible to compute the influence of nitrogen upon the model test parameters. First, Equation (6-24) combined with Equation (6-25) can be simplified to the following expression when substitutions are made for μ^*, Pr^* and V^*,

$$h^* = 1.41\ (\dot{m}^* v^{*2})^{0.25}\ m^{*0.5}\ C_p^* \tag{6-26}$$

Combining Equations (6-23) and (6-26) will result in

$$\theta^* = 1.41\ (\dot{m}^* v^{*2})^{0.25}\ \tau^*/m^{*0.5} \tag{6-27}$$

and substituting $m^* = 6.2$ gives

$$\theta^* = 0.57\ (\dot{m}^* v^{*2})^{0.25}\ \tau^* \tag{6-28}$$

From the continuity equation we have

$$\dot{m}^* = \rho_L^* A^* v^* = \rho_L^* v^* = 11.4 v^* \tag{6-29}$$

or

$$v^* = 0.088\ \dot{m}^* \tag{6-30}$$

Substituting (6-30) into (6-28) gives

$$\theta^* = 0.17\ \tau^* \dot{m}^{*0.75} \tag{6-31}$$

Also

$$\dot{m}^* = m^*/\tau^* = 6.2/\tau^* \tag{6-32}$$

which when substituted into (6-31) gives

$$\theta^* = 0.67\,\tau^{*0.25} \tag{6-33}$$

Finally, setting $\theta^* = 1$ we have

$$\tau^* = 4.96 \tag{6-34}$$

Substituting (6-34) into (6-32), and (6-32) into (6-30) gives

$$\dot{m}^* = 1.25 \tag{6-35}$$

$$v^* = 0.11 \tag{6-36}$$

Equations (6-34), (6-35), and (6-36) give the time, flowrate, and velocity ratios that must be applied if nitrogen rather than hydrogen is employed as a test fluid for prechill tests. These ratios can be used as multipliers for the respective ratios of Table 6-3 to obtain the combined influence of fluid and tank scale (for the stated tank scale assumptions). It is concluded that substituting LN_2 for LH_2 should not compromise prechill test results. In fact, there appears to be an advantage in using LN_2 because model test velocities will be substantially lower with this fluid, which may make it possible to achieve some of the conditions for a small scale test tank.

The primary disadvantage with using LN_2 as a test fluid is that there are fewer potential problems with prechilling a propellant tank with LN_2 than with liquid hydrogen. LN_2's thermo-physical properties are very similar to those of liquid oxygen, which was rejected as the preferred test fluid. The arguments against LO_2 as a test fluid also apply to LN_2.

6.2.4 PREDICTED PRECHILL TEST VARIATIONS FROM THE IDEAL. An exact prechill simulation cannot be achieved using liquid hydrogen and the 0.108 scale model hydrogen tank. It would be useful, however, to quantify the deviation from the ideal model test condition. To this end, the HYPRES computer program was employed to predict propellant tank temperature and pressure histories for several model test conditions. Program results are shown in Figures 6-11 and 6-12 for the assumed flow condition of saturated hydrogen vapor entering the propellant tank.

6.2.4.1 Zero-g test environment limitations. Figure 6-11 gives predicted propellant tank temperature versus time from prechill initiation for several test tank configurations and flow conditions. The abscissa represents the product of model test time and the time ratio obtained from Equation (6-22). Case 1 data represents an exact simulation

NOTES:

1. Curves are "HYPRES" computer runs simulating prechill tests conducted with a 0.108 scale LH_2 tank.

2. Prechill tests simulate the full scale conditions given in Table 6-4.

3. Predicted full scale tank prechill duration is "model test" time divided by the time scale factor of Table 6-4.

4. Case (1), (2) and (3) conditions are identified in Table 6-4.

5. Case (1) results exactly simulate the full scale prechill process.

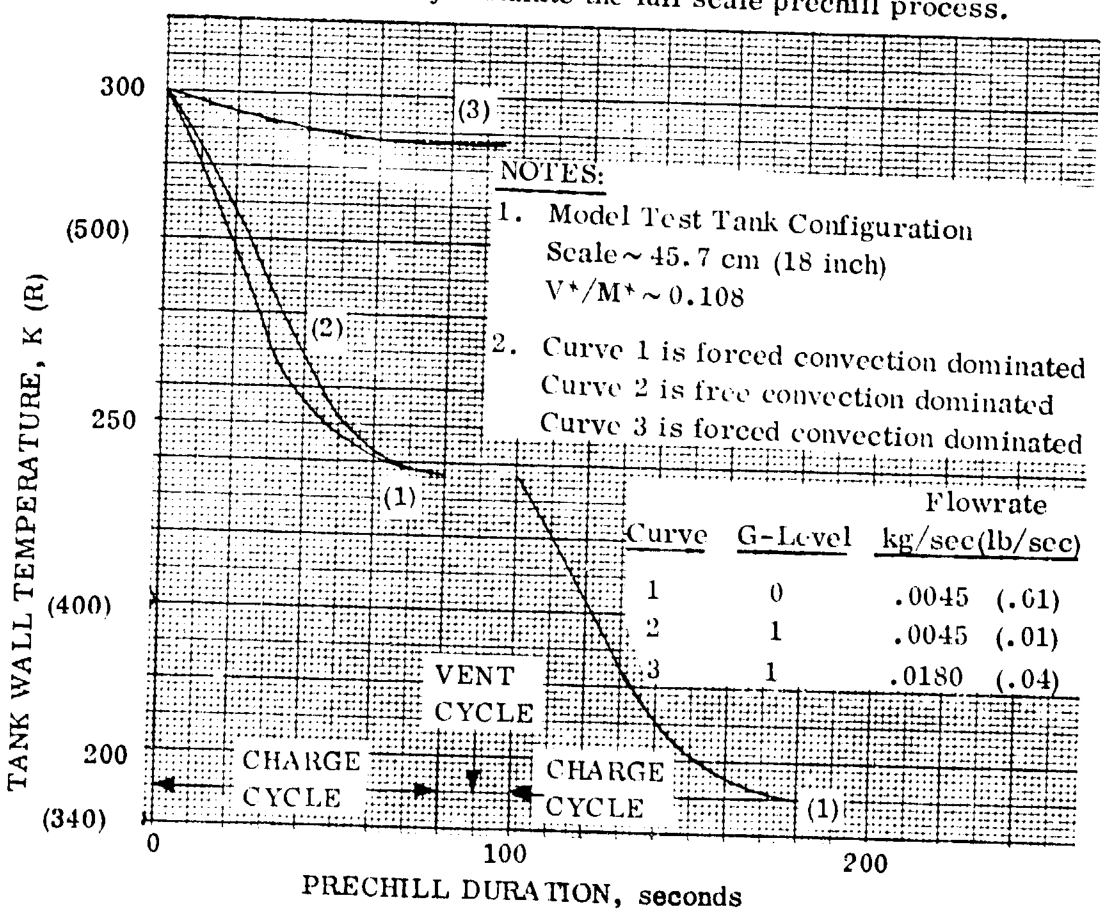


Figure 6-11. Predicted LH_2 OTV Tank Temperature Histories From "HYPRES" Computer Simulation of 0.108 Scale Model Prechill Tests

where $V^*/M^* = 1$. These results are identical to those obtained for the full scale tank conditions of Table 3. Case 2 shows how tank conditions would be altered if substantially lower model test flowrate and velocity conditions were imposed. This deviation is small compared to deviations which results when V^*/M^* is not matched (although velocity and flowrate are matched), Case 3.

Figure 6-12 gives predicted pressure histories for the same test tank and flow conditions identified in Figure 6-11. Again, only the first charge and vent cycle is shown for the non-ideal model test case. Also, as before, Case 1 results were identical to those of the full scale tank. Similarly, Case 2 and Case 3 results were patterned after those of Figure 1. That is, an order of magnitude change in incoming flowrate had a minor influence on results, whereas an order of magnitude change in V/M had a major influence.

6.2.4.2 One-g test environment limitations. Ground-based tests for determining the validity of the prechill process will be applicable only if a) the normal gravity free convection heat transfer process is dominated by the forced convection mechanism created by the entering propellant, and b) the fluid flow mechanism is the same for model tests as for the prototype configuration. To satisfy the former condition we must verify that the heat transfer coefficient given by equation (6-9) exceeds that free convection coefficient created by a normal gravity environment. Equation (6-9) is applicable to heat exchange between a propellant tank surface and its contained vapor.

Prechill Process. The primary concern with normal gravity prechill tests is that inflow conditions required for similarity on the basis of tank scale (Table 6-4) will not be sufficient to guarantee the dominance of forced convection heat transfer. This point is illustrated by Figure 6-13 which shows how prechill test tank pressures will be affected a) by the influence of a normal gravity environment, and b) by the increased inflow conditions necessary to provide a forced convection dominated environment. Curve 1 is the predicted 45.7 cm (18 inch) diameter test tank pressure at the indicated flow conditions in a zero gravity environment. Obviously, the heat exchange mechanism is forced convection dominated. Curve 2 shows the predicted tank pressure if the same test is performed in a normal gravity environment. The tank wall-ullage heat exchange mechanism is free convection dominated. However, a forced convection dominated environment is necessary for a valid model test. This will require a factor of four increase in entering flowrate and velocity. The outcome is that a greater difference results between the zero-g and one-g test tank pressure profiles, Curves 3.

It is questionable that results of one-g tests that meet the criterion for forced convection heat transfer dominance can be employed to scale the zero-g prechill process.

6.2.5 PRECHILL SUMMARY. It is concluded from the computer simulations that the 45.7 cm (18 in.) diameter test tank cannot be employed to obtain results that are directly extrapolated to a prototype OTV vehicle. This is so even if tests are conducted with LH_2 in a near zero-g environment. Direct extrapolation becomes even less likely

NOTES:

1. Curves are "HYPRES" computer runs simulating prech 'l tests conducted with a 0.108 scale LH_2 tank.
2. Prechill tests simulated the full scale conditions given in Table 6-4.
3. Predicted full scale tank prechill duration is "model test" time divided by Table 3 time scale factor.
4. Case (1), (2) and (3) conditions are identified in Table 6-4.
5. Case (1) results exactly simulate the full scale prechill process.

1. Model Test Tank Configuration
   Scale ~ 45.7 cm (18 inch)
   V*/M* ~ 0.108
2. Curve 1 is forced convection dominated
   Curve 2 is free convection dominated
   Curve 3 is forced convection dominated

Curve	G-Level	Flowrate kg/sec (lb-sec)
1	0	.0045 (.01)
2	1	.0045 (.01)
3	1	.0180 (.04)

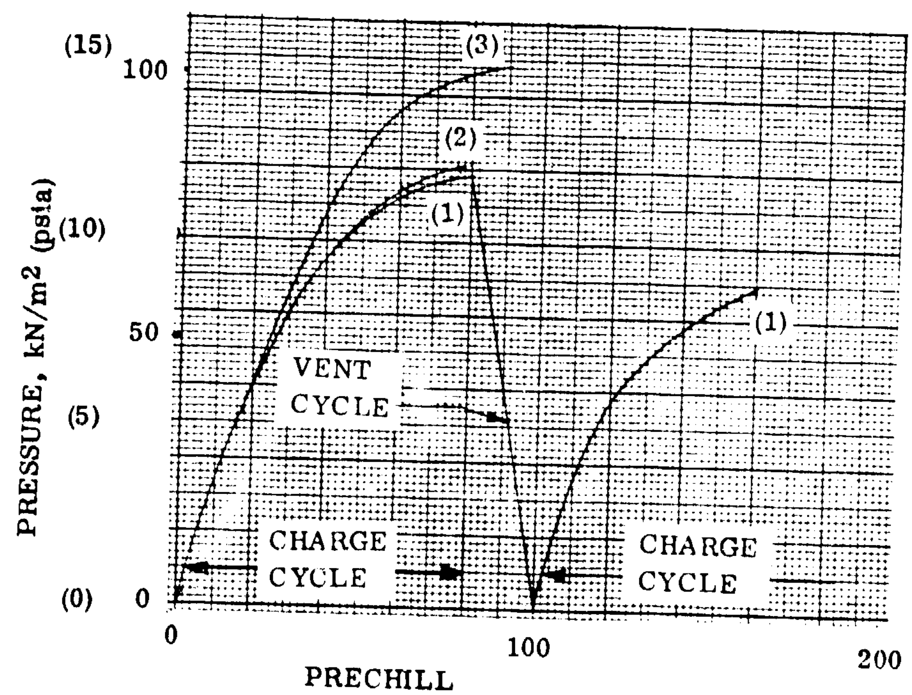


Figure 6-12. Predicted LH_2 OTV Tank Pressure Histories from "HYPRES" Computer Simulation of 0.108 Scale Model Prechill Tests

Table 6-4. Scale model test variables for simulating OTV LH_2 tank prechill.

	Incoming Flow Rate, kg/sec (lb/sec)	Incoming Velocity m/sec (ft/sec)	Tank Scale	V*/M*	Time Scale
Prototype Condition	0.71 (1.56)	6.7 (22)	1.0	1.0	1.0
Case (1)	0.045 (0.100)	62.2 (204)	0.108	1.0	0.0117
Case (2)	0.0045 (0.0100)	6.20 (20.4)	0.108	1.0	0.0021
Case (3)	0.045 (0.100)	62.2 (204)	0.108	0.105	1.28×10^{-4}

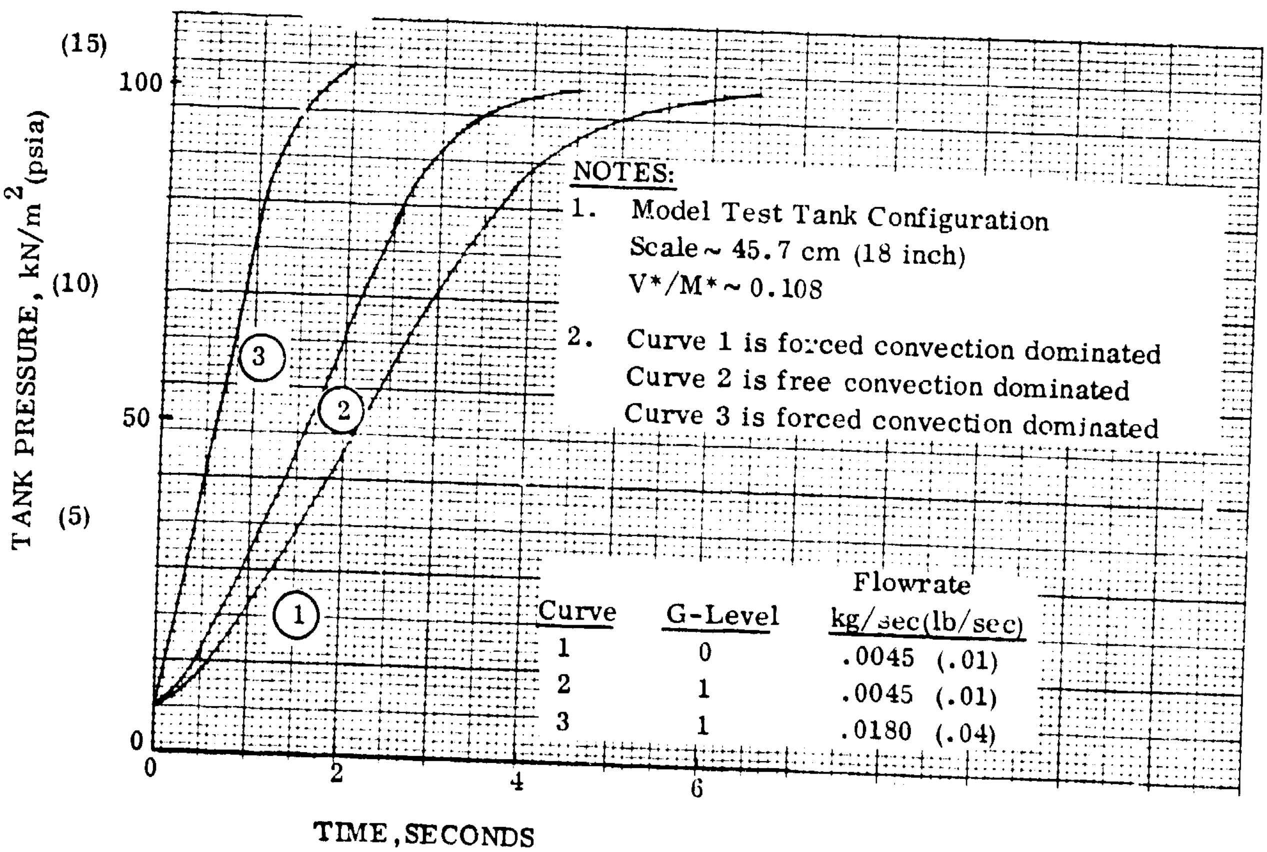


Figure 6-13. A normal gravity environment will influence OTV model tank prechill test results

if LN_2 and/or a one-g environment is imposed as a constraint. It is expected, however, that the heat transfer phenomenon involved in the prechill process can be evaluated. Empirical coefficients obtained from such tests can be applied to an analytical model, such as HYPRES. This model can then serve as a tool for full scale vehicle prechill predictions.

6.3 TANK FILL MODELLING

Tank fill will be initiated after the prechill requirements have been satisfied. The single requirement for tank fill is to maintain an acceptably low pressure during the process. Tank pressures will be at a minimum if thermal equilibrium conditions are maintained during fill.

The intent of the tank fill process will be to create turbulent conditions within the tank. These conditions will be achieved by introducing liquid into the tank at high velocities (and perhaps through a spray nozzle) to provide the high heat-transfer rates needed to attain near-thermal equilibrium. As tank fill continues, the internal tank fluid environment changes from liquid droplets in the ullage volume to vapor bubbles entrained within a liquid bulk. The transition from heat transfer dominated by liquid droplets to heat transfer dominated by vapor bubbles is expected to occur at about the 40% to 60% liquid fill. This latter mechanism is the only mechanism that will influence tank pressures toward the completion of tank fill.

Tank pressure near the end of fill is more critical than during the early stage (since pressure does not become excessive in the interim), because the end state must reside within an acceptable thermodynamic range to satisfy mission and propulsion system requirements. For this reason, an evaluation is made only of the bubble dominant tank fill process.

6.3.1 VAPOR BUBBLE DOMINANT HEAT EXCHANGE PROCESS.

Scaling parameters for this process can be developed by assuming tank fill to be equivalent to the transient heat conduction process of a lumped-mass system. Consider the heat exchange condition described by Figure 6-14 at some instant in time during fill.

1. Vapor is dispersed throughout the luquid bulk and resides at a higher temperature than the surrounding liquid.
2. Vapor dispersal and heat transfer is caused by fluid agitation, created either by a mechanical mixer or the entering liquid.
3. The temperature-time history of the liquid, as indicated by Figure 6-14, may be characterized by

$$\frac{T - T_\infty}{T_o - T_\infty} = e^{-(hA/mC_p)\tau} \qquad (6\text{-}37)$$

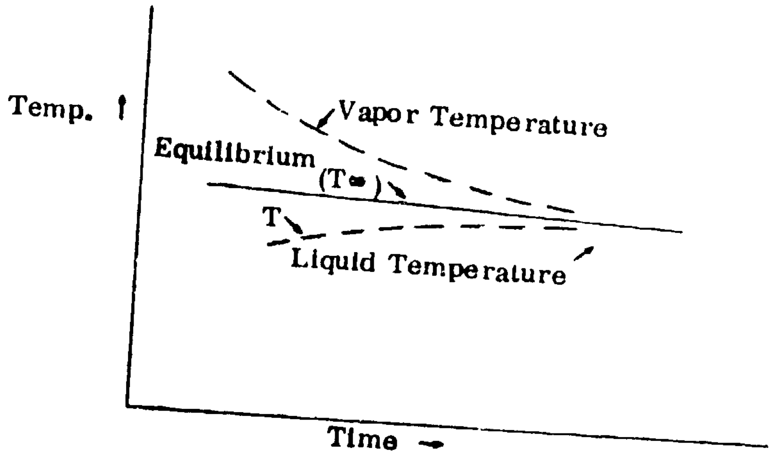


Figure 6-14. (hA τ/mCp) is applicable as time scale parameter for tank fill process.

where

T = liquid temperature at time, t

T_o = initial fluid temperature

T_∞ = equilibrium temperature of vapor and liquid

h = bubble-liquid heat transfer coefficient

A = total surface area of the dispersed vapor phase

m = liquid mass

Cp = liquid heat capacity

τ = time

θ = $(hA/mCp)\tau$, dimensionless time

Note that the development of a time scaling parameter for the fill process begins identically to the approach selected for prechill. This parallel to the prechill analysis will be maintained throughout this evaluation.

The intent of a model fill test is to simulate the thermodynamic exchange between liquid and vapor during fill. Such a simulation would enable one to relate the model fluid temperatures to the prototype (or full scale) conditions. Tank pressure will be influenced by fluid temperature during fill. And, if thermal equilibrium conditions can be attained, tank pressure will be a direct function of liquid temperature. Consequently, tank pressures observed during a model fill experiment can be directly related to the prototype only if near-thermal equilibrium conditions exist.

An inspection of equation (6-37) indicates that model fluid temperatures will be similar to those of the prototype if the dimensionless time parameter, θ, is the same for each condition. That is, if $\theta^* = 1$.

By taking the ratio of model-to-prototype for equation (6-37) we have

$$(T - T_\infty)^*/(T_o - T_\infty)^* = e^{-\theta_m}/e^{-\theta_p} = e^{-(\theta_m - \theta_p)} = e^{-(\theta^*-1)\,\theta_p} \tag{6-38}$$

For $\theta^* = 1$, equation (6-38) becomes

$$(T - T_\infty)^*/(T_o - T_\infty)^* = 1 \tag{6-39}$$

Now, if in addition to $\theta^* = 1$ conditions can be selected such that $T_o^* = T_\infty^* = 1$, then $T^* = 1$(from equation 6-39). This means that fluid temperatures will be identical for the model and prototype conditions at the same dimensionless time, θ. The remainder of this discussion is devoted to identifying model test conditions that will satisfy $T_o^* = T_\infty^* = \theta^* = 1$. Furthermore, all analysis is based upon the following assumptions:

1. Model tank geometry is identical to the prototype tank geometry.
2. Aluminum is material for both model and prototype.
3. Liquid hydrogen is fluid for both model and prototype.

6.3.1.1 Initial fluid temperature. T_o^* can be maintained at unity simply by providing the same propellant supply temperature for model and prototype tank resupply.

6.3.1.2 Equilibrium temperature. The following development will show that equilibrium temperature at any time during the tank fill process will be a function of entering liquid temperature, initial propellant tank stored energy and the percent liquid fill:

From the First Law

$$m_w (u_\infty - u_i)_w + (u_{L\infty} m_L) + (u_{v\infty} m_v) - (u_L m_L)_i - (u_v m_v)_i = h_L (m - m_i) \tag{6-40}$$

$$m = m_L + m_v \text{ (at time, } \tau) \tag{6-41}$$

$$m_i = (m_L + m_v)_i \tag{6-42}$$

where

u = internal energy

h = enthalpy

m = mass

subscript,

L = liquid

v = vapor

w = wall

i = conditions at tank fill initiation

∞ = equilibrium property conditions at time τ

The solution to equation (6-40) can be readily obtained by recognizing that m_{L_i}, m_{v_i} and u_{w_∞} are either zero or insignificant. Fluid mass at tank fill initiation will be essentially zero for filling an evacuated tank and u_{w_∞} will be near zero throughout much of the tank fill because tank wall temperatures will be at or very near liquid temperature. Thus, combining (6-40), (6-41) and (6-42) results in

$$u_{L_\infty} m_L + u_{v_\infty} m_v = h_L (m_L + m_v) + m_w u_{w_i} \tag{6-43}$$

Since $m_w u_{w_i}$ represents the initial tank stored energy, ΔQ_i, we can make this substitution in (6-43)

$$u_{L_\infty} m_L + u_{v_\infty} m_v = h_L (m_L + m_v) + \Delta Q_i \tag{6-44}$$

Dividing (6-44) by m_L gives

$$u_{L_\infty} + u_{v_\infty} (m_v/m_L) = h_L (1 + m_v/m_L) + \Delta Q_i/m_L \tag{6-45}$$

The left hand side of (6-45) is a function only of equilibrium temperature and vapor to liquid mass ratio, m_v/m_L. The right hand side of the equation is a function of entering fluid temperature ($h_L = f(T_o)$), $\Delta Q_i/m_L$, and m_v/m_L.

It is concluded from (6-45) that $T_\infty = f(T_o$ and $\Delta Q_i/m_L)$ for a given m_v/m_L. Therefore, $T_\infty{}^* = 1$ when $T_o{}^* = (\Delta Q_i/m_L)^* = (m_v/m_L)^* = 1$. As stated previously $T_o{}^*$ can be selected by controlling supply temperature. $(\Delta Q_i/m_L)^*$ can be selected by varying model tank mass and/or initial tank temperature. The variable $(m_v/m_L)^*$ imposes no restraint other than to stipulate the obvious, which is that a comparison of model to prototype tank fill behavior is applicable only at the same tank fill condition. However, this requirement does provide the following relationship between time and tank scale

$$m_v{}^* = m_L^* = m^* = 1 = (m_m/m_p) = (\rho V_T)_m/(\rho V_T)_p = V_T{}^* = L^{*3} \tag{6-46}$$

where

V_T = tank volume

L = characteristics tank dimension

ρ = fluid density

ρ_m = ρ_p (same fluid)

Time can be introduced by recognizing that

$$m_L = \dot{m}\tau \quad \text{or} \quad \tau = m_L/\dot{m} \tag{6-47}$$

where

$\dot{m}$ = entering flowrate

τ = flow duration

m_L = liquid mass introduced to tank during τ

Dividing model variables by prototype variables gives

$$\tau^* = m_L{}^*/\dot{m}^* \tag{6-48}$$

Substituting (6-46) into (6-48) results in

$$\tau^* = L^{*3}/\dot{m}^* \tag{6-48a}$$

6.3.1.3 Dimensionless time parameter. It is now necessary to identify conditions under which $\theta^* = 1$.

All variables of the dimensionless time parameter (θ), with the exception of h and A, can be described. Equation 3-11, Reference 3-3, relates h to fluid properties and tank inflow conditions. This relationship is given as

$$\frac{h}{\rho C_p}(N_{PR})^{2/3} = C_1\left[\frac{(\dot{m}v^2/V_L)\,\mu}{\rho^2}\right]^{1/4} \tag{6-49}$$

where all variables have been previously identified except:

V_L = fluid volume

Normalizing equation 6-49 provides the following expression

$$h^* = (\dot{m}^* v^{*2}/V_L^*)^{1/4} \qquad (6-50)$$

The total surface area, A, of the dispersed phase is

$$A = n \ A_b \qquad (6-51a)$$

$$n = V_u/V_b \qquad (6-52)$$

$$V_b = \pi d^3/6 \qquad (6-53)$$

$$A_b = \pi d^2 \qquad (6-54)$$

Therefore

$$A = 6V_u/d \qquad (6-51b)$$

or

$$A^* = V_u^*/d^* \qquad (6-51c)$$

where

n = total number of bubbles immersed in liquid

A_b = bubble surface area (assumed spherical)

V_v = total vapor volume

d = bubble diameter

Now, equation 3-36 can be written as

$$d = F_1 \epsilon^{.5}/(\dot{m} v^2/V_L)^{0.4} + C \qquad (6-55)$$

where

ϵ = V_u/V_T

$F_1 \quad = \quad 1.134\sigma^{0.6}/\gamma^{0.2}$ = constant based upon fluid properties

$C \quad = \quad$ empirical constant = 0.09

By assuming that C is insignificant,

$$d^* \quad = \epsilon^{*\,.5}/(\dot{m}^* v^{*2}/V_L{}^*)^{0.40} \tag{6-56}$$

Al'o since tank ull condition is the same for model and prototype, $V_L{}^* = V_v{}^* = V_T{}^*$ and $\epsilon^* = 1$. Thus (6-56) becomes

$$d^* \quad = \quad (\dot{m}^* v^{*2}/V_L{}^*)^{-0.40} \tag{6-57}$$

This, of course, assumes that the same proportion of vapor is entrained in liquid for both the model and prototype.

If one assumes that C is the dominant term in (6-55) the result is

$$d^* \quad = \quad 1 \tag{6-58}$$

Both 6-57 and 6-58 will be considered in evaluating experiment modelling requirements.

Normalizing the dimensionless time parameter, we have

$$\theta^* \quad = \quad h^* A^* \tau^*/m^* \ (Cp^* = 1 \text{ for same fluid}) \tag{6-59}$$

Substituting (6-48) into (6-59) gives

$$\theta^* \quad = \quad h^* A^*/\dot{m}^* \tag{6-60}$$

Consider first the condition where bubble diameter is primarily influenced by (mv^2/V_L). Substituting equations 6-50, 6-51 and 6-57 into 6-60 provides

$$\theta^* \quad = \quad (\dot{m}^* v^{*2}/V_L{}^*)^{0.25} (V_u{}^*/\dot{m}^*)/(\dot{m}^* v^{*2}/V_L{}^*)^{-0.40} \tag{6-61a}$$

$$= \quad (\dot{m}^* v^{*2}/V_L{}^*)^{0.65} (V_u{}^*/\dot{m}^*) \tag{6-61}$$

Now, from continuity

$$\dot{m} \quad = \quad \rho A_n v \tag{6-62}$$

where A_n = nozzle inlet diameter.

Normalizing (6-62) results in

$$\dot{m}^* = A_n^* v^* \text{ (where } \rho^* = 1 \text{ for same fluid)} \tag{6-63}$$

Substituting (6-63) into (6-61) and recognizing that

$$V_u^* = V_L^* = L^{*3}, \text{ and } A_n^* = L^{*2},$$

$$\theta^* = (\dot{m}^{*3}/A_n^{*2} V_L^*)^{0.65} (L^{*3}/\dot{m}^*)$$

$$= (\dot{m}^{*3}/L^{*7})^{0.65} (L^{*3}/\dot{m}^*) = (\dot{m}^{*0.95}/L^{*1.55}) \tag{6-64}$$

By setting $\theta^* = 1$, (6-64) becomes

$$\dot{m}^* = L^{*1.63} \tag{6-65}$$

Combining 6-63 with 6-65 and solving for v^*, we have

$$v^* = L^{*1.63}/A_n^* = L^{*1.63}/L^{*2} = L^{*-0.37} \tag{6-66}$$

Also, combining (6-48a) with (6-65) and solving for $\tau^* = L^{*1.37}$ (6-67)

Equations 6-65, 6-66 and 6-67 relate the primary test variables of flowrate, velocity and time to model tank scale for the condition where bubble diameter, d, is a function of fluid power input (or mechanical power). These relationships will exist at low power input levels. Results are given in Table 6-5.

Consider now the second condition where bubble diameter is independent of $(\dot{m}v^2/V_L)$.

Substituting equations (6-50), (6-51c), and (6-58) into (6-60) gives

$$\theta^* = (\dot{m}^* v^{*2}/V_L^*)^{0.25} (V_u^*/\dot{m}^*) \tag{6-68}$$

Substituting (6-63) into (6-68) and recognizing that

$$V_u^* = V_L^* = L^{*3}, \text{ and } A_n^* = L^{*2}$$

results in

$$\theta^* = (\dot{m}^{*3}/L^{*7})^{0.25} (L^{*3}/\dot{m}^*) = L^{*1.25}/\dot{m}^{*0.25} \tag{6-69}$$

By setting $\theta^* = 1$, (6-69) becomes

$$\dot{m}^* = L^{*5} \tag{6-70}$$

Combining (6-63) with (6-70) and solving for v^* we have

$$v^* = L^{*5}/L^{*2} = L^{*3} \tag{6-71}$$

Also, combining (6-49) with (6-70) and solving for *,

$$\tau^* = L^{*-2} \tag{6-72}$$

Results of equations (6-70), (6-71), and (6-72) are given in Table 6-6.

Table 6-5. Model tank scale influence upon fill tank variables. ($d^* = f(\dot{m}^* v^{*2}/V_L{}^*)$)

Model Test Variable	Model Tank Scale, L^*		
	0.108(1)	0.2	0.3
$\dot{m}^*$	0.0266	0.07	0.14
v^*	2.278	1.81	1.56
τ^*	0.0474	0.11	0.19

(1) Tank model for Spacelab experiment.

Table 6-6. Model tank scale influence upon fill tank variables. ($d^* = 1$)

Model Test Variable	Model Tank Scale, L^*		
	0.108(1)	0.2	0.3
$\dot{m}^*$	1.47×10^{-5}	0.0003	0.002
v^*	1.26×10^{-4}	0.008	0.027
τ^*	85.7	25	11.11

(1) Tank model for Spacelab experiment

- Prototype Tank Dimension, l_p = 4.29m (169 inches) (diameter)
- $\theta^* = 1$
- Assumed Test Conditions
  - o Same Propellant (LH_2)
  - o Same Initial Tank Temperature, 200K, (360R)

The question to be resolved is which set of scaling equations should be satisfied during a model fill experiment: the equations based upon $d^* = 1$, or upon $d^* = f(\dot{m} * v^{*2}/V_L)$? The answer depends upon determining the following:

1. What are the vapor to liquid heat transfer rates needed to guarantee thermal equilibrium during tank fill?
2. What fluid power (or equivalent mixer power) requirements will provide the needed heat transfer rates?
3. Will propellant tank fill durations be acceptable at the identified fluid power input conditions.
4. Is the calculated bubble diameter a function of or independent of fluid power (or mixer power) over the range of anticipated power input conditions?

First, Figure 6-15 gives the average vapor to liquid heating rate required during tank fill to assure near thermal equilibrium conditions. Note that heating rate requirements decrease as tank fill duration increases. This occurs because the total ullage energy removal requirement is independent of fill duration, and a longer fill duration means that the average heat removal rate can be decreased. A fill duration of 3 hours was selected as acceptable in section 3-3, and results in a required average heating rate of about 5 kw (~5 Btu/sec).

Second, it is seen from Figures 6-16 and 6-17 that the required heating rate can be

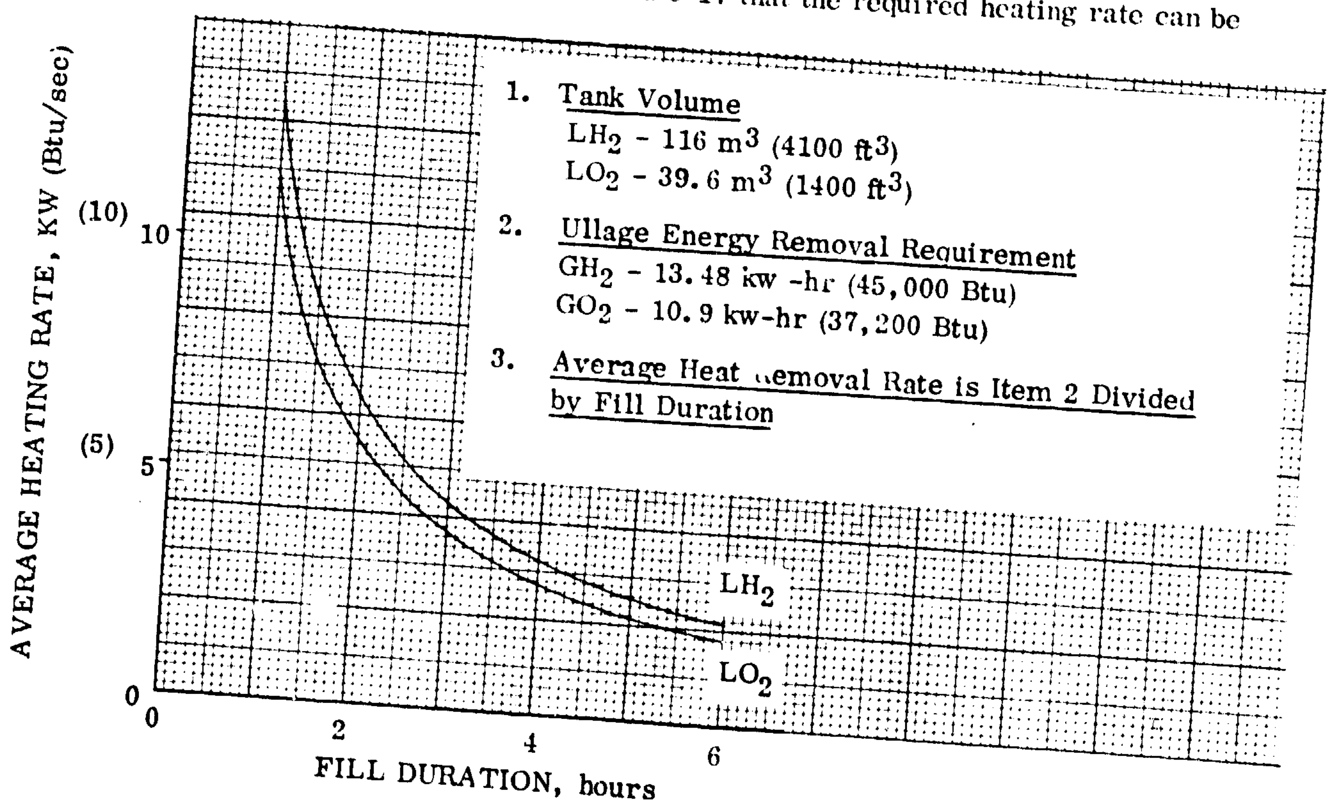


Figure 6-15. Average vapor to liquid heating rate needed to achieve thermal equilibrium.

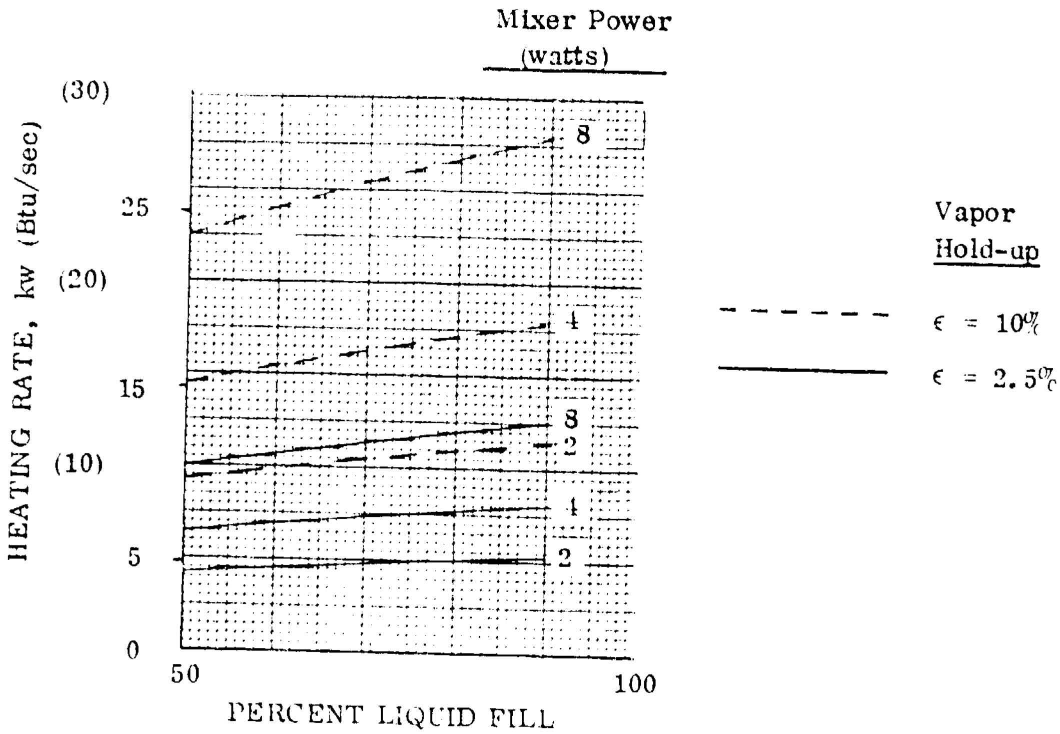


Figure 6-16. Mixer Power Influence Upon Entrained Vapor to Liquid Hydrogen Heat Transfer Rate

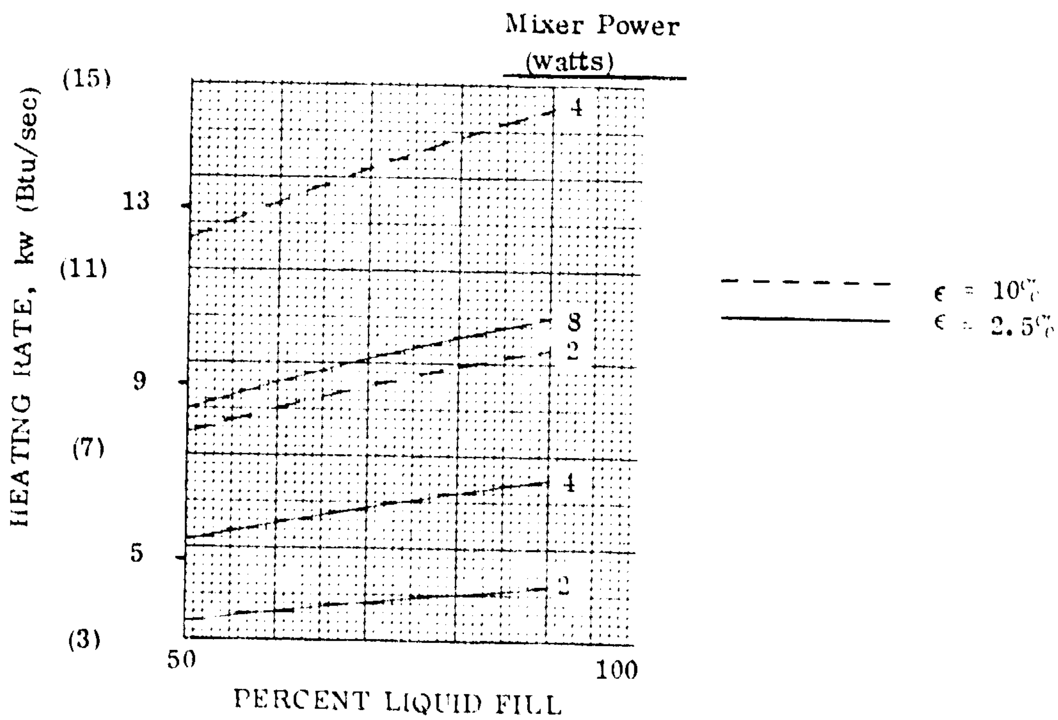


Figure 6-17. Mixer Power Influence Upon Entrained Vapor to Liquid Oxygen Heat Transfer Rate

achieved for a mixer power input less than 4 watts. Note thatinput power requirements to achieve a given heating rate are a strong function of vapor hold-up (percent vapor entrained in liquid), and a very weak function of the percent liquid fill condition. According to Figures 6-18 and 6-19, vapor bubbles generated within the OTV propellant tanks by a mixer (or its equivalent in fluid power) are a function of fluid power for power levels less than about 12 watts. Thus it appears that the model test parameters of Table 6-5 should be employed for tank fill experimentation. It is encouraging to note from Table 6-5 that the required model test flowrate and velocity requirement for the Spacelab experiment can be readily achieved.

6.3.1.4 Mixer power/fluid power relationship. Additional information applicable to orbital refill is presented in Figures 6-20 and 6-21. These curves show the equivalence between fluid power and mixer power, but using variables of fill duration and transfer line nozzle inlet diameter to describe fluid power entering the propellant tanks during fill. Equivalence was established in the following manner:

$$\text{Fluid Power} = \dot{m}\, v^2 = \dot{m}^3 / \rho_L^2 A_n^2 = C_1 \left(\dot{m}^3 / \rho_L^2 D_n^4\right) \qquad (6\text{-}72)$$

where

D_n = nozzle diameter

$\dot{m}$ = tanking flowrate

v = inlet velocity

ρ_L = propellant density

C_1 = constant

But

$$\dot{m} = m_T/\tau = \rho_L V_T/\tau \qquad (6\text{-}73)$$

Therefore

$$\dot{m}v^2 = C_1 \rho_L / D_n^4 \left(V_T/\tau\right)^3 \qquad (6\text{-}74)$$

where

m_T = tanked propellant mass

V_T = propellant tank volume

τ = tanking duration

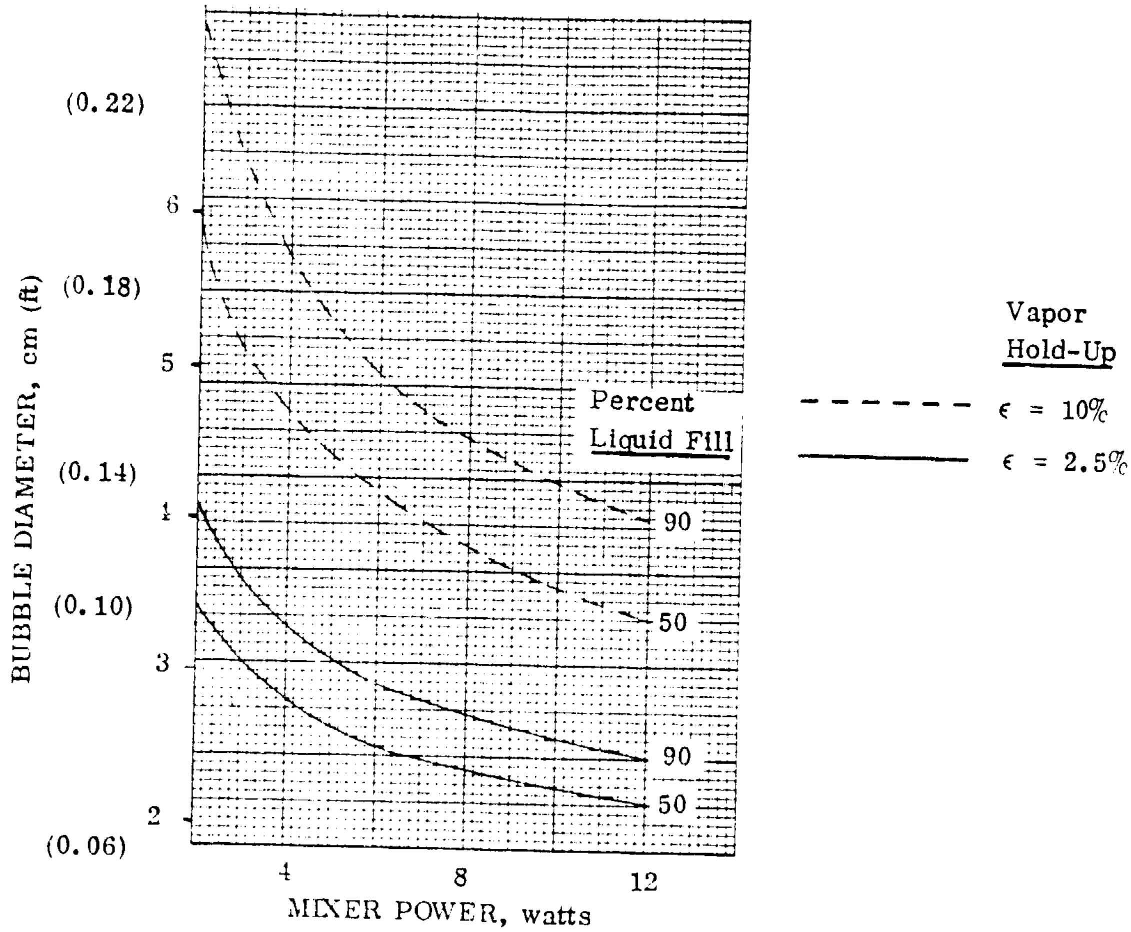


Figure 6-18. Mixer Power Influence Upon Hydrogen Bubble Diameter During Tank Fill

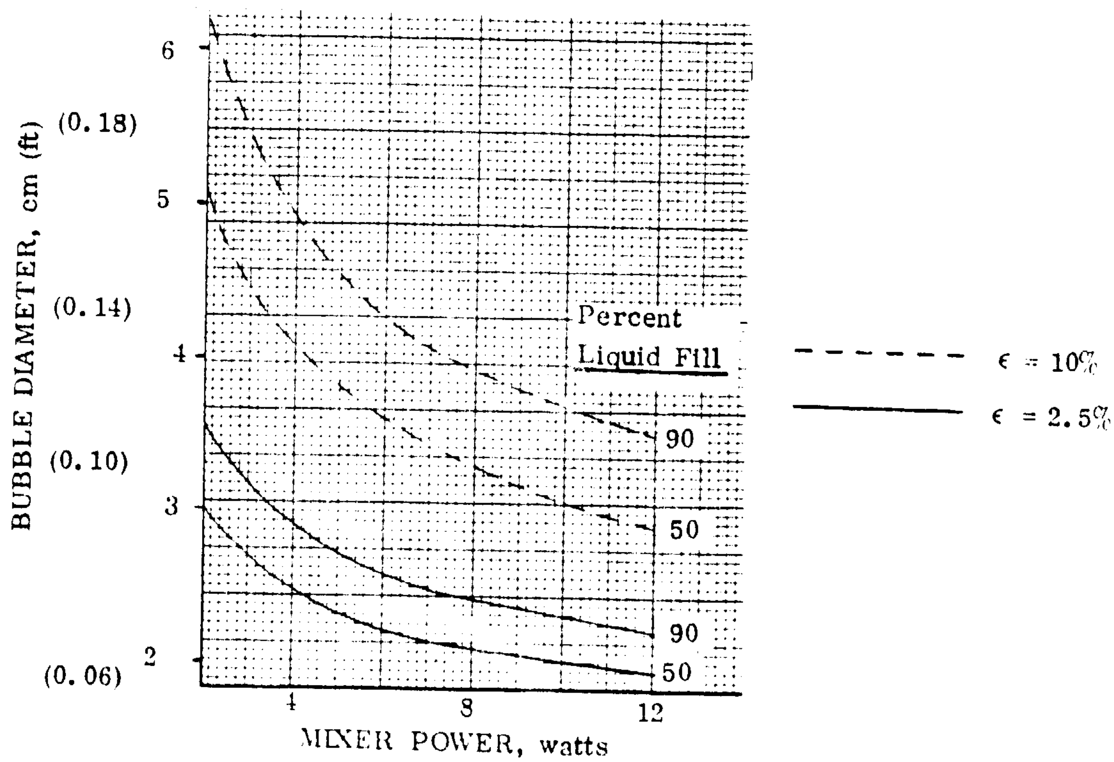


Figure 6-19. Mixer Power Influence Upon Oxygen Bubble Diameter During Tank Fill

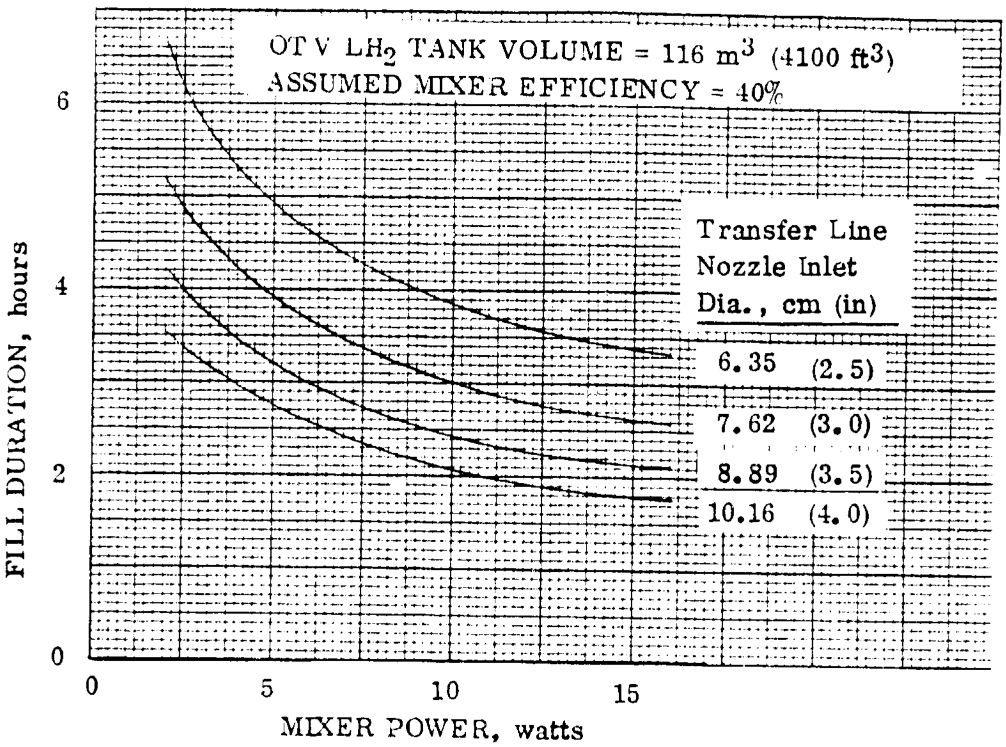


Figure 6-20. Fluid Power Input Equivalence to Mixer Power During OTV LH_2 Tank Fill

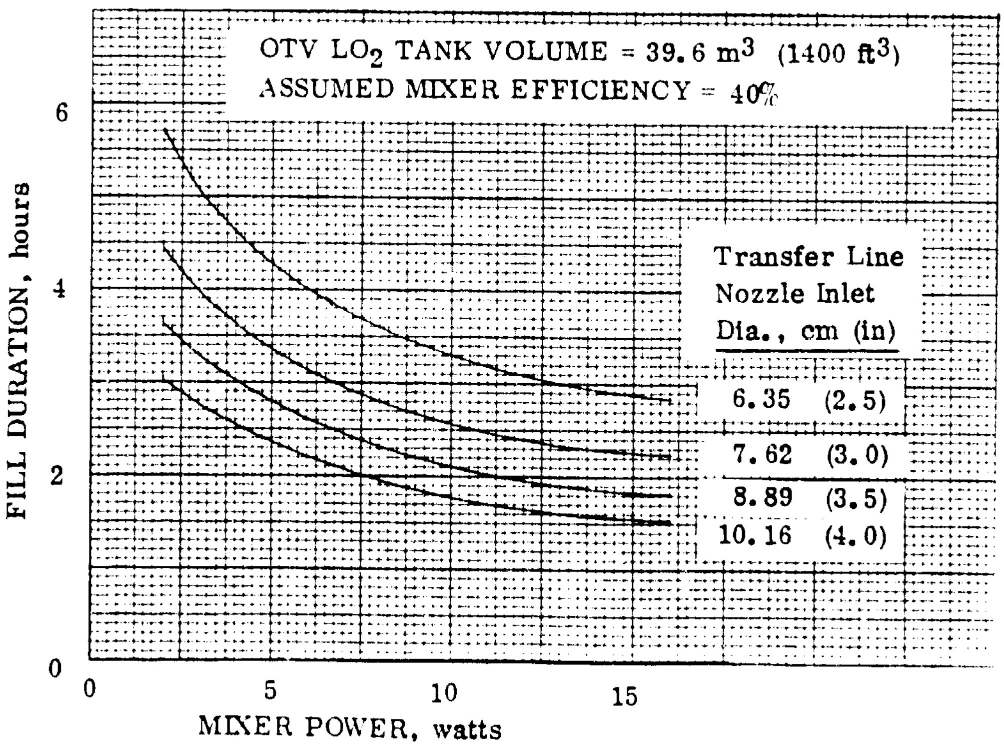


Figure 6-21. Fluid Power Input Equivalence to Mixer Power During OTV LO_2 Tank Fill

Finally, we can show that

Fluid Power = Mixer Power Output

= Efficiency × Mixer Power Input

Note that fluid power requirements of greater than 4 watts can be achieved with a reasonable selection of fill duration and nozzle diameter.

6.3.1.5 Model tank V*/M* influence. The inability to exactly scale POTV on the bases of tank volume and tank mass will have a significant influence on tank fill, as it will have upon the prechill process. Higher model tank pressures will occur during the initial fill transient than for the prototype LH_2 tank, as previously discussed and indicated by Figure 6-7. This peak pressure could be excessive, especially for the 0.108 scale tank which has V*/M* = 0.105. Equally significant is the fact that equilibrium pressures at the end of tank fill can also be heavily influnced by V*/M*. Figure 6-22 gives final tank pressure as a function of entering LH_2 vapor pressure and V*/M*. It is seen that final tank pressure could increase by about 69 kN/m^2 (10 psid) greater at a V*/M* of 0.108 than for the prototype tank. Conversely, if the goal is to achieve a given final pressure, the model test tank will require an entering LH_2 vapor pressure that is about 69 kN/m^2 (10 psid) lower than for the prototype tank.

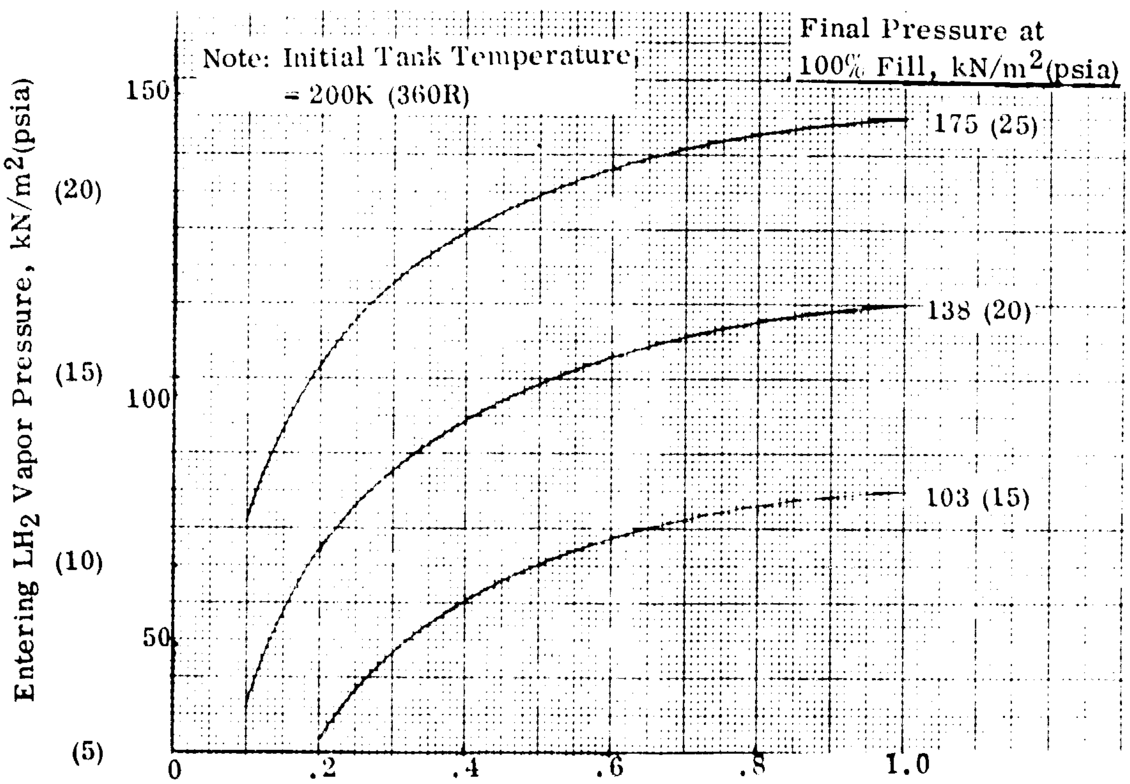


Figure 6-22. V*/M* influence upon POTV LH_2 tank pressure following thermal equilibrium fill process.

3.3.2 FLUID SUBSTITUTE. A determination was made in Section 6.3.1 that a propellant tank fill process can be simulated if the model-to-prototype dimensionless time parameter, θ^*, can be established as unity for experimentation. Model test parameters of flowrate, velocity and time scaling are identified in Table 6-5 which establish $\theta^* = 1$ for model tests conducted with liquid hydrogen. The purpose of this analysis is to quantify these same test parameters for the case where liquid nitrogen is substituted for liquid hydrogen. The following equations employed in Section 6.3.1 are also applicable to this analysis:

$$\theta = (hA/mCp)\ \tau \tag{6-10}$$

$$\frac{h}{\rho Cp}\ (N_{PR})^{2/3} = C_1 \left[\frac{(\dot{m}v^2/V_L)}{\rho^2}\right]^{1/4} \tag{6-49}$$

Normalizing equation (6-10) by taking the ratio of model-to-prototype conditions results in,

$$\theta^* = (h^*A^*/m^*Cp^*)\ \tau^* = (h^*A^*/\dot{m}^*Cp^*) \tag{6-73}$$

where:

$$\dot{m}^* = m^*/\tau^*$$

and superscript (*) = the ratio of model to prototype (in this case, LN_2 to LH_2 conditions for same tank scale)

Normalizing equation (6-49) and solving for h* gives,

$$h^* = (\dot{m}^*v^{*2}/V_L)^{.25}\ \mu^{.25}\ \rho^{*.5}\, Cp^*/(N_{PR}^*)^{2/3} \tag{6-74}$$

$$= (\dot{m}^*v^{*2})^{.25}\ \mu^{.25}\rho^{*.5}\, Cp^*/(N_{PR}^*)^{2/3} \tag{6-75}$$

where:

$$V_L^* = 1 \text{ (for same tank scale)}$$

Now, combining (6-73) and (6-75) results in

$$\theta^* = (A^*/\dot{m}^*)\ (\dot{m}^*v^{*2})^{.25}\mu^{*.25}\ \rho^{*.5}/(N_{PR}^*)^{2/3} \tag{6-76}$$

$$= 4.43\ (A^*/\dot{m}^*)\ (\dot{m}^*v^{*2})^{.25} \tag{6-77}$$

where

$$\mu^{*.25} = (11.85)^{.25} = 1.86$$

$$\rho^{*.5} = (11.4)^{.5} = 3.38$$

$$(N_{PR}^*)^{2/3} = (1.7)^{2/3} = 1.42$$

The normalized total surface area, A^*, of the dispersed phase is $A^* = Vu^*/d^*$ (6-51c)

Now, the expression for bubble diameter, d, was found to be

$$d = F_1 \epsilon^{.5} / (\dot{m} v^2 / V_L)^{0.4} + C \qquad (6\text{-}55)$$

where:

$\epsilon = V_u / V_T$

F_1 = constant based upon fluid properties

V_T = tank volume

C = empirical constant

It was determined in Section 6.3.1.3 that the expression for d can be simplified by eliminating C. Consequently, we have

$$d = F_1 \epsilon^{.5} / (\dot{m} v^2 / V_L)^{0.4} \qquad (6\text{-}78)$$

Normalizing equation (6-78) and recognizing that $V_L^* = V_u^* = V_T^*$,

$\epsilon^* = 1$ and,

$$d^* = F_1^* / (\dot{m}^* v^{*2})^{0.4} \qquad (6\text{-}79)$$

Now $F_1^* = .\sigma^{*.6} / \rho^{*.2} = 1.26$

where: σ = liquid vapor surface tension

and $\sigma^{*.6} = (3.3)^{.6} = 2.05$

and $\rho^{*.2} = (11.4)^{.2} = 1.63$

Combining equations (6-51c) and (6-79) and recognizing that

$$V_u^* = 1 \text{ we have}$$

$$A^* = 1/d^* = (\dot{m}^* v^{*2})^{0.4}/1.26 \qquad (6\text{-}80)$$

Substituting equation (6-80) into (6-77),

$$\theta^* = 3.52\,(\dot{m}^* v^{*2})^{.65}/\dot{m}^* \qquad (6\text{-}81)$$

Introducing the continuity equation, $\dot{m} = \rho A v$, and normalizing, we have

$$\dot{m}^* = \rho^* v^* A^*_{nozzle} \text{ where } A^*_{nozzle} = 1.0 \qquad (6\text{-}82)$$

Substituting (6-82) into (6-81) and solving for $\dot{m}^*$,

$$\dot{m}^* = .266 \rho^{*1.368} \theta^{*1.053} = 7.42\,\theta^{*1.053} \qquad (6\text{-}83)$$

Finally, in order to satisfy the requirement for $\theta^* = 1$, we find from equation that

$$\dot{m}^* = 7.42 \qquad (6\text{-}84)$$

$$v^* = .651 \qquad (6\text{-}85)$$

$$t^* = 1.536 \qquad (6\text{-}86)$$

Equations (6-84), (6-85), and (6-86) represent the ratio of nitrogen-to-hydrogen test variables. These factors were applied to the previously determined liquid hydrogen model test variables (of Table 6-5) to arrive at the data given in Table 6-7. An inspection of this data indicates that there appears to be no limitations nor disadvantages to conducting model tests with LN_2, other than the concern that the fluid properties difference could result in a non-scaleable heat and mass exchange difference between the propellants.

6.3.3 ONE-G TEST ENVIRONMENT LIMITATIONS. Ground-based tests for determining the validity of the tank fill process will be applicable only if a) the normal gravity free convection heat transfer process is dominated by the forced convection mechanism created by the entering propellant, and b) the fluid flow mechanism is the same for model tests as for the prototype configuration. To satisfy the former condition we must verify that the heat transfer coefficient given by equation (6-9) exceeds that free convection coefficient created by a normal gravity environment. Equation (6-9) is applicable to heat exchange between a propellant tank surface and its contained vapor, and to heat exchange between a liquid and entrained vapor bubbles. The latter condition applies to the similarity of fluid flow regimes such as laminar or turbulent flow in a flow inertia

Table 6-7. Model tank scale and fluid substitute influence upon fill test variables $d^* = f(\dot{m}^* v^{*2}/V_L^*)$, for LN_2.

Model Test Variable	Model Tank Scale, L*		
	.108 (1)	.2	.3
m*	.15	.52	1.04
v*	1.52	1.18	1.02
τ *	.06	.17	.29

(1) Tank model selected for Spacelab experiment.

- Reference data is from Table 6-5.

- Test fluid is LN_2

dominated environment.

It is expected that propellant tank fill in space will cause the propellant and its vapor to be intimately mixed due to the absence of gravity. Thus the fluid flow mechanism will be that of an inertia dominated process. Furthermore, this intimate mixture of liquid and vapor should serve to provide near-thermal equilibrium conditions during fill.

This same uniform mixing of liquid and vapor will not be possible for tests conducted in a normal gravity environment because gravity will tend to maintain the liquid phase separated from vapor. Analyses have indicated that incoming liquid velocities, for a 0.108 scale model tank, may have to be increased by a factor of five to ten greater than scaling would indicate just to provide liquid-vapor mixing. A velocity increase of this magnitude could invalidate the experiment because of a greatly increased vapor bubble-to-liquid heat exchange. As with the prechill process, it is questionable that useful data can be obtained for predicting the full-scale process.

6.3.4 START BASKET REFILL TEST CONSIDERATIONS. Refill of the propellant start baskets is an important requirement during the tank fill process. The start baskets must be free of vapor prior to first main start so that 100 percent liquid flow to the OTV main engines can be assured. It was determined in the analysis of Section 3.3.6 that an unknown quantity of propellant vapor may reside within these screen devices at tank fill completion. An extremely conservative assumption was made that the largest spherical bubble possible would reside within each device. Collapse times of about five minutes and four hours were calculated for the oxygen and hydrogen bubbles, respectively. It is obvious from the predicted collapse times that oxygen start basket refill will be assured. The long time to collapse hydrogen bubbles, however, resulted in a recommendation to actively assist hydrogen start basket refill. It was proposed that a small diameter line be routed from the tank fill port to the basket so that liquid would be sprayed into the volume during the fill process. Calculations, for a conservative fill model, indicated that all entrapped vapor would be condensed before tank fill completion.

The mechanism for removing vapor from the start basket is identical to that described for tank fill. That is, the incoming liquid momentum will serve the dual function of creating small bubbles and a high heat exchange environment between liquid and vapor, both of which enhance condensation. The strategy to assure start basket refill will be to create sufficient agitation within the screen volume to assure complete vapor condensation. It appears that there will be fewer restraints imposed upon start basket refill than tank refill, consequently, a propellant flow split can be selected to assure basket refill before tank fill completion. As an extreme example, tank fill could be performed by flowing all propellant through the screen device.

Since it appears that basket refill can be assured, and because tank fill experiments will also be useful in assessing condensation conditions within the start basket, a detailed test program is not recommended. Rather, it may be sufficient to provide a range of start basket flowrates as part of the tank fill experiment. There would be an advantage to performing several tank fill experiments without flowing propellant to the start basket. These tests would identify if an active means of start basket refill is necessary.

6.4 SPACELAB EXPERIMENT INTEGRATION

The Spacelab provides facilities for investigating fluid behavior in a low gravity environment. The experiment payload accommodated within the pressurized Spacelab, however, must adhere to strict safety regulations regarding the crew, the mission, and the payloads. Specifically, Reference 6-1 restricts the use of cryogens to the exterior of the Spacelab.

To confirm the Reference 6-1 guidelines, Merle Slayden, a NASA/MSFC Safety Engineer was contacted to discuss current directives regarding the use of cryogens. The proposed receiver tank experiment, employing a 457 mm (18 inch) diameter tank mounted within a double rack, was discussed and he stated such LN_2 quantities would require a waiver for Spacelab. Unfortunately a waiver can only be obtained during the experiment integration safety review period, which occurs after payload design and fabrication. As a guideline, Mr. Slayden offered the information that LN_2 quantities of about 3.6 kg (8 lb) would be acceptable. The proposed test tank would contain a maximum of 120 kg (264 lb) LN_2. Liquid hydrogen is unacceptable under any condition.

There will be additional restraints imposed upon the receiver tank experiment design other than the use of LN_2. These Spacelab general experiment accommodation factors are given in Appendix B and have been extracted from the "Life Sciences Guide to the Space Shuttle and Spacelab" experimenters handbook, and the Reference 6-1 handbook

6.5 MODELLING OF LTL REFUELING OPERATIONS

A mission scenario was selected in Section 5 for the purpose of analyzing orbital refill operations. A low thrust liquid (LTL) vehicle configuration was selected from existing vehicle hardware; principally, the OMS tanks were selected as the main propellant tanks, and the shuttle RCS bottles were selected for the same function on the LTL. N_2O_4 and MMH were the propellants selected for the main propulsion and RCS systems. Each of the storage tanks and bottles contained a screen acquisition device.

The analysis of Section 5 identified the refill requirements listed in Table 6-8 . It was determined that all potential problems, such as liquid venting and helium entering the screen devices, resulted from the need to expel helium from the storage vessels before initiating refill. Furthermore, it was found that refill could be performed in a very straightforward manner; venting would not be required, and tank pressures would not exceed normal operating pressures during refill.

Table 6-8 . LTL vehicle refueling requirements.

- Propellant Tank Venting Must Not Damage LTL or Orbiter
  - Liquid Venting Potentially More Damaging Than Vapor Venting
- Helium Must Not Enter Partial Screen Device Galleries
- Vehicle Control Must Be Maintained During Refill Operations
  - Liquid Venting is Undesirable
- Simplified Operations Are a Necessity
- Must Have Ability to Refill to ~95% Level

It was concluded from the refill analysis that virtually all potential problems would be eliminated through the use of procedures, if certain vehicle modifications were incorporated. These modifications included adding plumbing between storage vessels (see Figure 5- 5) to enable propellant transfer between common tanks. With these changes, procedures were established so that propellants could be transferred from one tank to another before initiating a tank vent.

The single concern of the selected refueling procedure was that propellant contained within the screen devices might boil during tank vent to expel helium. The solution would be to maintain sufficient liquid residual that it would replenish any screen device liquid lost by evaporation. This solution will be effective only if communication is maintained between liquid pool and screen device. Thus, the purpose of an orbital experiment would be to verify that communication is maintained during the tank vent process, and this will be a function of geometry (tank and screen) and acceleration environment.

Laboratory experiments conducted in a normal gravity environment will not be useful because a near zero-g environment is required. The appropriate acceleration environment can be provided by orbital experiments.

Orbital experiments to verify that the screen devices will remain filled with liquid are not recommended. Such tests would not have wide applicability because results would be very sensitive to geometry. It is recommended that experiments of this type not be conducted until an operational vehicle configuration can be established.

7

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4-1 "Orbital Propellant Handling and Storage Systems for Large Space Programs," JSC-13967, CASD-ASP-78-001, April 1978.

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APPENDIX A

IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

TABLE A-1. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

GENERAL SUMMARY (1975-1990) Per CASD-NAS76-014

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature Liquid		Temperature Gas		Flow Rate			Comment/Reference
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m²	Category	kN/m²	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m²-sec	< 100 kg/m²-sec	> 100 kg/m²-sec	
1	STS Orbiter (Space Shuttle) 2 OMS 6000 LBF ea 38 RCS 870 LBF ea 6 RCS 25 LBS ea	OMS	N_2O_4	7047 (15536)		x	(x)		x			1750	Med			0-1	<1		x			278 <T				10.6-11.4	x		2 Tanks (+3) each
			MMH	4462 (9836)		x	(x)		x			1758	Med			0-1	< 1		x			< 312	Avg Contr			5.6-6.4	x		Table A-1, NASA CR 134911, Ref. 1
		RCS	N_2O_4	1852 (4084)			(x)	x				1689	Med			0	< i		x			278 < T							3 Tanks each
			MMH	1158 (2553)			x	x				1689	Med			0	< 1		x			< 312	Avg Contr						Table A-1, NASA CR 134911, Ref. 1
2	STS External Tank (Logistics) OTV	Oxidizer Prop.	LO_2	603730 (1331000)	x				x	(x)				152	Low	0-3	> 1				x	90	Cryo						Table A-2, CR 134911, Ref. 1
		Fuel	LH_2	100700 (222000)	x				x					234	Low	0-3	> 1				x	20	Cryo						
3	Orbital Transfer Vehicle	MPS Oxidizer	LO_2	21-40K (47-89K)		x			x	x						0-3	> 1		x			90	Cryo			< 12000(i)			JSC 11568, Vol. I, 31 Aug 77, Ref. 2. See Items 21 thru 36 for additional versions
		MPS Fuel	LH_2	3.6-6.8K (8-150K)	x	x			x	x						0-3	> 1		x			20	Cryo			<100	x		
4	Reusable Tug (GD Centaur Derivative, one 15000 LBF Engine)	MPS Oxidizer	LO_2	30345 (66900)		x				x		138	Low	138	Low	0-3	> 1		x			90	Cryo			< 12.68	x		NAR Report SD 71-292-4, 3-22-71, Table A-3, CR 134911, Ref. 3, Ref. 1.
		MPS Fuel	LH_2	5035 (11100)		x				x		138	Low	138	Low	0-3	> 1		x			20	Cryo			< 2.11	x		
		RCS	O_2	22.7 (50)			x	x				6895	Med			0-3	> 1				x			> 211					Memo ER-78-004, 1-13-78, Cryo Stage Utilization, CASD-NAS-76-014, p. A-5, 5/76; Ref. 4, Ref. 1
			H_2	5.4 (12)			x	x				6895	Med			0-3	> 1				x			> 111					
5	Reusable Centaur, (2 ea. 15000 LBF Engine)	MPS Oxidizer	LO_2	17830 (39310)		x				x		231	Low			0-3	> 1	x				90-97	Cryo			< 25.4	x		Table A-5, CR 134911, Ref. 1, Memo ER-78-004, 1-13-78, Ref. 4.
		MPS Fuel	LH_2	3340 (7370)		x			x	x		159	Low			0-3	> 1	x				20-21 278 < T	Cryo Avg			< 4.2	x		
		Auxiliary Propulsion	H_2O_2	220 (484)			x		x			2000	Med			0-3	> 1			x		< 323	Contr						
6	Reusable Transtage, 2 ea. 8000 lbs Engine + SP 308 sec a2:1	MPS	N_2O_4	9662 (21300)		x			x			1427	Med			0-3	> 1		x			294	Avg			< 15.7	x		Table A-6, CR 134911, Ref. 1
		ACS	Aerozine 50	4853 (10700)		x			x			1289	Med			0-3	> 1		x			294	Avg			< 7.8	x		
			N_2H_4	121 (267)			x	x				2551	Med			0-3	> 1			x		294	Avg						

x* = Cylindrical Tank + Ellipsoidal Ends

(i) - inlet, (e) = exit

TABLE A-1. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

GENERAL SUMMARY (1975-1990) Per CASD-NAS76-014

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature Liquid		Temperature Gas		Flow Rate			Comment/Reference
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m^2	Category	kN/m^2	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m^2-sec	<100 kg/m^2-sec	>100 kg/m^2-sec	
7	Reusable Agena (One 16000 Lb Engine, ISP ~320 sec)	MPS Fuel	MMH	2254 (4970) to 6132 (13518)		x				x		276	Low			0-3	>1 <1				x	< 297	Avg			< 7.6	x		Table A-7, CR 134911, Ref. 1
		MPS Oxidizer	N_2O_4	4568 (10070) to 12433 (27441)		x			x			241	Low			0-3	>1				x	< 297	Avg			< 15.1	x		
		RCS	N_2H_4	46 (102)			x	x				2413	Med			0-3	>1					< 297	Avg						
8	Satellite Control Section Ops 1 ea 300 LBF Engines																< 1			x		198 < T	Avg			< 0.6	x		Table A-8, CR 134911, Ref. 1 (4 RCS Tanks)
		OAS Orbital Adj. Sys.	N_2H_4	1320 (2910)			x	x				2137	Med			0-1	< 1		x			< 294 278 < T < 333	Avg						
		RCS	N_2H_4	187 (412)			x	x				1820	Med			0-1	< 1			x									
9	Space Tub (NASA MSFC) (One 15000 Lb Engine)	MPS Fuel	LH_2	3459 (7626)	x			x				151.7	Low			0-3	< 1 < 1				x	20	Cryo			< 25.4	x		Table A-9, CR 134911, Ref. 1. CASD NAS 75-017, June 1975, Vol. III, p 6-10 & 6-11, Ref. 5.
		MPS Oxidizer	LO_2	19764 (43574)	x				x			141.3	Low			0-3	>1				x	90	Cryo			< 4.2	x		
		ACS	N_2H_4	152 (336)		x						2282	Med			0-3	>1			x		167 < T < 294	Avg Contr						
10	Chemical Inter-Orbital Shuttle (~SII Stage), Conf. A, 2nd Stage 2 ea 200000 LBF, 1st Stage 4 ea 200000 LBF, Engines	MPS Oxidizer	LO_2	399150 (880000)	x			x				193	Low			0-3	>1	x				90	Cryo			676 338		x	Table A-11, CR 134911, Ref. 1. SII Stage Inter-Orbital Shuttle Capability. Analysis, SD 71-145-1, Apr 21, 1971, Ref. 6.
		MPS Fuel	LH_2	72570 (160000)	x			x				179	Low			0-3	>1	x				20	Cryo			112.8 56.4		x	
		OMS Oxidizer	LO_2		x		x									0-3	>1	x				90	Cryo						
		OMS Fuel	LH_2		x		x									0-3	>1	x				20	Cryo						
		RCS Oxidizer	SC O_2				x					5516	Med			0-3	>1			x			Cryo						
		RCS Fuel	SC H_2				x					3447	Med			0-3	>1			x			Cryo						

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-1. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

GENERAL SUMMARY (1975-1990) Per CASD-NAS76-014

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m^2	Category	kN/m^2	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m^2-sec	<100 kg/m^2-sec	> 100 kg/m^2-sec	
11	Chemical Inter-Orbital Shuttle, 1-1/2 Stage Configuration, 4 ea, 200000 LBF ENG/HES	MPS Oxidizer	LO_2	118600 (261500)	x			x	?			186	Low			0-3	>1	x			90	90	Cryo			338		x	Table A-12, CR-134911, CASD-NAS76-014, Low-G Fluid Transfer Technology Study, 1976, Ref. 1
		MPS Fuel	LO_2	19700 (43500)	x			x	?			172	Low			0-3	>1	x			20	20	Cryo			56.4	x		
		Drop Tank MPS Oxidizer	LO_2	260500 (574300)	x				x							0-3	>1	x			90	90	Cryo			676.0		x	
		Drop Tank MPS Fuel	LH_2	43400 (95700)	x				x							0-3	>1	x			20	20	Crye			112.8		x	
12	Modular Space Station	RCS Oxidizer	GO_2	275 (606)			x	x				20685	High			0-0.1	<1			x	x			>211	Avg		x		Table A-15, CR-134911, Ref. 1, SD-71-214-4, Ref. 7.
		RCS Fuel	GH_2	34 (76)			x	x				20685	High			0-0.1	<1			x	x			>111	Avg		x		
13	Large Space	Hi Thrust RCS	N_2H_4	3180 (7000)			x		x			20679	High			0-0.2	<1			x	x	278 <T 312	Avg				x		Table A-16, CR 134911, MDC GO634, Vol. 1, Book 4, pages 118-122; Ref. 1, Ref. 8.
		Low Thrust RCS	H_2O	204 (450)			x	x								0-0.2	<1			x	x	274-372	Avg				x		
14	Large X-Ray Telescope (HE-01-A)	Orb Maint Propellant	N_2H_4	(250) × 4			x	x					Low			0-0.1	<1			x		278 <T 312	Avg				x		NASA Payload Descriptions Vol. I, p. 2-109, July, 1975, Table A-12, CR134911; Ref. 9, Ref. 1.
		RCS	N_2	100 (220)			x	x					High			0-0.1	<1				x			263-303	Wide Range		x		
15	Large High Energy Observatory B (Magnetic Spectrometer) HE-09-A	Orb Maint Propellant	N_2H_4	(250) × 4			x	x					Low			0-0.1	<1			x		278 < T 312	Avg				x		Payload Descriptions, Vol. I, p. 2-109, July 1975, Table A-17, CR 134911, Ref. 9, Ref. 1.
		RCS	N_2	100 (220)			x	x					High			0-0.1	< 1				x			263-303	Wide Range				
16	Geosynchronous Platform Observ. Class ~ 25000 Lbs	RCS	N_2H_2	80 (176)			x	x					Low			0-0.1	<1			x		278 < T 312	Avg				x		Table A-19, CR134911, Ref. 1
17	Upper Atmosphere Explorer	OAS + RCS	N_2H	175 (385)			x	x								0-1	< 1			x		278 < T 312	Avg				x		NASA Payload Descriptions, Vol. I, Automated Payloads, July '75, p. 4-6, Ref. 9.
18	1.5M Propellant Depot	RCS	LO_2	68000 150000)	x			x				138	Low			0-0.2	< 1				x	90	Cryo				x		Table A-22, P.A-14, CR 134911, CASD-NAS76-014, May '76; S-II Orb. Prop. Study, SD7-554, 3-31-72; Ref 1, Ref 10
		RCS	LH	(215000) × 2	x			x				138	Low			0-0.2	< 1				x	20	Cryo				x		

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-1. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

GENERAL SUMMARY (1975–1990) Per CASD-NAS76-014

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m^2	Category	kN/m^2	Category	x	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m^2-sec	< 100 kg/ m^2-sec	> 100 kg/ m^2-sec	
19	Space Taxi	MPS + RCS	N_2H_4	106 (234)			x	x				1724	Med			0-0.5	< 1			x		$278 < T \leq 312$	Avg				x		Table A-23, p. A-15, CR-134911, Ref. 1

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-2. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS — ORBITAL TRANSFER VEHICLES (OTV'S)

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: < 100 kg/m²-sec	Flow Rate: > 100 kg/m²-sec	Comment/Reference
20	Teleoperator (6 ea 25 LB Thrusters/Tank, 8 Triads X3-24 Cold Gas Thrusters	OAS + RCS	N_2H_4	680 (1500)×2 680 (1500)×4					x* x*							0-0.5	< 1			x		278 < T < 312	Avg			7.5	x		Teleoperator Retrieval System (TRS) Proj. Overview to Aero. + Space Engr'g Board (ASEB), by MSFC, 1-12-78, Ref. 11.
		RCS	N_2	49 (108)×4			x	x								0-0.5	< 1				x			160<T <320	Avg		x		
21	Initial Single Stage OTV (561), Unmanned, (Also - APOTV) One 15000 LBF Eng	MPS Oxid.	LO_2	21550 (47506)		x				x		111.7				0-1	< 1		x			90	Cryo			< 12.4 (e)	x		Veh. Data Book, 4-26-77, Boeing D180-20242-3, p. 71; NASA TMX 73394, p. 15-16, 4-77, MSFC, p. 37-41; NAS9-29676 GDC, 1-74; Ref 12, Ref 13 (Ref 14), Ref 15, Ref 16.
		MPS Fuel	LH_2	3990 (8797)		x			x*			119							x			20	Cryo			<2.3	x		
		RCS	N_2H_4	90 (198)			x	x				2282				0-1	< 1			x		278 < T <312	Avg						
22	Common Stage OTV (117E/117E), Manned (POTV), 1st Stage 4 ea 20000 LBF Eng., 2nd Stage 2 ea 20000 LBF Eng.	MPS Oxid.	LO_2	44430 (97940)×2		x				x						0-1	< 1		x			90	Cryo			67.6 / 33.8		x	Convair Veh. Data Book, 4-26-77, Convair CASD-ASD-78-001, JSC-13567, p. 7-16, -17, -20; JSC-11568, p. VI-E-21, 22, F-6, 8-31-76, Ref. 12, Ref. 17, Ref. 2.
		MPS Fuel	LH_2	8280 (18255)×2		x			x*										x			20	Cryo			5.6 / 5.6	x (e)		
		RCS	N_2H_4	431 (950)			x									0-1	< 1					278 < T <312							
23	Modular Common Stage OTV 2 Stages (ea 2 LH_2 tanks, 1 LO_2 tank) (manned)	MPS Oxid.	LO_2	57300 (126400)×2	x				x*							0-1	<1					90	Cryo			50.6 / 25.3		x	Vehicle Data Book, 4-26-77; Davis H. P., Solar Power Satellite Demo. Prog. JSC Memo, 3-24-77, Ref. 12, Ref. 19. 1st Stage 4 ea 15000 LB engine, 2nd Stage 2 ea 15000 LB engine.
		MPS Fuel	LH_2	5340 (11780)×4		x			x*							0-1	<1					20	Cryo			8.4 / 4.2	x (e)		
		RCS	N_2H_4				x	x								0-1	< 1					278 < T <312	Avg						
24	OTV Common Stage (170E/170E), (Manned), 1st Stage 4 ea 20000 LBF eng., 2nd Stage 2 ea 20000 LBF eng.	MPS Oxid.	LO_2	65000 (143310)	x					x						0-1	< 1		x			90	Cryo			67.6 / 33.8			Vehicle Data Book, 4-26-77, Ref. 12.
		MPS Fuel	LH_2	12100 (26700)		x			x*										x			20	Cryo			11.2 / 5.6	x (e)		
		RCS	N_2H	495 (1092) 1612 (3554)				x								0-1	< 1			x x		278 < T 312	Avg Avg						

x* Cylindrical Tank With Ellipsoidal Bulkheads

(I) = inlet, (e) = exit

TABLE A-2. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

ORBITAL TRANSFER VEHICLES (OTV'S)

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature Liquid		Temperature Gas		Flow Rate			Comment/Reference
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m²	Category	kN/m²	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m²-sec	< 100 kg/m²-sec	> 100 kg/m²-sec	
25	OTV - 1-1/2 Stage (85K/220K/220K), 5 LH_2, 5 LO_2 Tanks, (Geo Assy), 4 ea. 20000 LBF Engines	MPS Oxid.	LO_2(4)	85550 (188500)×4	x					x						0-1	< 1					90	Cryo			67.6			(4 Init. Stage LO_2 Tanks, 1 Final Stage LO_2; 4 Init. Stage LH_2, 1 Final Stage LH_2). Vehicle Data Book, 4-26-77; D180-20242-4, p. 259; D180-20242-3, p. 79-83; JSC 11568, D180-2024202, p. 522, SPS Concepts, Mid-Term Briefing Boeing 6-7-76; Ref. 12, Ref. 20, Ref. 14, Ref. 21, Ref. 2.
		MPS Fuel	LH_2(4)	14250 (31420)×4	(x)	x			x*													20	Cryo			11.3	x(e)		
		MPS Oxid.	LO_2	32500 (71650)		x				x						0-1	< 1					90	Cryo			67.6			
		MPS Fuel	LH_2	6060 (13360)		x			x*													20	Cryo			11.3	x(e)		
		RCS	N_2H_4													0-1	< 1					278 < T <312							
26	OTV-475T (520K/520K) (1st + 2nd Stage Tanks Similar +1/2 Stage) Stage 1: 11 ea. 20K LBF Eng. Stage 2: 8 ea. 20K LBF Eng. Similar to #33	MPS Oxid.	LO_2	198800 (438300)×2	x					x						0-1	< 1	x				90	Cryo			< 185.9		x(e)	Vehicle Data Book, 4-26-77, Similar to Item 33. Also Ref. JSC 11568, Vol. II, p. VI-D-2-5; Ref. 12, Ref. 2. Stage 1 of 11 Engines. Stage 2 of 8 Engines, 20000 lbs/each.
		MPS Fuel	LH_2	37060 (81700)×2	(x)	x			x*									x				20	Cryo			< 30.98	x(e)		
		1/2 Stage MPS Oxid.	LO_2			x	x									0-1	<1	x				90	Cryo			<67.6	x(e)		
		MPS Fuel	LH_2			x	x											x				20	Cryo			<11.26	x(e)		
		RCS	N_2H_4			x	x									0-1	< 1		x			278 < T 312	Avg						
27	Lunar Transfer Vehicle (LTV), (32K/40K), 2 ea 20000 LBF Engines	Drop Tank Oxid.	LO_2	15100 (33280)	x					x		170				0-2	1	x				90	Cryo			42.26	x(e)		Vehicle Data Book, 4-26-77; Boeing D180-2024-4, p. 227 (Loading Reported to be at 8000 kg/m²/sec) for LO_2 and 700 kg/m²/sec for LH_2. NAS8-29676, p. 227; Ref. 12, Ref. 20, Ref. 16.
		Drop Tank Fuel	LH_2	2820 (6220)	x				x*			140						x				20	Cryo			7.54	x(e)		
		Main Stage Oxid (2)	LO_2	6100 (13440)×2	x				x*			170				0-2	1	x				90	Cryo			42.26	x(e)		
		Main Stage Fuel (2)	LH_2	1140 (2510)×2		x			x*			140						x				20	Cryo			7.54	x(e)		
		APS(4)	N_2H_4			x	x									0-1	1		x			278 < T < 312	Avg						
28	Pilot Plant - Chemical OTV, (LEO Ass'y), (LTL), 20 ea 1000 lb Engines, Payload 2096K lbs	MPS Oxid	LO_2	4540 (10000)×20	x				x*							0-1	< 1	x				90	Cryo			<16.9(e)	x		Vehicle Data Book, 4-26-77; Future Space Transportation Systems Analysis Study, Final Briefing NAS9-14323, Dec. '76, D180-20242-4, p. 165, Ref. 12, Ref. 20.
		MPS Fuel	LH_2	850 (1870)×20		x			x*									x				20	Cryo			<2.8	x		
		RCS	N_2H_4			x	x									0-1	1		x			278 < T 312	Avg						

x* - Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-2. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

ORBITAL TRANSFER VEHICLES (OTV's)

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m²	Category	kN/m²	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m²-sec	< 100 kg/m²-sec	> 100 kg/m²-sec	
29	On Orbit Assembly Chemical OTV, 14000 lbs Payload, 1000 LBF Pump Fed Cryogenic Eng.	MPS Oxid.	LO_2	15300 (33720)		x					x					0-.05	< 1		x		x	90	Cryo			0.2245	x		CASD-AFS-77-005-10, p. 259, 272, Ref. 22.
		MPS Fuel	LH_2	2856 (6280)		x			x*							0-.05			x		x	20	Cryo			0.0321	x		
		RCS	N_2H_2													0-.05	< 1			x		278 < T < 312	Avg			33.8 e 5.6 i	x x		
30	SPS Geo Ass'y OTV Common Stage, Payload 500000 lbs, 60000 LBF Engines	MPS Oxid.	LO_2	187390 (413000)	x					x						0-1	< 1		x			90	Cryo			202.8 101.4		x	D-180-20242-4, p.353, 1st Stage - 4 each, 60000 lb Eng., 2nd Stage - 2 each, 60000 LBF Engine, Ref. 20.
		MPS Fuel	LH_2	34900 (77000)	x				x*										x			20	Cryo			33.5 16.9	x		
		RCS	N_2H_4				x	x								0-1	< 1			x		278 < T < 312	Avg						
31	STS Orbiter (Space Shuttle) (OTV Ser.) 2 ea OMS Eng of 6000 lbs, 38 RCS Eng of 870 lbs, 6 RCS Eng of 25 lbs.	OMS	N_2O_4	3526 (7779)×2			x		x			1758	Med			0-1	< 1		x			294 ± 10	Avg Con			10.6-11.4			Table A-1, NASA CR134911; CASD-ASD-75-001 (JCS-13967) p. 7, Vol. I, Vol. II, p. 4-11 thru 4-14; Ref. 1, Ref. 17.
			MMH	2130 (4696)×2			x		x			1758	Med						x				Avg Con			5.4-6.4			
		RCS	N_2O_4	620 (1360)×3			x	x				1689	Med			0-1	< 1			x		294 ± 10	Avg Con						
			MMH	390 (850)×3			x	x				1689	Med										Avg Con						
32	STS Orbiter External Tank (OTV), Modified W Add'l MLI	Oxidizer	LO_2	1331000 (2934000)	x				x*					152	Low	0-1	< 1	x			x	90	Cryo						Table A-2, CR134911; Ref. 1
		Prop Fuel	LH_2	222000 (489400)	x				x*					234	Low	0-1	< 1	x			x	20	Cryo						
33	Cargo Orb. Transf. Vehicle (COTV) 2-1/2 Stage LO_2/LH_2, (225 Ton Payload) 1st Stage 11 ea 20K eng., 2nd stage 8 ea. 20K Engine	MPS Oxid	LO_2	195000 (429900)×2	x					x						0-1	< 1		x			90	Cryo			< 185.9 e		x	JCS-12973, p. VI-C-32, 33; D180242-3, Vol.3, Sec 6.3, 12/31/76, Ref. 23, Ref. 14. JCS 11568, 8-31-76, p. VI-D-2-9, Grumman ECON-100, 3-77, p. 105; Ref. 2, Ref. 24.
		MPS Fuel	LH_2	36300 (80120)×2	x				x*										x			20	Cryo			< 30.98	x		
		APS/RCS	LO_2				x	x								0-1	< 1		x			90	Cryo			< 67.6 e	x		
			LH_2				x	x											x			20	Cryo			< 11.26	x		

x* - Cylindrical Tank + Ellipsoidal Ends

(i) - inlet, (e) - exit

TABLE A-2. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

ORBITAL TRANSFER VEHICLES (OTV's)

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid Type	Fluid Quantity kg (lb)	SI Large	SI Medium	SI Small	Geometry Sphere	Geometry Cylinder	Geometry Ellipse	Geometry Toroid	Operating Pressure kN/m^2	Operating Pressure Category	Vent Pressure kN/m^2	Vent Pressure Category	Accel. g	Accel. Category	Acquisition System Acceleration	Acquisition System Capillary	Acquisition System Bladder	Acquisition System Other	Temperature Liquid K	Temperature Liquid Category	Temperature Gas K	Temperature Gas Category	Flow Rate kg/m^2-sec	Flow Rate < 100 kg/m^2-sec	Flow Rate > 100 kg/m^2-sec	Comment/Reference
34	Personnel Orbit Transfer Vehicle (POTV), (75 passengers + 20 tons UP), Stage 1 - 4 ea. 15000 lb eng. Stage 2 - 2 ea 15000 LBF eng.	MPS Oxid.	LO_2	40530 (89350)×2	x					x						0-1	<1	x				90	Cryo			<63.6 / 31.8 (e)	x		JSC 11568, 8-31-78, p. VI-E-16, VI-E-17; JSC 12973, July 1977, p. VI-C-13, Ref. 2, Ref. 23.
		MPS Fuel	LH_2	7550 (16650)×2	x				x													20	Cryo			<9.1 / 4.54	x		
		APS	N_2H_4	430 (950)			x	x								0-1	< 1	x				278 < T < 312	Avg Con						JCS 11568, 8-31-76, p. VI-E-17, VI-E-18, Ref. 2
				2610 (5750)			x	x																					
35	Crew Rotation POTV, Stage 1 = 4 ea 15K Eng. Stage 2 = 2 ea 15K Eng.	MPS Oxid	LO_2	40530 (89350)	x					x						0-1	1	x				90	Cryo			< 63.6 / 31.8 (e)	x		Grumman Econ-100, Mar 77, P109, Ref. 24, JSC-11568, p. VI-E-10, 8-31-76, Ref. 2
		MPS Fuel	LH_2	7550 (16650)	x				x*									x				20	Cryo			< 9.1 / 4.54	x		
		APS	N_2H_4	2610 (5750)			x	x								0-1	1		x			228 < T < 312	Avg Con						
36 A	SPS GEO Ass'y OTV, Common Stage LO_2/LH_2 500,000 lb payload capability)(also $COTV_G$) 1 ea 90000 lb F engine	MPS Oxid.	LO_2	185200 (408300)×2	x					x						0-0.4	1	x				90	Cryo			75.8	x		Boeing D180-20242-2, Ref. 21, P 577; D180-20242-4, p. 355, Ref. 20. JSC 12973, p. VI-C-30, 31, Ref. 23; D180-20242-3, NAS9-14323, Vol. 3, Sec. 2.1, Ref. 20.
		MPS Fuel	LH_2	37050 (81670)×2	x				x*									x				20	Cryo			12.68	x		
		APS	N_2H_4	1304 (2875)×4			x	x								0-0.4	1		x			278 < T 312	Avg. Cont						
B	Space Based Common Stage Construction, 400000 KG payload capability, 1st stage of 4 ea 105K LBF engines, 2nd stage of 2 ea 105K LBF engines. I_{sp} = 470	MPS Oxid	LO_2	784400 (1729000)						x		13790	High									90				343.38		x	Boeing Report D180-22876-2, Dec. 1977, Vol. II, pp 137, Ref. 29.
		MPS Fuel	LH_2	130700 (288100)	x				x*			13790	High			0-0.5	1					20				57.89	x		
		MPS Oxid.	LO_2	784400 (1729000)						x		13790	High									90				178.66		x	Boeing Report D180-22876-5, Ref. 30. Dec. 1977, Vol. V, p. 48, p. 265 thru 272.
		MPS Fuel	LH_2	130700 (288100)	x				x*			13790	High			0-0.5	1					20				28.94	x		
		APS Thrusters (400 I_{sp})	GO_2				x	x				689.5	Low			0-0.5	1			x		100-320							
			GH_2									689.5	Low			0-0.5	1					30-20							

x* Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-2. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

ORBITAL TRANSFER VEHICLES (OTV's)

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m²	Category	kN/m²	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m²-sec	< 100 kg/m²-sec	> 100 kg/m²-sec	
36 C	JSC Cargo Orbit Transfer Vehicle, $COTV_6$	MPS Oxid.	LO_2	243000 (535700)×2	x					x						0-0.5		x				90							JSC 12973, Solar Power Satellite Concept Evaluation Vol. 2, detailed report July '76 to July '77, p. VI-C-18, Ref. 23.
		MPS Fuel	LH_2	40500 (89300)×2	x				x*							0-0.5		x				20							
			RCS				x	x													x								
37 A	Low Thrust Liq OTV Press. Fed Storable 6000 LBF Eng, Payload 6000 Lb	MPS Oxid	N_2O_4	4670 (10300)×3		x			x							0-0.4	1		x		x	297 ± 10	Avg Cont						Convair, R.H. Thomas, Concept Review, FO4701-77-C-0-0175, CASD/AFS-77-005-10, 1-18-78, Ref. 25., p. 258-268, GDC PIN 78-029, p. 2-28, 2-29, Ref. 26.
		MPS Fuel or	MMH	2310 (5100)×3			x		x										x		x								
		MPS Oxid	N_2O_4	31000 (68300)		x		x		x*						0-0.05	< 1		x		x	297 ± 10	Avg			12.9	x		
		MPS Fuel	MMH	16000 (35300)		x		x											x		x		Cont			6.45	x		
B	Low Thrust Liquid OTV, Pump Fed Storable 700 LBF Engine, Payload - 6000 Lb	RCS	N_2H_4													0-0.05	< 1			x									
38	Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS), Payload 10000 Lbs, one 20000 Lb Eng. Isp 460	MPS Oxid.	LO_2	18540 (40870)		x				x						0-2	1		x			90				16.9	x		NASA TMX-73394, Orbit transfer systems, MSFC, 1977 p. 18-21, 42-48, Ref. 15; Ref. 27, p. 154 of Ref. 28.
		MPS Fuel	LH_2	3460 (7630)		x			x*							0-2	1		x			20				2.8	x		
		RCS														0-2	1			x									
39	LTL Orbital Veh. Based on OMS Components, 2 ea OMS Engines @ 6000 LBF, 38 ea RCS Engines of 870 LBF, payload 6000 to 24000 Lbs	MPS Oxid	N_2O_4	3520 (7770)×2			x		x			1760							x			225 < T	Avg			10.6-12.9	x		OMS from STS Orbiter TBL A-1 Ref. 1, (CR 134911), Config. from GDC PIN 78-029, p. 2-7, Ref. 26.
		MPS Fuel	MMH	2130 (4690)×2			x		x			1760	Med			0-1	1		x			< 312				5.6-6.45	x		
		RCS	N_2O_4	617 (1360)×3			x	x				1690							x										
			MMH	386 (850)×3			x	x				1690	Med			0-1	1		x										

x* - Cylindrical Tank + Ellipsoidal Ends

(i) - inlet, (e) - exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

SPACE PLATFORMS/SPACE STATIONS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: < 100 kg/m²-sec	Flow Rate: > 100 kg/m²-sec	Comment/Reference
40	Propellant Depot, 1.5M, 4 Dual Propellant Tanks (4×425K Lbs) (1986)	Orbit Maint.	GO_2									140	Low			0-0.2	< 1		x			90							Uses boiloff from LO_2 and LH_2 tanks. Vehicle Data Books, Drowns, 4-26-77, Ref. 12.
			GH_2									140	Low			0-0.2	< 1		x			20							
		RCS (Attitude Control)	GO_2									140	Low			0-0.2	< 1		x			90							
			GH_2									140	Low						x			20							
41	Propellant Depot, 1.5M, 15 ft Dia Tanks (1986)	Orbit Maint.	GO_2		x		x					140	Low			0-0.2	< 1		x			90							Vehicle Data Book, Drowns, 4-16-77, Uses Boiloff From LO_2 and LH_2 Tanks, Ref. 12.
			GH_2		x			x*				140	Low						x			20							
		RCS	GO_2		x		x					140	Low			0-0.2	< 1	x				90							
			GH_2		x			x*				140	Low					x				20							
42	Propellant Depot, 1.5M, Dual Propellant Tanks (1986)	Orbit Maint.	GO_2									140	Low			0-0.2	< 1	x				90							Vehicle Data Book, Drowns, 4-26-77. Uses Boiloff from LO_2 and LH_2 tanks; Ref 12, AIAA/MSFC Symp. on Space Industrialization, 26-27 May 76, Ref. 31.
			GH_2									140	Low					x				20							
		RCS	GO_2									140	Low			0-0.2	< 1	x				90							
			GH_2									140	Low					x				20							
43	1.5M - Modular Tank D... (1986)	Orbit Maint.	GO_2									140	Low			0-0.2	< 1	x				90							Vehicle Data Book, Drowns, 4-26-77, uses boiloff from LO_2 and LH_2 tanks, Ref. 12. Space station systems analysis, Grumman Review Presentation 6-17-76, Ref. 32
			GH_2									140	Low					x				20							
		RCS	GO_2									140	Low			0-0.2	< 1	x				90							
			GH_2									140	Low					x				20							

x* - Cylindrical Tank + Ellipsoidal Ends

(I) = Inlet, (e) = exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

SPACI PLATFORMS/SPACE STATIONS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: <100 kg/m²-sec	Flow Rate: >100 kg/m²-sec	Comment/Reference
44	Pilots SPS Plant (15M W), 347 X 374 M, 340,000 KG (750,000 LBS) (1987)	Orbit Maint.	O_2 or N_2	812 (1970) per year 29000 (63900)/yr												0-0.2	<1												Boeing D180-20242-4, P245, 247, Ref. 20.
		RCS	LA_r	121000 (266800)/yr												0-0.2	<1					29				1.7×10^{-5}/SEC	x		
45	Low Earth Orbit Space Station (Unitary, 12 Man), (208,000 Lbs), 24 ea, 50 lb. Thrusters	Orb. Maint. & Art. G Manv.	N_2H_4	2840 (6250)			x	x				1380	Med			0-1	< 1			x		275-375							Boeing D180-20242-2, P45, PP 3.2-3 and 3.2-4; MDC GD 634, P 99-123, Vol. 1, Book IV, July 1970, Ref. 21, Ref. 8.
		RCS	H_2O	102 (225)×2				x	x			1380	Med			0-0.2	< 1				x								
			N_2H_4	750 (1650)				x	x			1380	Med								x								
			Bio-waste					x	x			1380	Med																
46	Low Earth Orbit Space Station, (Modular), (12 man, 252,000 lbs) (1985)	Orbit Maint.	LO_2	381000 (840000)/yr												0-0.2	< 1												Boeing D180-20242-2, P 44, 45, Ref. 21.
			LH_2	63500 (14000)/yr																									
		RCS	LA_r	1.36×10^6 (3×10^6)/yr												0-0.2	< 1												
47	Space Base, (60 man), (wt ~ 912,000 lbs)	Orbit. Maint.														0-0.2	< 1												Boeing D180-20242-4, Ref. 20, MSFC Preliminary Study, Feb. 1977, Ref. 33
		RCS																											
48	GSS-Geosyn. Sp. Sta. (8 Man, Earth Applic. + Science), (283,000 - 348,000 lb) 1988)	Orbit. Position Adjust																											D 180-20242-2, P 107, Ref. 21
		RCS	N_2H_4	1840 (4050)												0-0.2	< 1					278 < T <312							

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

SPACE PLATFORMS/SPACE STATIONS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: < 100 kg/m²-sec	Flow Rate: > 100 kg/m²-sec	Comment/Reference
49	Orbiting Lunar Sta. (12 Man), (277,000 - 386,000 Lbs) (1992)	Orbit Position Adjust																											D 180-20242-4, P 194, Ref. 20
		RCS	N_2H_4	1430 (3150)												0-0.2	< 1					278 < T < 312							D 180-20242-2, Ref. 21.
50	Low Earth Staging Depot/Laboratory (72 man) (1987)	Orbit Maint.														0-0.2	< 1												D 180-22876-5, P150-155, Dec 1977, Ref. 30.
		RCS														0-0.2	< 1												
51	Low Earth Manufacturing Facility (1989)	Orbit Maint.														0-0.2	< 1												SD-77-AD-0094, Ref. 34.
		RCS														0-0.2	< 1												SD-78-AD-0055-2, 4-14-78, Ref. 35
52	Low Earth 300 NM Const. Base (Fac.) (1.8×10^6 lbs to 20.86×10^6 lbs) (1987)	Orbit Keeping (per yr)	N_2H_4 or Argon	160000 (352700) 9000 (19800)												0-0.2	< 1					278 < T < 312							D 180-20242-2, P 382; Grumman Econ E100, 3-77, P 93, (LO_2/LH_2 @ 800 KG/day), (Ref. 2, Ref. 24).
		RCS (per yr)	N_2H_4 or Argon	31×10^6 (68×10^6) 1.4×10^6	(3.1×10^6)																	278 < T < 312							D180-22876-5, 12-77, P 34, 35 (Fig. 2-34), P120, Ref. 30.
53	Geosyn. Space Pwr Satellite, 10,000 MW, 86×10^6 KG (190×10^6 lb) to 97 $\times 10^6$ KG	OAS + RCS	LAr	40900 (90200)					x*							0-0.2	< 1												D180-22876-2, Vol. II, 12-77, P 59, Ref. 29.
		Equinoctal Occultation Control	LO_2	1280 (2819)			x	x								0-0.2	< 1					90							D180-22876-5, Vol. V, 12-77, P. 59, Ref. 30.
			LH_2	214 (471)			x	x														20							
54	1200/500 MW Space Power Satellite (Sta.), 19.5 to 22 Million Lb (Prototype Satellite) (1990-1995)	Equinoctal Oscillation Control	LO_2	1000 (2200)												0-0.2	< 1					90							D180-20242-2, P373, 374, 380, 536, Ref. 21.
			LH_2	to 5000 (11000)/yr																		20							

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

SPACE PLATFORMS/SPACE STATIONS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m²	Category	kN/m²	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m²-sec	< 100 kg/ m²-sec	> 100 kg/ m²-sec	
55	Geosynchronous Satellite Maintenance Veh. 2-4 Men, 1-2 weeks 6710 KG (14790 lbs) (1996)	OAS														0-0.2	< 1												D180-20242-2 (NAS9-14323), Vol. 2, 31 Dec. 1976, pp. 73-92, Ref. 21.
		RCS														0-0.2	< 1												
56	LEO Solar Powered Propellant Processor, ($H_2O - O_2 + H_2$) (~1×10⁷ lbs Plant Weight)	Drag Adj + ACS or	GO_2	160000 (352000/yr)	x			x	x							0-0.2	~ 1								Wide Range				CASD-ASD-78-001, JSC 139, p.7-14, 7-56, JSC Orbital Prop. Handling P15-18, Ref. 17.
			GH_2	32000 (70000/yr)	x			x																					
		Drag Adj + RCS	Excess GO_2	1030000 (2270000/yr)	x			x								0-0.2	< 1								Wide Range				
57	Power Satellite GEO Synchronous Orbit Support Sta. (~600,000 lbs) (1991)	RCS														0-0.2	< 1												
58	Radar Platform (OOA)	RCS	N_2H_4																										CASD-AFS-77-005-10, 1-17-78, Ref. 25.
59	Earth Observation Platform															0-0.2	< 1												Eo-8A, LANDSAT D, NASA Prel. Payload Descriptions, Vol. 1, Automated Payloads, July 1975, P 5-1, Ref. 9.
60	Geosynchronous Construction Base, Mass: 855,000 KG Crew Size 65-67	Orbit Keeping														0-0.2	< 1												D180-22876-5, Dec. 1977, Figure 2-33, P 33, P 145, Ref. 30.
		RCS	N_2H_4 or Argon	3500 (7700)												0-0.2	< 1												

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = Inlet, (e) = Exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

SPACE PLATFORMS/SPACE STATIONS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: < 100 kg/m²-sec	Flow Rate: > 100 kg/m²-sec	Comment/Reference
61	Geosynchronous Orbital Antenna Farm (OAF)	Orbit Position Adjust																											G.E. Stationary Platform, Public Service/Antenna Farm, P 20 of OTV (Concepts and Questions).
		RCS																											GDC Briefing 1502 8000J0604A, June 1976, D.A. Heald, Ref. 26, Ref. 36, pgs 20-29, Vol. 15, No. 9, Astro. & Aero., 9-77, COMSAT, Ref. 37.
62	25-35 KW Power Module (Small Power Module)	Backup RCS, Moment-um Dump	GN_2																										NASA S-77-5339, Small Power Module Status Review/ DR Kraft, 10-77, Ref. 38.
63	100 KW Power Module	Backup RCS																											Auxiliary Payload Power Sys. Study, DPD 533, Configuration Definition. Ref. 39. AAPS Final Rpt, MDC 1466, 2-29-76 Ref. 40.
64	Antenna Farm Construction Platform	RCS																											Dodecagon Structure Assy. Concept; Viewgraph, P 20, Ref. 41.
65	Orbital Antenna Farm																												Page 20 of Ref. 41. Page 19 of Ref. 36.
66	Logistics Tank, 55K Gross Wt. 55994 Lbs (Used With OTV's)		LO_2	20000 (44000)	x			x*										x				90				<8000(l)		x	NASA/Convair CR-134911, P 4-43, 5-76, CASD-NAS77-025, 12-77, P 1-2, Ref. 1. D.A. Heald, "Orb. Prop. Handling & Stor. Sys. for Lgr Space Programs," JSC 13967, 4-14-78, Ref. 42
			LH_2	3600 (8000)	x			x*										x				20				< 600(l)		x	
			N_2H_4	181 (400)		x	x												x			278 < T							
			He	54 (120)		x	x													x		< 333							

x* = Cylindrical Tank + Ellipsoidal Ends

(l) = inlet, (e) = exit

TABLE A-3. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m^2	Category	kN/m^2	Category	g	Category	Acceleration	Capillary	Bladder	Other	Liquid K	Liquid Category	Gas K	Gas Category	kg/m^2-sec	< 100 kg/ m^2-sec	> 100 kg/ m^2-sec	
67	Logistics Tank, 100K		LO_2	36700 (81500)		x			x*													90				< 8000(i)		x	Convair Vehicle Data Book, 4-26-77, Ref. 1,
			LH_2	6710 (14800)		x			x*													29				< 600(i)		x	
68	Orbital Service Module	Station Keeping																											GDC CM78-007, Orbital Service Module (OSM) Systems Analysis, 12-22-77, Ref. 43.
		RCS	GN_2																										
69	Water Delivery Tank	Delivered Water RCS	H_2O	4.25×10^6 (9.36×10^6)				x				344.8								x		274 -370							JSC 13967, CASD-ASD PUC Ref. 42.

x* - Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-4. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

AUTOMATED SPACECRAFT

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid: Type	Fluid: Quantity kg (lb)	Size: Large	Size: Medium	Size: Small	Geometry: Sphere	Geometry: Cylinder	Geometry: Ellipse	Geometry: Toroid	Operating Pressure: kN/m²	Operating Pressure: Category	Vent Pressure: kN/m²	Vent Pressure: Category	Accel.: g	Accel.: Category	Acquisition System: Acceleration	Acquisition System: Capillary	Acquisition System: Bladder	Acquisition System: Other	Temperature Liquid: K	Temperature Liquid: Category	Temperature Gas: K	Temperature Gas: Category	Flow Rate: kg/m²-sec	Flow Rate: < 100 kg/m²-sec	Flow Rate: > 100 kg/m²-sec	Comment/Reference
70	MSR-MARS Sample Return (Automated Planetary) Spacecft (1989)	Orbit Adj.																											
		RCS																											PP 309-322, D180-20242-2, Ref. 21.
71	Automated Geosynchronous Satellite (Disaster Comm, Veh. Tracking, Energy Monitor) (1986)	Orbit Adj																											
		RCS																											Aerospace Co HTR-76(7365)-1, PP 46,47,48, 4-76, Ref. 44
72	Low Earth Orbit Large High Energy Observatory - 3000 lb (1980-1985)	Orbit Alt Maint.	H_2H_4	113 (250) × 4		x	x						Low			0.0.1	<1					278-312	Avg						
		RCS	N_2	91 (200)		x	x						High			0.01	< 1												Payload Descriptions, Vol. I, P2-109, July 1975, Table A-17 CR 134911, Ref. 9
73	Nuclear Waste Disposal Vehicle (30 KLB Payload) (1991)	Orbit Direction																											
		RCS																											Aerospace Co, ATR-76-(7365)-1, PP 84,85, 4-76, Ref. 44
74	Advanced TV Broadcast (Geo-synch) Satellite 14000 Lb	Orbit Adjust.																											
		RCS																											Aerospace Co. ATR-76(7365)-1, P 53, April 76, Ref. 44.
75	High Resolution Earth Mapping (~200 NMI Orbit), Antenna/Satellite Wt 110,000 Lb, 16×100 ft Antenna	Orbit Maint.																											
		RCS																											Aerospace Co. ATR-76(7365)-1, Vol. II, P 58, April 76, Ref. 44.
76	Adv. Resources/ Polution Obs., Wt 30000 Lb, 500 NMI Orbit (Sun Sync) Size: 30 × 60 ft	Orbit Maint.																											
		RCS																											Aerospace Co ATR-76(7365)-1, Vol. II, P 57, April 76, Ref. 44.

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

TABLE A-4. IDENTIFICATION OF CANDIDATE VEHICLE RECEIVER TANKS

AUTOMATED SPACECRAFT

Identification No.	Candidate Vehicle Receiver	Tank Function	Fluid		Size			Geometry				Operating Pressure		Vent Pressure		Accel.		Acquisition System				Temperature				Flow Rate			Comment/Reference
																						Liquid		Gas					
			Type	Quantity kg (lb)	Large	Medium	Small	Sphere	Cylinder	Ellipse	Toroid	kN/m^2	Category	kN/m^2	Category	g	Category	Acceleration	Capillary	Bladder	Other	K	Category	K	Category	kg/m^2-sec	<100 kg/ m^2-sec	> 100 kg/ m^2-sec	
77	Coastal Anti-Collision Radar, 1000 × 1000 Ft, Synch Equat. Orbit	Orbit Adjust																											Aerospace Co. ATR-76(7365)-1 Vol. II, P. 46, April 1976, Ref. 44.
		RCS																											
78	Intentionally Left Blank																												
79	Intentionally Left Blank																												
80	Geosynchronous Manned Mission Module																												Adv. Study Prog Meeting, R.O. Piland et al, JSC, 8/30-31/77, Ref. 45.
81	Manned Utility Spacecraft (MUS)																												Same as above.
82	Orbital Construction Demonstration Article Mass 37073 KG (81790 Lb)	ACS (per 6 mo)	N_2H_4	7580 (16700)	x							1400				0–0.2	<1					278 < T <312	Avg						Orbital Construction Demonstration Study, Final Report, N55-DC-RP-008, 12-1-76, Grumman, Ref. 46.
		Pressurization	GHe	39 (86)		x	x									0–0.2	<1												
		Ion RCS Thrusters (per 6 mo)	Ar	523 (1152)		x	x									0–0.2	<1												

x* = Cylindrical Tank + Ellipsoidal Ends

(i) = inlet, (e) = exit

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APPENDIX B

SPACECRAFT ACCOMMODATIONS

Equipment Mounting and Software

- The Spacelab double racks (38") can hold up to 580 Kg (1276 lbs) of equipment with a maximum volume of 1.75 m^3 (62 ft^3)
- Equipment mounted in racks may be as large as:
  - Width (double rack) = 94 cm (36.7 in)
  - Height = 149 cm (58.1 in)
  - Depth = 61 cm (23.8 in)
- Rack accessories include: interfaces with Spacelab data collection/processing/transmission systems; power outlets; connectors for forced-air cooling of flight experiment equipment.
- Equipment unsuitable for rack mounting can be mounted directly to the floor of the Spacelab.

Electric Power

Spacelab electrical power is routed to flight experiments through experiment switching panels which can be mounted to individual racks, or under the Spacelab floor to service floor-mounted equipment.

- Electrical power: 28 v DC; 115/200v AC @ 400 H_z/3 phase
- Power conditioning equipment
- Power available for payload and mission-dependent equipment in the Module is payload dependent

Ascent /Descent: 1 kw (continuous)
1.5 kw (peak)

On-orbit (all module configurations): 2.6 to 3.5 kw (continuous)
7.2 to 8.1 kw (peak)
160 to 300 kwh of energy available/mission

Thermal Control

The Spacelab Thermal Control Subsystem (TCS) consists of four thermal control loops:

- The mdoule cabin air loop provides conditioned air, within established comfort criteria, for the crew within the module
- The module avionics loop provides forced-air cooling for equipment mounted in racks.
- The freon loop supplies cooling to cold plates mounted on any pallets that might be carried on the flight
- The module water loop exchanges heat with the three loops already discussed and, in turn, transfers Spacelab heat into the Orbiter coolant loop. The water loop also provides cold plate cooling for Spacelab subsystems, and can accommodate an Experiment Heat Exchanger and one Experiment Cold Plate.

The TCS transfer 8.5 kw, maximum, continuously to the Orbiter. It can also accommodate peak heat loads of up to 12.4 kw for 15 minutes every 3 hours. However, for Spacelab configurations incorporating a Module, part of the water loop heat rejection capability is used to cool Module subsystems. The remaining capability may be used by experiments and other mission dependent equipment. The water loop heat rejection available to such equipment is 3.7 to 5.2 kw, continuous; and 7.6 to 9.0 kw peak. This range of values covers all possible Spacelab Module configurations. Within these limits, the remaining thermal control loops have their individual capacities to provide experiment cooling. These are:

- <u>Cabin air-loop</u> - 0.4 to 1.6 kw, continuous; 2.4 to 3.6 kw, peak
- <u>Avionics loop</u> - 3.3 kw, continuous; 8.7 kw, peak
- <u>Freon loop</u> - 8.0 kw, continuous; 11.9 kw, peak (Note: In pallet-only configurations, the freon loop exchanges heat directly with the Orbiter loop. In this case the Spacelab total heat rejection capability is 7.0 kw, continuous and 10.9 kw, peak).

The Experiment Heat Exchanger can be mounted only in experiment double rack number 4 in the Spacelab Core segment. Its nominal heat transfer capability is 4kw. However, its actual performance depends on where it is connected into the water loop on the other loads along the water loop.

The Experiment Cold Plate must also be mounted in rack 4. When both the Heat Exchanger and the Cold Plate are flown, they must be connected in series and must share the same connection into the water loop. The cold plate is 500 mm × 390 mm × 4.4 mm and has a 70 × 70 mm hole pattern. When filler is used to improve the contact between the experiment and plate, the conductance will be 0.07 watts per cm^2 per °C. Without filler the conductance is approximately 1.0 watt per °C per bolt area

Digital Data

The Spacelab Modules's data management system enables the collection, processing, recording, on-board display, and transmission of low-rate digital data (including digitized analog) received from the flight experiments at 1 Mb/sec or less. Experiment produced data with rates between 1 Mb/sec and 16 Mb/sec can be stored and/or transmitted to the ground, but these data cannot be processed or displayed on-board the spacecraft.

Depending on the needs of the experiment, low-rate digital data can be:

- Processed and analyzed on-board the spacecraft using experiment supplied software
- Formatted and displayed for review and analysis by the Payload Specialists on-board the spacecraft.
- Annotated with voice recordings and/or time marks.
- Recorded on data tapes or transmitted to the ground.

High-rate digital data can be recorded on tapes or transmitted to the ground. Additional information about low-rate data and high-rate data follows:

Low rate (≤ 1 Mb/sec)

- Remote Acquisition Units (RAU's) receive experiment data and deliver them to the Spacelab data management system for computer processing display, storage and/or transmission to the ground. RAU's may be mounted in equipment racks, under the Spacelab floor, and inside an Airlock.

- RAU's provide an 8 bit resolution, analog-to-digital conversion capability, and can accept serial digital data.

- The RAU data acquisition function is under experiment-supplied software control

- Signal conditioning equipment will be available, as required, to interface experiment hardware with the RAU

- A data processing computer (Mitra 125 S, 64K core, 16 bitwords, 3.5×10^5 operations/sec) which can analyze experiment data onboard the spacecraft, format data for on-board display, and format data for transmission to the ground will be on-board.

- A mass memory unit for storage of software will be on-board

- A data display unit and keyboard which permit on-board review of experiment data will be available on-board.

- Data may be stored on tape or transmitted to the ground at up to 64 Kb/sec.

High rate (62.5 Kb/sec to 16 Mb/sec)

- On-board data recording and transmission to the ground will be provided.

- Experiment data can be automatically annotated with time references, and digitized voice signals via the high data rate system.

Voice Data and Communications

The spacecraft provides voice recording, voice communications between the spacecraft and the ground, and an intercom system within the spacecraft.

Analog Data

Investigations which produce analog data will have two basic options. If feasible the analog data should be digitized, to take full advantage of the Spacelab Module's extensive digital data recording processing, and transmission capabilities.

Analog data, that cannot be digitized, may be recorded on-board and/or transmitted to the ground. Planned capabilities include:

- Analog to digital conversion. Data may be stored, transmitted to the ground, or processed and displayed on-board the spacecraft. Conversion rates are programmable at 1,10, and 100 samples/sec.

- Analog-down-link transmission for up to 85% of the mission. Data may be real-time or tape records (3 H_Z to 4.2 Mh_Z).

Video Data

An Orbiter-to-Spacelab interface has been provided that would enable the collection monitoring and recording of black-and-white and color video signals. Also, black-and-white video data could be transmitted to the ground, one channel at a time, through this interface. Spacelab video capabilities currently under consideration include:

- Black-and-white video cameras

- Black-and-white video monitor

- Black-and-white video tape recorder plus tapes

- Camera Control - Provides automatic and manual camera/recorder/monitor switching and tags video records with time and experiment identification.

- Camera Timer - Provides automatic time-controlled activation/deactivation of cameras and recorders.

- Accessories - Mounting brackets, lens assortments, cabling, and remote controls.

- Color video camera

- Color video montior

- Color video tape recorders plus tapes

Table B-1 summarizes the Spacelab module environment that the fluid behavior experiment designs will consider.

Table B-1 Spacelab Module Interior Environment

Gravity

On-orbit operations: 10^{-4}g to 10^{-5} (spacecraft drifting)
10^{-3}g (spacecraft maneuvering)

Other phase. 3g (maximum, launch); 1.5g (maximum, reentry)

Acoustic

On-orbit operations: < NC-50 curve; overall noise level < 75 db
Incidental equipment-produced noise is dependent on equipment included in the final payload complement.

Other phases: 137 db maximum - during launch (ref: 20 $\mu N/m^2$)

Vibration

On-orbit operations: Incidental equipment produced vibration is dependent on equipment included in the final payload complement.

Other phases: Launch and ascent
- Sinusodial $\pm$ 0.25g (5-35 Hz) - System level
- Random - 3.3g RMS (composite) - Equipment racks. 4.77g RMS (composite) - floor mounted. For ~6 seconds after engine ignition.

Temperature

On-orbit: (Adjustable) Min = $18^{\circ} \pm 1^{\circ}C$ (~ $64.6^{\circ}F$)
Max = $27^{\circ} \pm 1^{\circ}C$ (~ $80.6^{\circ}F$)

Other phases: Within the on-orbit range.

Humidity

On-orbit: 25% to 70% RH (not adjustable).

Other phases: Within the on-orbit range.

Atmospheric pressure

On-orbit: Total pressures = 14.7 psia (O_2/N_2)
Composition: O_2 = 21%
N_2 = 79%

Other phases: Approximately the same as on-orbit

Cleanliness

On-orbit: Maintained by 280 micron filters

Other phases: Most NASA operations - class 100,000

Radiated Emissions

On-orbit: Narrow band - 0.1 V/m (peak at 1 to 10 MHz & S-band)
Broad Band - 90db $\mu V/m/MHz$ (peak in 100 MHz range)
This is the estimated upper limit.

Other phases: Not presently available.

Magnetic Environment (AC)

On-orbit: 146 db above a pico-tesla at 30 Hz, decreasing linearly to 80 db above a pico-tesla at 50 KHz.

Other phases: Within the on-orbit range.

Radiation

On-orbit: Exposure to cosmic and trapped radiation, and solar flare particle fluxes through a minimum shielding of 0.45 g/cm^2. Actual shielding depends on location and configuration of equipment.

Lighting

On-orbit: 200-300 lumens/meter^2, but increases to 400-600 lumen/meter^2 at Spacelab workbenches.

Other phases: Lights will be turned off.

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