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16. Abstract

Technology gaps and system characteristics critical to cryogenic and non-cryogenic in-orbit fluid transfer were identified. Four different supply systems were conceptually designed as Space Shuttle payloads. These were; (1) Space Tug Supply - LH_2, LO_2, N_2H_4, He - linear acceleration for liquid acquisition with supply module and Tug separated from Shuttle, (2) Tug supply using Orbiter drag, (3) Orbiter Supply - N_2O_4, MMH, He, H_2, O_2 - surface tension screens, (4) Multiple Receivers Supply - Solar Electric Propulsion Stage, Hg, diaphragm - HEAO B, LHe, paddle fluid rotation-Satellite Control Section, N_2H_4, screens. It was found that screens had the best overall potential for low weight and simplicity, however, thermal problems with cryogenics still need final resolution. A paddle system also has many advantages and should provide a good back-up to screens, however, feasibility demonstration is needed. For Tug transfer at 296 Km or lower, use of Orbiter drag is best, and to minimize residuals, long, small diameter tankage should be used. Unresolved problems exist with low-g chilldown and filling; prevention of excessive fluid loss and insuring that receiver screens are filled. In-orbit supply can increase Shuttle performance by 75%. Supply to recover a disabled Orbiter can save \$472M. Supply of H_2 and O_2 can significantly extend spacelab missions. In-orbit supply can increase Tug performance significantly for most missions. For a Mars Sample Return mission, payload can be increased up to 108% and 83% respectively for reusable and expendable Tugs. Applying in-orbit supply and low cost design principles to two Mars Missions results in potential savings of \$120M.

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FOREWORD

This report was prepared by the Convair Division of General Dynamics Corporation in partial fulfillment of Contract NAS 3-17814. The contract was administered by the Lewis Research Center of the National Aeronautics Space Administration, Cleveland, Ohio. The contract period covered by this report is May 1974 through February 1976. The NASA-LeRC Project Manager was Mr. John C. Aydelott.

All data, except in Appendix A, are presented with the International System of Units as the primary system and English units as secondary. The English system was used for the basic calculations. The data in Appendix A is entirely from other reports and was therefore used in its existing form.

Three companion reports were published in December 1974 covering the literature survey portion of this contract. These reports are NASA CR-134746, "Low-G Fluid Behavior Technology Summaries;" NASA CR-134747, "Cryogenic Thermal Control Technology Summaries;" and NASA CR-134748, "Fluid Management Systems Technology Summaries." An executive summary covering the overall program results is published as NASA CR-135020.

In addition to the project manager, Mr. John A. Stark, a listing of the Convair personnel which contributed to the study is presented below, including their primary areas of contribution.

R.E. Downs	-	Receivers Configurations and Characteristics Investigations and Benefits Analyses
M. D. Walter	-	Design
R. L. Pleasant	-	Thermal Analysis
R. D. Bradshaw	-	Low-G Fluid Dynamics Technology Evaluations
M.H. Blatt	-	Low-G Liquid Acquisition Technology Evaluations
B. J. Campbell	-	Instrumentation Technology Evaluations
K. E. Leonhard	-	Cryogenic Thermal Control Technology Evaluations

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SUMMARY

Reported here are the final results of a program to identify technology gaps, system characteristics, components, and operations critical to the design and performance of efficient and predictable in-orbit fluid transfer systems. The results of this program could significantly contribute to increased use and applications of current and future space systems. The initiation of the program was timely in that shortcomings and deficiencies in the technologies necessary to support effective in-orbit fluid transfer are identified in time to allow for their resolution in a planned and orderly manner.

The primary problem of transferring fluids in space is the absence of unbalanced body forces to provide a natural orientation of the liquid and vapor in a tank. This results in requirements for systems to orient or collect the liquid to be transferred and for receiver tank vent systems that prevent excessive liquid loss.

For purposes of this study, transfer systems are defined in terms of the method used for liquid acquisition in the supply, however, a complete system consists of supply storage, transfer lines and up to three different receivers; as well as auxiliary support systems such as required for tank pressure control and venting. Both cryogenic and non-cryogenic fluids are included and supply modules are assumed to be payloads of the Space Shuttle manned transportation system.

The overall study was made up of the individual tasks listed below.

a. Comprehensive literature analysis to provide a sound base for all subsequent work.

b. Technology evaluation, in general terms, of the adequacy of existing technology to design cryogenic and noncryogenic in-orbit fluid transfer systems.

c. Receiver configurations and characteristics definitions to determine which receivers would need or could benefit from in-orbit fluid transfer, along with their relevant characteristics and specific transfer benefits.

d. Transfer systems studies to conceptually design overall transfer systems that appear most likely to provide efficient and predictable in-orbit supply of representative receivers determined in c. As a result of initial definition and screening, four different transfer systems were conceptually designed, as listed below.

System 1 Space Tug Supply (LH_2, LO_2, N_2H_4, He) with linear acceleration of supply module and Tug separated from the Shuttle Orbiter.

System 2 Space Tug Supply (LH_2, LO_2, N_2H_4, He) with linear acceleration from Shuttle drag with the Tug attached to the Orbiter.

System 3 Space Shuttle Orbiter Supply (N_2O_4, MMH, He, H_2, O_2) using surface tension screens for liquid acquisition.

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System 4 Multiple Receivers Supply of the Solar Electric Propulsion Stage (Hg) using a diaphragm, Large High Energy Observatory-B (LHe) using a paddle for liquid acquisition, and the Satellite Control Section (N_2H_4) using surface tension screens.

e. Systems evaluation to determine technology requirements and programs necessary for final design and development of the specific transfer systems defined in d.

f. Analysis of Shuttle/Tug fluid transfer benefits as to specific performance improvements and potential cost savings of in-orbit fluid supply using supply systems 1, 2 and 3 defined in d.

Major study results are presented below.

a. There are a large number of existing and future space systems which would need or could benefit from in-orbit fluid transfer. In general, cost effectiveness (reduced cost, increased performance and/or mission capability) and safety are the benefits which can be realized. A representative sampling (29 receivers) indicated that liquid oxygen would be the fluid, by mass, used most in space; with hydrogen a close second. Hydrazine was used on the greatest number of different receivers and there was an average of three different fluids per spacecraft. The number of applications of cryogenics and noncryogenics was about equal.

b. In-orbit fluid supply can increase the Shuttle performance envelope by 75%. Applying in-orbit supply to recovery of a single disabled Orbiter can result in savings of \$472M. Supply of cryogenic H_2 and O_2 and some OMS/RCS fluids to the Shuttle Orbiter can extend uninterrupted spacelab missions indefinitely.

c. Tug performance can be significantly increased for most missions. For example, for in-orbit supply of the Tug-only, for a Mars Sample Return Mission, payload can be increased by 35% for a reusable Tug and by 53% for an expendable Tug. With supply of both the Tug and Orbiter, respective Tug payload increases of 108% and 83% are possible. Applying low cost design concepts to two Mars Sample Return Missions, assuming supply of a reusable Tug, results in estimated savings of \$120M over no supply.

d. Use of surface tension screens for low-g liquid supply has the best overall potential for low weight and simplicity for both cryogenics and noncryogenics, however, potential thermal problems with cryogenics still need final resolution.

e. A paddle rotation system appears to be a good back-up to surface tension screens. Advantages are a potential minimization of problems associated with heat transfer, mass gauging, low-g venting and vehicle disturbances, as compared to screens. Little work has been done on the paddle system and feasibility demonstration is needed.

f. For large systems such as the Space Tug, use of linear acceleration for liquid orientation has the advantage of being nearer to current state-of-the-art. A Tug supply system using Shuttle drag was found to be slightly lower in weight than one with the Tug/supply module separated from the Shuttle and accelerated by a separate propulsion system. Thus, unless transfer in orbits higher than 296 km (160 n.mi.) were required, the drag system would be the likely choice. For both cases, supply module weights are less than the baseline Space Tug supplied, allowing more payload with the transfer module than with the Tug.

g. For linear acceleration systems it was determined to be optimum to use long, small diameter tankage rather than tankage characteristic of current vehicles. Savings in liquid residuals more than offset the increased weights of the small diameter tanks. Additional work on low-g outflow could likely reduce residuals even further.

h. A significant problem, for which final solutions have not yet been demonstrated, is receiver tank chilldown and filling. Due to the low-g environment, preventing direct liquid loss at receiver vents may be a problem. For most of the cryogenic receivers a non-vent chilldown is impractical. Also, since the Shuttle and Large HEAO-B receivers are quite heavy, the quantity of fluid required for chilldown, even without direct liquid loss, is sensitive to the thermodynamic condition of the vent fluid (saturated versus superheated vapor). This is especially critical with helium and due to uncertainties in expected chilldown efficiency, LN_2, representing an additional fluid system, must be used for pre-chill of the HEAO-B.

   Another potential problem is to insure that screen surface tension devices, such as exist in the Shuttle N_2O_4 and MMH tanks and the Satellite Control Section N_2H_4 tank, are full at the completion of transfer. Premature screen wicking and trapping of non-condensible vapor are problems for which solutions have not yet been developed.

i. Due to its very low heat of vaporization and surface tension, as compared to other cryogenics, helium represents potentially unique problems needing further investigation; primarily in relation to use with surface tension screens and in receiver tank chilldown.

A listing is presented below of the most pertinent technology work recommended to develop in-orbit fluid transfer capability.

a. Receiver Chilldown and Fill (Cryogenic and Noncryogenic)
   1. Analytical Model Development
   2. One-g Thermodynamic Testing
   3. Drop Tower Testing
   4. One-g Prototype Demonstration of Practical System(s)
   5. Orbital Demonstration

b. Surface Tension Screen Systems (Cryogenic and Noncryogenic)
   1. Develop Low-g Refill Capability for Supply Channels and Receiver Channels and Baskets
   2. Demonstrate Compatibility With Realistic Vibration and Thermal Environments, Including Integration With Operational Type Tank Pressure Control Systems

3. Orbital Demonstration of Complete Supply System Concept

c. Paddle Rotation Liquid Orientation

   1. Demonstrate Feasibility and Generate Basic Design Data in Subscale One-g Tests
   2. Overall System Analysis and Design
   3. One-g Prototype Testing and Orbital Demonstration

d. Low-g Pressure Control - Orbital Demonstration of Bulk Heat Exchanger Type Vent System

e. Low-g Outflow to Improve Prediction and Minimization of Liquid Residuals
   1. Analytical Model Development
   2. One-g and Drop Tower Testing
   3. Orbital Demonstration

f. Investigation of Special Problems of Helium Transfer
   1. Demonstrate Compatibility With Screen System
   2. Investigate Practicality of Other Than Liquid Transfer
   3. Develop Methods for Improved Thermal Chilldown Efficiency

g. Low-G Boiling, Condensation, Convection, and Two-Phase Flow Heat Transfer - Orbital Experimentation Required

h. Orbital Demonstration of Low-G Mass Gauging

i. Orbital Demonstration Test of Overall Transfer Concept
   1. Prototype Hardware One-g Test
   2. Instrumentation/Observation Demonstration
   3. Development of Orbital Test Techniques

1

INTRODUCTION

This report presents the final results of a program to identify technology gaps, system characteristics, components, and operations critical to the design and performance of efficient and predictable in-orbit fluid transfer systems. The results of this program could significantly contribute to increased use and applications of current and future space systems. The initiation of the program was timely in that shortcomings and deficiencies in the technologies necessary to support effective in-orbit fluid transfer are identified in time to allow for their resolution in a planned and orderly manner.

The primary problem of transferring fluids in space is the absence of unbalanced body forces to provide a natural orientation of the liquid and vapor in a tank. This results in requirements for systems to orient or collect the liquid to be transferred and for receiver tank vent systems that prevent excessive liquid loss.

For purposes of this study, transfer systems are defined in terms of the method used for liquid acquisition in the supply, however, a complete system consists of supply storage, transfer lines and up to three different receivers; as well as auxiliary support systems such as required for tank pressure control and venting. Both cryogenic and non-cryogenic fluids are included and the supply module is assumed to be a payload of the Space Shuttle manned transportation system.

The overall study was made up of the individual tasks listed below.

a. Literature Analysis — This task was designed to provide a sound base for all subsequent in-orbit fluid transfer work and consisted of the collection, screening, summarization and categorization of the available data on low-g fluid behavior, cryogenic thermal control and fluid management systems technology. Listings of reports reviewed, along with detailed summarizations and categorizations of the most pertinent ones, are contained in NASA CR-134746, "Low-G Fluid Behavior Technology Summaries;" NASA CR-134747, "Cryogenic Thermal Control Technology Summaries;" and NASA CR-134748, "Fluid Management Systems Technology Summaries." These reports were published as part of the current contract in December 1974.

b. Technology Evaluation — In this task the technology within each category, as determined in the Literature Analysis, was evaluated in general terms with respect to defining the adequacy of existing technology to design cryogenic and non-cryogenic in-orbit fluid transfer systems. Results are presented in Section 2.0.

c. Receiver Configurations and Characteristics — Here, spacecraft and vehicles were identified which need or could benefit from in-orbit fluid transfer. Specific characteristics and benefits of in-orbit fluid transfer were then determined for selected vehicles. Selection was on the basis of obtaining receivers representative of different classes (manned, unmanned, etc.) using different fluids and which are potentially viable hardware for eventual usage in space. Twenty-nine different receivers were so identified. Overall results are reported in Section 3.0. Detailed receiver characteristics, including fluids and quantities required for resupply, are presented in Appendix A. As a result of the receivers studies, supply systems for the Space Tug (LH_2, LO_2, N_2H_4, He), Space Shuttle Orbiter (N_2O_4, MMH, He, H_2, O_2), Solar Electric Propulsion Stage (Hg), Large High Energy Observatory - B (LH_e) and Satellite Control Section were chosen for conceptual definition.

d. Transfer Systems Studies — This task consisted of the conceptual definition of overall transfer systems that appeared most likely to provide efficient and predictable in-orbit supply of the five receivers chosen in c. above. As a result of initial definition and screening presented in Paragraph 4.1, four different transfer systems were conceptually designed. These are listed below.

   System 1 — Space Tug Supply with linear acceleration of supply module and Tug separated from the Shuttle Orbiter.

   System 2 — Space Tug supply with linear acceleration from Shuttle drag with the Tug attached to the Shuttle Orbiter.

   System 3 — Space Shuttle Orbiter supply using surface tension screens for liquid acquisition.

   System 4 — Multiple Receivers supply of the Solar Electric Propulsion Stage using a diaphragm, Large High Energy Observatory - B using a paddle for liquid acquisition and the Satellite Control Section using surface tension screens.

   Conceptual definitions and operational procedures for these four systems are presented in Paragraph 4.2.

e. Systems Evaluation — This work, presented in Section 5.0, defines technology requirements and programs necessary for final design and development of the specific transfer systems defined in the Transfer Systems Studies task.

f. Analysis of Shuttle/Tug Fluid Transfer Benefits — This task determined specific performance improvements and potential cost savings of in-orbit fluid supply of the Space Tug and Shuttle Orbiter using supply systems 1, 2 and 3 as defined in d. above. Results are presented in Section 6.0.

A summary of overall program conclusions and recommendations is presented in Section 7. An executive summary report covering the total program is published as NASA CR-135020.

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TECHNOLOGY EVALUATION

This section presents an evaluation, in general terms, of the adequacy of existing technology to design cryogenic and non-cryogenic in-orbit fluid transfer systems. Evaluation is divided into technology categories as developed during the literature analysis task, results of which are reported in References 2-1, 2-2, and 2-3. These categories are repeated below.

1. Low-G Fluid Behavior

    a. Interface Configuration
    b. Interface Stability
    c. Natural Frequency and Damping
    d. Liquid Reorientation
    e. Bubbles and Droplets
    f. Fluid Inflow
    g. Fluid Outflow
    h. Convection Heat Transfer
    i. Boiling Heat Transfer
    j. Condensation Heat Transfer
    k. Venting Effects
    l. Fluid Properties

2. Cryogenic Thermal Control

    a. Multilayer Insulation
    b. Other Insulation
    c. Fluid Lines
    d. Tank Supports and Penetrations

3. Fluid Management Systems

    a. Fluid Line Dynamics and Thermodynamics
    b. Instrumentation
    c. Stratification/Pressurization
    d. Low-G Vent Systems
    e. Fluid Mixing
    f. Refrigeration and Reliquefaction
    g. Interface Control and Liquid Acquisition

For each of these individual technologies, discussions are presented covering the requirements for in-orbit fluid transfer and existing deficiencies with respect to fluid transfer systems design. The information is based on the technology summaries presented in References 2-1, 2-2 and 2-3.

2.1 LOW-G FLUID BEHAVIOR

The technology work reported in this section is concerned only with fluid dynamics, thermodynamics and fluid properties associated with low-g or the effects of variation in g-level which could be useful in predicting fluid behavior at low-g.

2.1.1 INTERFACE CONFIGURATION - Covered here is the determination of equilibrium low-g interface geometries under both linear and rotational accelerations, including relaxation times to attain equilibrium configurations. Analytical studies utilize a force balance on the fluid considering the acceleration field and surface tension forces. The interface configuration must be defined when examining propellant management evolutions such as venting, mixing, outflow, or settling. The interface shape is a geometric characteristic which is required in most analyses just as the tank shape is required. The low-g configuration of the fluid results in increased surface area and increased wall wetting, both of which must be considered in tank thermodynamic evaluations. The interface shape enters into settling and venting considerations where the initial equilibrium condition is of importance to predict transient fluid behavior. In propulsion design, the initial fluid configuration is a consideration in determining the engine start procedures. For fluid transfer, the initial location of the fluid in the tank is a departure design point, while transfer with minimum acceleration uses the configuration to define residuals. Tank heating analyses for coast requires definition of the fluid configuration for determining wetted areas.

The state-of-the-art is basically adequate for design of transfer systems. Some particular configurations may require interpolation when the desired Bond number or fill quantity has not been analyzed with the computer codes and graphical results available and verified by drop tower tests; however sufficient data are in general readily available for spheres, ellipsoids, or cylinders. For other configurations, some approximations to the exact interface shape will be required since this data is more qualitative. The subject area of transient interface conditions and the time to readjust to a new acceleration field have not been given adequate study, particularly verification of the empirical correlations postulated by Hastings, 1969, (Ref. 2-4) or Paynter, 1964 (Ref. 2-5). These transients may occur rapidly during settling or slowly during maneuvers from attitude control firings.

2.1.2 INTERFACE STABILITY - The area of interface stability covers a range of topics: tank interface behavior under various acceleration environments (lateral, axial and rotational), static and dynamic stability for various geometric configurations, and stability for gaseous jets impinging on surfaces. These topics are of importance to fluid transfer. Other fluid instabilities which may be gravity sensitive but are not necessarily of significant importance to fluid transfer are convective stability and stability of moving films.

The requirements for interface stability criteria in the design of fluid transfer systems manifests itself in interface control. A stable interface in the acceleration environment during the transfer will allow easier prediction of fluid location, will be less likely to contribute to ullage collapse; and will not induce vehicle torques requiring attitude control propulsion. The designer must know the limits of axial, lateral, rotational, and vibrational acceleration under which this stable interface is maintained. The design of surface tension acquisition devices utilizes data on stable fluid configurations in holes and channels and the response of the fluid to the operational environments. Rotational stability is important to the design of fluid and tank rotation type acquisition devices. The geyser behavior of flow fields may effect the selection of baffles for inflow and outflow. Finally, the selection of the pressurization system for low-g involves an assessment of the gas-flow interaction with the interface as it relates to ullage collapse or interface break-up leading to vapor ingestion, bubble entrainment or early pull-through. Conditions for liquid breakup from gas impingement may also be applicable to defining low-g mixer performance when encapsulated in vapor.

In most aspects of interface stability, sufficient data have been generated from analytical efforts, with drop tower verification, to select operational conditions defined by dimensionless variables which are within stable regimes. The effect of geyser formation during termination of outflow may be influenced by the presence of inflow or outflow baffles and may require further stability evaluation, since the latest work in this area (Ref. 2-6) was limited to non-baffled tanks. Current one-g vent systems maintain a settled liquid during venting; whereas rapid ventdown under low-g conditions could result in bubble ebullition and interface breakup. This instability phenomena has been discussed but no design criteria or test data have been developed in low-g; excess liquid carryover may result, depending on the vent system selected.

2.1.3 NATURAL FREQUENCY AND DAMPING - This technology includes lateral and longitudinal sloshing, slosh waves and tank elasticity effects, and slosh suppression by baffles and viscous damping.

The fluid transfer mission requirements include a design requirement to be able to define the natural frequency and damping coefficients for various shape tanks and disturbances in low-gravity environments down to and including zero Bond number. Similarly, adquate information on baffle effectiveness will be required. These needs exist because of the interactions with vehicle control during docking and maintenance of vehicle position.

Extensive investigations have resulted in some design data for orbital transfer applications. The range of data presented in Figure 2-1 (Bo_N = 1 to 1000) indicate the broad

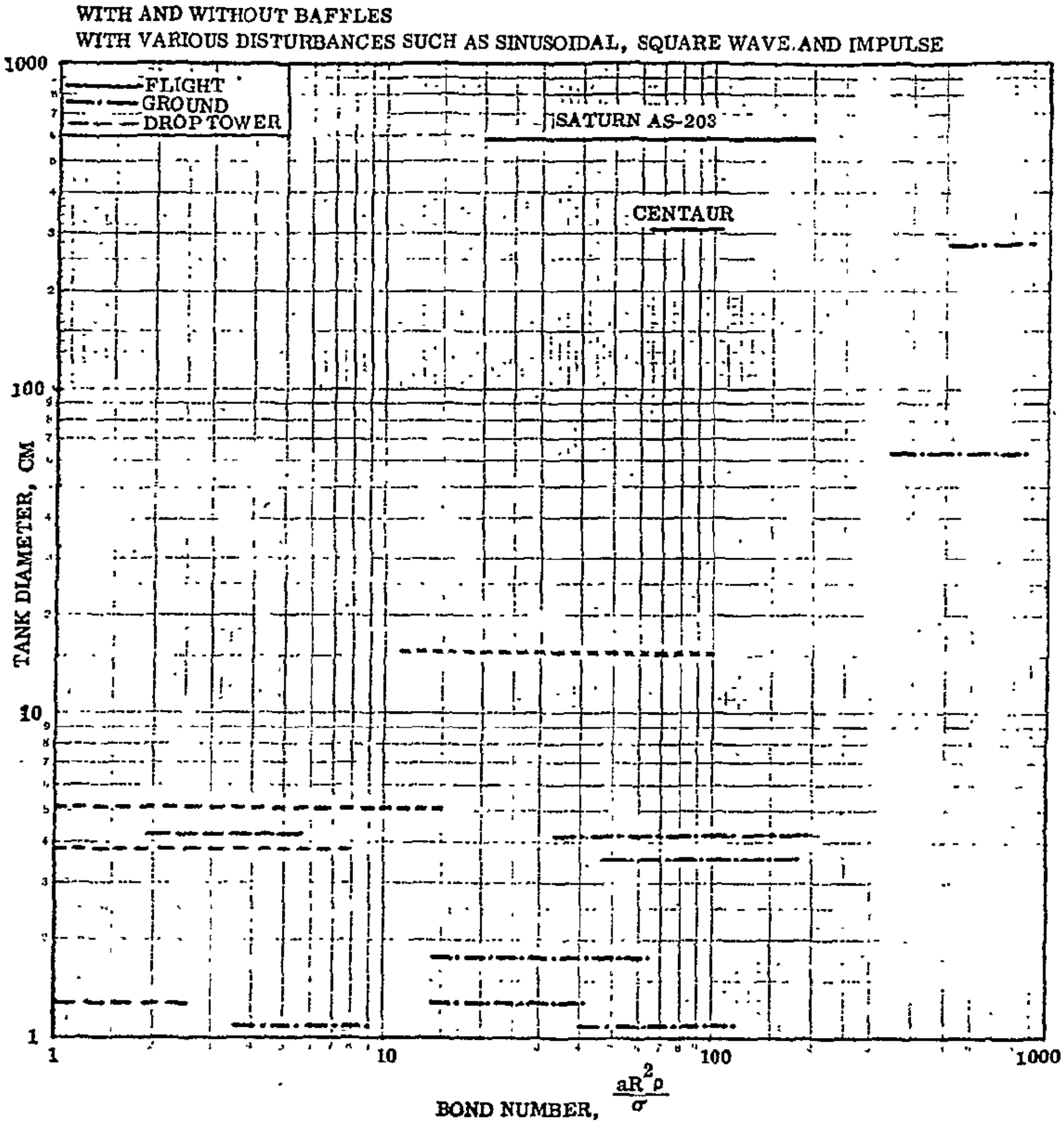


Figure 2-1. Natural Frequency and Damping Testing (Sloshing-Wave Motion)

extent of the investigations. The only major gaps in the data occur at low Bo_N and large tank sizes.

Two other technology deficiencies are significant - (1) low-g time has been insufficient to verify the low natural frequencies at low Bond numbers, and (2) analytical solutions in arbitrary containers are impossible below contact angles of 5°. Experts indicate this latter factor is not a serious deficiency and that solutions at 5° contact angle are adequate.

2.1.4 LIQUID REORIENTATION - Liquid reorientation is concerned with motion of a liquid/vapor interface during acceleration from the forward end of a tank to the aft end, over the outlet. Accelerations can be impulsive or sustained in nature. Liquid reorientation may be needed before venting or expulsion is accomplished. Details of the collection process are required for determining supply tank operations and weight penalties for venting and fluid transfer.

Low Bond number ($Bo_N < 450$) and Weber number ($We_N < 30$) cases can be adequately handled by using NASA/LeRC drop tower correlations. For higher Bond number and Weber numbers where geysering and recirculation become dominant, drop tower test times are inadequate. For these cases the diaphragm test results of Blackmon, et al, 1968, (Ref. 2-7) should be applicable. Unfortunately the empirical coefficients needed to solve the equations presented are not obtainable from the data as presented. Analytically, the SMAC model discussed in Bradshaw and Kramer, 1974 (Ref. 2-8) is a promising tool for analyzing specific vehicle cases. However, due to its running time and complexity, the SMAC model is not a predesign tool. In order to develop an adquate predesign tool more work needs to be done in obtaining and correlating test data for predicting geysering and recirculation effects on liquid reorientation or simplifying the SMAC model to permit its use as a predesign tool. Current predesign predictions force the use of some conservative multiple (3 to 5) of the free fall time (time for liquid to impact the aft bulkhead from its initial position on the forward bulkhead).

2.1.5 BUBBLES AND DROPLETS - The subject area of bubbles and droplets is difficult to scope because of its interaction with the heat transfer process. In the bulk liquid or the ullage, bubbles or droplets can be considered individually or as a population. Boiling heat transfer and condensation on surfaces involves the behavior of bubbles and droplets, respectively.

Several applications have been identified where bubble behavior data in low gravity is required. In settling flow dynamics, bubbles are entrained in the liquid due to turbulence. In order to transfer or pump liquid which is free of gas, a sufficient period must be allowed for the bubbles to clear or coalesce. The collapse of bubbles is pertinent to the re-filling or thermal conditioning of acquisition devices. The collapse times must be related to the degree of subcooling. Most important in this technology area is the behavior of droplets in the chilldown process for a cryogenic receiver in reduced-g. The behavior of the droplets as they interact with the tank wall has not been reported for reduced gravity investigations. Droplet heat transfer aspects strongly influence tank pressure. Droplet behavior also influences ullage collapse in the above heat transfer/wall interaction.

A model for bubble clearing the liquid was developed by Blackmon, 1968 (Ref. 2-7), however the bubble modeling was based on one-g data and no reduced-g data has been correlated to verify the model. Bubble clearing is a phase of liquid reorientation requiring further study. Model development and one-g and reduced-g testing are required in the area of the collapse behavior of bubbles in temperature gradient fields and on pressure variation resulting in bubble subcooling. Analytical modeling and experimental verification including reduced-g testing is required on droplet/wall interactions. This should include testing to obtain both fluid mechanics and heat transfer data such that it might be extended to large tank chilldown.

The static growth of bubbles in a bulk liquid at low-g has been investigated through drop tower tests and the data obtained was in agreement with basic theory (Ref. 2-9).

2.1.6 FLUID INFLOW - The fluid inflow topic covers the fluid mechanics and heat transfer aspects of liquid entering a tank and impacting on the walls and/or inlet baffling, including resultant flow patterns. Ullage gas pressure/temperature effects are included.

Analytical models for the fluid behavior, pressure history, and heat transfer aspects of fluid inflow to baffled and unbaffled tanks is required. These models are required to scale test data to full-scale vehicles; current data is primarily qualitative in nature and addresses mainly the stability and flow patterns of the inflow process with some data on the effect of vent location on potential liquid loss. Extensive data is needed on pressure histories and tank chilldown requirements during inflow for the transfer process. The influence of droplet behavior on wall cooling in the inflow process is needed for the design of adequate vent systems. The effectiveness of baffles in permitting non-vent transfers requires additional modeling and low-g model verification. This information will lead to a selection of optimum inlet baffle arrangements for the particular tank/fluid combination.

Current analytical model development and verification is inadequate for this complex process which is of major importance to fluid transfer hardware design. An existing model, Vernon, 1967, (Ref. 2-10) has major deficiencies in droplet phenomena, baffle treatment, and heat transfer considerations. Extensive reduced-g data has been generated since the model was developed, but the model has not been updated nor verified with this data.

Further testing in reduced-g is required to analyze the heat transfer aspects of the inflow process. The existing data must be used for model verification to assess the areas where additional analytical/experimental effect is required. It is observed that more extensive deficiencies exist in the heat transfer aspects than in the flow field definition.

A proposed approach is to use recently generated data on jets and data on fluid behavior from baffled drop tower filling cases to develop an analytical model. The higher filling rates with baffled arrangements indicate their desirability. Although inertia effects dominate in the entrance sectors where the jet is well-defined and gravity effects are not important, the phenomena near the wall and at jet stagnation will be strongly influenced by the acceleration field.

2.1.7 FLUID OUTFLOW - Fluid outflow is concerned with motion of the liquid/vapor interface during the draining both with and without baffles. Pullthrough and vortexing were considered. Both phenomena manifest themselves by interface deformation during draining that can cause premature entrance of vapor into the outlet. Vapor pullthrough is caused by inertia forces accelerating the fluid in the region of the outlet. Vortexing is caused by initial fluid circulation. Suppression of pullthrough and vortexing is essential to minimize liquid residuals during draining. Liquid residuals, in low gravity are sensitive to liquid outflow rate and fluid conditions (such as sloshing) and tank geometry.

For high Bond number ($Bo_N > 10$) draining, some evaluation has been done on different tank shapes. Froude number correlations are applicable here. Work should be extended to include all possible tank shapes considered for propellant transfer purposes, including those with contoured outlets. Pullthrough suppression should be investigated much more extensively in order to provide design criteria for designing pullthrough suppression baffles, screens or sumps. Currently published studies do not provide sufficient information for this purpose.

For low Bond number ($Bo_N < 0.1$) draining, unbaffled cylindrical tanks with flat and hemispherically ended bottoms have been adequately handled. Weber number correlations are applicable. Other tank shapes need to be studied. Also, pullthrough suppression in low Bondnumber draining is even more important than in high Bond number draining because of potentially high residuals. Minimum residuals with current designs are on the order of 40% of tank volume. No adequate baffle configurations have been developed. More work needs to be done to minimize residuals both through baffling, screens and sumps and through outflow throttling. Correlations need to be developed comparing the various residual suppression techniques. Design principles for optimizing each technique need to be developed. Empirical evaluation and correlation of dimensionless groups would appear to be more fruitful in this area than analytical studies. Studies to date have only considered quiescent interfaces. The work by Symons, 1974 (Ref. 2-11) on outflow throttling indicated that interface disturbances over the outlet could have a significant effect upon residuals. Interfacial disturbances during draining, such as sloshing, should be studied to determine their effect on vapor pullthrough in low gravity.

For intermediate Bond number ($0.1 < Bo_N < 10$) cases both Weber number and Froude number are important. No work has been done of note except for identification that both dimensionless parameters should be considered. Should this regime appear to be of interest for fluid transfer, extensive correlations in the form of residuals versus ($We/1 + Bo$) will be required. For vortexing, pragmatically speaking, no problems exist. Normal baffling and operations have eliminated vortexing during draining. Unforeseen problems may occur however, that cannot be anticipated because of the lack of theoretical knowledge or empirical correlations predicting the onset and severity of vortexing.

2.1.8 CONVECTION HEAT TRANSFER - The technology covered here is concerned with both free and forced convection in single phase fluids, including supercritical fluids.

The two main areas of concern for in-orbit fluid transfer are associated with cryogens and are; (1) determination of the incipient boiling point in order to know when vapor may be formed at tank walls or within positive expulsion and surface tension liquid acquisition devices, and (2) determination of fluid temperature stratification and tank pressure variations to be expected at low-g.

In order to fully verify existing convection heat transfer equations, long term orbital experimentation would be required. There are no drop tower or aircraft flight data available, since times of these ground based tests are insufficient to develop stable boundary layers. From supercritical O_2 data obtained from Apollo flights, it appears that classical convection equations containing an acceleration term should be applicable at low-g. However, data from small lab type experiments aboard Apollo 14, 17 and Skylab 3, where poor correlation with analysis occurred, indicated that overall heat transfer may be somewhat affected by acoustic waves which would not be significant at one-g. This phenomenon needs further investigation through long term orbital testing.

2.1.9 BOILING HEAT TRANSFER - The technology covered here is concerned with transition, nucleate, peak, minimum and film boiling, including both transient and steady state conditions as well as bubble dynamics and other characteristics associated with boiling at a solid surface. Both pool and forced flow boiling are included.

2.1.9.1 Pool Boiling - The main requirements of pool boiling technology, for application to in-orbit fluid transfer, are associated with tank fluid temperature stratification and pressurization, vapor formation within acquisition devices and receiver tank chilldown. The boiling curve shown in Figure 2-2 illustrates the consensus of opinion as to what happens at low-g based on drop tower, aircraft and magnetic simulation testing. However, there is not much data at very low-g levels ($a/g < 0.01$). Also, there are indications that for longer test times at very low-g the surface would become vapor bound. This is further indicated from the results of a significant amount of work performed at the University of Kentucky where analyses and test (one-g and elevated g) indicated that very small wires tested at one-g would simulate the boiling heat transfer to be expected on larger systems at low-g. This work says that at zero-g the nucleate boiling regime vanishes altogether.

Orbital testing would be required to verify and/or resolve the above conclusions. In any such experimentation, emphasis should be placed on accurate and controlled data measurements at both low-g and one-g.

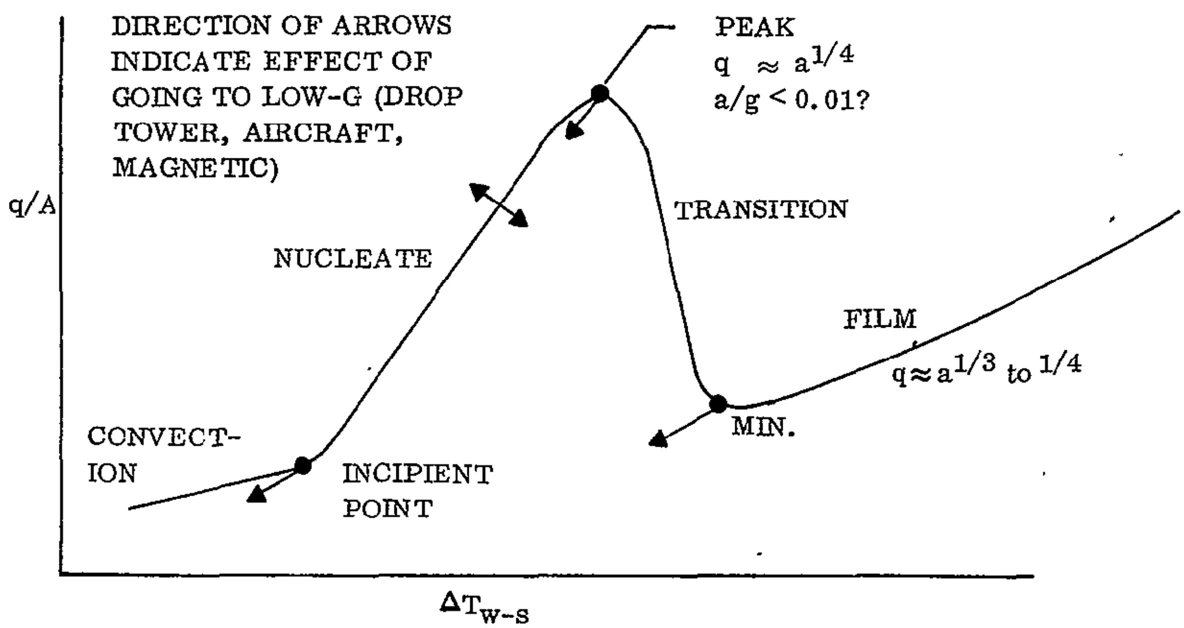


Figure 2-2. Pool Boiling Heat Transfer Test Results

2.1.9.2 Forced Flow Boiling - Forced flow boiling heat transfer data are needed in the design of heat exchangers operating at low-g (such as might be used with auxiliary power systems and thermodynamic type zero-g vent systems), for analysis of boiling at a tank wall where fluid mixers are used and for analysis of transfer line and receiver tank chilldown. The dynamics and thermodynamics specifically associated with transfer line chilldown are covered in Paragraph 2.3.1.

Very little quantitative data have been obtained to-date. Existing analytical heat transfer and pressure drop models are based on a knowledge of flow regimes (slug, plug, mist, etc). Longer test times than are available with drop tower and aircraft tests are needed for a final determination of the effect of low-g on the existance of the various flow regimes.

From the available drop tower and aircraft flight data it appears that for reasonably high flow rates and/or by the use of means to vortex the flow within tubing passages the performance of boiling heat transfer equipment at zero-g can be predicted by testing at one-g. These assumptions would, however need to be verified by orbital testing.

In the case of boiling where fluid motion exists, such as during receiver tank chilldown, essentially no published data exist. However, work is presently being accomplished in the NASA/LeRC drop facility with LN_2 inflow to a 0.6 m (2 ft) dia tank which should give some insight into this phenomenon.

2.1.10 CONDENSATION HEAT TRANSFER - The technology covered here includes dropwise and film condensation under flow and non-flow conditions at both liquid and solid surfaces.

No non-flow data is available. Drop tower and aircraft test times are insufficient to develop stable boundary layers. Some aircraft tests were accomplished under forced flow conditions with mercury and with water. These tests indicated that heat transfer and pressure drop were not g-sensitive. Flow rates, however, were quite high and thus the conclusions made could not be taken to be generally applicable without low-flow testing requiring longer test times than available from aircraft flights.

The application of condensation heat transfer technology to in-orbit fluid transfer would be in the following areas;

a. Performance of condensing heat transfer equipment. The primary example of this is where either a bulk (compact) or distributed type heat exchanger vent system is employed for tank pressure control at low-g. In the case of the bulk exchanger system a mixer is used to force fluid through the hot side of the exchanger and thus the design can employ forced vortexing type flow which should not be sensitive to variations in g-level. In the case of the distributed exchanger without a mixer, significantly conservative assumptions must be made in design and the final performance demonstration of such systems must await relatively long term orbital testing.

b. Thermal conditioning of screen type liquid acquisition devices. Such can be significantly dependent on the potential for condensation heat transfer of cooling surfaces located inside the storage tank. At present, designs are accomplished assuming much higher condensation rates than should occur at low-g. In order to relax such assumptions relatively long term orbital testing would be required.

c. Condensation heat transfer at tank walls and at liquid surfaces. Such phenomena may play a major role in overall tank pressure and pressurization predictions for both supply and receiver tanks. At present, somewhat conservative assumptions must be made in such calculations. Before a change in present design methods could be justified a significant amount of orbital test data would be required on near-full-scale systems.

2.1.11 VENTING EFFECTS - This technology covers bulk and surface vapor generation caused by tank venting as illustrated in Figure 2-3. Such vapor generation can affect acquisition system performance and cause liquid level rise with subsequent loss of liquid at the vent.

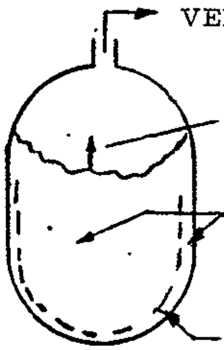


Figure 2-3. Potential Vapor Generation by Venting

Knowing under what vent conditions only surface evaporation will occur would allow the specification of systems to prevent liquid loss and vapor formation within screen acquisition systems. Vapor formation within a screen acquisition device can cause reduced transfer efficiency and even complete breakdown of the liquid transfer process. Some drop tower and SIV flight data are available on the venting effects phenomenon. The most extensive work was at the NASA/LeRC where drop tower tests were accomplished and an analytical model developed (Ref. 2-12). The analytical model was designed to determine pressure decay rates when only surface evaporation takes place. Even though the analytical model developed represented a significant improvement over a simple adiabatic blowdown model the correlation with test results could still be significantly improved. One factor is that the model does not now account for wall heating.

Further work is also needed in this area to include the effects of pressure cycling, non-condensible pressurant, variations in ullage temperature, fluid mixing and the existance of submerged surfaces such as screens.

2.1.12 FLUID PROPERTIES - Consideration was given here to any fluid properties which might be sensitive to reductions in gravity. No specific data demonstrating low-g sensitivity of any fluid properties was found in the literature. The only low-g testing (drop tower) was on dynamic contact angle which was found to be insensitive to gravity.

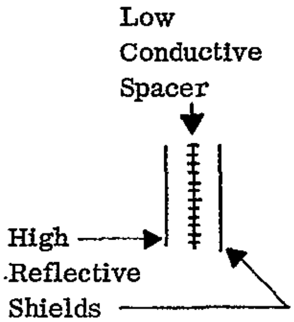
However, supercritical fluid properties are suspect and orbital testing has been proposed. Superfluid helium properties are to be examined during rocket tests by the Los Alamos Scientific Laboratory.

In general, it is considered unlikely that normal fluid properties will be affected by gravity to the extent where transfer system design and performance is significantly influenced.

2.2 CRYOGENIC THERMAL CONTROL

This section is only concerned with technology having potential application to in-orbit fluid transfer systems and/or associated space storage. Insulation systems only applicable to atmospheric operation such as Batting, Dyna Quartz, etc. are not included. Also, fluid tankage and associated structural details are not included. Thermal control of tankage (side wall multilayer and other insulation and tank supports and penetrations) and fluid lines are covered, as discussed in the following paragraphs.

2.2.1 MULTILAYER INSULATION (MLI) SYSTEMS - The most efficient form of passive insulation for cryogenic storage systems operating in a vacuum are the multilayer insulation systems. They consist of highly reflective metallized films which minimize radiation heat transfer, and some technique for spacing the films to minimize conduction heat transfer,as shown below. Considerable work has been carried out with regard to the analysis, development, and prototype testing of MLI systems. These systems have been under serious study for more than ten years. Numerous system configurations have been developed and many have been discarded for lack of competitive performance. Any of those remaining could be developed to provide adequate, if not optimum, thermal performance for specific fluid transfer mission applications.



Further work to increase performance and/or reduce weight and cost is outlined below.

a. Development of low cost reusable MLI insulation systems. Present reusable systems employ expensive goldized Kapton.

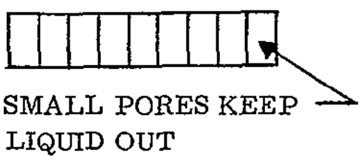
b. Weight and cost reduction of purge bags. Purge bags are used in conjunction with the MLI to protect the insulation from moisture condensation during ground hold, boost and re-entry. Such systems are in the early stages of development and further work should result in an overall reduction of weight and cost. In conjunction with purge systems the use of sealed leakage containment membranes (LCM) should be considered to prevent the possibility of gas leakage from the tanks degrading insulation performance. These membranes would be designed to channel any leakage to a convenient point for dumping to space.

c. Demonstrate inspection and refurbishment of MLI. For reusable applications, reliable procedures will be needed to insure that insulation performance is satisfactory prior to each flight.

d. Final development of lightweight reusable vacuum jackets for use with MLI. Such systems can significantly reduce ground hold and launch heat flux, increase launch flexibility (tank topping prior to launch minimized) and protect the MLI from adverse external environments. Significant progress has been made toward the development of such a system under Contract NAS3-15848 (Ref. 2-13).

2.2.2 OTHER INSULATIONS - MLI systems are discussed above. This section covers the other forms of storage vessel thermal control. This topic is broad, being intended to include all forms of tank cryogenic insulation systems and concepts other than multilayer insulation. The types of systems included here are conventional closed-cell insulation materials, internal gas-layer systems, microspheres, active cooling in the form of vent gas heat interception or the use of secondary cryogenic fluids, and shadow shields.

Conventional closed-cell foam systems have received the most development effort, primarily due to the Saturn vehicle program. Comparatively efficient, reliable, and inexpensive systems are available for one-shot use. Additional effort is required to develop a truly reliable reusable closed-cell foam insulation system. The internal, open-cell, gas layer insulations (shown below) have never been flown, although a polyphenylene oxide foam system appears to be promising for reusable applications.



Microspheres, an LMSC proprietary system, is apparently still in the comparatively early stages of development, and considerable additional work would be required to demonstrate its feasibility.

The use of vented boiloff gas or a secondary cryogenic fluid to intercept incoming heat has been shown by analysis and test to be feasible and highly efficient from a thermal standpoint. Unknowns exist with regard to trading off the added cost and complexity associated with these systems against the reduced fluid loss. A similar statement can be made with regard to solar shadow shields. They are usually considered for long term deep space coast missions where the vehicle orientation with the sun can remain constant for long periods of time. Although clearly feasible, the benefit for a short term orbital resupply mission would need to be evaluated in light of the additional system complexity and cost.

Use of MLI with a foam substrate may prove desirable for reducing ground hold and boost heat leak. Also, by employing a foam substrate with low temperature cryogens such as LH_2 and LHe a GN_2 purge can be employed rather than GHe. GN_2 has a lower conductivity and thus lower heat leak during purging than GHe. Complete reusability of such systems would need to be demonstrated.

2.2.3 FLUID LINES - The lines used to transfer fluids from a supply to a receiver tank must have high reliability and low weight. For use with cryogenic fluids the lines should also have a low thermal mass and be well insulated to minimize fluid boiloff.

Extremely efficient, light weight composite fluid lines have been developed and tested. These lines employ MLI to minimize external heat leakage. Vacuum jacketed line systems have also been built and tested. No deficiencies are known to exist with regard to adapting existing fluid line technology to space fluid transfer systems. However, assuming a requirement for vacuum jacketed lines their long term reusability would need to be demonstrated.

2.2.4 TANK SUPPORTS AND PENETRATIONS - Insulation systems for space cryogenic storage tanks, MLI systems in particular, are extremely efficient with regard to minimizing longitudinal heat flux to a tank. Thus it is imperative that tank support systems, fluid flow penetrations, instrumentation harnesses, etc, also possess high thermal efficiency to avoid contributing significantly to total heat flow and fluid loss.

A considerable amount of work has been performed with respect to analysis and test of insulation penetrations and support struts. Analytical techniques for determining the effect of penetrations on local MLI performance and overall heat flow to the tank are available. Highly thermally efficient, structurally sound, and low weight composite tank support struts have been tested.

No significant, identifiable deficiencies with regard to tank supports and insulation penetrations have been found, however, long term structural reusability of composite support struts would need to be verified prior to operational use.

2.3 FLUID MANAGEMENT SYSTEMS

Fluid management technology associated with in-orbit fluid transfer and/or associated storage at low-g is covered here. The work reported here is more hardware oriented, as compared with the low-g fluid behavior technology discussed in Section 2.1.

There have been a number of fairly large studies accomplished to develop data on overall fluid management for in-orbit transfer. These have been mostly operations type studies with individual systems technology based on the existing state-of-the-art at the time the study was accomplished.

The only new or unqiue overall transfer concept found in the literature was by Yoshikawa and Madison, 1971 (Ref. 2-14) which proposes to utilize the heat sink in the receiver (vaporized fluid used to drive compressor) in connection with a thermodynamic compression-expansion flow cycle (Figure 2-4).

The main problems with this system are (1) the supply tank fluid must be maintained in a mixed condition during transfer, (2) the required distribution of wall cooling and ullage cooling in the receiver would be difficult to maintain, (3) the system as proposed here would not work with an initially cold receiver tank, and (4) design of a compressor to operate with a liquid/vapor inlet would be difficult. This system is considered further under the transfer systems studies task (Section 4) where an external source of energy is used in place of reliance on the receiver wall heat. Use of a heat exchanger upstream of the compressor was also considered to insure 100% vapor to the compressor.

Typical of overall systems studies are those by Sexton, et al, 1972, (Ref. 2-15) and Stark, 1972, (Ref. 2-16). These studies were limited by existing technology and came to the following conclusions.

a. For large quantities of cryogenics, linear acceleration for liquid acquisition looks to be the most promising if the requirement existed now. For small quantities of cryogenics and non-cryogenics, bellows or metallic diaphragms look promising for current use, as do bladders and surface tension devices for non-cryogenics.

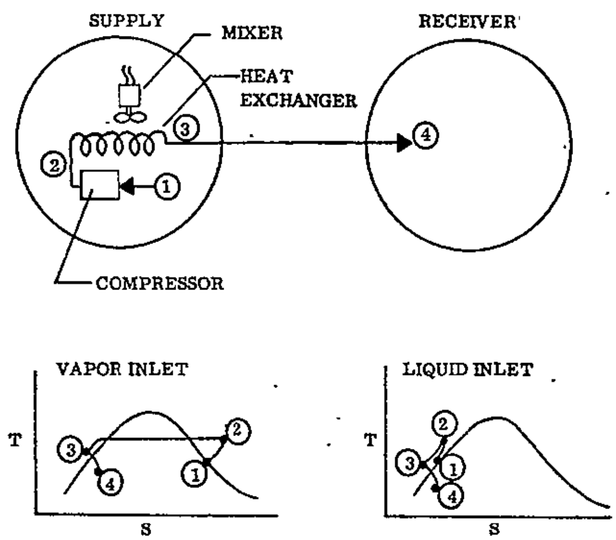


Figure 2-4. Thermodynamic Transfer Concept

b. Surface tension screens and tank and/or fluid rotation look promising for future development for use with a large range of fluid quantities for both cryogenics and non-cryogenics.

c. The study by Sexton, et al, (1972) showed promise to using the receiver vent fluid to pressurize the supply tank.

d. The study by Stark (1972) showed that the transfer of single phase fluids at high pressure (above critical) suffer large weight and power penalties.

Specific technologies under fluid management are discussed in the following paragraphs.

2.3.1 FLUID LINE DYNAMICS AND THERMODYNAMICS - Covered here are start/shutdown transients, two-phase flow surging, pumping, pressure drop and critical flow or choking, and line chilldown.

Such data are required to design transfer lines to minimize loss of cryogenic fluids due to flashing or system chilldown, predict pressure surges which could effect acquisition system operation, and determine receiver inlet conditions and transfer time. A knowledge of the state-of-the-art of fluid pumping allows supply tank driving pressures and pump power requirements to be minimized. In the case where transfer line and associated hardware are not wet prior to transfer, chilldown could possibly cause pressure surges, especially when using subcooled liquid, which could result in reverse flow and possible vapor entrainment in a screen capillary device. This has not been satisfactorily investigated to-date.

Analytical models are available to predict two-phase flow pressure drops and heat transfer rates with reasonable accuracy at one-g. The accuracy of these models depend on a knowledge of the flow regimes (slug, plug, mist, etc) involved. There is very little experimental data applicable to defining fluid line dynamic and thermodynamic conditions expected at low-g. Some KC-135 aircraft flight tests were accomplished (Ref. 2-17) with non-cryogenic fluids to determine the effect of g-level on flow regimes. These data gave some insight into the effects of g-level, however, test section sizes and test times were not sufficient to reach final conclusions.

Calculation techniques are available for predicting peak line pressure surges for both single phase and two-phase flow at one-g. Along with this, techniques for minimizing pressure surges have been identified. This data, however, has not been related to surface tension system limitations nor verified in a low-g environment.

Pumping of saturated fluids has been demonstrated at one-g such that pressurant requirements can be minimized, however, final system design would need to be verified through low-g orbital testing.

2.3.2 INSTRUMENTATION - Technology information covering low-g mass gauging, liquid/vapor sensing, two-phase flow quality/density gauging, flow metering, and temperature and acceleration measurement is presented in the following paragraphs.

2.3.2.1 Low-G Mass Gauging - This is the single most important piece of instrumentation required for in-orbit fluid transfer. Accurate gauging of fluid quantity at low-g also presents the greatest problem, since at low-g the orientation of liquid and vapor may be unknown.

Numerous methods of propellant mass gauging have been proposed and investigated; among which are the following:

a. Radio Frequency or RF Gauging

b. Nucleonic Absorption

c. Resonant Infrasonic (RIGS)

d. Pressure - Volume - Temperature (PVT)

e. Titration

f. Ultrasonic Gauging

g. Light Attenuation

h. Capacitive Gauging

Evaluation of the state-of-the-art of each of the many systems proposed must take into consideration specific application parameters such as propellant configuration control, tank size and configuration, fluid characteristics, etc. As a result, a system or method that may be well adapted to small tanks (less than 2.8 m^3, 100 cu ft), such as capacitive gauging, appears quite impractical when considered for much larger tanks. In light of this, a favorable evaluation of the state-of-the-art of a particular method does not imply applicability to all transfer systems.

For purposes of this evaluation, three sigma performance accuracies of $\pm$ 1.0% were considered desirable for support of low-g fluid transfer operations. Static calibration runs of some of the propellant gauging systems evaluated have demonstrated accuracies better than ± 1.0% under optimized tank configurations and environmental conditions. Complicating the propellant geometry, as may be expected under low-g conditions or complicating tank internal configurations, quickly deteriorates this accuracy. It is estimated that, in practice, it will be difficult to realize a three sigma accuracy of even ± 3.0% with any of the proposed systems for use with large quantities of cryogenics; with the possible exception of the RF gauging method.

A fairly large amount of work has been done on this concept at one-g for use with LO_2 and LH_2. The latest testing (Ref. 2-18) included demonstration of performance with LH_2 and LO_2 under various conditions of liquid sloshing and liquid orientation, including the effect of a screen acquisition system located inside the tank. In the testing the screen reflected the RF waves so that accurate gauging of the fluid outside the screen device was accomplished without including the fluid inside the screen. Testing was also accomplished with benzene in several KC-135 low-g flights. Orbital testing should be the final step in demonstrating this concept. Such systems have not been tested with most non-cryogenics and the properties of many of these fluids may not be compatible with RF gauging. Each fluid would have to be investigated on an individual basis.

The second most promising system for use with cryogenics is considered to be the nucleonic absorption concept. Such systems have been thoroughly tested at one-g and are presently scheduled for low-g flight tests in a KC-135 (Ref. 2-19). The main limitation anticipated for this system is poor accuracy with variable liquid configurations in large tanks. The target accuracy for the system to be tested is 0.25% of total tank capacity.

For application to non-cryogenic fluids it is anticipated that a simple PVT system would meet the in-orbit fluid transfer requirements.

2.3.2.2 Low-G Liquid/Vapor Sensing - The main problem here is that at low-g, liquid tends to cling to the sensor element even though the sensor is basically surrounded by vapor. Some drop tower and flight testing has been accomplished on a number of concepts. The results to-date have not been conclusive, in that none of the concepts performed satisfactorily under all test conditions. Thus further work is needed to develop and demonstrate satisfactory liquid/vapor sensing for the low-g application.

Such elements are needed for in-orbit fluid transfer to determine when vapor is being ingested into the outlet line and/or when liquid is being vented overboard from the receiver tank. Such knowledge may allow corrective action to be taken to minimize liquid loss and/or to complete the transfer.

2.3.2.3 Two Phase Flow Quality/Density Gauging - Beta ray, gamma ray and resonant cavity systems have been the prime methods considered for two phase quality measurements. Gamma ray and resonant cavity methods have been investigated for measuring LH_2 density.

None of these systems have been developed to a state-of-the-art that would qualify for measuring two phase quality during flight operations. Considerable development work will be required to establish the accuracies to which these systems may be expected to perform. It is considered that these systems, while promising, do not as yet meet the requirement for low-g quality gauging. Nucleonic and microwave cavity methods, however, appear to have the potential to provide a satisfactory basis for two-phase density gauging in LH_2 in support of space applications. One basic problem which has not been solved to-date is that high accuracies have not been demonstrated over the full range of fluid quality and types of flow (slug, mist, etc) which may occur. In any case, further work would be required to verify low-g operation and to extend capability to include other fluids.

2.3.2.4 Flow Metering - The technology of flow metering is well developed as a result of the need to support expanding commercial requirements and space operations. For single phase fluids no specific effect of low-g is expected. Mass flowmetering featuring standard turbine meters and an in-line densitometer with electronic conversion appear to be as well advanced in development as flowmeters in general. Performance (based on limited experimental data) compares well with volumetric flow test data.

The realizable performance accuracies of any flow metering system are ultimately dependent upon system calibrations. There is a reported deficiency in available cryogenic flow calibration services in the following areas:

a. Interlaboratory calibration standardization.

b. Calibration technology involving surrogate fluids.

c. Calibration of large flow (> 12.6 ℓ/s, 200 gpm) instrumentation and systems.

2.3.2.5 Temperature Measurement - A review of thermometry technology reveals that there are several types of commercially available instruments suitable for support of low-g fluid transfer operations. Among these are the platinum, copper, indium, and carbon resistance types and various thermocouple types. The inherent accuracies of these types of instruments readily meet the anticipated accuracies required to support space transfer operations. Adequate calibration services, calibration standards, procedures and instrumentation are available to insure accurate transducer calibrations.

Although not considered a critical deficiency, it is thought that the further development of thin film carbon type transducers for application in the cryogenic temperature ranges would be beneficial. The high sensitivity, fast time response, small size and ruggedness of these units could warrant the costs of such development for specific applications.

2.3.2.6 Acceleration Measurement - The technology survey on acceleration measurements revealed little new in the area of improved instrumentation for low-g, ie below 0.001 g, acceleration measurements. Certain conceptual ideas have been studied but these appear to be related to advanced science-physics applications and are quite unfit for space applications. Wallace L. VanderVelde (Ref. 2-20) anticipated the need for accurate low-g calibrations and proposed an experimental method. The accuracies to be attained by this method (0.1% from 0.1 g to 0.0002 g's increasing up to 0.7% at 10^{-5} g's) should be adequate to support low-g fluid transfer requirements. However, this method, involving zero-g flight trajectories, would be extremely expensive.

The specific requirements for acceleration measurements for support of low-g propellant transfer are, as yet, undetermined. It is anticipated that the needs would be associated primarily with system testing prior to operational flights and in determining if design allowables of acquisition systems are exceeded during operational flights. It is considered that if the actual requirements fall in the milli-g ranges they can be met by improved versions of force balance accelerometers now available without additional research effort. In the event that accurate acceleration measurements will be required in the range below 1×10^{-4} g's, basic development work may be required.

2.3.3 STRATIFICATION/PRESSURIZATION - This section covers fluid temperature stratification and tank pressurization with and without liquid outflow, with and without venting and with and without external pressurization. Tank pressurization may be with non-condensible and/or condensible gases.

Considerable work was accomplished in this area prior to 1970. Most, however was not directly applicable to low-g.

One important application of this technology to in-orbit fluid transfer would be in the determination of tank pressurant quantities for providing pump NPSH and/or expulsion of the transfer fluid. The main problem here is that the liquid in the supply tankage may not be settled and, even when settled, the liquid/vapor interface will likely not be flat. This imposes unknowns of liquid/vapor interface surface area and the potential for liquid slosh tending to increase the effective L/V interface area. This increases the heat transfer between liquid and vapor, tending to promote ullage pressure collapse. These same factors also tend to result in intermittent and/or unpredictable contact of liquid with the tank walls. Further, transfer times are likely to be significantly longer than engine firing times for which most of the existing pressurization data was generated.

Another important application to in-orbit transfer is to predict pressure rise and/or venting requirements during boost and orbital storage prior to the transfer. The same problems associated with the uncertainty of liquid/vapor orientation, as mentioned above for tank pressurization, apply here. The relation between liquid and/or vapor orientation and external heating was found by Aydelott, 1967 (Ref. 2-21) to have an important bearing on tank pressure rise rates.

Pressure rise at low-g, where the liquid interface is other than flat, can not now be predicted with a high degree of accuracy. The basic problems are (1) knowing or predicting liquid configuration under various supply tankage orientations and disturbances and (2) calculating tank pressure history assuming such information is available. A comprehensive computer model is needed which can consider various potential liquid orientation, sloshing and heat distribution conditions. In any case a significant amount of orbital data (from tests and/or vehicle flights) will be needed to verify the models and/or allow development of statistical correlations.

The ability to accurately predict pressurant requirements at low-g, especially where sloshing and/or relatively long transfer times exist, is also lacking. The best information to-date is a qualitative evaluation of the effects of sloshing at one-g on pressurant requirements and ullage pressure collapse. The best way to get final data is through an extensive orbital experiment program and/or operation of transfer systems at low-g.

In the final analysis, even with the most sophisticated of models, there would still be unknowns as to liquid orientation and disturbances associated with any specific transfer mission. Thus the best which could be done would be to define a range of pressure rise and pressurant requirements to be expected. This is the approach currently taken. However, computer models oriented to handle low-g conditions, along with additional low-g data, would serve to reduce the amount of conservatism necessary in system design. This should then reduce the basic design weight of the overall transfer system, and possibly reduce complexity since in some cases the need for low-g venting may be eliminated.

2.3.4 LOW-G VENT SYSTEMS - This technology covers the analysis, design, fabrication and test of systems to vent and/or control fluid storage and/or receiver tank pressures at low-g.

The basic problem with respect to venting is that at low-g the orientation of the liquid and vapor within a tank may be unknown. This requires that special means be provided to insure the venting of vapor. Venting of liquid imposes intolerable weight penalties and can result in vent thrust imbalance and safety problems. Fluid mixing is closely tied to vent requirements and is required for the basic operation of some low-g vent concepts. However, due to its importance and general application, this subject is covered in detail in a separate section (Paragraph 2.3.5).

Another closely related technology is low-g fluid inflow. Fluid flow patterns existing in a receiver tank during chilldown and/or transfer are important to the design of vent systems to control receiver tank pressures without excessive liquid loss. In fact, a potential method of receiver pressure control is to control the inlet liquid flow patterns such that conventional type vents, possibly extended part way into the tank, can be used. Drop tower investigations in this regard are being carried out at the NASA/LeRC and are covered under low-g fluid inflow technology. The following discussions are concerned primarily with liquid/vapor separation and/or pressure control systems which are for the most part independent of whether or not there is inflow to the fluid tankage involved.

Most of the work accomplished to-date has been in connection with relatively long term storage of cryogens at low-g. For this application bulk heat exchanger type prototype vent systems have been tested at one-g for both LH_2 and LO_2. A typical system is illustrated in Figure 2-5. These systems employ forced mixing of the tank fluid, and testing was accomplished under what should be worst case conditions (at one-g with system immersed in liquid). The only major area remaining to be investigated with respect to these systems is low-g mixing which is covered in Paragraph 2.3.5.

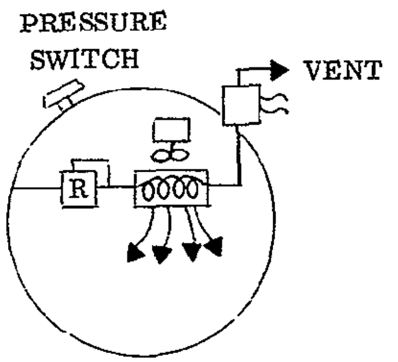


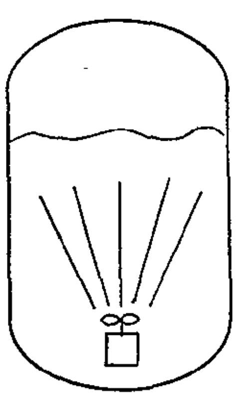
Figure 2-5. Bulk Exchanger Low-G Venting

Assuming use of a wall type exchanger vent system without a mixer, orbital testing would be required to demonstrate basic performance, since operation of this system is highly dependent on natural convection heat transfer coefficients which are gravity dependent.

With respect to receiver tank venting very little work has been accomplished. Other than the investigations of vent location and inflow patterns performed at the NASA/LeRC, the only work has been a paper study for a particular application (Ref. 2-22) resulting in the recommendation to employ an exchanger and/or mechanical vapor/liquid separator at the receiver vent to prevent direct liquid loss.

2.3.5 FLUID MIXING - Covered here are systems to destroy fluid temperature stratification and/or minimize tank pressure rise by fluid mixing.

Fluid mixing is required in order to insure the efficient operation of the bulk heat exchanger type low-g vent system discussed in Paragraph 2.3.4. Tank fluid mixing to destroy temperature stratification can also minimize pressure rise and in some cases eliminate altogether the need for venting at low-g. Such mixing may also be used to minimize or eliminate vapor formation within or at the surface of a screen type acquisition system.



There has been a significant amount of one-g testing accomplished to-date for the case where a jet type mixer is located in the liquid, as shown here. Test data using LH_2, LO_2, water and Freon 113 are available and some drop tower tests were also accomplished to measure the jet flow criteria for providing complete liquid circulation within a spherical tank. However, a comprehensive correlation of the data with an analytical model has not been accomplished.

Where it is desired to minimize the pressure rise in a tank by mixing, it is also important to determine the mixing requirements when the mixer is encapsulated in the vapor. One proposed approach to the problem has been to locate a mixer at each end of the tank such that one is always in the liquid. Such a solution does not cover all liquid orientations and is particularly deficient where spherical tanks are involved. In general, the case where the mixer is located in vapor has not been well explored. Current one-g data show that with the mixer in vapor, pressure rise rates are actually increased over that of the unmixed case.

The information required with the mixer in vapor and in liquid is concerned with power and flow rates required to mix or destroy temperature stratification as well as that to break the liquid/vapor interface. Also, the time to accomplish mixing as a function of power and flow rate is important.

2.3.6 REFRIGERATION AND RELIQUEFACTION - This technology covers refrigeration and/or reliquefaction systems to control tank pressure of space-stored fluids.

Such systems have high fixed weight and power requirements relative to other pressure control systems and would only be competitive for long term storage (greater than 30 days). The only anticipated application for long term storage in support of in-orbit fluid transfer would be if an orbital propellant depot or tank farm were to be used. Most of the work in this area has consisted of the analytical development of comparative weight and performance data over a range of heating for various refrigeration and reliquefaction cycles employing a number of different fluids. As a result of various analyses a flight weight partial reliquefaction system was built and tested at one-g for long term storage of large quantities of LH_2 (1.7 lb/hr reliquefied and 2.0 lb/hr vented overboard). During testing, adequate compressor piston ring sealing could not be achieved. It was concluded that extensive design modifications would be required to solve the problem and no further work was accomplished.

Assuming the specific requirement for such a system, the next step in the evolution would be to carry on the development as initiated.

Assuming the use of a zero-g vent system to insure a gas inlet to the liquefaction system in space, complete development testing of the system should be able to be accomplished at one-g.

2.3.7 INTERFACE CONTROL AND LIQUID ACQUISITION - Interface control and liquid acquisition as used here is concerned with collection and maintenance of liquid for expulsion at low-g. Means of providing liquid over the outlet of a supply tank so that pressurization or pumping systems can efficiently transfer the fluid to the receiver tank were considered. Work can be divided into several categories based on the type of acquisition system employed. These are: capillary acquisition, positive expulsion, and other systems. A representative sampling of the various concepts considered is presented in Figure 2-6. Linear acceleration was not considered here, since the technology applicable to this mode of transfer is covered by low-g liquid orientation and outflow (Paragraph 2.1.4 and 2.1.7).

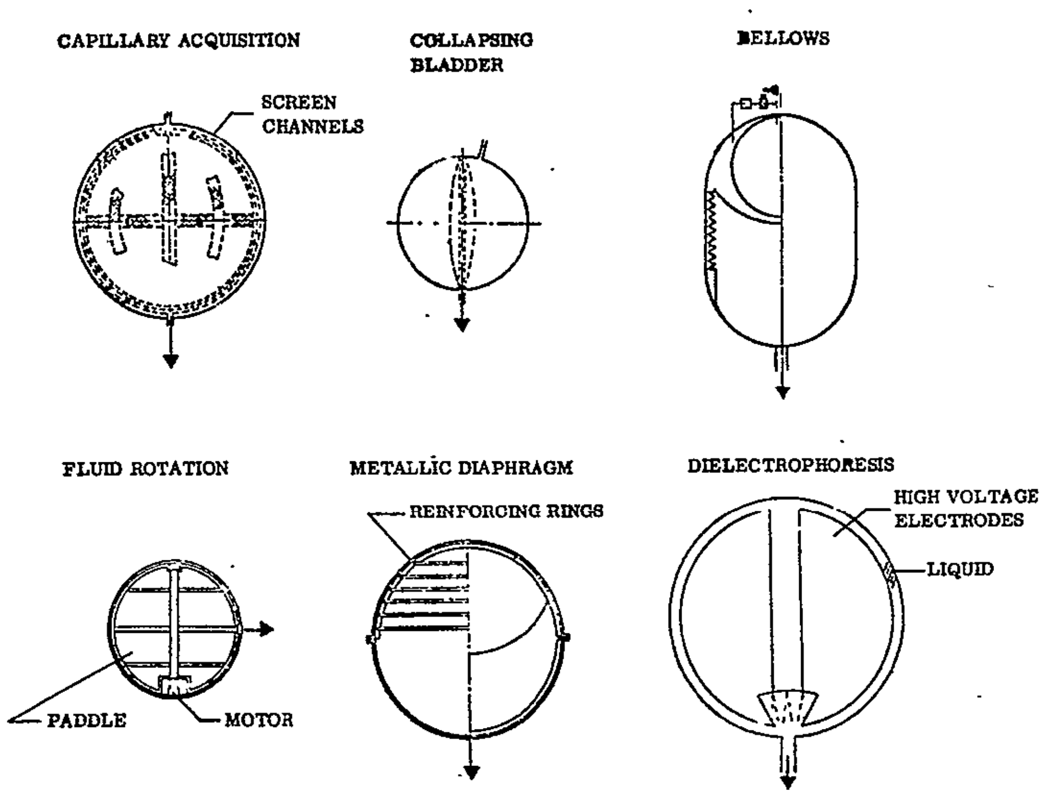


Figure 2-6. Interface Control and Liquid Acquisition Systems

Capillary acquisition uses the surface tension retention capability of screen, perforated plate or baffles to position liquid within a tank. Positive expulsion devices use a contracting, expanding (or moving) barrier to control and expel liquid. The devices include bellows, bladders, diaphragms and pistons. Bellows are thin walled convoluted tubes composed of circumferential corrugated elements. Fluid to be transferred is generally stored inside the bellows. Bladders are balloon shaped membranes that completely enclose the liquid or ullage and are contracted or expanded to expel the liquid. Diaphragms are membranes that completely reverse during liquid expulsion, forming a mirror image of themselves. Pistons are rigid bodies whose external shape closely fits the inside surface of a tank, sealed around the periphery of the piston. During piston travel along the tank wall (driven mechanically or by pressurant) liquid is expelled from the outlet opposite the initial piston position. Devices in the "other" category are those that have not seen as widespread use as capillary or positive expulsion devices. These devices include those using acoustic forces, magnetic forces, electrical forces and centrifugal forces. Dielectrophoretic fluid control devices rely upon dielectric properties of fluids to preferentially orient liquid and vapor within an electric field. Centrifugal devices rely upon fluid vortexing, tank rotation or rotating members to impart centrifugal force to a liquid vapor mixture causing the denser liquid to be positioned on the periphery of the device or tank wall.

Discussions of the current state-of-the-art and technological deficiencies of the above systems are presented in the following paragraphs.

2.3.7.1 Capillary Devices - Over 500 low-g operational flights have been accomplished with screen type capillary devices using non-cryogenic fluids. Most of these have been with engine start type systems. Some operational flights, however, have been accomplished with a 1.5 m (5 ft) diameter channel type screen system using hydrazine.

Development work is required in promoting low gravity refilling for applications where a receiver tank contains a capillary device or where a supply tank containing a capillary device is subjected to disturbing accelerations between transfers that dislodge fluid from the contained volume. This work must be done if capillary devices are to be successfully used as acquisition devices for space based refillable tankage. Other areas cited in the literature (for storable fluids) that warrant further attention are start transient and vibrational effects on capillary device retention, pressure drop of vapor flow across a wetted screen and screen dryout limits as a function of flow rate, long term propellant storability with reactive propellants, the effect of impulsive loading on screen retention, impact loads on capillary devices due to jets or waves caused by settling or sloshing, pressure drop effects in a channel due to mass injection, development of a standard procedure for conducting film bubble point testing, and continuing development of scale model simulation techniques to permit evaluation of interacting phenomena. Fabrication techniques are well developed but effort could fruitfully be spent on developing screen repair techniques for oxidizers, fine mesh forming for designs requiring compound curvature, dissimilar metal joining, holding close spacing on multiple fine mesh screen layers, and joining of screen materials such as titanium.

Cryogenic capillary devices are still in the development stage. In addition to the problems with non-cryogenics, use with cryogenics imposes the added burden of maintaining vapor free screens under external heating and pressure cycle conditions which can cause unwanted vapor formation. Cryogenic systems have not been flight tested. A significant amount of analysis and small scale testing, however, has been accomplished in the areas of steady state outflow, screen retention and flow pressure drop characteristics, thermal conditioning, wicking and the effects of pressurant and vibration environments on screen retention. In some areas, however, such as vibration effects, insufficient work has been accomplished to reach final conclusions. Further work is required in the areas of fluid dynamics, thermodynamics and fabrication. In the fluid dynamics area investigations should be initiated into wicking of screen/perforated plate combinations for providing capillary device thermal conditioning and on the effects of vibration on screen retention. Thermodynamic studies to be undertaken include; the effect of pressure excursions on bulk boiling, surface evaporation, screen drying and retention, thermal conductivity testing of screen plate combinations, line chilldown pressure surge effects on capillary device retention, effect of unpressurized outflow on capillary device thermal conditioning, capillary device/vent system integration vent flow control to supply sufficient coolant for keeping screens wet without venting excessive mass overboard, closed cycle refrigeration system for capillary device cooling using pumping to return the conditioning coolant back to the tank, mixing investigations to determine capillary device interactions, and feedline heating control using screen barriers and bypass flow lines. In the fabrication area, work should mainly be devoted to methods of attaching thermal conditioning cooling coils to the capillary device and use of conducting materials to produce thermally efficient repair joints between the coils and the device.

2.3.7.2 Positive Expulsion Devices -

a. Bellows - Bellows are well developed devices for sizes below 0.6 m (2 ft) in diameter. Over 400 such devices have been installed in flight vehicles. The main problem areas in designing and fabricating bellows that require additional development are; fabrication processes for titanium and aluminum bellows, tooling and fabrication processes for bellows over 1.02 m (40 in) in diameter, design information on the effects of differential pressures and stroke on bellows cycle life and the extension of bellows design to more severe dynamic environments than those tested to date. Bellows for expulsion of cryogencs are still in the development stage but no work is currently being done. Life cycles at cryogenic temperatures were found to exceed room temperature life cycles (4000 to 5000 cycles compared to 1000 cycles). Thermal conditioning of cryogenic positive expulsion devices is a general area requiring attention. Methods of preventing and/or collapsing vapor bubbles formed due to heat input should be explored.

b. Bladders - Bladders for cryogens are not well developed. Problems have centered on confusion related to the LOX impact sensitivity of polymeric bladders and interply inflation problems with laminated bladders. Other improvements in the areas of controlled collapse,thermal conditioning, improved material flexibility, improved seals and attachments, and fabrication of seamless bladders should be undertaken. But initially the impact sensitivity and interply inflation problems must be solved.

   For storable fluids, Teflon and elastomeric bladders are well developed. Over 6000 units up to 0.91 m (3 ft) dia have been delivered for flight. Life cycle capability is in excess of 1000 cycles. Improvements can be provided by concentrating on improved sealing techniques between the bladder and flange, prevention of gas bubble formation in propellants which are stored in Teflon bladders, the effect of cleaning and different fluids on bladder performance, reduction of interply inflation in laminated Teflon bladders, determination of long term storage capability, permeability and gas formation in elastomeric bladders, and development of readily available elastomeric compounds for fluids that attack existing elastomers.

c. Metallic Diaphragms - For storable fluids the technology is well developed. Several flight tests have been accomplished on systems up to 0.46 m (1.5 ft) diameter. Cryogenic (LH_2 and LN_2) and non-cryogenic (water) tests at one-g have been performed on systems up to 0.58 m (23 in.) diameter. For both cryogenics and non-cryogenics, life cycle capability was 5 to 10 cycles. Non-cryogenic tests on a 1.8 m (6 ft) diameter system have also been accomplished. Improvements could be instituted in supporting diaphragms against shock, slosh and vibration in a partially expelled condition and in establishing their long term storage capability. Better tooling should be developed for tack welding reinforcing wires to the steel. Improved materials should be sought. Weld area thickness should be increased to minimize structural deficiencies. Thermal conditioning problems need to be addressed for cryogens.

d. Elastomeric Diaphragms - None have been developed for use with cryogenics. In the case of non-cryogenics a number have been flown in sizes up to 0.76 m (2.5 ft) diameter. Further work is needed if larger sizes are required. Also, flange sealing could be improved. Long term compatibility would need to be demonstrated, if not already done, for the particular fluids to be employed in the transfer.

e. Pistons - In the case of non-cryogenics several flight tests have been accomplished with sizes up to 0.61 m (2 ft) diameter. For cryogenics none have been flown, however, one-g testing has been performed with a 0.61 m (2 ft) diameter system over a limited number of cycles. Further work would be required to reduce friction, demonstrate increased cycles and verify satisfactory thermal conditioning prior to as well as during the transfer.

2.3.7.3 Other Systems -

a. Dielectrophoresis - No operational systems are available. Successful tests with Freon 113 and with LN_2 were accomplished in KC-135 aircraft flights using an 0.34 m (13.5 in) diameter sphere. No work has been done on these systems in the last seven years. Further work would be needed to conclusively demonstrate safety for use with oxidizers such as LO_2. The demonstration of reasonable structural weight in large sizes and with actual environments may also be a problem with this concept.

b. Fluid Rotation - Operational systems are not available. A 0.15 m (6 in) diameter vortex separator system has been demonstrated in KC-135 aircraft tests for application to air/water separation for in-space life support. Studies reported in Reference 2-17 indicate that such systems could be applicable to much larger sizes. Further work, however, such as scale model testing to better determine power requirements and fluid residuals, would be required to demonstrate the feasibility of larger systems.

c. Tank Rotation - Several flight vehicles up to 0.46 m (1.5 ft) diameter have used this concept for liquid orientation. Further work, similar to that required for fluid rotation would be required to demonstrate feasibility of this concept for larger systems.

d. Acoustic And Magnetic - Attempts to demonstrate the feasibility of these systems were unsuccessful and no recent work is reported.

2.4 SUMMARY OF RECOMMENDED TECHNOLOGY WORK

Based on the technology evaluations discussed in the foregoing paragraphs, the following is a list of recommended technology work which could be accomplished by ground based testing and analysis. No attempt was made to assign priorities and thus this listing is not meant to imply any.

a. Bubble Collapse in Cryogenic Bulk Fluid - This work is pertinent to low-g refilling and thermal conditioning of surface tension and positive expulsion devices. One-g testing should be performed to determine basic collapse characteristics of isolated bubbles, including collapse times as function of cooling and pressurization (sub-cooling). The effects of helium in the ullage should also be considered, as well as the use of magnetic simulation of low-g as a test method. Drop tower testing should be accomplished where collapse times are projected to be reasonably short.

b. Low-G Fluid Inflow - In this area it is recommended to continue the drop tower testing at NASA/LeRC on receiver tank chilldown and vent system performance and to develop an analytical model, utilizing current and previous (non-cryogenic) drop tower data, to allow prediction of tank pressure and/or vent requirements during receiver chilldown and fill.

c. Low-G Fluid Outflow - Perform further drop tower testing at low Bond numbers ($Bo < 0.1$ to 1.0) to define efficient (low residuals) pullthrough suppression devices. The use of screens for this purpose should be investigated.

d. Vapor Formation Within Bulk Fluid - This work has primary application to determining the effects of venting and/or pressure cycling on the formation of vapor within surface tension liquid acquisition devices. Some drop tower work has been accomplished with respect to venting and an analytical model developed to determine pressure decay rates when only surface evaporation takes place. This work was accomplished at the NASA/LeRC. It is recommended to perform further development of the LeRC analytical model to allow for other pertinent effects such as wall heating, along with further drop tower testing with and without non-condensible gases and with and without fluid mixing. Addition of surface tension surfaces to the test setup should also be included.

e. Line Dynamics and Thermodynamics - Line chilldown can cause pressure surges, particularly when using subcooled liquid, that could result in reverse flow and possible vapor entrainment in a capillary device. Analytical determination and testing of system pressure surges in conjunction with screens should be made in order to define the magnitude of any potential problems and any changes in system operation or configuration that could be used to solve them. Line, pump and/or valving chilldown at initiation of transfer should be included.

f. Low-G Liquid/Vapor Sensing - The satisfactory demonstration of such systems has not been accomplished. It is recommended to accomplish further drop tower testing and possible development of the most promising concepts such as a tapered probe with heating.

g. Fluid Mixing - Fluid mixing is required in order to insure the efficient operation of the bulk heat exchanger type low-g vent system. Tank fluid mixing to destroy temperature stratification can also minimize pressure rise and in some cases eliminate altogether the need for venting at low-g as well as minimize or eliminate vapor formation within or at the surface of a screen type acquisition system. Development of an analytical mixing model and correlation with all existing data (one-g and drop tower) should be accomplished. Also, tests should be performed to determine the degree of liquid surface agitation required to accomplish destratification and reduce pressure rise rates as a function of jet/interface orientation and ullage volume. Drop tower tests should also be performed to determine flow patterns and conditions for liquid breakup and/or mixing for conditions simulating a mixer located in the vapor.

h. Screen Wicking - Data on screen wicking mass flow rate as function of distance from liquid source are required to design surface tension acquisition devices for the supply tank and to design for filling of surface tension devices which may be located in the receiver tank. Such data are needed for both cryogenic and non-cryogenic applications. Testing would be accomplished at one-g with screen specimens at various orientations with respect to the gravity vector to simulate different low-g conditions. In addition to tests with individual screen samples, tests with various elements of a complete system should be accomplished; e.g. (1) screens backed with perforated plates, (2) structural joints, (3) supports and (4) wicking jumpers.

i. Low-G Refilling of Screen Devices (Cryogenic and Non-Cryogenic) - This work is applicable to cases where the receiver has a surface tension device, such as for engine feed, and also for the supply where liquid may be lost from the screen device between transfers. The basic tasks would be to investigate and define the most promising methods for accomplishing low-g refilling and then to perform one-g and possibly low-g tests to verify feasibility.

j. Liquid Rotation for Low-G Acquisition - System trade-off studies have shown such systems to have potentially low weight and cost and high reliability for the in-orbit transfer application. However, little work has been done on such systems. Proposed work would be to first define the most promising system configuration(s) and then to perform small scale prototype testing to determine concept feasibility.

The following technologies are considered to be at a stage where any significant advancement in the state-of-the-art would require orbital experimentation.

a. Interface Configuration

b. Interface Stability

c. Natural Frequency and Damping

d. Convection Heat Transfer

e. Boiling Heat Transfer

f. Condensation Heat Transfer

g. Fluid Properties

h. Two-Phase Flow

i. Mass Gauging

j. Stratification/Pressurization

k. Supply Tank Zero-G Vent Systems

3

RECEIVER CONFIGURATIONS AND CHARACTERISTICS

The objectives of this task were to identify the characteristics of potential receivers of fluids transferred in space, and to determine the potential benefits that might be derived from such transfer. Specific candidate receiver vehicles were also recommended for further transfer systems studies (Section 4). The overall study approach used is shown in Figure 3-1.

Only Non-DoD missions are discussed herein. Benefits can be derived for DoD missions, but these missions are considered beyond the scope of this study. Spacelab is not considered as a receiver here because the spacelab is attached to the shuttle from launch through return to earth and is therefore not a direct candidate for fluid transfer from a separate system. However, spacelab experiments are included because they are potential receiver candidates for future spacecraft.

"Fluid" is defined herein as being either the liquid or gaseous state. "Fluid Transfer" as used here does not imply any particular method of transfer.

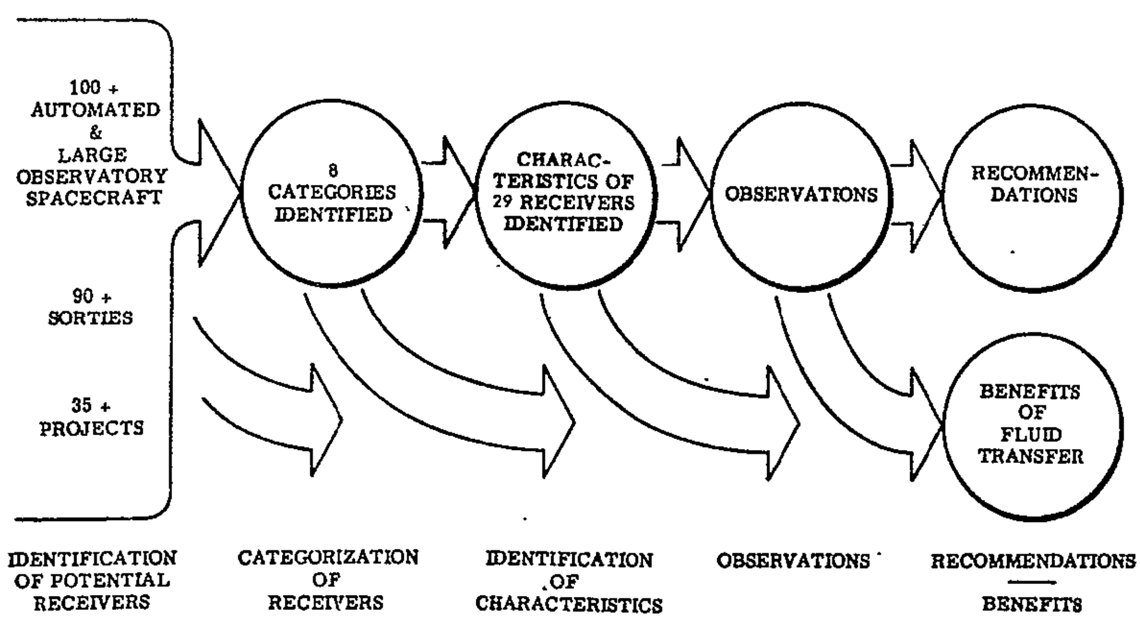


Figure 3-1. Receiver Configurations and Characteristics Study Approach

The characteristics of receiver vehicles are identified on the basis of selecting representative potential receivers from different categories. This task required the identification of as many potential candidate receivers as possible, and then the determination of categories for these potential receivers. Selection of potential receivers for further identification of fluid receiver characteristics is on the basis of receivers representative of their category and receivers which have appreciatively different characteristics, and which may be considered as potentially viable hardware for eventual usage in space.

3.1 IDENTIFICATION OF CANDIDATE RECEIVERS

Candidate receivers were identified by reviewing applicable documentation, literature, and personal communications with others within both government and industry. No time period (year) limitation was assumed for vehicle viability, so all concepts were considered. Literature review included mission models (References 3-1 through 3-7); funded advanced studies (References 3-8 through 3-27); industry studies and presentations (References 3-28 through 3-44); and personal communications.

A total of 238 automated spacecraft, large observatories, sorties, orbital propulsion vehicles, manned space flight projects and spacecraft, orbital propellant depots, and maneuvering units were investigated as potential receivers. Some spacecraft are current vehicles and may not be viable candidates for fluid transfer themselves, however, they were identified since a future spacecraft or vehicle with similar functions could become a candidate for fluid transfer. Commercial spacecraft are also included as they could be potential receivers of fluid transferred during a supply mission.

The sorties and the spacelab are not in themselves considered candidates for fluid transfer herein as they are launched with the orbiter and are returned with the orbiter on the same flight. Nevertheless, the sorties and spacelab missions include some experiments and instruments which require fluid transfer during operations in space and may someday be flown as a spacecraft (or space station) experiment and hence would then become a candidate for fluid transfer from a resupply source. Sortie missions are therefore reviewed for the purpose of identifying the experiment instrumentation and supporting fluid requirements.

Often a proposed vehicle was found to have multiple design concepts (or approaches) as a result of different contractor studies and/or alternate configurations. These alternative concepts of basically the same vehicle are normally represented herein by only one concept. On the other hand, some vehicles are in direct competition with others and possibly only one concept among several presented will prevail, such as the Interim Upper Stage vehicles; however, different competing systems are included herein since the fluid type and/or quantity may vary among the candidates. Selection or identification of one candidate or concept over another in no way is intended to imply a preference nor a recommendation unless specifically stated as a recommendation.

3.2 CATEGORIZATION OF RECEIVERS

Several categorization options for the fluid receivers were considered including (1) receiver function, (2) fluid function, (3) fluid type, and (4) fluid quantity. The receiver function was chosen as the method of categorization since (a) the receivers were initially identified for studies based upon vehicle function, (b) the receiver characteristics, and future changes to those receivers can more easily be traced by using the functional categorization, (c) an operational vehicle (receiver) in space which will be resupplied nominally will have more than one fluid to be resupplied on a given mission, and the multiple fluids are easier to identify if receivers are categorized by vehicle function, and (d) the supply would most likely be sized and designed to match the receiver vehicle, particularly including the presence of man with the commensurate potentially operational differences, versatility, and safety requirements.

The resultant categories are as follows:

- Propulsion Vehicles
  - Manned
  - Automated
- Spacecraft
  - Manned
  - Large Observatory (Manned Tended)
  - Automated
- Other
  - Storage
  - Maneuvering Units
  - Experiments

3.3 RECEIVER CHARACTERISTICS

Specific receivers are selected and their pertinent characteristics are identified in Appendix A. These receivers are selected from the many potential receivers on the basis of receiver utilization of fluids, selection of representative examples from each category, selection to identify different types and different quantities of fluids, and the availability of receiver design data.

Characteristics which are identified include the fluid characteristics (function, mass, temperature, and pressure), fluid system operating characteristics (fluid zero-g feed subsystem, mass measurement subsystem), tank characteristics (thermal control, pressurization method, operating pressures, number of tanks, material, weight, length, diameter, capacity), fill and vent characteristics (zero-g vent type, fill and vent line diameters), and any other characteristics which could affect the low-g transfer system design. Data sources for each receiver are identified on the data sheets. All the above characteristics were not available due to the preliminary nature of the conceptual designs under consideration. It should also be recognized that technology has advanced since some of these studies were conducted, and a current design study might well modify the subsystems, components, and operating characteristics presented.

Near-term systems which are presented herein, such as the Shuttle and Interim Upper Stages (IUS), are based upon a ground serviced and launched design. These ground-based designs are presented herein because potential benefits could be derived from fluid transfer through modification to the vehicle to provide a capability for resupply of fluids in space.

The orbiter (and external tank data) are based on the most recent design as identified in the Shuttle Systems Definition manuals. The orbiter fluid tank designs are not final and are subject to change. There has been some consideration to a hydrogen and oxygen bipropellant reaction control subsystem (RCS) and orbital maneuvering subsystem (OMS); however, only the current design is presented. Orbiter subsystems which are not normally subject to resupply (such as auxiliary power unit hydraulics, water coolant loops, thermal control subsystems, fire extinguisher, etc.) are presented on the data sheets for information purposes and because resupply may be required to return to earth a disabled orbiter. The external tank is not normally carried into space with the orbiter, but is included since benefits may be derived by a resupplied shuttle tank.

The IUS vehicles, space tug, and manned space based tug are anticipated to be a time phase series of shuttle upper stages with appropriate capability and technology upgradings. No attempt has been made to indicate when one will replace another, nor to indicate which design from many concepts is the one most likely to be selected.

Several competing designs for the IUS are presented (Centaur, Transtage, and Agena) to indicate the diversity of potential IUS receivers. Only the reusable IUS concepts are presented since fluid resupply to an IUS would reap its greatest benefits from vehicle reusability (i.e., lower operating costs).

Both a manned modular nuclear vehicle (Mars direct landing mission vehicle) and a single (hydrogen) tank reusable nuclear shuttle vehicle are presented. Probably only a single stage reusable shuttle concept, similar to that presented, would be developed first (if at all), and then the modular vehicle would be assembled from this single vehicular design. In any case, the type and quantities of fluids for nuclear propulsion would be similar to that presented.

Two concepts for a chemical interorbital shuttle are presented, both concepts are modifications to hardware developed from other propulsion systems (Saturn V and

Space Shuttle). Similarly, two concepts for the solar electric propulsion stage and two concepts for a permanent manned space station are presented. Two concepts for each of these space systems are presented to highlight the possibility that one design may or may not be representative of an advanced system, and to identify similarities as well as dissimilarities of the proposed systems.

The space station similarities are seen in that both concepts use nitrogen resupply for repressurization and leakage make-up. Differences are seen in the RCS propellants and the resupplied fluids. The Modular Space Station required water as the resupplied fluid, which is subsequently broken down by electrolysis for the ECLSS (environmental control and lift support subsystem) and the RCS, while the Large Space Station uses gaseous oxygen as the resupplied fluid for the ECLSS and hydrazine (N_2H_4) for the RCS. Thus, oxygen used for the space station ECLSS could be resupplied in the gaseous, supercritical or liquid state, or as water (H_2O).

Three large observatory spacecraft are presented to provide a cross-section of different fluids that may have to be resupplied. In the case of the Geosynchronous Platform, only one fluid (N_2H_4) needs to be resupplied; whereas, for the Large High Energy Observatory B (Magnetic Spectrometer) and the Large X-Ray Telescope three different fluids may need to be resupplied.

Only two automated spacecraft were selected from among the over one-hundred candidates available. The Gravity and Relativity Satellite and the Upper Atmosphere Explorer were selected because of the appreciable quantity [136 to 182 kg, (300 to 400 lb)] of fluids needed for these missions, whereas most automated NASA spacecraft do not use such large quantities of fluids. It should be noted that the Upper Atmosphere Explorer is a spin stabilized spacecraft and not designed for resupply; however, a similar spacecraft could be modified for resupply by utilizing a spin/de-spin subsystem or three-axis stabilization.

One concept of an orbital propellant depot is presented. This concept was based on utilizing the SII stage system, whereas a current design probably would be based on the Shuttle External Tank design. Nevertheless, the propellants and quantities would be similar to that presented.

Two maneuvering units, manned and automated, are presented. It is interesting to note that both the Space Taxi (1966 study) and the Remote Maneuvering Unit (1973 study) use hydrazine (N_2H_4) monopropellant as the main propulsion fluid.

Six experiments are presented, although only sketchy detail was available during the course of this study, to indicate the variety of fluids and fluid applications that may be utilized in space. These experiments are being considered for sortie (spacelab) flights and will not be resupplied in the sortie flight mode; however, a sortie flight of an experiment can be a harbinger of future automated or large observatory spacecraft experiments, and automated and large observatory spacecraft are candidates for space resupply. It should be noted that the fluids for the remote sensing platform are listed as cryogenic because the specific fluid has not been selected at this time, and this is interpreted as meaning that any appropriate cryogenic such as LN_2, LNe, LH_2 or LHe might be selected..

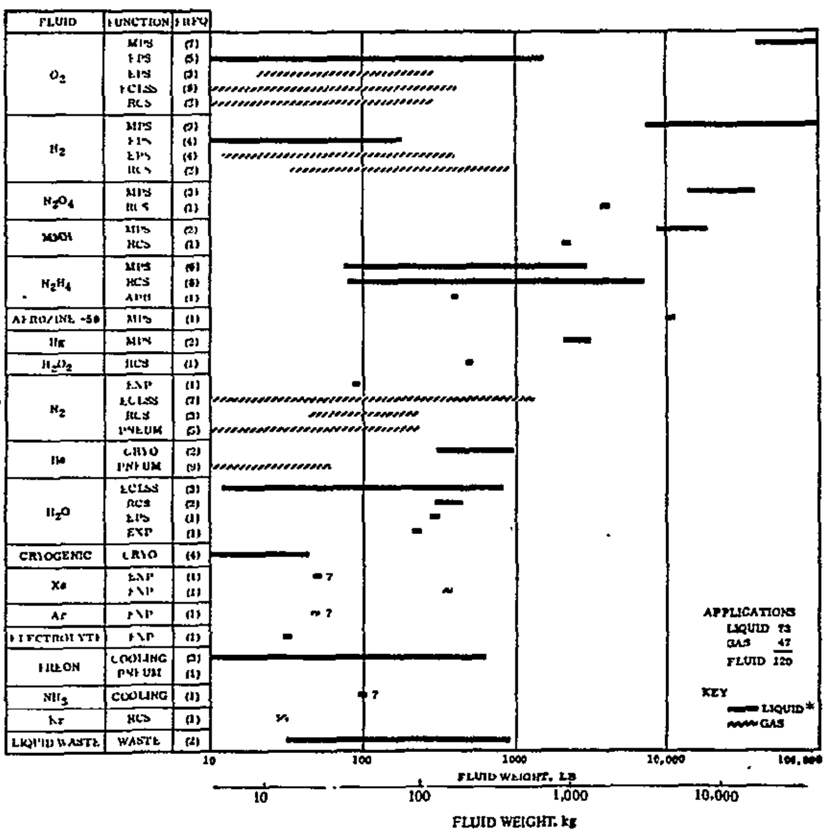
The electrolyte for the biological experiments are presented because of special handling characteristics which may be inherent to this fluid. Also identified as a candidate for low-g fluid transfer is liquid waste water. Normally liquid waste water may be dumped into space; however, some instrumentation (e.g. optics) may be too sensitive and/or false readings may result from having water vapor and contaminants in the area where observations are being made. Therefore, liquid waste water may have to be transported away from the space station or spacecraft.

3.4 OBSERVATIONS CONCERNING TYPES OF SPACE VEHICLE FLUIDS

Fluid functions and quantities are compiled based upon the characteristics of the selected receiver vehicles and used as a guide for recommending vehicles to be studied in the following task. Compilation of fluid characteristics for the selected sample is presented in Figure 3-2.

It is cautioned that the fluid characteristics identified are based upon a small selected sample of only 29 potential receiver vehicles and experiments. The sample itself is selected to provide a cross-section of different fluid types, different fluid quantities, and examples from different receiver categories and viable vehicles, as discussed in Paragraph 3.3. It is also cautioned that the frequency of resupply (pounds per year) must be considered to quantitatively evaluate the benefits of resupply, but frequency of resupply is not considered herein. The sample also includes some systems which are

FLUID QUANTITY AND FUNCTIONS (BASED ON SAMPLE OF 29 RECEIVERS)



* Includes supercritical fluids

Figure 3-2. Sample Fluid Quantity and Function duplicative (e.g. chemical interorbital shuttle, various IUS concepts, etc.) and development of one system could eliminate the further consideration of others. Nevertheless, with these cautions in mind, some valid observations may be made concerning potential fluids in space.

The fluid applications can be compiled into eight functions and seventeen fluids as presented in Table 3-1.

Table 3-1. Fluid Functions and Identification

EIGHT FLUID FUNCTIONS IDENTIFIED

- MPS - MAIN PROPULSION SUBSYSTEM (INCLUDES OMS-ORBITAL MANEUVERING SUBSYSTEM AND PROPELLANT STORAGE)
- RCS - REACTION CONTROL SUBSYSTEM
- EPS - ELECTRICAL POWER SUBSYSTEM
- ECLSS - ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM
- THERMAL CONTROL SUBSYSTEM
- PNEUMATICS
- EXPERIMENTS (INCLUDES DETECTOR COOLING)
- LIQUID WASTE

SEVENTEEN DIFFERENT FLUIDS IDENTIFIED

NINE LIQUID* ONLY		SIX LIQUID* AND GAS		TWO GAS ONLY
N_2O_4	H_2O_2	O_2	He	AR
MMH	H_2O	H_2	Xe	KR
N_2H_4	NH_3	N_2	FREON	
Hg	ELECTROLYTE			
	AEROZINE-50			

* Includes supercritical fluids

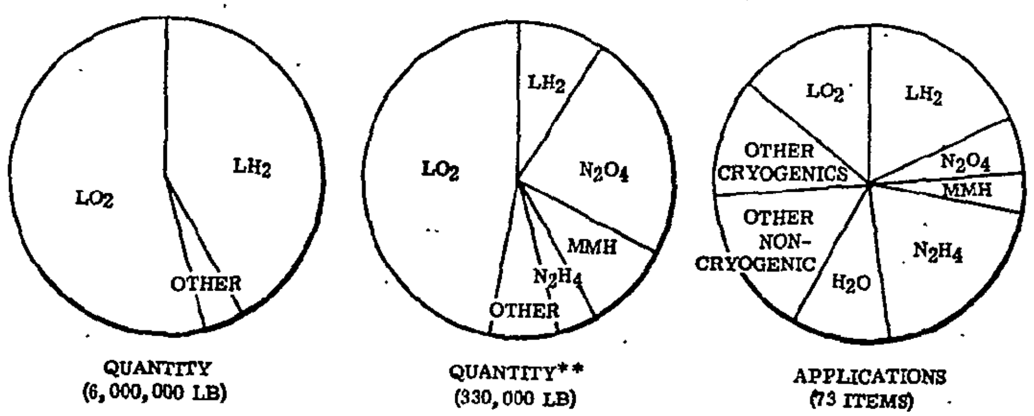
The main propulsion subsystem (MPS) function includes the shuttle OMS since the shuttle in-space maneuvering propulsion is performed with the OMS. The propellants stored in an orbital propellant depot are also considered as having an MPS function since their eventual usage will be for propulsion in a tug, chemical interorbital shuttle, or nuclear shuttle. Fluids used as cryogens to cool experiment detectors are considered as experiment fluids. However, the function of fluids (e.g., freon) used to control the temperature of a vehicle are classified as a thermal control function.

Table 3-1 also lists the 17 different fluids that are identified in the sample. Of the 17 fluids, nine are identified in the liquid state only, two in the gas state only, and six are identified in both the liquid and gas state. It is noted that some experiments called for cryogens for detector cooling without identifying which cryogenic. Liquid neon (LNe) is one such potential cryogen which is not included in Table 3-1. There has been consideration of slush hydrogen for propulsion, and solid cryogens such as solid hydrogen for use in cooling experiment detectors, neither of which is included in Table 3-1.

A summary of the liquid quantities and applications for the 15 liquids identified are shown in Figure 3-3. It can be seen that on a quantity (mass) basis, liquid oxygen and liquid hydrogen dominate the lqiuid usage. However, a large portion (95%) of the liquid quantity in this sample is composed of propellants for the shuttle external tank, chemical interorbital shuttle, nuclear shuttle, and the orbital propellant depot. Removal of these large propellant receivers shows that liquid oxygen is still a dominant liquid.

The summary of the applications of liquid indicates wide usage (20%) of hydrazine (N_2H_4) in the sample. An application is considered herein as the usage of a particular fluid to perform a function (e.g., reaction control) in an identified potential receiver (e.g., Upper Atmosphere Explorer). Thus, since N_2H_4 is used for both the main propulsion (orbit adjustment) and the reaction control for the Upper Atmosphere Explorer, then two applications are counted; whereas N_2H_4 application for the Large X-Ray Telescope is counted as one since N_2H_4 is used as the orbit adjust fluid and nitrogen gas (N_2) is used for the reaction control. It is observed that the application of cryogenic and non-cryogenic liquids are about equal in the sample selected.

A histogram of the number of receivers that utilize fluids compared to the number of fluids per receiver is shown in Figure 3-4. It can be seen that out of the sample of 29 receivers that over 80% use more than one fluid, the mode is two fluids per receiver, and the arithmatic mean and the median is three fluids per receiver. The one receiver with ten fluids shown on Figure 3-4 is the shuttle orbiter.



15 DIFFERENT LIQUIDS
29 DIFFERENT VEHICLES & EXPERIMENTS

*INCLUDES SUPERCRITICAL FLUIDS

**WITHOUT SHUTTLE EXTERNAL TANK, NUCLEAR SHUTTLE, CHEMICAL INTERORBITAL SHUTTLE, ORBITAL PROPELLANT DEPOT

Figure 3-3. Summary of Liquids* Identified

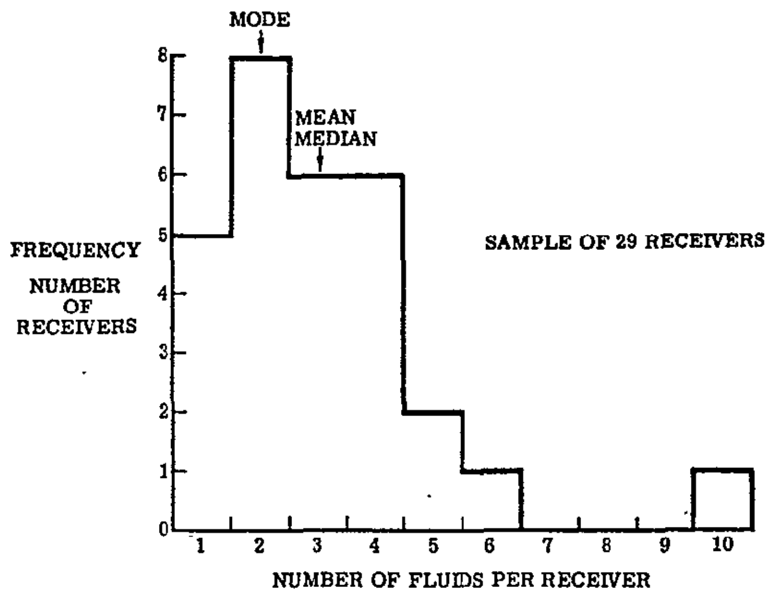


Figure 3-4. 80% of Receivers Utilized More Than One Fluid

3.5 BENEFITS OF FLUID TRANSFER

Planned and potential future space operations cover a wide range of activities from earth orbital operations to lunar and planetary operations, as sketched in Figure 3-5.

Earth orbital and interorbital operations include automated spacecraft, large observatory, space station, propellant storage, space basing, and spacecraft servicing operations. Lunar and planetary operations include lunar space basing and exploration, and automated and manned planetary exploration. Some of these operations are now being performed in space, but increased capabilities will result as well as new opportunities opened with the emergence of newer systems such as the shuttle and tug. These planned and potential future space operations as well as the planned and potential future space vehicles were investigated, and the potential benefits of fluid transfer in space are categorized and identified.

The potential benefits of fluid transfer which are identified are found to be in the general category of cost effectiveness or safety. Cost effectiveness means that the benefit is a reduction in cost, increase in life time, provides reusability, performance improvement, or a desirable capability is attained. The identified benefits which may be attained from fluid transfer are summarized in Figure 3-6.

The potential benefits are discussed in further depth in following paragraphs. Quantitative evaluation of these benefits is not intended. Benefits are identified as potential benefits because quantitative evaluation may be required to evaluate whether the benefit could be realized. Specific quantitative benefits for Space Tug and/or

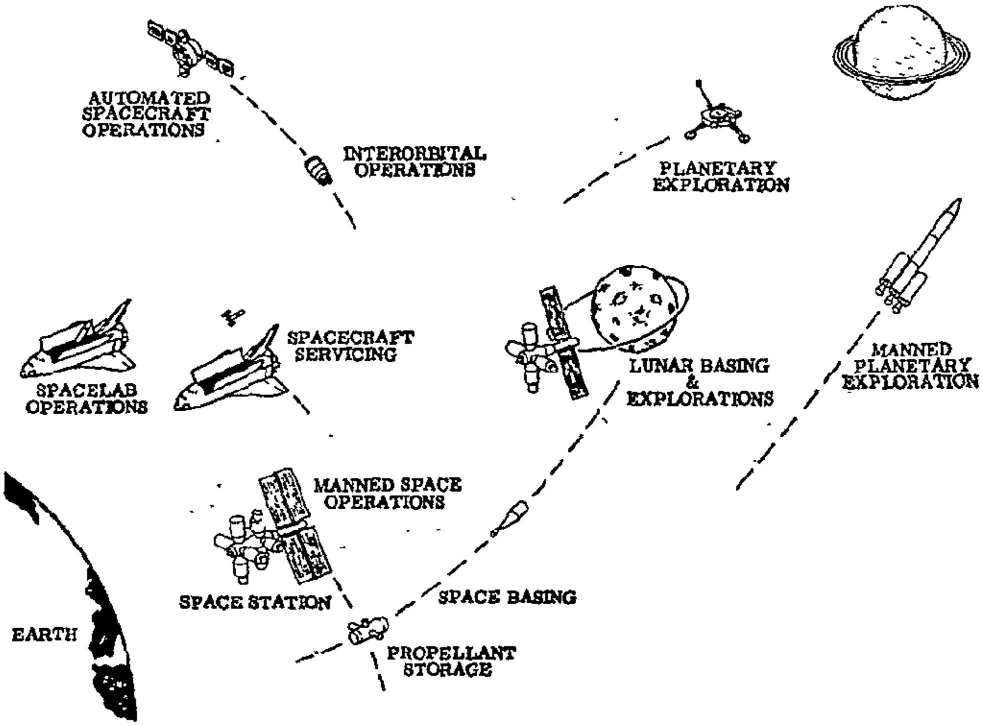


Figure 3-5. Planned and Potential Future Space Operations

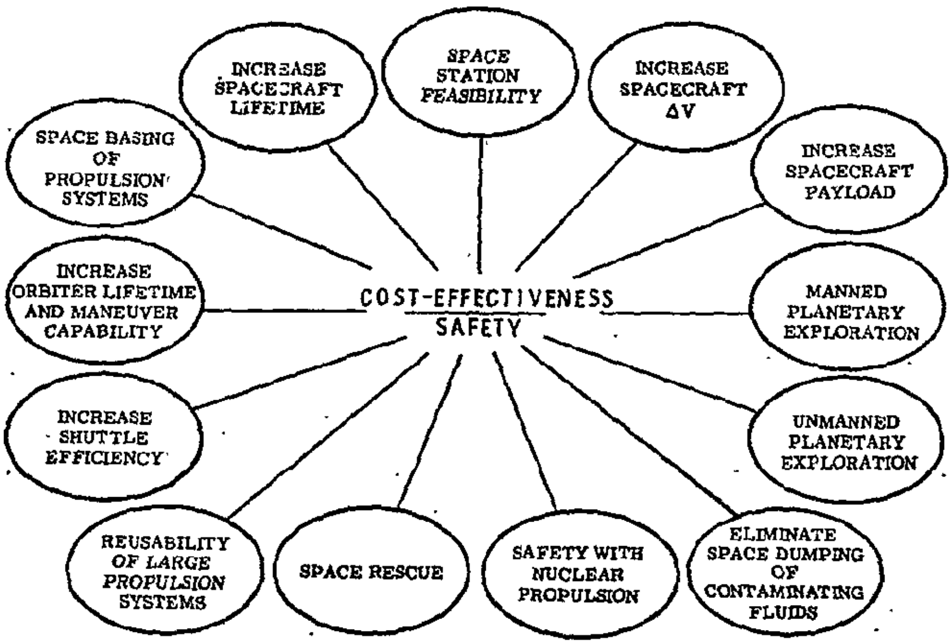


Figure 3-6. Potential Benefits of Fluid Transfer

Shuttle Orbiter resupply were developed and are presented in Section 6. Only non-DoD missions are referred to herein. Significant benefits may be derived by the DoD utilizing fluid transfer, but they are beyond the scope of this study.

3.5.1 SPACE BASING OF PROPULSION SYSTEMS - Fluid transfer in space is inherent to space basing of propulsion systems. Space basing of propulsion has been considered as a means to reduce the costs of space operations by not having to transport the Tug (or other such propulsion system) from earth to orbit and then return to earth for refurbishment and refilling. Space basing means having the propulsion vehicle stay in orbit and then only the propellants and payload would be transported to orbit in what is intended to be more efficient operations.

Propellants can be stored in space until the propulsion vehicle rendezvous with the storage system for propellant resupply, or propellants can be transfered directly to the propulsion vehicle after orbit insertion of the resupply vehicle. Propellant storage in space implies an orbital storage system in specified orbits and operationally will require the propulsion vehicle to initiate and terminate its missions from these specified orbits. Cost savings with propellant depots can be derived if sufficient missions are flown to amortize the development and operations of a space storage system, and overcome the inefficiency for any orbital plane changes imposed upon the propulsion stage. If propellants are transferred to the propulsion vehicle via the shuttle only when required, then the traffic rates may not have to be as high for amortization of the inherently lower development costs.

The major propulsion systems for which space basing is planned include the reusable space based tug, reusable nuclear shuttle, and the chemical interorbital shuttle. The interim upper stage and the space tug are planned to be fueled at the ground launch facilities, but space basing of these systems is feasible. The fluids which are predominant candidates for fluid transfer for space basing of propulsion systems are liquid oxygen and hydrogen. Other fluids, such as helium, are required dependent upon the specific orbital system which is space based.

3.5.2 INCREASE SPACECRAFT LIFETIME - The lifetime of some satellites may be increased with fluid resupply. A spacecraft lifetime is often determined by the fluid quantity available on-board. When this fluid is expended, the spacecraft (or on-board experiment) is no longer useful. The functions that these spacecraft fluids perform include attitude control, orbit maneuvers, drag makeup, electrical power generation, and/or experiment support.

Fluid resupply to spacecraft could be attained by retrieving the satellite with the shuttle and/or tug and returning it to earth, replenishing the expended fluids, and then relaunch of the spacecraft. However, in-orbit fluid transfer may prove to be more economical as orbital refueling could result in less shuttle payload bay space required, or possibly less launch weight required, and thus allowing more efficient use of the shuttle. Retrieval of a spacecraft will require refolding of solar arrays or other protuberances. Retrieval of spacecraft implies that spacecraft operational data will be lost during the time that it is being refurbished on earth unless a replacement satellite is utilized (added cost), whereas in-orbit fluid transfer will minimize spacecraft down-time.

The IUS currently is planned as expendable and no payload requiring IUS performance is retrieved, and for these satellites there is no alternative to fluid supply other than replacement. During the tug era, some satellites such as heavy synchronous equatorial satellites may not be retrievable by the tug, and these satellites which have fluid limited lifetimes must be replaced by new satellites if fluid transfer is not available.

Spacecraft which have been identified as using fluids include most of the large observatores, and many automated spacecraft including the Gravity and Relativity Satellite and Upper Atmosphere Explorer which are presented in Appendix A. Fluids which are potential candidates for fluid transfer for the purpose of increasing spacecraft lifetime include hydrazine, liquid helium, and nitrogen.

3.5.3 SPACE STATION FEASIBILITY - Fluid transfer in orbit is required for space station operations. A space station will require consumables for life support and attitude control. Consumables are also required in some space station designs for electrical power and experiment support.

Consumable usage in space station operations has been studied, and alternative subsystems have been proposed to recycle consumables as a means to reduce consumption. As an example, water is resupplied to the modular space station, broken down by electrolysis into hydrogen (used in the reaction control system and fuel cells for power) and oxygen (used for station repressurization, reaction control system, fuel cells for power, and EVA resupply). The water from the fuel cells are then used for drinking and recycled through the electrolysis unit. Nitrogen gas must be replenished due to space station leakage.

Consumables transferred to the space station are primarily nitrogen, oxygen and/or water. The nitrogen and oxygen fluids may be supplied either in the cryogenic or gaseous state dependent upon the space station concept. Other fluids which may be transferred could include hydrazine (attitude control/drag makeup) and freon (cooling reservoir and fire control).

3.5.4 INCREASE SPACECRAFT MANEUVER VELOCITY (ΔV) - Spacecraft maneuver velocity capability may be increased with fluid transfer. Spacecraft with maneuver ΔV capability, such as the Upper Atmosphere Explorer, can be resupplied before its onboard propellant is exhausted and thus increase its useful life. This additional velocity capability can be used for drag makeup, orbit phasing, and orbital plane changes. Thus, resupply propellant capability to the Upper Atmosphere Explorer could mean increased lifetime and/or a lower perigee altitude allowing deeper penetration into the atmosphere.

An alternative to fluid transfer is retrieval of the spacecraft and return to earth for refueling. Return of the spacecraft to earth will mean a time break in the operational data unless a second spacecraft is used to replace the first, but launch and retrieval with one or two spacecraft may mean increased costs and less efficient use of the shuttle.

Hydrazine was observed to be the most predominant of the maneuvering propellants used by the spacecraft.

3.5.5 INCREASE SPACECRAFT PAYLOAD - Spacecraft payload weight may be increased for some spacecraft through the utilization of fluid transfer. In the preshuttle era, a spacecraft which utilizes an appreciable quantity of fluid and which is payload weight limited due to launch vehicle limits might be aided by fluid transfer. The payload weight could be increased in this case by offloading the main spacecraft fluid tank and launching with the same gross weight (increased payload) and partially empty tanks which are later filled via in-orbit fluid transfer. The benefit attained would be increased payload. An additional benefit could also be lower cost by launching with two less expensive (or existing) launch vehicles instead of one larger launch vehicle (more cost if development is required).

Payload may also be increased similarly in the shuttle era. A payload carried by the shuttle upper stage (IUS or tug) may exceed the weight carrying capability of the shuttle to the desired orbit. The payload weight could be increased in this case by offloading the shuttle upper stage to stay within the shuttle launch constraints, and then topping-off the upper stage tanks in orbit from a second shuttle flight before proceeding with the mission. An alternative to fluid resupply is to launch the payload in one shuttle, the upper stage in another shuttle and mate the upper stage and payload in orbit.

The fluid which would be the principal candidate for transfer to the spacecraft would be hydrazine, while the fluid for transfer to the shuttle cryogenic upper stage or tug would be oxygen and possibly hydrogen. Oxygen is suggested as possibly the only fluid transferred since its weight savings would be greater than that of hydrogen and this weight savings alone would probably be sufficient to meet shuttle weight constraints. Thus, the hydrogen would be loaded on the tug at launch and the oxygen transferred in orbit. Other potential fluids would be those fluids which are used by the non-cryogenic interim upper stage candidates, such as nitrogen tetroxide (N_2O_4), Aerozine -50, or mono-methyl-hydrazine (MMH).

3.5.6 INCREASE ORBITER LIFETIME AND MANEUVER CAPABILITY - The on-orbit lifetime and maneuver capability of the shuttle orbiter can be increased with fluid transfer. Currently, the orbiter is being designed for a nominal orbit lifetime of seven days, and 30-day lifetime with reduced orbiter capability. The lifetime of the orbiter is controlled by the available fluids (oxygen and nitrogen) used for the environmental control and life support subsystem (ECLSS), oxygen and hydrogen used for experiment electrical power subsystem (EPS), and/or nitrogen tetroxide (N_2O_4) and mono-methyl-hydrazine (MMH) used for the orbital maneuvering subsystem (OMS) and reaction control subsystem (RCS).

Orbiter lifetime is usually increased from the nominal seven-day mission by adding kits with the required fluids for the mission. However, adding kits (increased weight) reduces the weight available for payload; also space in the cargo bay is reduced if OMS kits are added. Thus, increased lifetime and/or maneuver capability could be increased for the orbiter with fluid transfer as an alternative to reduced orbiter payload weight capability and/or payload volume.

Orbital plane changes and increased orbital altitude capabilities may be attained by retaining the external tank with the orbiter and resupplying it with liquid after orbital insertion. This added maneuver capability would be helpful to perform manned spacelab flights in orbits with a higher altitude than those within the current orbiter capability. Plane changes could be beneficial for multiple orbiter passes over selected earth targets in earth resources and surveillance missions (e.g. disaster flood, etc.) where frequent passes within a short time period are desirable. Multiple passes within a short time period (one day) may be useful to attain observations of short period changes, attain different lighting conditions, or to make observations under the correct weather conditions.

3.5.7 INCREASE SHUTTLE EFFICIENCY - Shuttle efficiency can be increased with fluid resupply. Without fluid transfer, large observatory spacecraft would have to be retrieved and relaunched to extend their lifetime after they exhausted their attitude control propellants and/or cryogenics used in experiments. The large observatories take a large portion of the cargo bay in addition to taking a significant portion of the payload weight capability, thus reducing shuttle flight capability for multiple (efficient) payload launches. However, with a fluid resupply capability, the fluid resupply tank could take less volume and weight from a shuttle flight,providing greater opportunity for multiple missions and more efficient utilization of the shuttle.

All large observatories use fluids in some function. These fluids which are candidates for fluid resupply to the large observatories include liquid helium, xenon, nitrogen, and hydrazine.

3.5.8 REUSABILITY OF LARGE PROPULSION SYSTEMS - Fluid resupply is required to reuse large propulsion systems, and thus to provide more efficient space operations. Many large propulsion systems have been proposed for future space operations including the chemical interorbital shuttle, nuclear shuttle, and manned planetary exploration vehicles. These propulsion stages would be launched with a large expendable launch vehicle (probably a derivative of the orbiter booster) and could not be returned to earth in the orbiter. Thus, economics (i.e., reusability) of large propulsion systems will require orbital fluid transfer capability before manned planetary exploration or lunar space basing is economically feasible.

The primary fluids which must be transferred in space to provide reusability of large propulsion systems are liquid hydrogen and oxygen. Additional fluids such as helium may be required for systems support (e.g., pneumatics).

3.5.9 SPACE RESCUE - Space rescue capability can be increased for orbits beyond the nominal range of the orbiter with low-g fluid transfer. A vehicle similar to the tug (or other interorbital vehicle such as nuclear shuttle, etc.) which transported the men to their current whereabouts might be used for the rescue. However, special rescue equipment might have to be transported to perform the rescue. This added weight might exceed the carrying capability of the orbiter with fully loaded tug, or put the performance requirement out of the nominal range of a single tug transported to earth orbit; thus, multiple shuttle flights would be required to perform the rescue mission.

Perchonok has shown (References 3-42 and 3-43) that the orbiter may be used for rescue from lunar orbit or geosynchronous orbit with orbital refueling of the external tank. Efficient refueling for rescue with the orbiter implies using a propellant depot (fluid transfer required) or capability to launch large quantities (0.68 Gg, 1.5 M lb) of propellants (fluid transfer required) since the number of flights, and thus time consumed, to refuel the external tank from the orbiter would be prohibitive for rescue time-critical operations. An alternative to the fluid transfer mode would be to rendezvous and dock the orbiter to a loaded external tank launched separately.

Fluids which would be transferred in space in support of space rescue would predominantly be liquid oxygen and liquid hydrogen.

3.5.10 SAFETY WITH NUCLEAR PROPULSION - The introduction of nuclear propulsion will bring with it the radiation problems that result after a nuclear engine has been used. Radiation safety is aided by parking the nuclear shuttle vehicle in a holding orbit to allow sufficient engine cooldown. Keeping the nuclear engine in orbit for purposes of safety requires that fluid be transferred to the nuclear vehicle in orbit. Safety with nuclear propulsion is a benefit attained with fluid transfer and is presented as such, although the anticipated large size of nuclear vehicles is probably a more compelling argument for in-orbit fluid transfer.

Liquid hydrogen is the fluid which would require transfer in orbit to support nuclear propulsion.

3.5.11 ELIMINATE SPACE DUMPING OF CONTAMINATING FLUIDS - The purpose of many operations in space is to make observations of a remote target (earth, planet, or star) and measure its radiation. Fluids near the observing platform and in the path of the incoming radiation may cause incorrect readings and may act as a filter. Waste fluids dumped overboard could also result in a residual coating on sensitive optics. Waste fluids which may be generated onboard a space station, a biological facility, or a space manufacturing facility may have to be transferred either back to earth or away from the facility before they can be dumped. Temporary on-board storage and selective dumping may be acceptable for some spacecraft, but for facilities where observations may be effected, fluid transfer of waste fluids will be required.

Waste fluids would primarily include water and electrolytes from biological and space manufacturing activities.

3.5.12 MANNED PLANETARY EXPLORATION - Manned exploration of the planets has long been one of man's goals. One day in the post-shuttle era man will set out from earth and journey to the planets, his main propulsion most likely will be nuclear,and fluid transfer in space will probably be required. An alternative to fluid transfer would require the launching of loaded modular stages by a large (probably expendable) booster, assembly in orbit of these modular stages, and an assembly (storage) period of probably months before the launch to a planet. The economics, as well as the technology and systems available at the time, is expected to dictate that fluid transfer be employed in support of manned exploration of the planets.

The major fluids which will be used for manned planetary exploration, and thus fluid transfer, include liquid hydrogen for propulsion plus oxygen and nitrogen for the environmental control and life support subsystems.

3.5.13 UNMANNED PLANETARY EXPLORATION - Unmanned planetary exploration could benefit from fluid transfer in orbit. Future planetary missions will utilize spacecraft which are heavier than those flown today, and all planetary missions will obviously have to use a shuttle upper stage (e.g., Tug). Some of these spacecraft, such as the (Mars) Satellite Sample Return spacecraft, plus a fully loaded tug (Reference 3-9) will require offloading the Tug to stay within the shuttle performance capability limits. Offloading the tug is expected to be the nominal approach to meet the launch capability of the Shuttle. However, if the spacecraft should increase in weight as has historically been the trend, then the shuttle launch capability limitation may require offloading more propellants from the tug than would be acceptable to make the mission. A fluid transfer capability means that the tug could be fueled in orbit, allowing the mission to be flown with increased spacecraft weight up to the fully loaded tug performance capability limit.

The alternatives to fluid transfer include constraining the payload weight (may cost more money and/or result in reduced experimental data return), or mating the payload and the tug in orbit.

The propellant most appropriate for fluid transfer to the Tug would be liquid oxygen. The Tug uses both liquid oxygen and liquid hydrogen, offloading the liquid oxygen only, allows up to 21,600 kg (47,500 lb) of payload to be carried. Liquid oxygen is also selected because the liquid oxygen in the refueling flight requires less volume than liquid hydrogen, thus allowing more room for other payloads to be carried for efficient use of the shuttle. However, further analysis is needed before selection of transferring liquid-oxygen-only, since the time waiting for the refueling vehicle to rendezvous with tug may conflict with the storability capability of liquid hydrogen in the tug.

3.5.14 CONCLUSION - There are many benefits which may be derived from fluid transfer in space, including space station feasibility, increased spacecraft lifetime, increased spacecraft maneuver velocity (ΔV), increased spacecraft payload, space basing of propulsion systems, increased orbiter lifetime and maneuver capability, increased shuttle efficiency, reusability of large propulsion systems, space rescue, safety with nuclear propulsion, eliminating space dumping of contaminating fluids, and manned and unmanned planetary exploration. In general, cost effectiveness (reduced cost, increased performance and/or mission capability) and safety are the measurement of fluid transfer benefits. It is concluded that development of in-space fluid transfer technology will avail many cost-effectiveness and safety benefits for exploitation and exploration of space.

3.6 RECOMMENDATIONS FOR TRANSFER SYSTEMS STUDIES

Receiver vehicles to be used in support of the Transfer Systems Study activity are recommended. The recommendations were originally to include only cryogenic fluids; however, non-cryogenic fluids are also recommended for further study based on the observations (Paragraph 3.4) of the potential space fluids and the realization that a receiver vehicle with cryogenics often also contains non-cryogenics. It was found that the mean (and median) is three different types of fluids in each receiver out of the sample of 29 receivers, as shown in Figure 3-4.

The receivers recommended for further study are therefore based on both cryogenic and non-cryogenic categories. The systems are selected to be representative of others by fluid type and quantity, but a difference in fluid type among the recommended receivers is desirable. Finally, the selection is based on the receiver having a reasonable likelihood of being flown in the next decade.

3.6.1 CRYOGENIC RECEIVERS - Five cryogenic receivers are recommended as presented in Table 3-2. The shuttle external tank is considered separate for the orbiter because of the large fluid quantities required for resupply. The shuttle orbiter is an excellent candidate because it is now being developed and requires cryogenics (H_2 and O_2). The orbiter is not designed for in-orbit fluid transfer, but longer on-orbit life of the orbiter (e.g., spacelab/sortie missions) can prove beneficial.

The shuttle external tank is considered an excellent receiver to include since a direct benefit would be additional orbital ΔV maneuver capability for the orbiter. It is also possible that the chemical interorbital shuttle tank, the nuclear shuttle tank,

Table 3-2. Recommendations for Cryogenic Receivers

Receiver Vehicle	Cryogenic Fluid	Fluid Weight kg		Fluid Weight lb		Comments
Space Shuttle Orbiter* (With Kits)	H_2 O_2	84 709	(334) (2837)	184 1563	(736) (6248)	• Orbiter in Development
Space Shuttle External Tank	LH_2 LO_2	100,788 604,274		222,000 1,331,000		• Large Fluid Requirements • Similar to Other Large Propulsion Systems Such as Chemical Interorbital Shuttle, Nuclear Shuttle, and Storage Depot Tank • In Development
Space Tug *	LH_2 LO_2	3,462 19,783		7,626 43,574		• Near Term NASA Planning • Space Based System Will Probably Evolve From Ground Based Design
Modular Space Station*	H_2 O_2 N_2	596 44 620		1,310 96 1,365		• Likely Post Shuttle Development • Supplied in Cryogenic State a Possibility
Large High Energy Observatory B (Magnetic Spectrometer)	LHe Xe N_2	431 168 200		950 370 221		• Near Term NASA Planning • Typical of S/C and Experiment Cryogenic Requirements

* Other Noncryogenics Required

and/or the storage depot tank may be based on modifying and utilizing the external tank as a means of reducing development and production costs.

The Space Tug is recommended because the space transportation system inherently requires a shuttle upper stage, and the Space Tug is NASA's planned vehicle to fill that primary role (the interim upper stage is planned to be just interim until the tug is developed).

The modular space station is recommended because a manned space station has long been in NASA long range plans, and with the shuttle era the transportation economics makes a space station more viable. The oxygen, hydrogen, and nitrogen fluid are utilized in the space station as gases. However, these fluids could be resupplied in the cryogenic state.

The Large High Energy Observatory B (magnetic spectrometer) is recommended because it is a typical large observatory spacecraft, requires a significant quantity of cryogenics, and requires different cryogens (i.e., helium and xenon) from the other recommended receivers.

3.6.2 NON-CRYOGENIC RECEIVERS - The recommended non-cryogenic receivers are presented in Table 3-3, and include two receivers, the shuttle orbiter and the modular space station, which are recommended for investigation as cryogenic receivers. They are also recommended for non-cryogenic investigation because these vehicles require non-cryogenics to function and resupply of one (cryogenics) without the other (non-cryogenics) will likely not result in the full realization of transfer benefits.

Table 3-3. Recommendations for Non-Cryogenic Receivers

Receiver Fluid	Non-cryogenic Fluid	Fluid Weight - kg	lb	Comments
Space Shuttle Orbiter*, ** (with kits)	N_2O_4	8907 (19488)	19620 (42924)	• Orbiter in Development
	MMH	5418 (11805)	11933 (26003)	
	He	42 (104)	92 (230)	
	N_2H_4	263	579	
	N_2	?	?	
Modular Space Station*, **	H_2O	451	993	• Likely Post-Shuttle Development
	Freon	274	604	
Satellite Control Section	N_2H_4	1508	3322	• Operational System • Characteristics of N_2H_4 3-Axis Stabilized Spacecraft Exhibited
	N_2	20	45	
Solar Electric Propulsion Stage	Hg	1498	3300	• Potential Post-Shuttle Development • Differing Fluid Characteristics From Others Selected
	H_2H_4	36	80	
	Kr	14	30	
	Freon 113	5	10	

* Vehicle also recommended for study under cryogenic receiver category.
** Not all vehicle fluids listed, only those which would probably need replenishing.

The Satellite Control Section is recommended because it is an operational system and requires a significant quantity of hydrazine. Hydrazine was found to have the most applications in the sample receivers surveyed.

The Solar Electric Propulsion Stage is recommended because of its unique fluids, and it is still under consideration for development because of its high performance (specific impulse) characteristics. Solar electric propulsion has been considered for orbital operation with trips to and from higher orbital altitudes. A solar electric propulsion stage could be economical for some of these orbital operations and is therefore a good candidate for fluid transfer.

4

TRANSFER SYSTEMS STUDIES

This section covers the work performed to conceptually define overall in-orbit fluid transfer systems which are most likely to provide efficient and predictable in-orbit fluid supply. General ground rules used are presented in Appendix B. The supply modules or fluid tankers are assumed to be payloads of the Space Shuttle manned transportation system having a maximum allowable payload of 29,510 kg (65,000 lb). The overall payload bay size is 18.3 m (60 ft) long by 4.6 m (15 ft) diameter.

Based on the results presented in Section 3, the three receiver systems listed in Table 4-1 were chosen for supply system design. Also presented in Table 4-1 are the basic fluids and fluid quantities to be supplied along with other receiver characteristics pertinent to transfer system design.

The Space Tug was taken to be representative of high energy upper stages requiring fairly large quantities of cryogenic fluids and small amounts of auxiliary fluids. The Space Shuttle Orbiter is a near term vehicle which could benefit from in-orbit fluid supply and is also representative of space systems where a number of different fluids (both cryogenics and non-cryogenics) may be supplied in intermediate quantities. Since it is not possible to supply all the fluids listed in Table 4-1 to the Orbiter in a single supply mission, two different transfer cases were considered for supplying this receiver. Case 1 assumes the supply of all OMS fluids and GHe with no H_2 and O_2 supplied. Case 2 assumes the supply of all H_2, O_2 and GHe with the N_2O_4 and MMH off-loaded to the extent necessary to meet the 29,510 kg (65,000 lb) Shuttle payload limitation. The multiple receivers supply system covers cases where several small receivers containing a variety of fluids are to be supplied in a single transfer mission.

The work performed is divided into (1) initial definition and screening to determine the best method(s) of liquid acquisition for each transfer system and (2) overall conceptual system definitions to the extent necessary to identify associated technology, critical system characteristics, components and operational constraints.

Initial definition and screening data are presented in the following paragraph.

4.1 INITIAL DEFINITION AND SCREENING

Weight, performance and operations data were generated for a number of different acquisition concepts designed to supply each of the fluids and receivers listed in Table 4-1. Comparisons were then made between each of the concepts and the "best" one chosen for each transfer case. The only limitation was that, in total, a minimum of three different liquid acquisition concepts were to be selected.

Table 4-1. Basic Receivers for Transfer Systems Design

Receiver(s)	System(s) Supplied	Fluid	Total Amount Supplied kg	Total Amount Supplied lb	Single Tank Volume m^3	Single Tank Volume ft^3	No. of Tanks	Tank (1) Material	Single Tank Weight (1) kg	Single Tank Weight (1) lb	Tank Maximum Fill Pressure kN/m^2	Tank Maximum Fill Pressure psia	Initial Wall Temp.(5) °K	Initial Wall Temp.(5) °R	Ext. Heating watts	Ext. Heating Btu/hr
Space Tug	Main Propulsion(3)	LH_2	3462	7626	49.5	1748	1	Al Aly	228	502	152	22	256	460	39.6	135
	Main Propulsion(3)	LO_2	1978	43574	18.1	640	1	Al Aly	133	294	141	20.5	256	460	27.0	92
	Auxiliary Propulsion	N_2H_4	153	336	-	-	1	-	-	-	-	-	256	460	-	-
	Tank Pressurization	GHe	4.1	9	-	-	1	CRES	-	-	22754	3300	256	460	-	-
Space Shuttle	OMS	N_2O_4	17633	38840	-	-	5	Ti	-	-	-	-	-	-	-	-
Orbiter With Kits(2)	OMS	MMH	10646	23450	-	-	5	Ti	-	-	-	-	-	-	-	-
	OMS Pressurization	GHe	104(4)	230(4)	0.48	16.8	5	Kevlar Wrapped Ti	133	294	33096	4800	311	560	-	-
	EPS & ECLSS	LO_2	2837	6248	0.32	11.3	8	Inconel 718	42.2	93	6550	950	350	630	Low	Low
	EPS	LH_2	334	736	0.61	21.7	8	Al Aly	33.6	74	1965	285	350	630	Low	Low
Multiple Receivers																
SEPS	Propulsion	Hg	1498	3300	0.029	1.02	4	CRES	18.2	40	190	27.5	-	-	-	-
Large HEAOB	Magnet Cooling	LHe	431	950	3.6	128	1	Al Aly	445	981	110	16.0	267	480	Low	Low
SCS	Propulsion	N_2H_4	1508	3322	2.4	85	1	Al Aly	-	-	2137	310	-	-	-	-

NOMENCLATURE:

- OMS = Orbit Maneuvering System
- EPS = Electrical Power System
- ECLSS = Environmental Control Life Support System
- SEPS = Solar Electric Propulsion Stage
- HEAOB = Large High Energy Observatory B
- SCS = Satellite Control Section

NOTES:

(1) Equivalent values used for calculating fluid chilldown requirements

(2) Two supply cases considered; (a) complete OMS supply with no EPS or ECLSS, and (b) complete EPS & ECLSS with OMS off-loaded as required to meet the 29,510 kg (65,000 lb) payload limitation.

(3) Also includes that required for electrical power supply (fuel cells).

(4) Numbers shown are based on the latest data available. For initial definition and screening 56 kg (123 lb) was used. This did not effect the final study results.

(5) Based on estimates of maximum receiver wall temperatures which could exist at initiation of chilldown, used for calculating fluid chilldown requirements.

Capillary acquisition, fluid rotation, positive expulsion (bladder, bellows, diaphragm), linear acceleration and thermodynamic methods of acquisition were considered. Storage tank, liquid residual, acquisition hardware, and power supply weights were generated for each system. Other systems, such as tank pressurization, thermal control, fluid lines and support structure were not included, since they were assumed to be similar between the various acquisition concepts. Spherical tankage was employed where possible, for uniformity and simplicity in the screening.

The quantities of fluids to be transferred include the basic requirements from Table 4-1 plus that required for receiver childown. Final values used are presented in Table 4-2. In each case, storage and transfer of liquid was assumed, even though final storage in the receivers may be as a supercritical fluid or high pressure gas.

Childown fluid requirements were calculated assuming childown occurs while venting the receivers at $10.3 \times 10^4 \, N/m^2$ (15 psia). The quantity of fluid vented during childown was calculated using two different assumptions; (1) that vapor is vented at a saturation temperature corresponding to $10.3 \times 10^4 \, N/m^2$ (15 psia) and (2) that vapor is vented at a temperature corresponding to the tank wall as it chills. Vented quantities for the two conditions were obtained from the curves of Reference 4-1. In all cases, except for LHe, an average value between conditions 1 and 2 was used as an estimate of the actual fluid loss. In the case of LHe, venting of saturated vapor resulted in such high losses it was assumed that means would need to be provided to insure the most efficient vent condition (condition 2). This could possibly be accomplished by the addition of a heat exchanger to the receiver wall to allow maximum heat absorption of the vented vapor. In this and all subsequent calculations, integrated specific heats were used for the receiver hardware being chilled.

Descriptions, calculation methods, weights and performance data for each acquisition concept are presented in the following paragraphs.

Table 4-2. Fluid Usage for Screening Analysis

Receiver	Maximum Childown		Minimum Childown		Assumed Childown		Total Liquid Transferred	
	kg	lb	kg	lb	kg	lb	kg	lb
<u>Space Tug</u>								
Main Tank, LH_2	72.19	159	15.89	35	44.03	97	3506.24	7723
Main Tank, LO_2	74.91	165	54.93	121	64.92	143	19847.52	43717
APS, N_2H_4	-	-	-	-	-	-	152.54	336
Pressurant, LHe	125.30	276	4.09	9	4.09	9	8.17	18
<u>Shuttle Orbiter</u>								
OMS, N_2O_4	-	-	-	-	-	-	17633.36	38840
OMS, MMH	-	-	-	-	-	-	10646.3	23450
OMS, Press., LHe	1022.41	2252	33.14	73	33.14	73	88.99	196
EPS, ECLSS, LO_2	283.30	624	207.93	458	245.61	541	3022.20	6789
EPS, LH_2	190.68	420	41.77	92	116.22	256	450.37	992
<u>Multiple Receivers</u>								
SEPS, Hg	-	-	-	-	-	-	1498.2	3300
HEAOB, LHe	2494.28	5494	80.36	177	80.36	177	511.66	1127
SCS, N_2H_4	-	-	-	-	-	-	1506.10	3322

4.1.1 CAPILLARY ACQUISITION - Based on previous work (Ref. 4-2) the screen system illustrated in Figure 4-1 was chosen as being representative of capillary type acquisition. This system employs a double screen liner separated from the wall by low conductive supports. The screens are assumed to be kept wet at all times by wicking. The primary advantages of capillary systems over other systems are generally low weight, simplicity, fluid compatibility and life or reusability. Such devices have previously been applied to non-cryogenic propellant acquisition for low-g engine start. The basic problem is to prevent vapor from becoming trapped within the screens and causing breakdown of the capillary barrier such that a direct vapor path is formed between the tank ullage and the tank outlet. Such vapor formation or screen unwetting can result from liquid disturbances or, in the case of cryogenics, heat transfer to the collected liquid.

Even though mercury is a non-wetting fluid, screen systems can theoretically be designed for this application (Ref. 4-3). However, there are a number of disadvantages, and even though weight data for the mercury case were developed, the surface tension device is not a serious contender for this application.

Weight data generated for the Figure 4-1 acquisition system for each reciever are presented in Table 4-3. Screen, liquid residual, and tank weights were obtained respectively from Figures 5-35, 5-36 and A-1 of Reference 4-2. Non-cryogenic weights may be somewhat conservative since for these cases final design will not likely require a full double liner.

4.1.2 FLUID ROTATION - Based on data from Reference 4-2, the system presented in Figure 4-2 was taken to be the best representative of fluid rotation methods of liquid acquisition. This system employs a motor driven paddle to force the liquid to the tank outlet. Rotation of the entire Shuttle and receiver was not considered practical due to adverse dynamic effects and changing c.g. while transferring. Rotation of the tankage within the Shuttle is possible but was not considered desirable in comparison with fluid rotation due to the requirement for stationary to rotational connections.

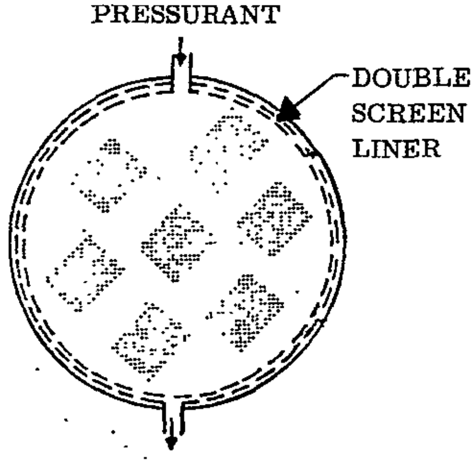


Figure 4-1. Typical Capillary Acquisition System

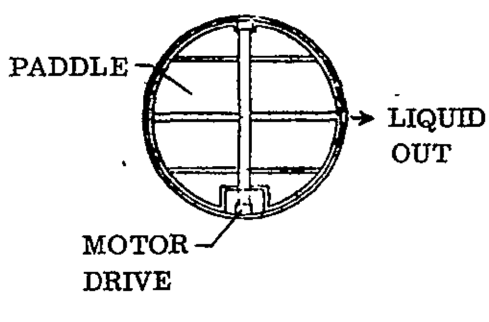


Figure 4-2. Fluid Rotation Using Paddle

Table 4-3. Capillary Acquisition Weights

Receiver	Supply Tank Volume		Tank Dia.		Tank Weight		Residual Weight		Screen Syst. Weight		Total Weight	
	m^3	ft^3	m	in	kg	lb	kg	lb	kg	lb	kg	lb
Space Tug												
Main Tank, LH_2	50.3	1775	4.6	180.3	395	870	36	80	186	410	617	1360
Main Tank, LO_2	17.7	623.7	3.2	127.2	145	320	244	538	93	205	482	1063
APS, N_2H_4	0.16	5.52	0.67	26.3	13	29	5	10	4	8	21	47
Pressurant, LHe	0.07	2.43	0.51	20.0	12	26	0.5	1	2	5	15	32
Shuttle Orbiter												
OMS, N_2O_4	12.3	433.8	2.9	112.7	109	240	229	505	73	160	411	905
OMS, MMH	12.4	437.5	2.9	113	109	240	138	305	73	160	320	705
OMS Press., LHe	0.75	26.4	1.12	44	19	41	3	7	2	5	24	53
EPS/ECLSS, LO_2	2.76	97.3	1.74	68.5	34	76	50	111	27	60	112	247
EPS, LH_2	6.48	228.9	2.3	91.1	64	140	6	14	48	105	117	259
Multiple Receivers												
SEPS, Hg	0.114	4.03	0.6	23.7	12	27	47	103	4	8	63	138
HEAOB, LHe	4.21	148.6	2.0	78.9	47	103	8	17	36	79	90	199
SCS, N_2H_4	1.53	54.0	1.43	56.3	25	56	28	61	18	40	71	157

The primary advantage of the paddle system is the positive positioning of the liquid. It is also convenient to incorporate low-g venting and mass gaging functions into this system. The primary disadvantages are its requirement for a motor drive and potentially higher residual weights than for a surface tension system.

To-date, significant quantitative analysis has not been performed on these systems. The fluid dynamics of the process are quite involved.

Some analysis was accomplished under Contract NAS8-26236 and is reported in Reference 4-2. Weight, power and fluid residuals were estimated for each of the present receivers using the methods presented in Reference 4-2.

The basic approach was to determine at what rate the fluid must be rotated to insure liquid at the wall and then to determine the power and hardware required to accelerate and to maintain the fluid at this rotation. For each case, required rotation rates were calculated for two different conditions; (1) that required to overcome surface tension forces and (2) that required to pump liquid away from the center shaft. The greater of the two rates was then used for design and calculation of power requirements. Equations used are presented below.

Rotation Required to Overcome Surface Tension

$$\omega = \sqrt{\frac{4\,\sigma\, g_c}{\rho_L\, \bar{R}_t^3} + \frac{5a}{R_t}} \qquad (4\text{-}1)$$

$\bar{R}_t$ used in the above equation is an average value; i.e., $\bar{R}_t = R_t/2$.

Rotation for Pumping Liquid From Center Shaft

$$\omega = \sqrt{a/R_l} \tag{4-2}$$

In the calculations made it was assumed that the radius of the center shaft is 0.017 times the tank diameter, D_t.

A maximum adverse acceleration, a, of 1×10^{-4} g's, was assumed.

In all cases, except for the small Space Tug APS N_2H_4 requirement, Equation 4-1 resulted in the maximum rotation rate.

From Reference 4-2, which assumes that paddle form drag provides the greatest resistance to paddle motion, the following equation was developed for the required power.

Motor input power

$$P, \text{ watts} = 1.91 \times 10^{-11} (\rho_L, \text{ lb/ft}^3) (\omega, \text{ rpm})^3 (D_t, \text{ in})^5 \tag{4-3}$$

$$\dot{P}, \text{ watts} = 3.7 \times 10^{-21} (\rho_L, \text{ kg/m}^3) (\omega, \text{ rad/sec})^3 (D_t, \text{ m})^5$$

This equation assumes a motor drive efficiency of 50 percent.

To allow for uncertainties in the analysis the actual rotation rate used in design and in calculation of power is increased by a factor of 3 over that determined from Equations 4-1 and 4-2. In addition, based on data from Reference 4-2, the required power for LH_2 and LHe cases is doubled to allow for start up.

Power supply weights were calculated, assuming use of fuel cells, with

$$\text{English Unit Weight} = 94 \text{ lb/kw} + 2.9 \text{ lb/kw - hr} \tag{4-4}$$

$$\text{S.I. Unit Weight} = 42.7 \text{ kg/kw} + 0.37 \text{ kg/MJ}$$

Liquid residuals were calculated per Reference 4-2 as the sum of that due to normal vapor pull-through and that which could be trapped between the paddle and the tank wall. Minimum clearance between paddle and wall is taken to be the same as for the surface tension screens.

Paddle rotation rates, power requirements and system weights are presented in Table 4-4. Paddle system (paddle plus motor) weights are taken from Figures 5-67 and 5-65 of Reference 4-2. Use of a DC motor with a 200:1 harmonic drive reducer

Table 4-4. Paddle Acquisition Weights

Receiver	Tank Dia		Rotation		Input Power	Tank Weight		Residual Weight		Paddle System Weight		Power Supply Weight		Total Weight	
	m	in	rad/sec	rpm	watts	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb
Space Tug															
Main Tank, LH_2	4.6	180.3	0.34	3.2	1044	395	870	66.7	147	115	254	59.0	130	636.0	1401
Main Tank, LO_2	3.2	127.2	0.40	3.8	2559	145	320	349	768	95.3	210	145	320	735	1618
APS, N_2H_4	0.67	26.3	0.98	9.4	12.7	13	29	6.4	14	4.5	10	0.9	2	25.0	55
Pressurant, LHe	0.51	20.0	1.02	9.7	0.85	12	26	0.9	2	3.2	7	0.5	1	16.3	36
Shuttle Orbiter															
OMS, N_2O_4	2.9	112.7	0.43	4.1	2141	109	240	276	608	80.8	178	123	270	588	1296
OMS, MMH	2.9	113	0.43	4.1	1268	109	240	166	366	66.3	146	72.6	160	414	912
OMS, Press., LHe	1.12	44	1.01	9.6	0.97	19	41	4.1	9	3.6	8	0.5	1	26.8	59
EPS/ECLSS, LO_2	1.74	68.5	0.54	5.2	291	34	76	58.6	129	27.7	61	16.3	36	137	302
EPS, LH_2	2.3	91.1	0.47	4.5	97	64	140	9.5	21	32.2	71	5.4	12	111	244
Multiple Receivers															
SEPS, Hg	0.6	23.7	0.93	8.9	84	12	27	63.6	140	9.1	20	4.5	10	89.4	197
HEAOB, LHe	2.0	78.9	0.51	4.9	104	47	103	23.6	52	26.8	59	5.9	13	103	227
SCS, N_2H_4	1.43	56.3	0.61	5.8	130	25	56	49.9	110	19.1	42	7.3	16	102	224

was assumed. Power system weights are based on a 36,000 sec (10 hr) transfer.

4.1.3 POSITIVE EXPULSION - Positive expulsion systems are designed to provide a positive barrier between the pressurant and liquid to be transferred. The bladder, bellows and diaphragm systems illustrated in Figure 4-3 were considered. Pistons were not included due to their combination of high weight and moving seal problems, especially with cryogenics. Details of the individual systems are presented below.

4.1.3.1 Bladders - A significant amount of development work has been accomplished on bladder systems. Data from Reference 4-2 indicates the collapsing type bladder, as shown in Figure 4-3, is preferred over the expanding type. In the Figure 4-3 system, pressurant is applied external to the bladder and liquid is expelled internally as the bladder collapses around a perforated outflow standpipe. Such systems have been satisfactorily demonstrated for non-cryogenic fluids, having been used in operational systems. Completely satisfactory systems have not yet been demonstrated for use with LO_2 and LH_2. Materials which are flexible at LO_2 temperatures are not 100% compatible from a safety standpoint. The main problem with LH_2 is inter-ply inflation from gas being trapped at cryogenic temperatures and causing separation of the plies on warm-up.

Bladder system weights are presented in Table 4-5. Based on existing systems the percentage of liquid residuals was assumed to be 2%. The weight of the bladder assembly consists of the weight of the bladder itself plus the standpipe and associated flanges and fittings. A detailed examination of the weight elements of the Centaur H_2O_2 bladder system resulted in the development of the following parametric weight equation for the standpipe, flanges and fittings.

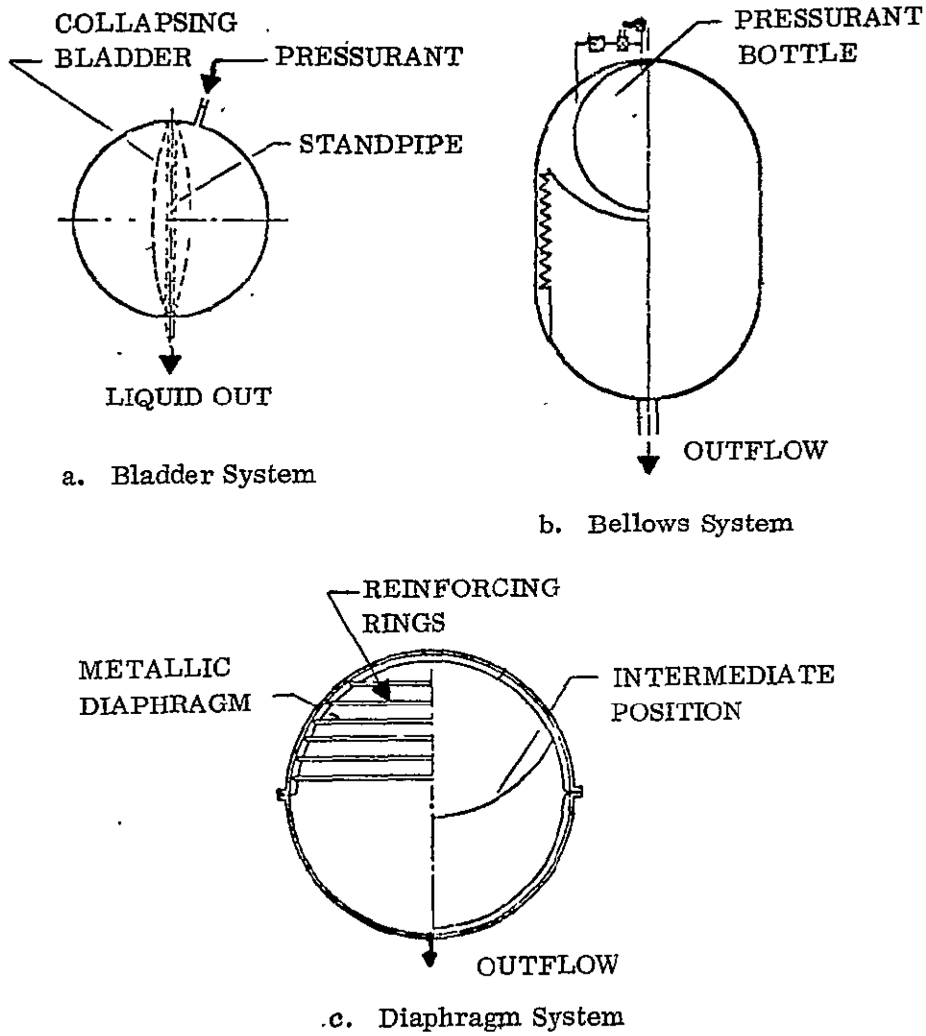


Figure 4-3. Positive Expulsion Systems

$$\text{Bladder System Weight, lb} = \frac{1.55\times 10^{-5}\,(D_t,\text{in.})^3 + 0.0021\,(D_t,\text{ in.})^2 + 0.25\,(D_t,\text{in.}) + 0.25}{0.95}$$

$$\text{Bladder System Weight, kg} = \frac{2.5\times 10^{-4}\,(D_t,\text{cm})^3 + 0.014\,(D_t,\text{cm})^2 + 0.64\,(D_t,\text{cm}) + 0.25}{2.09} \qquad (4\text{-}5)$$

For weight estimating, the bladder itself was assumed to consist of two plies of Teflon with a total thickness of 0.036 cm (0.014 in) or 0.018 cm (0.007 in) for each ply. The bladder weight is then taken to be 0.826 kg/m^2 (0.169 lb/ft^2).

Table 4-5. Bladder System Weights

Receiver	Tank Dia.		Tank Weight		Residual Weight		Bladder Assembly		Total Weight	
	m	in	kg	lb	kg	lb	kg	lb	kg	lb
Space Tug										
Main Tank, LH_2	4.6	180.3	395	870	69.9	154	152	335	617	1359
Main Tank, LO_2	3.2	127.2	145	320	397	874	73.5	162	616	1356
APS, N_2H_4	0.67	26.3	13	29	3.2	7	5.4	12	21.8	48
Pressurant, LHe	0.51	20.0	12	26	0.2	0.4	3.6	8	15.4	34
Shuttle Orbiter										
OMS, N_2O_4	2.9	113	109	240	353	777	58.6	129	520	1146
OMS, MMH	2.9	113	109	240	213	469	58.6	129	380	838
OMS Press., LHe	1.12	44	19	41	1.8	4	11.4	25	31.8	70
EPS/ECLSS, LO_2	1.74	68.5	34	76	61.7	136	23.2	51	119	263
EPS, LH_2	2.3	91.1	64	140	9.1	20	38.6	85	111	245
Multiple Receivers										
SEPS, Hg	0.6	23.7	12	27	30.0	66	4.5	10	46.8	103
HEAOB, LHe	2.0	78.9	47	103	10.4	23	30.0	66	87.2	192
SCS, N_2H_4	1.43	56.3	25	56	30.0	66	16.8	37	72.2	159

4.1.3.2 Bellows - Although bellows systems are in general heavier than other systems they have the advantage of reusable operation over a large number of cycles. Also, low-g mass gaging can be accomplished by measuring the stroke of the bellows. Based on data from Reference 4-2, a nested-hydroformed type of CRES bellows with L/D of 1.5 was selected for the current application. Parametric weight data for such a system are presented in Reference 4-2. This data was applied to the current supply requirements and the weights presented in Table 4-6 generated. Liquid residuals were taken to be 1% of total liquid volume.

4.1.3.3 Diaphragm - The diaphragm employed here consists of a hemisphere which reverses itself from one end of the tank to the other during expulsion. For weight comparisons a metallic diaphragm with reinforcing rings was used. Weight data were obtained from Figure 5-60 of Reference 4-2 and are presented in Table 4-7. The main disadvantage to this system is the limited number of reusable cycles (5 to 10). The primary advantages are compatibility with all fluids and low residuals, taken as 0.5% for the present case. CRES tankage is used here, instead of Al Aly, to allow welding of the diaphragm into the tank.

Table 4-6. Bellows System Weights

Receiver	Supply Tank Volume		Cylindrical Tank Dia		Bellows and Tank Weight		Residual Weight		Total Weight	
	m^3	ft^3	m	in	kg	lb	kg	lb	kg	lb
Space Tug										
Main Tank, LH_2	50.2	1775	3.5	138	1679.8	3700	34.96	77	1714.76	3777
Main Tank, LO_2	17.6	622	2.5	97	544.8	1200	198.40	437	743.20	1637
APS, N_2H_4	0.15	5.4	0.51	20	19.52	43	1.36	3	20.88	46
Pressurant, LHe	0.07	2.4	0.38	15	15.44	34	0.09	0.2	15.44	34
Shuttle Orbiter										
OMS, N_2O_4	12.3	433	2.2	86	381.36	840	176.15	388	557.51	1228
OMS, MMH	12.3	436	2.2	86	385.9	850	106.69	235	492.59	1085
OMS Press., LHe	0.74	26	0.86	34	52.21	115	0.91	2	53.12	117
EPS/ECLSS, LO_2	2.7	97	1.3	52	122.58	270	30.87	68	153.45	338
EPS, LH_2	6.5	228	1.8	69	227	500	4.54	10	231.54	510
Multiple Receivers										
SEPS, Hg	0.11	3.9	0.46	18	15.44	34	14.98	33	30.41	67
HEAOB, LHe	4.2	148	1.5	60	163.44	360	4.99	11	168.43	371
SCS, N_2H_4	1.5	54	1.1	43	83.08	183	14.98	33	98.06	216

Table 4-7. Diaphragm System Weights

Receiver	Tank Dia		CRES Tank Weight		Residual Weight		Diaphragm Weight (1)		Total Weight	
	m	in.	kg	lb	kg	lb	kg	lb	kg	lb
Space Tug										
Main Tank, LH_2	4.6	180	432.20	952	17.71	39	543.89	1198	993.81	2189
Main Tank, LO_2	3.2	127	158.9	350	99.43	219	227	500	485.33	1069
APS, N_2H_4	0.7	26.2	24.07	31	0.68	1.5	5.90	13	20.88	46
Pressurant, LHe	0.5	18.0	12.71	28	0.05	0.1	1.82	4	14.53	32
Shuttle Orbiter										
OMS, N_2O_4	2.9	113	114.86	253	88.08	194	167.98	370	370.92	817
OMS, MMH	2.9	113	114.86	253	53.12	117	167.98	370	335.96	740
OMS Press., LHe	1.1	44	19.52	43	0.45	1	13.17	29	33.14	73
EPS/ECLSS, LO_2	1.7	68	38.14	84	15.44	34	41.77	92	95.34	210
EPS, LH_2	2.3	91	67.64	149	2.27	5	93.52	206	163.44	360
Multiple Receivers										
SEPS, Hg	0.6	23.5	12.71	28	7.72	17	4.09	9	24.51	54
HEAOB, LHe	2.0	79	50.85	112	2.72	6	63.56	140	117.13	258
SCS, N_2H_4	1.4	56.2	26.33	58	7.72	17	25.42	56	59.47	131

(1) Hollow wires are used for all tank diameters over 30 inches.

4.1.4 LINEAR ACCELERATION - The linear acceleration concept utilizes external forces to maintain liquid at one end of a tank for transfer (Figure 4-4). Therefore, other than that orientation forces are significantly lower than at 1-g, transfer proceeds in the same manner as it would on earth. Two different methods of providing orientation forces were investigated; (1) thrusting with an auxiliary propulsion system, and (2) utilizing drag forces on the Shuttle Orbiter.

Settling acceleration produced by drag alone was determined (Appendix B) to be a maximum of 1.2×10^{-6} g's at the 296 km (160 n.mi.) transfer orbit. When utilizing auxiliary propulsion it was assumed, except for Shuttle to Shuttle transfer, that the supply module and receiver are separated from the Shuttle and accelerated in a cross-plane direction. Separation from the Shuttle minimizes auxiliary propellant requirements and was determined to be better than acceleration of the entire Shuttle (Reference 4-4). Also, from Reference 4-4, cross-plane thrusting was determined to be desirable since the supply-module-receiver will return to the vicinity of the Shuttle once every orbital cycle. This minimizes rendezvous requirements for re-acquiring the supply module.

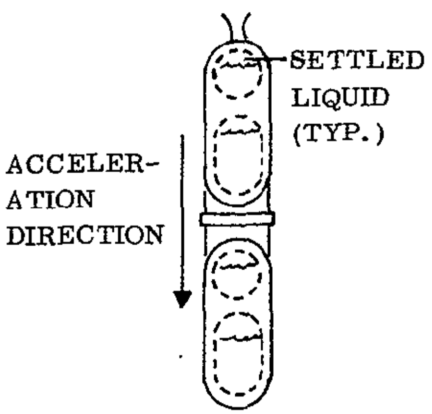


Figure 4-4. Linear Acceleration System

In both drag and thrusting cases, liquid residuals were calculated using the basic methods described in Reference 4-4. Tank outlets were taken to be hemispherical employing baffles to minimize residuals. The procedure was to calculate residuals at vapor pull-through assuming a flat liquid interface and then adding those due to low-g interface curvature (Figure 4-5). Froude number correlations from Figure 6.3.2-1 of Reference 4-4 were used to determine the flat interface residuals and the liquid height at the center of the tank when pull-through occurs. Equations from Reference 4-5 used to calculate curved interface residuals are presented below.

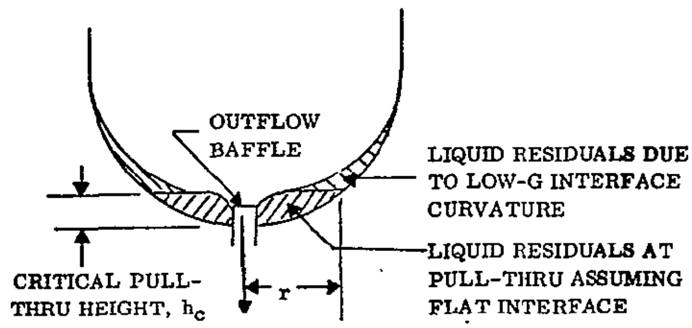


Figure 4-5. Low-g Liquid Residuals at Vapor Pull-Through

$$\text{Curved Interface Residual Volume} = \frac{\pi}{3}\left(\frac{1}{\xi}\right) \cdot r^3$$

where

$$\xi^3 + \xi = 2 + Bo_r$$

and r refers to the local tank radius where pull-through occurs (Figure 4-5). A further discussion of this method of calculating residuals is presented in Appendix C.

A step throttling down of the outflow by a ratio of 10:1 was accomplished near the end of transfer to minimize residuals.

A hydrazine propulsion system was used to provide acceleration for the non-drag case. Propulsion system weight data were taken from Reference 4-6 and are presented in Figure 4-6 as a function of the total impulse required. To determine minimum overall

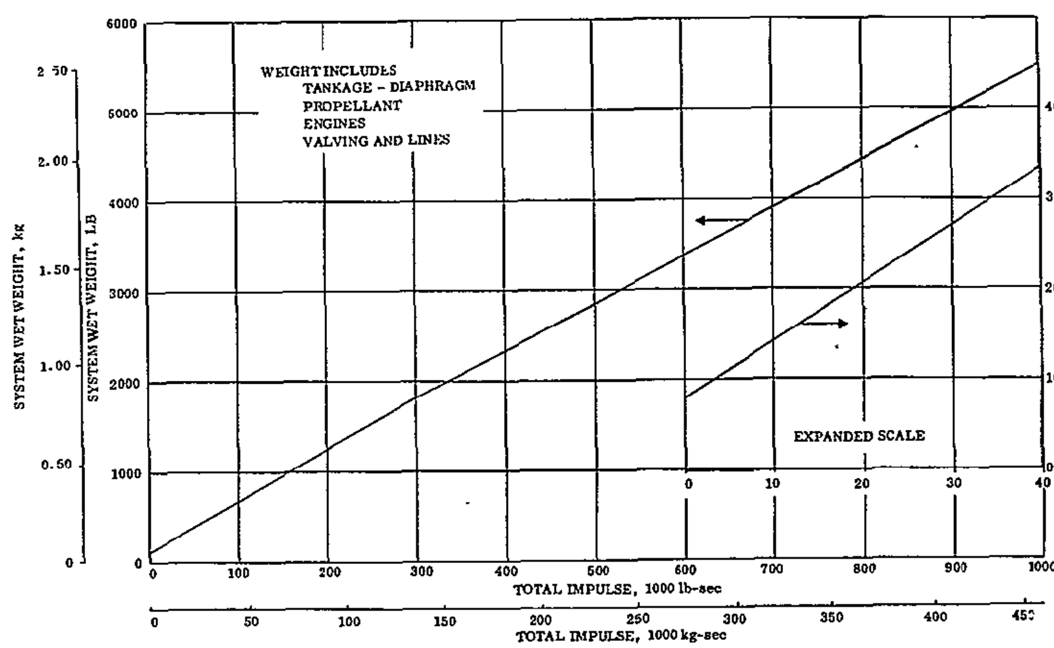


Figure 4-6. Auxiliary Propulsion System Weights (N_2H_4 Monopropellant), Ref. 4-6

system weights, trade-offs were made between residual and propulsion system weights as a function of transfer time and acceleration level. Residuals decrease with increases in transfer time (reduced flow rate) and acceleration level. Propulsion system weights increase with increased transfer time and acceleration level.

Typical weight versus transfer time curves are presented in Figure 4-7 for different accelerations for the Space Tug supply case. Weights are the sum of propulsion system (where used) weight and propellant plus the LO_2 and LH_2 residuals. The LO_2 and LH_2 are assumed to be transferred simultaneously. Hydrazine and helium weights were not significant with respect to their influence on optimizing transfer time. From Figure 4-7 a reasonable compromise between transfer time and weight for the auxiliary propulsion or separated Tug case was taken to correspond to an acceleration of 1×10^{-4} g's. Weights generated in this manner are presented for the various cases in Tables 4-8 through 4-10.

Optimum transfer times for the drag cases were very long (Ref. Figure 4-7). The transfer times presented in Tables 4-8 and 4-9 are not necessarily optimum. They were chosen as reasonable maximums without having to consider mission limitations and the effects of external heating.

Drag-only was not considered for the relatively small multiple receivers case.

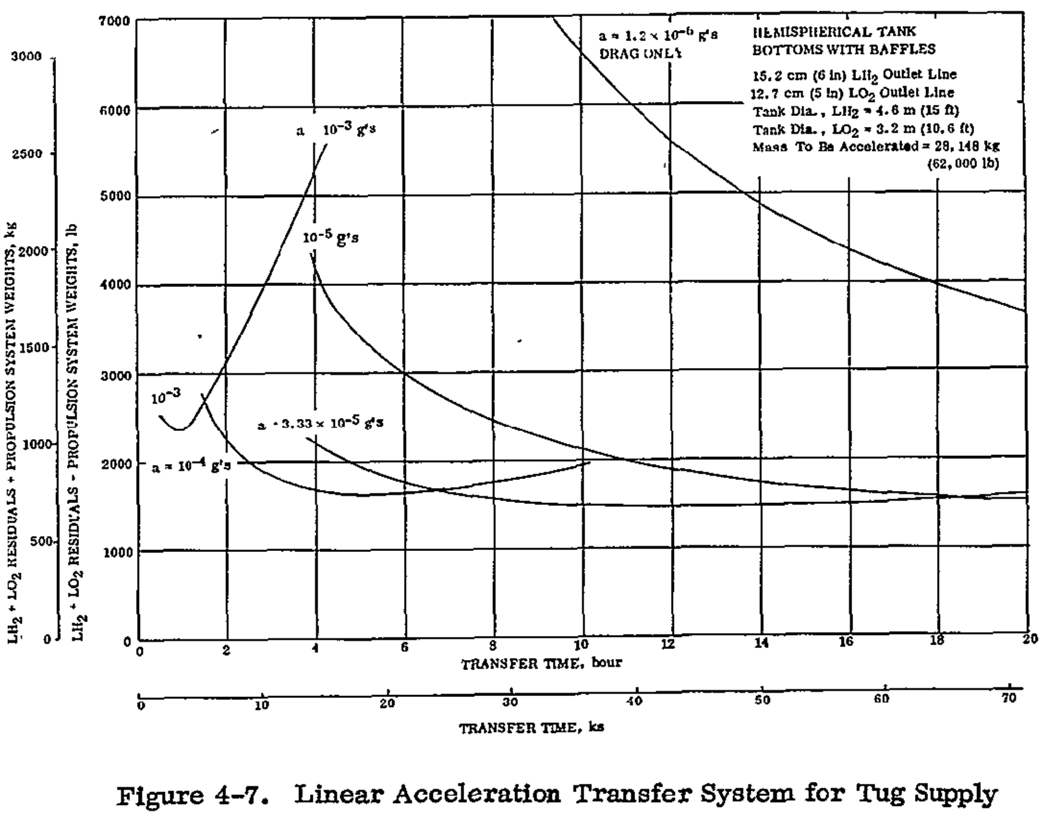


Figure 4-7. Linear Acceleration Transfer System for Tug Supply

Table 4-8. Tug Supply Linear Acceleration System Weights

	Transfer Time (θ_{TR}) = 5 hrs a = 10^{-4} g's		Drag-Only Transfer Time (θ_{TR}) = 20 hrs a = 1.2×10^{-6} g's	
	kg	lb	kg	lb
LH_2 Residuals	86.26	190	376.82	830
LH_2 Tankage	394.98	870	394.98	870
LO_2 Residuals	317.80	700	1271.20	2800
LO_2 Tankage	145.28	320	145.28	320
N_2H_4 Residuals	4.54	10	4.54	10
N_2H_4 Tankage	13.17	29	13.17	29
LHe Residuals	.45	1	.45	1
LHe Tankage	11.80	26	11.80	26
Propulsion System	340.50	750	-	-
Total	1314.78	2896	2218.24	4886
	D_ℓ, LO_2 = 12.7 cm (5 in) D_ℓ, LH_2 = 15.2 cm (6 in)		D_t, LO_2 = 3.2 m (10.6 ft) D_t, LH_2 = 4.6 m (15 ft)	

Table 4-9. Shuttle Supply Linear Acceleration System Weights

	a = 10^{-5} g's θ_{TR} = 20 hrs		a = 1.2×10^{-6} g's θ_{TR} = 40 hrs	
	kg	lb	kg	lb
N_2O_4 Residuals	472.16	1040	835.36	1840
N_2O_4 Tankage	108.96	240	108.95	240
MMH Residuals	413.14	910	526.64	1160
MMH Tankage	108.96	240	108.96	240
LHe Residuals	3.18	7	3.18	7
LHe Tankage	18.61	41	18.61	41
LO_2 Residuals	72.64	160	118.95	262
LO_2 Tankage	34.50	76	34.50	76
LH_2 Residuals	14.98	33	24.06	53
LH_2 Tankage	63.56	140	63.56	140
Propulsion System	844.44	1860	-	-
Total	2155.14	4747	1842.79	4059

Accelerated Mass: 198,398 kg (437,000 lb)
N_2O_4: D_t = 2.87 m (9.42 ft), D_ℓ = 10.2 cm (4 in)
MMH: D_t = 2.87 m (9.42 ft), D_ℓ = 10.2 cm (4 in)
LO_2: D_t = 1.7 m (5.7 ft), D_ℓ = 6.4 cm (2.5 in)
LH_2: D_t = 2.3 m (7.6 ft), D_ℓ = 7.6 cm (3 in)

Table 4-10. Multiple Receivers Linear Acceleration System Weights

	SEPS D_t = 0.6 m (2 ft) D_ℓ = 2.5 cm (1 in) a = 10^{-4} g's θ_{TR} = 4 hrs Mass = 2229 kg (4910 lb)		HEAO B D_t = 2 m (6.6 ft) D_ℓ = 7 cm (2.75 in) a = 10^{-4} g's θ_{TR} = 2 hrs Mass = 3405 kg (7500 lb)		SCS D_t = 1.4 m (4.7 ft) D_ℓ = 5.1 cm (2 in) a = 10^{-4} g's θ_{TR} = 4.5 hrs Mass = 3201 kg (7050 lb)	
	kg	lb	kg	lb	kg	lb
Residuals	32.69	72	14.53	32	46.31	102
Tankage	12.26	27	46.76	103	25.42	56
Propulsion	55.84	123	51.30	113	68.10	150
Total	100.79	222	112.60	248	139.83	308

4.1.5 THERMODYNAMIC - The thermodynamic concept is illustrated in Figure 4-8, along with T-S diagrams describing fluid conditions in various parts of the system for both vapor and liquid inlets. In this system the fluid is compressed to a higher pressure and temperature than the tank fluid and then allowed to exchange heat with the bulk fluid to condense any vapor which may be present. The liquid at state 3 (Figure 4-8) is then supplied to the receiver tank. This system was only considered where cryogenics were involved.

A similar system was analyzed in Reference 4-7, where it was assumed that warm vapor generated in the receiver was used to drive the compressor. However, even though for some conditions there is sufficient total receiver energy available, the available rate does not correspond with the demand rate. This means that the produced energy must be controlled or stored, which would be extremely difficult.

In the present case, an independent power source was used to drive the compressor. To determine system feasibility, calculations were made for Tug LH_2 and LO_2 and HEAO-B LHe supply. Resulting weights and operating conditions are presented in Table 4-11. In each case, transfer time was 24 hours and it was assumed that a vapor

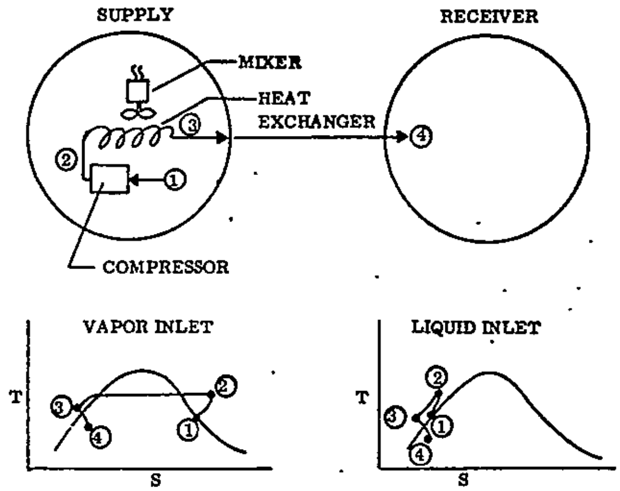


Figure 4-8. Thermodynamic Transfer Concept

inlet existed throughout the entire transfer.

Due to its uniquely low heat of condensation and high vapor specific heat, helium appears to be the only fluid requiring reasonable power.

In an actual case a major problem is that the inlet to the compressor may be either gas or liquid or a mixture of both. It is not reasonable at present to design a compressor to provide the needed head rise where such uncertainty exists. One way to overcome this problem would be to locate a heat exchanger upstream of the compressor and to throttle the inlet and exchange heat with warm fluid to vaporize any liquid which may be present. Analysis was performed for the HEAO-B case to determine the effect on system weight of adding such an inlet heat exchanger. In this case the inlet fluid is throttled from 103.4 kN/m^2 (15 psia) to 69 kN/m^2 (10 psia) and

Table 4-11. Estimated Thermodynamic Transfer System Weights and Operating Conditions (24 Hour Transfer)

	Tug				HEAO-B	
	LH_2		LO_2		LHe	
Transfer Time, sec (hr)	(86.4×10^3	(24)	86.4×10^3	(24)	86.4×10^3	(24)
ΔT_{1-2}, °K (°R)	5.6	(10)	5.6	(10)	0.56	(1)
ΔP_{1-2}, kN/m^2 (psid)	75.8	(11)	82.7	(12)	69.0	(10)
Circulating Pump Power, kw-hr	1.2		377		0.12	
Compressor Power, kw-hr	322		174		4	
Exchanger Weight, kg (lb)	65.8	(145)	183	(404)	4.1	(9)
Circulation Pump Weight, kg (lb)	2.3	(5)	22.7	(50)	0.5	(1)
Compressor Weight, kg (lb)	20.9	(46)	15.0	(33)	2.7	(6)
Power Supply Weight, kg (lb)	174	(384)*	318	(700)*	10.4	(23)**
Tankage Weight, kg (lb)	395	(870)	145	(320)	46.8	(103)
Total Weight, kg (lb)	658	(1450)	684	(1507)	64.5	(142)

* Assumes existing Shuttle power supply used with weight penalty only for usage in excess of 50 kw-hr, and then only for additional reactants.

** Provides for own power supply.

then compressed from 69 kN/m^2 (10 psia) to 138 kN/m^2 (20 psia). Increases in weights and powers for this case were considerable. The total weight was found to be approximately 136 kg (300 lb) and the power was approximately doubled.

An overall summary of the screening data and concept selections for further work are presented in the following paragraph.

4.1.6 SCREENING SUMMARY AND CONCEPT SELECTIONS - Comparative weight data generated for each acquisition concept, as described in the preceding paragraphs, are presented in Table 4-12. The system weights include supply tank residuals, supply tankage, and acquisition system. Residuals, as percentage of total mass transferred are either presented in the headings of each system or in parenthesis beside the total weights. In some cases, for comparison, total weight data are presented for more than one percentage of residuals. Also, there is some uncertainty as to whether or not the existing Shuttle power supply can be used without penalty. Thus, for the fluid rotation concept, weights are shown with and without a penalty for the power supply.

Overall qualitative comparisons of the various acquisition concepts are presented in Tables 4-13 through 4-17. The various comparison criteria are discussed below.

a. Weight - this relative rating is based on the data from Table 4-12.

b. Reliability - is the measure of the probability of the system to operate as required in an operational situation, once it has been developed.

c. Reusability - measures the useful life in terms of the number of transfer missions which can be accomplished with the same hardware.

d. Safety - measures the potential for hazardous failure with respect to manned operation of an operational system.

e. Development Risk - is the risk that a satisfactory system cannot be developed with a reasonable amount of resources.

f. Receiver Impact - pertains to the extent of modifications and/or technology advancement required, with respect to the receiver(s), to allow transfer.

g. Shuttle Impact - pertains to the modifications or operations, such as transfer module deployment and/or receiver docking, which must be accomplished with respect to the Shuttle Orbiter.

h. Development Cost - is the cost to develop an operational system.

i. Unit Cost - is the cost of a transfer system once it is developed.

Table 4-12A. Overall Transfer Systems Weight Summary (kg)

	Individual System Weights[a], kg											
			Fluid Rotation (% Residuals)		Bladder		Bellows	Metallic Diaphragm		Linear Acceleration		
	Mass Transfer	Surface Tension (% Residuals)	With Power Supply	W/O Power Supply	2% Residuals	Same Resid. as S. T.	1% Residuals	1% Residuals	0.5% Residuals	10^{-4} g's (Time, hrs)	Shuttle Drag (Time, hrs)	Thermodynamic
<u>Tug Supply</u>												
LH_2	3506	617 (1.0)	636 (1.9)	577	617	583	1715	1011	994			658 +
LO_2	19848	483 (1.2)	735 (1.8)	589	616	463	743	584	485			684 +
N_2H_4	153	21 (3.0)	25 (4.2)	24	22	23	21	21	21			-
LHe	8	15 (3.3)	16 (8.8)	16	15	15	15	15	15			-
Total		1136	1412	1206	1270	1085	2494	1631	1515	[c] 1315 (5)	[d] 2218 (20)	1342 +
Shuttle <u>Supply</u>												
N_2O_4	17633	411 (1.3)	588 (1.6)	466	520	397	5576	459	371			-
MMH	10646	320 (1.3)	414 (1.6)	341	380	306	493	389	336			-
LHe	89	24 (3.6)	27 (4.6)	26	32	33	53	34	33			-
LO_2	3082	112 (1.6)	137 (1.9)	121	119	108	134	111	95			
LH_2	450	118 (1.4)	111 (2.1)	105	111	109	232	166	163			-
Total		984	1277	1060	1163	952	1488	1158	999	[b] 2155 (20)[e]	1843 (40)[f]	
Multiple <u>Supply</u>												
SEPS, Hg	1498	63 (3.1)	89 (4.2)	85	47	64	30	32	25	[g] 101 (4)		-
HEAOB, LHe	512	90 (1.5)	103 (4.6)	97	87	84	168	120	117	[g] 113 (2)		136
SCS, N_2H_4	1509	71 (1.8)	102 (3.3)	94	72	70	98	67	59	[g] 140 (4.5)		-

a. Weights include residuals, bare tankage and propulsion system, where applicable.
b. In this case the optimum acceleration with thrust was 10^{-5} g's.
c. Residuals: LO_2 (1.6%), LH_2 (2.5%), N_2H_4 (3.0%), LHe (3.3%).
d. Residuals: LO_2 (6.4%), LH_2 (10.7%), N_2H_4 (6.4%), LHe (11.0%).
e. Residuals: N_2O_4 (2.65%), MMH (3.9%), LHe (3.3%), LO_2 (2.4%), LH_2 (3.3%).
f. Residuals: N_2O_4 (4.7%), MMH (4.9%), LHe (5.1%), LO_2 (3.9%), LH_2 (5.2%).
g. Residuals: Hg (2.2%), LHe (2.9%), N_2H_4 (3.1%).

Table 4-12B. Overall Transfer Systems Weights Summary (lbs)

	Individual System Weights[a], lb											
	Mass Trans-fer	Surface Tension (% Residuals)	Fluid Rotation (% Residuals): With Power Supply	Fluid Rotation (% Residuals): W/O Power Supply	Bladder: 2% Resid-uals	Bladder: Same Resid. as S. T.	Bel-lows: 1% Resid-uals	Metallic Diaphragm: 1% Resid-uals	Metallic Diaphragm: 0.5% Resid-uals	Linear Acceleration: 10^{-4} g's (Time, hrs)	Linear Acceleration: Shuttle Drag (Time, hrs)	Thermo-dynamic
Tug Supply												
LH_2	7,723	1,360 (1.0)	1,401 (1.9)	1,271	1,359	1,285	3,777	2,227	2,189			1,450+
LC_2	43,717	1,063 (1.2)	1,618 (1.8)	1,298	1,356	1,020	1,637	1,287	1,069			1,507+
N_2H_4	336	47 (3.0	55 (4.2)	53	48	51	46	47	46			-
LHe	18.1	32 (3.3)	36 (8.8)	35	34	34	34	32	32			-
Total		2,502	3,110	2,657	2,797	2,390	5,494	3,593	3,336	[c] 2,896 (5)	[d] 4,886 (20)	2,957+
Shuttle Supply												
N_2O_4	38,840	905 (1.3)	1,296 (1.6)	1,026	1,146	874	1,228	1,011	817			-
MMH	23,450	705 (1.3)	912 (1.6)	752	838	674	1,085	857	740			-
LHe	196	53 (3.6)	59 (4.6)	58	70	73	117	74	73			-
LO_2	6,789	247 (1.6)	302 (1.9)	266	263	238	338	244	210			-
LH_2	992	259 (1.4)	244 (2.1)	232	245	239	510	365	360			-
Total		2,169	2,813	2,334	2,562	2.098	3,278	2,551	2,200	[b] 4,747 (20)[e]	4,059 (40)[f]	
Multiple Supply												
SEPS, Hg	3,300	138 (3.1)	197 (4.2)	187	103	140	67	71	54	[g] 222 (4)		-
HEAOB, LHe	1,127	199 (1.5)	227 (4.6)	214	192	186	371	264	258	[g] 248 (2)		300
SCS, N_2H_4	3,322	157 (1.8)	224 (3.3)	208	159	154	216	148	131	[g] 308 ($4\frac{1}{2}$)		-

a. Weights include residuals, bare tankage and propulsion system, where applicable.
b. In this case the optimum acceleration with thrust was 10^{-5} g's.
c. Residuals: LO_2 (1.6%), LH_2 (2.5%), N_2H_4 (3.0%), LHe (3.3%).
d. Residuals: LO_2 (6.4%), LH_2 (10.7%), N_2H_4 (6.4%), LHe (11.0%).
e. Residuals: N_2O_4 (2.65%), MMH (3.9%), LHe (3.3%), LO_2 (2.4%), LH_2 (3.3%).
f. Residuals: N_2O_4 (4.7%), MMH (4.9%), LHe (5.1%), LO_2 (3.9%), LH_2 (5.2%).
g. Residuals: Hg (2.2%), LHe (2.9%), N_2H_4 (3.1%).

Table 4-13. Qualitative Comparisons of Acquisition Concepts for Tug Supply (Same Concept for all Fluids)

Comparisons Criteria**	Surface Tension	Fluid Rotation	Bladder	Bellows	Diaphragm	Linear Acceleration*	Thermodyn.
Weight	2	3	2	5	4	3	4
Reliability	2	2	3	2	2	2	2
Reusability	1	1	3	2	4	1	1
Safety	1	2	3	2	1	2	2
Development Risk	3	2	4	5	4	1*	4
Receiver Impact	4	4	4	4	4	2*	4
Shuttle Impact	2	2	2	2	2	3	2
Development Cost	3	3	4	5	4	2	4
Unit Cost	2	2	2	5	4	2	2
Operational Complexity	2	2	2	2	2	4	2
	22	23	29	34	31	23	28

* Concept selected for conceptual design and most important criteria used in selection.

** Ratings are relative on a scale of 1 to 5 where 1 represents the best.

Table 4-14. Qualitative Comparisons of Acquisition Concepts for Shuttle Supply (Same Concept for all Fluids)

Comparisons Criteria **	Surface Tension*	Fluid Rotation	Bladder	Bellows	Diaphragm	Linear Acceleration
Weight	1*	2	1	3	2	5
Reliability	2	2	3	2	2	2
Reusability	1*	1	3	2	4	1
Safety	1*	2	3	2	1	2
Development Risk	3	2	4	5	4	2
Receiver Impact	4	4	4	4	4	3
Shuttle Impact	2	2	2	2	2	4
Development Cost	3	3	4	5	4	3
Unit Cost	2	2	2	4	3	2
Operational Complexity	2	2	2	2	2	4
	21	22	28	31	28	28

* Concept selected for conceptual design and most important criteria used in selection.

** Ratings are relative on a scale of 1 to 5 where 1 represents the best.

Table 4-15. Qualitative Comparisons of Acquisition Concepts for Solar Electric Propulsion, Hg

Comparisons Criteria**	Surface Tension	Fluid Rotation	Bladder	Bellows	Diaphragm*	Linear Acceleration
Weight	3	4	3	2	1*	5
Reliability	3	2	2	2	1*	2
Reusability	1	1	3	2	4	1
Safety	2	2	2	2	2	2
Development Risk	4	2	2	2	2	2
Receiver Impact	3	3	3	3	3	3
Shuttle Impact	2	2	2	2	2	3
Development Cost	3	3	3	3	3	3
Unit Cost	2	2	2	3	2	4
Operational Complexity	2	2	2	2	2	4
	25	23	24	23	22	29

* Concept selected for conceptual design and most important criteria used in selection.

** Ratings are relative on a scale of 1 to 5 where 1 represents the best.

Table 4-16. Qualitative Comparisons of Acquisition Concepts for Large High Energy Observatory B, LHe

Comparisons Criteria**	Surface Tension	Fluid Rotation*	Bladder	Bellows	Diaphragm	Linear Acceleration	Thermodynamic
Weight	2	3	2	5	4	4	3
Reliability	3	2	3	3	2	2	2
Reusability	1	1	4	2	3	1	1
Safety	1	1	1	1	1	2	1
Development Risk	3	2*	4	4	3	1	4
Receiver Impact	4	4	4	4	4	2	4
Shuttle Impact	2	2	2	2	2	3	2
Development Cost	3	3	5	4	4	3	3
Unit Cost	2	2	2	4	3	3	3
Operational Complexity	2	2	2	2	2	4	2
	23	22	29	31	28	25	25

* Concept selected for conceptual design and most important criteria used in selection

** Ratings are relative on a scale of 1 to 5 where 1 represents the best.

Table 4-17. Qualitative Comparisons of Acquisition Concepts for Satellite Control Section, N_2H_4

Comparisons Criteria**	Surface Tension*	Fluid Rotation	Bladder	Bellows	Diaphragm	Linear Acceleration
Weight	1	2	1	2	1	3
Reliability	2	2	2	2	2	3
Reusability	1*	1	2	2	4	1
Safety	2	2	2	2	2	2
Development Risk	2	2	2	3	3	2
Receiver Impact	4	4	4	4	4	2
Shuttle Impact	2	2	2	2	2	4
Development Cost	2	2	2	3	2	2
Unit Cost	2	2	2	3	2	3
Operational Complexity	2	2	2	2	2	4
	20	21	21	25	24	26

* Concept selected for conceptual design and most important criteria used in selection.
** Ratings are relative on a scale of 1 to 5 where 1 represents the best.

j. Operational Complexity - measures the manned and/or remote control tasks involved in accomplishing the overall transfer.

All ratings are only relative and are on a potential scale of 1 to 5, where 1 represents the best.

Systems chosen for detailed conceptual design, along with reason(s) for their choice, are also indicated in the tables.

Data for Tug supply is presented in Table 4-13. It is noted that the data here assumes that the same acquisition concept is used for all fluids. This is also the case for Shuttle supply (Table 4-14). Each fluid was initially considered separately, however, for simplicity it would be advantageous to use a single concept throughout a single transfer system, and for these systems no strong reasons were found for not doing this. The multiple receivers are significantly different, and data are presented separately in Tables 4-15 through 4-17.

Referring to Table 4-13 it is seen that on the whole the surface tension, fluid rotation and linear acceleration systems are the most promising for Tug supply. Giving a fairly high importance to low development risk and receiver impact resulted in the choice of the linear acceleration system for conceptual design. The choice between the drag and auxiliary propulsion versions is sensitive to the actual quantities of liquid residuals to be expected. Such residuals are highly dependent on tank diameter and shape, which are functions of detailed packaging into the Shuttle. Other factors affecting a choice between drag and auxiliary propulsion are;

a. Reaction control system penalties for maintaining the Shuttle in its maximum drag orientation.

b. Fluid boil-off over long transfer times.

c. Optimization of outflow throttling schedules near the end of transfer.

d. Use of Shuttle vernier engines near the end of transfer to minimize residuals.

e. Potential of transferring at altitudes less than 296 km (160 n.mi.) to increase drag.

f. Liquid settling requirements prior to start of transfer.

In order to resolve the many unknowns described above, detailed conceptual designs were developed for both the drag and auxiliary propulsion concepts.

In the case of Shuttle supply (Table 4-14) the surface tension and fluid rotation concepts appear to have the best potential. It is noted that in this system, weight is most critical and therefore the surface tension concept was chosen for its low weight and high potential safety over the fluid rotation concept.

For the Solar Electric Propulsion receiver (Table 4-15) the diaphragm system was chosen as best, primarily on the basis of low weight and potentially high reliability. For the mercury application, residual fluid weights, which are lowest for the diaphragm system, are a significant factor.

In the case of the Large High Energy Observatory B (Table 4-16) the fluid rotation concept was chosen for further consideration, primarily due to its lower development risk which is based on the fact that it is a positive force system.

The surface tension device was chosen for the Satellite Control Section supply (Table 4-17) over fluid rotation due to a slightly lower weight and over the bladder due to reusability. A channel type surface tension system for low-g engine feed is currently employed in the SCS.

Work to generate conceptual designs of overall transfer systems employing the above acquisition schemes is presented in the following paragraphs.

4.2 CONCEPTUAL SYSTEMS DEFINITIONS

Conceptual definitions were accomplished for the four supply cases listed below.

System No. 1 (Tug In-Orbit Supply Separated From Shuttle) - H_2, O_2, He, N_2H_4 - Linear Acceleration With Supply Module and Tug Separated From the Shuttle Orbiter.

System No. 2 (Tug In-Orbit Supply Using Shuttle Drag) - H_2, O_2, He, N_2H_4 - Linear Acceleration With Tug Attached to the Shuttle Orbiter.

System No. 3 (Shuttle Orbiter In-Orbit Supply System). N_2O_4, MMH, He, H_2, O_2 for OMS, EPS, ECLSS Systems - Surface Tension Screens.

System No. 4 (Multiple Receivers In-Orbit Supply System) - Solar Electric Propulsion Stage, Hg, Diaphragm - Large High Energy Observatory, LHe, Paddle Vortex - Satellite Control Section, N_2H_4 - Surface Tension Screens.

Trade-offs and final system designs for each of the above systems are presented in the following paragraphs. A general thermal anslysis to define insulation systems for the cryogenic systems is contained in Appendix D.

4.2.1 TUG IN-ORBIT SUPPLY SEPARATED FROM SHUTTLE - This system concept, illustrated in Figure 4-20, is essentially the same as presented in Reference 4-4, except that in the current system the supply tanks are not constrained to any particular shape. For settling and transfer of LO_2 and LH_2 the supply module and Tug are completely separated from the Shuttle Orbiter. Linear acceleration is applied in a direction perpendicular to the Shuttle orbit plane, resulting in a cyclic path which, under ideal conditions, is coincident with the Shuttle position at one point in each revolution. The propulsion module providing linear acceleration is assumed to be part of the supply module.

Other ground rules used in the current analyses are presented below.

a. Maximum supply module length is 12.2 m (40 ft) to allow for Tug payloads up to 6.1 m (20 ft).

b. Initial liquid settling propulsion requirements are negligible in relation to the overall transfer (Ref. 4-4).

The major trade-offs and analyses performed to define the overall transfer system are presented in the following paragraphs.

4.2.1.1 Thrust System Investigation - Any of the following propellants could be used to provide the linear acceleration necessary for transfer.

a. N_2H_4 stored as part of the supply.

b. N_2H_4 stored in dedicated tankage.

c. N_2O_4/MMH in dedicated tankage.

d. GO_2/GH_2 in high pressure bottles.

e. LO_2/LH_2 stored as part of the main supply.

f. LO_2/LH_2 stored in dedicated tankage.

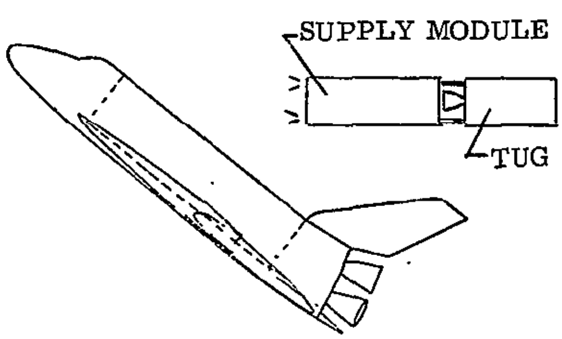
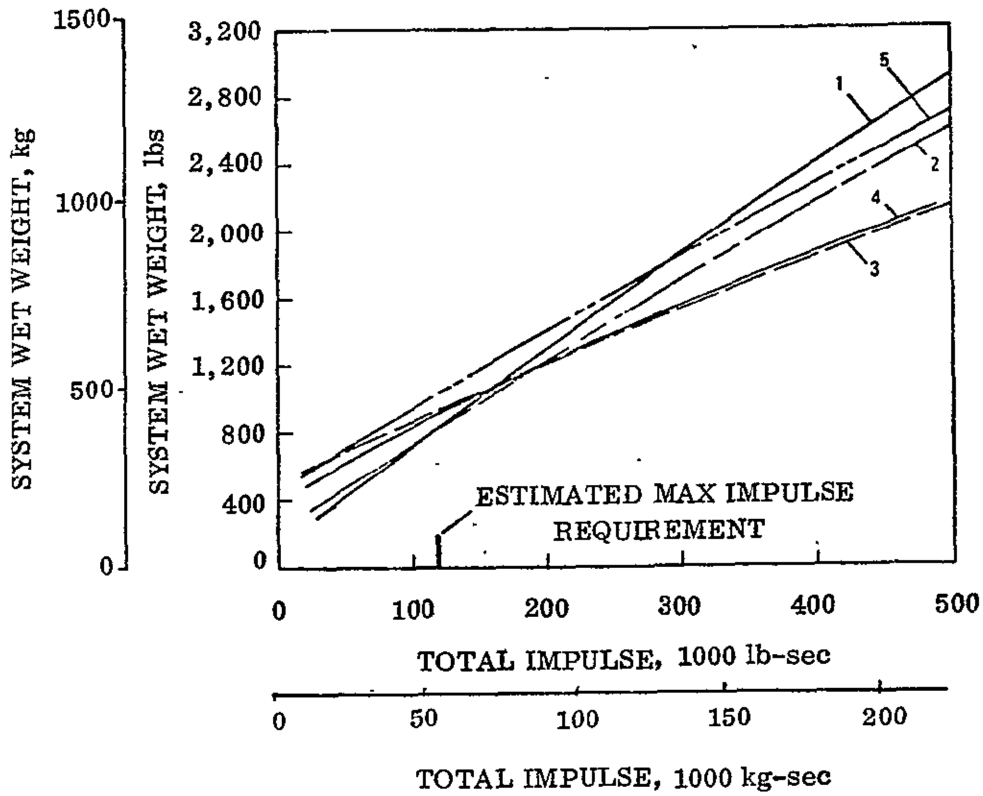


Figure 4-9. Separated Space Tug Supply

Tradeoffs between b. through f. type systems were accomplished by Convair in connection with

Space Tug attitude control. Results are reported in Reference 4-8. Comparative weight data from this reference are presented in Figure 4-10 as a function of total impulse.

For the current case the Tug plus supply module weight is estimated at 28,148 kg (62,000 lb). Then, assuming a reasonable transfer time of 18,000 sec (5 hr) at an acceleration of 1×10^{-4} g's (Ref. Figure 4-7), the total required impulse is 50,848 kg-sec (112,000 lb-sec). Therefore, from Figure 4-10, the minimum weight system would use either N_2H_4 or N_2O_4/MMH as propellants. For the present application the N_2H_4 could be stored as part of the supply with the potential of even lower overall system weight. Also, the N_2H_4 monopropellant system is the simplest and was therefore chosen for use in the current study.



1. N_2H_4 - Diaphragm Tankage
2. N_2O_4 /MMH - Bellows Tanks
3. Gas/Gas O_2/H_2 - Turbopump-Fed Hot Gas Conditioning
4. Liquid/Liquid O_2/H_2 - Pumped Main Tank LH_2 Pressurization Dedicated Storage LO_2
5. Liquid/Liquid O_2/H_2 - Pressurized Dedicated Storage LH_2 and LO_2

Figure 4-10. Tug ACS Wet Weight Versus Total Impulse

4.2.1.2 Tank Geometry and Residuals Optimization - The analysis in this section is concentrated on the transfer of O_2 and H_2 since they represent by far the bulk of the fluids to be transferred. The O_2 and H_2 are assumed to be transferred simultaneously. Transfer of the He and N_2H_4 is studied separately.

A trade-off was made to determine the optimum shapes and packaging of the LO_2 and LH_2 supply tanks. The weight variation with L/D of a constant volume tank was estimated to be

$$\text{Tank Weight} = \text{Constant}\left[\frac{2\,(L/D) - 1}{3\,(L/D) - 1}\right] \qquad (4\text{-}6)$$

The following assumptions were used in the derivation.

a. Tanks are cylindrical with hemispherical ends. A sphere is a special case with L/D = 1.

b. The cylindrical portion of the tank has a wall thickness, $t_c = PD/2\,S_{Tu}$. For the spherical ends, $t_s = PD/4\,S_{Tu}$.

Using Equation 4-6 and the NAR residual calculation methods described in Appendix C, it was determined that the smaller the tank diameters the lower should be the total system weight. Maintaining a maximum overall supply module length of 12.2 m (40 ft) resulted in the following minimum tank diameters for the current fluid requirements.

LO_2 tank = 1.52 m (60 in.) by 10.7 m (421 in.) long

LH_2 tank = 2.64 m (104 in.) by 10.7 m (421 in.) long

Residuals calculated for these tanks are presented in Figure 4-11 as a function of transfer time. Propulsion system weight data from Figure 4-6 are then used to obtain residual plus propulsion weights as a function of transfer time. These data are presented in Figure 4-12. From Figure 4-12 the optimum transfer time is 2.5 hours. Comparing weights from Figures 4-7 and 4-12 shows a residual plus propulsion weight saving of 334 kg (735 lb) for the small diameter tanks.

Based on Equation 4-6, the maximum weight penalty of the small diameter tanks (Figure 4-13b) over the larger diameter tanks (Figure 4-13a) was determined to be 133 kg (294 lb). The net saving is thus 200 kg (441 lb) and the tank configuration presented in Figure 4-13b was chosen for further detailed design.

Use of a transfer acceleration of 10^{-4} g's is based on the trade-offs performed in Paragraph 4.1.4.

4.2.1.3 Receiver Pressure Control - Initially, calculations were made to determine the feasibility of a non-vent fill of the Tug H_2 and O_2 tanks. Using tank volumes, weights, materials and initial wall temperatures from Table 4-1, energy calculations for non-vent filling resulted in the final tank pressures presented at the top of the next page. Mixed fluid conditions were assumed to exist in the receivers at all times.

Referring to Table 4-1, maximum tank pressure allowables are exceeded for the H_2 case, but not for the O_2 case. Therefore H_2 tank venting would be required during chilldown. Even though it would theoretically be possible to not vent the O_2 receiver,

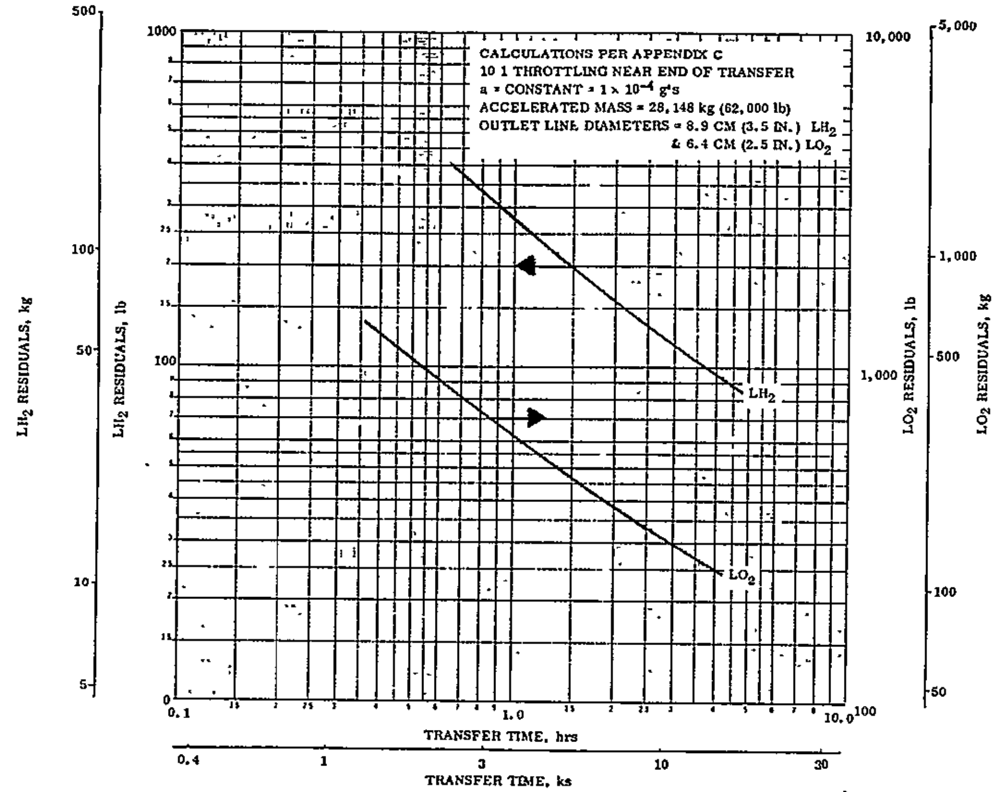


Figure 4-11. Residuals Versus Transfer Time for Separated Tug Supply

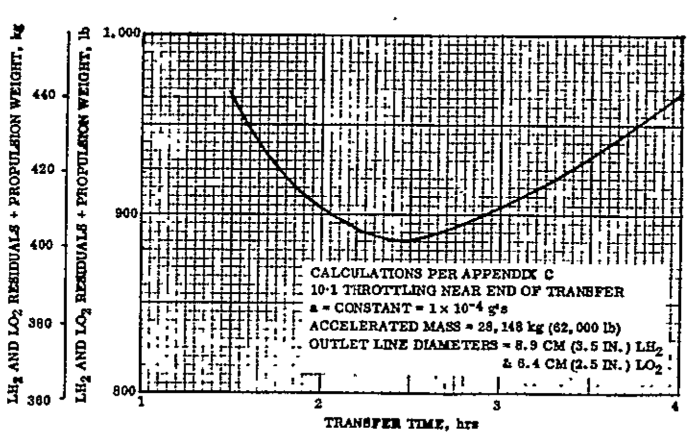


Figure 4-12. Variable System Weight Versus Transfer Time for Separated Tug Supply

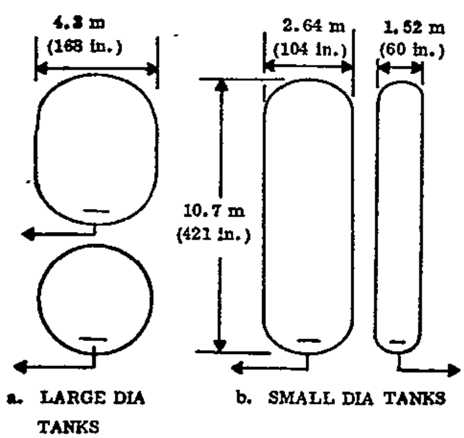


Figure 4-13. Large and Small Diameter Tug Supply Tankage

Initial Receiver Fluid Condition	Final Tank Pressure, kN/m^2 (psia) LO_2 Tank	LH_2 Tank
Completely Empty (Vacuum)	116 (16.8)	155 (22.5)
Gas at 69 kN/m^2 (10 psia)	117 (17.0)	166 (24.0)

it was determined that, without venting, a full load of liquid could not be transferred. This is due to reduced density of the liquid at final saturation pressures, without venting, higher than the initial load. Therefore it was assumed that venting would be accomplished for both H_2 and O_2 cases to maintain a final saturation pressure in the receiver equal to that in the supply.

Assuming a settling acceleration of 10^{-4} g's, calculations were made to determine the possibility of the liquid inflow jet rising to the vent during chilldown. Following chilldown the normal vents in both H_2 and O_2 tanks can be closed, and during fill, pressure control is maintained through use of a thermodynamic vent system. Jet rise calculations were made using the methods described in Reference 4-9. Results are presented in Table 4-22 as a function of inlet line diameter.

The flow rates used here are based on a total transfer time of 9 ks (2.5 hrs). Since the actual inlet lines are 15.2 cm (6 in.) and 12.7 cm (5 in.) respectively, for H_2 and O_2 cases, flow rates will need to be reduced during chilldown and/or inlet flow baffles used. Additional work is needed to accurately define the effect of inlet baffling. Data from References 4-10 and 4-11 indicate that, by use of baffles, inflow velocities can be increased by factors of 5 to 11 and still maintain a stable inflow. Assuming an allowable inlet velocity increase of 10:1 without effecting jet height, then referring to Table 4-18, line diameters greater than 9.7 cm (3.8 in.) would result in jet liquid heights less than the tank lengths (70.2% for H_2 and 9.8% for O_2).

4.2.1.4 Tank Pressurization and Helium Transfer - Various O_2 and H_2 supply tank pressurization schemes were considered in conjunction with transfer of helium to the Tug. In the initial screening (Paragraph 4.1) LHe was supplied, however, further analysis showed that for the small quantity involved it was optimum to combine the He supply with the LH_2/LO_2 GHe pressurant system. Methods used for calculating pressurant requirements and storage bottle weights are presented in Appendix E. Collapse factors calculated for the H_2 and O_2 tanks shown in Figure 4-13b are presented in Table 4-19 for different pressurant conditions. The collapse factor is defined below.

Actual Pressurant Required = Ideal Pressurant × Collapse Factor (CF)

Ideal Pressurant = Expelled Liquid Volume × Pressurant Density at Expulsion Press. and at Temp. of Pressurant Inlet

Pressurant, storage and heating system weights (where applicable) were then calculated for the pressurant plus helium transfer schemes listed below. Corresponding data are presented in Table 4-20.

Table 4-18. Inflow Jet Rise Heights

LO_2 Tank Length = 2.6 m (101.8 in.)
LH_2 Tank Length = 4.4 m (174.5 in.)
a = 10^{-4} g's
LO_2 Inlet Flow = 4.7 m^3/ks (10 cfm)
LH_2 Inlet Flow = 1.2 m^2/ks (2.5 cfm)

Inlet Line Dia.		Jet Rise in % of Tank Length	
cm	(in.)	LH_2	LO_2
15.2	(6)	-	117.0
30.5	(12)	70.2	9.8
40.6	(16)	25.5	3.7
61.0	(24)	6.4	1.0

Table 4-19. Pressurant Collapse Factors

Tank Pressure
138 kN/m^2 (20 psia) for H_2 Tank
172 kN/m^2 (25 psia) for O_2 Tank

Tank	Pressurant Inlet Temp. K	(R)	Pres-surant	Collapse Factor
H_2	289	(520)	He	2.71
H_2	167	(300)	He	1.88
H_2	90.6	(163)	He	1.37
H_2	167	(300)	H_2	1.78
O2	289	(520)	He	2.12
O_2	167	(300)	He	1.42

Table 4-20. Pressurant Plus Helium Transfer System Weights (4.1 kg, 9 lb of Helium Supplied to Tug)

System	Total He kg	(lb)	Storage Bottle(s) Weight kg	(lb)	Other Fluids and Hardware kg	(lb)	Total Weight kg	(lb)
1	46.8	(103)	278	(612)	-		325	(715)
2	48.6	(107)	69.9	(154)	27.2	(60)	146	(321)
3	72.2	(159)	200	(440)	-		272	(599)
4	20.9	(46)	108	(238)	52.2	(115)	181	(399)

1. Helium for pressurant and transfer stored and used at ambient temperature. Helium supplied to Tug by bottle blowdown from 33,120 kN/m^2 (4800 psia) to 22,770 kN/m^2 (3300 psia), followed by O_2 and H_2 supply tank pressurization.

2. Helium for pressurant and transfer stored at 33,120 kN/m^2 (4800 psia) in H_2 tank. Helium supplied cold to Tug by blowdown prior to O_2 and H_2 tank pressurization. H_2 and O_2 tank pressurization with He heated to 289K (520R) using a hydrazine gas generator and heat exchanger.

3. Helium stored at 33,120 kN/m^2 (4800 psia) at LO_2 temperature. Helium supplied cold to Tug by blowdown, followed by unheated pressurization of LH_2 and LO_2 supply tanks.

4. Helium for Tug supply stored in H_2 tank at 22,770 kN/m^2 (3300 psia) and electrically heated to maintain pressure for transfer. LH_2 from main tank converted to GH_2 at 167K (300R) for H_2 tank pressurization. Helium stored at 33,120 kN/m^2 (4800 psia) and ambient temperature used without heating for O_2 tank pressurization.

In the previous calculations the ambient temperature is assumed to be 289K (520R). Based on trade-offs between line and line residual weights (line size) and pressurant system weight the optimum supply tank pressures were found to be 173 kN/m^2 (25 psia) for the LO_2 tank and 138 kN/m^2 (20 psia) for the LH_2 tank. These pressures were used to determine weights of the above systems.

Titanium bottles were employed for cold storage of helium and composite type Kevlar wrapped bottles were used for ambient storage (Appendix E).

From Table 4-20, system number 2 is the lightest. It is also a relatively simple system, since N_2H_4 is already carried for transfer. This system was thus chosen for the current application.

4.2.1.5 Separated Tug Final Conceptual Design Data - An overall schematic of the system and conceptual design drawings are presented respectively in Figures 4-14 and 4-15. System weights and a fluids inventory are presented in Tables 4-21 and 4-22. Fluid quantities presented are based on supplying initially empty receivers. Additional hardware weight data are presented in Appendix F.

This system employs linear acceleration of the Space Tug and supply system at 10^{-4} g's separated from the Shuttle Orbiter. A hydrazine propulsion system is employed to provide the necessary accelerations. The N_2H_4 storage tank is also used to supply the gas generator for pressurant heating and the N_2H_4 required for transfer to the Tug. The N_2H_4 tank, for settling and for gas generator operation employs a pressure of 2070 kN/m^2 (300 psia). A bladder is used to insure start and operation prior to application of liquid settling acceleration. Transfer of hydrazine to the Tug is accomplished following linear acceleration, after LO_2 and LH_2 transfer, with the N_2H_4 tank allowed to blowdown from 2070 kN/m^2 (300 psia) to approximately 689 kN/m^2 (100 psia).

Table 4-21. Weights for Separated Tug Supply

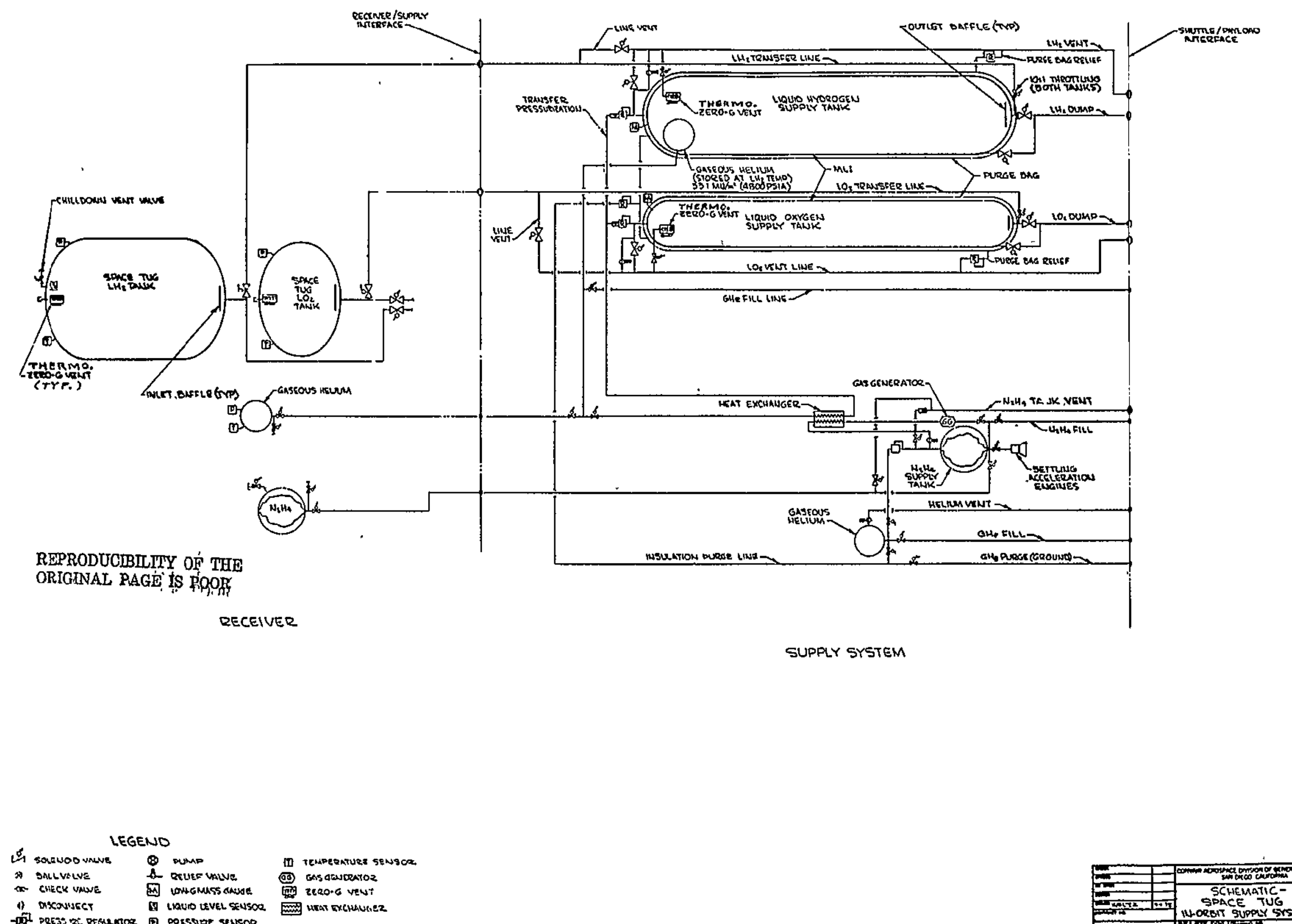
Item	Weight kg	Weight lb
LH_2 System (Dry)	472	1039
LO_2 System (Dry)	528	1164
N_2H_4 System (Dry)	40	89
Ambient GHe System (Dry)	37	81
Cold He Pressurization and Transfer System (Dry)	103	226
N_2H_4 Settling Thrusters and Plumbing	6	14
Tug Attach Fittings and Legs	11	24
Swing-Out Ring	132	290
Overall Module Support Structure	312	688
Total Dry Weight	1641	3615
Total Fluids (Ref. Table 4-26)	23979	52818
Lift-Off Weight	25620	56433
Total Fluids Supplied (Ref. Table 4-26)	23401	51545
Total Fluid Residuals (Ref. Table 4-26)	266	587
Total Return Weight	1908	4202
Fluid Supplied/Lift-Off, %		91.3
Total Fluid Residuals, %		1.1

Helium stored at 33120 kN/m^2 (4800 psia) and ambient temperature is used for pressurization of the N_2H_4 bladder tank and for purge pressurization of the LH_2 and LO_2 insulation systems during re-entry.

Helium is transferred to the Tug from a high pressure 33120 kN/m^2 (4800 psia) bottle stored

FOLDOUT FRAME 1

FOLDOUT FRAME 2



CONVAIR AEROSPACE DIVISION OF GENERAL DYNAMICS SAN DIEGO CALIFORNIA
SCHEMATIC - SPACE TUG IN-ORBIT SUPPLY SYSTEM
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FIGURE 4-14. SCHEMATIC - SEPARATED TUG SUPPLY

C-2

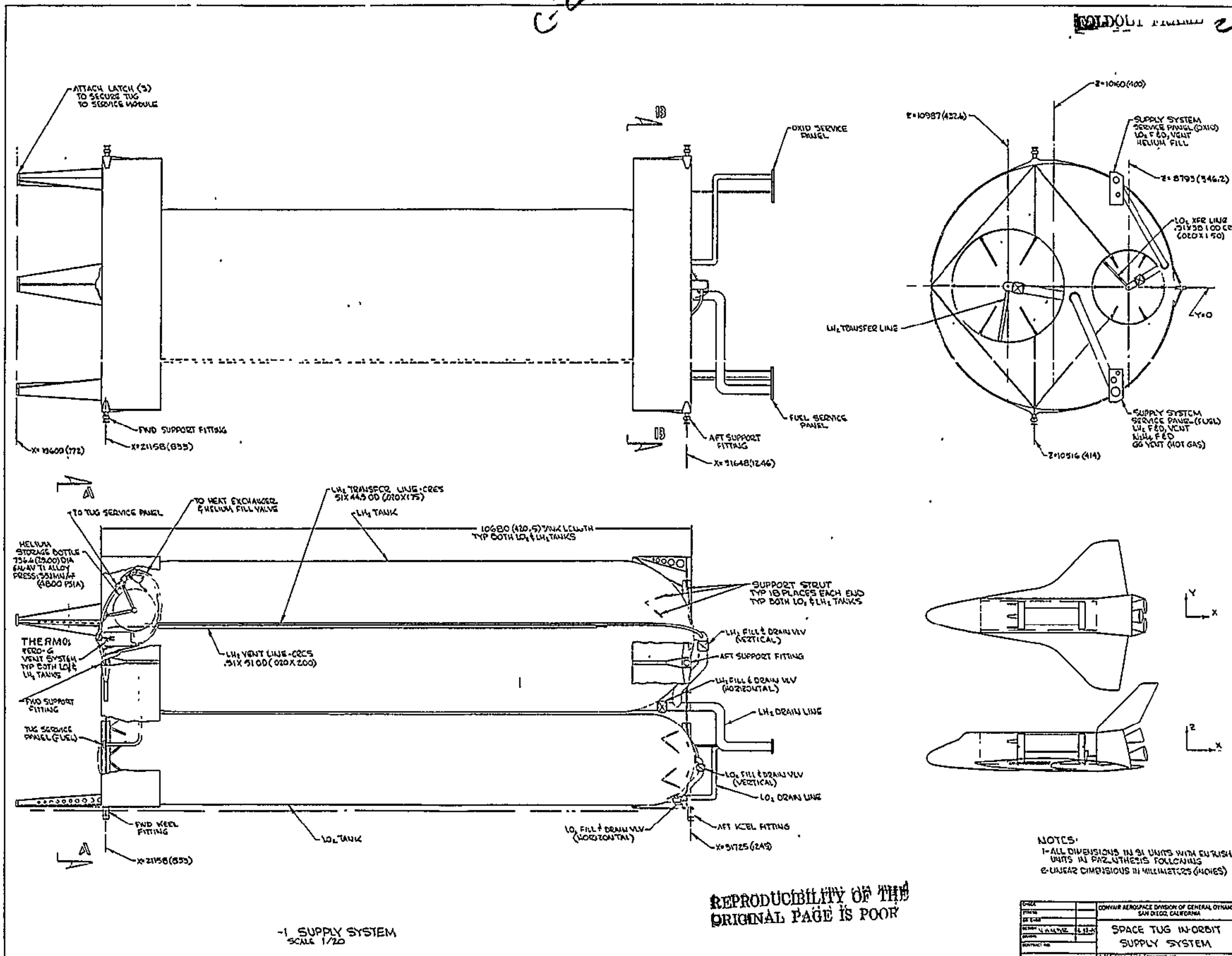

FIGURE 4-15. SEPARATED TUG SUPPLY

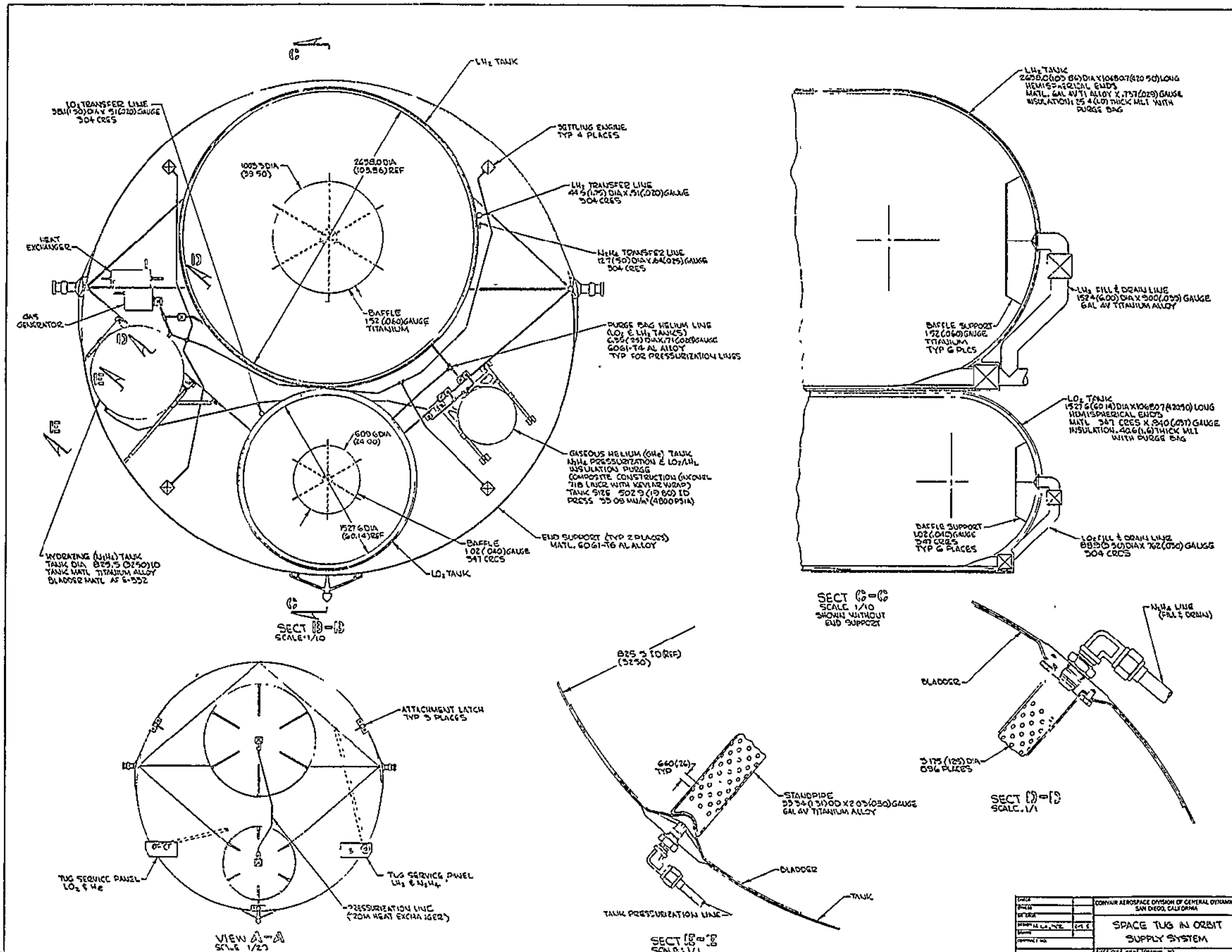

FIGURE 4-15. SEPARATED TUG SUPPLY-CONT.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 4-22. Fluids Inventory

Use	Fluid, kg (lb)									
	LH_2		LO_2		Amb GHe		Cold GHe		N_2H_4	
Required for Receiver Mission	3462	(7626)	19782	(43574)	-		4.1	(9)	153	(336)
Transfer Line and Receiver Childown	58	(128)	70	(154)	-		0	0	-	
Receiver Boil-Off During Transfer	0.9	(2)	1.4	(3)	-		-		-	
Transfer Line Residuals	1.8	(4)	20	(44)	-		-		0.1	(2)
Total Transferred	3523	(7760)	19874	(43775)	-		4.1	(9)	153	(338)
Supply Residuals	60	(133)	151	(333)	0.05	(0.1)	3.2	(7)	6	(13)
Supply Vented Prior to Transfer	31	(68)	-		-		-		-	
Fluid Used for Press., Purge & Safing	-		-		3.2	(6.9)	42.7	(94)	-	
Liquid for Gas Generator	-		-		-		-		18	(39)
Liquid for Acceleration	-		-		-		-		110	(242)
Total Supply Module Load	3614	(7961)	20025	(44108)	3.2	(7.0)	50	(110)	287	(632)

in the LH_2 tank and which is also used for LO_2 and LH_2 supply tank pressurization. Helium transfer is accomplished prior to the LO_2 and LH_2 transfer. For LO_2 and LH_2 tank pressurization,for transfer and abort dump,the helium pressurant is heated to 289K (520R) by a hydrazine gas generator.

The LH_2 and LO_2 tanks are long cylinders, 2.6 × 10.7 m (104 × 420 in.) and 1.5 × 10.7 m (60 × 420 in.) respectively, with hemispherical ends employing outlet baffles to minimize liquid residuals. The use of long cylinders significantly reduces residuals, for a given pull-through height, over that expected from spheres or large diameter tanks. Flow control valves are located at the tank outlets to throttle the liquid flow rate near the end of transfer to one-tenth of full-flow, to further reduce residuals. The optimum transfer time was found to be 9 ks (2.5 hrs) with the LH_2 and LO_2 transferred simultaneously over this time period. Both tanks employ Superfloc multilayer insulation [2.5 cm (1.0 in.) for LH_2 and 4.1 cm (1.6 in.) for LO_2] enclosed by rigid purge bags to prevent moisture condensation and/or freezing during ground hold, boost and re-entry.

The receiver oxygen tank is assumed to be locked-up during transfer, except that the thermodynamic vent system is used to maintain a nominal 107 kN/m^2 (15.5 psia) liquid saturation pressure to maximize the amount of liquid received. The Tug hydrogen tank is assumed to be vented during chilldown, and liquid inlet baffles are provided to prevent direct liquid loss at the vent. Following chilldown the tank is locked-up, except for the thermodynamic vent which will operate to maintain the required liquid vapor pressure for maximum loading.

LH_2 and LO_2 transfer lines are 4.4 cm (1.75 in.) dia by 0.05 cm (0.010 in.) wall and 3.8 × 0.05 cm (1.5 × 0.020 in.) CRES respectively, except at the supply tank outlets where diameters are 8.9 and 6.4 cm (3.5 and 2.5 in.) respectively.

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The two long cryogenic tanks are placed side-by-side and attached to a common bulkhead at each end. These bulkheads provide the support fittings to attach the tanks to the shuttle cargo bay. The spherical tanks for storage of the GHe and N_2H_4 and other hardware; settling acceleration engines, etc, are also attached to the bulkheads.

The two cryogenic tanks, one for liquid oxygen and one for liquid hydrogen, are of equal length, 10680 mm (420 in.), allowing easy attachment to the bulkheads. They are attached to the bulkheads by fiberglass struts for minimum heat transfer between the tank and bulkhead. Both tanks have hemispherical ends with a flat plate baffle located over the outlet end. Each tank has a dump line which also acts as a sump during in-orbit fluid transfer since the dump line is much larger than the transfer line. The dump line is arranged to allow draining the tank in either the vertical or horizontal position, although the baffle covers only the vertical drain.

The hydrogen tank material is titanium alloy with a skin gauge of 0.737 mm (0.029 in.) A gaseous helium storage bottle is mounted inside the hydrogen tank at the forward end and is of conventional design titanium alloy 736.6 mm (29 in.) in diameter.

The oxygen tank material is 347 CRES with a skin gauge of 0.940 mm (0.037 in.)

The bottle mounted on the aft bulkhead containing ambient temperature helium is of a composite construction consisting of an Inconel 718 liner with a Kevlar overwarp. It is 503 mm (19.8 in.) in diameter.

The spherical hydrazine tank located on the aft bulkhead is titanium, 825 mm (32.50 in.) in diameter with a skin gauge of 2.38 mm (0.093 in.) It has an internal bladder (material AF-E-332) and a standpipe for zero-g operation.

The gas generator and heat exchanger (used for heating the helium for tank pressurization) are of conventional design and are mounted on the aft bulkhead.

Three struts are attached to the forward bulkhead to allow the supply module to attach to the Space Tug. These struts are 1549 mm (61 in.) long to allow clearance for the Space Tug engine. Fuel and oxidizer service panels similar to those in the Shuttle cargo bay, are mounted on the supply module forward bulkhead so that the Tug can be connected to the supply module in the same manner as it is connected in the shuttle cargo bay.

4.2.1.6 Separated Tug Operating Procedure - A functional flow diagram of the overall supply operation is presented in Figure 4-16. The various operations outlined in Figure 4-16 are expanded and discussed in the following paragraphs. A schematic of the complete transfer system is presented in Figure 4-14.

Supply Module Ground Preparation - Leak checks and purging and loading of N_2H_4 into the bladder tank are performed. LO_2 and LH_2 tank purging is designed to insure

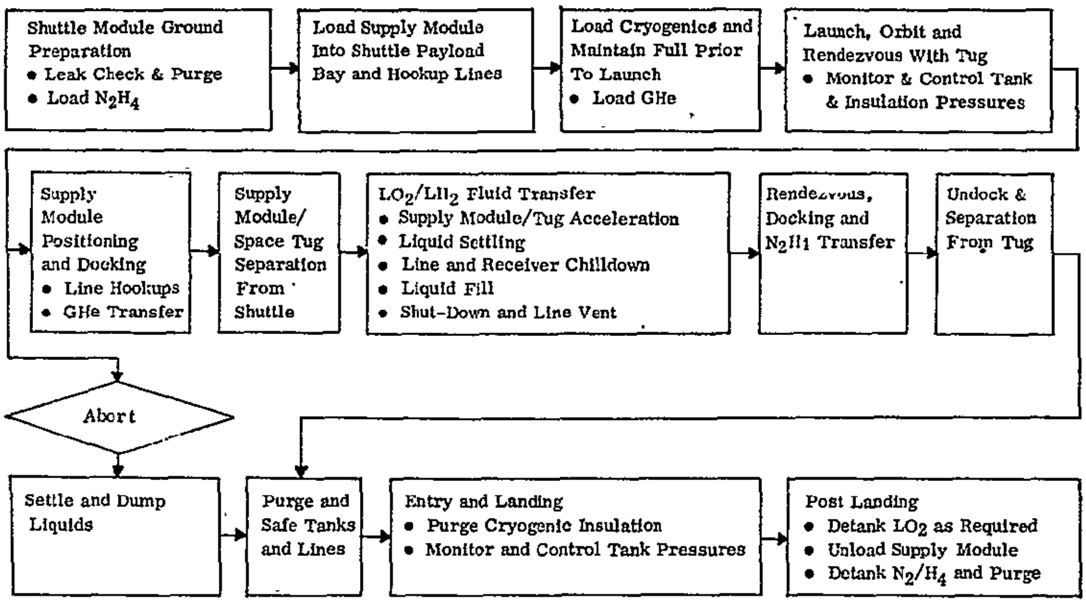

Figure 4-16. Overall Operations for Separated Tug In-Orbit Supply

that gases and/or liquids which are condensible and/or which may solidify at cryogenic temperatures are eliminated.

Load Supply Module Into Shuttle - This simply consists of the installation of the supply module in the Shuttle Orbiter cargo bay and the hookup of all ground supply, abort and in-orbit transfer lines.

Load and Maintain Cryogenics - At this point the cryogens, LO_2 and LH_2 are loaded into the supply module. This operation consists of initial line and tank chilldown and final loading. It is assumed that the maximum fluid loading is maintained by topping up to 180 seconds prior to launch. The high pressure helium storage bottles are also filled at this time. The cryogenic insulation for the LO_2 and LH_2 tanks is maintained in a purged condition with GHe from the ground prior to boost.

Launch, Orbit and Rendezvous - During this phase of the resupply operation the main problems are to control tank and insulation pressures. Venting of the LH_2 and LO_2 tanks is allowed for during the boost phase through the normal ground boiloff vents, although under normal heating conditions O_2 tank venting should not occur. Following entry into orbit, the thermodynamic vent systems are designed to accommodate any venting required. Tank overpressure is a safety critical failure and provisions are made to allow the immediate dumping of liquids as well as emergency venting.

Abort (Only as Necessary) - If for some reason it becomes necessary to terminate the mission during boost or while in orbit, all liquids are dumped. Liquid settling to the aft end of the Shuttle is provided by the normal Shuttle abort procedure. Redundant valving is provided to insure that abort dumping can be accomplished.

Supply Module Positioning and Docking - Subsequent to orbital rendezvous and prior to docking, the supply module is rotated out of the cargo bay to a position which allows docking of the Tug to the exposed end of the module. The remote manipulator is used for both module rotation and Tug docking. Prior to rotation of the module, all vent and abort lines interfacing with the Shuttle are disconnected. Orbital transfer line hookups will be accomplished upon docking of the Tug or immediately thereafter. It is assumed that all line and electrical connections and disconnections are accomplished remotely, and that all lines are empty and vented to the atmosphere (vacuum) prior to hookup. Following connection to the Tug all electrical hookups to the Shuttle are disconnected.

Following rendezvous and docking and prior to separation from the Shuttle, the high pressure 22770 kN/m^2 (3300 psia) Tug helium bottle is supplied from the 33120 kN/m^2 (4800 psia) LH_2/LO_2 supply module pressurant bottle stored in the LH_2 tank. Since the helium being supplied is cold, the Tug helium bottle fluid temperature and pressure must be closely monitored to determine when sufficient mass has been transferred.

Separation From Shuttle - In preparation for settling and transfer of LO_2 and LH_2 the supply module and Tug are completely separated from the Shuttle Orbiter. A safe separation distance prior to initiation of supply module/Tug linear acceleration is assumed, from Reference 4-4, to be 8 km (5 miles).

LO_2/LH_2 Fluid Transfer - The first operation following separation from the Shuttle is to initiate linear acceleration and settle the LO_2 and LH_2 at the supply tank outlet for transfer. Based on results of the study described in Reference 4-4, the linear acceleration is assumed to be applied in a direction perpendicular to the Shuttle orbit plane. This results in a cyclic orbital path which, under ideal conditions, is coincident with the initial orbit at one point in each revolution. The total thrusting time for the present case is estimated to be between 9 and 10.8 ks (2.5 and 3.0 hrs) over approximately two complete orbits.

The LO_2 and LH_2 are transferred simultaneously and the following basic series of events are employed.

a. LH_2

(1) Activate supply tank pressurization to 138 kN/m^2 (20 psia).

(2) Open receiver vent. A normal ground type vent will be used here, possibly in connection with a liquid/vapor separator. Further technology work is needed to determine the need for and/or configuration of such a system.

(3) Open receiver tank inlet valves and supply tank low-flow valve and allow line and receiver tank to chill down to LH_2 temperature. Inlet baffles are provided to minimize liquid at the vent.

(4) Close receiver tank vent, except for the thermodynamic vent, and increase transfer flow to the maximum allowed. The receiver thermodynamic vent is used to maintain the liquid saturation pressure at near 110 kN/m^2 (16 psia) for maximum loading density.

(5) Reduce flow rate by 10:1 to prevent vapor pull-through as tank nears depletion.

(6) Close receiver tank inlet valves and supply tank outlet valves and vent the transfer lines to vacuum.

b. LO_2

(1) Actuate supply tank pressurization to 172 kN/m^2 (25 psia).

(2, 3, 4) Open receiver tank inlet valves and supply tank outlet valves to maximum flow conditions. The Tug O_2 tank is assumed to be locked-up during both chilldown and transfer, except that the receiver zero-g heat exchanger vent system is used to maintain the liquid saturation pressure at near 110 kN/m^2 (16 psia) for maximum loading density.

(5) Same as for LH_2.

(6) Same as LH_2.

Rendezvous, Docking and N_2H_4 Transfer - Following transfer of the LO_2 and LH_2, the supply module and Tug are rendezvoused with the Shuttle within reach of the Shuttle remote manipulator. Using this manipulator the supply module/Tug is locked into the cargo bay in the rotated position which existed before separation. At this point, N_2H_4 from the bladder supply tank is transferred to the Tug bladder tank. During this transfer the supply tank pressure of 2067 kN/m^2 (300 psia) used to supply the linear acceleration thrusters is allowed to blowdown to a final pressure of 689 kN/m^2 (100 psia).

Undock and Separation From Tug - The Tug is separated from the Shuttle Orbiter and then the supply module is rotated back into the cargo bay and any required overboard vent and dump lines interfacing with the Shuttle, are reconnected. Electrical lines are connected to the Shuttle Orbiter prior to undocking of the Tug so that a continuous supply of power will be available to the supply module.

Purging and Safing - Prior to entry and landing, LO_2 and LH_2 residuals are completely dumped and the tanks purged with helium. This is especially critical in the case of H_2, where venting cannot be accomplished immediately upon landing, as would be required due to entry heating if LH_2 were remaining.

Entry and Landing - The primary operations here are to (1) control pressure in the insulation purge bags to slightly above ambient to prevent entry of air with subsequent moisture condensation and insulation damage, and (2) maintain tank pressures above ambient to prevent implosion and below the design pressure to prevent bursting.

Post Landing - Following landing of the Shuttle Orbiter, any remaining LO_2 is detanked. The complete supply module is then unloaded from the Shuttle Orbiter and any remaining N_2H_4 drained. All tanks are then purged and locked-up with inert gas and the complete system stored.

4.2.2 TUG IN-ORBIT SUPPLY USING SHUTTLE DRAG - This system is designed to utilize Shuttle Orbiter drag to orient the LO_2 and LH_2 at tank outlets; such that transfer can be accomplished without removal of the supply module from the Shuttle. This also eliminates the need for rendezvous of the Tug/Supply Module with the Shuttle following transfer and the incorporation of a propulsion system into the supply module.

For maximum drag during transfer, the Shuttle Orbiter is assumed to be oriented as shown in Figure 4-17. This orientation also allows the Shuttle 11.4 kg (25 lb) vernier RCS engines to be used to provide initial liquid settling and scavenging of residuals near the end of transfer. Use of the 431 kg (950 lb) Shuttle RCS engines was not found to be weight effective; i.e., propellant usage is much greater than savings in residuals.

Calculations are performed for transfer at both 185 km (100 n. mi.) and 296 km (160 n.mi.). Drag, shuttle RCS requirements to maintain the required Shuttle orientation, and vernier engine characteristics used here are presented in Table 4-23. Other ground rules used are listed below.

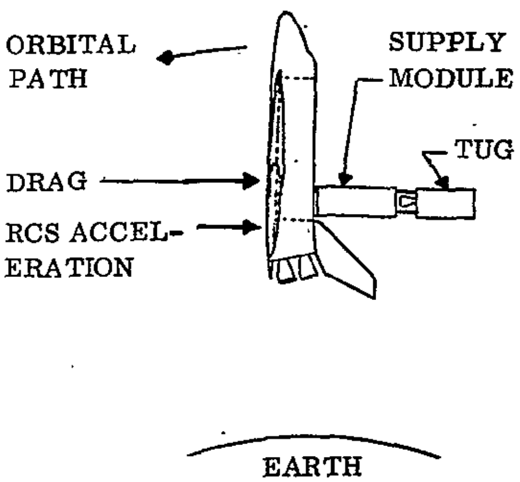

Figure 4-17. Space Tug Supply Using Shuttle Drag

a. Maximum supply module length is 12.2 m (40 ft) to allow for Tug payloads up to 6.1 m (20 ft). Based on analyses described in Paragraph 4.2.1.2 use of the minimum diameter tankage allowable (maximum length) was found to be optimum.

b. There are no limitations on transfer time as a result of sun heating. This was determined from an evaluation of all missions presently anticipated for the Tug.

c. At present, it is unknown whether or not RCS vernier thrusting must be charged as a weight penalty to the supply module. For final system design a weight penalty is assumed, however for comparison,

Table 4-23. Shuttle Orbiter Liquid Settling and Orientation Capabilities

	Orbital Altitude	
	296 km (160 n.mi.)	185 km (100 n.mi.)
Drag, g's(1)	1.09×10^{-6}	2.27×10^{-5}
RCS Usage to Maintain Vehicle Orientation(2), mg/sec (lb/hr)	47.9 (0.38)	189 (1.5)
Vernier Thrust, g's(1)(3)	2.92×10^{-4}	2.92×10^{-4}
Vernier Propellant Usage(3), kg/sec (lb/min)	0.14 (18.5)	0.14 (18.5)
Loss in Altitude Per Orbit Due to Drag, km (n.mi.)	Negligible	2.3 (1.25)

(1) Based on total accelerated weight of 99,880 kg (220,000 lb).
(2) From personal communication between R. Drowns of Convair and K. Lindsey at NASA/JSC on 7 Oct. 1975.
(3) From personal communication between J. Stark of Convair and Bob Budica of North American Rockwell on 25 September 1975.

preliminary data are presented for both cases.

d. The total Shuttle plus Tug plus supply module weight to be accelerated or decelerated is assumed to be 99,880 kg (220,000 lb).

The major trade-offs and analyses performed to define the overall system are listed below.

a. Requirements for liquid settling prior to transfer.

b. Optimization of overall transfer and vernier acceleration times and flow throttling.

c. Comparison of transfer at 185 km (100 n.mi.) and 296 km (160 n.mi.).

Basic tankage, transfer line and insulation configurations, receiver tank filling methods, supply tank pressurization and helium and hydrazine transfer systems are taken to be the same as defined for System No. 1 in Paragraph 4.2.1.

4.2.2.1 Initial Liquid Settling - Assuming supply tank diameters from Paragraph 4.2.1 (2.64 m, 104 in. for LH_2 and 1.52 m, 60 in. for LO_2), Bond numbers resulting from drag-only were calculated at 296 km (160 n.mi.) and at 185 km (100 n.mi.). Results are presented in Table 4-24.

Assuming Bond numbers greater than 0.84 are required to settle, it is seen from Table 4-24 that auxiliary settling would be required at 296 km (160 n.mi.), but not at 185 km (100 n.mi.). From Figure B-1 drag data, the maximum altitude without the need for auxiliary settling was determined to be 276 km (150 n.mi.).

Required settling times were estimated using two different methods; (1) five times liquid free fall (Ref. 4-12) and (2) vapor bubble motion (Ref. 4-13). Acceleration is applied using the vernier engines per Table 4-23. Calculated settling times are presented below.

Table 4-24. Bond Numbers Due to Drag-Only

	Tankage	Orbital Altitude	
		296 km (160 n.mi.)	185 km (100 n.mi.)
Bo Based on Tank Radius	H_2	0.66	13.7
Bo Based on Tank Radius	O_2	0.54	11.3

Five times free fall, time = 432 sec
Vapor bubble motion, time = 477 sec

In each case, oxygen dictated the maximum time. Tank lengths for both O_2 and H_2 were 10.7 m (421 in.).

From Table 4-23, RCS propellant usage is then 60.4 kg (133 lb) and 66.7 kg (147 lb). It is likely that in an operational case and with further technology work the actual thrust time and thus propellant usage could be significantly reduced by employing impulsive settling; i.e., auxiliary thrust application over only a portion of the total settling time.

4.2.2.2 Overall Transfer System Weight Optimization - The major variables affecting overall transfer system weight are liquid residuals, outflow rate or transfer time, flow throttling and scavenging acceleration. The current analysis assumes that the O_2 and H_2 are transferred simultaneously and that the He and N_2H_4 are transferred at another time using other means than linear acceleration. The basic sequence of events for the O_2 and H_2 transfer is to (1) outflow at maximum rates with drag-only settling until pull-through is imminent, (2) reduce flow rates by 10:1 to prevent pull-through as the tanks empty further and (3) where vernier engine scavenging is employed, actuate vernier engines, while increasing outflow if necessary to minimize total system weight.

RCS propellant weight penalties for scavenging are determined from Table 4-23. Overall supply module weights as a function of liquid residuals and transfer time are determined by reducing the weight data in Paragraph 4.2.1.5 to a parametric form. The resulting equation is presented below.

$$\text{Total Wt, lb} = 55,606 + 1.17\ (\text{LH}_2\ \text{Residual, lb}) + 1.037\ (\text{LO}_2\ \text{Residual, lb})$$

$$+ 3.64\ (\text{Transfer Time, hr}) \qquad (4\text{-}7)$$

$$\text{Total Wt, kg} = 25,245 + 1.17\ (\text{LH}_2\ \text{Residual, kg}) + 1.037\ (\text{LO}_2\ \text{Residual, kg})$$

$$+ 4.6 \times 10^{-4}\ (\text{Transfer Time, sec})$$

The above weight includes all fluids and hardware per Table 4-21, except that which is associated with the N_2H_4 acceleration system.

Using both NAR and LeRC data for calculating liquid residuals, per Appendix C, along with Equation 4-7 and data from Table 4-23, total system weights were determined as a function of total transfer time for a number of different cases. Results are presented in Figure 4-18. The total transfer time includes that allowed for scavenging. It does not include that required for liquid settling or that due to any reduced outflow rate which may be required during receiver chilldown. Total weights do not include the RCS propellant required for settling. The maximum flow throttling ratio is taken to be 10:1. For the scavenging cases, curves are shown with and without including a penalty for RCS usage. Corresponding liquid residuals, scavenging times and throttling-up ratios are presented in Figures 4-19 through 4-22.

Analysis showed that where scavenging is accomplished and an RCS weight penalty included, the minimum weight system results when outflow is throttled up as the

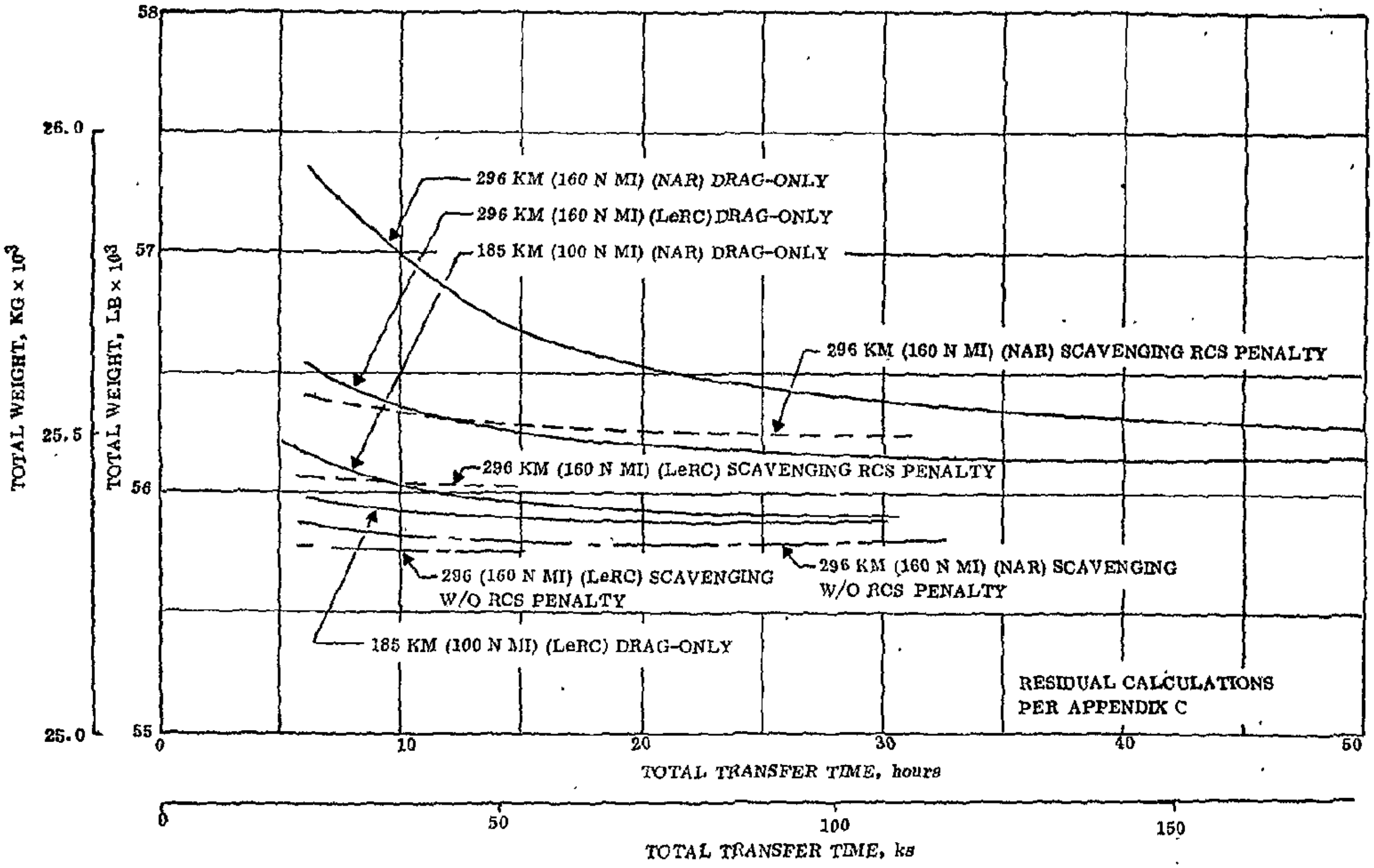

Figure 4-18. Drag System Supply Module Weight Optimization

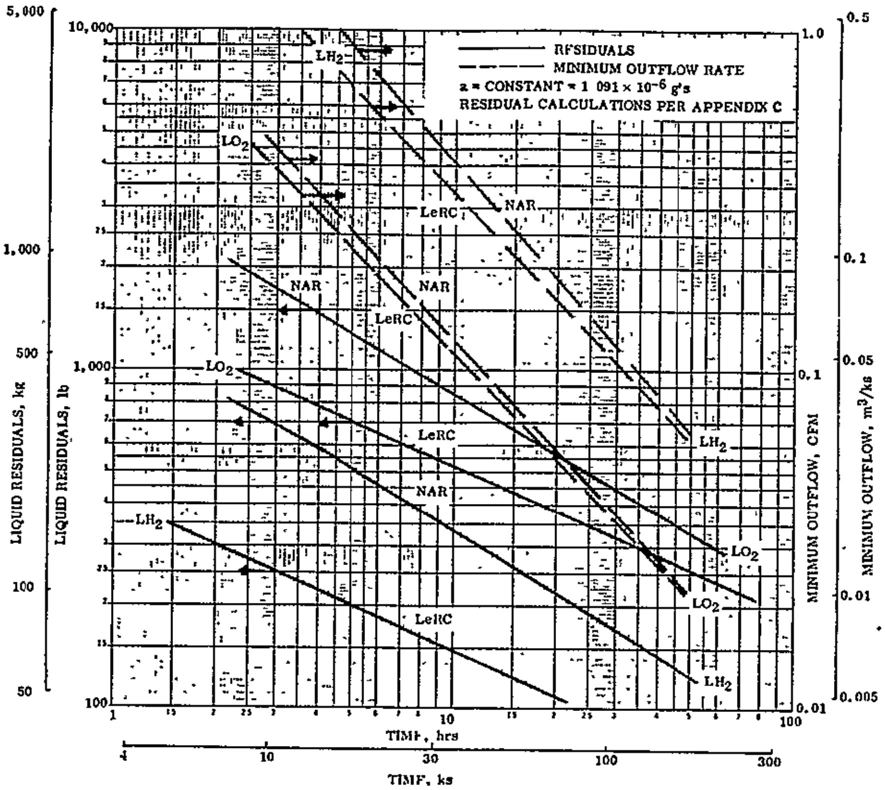

Figure 4-19. Liquid Residuals and Minimum Outflow Rates (Drag-Only 296 km, 160 n mi)

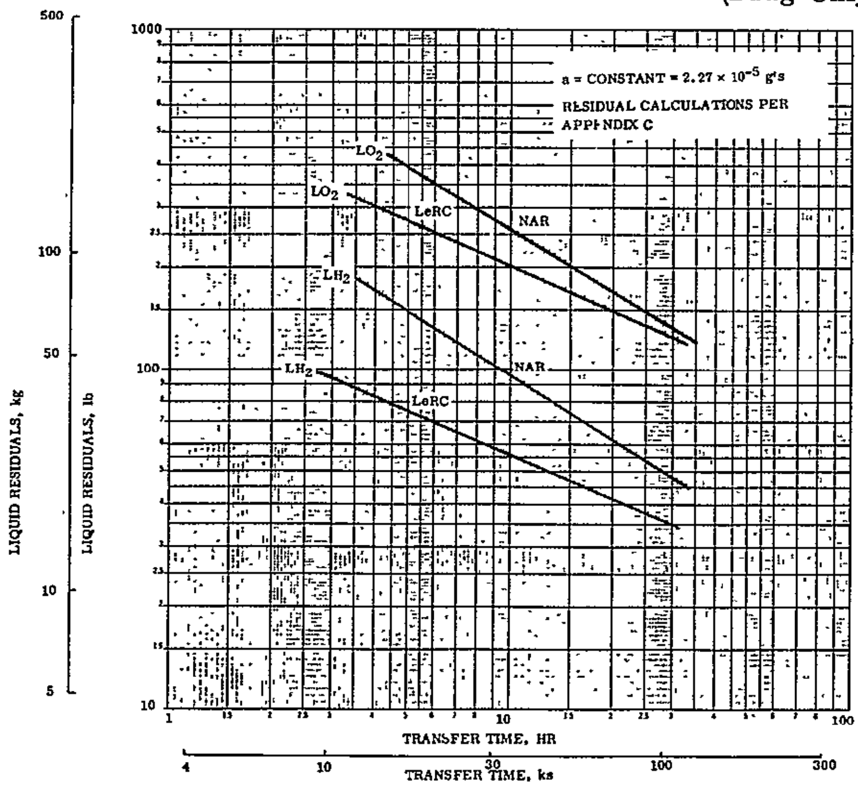

Figure 4-20. Liquid Residuals for Drag-Only at 185 km (100 n mi)

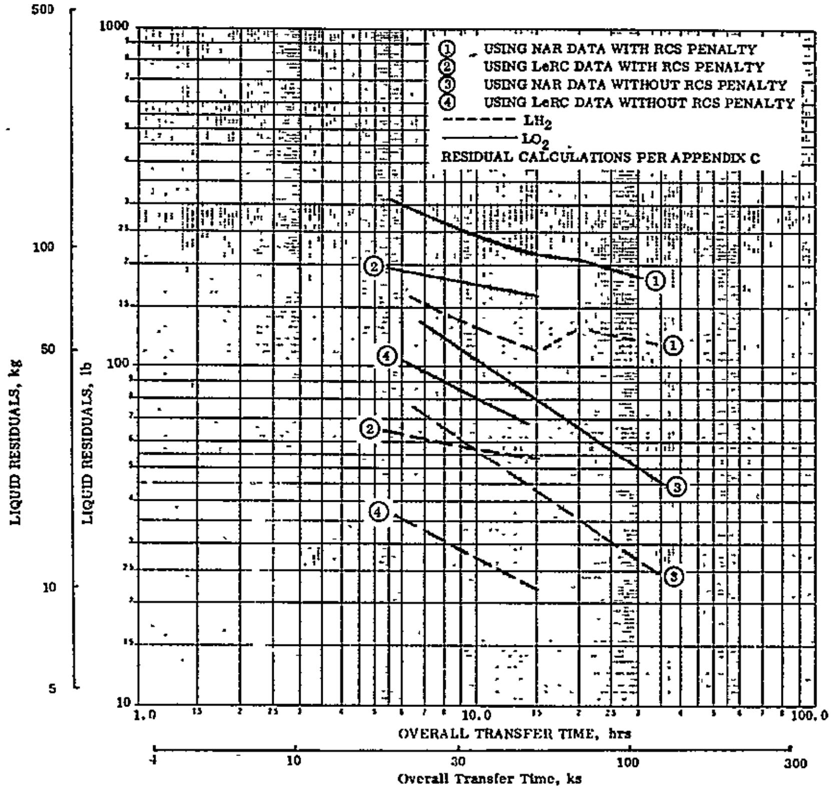

Figure 4-21. Liquid Residuals at 296 km (160 n mi) With RCS Scavenging

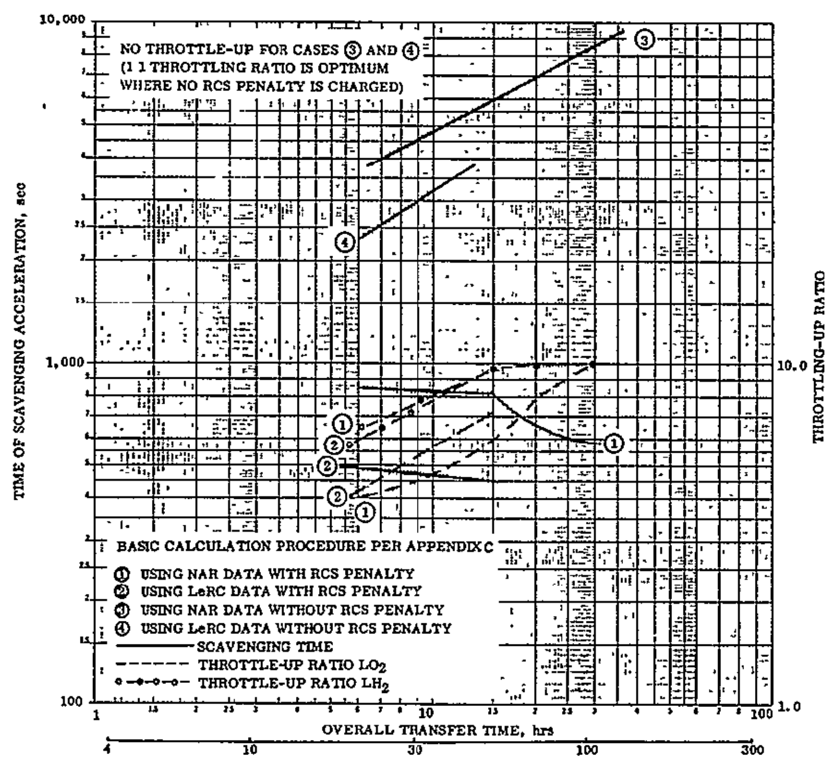

Figure 4-22. RCS Scavenging Time and Optimum Throttling-Up for Transfer at 296 km (160 n mi)

vernier engines are actuated. The optimum increase in flow at this time was found to be different for H_2 and O_2 and was not necessarily the maximum allowable. This is shown in Figure 4-22. The differences in flow-increase-ratios between H_2 and O_2 result from specifying that O_2 and H_2 pull-through occur at the same time. This minimizes total residuals.

4.2.2.3 Transfer at 185 km (100 n. mi.) versus 296 km (160 n. mi.) - From Figure 4-18, for drag-only transfer, the maximum weight saving by transferring at 185 km (100 n. mi.) rather than 296 km (160 n. mi.) is 163 kg (360 lb). Calculations where scavenging is employed show slightly less savings (maximum of 139 kg, 306 lb). Another advantage of transfer at the lower orbit is that transfer times can be shorter without compromising overall system weight. Data from Figure 4-23, however show a Tug payload penalty of 259 kg (570 lb) for a typical mission when transferring at the lower orbit.

Transfer at orbits lower than 296 km (160 n. mi.) would only be advantageous if the Shuttle could take the Tug to 296 km (160 n. mi.) following fill. The primary problem with this would be that due to space limitations the Tug would need to be carried outside the Shuttle between the two orbits. The possibility of doing this was analyzed for the configuration shown in Figure 4-24. It was found that the structural and payload carrying requirements were easily met. However, the existing Orbit Maneuvering Engines could not be gimballed sufficiently to accommodate the resultant shift in center of gravity of the overall Shuttle/Supply Module/Tug assembly. The maximum gimbal angle is now 0.07 rad (4°) while calculations show that 0.12 rad (6.8°) may be required. Further detailed work would be needed for a final resolution of this problem.

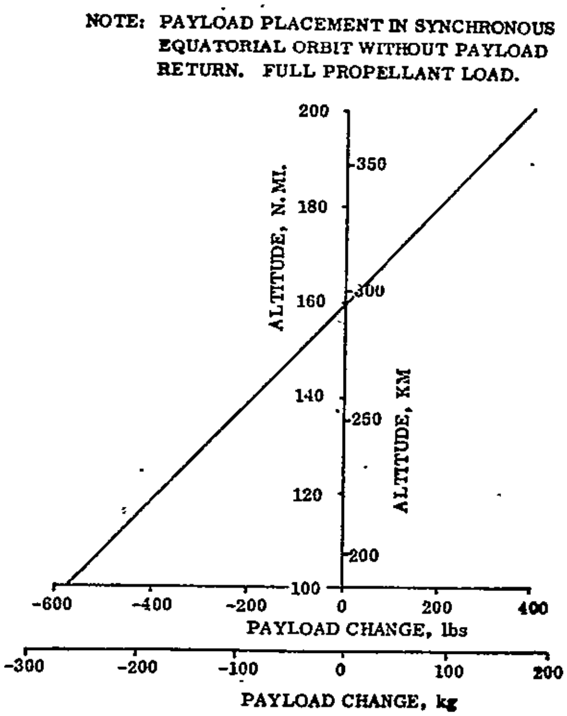

Figure 4-23. Change in Tug Payload Capability as a Function of Initial Altitude

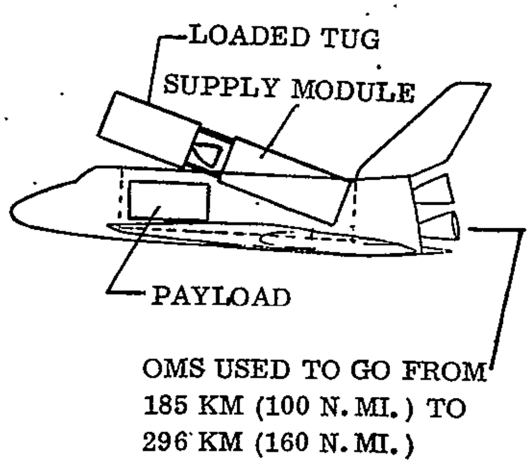

Figure 4-24. Tug Transported External to Shuttle Orbiter

FOLDOUT FRAME 1

FOLDOUT FRAME 2

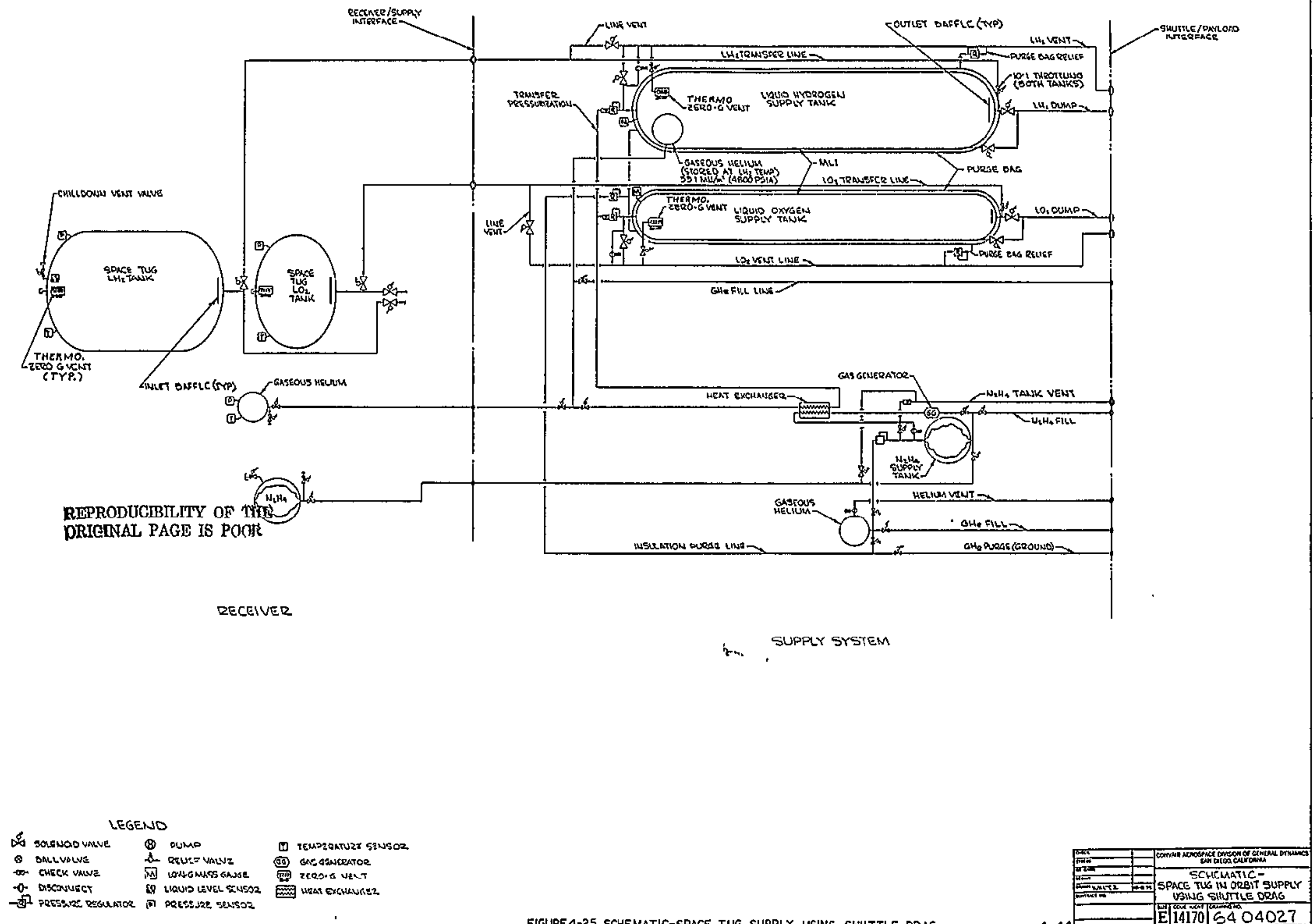

FIGURE 4-25. SCHEMATIC-SPACE TUG SUPPLY USING SHUTTLE DRAG

4.2.2.4 Drag System Final Conceptual Design Data - An overall schematic of the system and conceptual design drawings are presented respectively in Figures 4-25 and 4-26. System weights and a fluids inventory are presented respectively in Tables 4-25 and 4-26. Fluid quantities presented are based on supplying initially empty receivers. Additional hardware weight data are presented in Appendix F. Comparing the Table 4-25 and 4-12 data, it is seen that, even with the addition of tank pressurization, thermal control, plumbing and support structure, tankage optimization and use of the Orbiter RCS for scavenging results in a total weight reduction for the final design.

Table 4-25. Weights for Tug Supply Using Shuttle Drag

Item	Weight kg	Weight lb
LH_2 System (Dry)	472	1039
LO_2 System (Dry)	528	1164
N_2H_4 System (Dry)	30.4	67
Ambient GHe System (Dry)	36.8	81
Cold He Pressurization Transfer System (Dry)	103	226
Tug Attach Fittings & Legs	10.9	24
Swing-Out Ring	244	537
Overall Module Support Structure	175	386
Total Dry Weight	1600	3524
Total Fluids Table 4-21)	23821	52470
Lift-Off Weight w/o RCS Penalty	25421	55994
Potential RCS Penalty	154	340
Lift-Off Weight With RCS Penalty	25576	56334
Total Fluids Supplied (Ref. Table 4-21)	23401	51545
Total Fluid Residuals (Ref. Table 4-21)	205	452
Total Return Weight	1805	3976
Fluid Supplied/Lift-Off W/O RCS Penalty, %		92.1
Fluid Supplied/Lift-Off With RCS Penalty, %		91.5
Total Fluid Residuals, %		0.86

This system basically utilizes maximum Shuttle Orbiter drag at 296 km (160 n. mi.) to orient the LO_2 and LH_2 at tank outlets for transfer. Complete removal of the supply module from the Shuttle is not required. Hydrazine transfer is accomplished with a positive expulsion bladder. The hydrazine is stored in a common tank also used to supply a gas generator for heating of the LO_2 and LH_2 tank pressurant. Transfer of the N_2H_4 is accomplished following transfer of LO_2 and LH_2 without any need for liquid orientation. Initial hydrazine tank pressure is 2069 kN/m^2 (300 psia) for gas generator operation. This pressure is allowed to blowdown to 689 kN/m^2 (100 psia) during transfer of N_2H_4 to the Tug. Helium is transferred to the Tug from a high pressure 33096 kN/m^2 (4800 psia) bottle stored in the LH_2 tank which is also used for LO_2 and LH_2 supply tank pressurization. Helium transfer is

Table 4-26. Fluids Inventory for Tug Supply Using Shuttle Drag

Use	Fluid									
	LH_2		LO_2		Ambient GHe		Cold GHe		N_2H_4	
	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb
Required for Receiver Mission	3462.20	7626	19782.60	43574	-	-	4.09	9	152.54	336
Transfer Line and Receiver Chilldown	58.11	128	69.92	154	-	-	0	0	-	-
Receiver Boil-off During Transfer	6.36	14	9.08	20	-	-	-	-	-	-
Transfer Line Residuals	1.82	4	19.98	44	-	-	-	-	0.91	2
Total Transferred	3528.49	7772	19881.57	43792	-	-	4.09	9	153.45	338
Supply Residuals	58.57	129	94.87	209	0.05	0.1	3.18	7	3.18	7
Supply Vented Prior to Transfer	30.87	68	0	0	-	-	-	-	-	-
Fluid Used for Pressurant, Purge and Safing	-	-	-	-	2.68	5.9	42.68	94	-	-
Liquid for Gas Generator	-	-	-	-	-	-	-	-	17.71	39
Total Supply Module Load	3617.93	7969	19976.45	44001	2.72	6.0	49.94	110	174.34	384

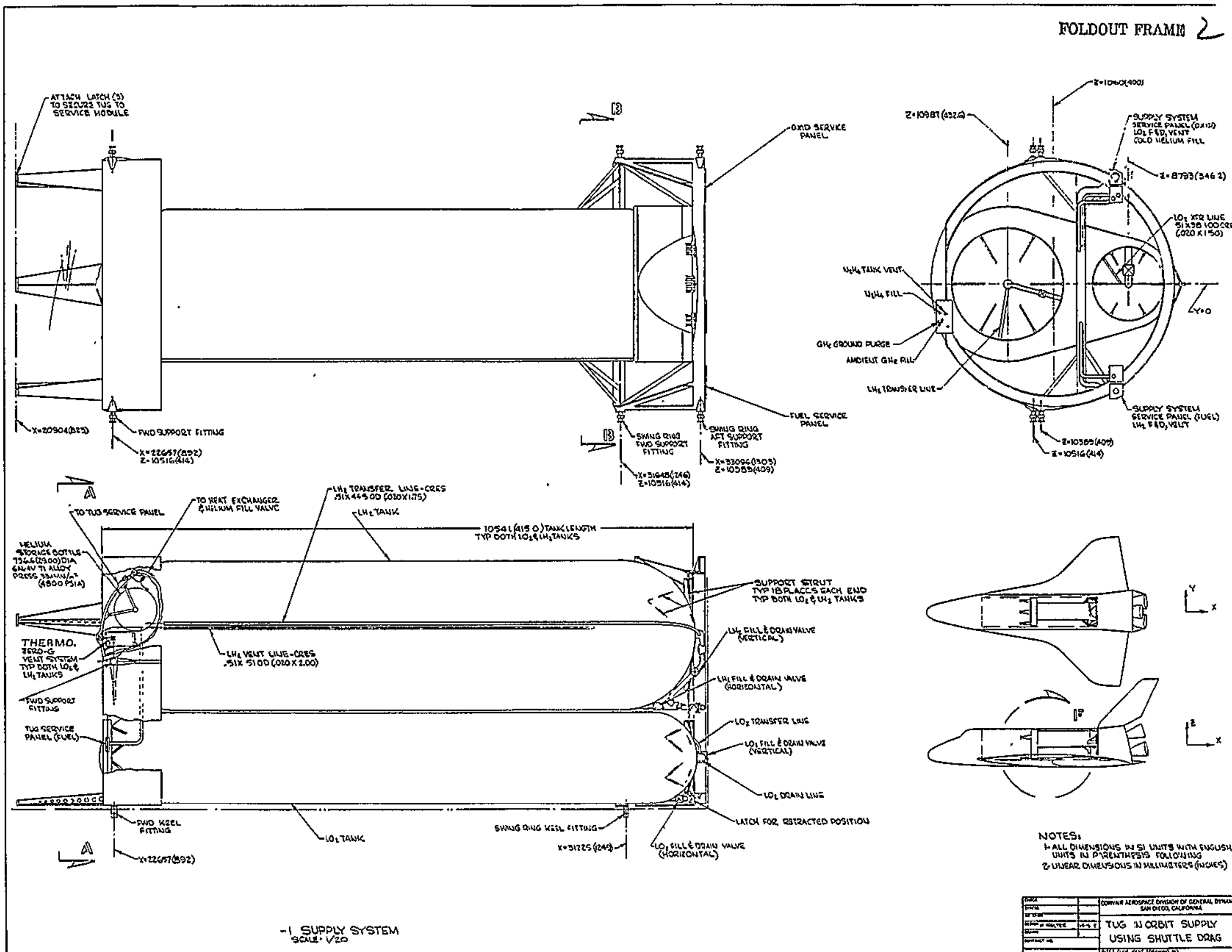

FIGURE 4-26 TUG SUPPLY USING SHUTTLE DRAG

FOLDOUT FRAME 3

FOLDOUT FRAME 4

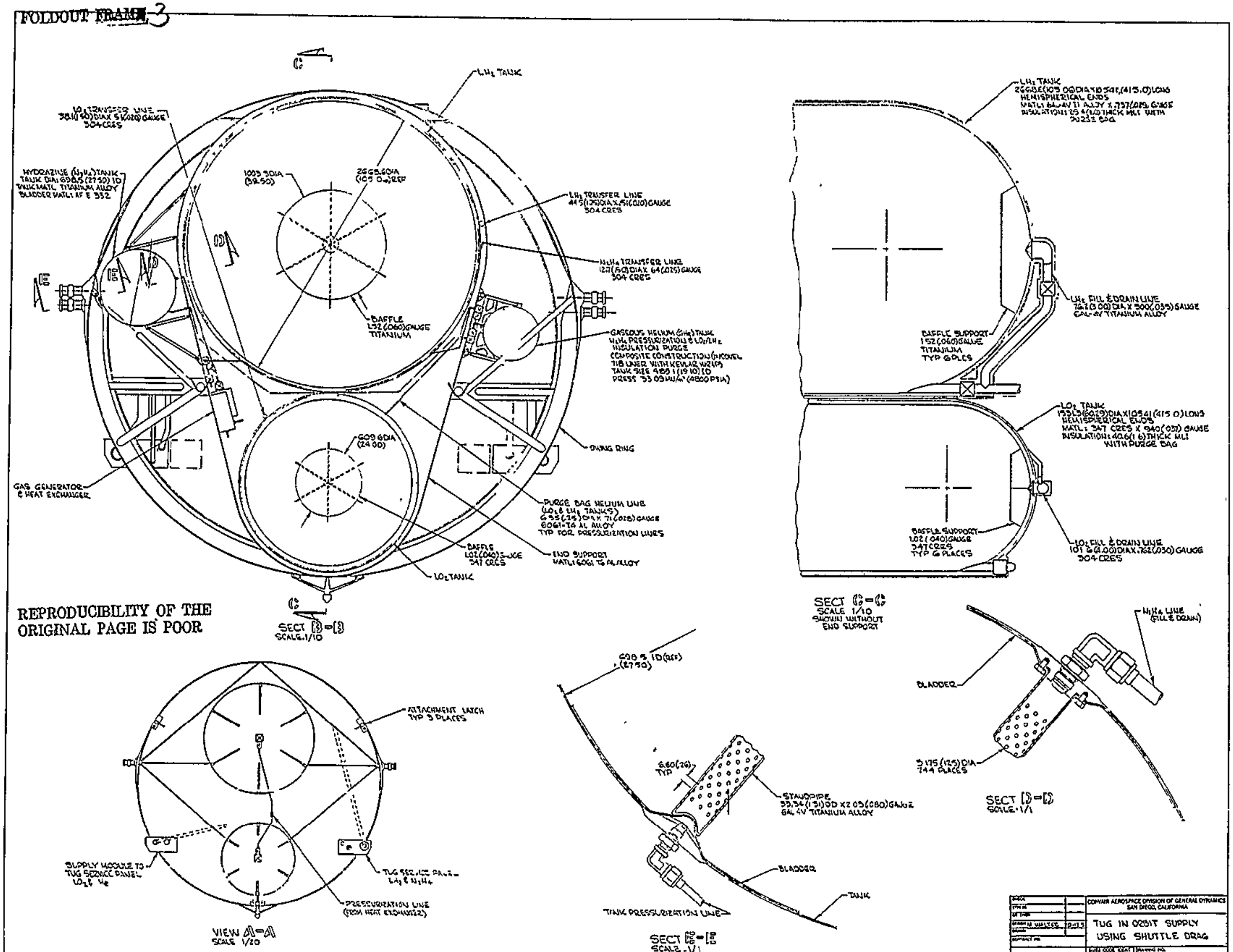

FIGURE 4-26. TUG SUPPLY USING SHUTTLE DRAG-CONT.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

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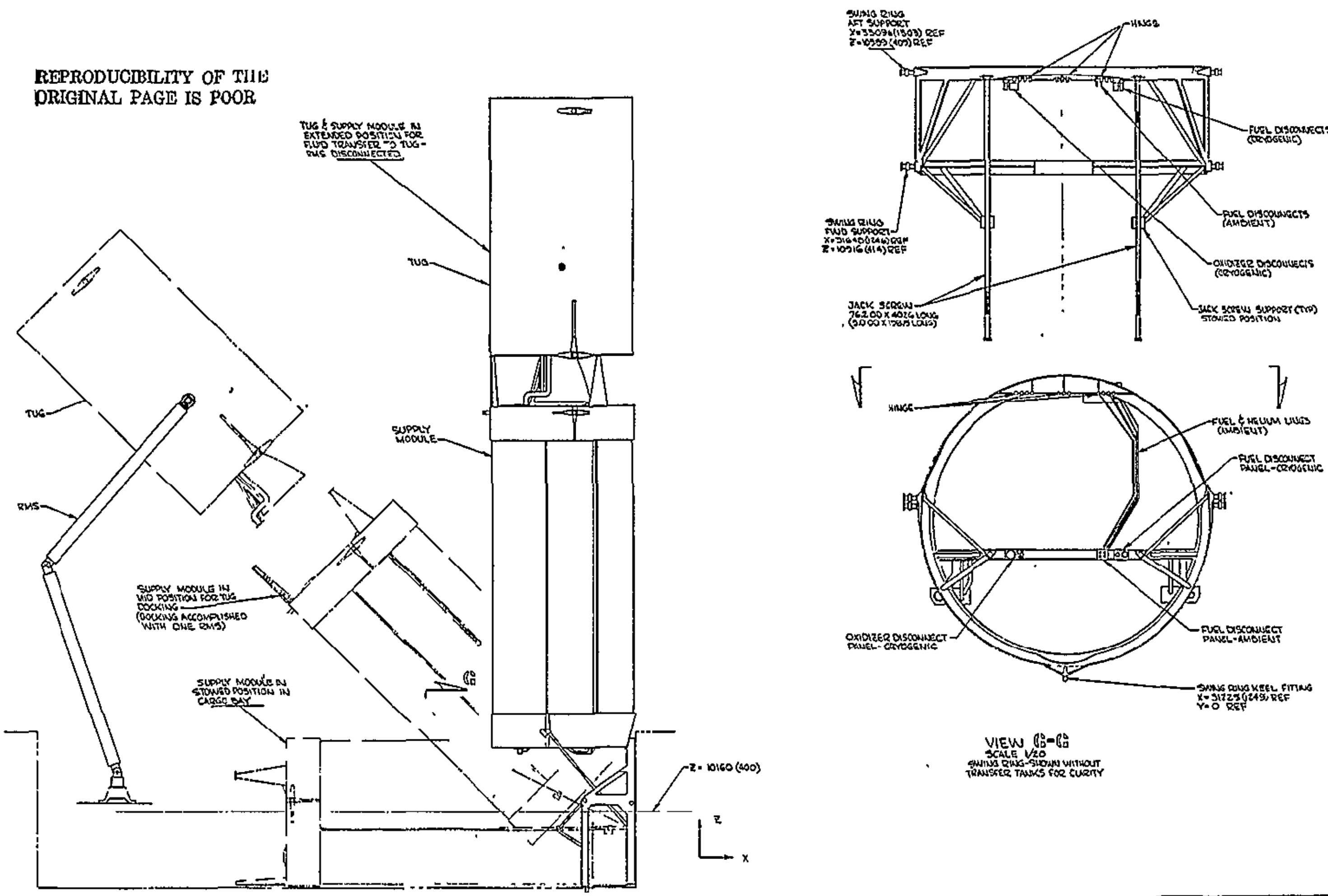


VIEW G-G
SCALE 1/20
SWING RING-SHOWN WITHOUT TRANSFER TANKS FOR CLARITY

DETAIL F
SUPPLY SYSTEM WITH TRANSFER TANKS IN STOWED, DOCKING & TRANSFER POSITIONS
SCALE 1/40

CONVAIR AEROSPACE DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

TUG IN ORBIT SUPPLY USING SHUTTLE DRAG

E 14170 64-04020

FIGURE 4-26. TUG SUPPLY USING SHUTTLE DRAG-CONT.

accomplished prior to the LO_2 and LH2 transfer.

The LH2 and LO_2 tanks are long cylinders, 2.7 m (105 in.) × 10.5 m (415 in.) and 1.5 m (61 in.) × 10.5 m (415 in.) respectively, with hemispherical ends employing outlet baffles to minimize liquid residuals. The use of long cylinders significantly reduces residuals, for a given pull through height, over that expected from spheres or large diameter tanks. Flow control valves are located at the tank outlets to throttle the liquid flow rate near the end of transfer to one-tenth of full-flow, to further reduce residuals. Tank lengths are limited by a requirement to reserve at least 6.1 m (20 ft) of the cargo bay for Tug payload.

The maximum drag acceleration is 1.1×10^{-6} g's. This results in Bond numbers of 0.66 and 0.54 for LH_2 and LO_2 respectively. Initial liquid settling and scavenging of residuals near the end of transfer are accomplished using the Shuttle Orbiter 11.4 kg (25 lb) thrust vernier engines. In the present case, a settling acceleration of 2.9×10^{-4} g's is provided by these engines. Firing times are 180 seconds for settling and 660 seconds for scavenging. The overall transfer time following settling is 72 ks (20 hours). This comes from Figure 4-18 where total weights are conservatively based on NAR residuals and including RCS propellant as a penalty for scavenging. Transfer is at 296 km (160 n.mi.). For this case, it is seen from Figure 4-18 that 108 ks (30 hr) is actually optimum, however, the weight penalty in going to 72 ks (20 hr) is very small. The shorter transfer time reduces required Orbiter time in orbit and is also the optimum time in case RCS propellants do not need to be charged to payload. Whether or not these propellants must be charged is still open to question. Outflow rates are throttled up by 8:1 for O_2 and 10.1 for LH_2 during scavenging.

Transfer of LO_2 and LH_2 is accomplished simultaneously with the supply module rotated 1.6 rad (90°) from its stowed position and docked to the Tug. The basic procedure is to (1) rotate the supply module to an intermediate position (approximately 0.79 rad (45°), (2) dock the Tug to the module using the single Shuttle Remote Manipulator System (RMS), (3) disengage the RMS from the Tug, since it will not reach over the full rotation, (4) rotate the Tug-supply module assembly to 1.6 rad (90°), and (5) settle and transfer.

Removal of the Tug after transfer is a reverse of the above. The supply module is rotated in until the RMS can be attached to the Tug and the Tug then released from the supply module.

Screw jacks are used to rotate and position the supply module. Power requirements to operate these jacks were found to be very small, i.e., 6W and 1872J (0.52 W-hr) for one complete rotation in 600 seconds.

The receiver oxygen tank is assumed to be locked-up during transfer, except that the thermodynamic vent system is used to maintain a nominal 107 kN/m^2 (15.5 psia) liquid saturation pressure to maximize the amount of liquid received. The Tug hydrogen tank is assumed to be vented during chilldown, and liquid inlet baffles are provided to minimize liquid at the vent. Following chilldown the tank is locked-up, except for the thermodynamic vent which will operate to maintain the required liquid vapor pressure for maximum loading.

Helium stored at 33096 kN/m^2 (4800 psia) and ambient temperature is used for pressurization of the N_2H_4 bladder tank and for purge pressurization of the LH_2 and LO_2 insulation systems during re-entry. LO_2 and LH_2 supply tank pressurization for transfer and abort dump uses helium stored at 33096 kN/m^2 (4800 psia) in the hydrogen tank and heated to 289K (520R) by a hydrazine gas generator system.

Both LO_2 and LH_2 supply tanks employ Superfloc multilayer insulation [2.5 cm (1.0 in.) for LH_2 and 4.1 cm (1.6 in.) for LO_2] enclosed by rigid purge bags to prevent moisture condensation and/or freezing during ground hold, boost and re-entry. LH_2 and LO_2 transfer lines are 4.4 × 0.05 cm (1.75 × 0.020 in.) and 3.8 × 0.05 cm (1.5 × 0.020 in.) CRES respectively, except at the supply tank outlets where diameters are 8.9 and 6.4 cm (3.5 and 2.5 in.) respectively.

The two long LO_2 and LH_2 tanks are placed side by side and attached to a common bulkhead at each end. Being of equal length allows easy attachment to the bulkheads. The tanks are attached to the bulkheads by fiberglass struts for minimum heat transfer between the tanks and bulkhead. The dump lines are arranged to allow draining the tanks in either the vertical or horizontal position. The aft bulkhead is designed to allow the inclusion of jackscrews and a swing ring. The aft bulkhead conforms to the shape of the two tanks to allow room for the jackscrews and bracing struts along each side. The basic construction of the bulkhead is a flanged rear plate, a forward flanged support and a connecting skirt with appropriate stringers and formers. The helium and the hydrazine tanks, and the heat exchanger are located on the skirt. The aft bulkhead is attached to the swing ring by an upper hinge and two lower jackscrews. Disconnects are provided in the fluid lines between the bulkhead and the swing ring to allow the bulkhead (and the supply tanks) to be rotated.

The design of the swing ring consists of an aft basic support ring and a forward stabilizing half ring connected by longitudinal beams and braces. Two jack screws and the fluid lines and disconnects are mounted on a cross beam attached to the main support ring. The main support ring has an H cross section, 203 × 203 mm (8 × 8 in.). The forward stabilizing ring has a T cross section, 203 × 152 mm (8 × 6 in.) and the cross beam has a channel cross section, 152 × 102 mm (6 × 4 in.). Electric power lines (for operation of valves, etc) do not disconnect when the supply tanks are rotated. These lines are routed along the main support ring and cross over to the supply tanks aft bulkhead over the hinge line. The necessary movement of the lines is provided by the flexibility of the wiring.

Cargo bay support fittings are provided on each side of both the main support ring and the stabilizing half ring. In addition, a fifth support, a keel fitting, is provided on the bottom of the stabilizing half ring. These five fittings provide support for the aft end of the supply module during ascent and descent of the Shuttle and support the entire module when the module is rotated out of the cargo bay.

A remotely actuated latching mechanism is provided on the bottom of the main support ring between the swing ring and the supply tanks aft bulkhead. This latch allows the supply tanks to be locked in place in the cargo bay during ascent and descent of the Shuttle. When the supply tanks are rotated out of the cargo bay, either fully or in an intermediate position, the jack screws are locked to hold the tanks in the required position.

The hydrogen tank material is titanium alloy with a skin gauge of 0.737 mm (0.029 in.). The oxygen tank material is 347 CRES with a skin gauge of 0.940 mm (0.037 in.).

Gaseous helium is stored in two spherical bottles, one inside the liquid hydrogen tank and one on the aft skirt. The bottle mounted inside the LH_2 tank is of conventional design titanium alloy. The gaseous helium bottle mounted on the aft skirt is of a composite construction with an Inconel 718 liner with a Kevlar overwrap.

The hydrazine tank is titanium with a skin gauge of 2.38 mm (0.094 in.). It has an internal bladder (material AF-E-332) and a standpipe for zero-g operation.

The gas generator and heat exchanger (used for heating the helium used for tank pressurization) are of conventional design.

Three struts are attached to the forward bulkhead to allow the supply module to attach to the Space Tug. These struts are 1549 mm (61 in.) long to allow clearance for the Space Tug engine. Fuel and oxidizer service panels, similar to those in the Shuttle cargo bay, are mounted on the supply module forward bulkhead so that the Tug can be connected to the supply module in the same manner as it is connected in the Shuttle cargo bay.

4.2.2.5 Drag System Operating Procedure - A functional flow diagram of the overall supply operation is presented in Figure 4-27. The various operations outlined in Figure 4-19 are expanded and discussed in the following paragraphs. A schematic of the complete transfer system is presented in Figure 4-25.

Supply Module Ground Preparation - Leak checks and purging and loading of N_2H_4 into the bladder tank are performed. LO_2 and LH_2 tank purging is accomplished to insure that gases and/or liquids which are condensible and/or which may solidify at cryogenic temperatures are eliminated.

Load Supply Module Into Shuttle - This simply consists of the installation of the supply module into the Shuttle Orbiter cargo bay and the hookup of all ground supply, abort dump and in-orbit transfer lines.

Load and Maintain Cryogenics - At this point the cryogens, LO_2 and LH_2, are loaded into the supply module. This operation consists of initial line and tank chilldown and final loading. It is assumed that the maximum fluid loading is maintained by topping

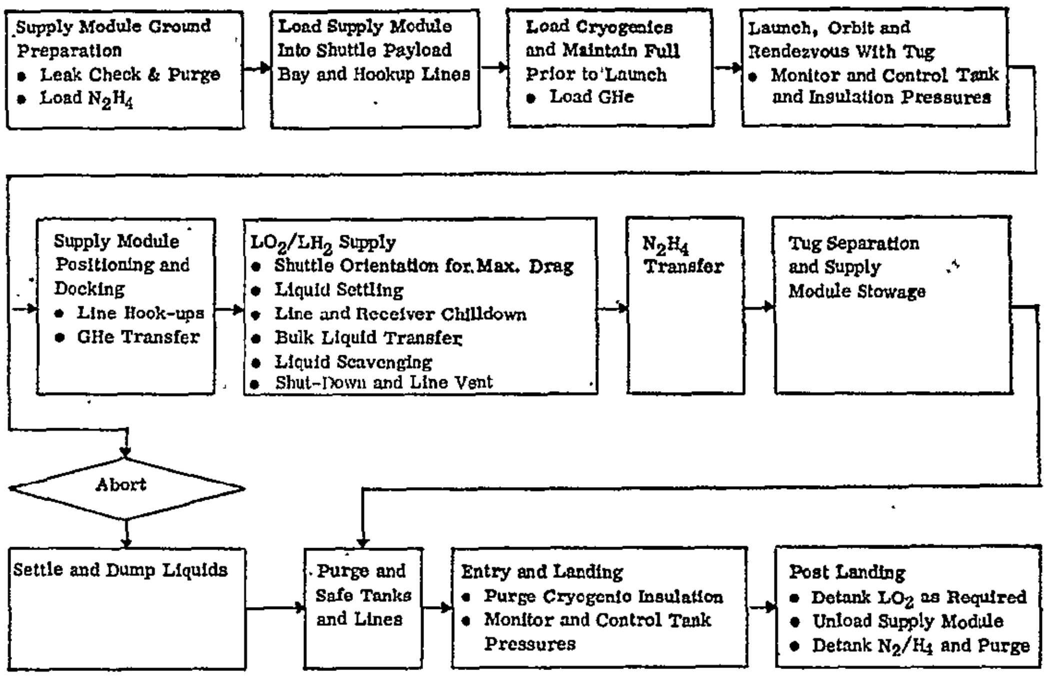


Figure 4-27. Overall Operations for Tug In-Orbit Supply Using Shuttle Drag

up to 180 seconds prior to launch. The high pressure helium storage bottles are also filled at this time. The cryogenic insulation for the LO_2 and LH_2 tanks is maintained in a purged condition with GHe from the ground prior to boost.

Launch, Orbit and Rendezvous - During this phase of the supply operation the main problems are to control tank and insulation pressures. Venting of the LH_2 and LO_2 tanks is allowed for during the boost phase through the normal ground boiloff vents, although under nominal heating conditions O_2 tank venting should not occur. Following entry into orbit, the thermodynamic vent systems are designed to accommodate any venting required. Tank overpressure is a safety critical failure and provisions are made to allow the immediate dumping of liquids as well as emergency venting.

Abort (Only as Necessary) - If for some reason it becomes necessary to terminate the mission during boost or while in orbit, all liquids are dumped. Liquid settling to the aft end of the Shuttle is provided by the normal Shuttle abort procedure. Redundant valving is provided to insure that abort dumping can be accomplished.

Supply Module Positioning and Docking - Following rendezvous of the Tug and Shuttle the following sequence of events are accomplished in preparation for actual LO_2/LH_2 supply.

a. Attachment to the Tug of the Shuttle Remote Manipulation System (RMS).

b. Partial 0.79 rad (45°) rotation of supply module out of payload bay (Ref. Figure 4-18). Disconnection of fluid services between Shuttle and supply module is accomplished remotely at rotation initiation. Electrical services remain connected.

c. Docking of Tug to supply module using the RMS. Transfer line hook-ups are accomplished remotely on docking or immediately thereafter.

d. Disengagement of RMS and complete [1.57 rad (90°)] rotation of tug-supply-module assembly. At this point the assembly is locked into position for LO_2/LH_2 fluid transfer. Prior to LO_2/LH_2 transfer the high pressure [22750 kN/m^2 (3300 psia)] Tug helium bottle is supplied from the 33096 kN/m^2 (4800 psia) supply module pressurant bottle stored in the LH_2 tank. Since the helium being supplied is cold, the Tug helium bottle fluid temperature and pressure must be closely monitored to determine when sufficient mass has been transferred.

LO_2/LH_2 Supply - Following orientation of the Shuttle for maximum drag, liquid settling is accomplished by firing the Orbiter 11.4 kg (25 lb) vernier engines for 180 seconds. This time, by itself, is not sufficient for complete settling, however, it is expected that drag will be sufficient to complete the settling. Further technology work needs to be accomplished in this area. Following settling, the LO_2 and LH_2 are transferred simultaneously through the following series of events.

a. LH_2

(1) Activate supply tank pressurization to 138 kN/m^2 (20 psia).

(2) Open receiver vent. A normal ground type vent will be used here, possibly in connection with a liquid/vapor separator. Further technology work is needed to determine the need for and/or configuration of such a system.

(3) Open receiver tank inlet valves and supply tank low-flow valve and allow line and receiver tank to chill down to LH_2 temperature. Inlet baffles are provided to minimize liquid at the vent.

(4) Close receiver tank vent, except for the thermodynamic vent, and increase transfer flow to the maximum allowed. The receiver thermodynamic vent is used to maintain the liquid saturation pressure at near 110 kN/m^2 (16 psia) for maximum loading density.

(5) Reduce flow rate by 10:1 to prevent vapor pull-through as tank nears depletion.

(6) Increase outflow rate by 10:1 and activate vernier engines for 660 seconds to minimize liquid residuals.

(7) Close receiver tank inlet valves and supply tank outlet valves and vent the transfer lines to vacuum.

b. LO_2

(1) Actuate supply tank pressurization to 173 kN/m^2 (25 psia).

(2,3,4) Open receiver tank inlet valves and supply tank outlet valves to maximum flow conditions. The Tug O_2 tank is assumed to be locked-up during both chill-down and transfer, except that the receiver zero-g heat exchanger vent system is used to maintain the liquid saturation pressure at near 110 kN/m^2 (16 psia) for maximum loading density.

(5) Same as for LH_2.

(6) Same as for LH_2 except outflow throttled up by 8:1 rather than 10:1.

(7) Same as LH_2.

The overall transfer time is approximately 72 ks (20 hrs).

N_2H_4 Transfer - Following transfer of the LO_2 and LH_2 , N_2H_4 from the bladder supply tank is transferred to the Tug bladder tank. During this transfer the supply tank pressure of 2070 kN/m^2 (300 psia) used to supply the gas generator is allowed to blowdown to a final pressure of 690 kN/m^2 (100 psia).

Tug Separation and Supply Module Stowage - This operation is basically a reversal of the supply module positioning and docking procedure described above.

Purging and Safing - Prior to entry and landing, LO_2 and LH_2 residuals are completely dumped and the tanks purged with helium This is especially critical in the case of H_2 where venting cannot be accomplished immediately upon landing, as would be required due to entry heating if LH_2 were remaining.

Entry and Landing - The primary operations here are to (1) control pressure in the insulation purge bags to slightly above ambient to prevent entry of air with subsequent moisture condensation and insulation damage, and (2) maintain tank pressures above ambient, to prevent implosion, and below the design pressure, to prevent bursting.

Post Landing - Following landing of the Shuttle Orbiter, any remaining LO_2 is detanked. The complete supply module is then unloaded from the Shuttle Orbiter and any remaining N_2H_4 drained. All tanks are then purged and locked-up with inert gas and the complete system stored.

4.2.3 SHUTTLE ORBITER IN-ORBIT SUPPLY - In the Orbiter supply system, liquid acquisition is with surface tension screens.

In order to determine reasonable transfer line lengths, an investigation was made to determine a reasonable Shuttle to Shuttle docking configuration. The configuration chosen is illustrated in Figure 4-28. The choice is based on data from Reference 4-14. Based on this docking configuration, transfer lines to be used in the following studies were determined to have the following characteristics.

N_2O_4 - 53.1 m (174 ft) long with eleven 90° bends
MMH - 53.1 m (174 ft) long with eleven 90° bends
He - 53.1 m (174 ft) long with eleven 90° bends
O_2 - 36.9 m (121 ft) long with eleven 90° bends
H_2 - 36.9 m (121 ft) long with eleven 90° bends

This assumes the supply module is located in the aft portion of the payload bay to meet the c. g. requirements for both boost and landing.

Two basic transfer cases are considered here. Case 1 assumes all OMS fluids (N_2O_4, MMH, He) are supplied without any O_2 and H_2 supply. Case 2 assumes all EPS/ ECLSS cryogenic O_2 and H_2 are supplied with OMS fluids (N_2O_4 and MMH) off-loaded as required to meet the 29,510 kg (65,000 lb) Shuttle payload limitation.

Initial tank packaging trade-offs resulted in the selection of spherical supply tankage for all fluids. This represents minimum weight when compared to cylindrical tanks of various shapes and orientations (lengthwise and crosswise). Weight was determined to be critical to meet Case 1 conditions and spherical tanks are compatible with surface tension liquid acquisition. Use of such tankage also allows the Shuttle landing c. g. requirements to be met with full tanks, in case of an emergency abort where the fluids are not able to be dumped.

Weight and design studies also showed that a minimum weight system occurs when the cryogenic supply tanks are not carried for the Case 1 supply. The basic supply module is designed to accommodate all tankage as necessary. This concept results in reasonably simple design and allows the 29,510 kg (65,000 lb) supply module weight limitations to be met for Case 1.

Further overall systems analyses and conceptual design are presented in the following paragraphs.

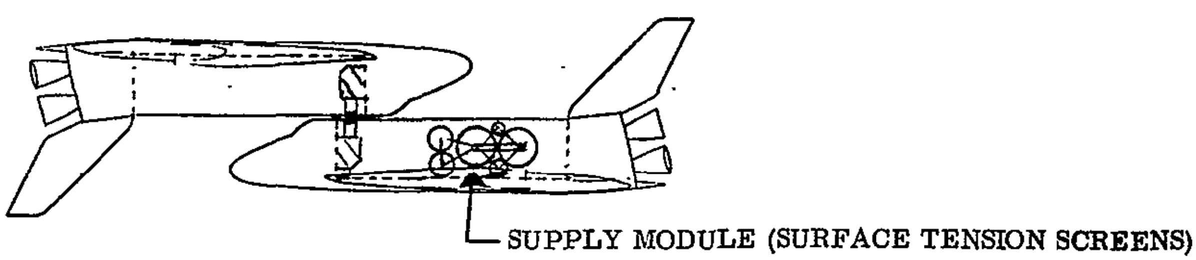


Figure 4-28. Shuttle Orbiter Supply Docking Configuration

4.2.3.1 Pressure Control of Cryogenic Receivers - The initial trade-offs performed were concerned with non-vent versus vent requirements during receiver chilldown and fill. Even though the O_2 and H_2 receivers are for supercritical storage and the He receivers for high pressure storage, it is assumed that in each case saturated liquid is supplied. Following supply the transferred fluid is heated or allowed to warm up to its final use condition.

To determine the venting requirements, if any, a balance is made between the energy removed from the receiver and that absorbed by the incoming liquid.

The receivers are assumed to be initially empty. Initial wall temperatures, allowable pressures and volumes are taken from Table 4-1. The basic energy equation used is presented below.

$$m_f \,(u_f - h_{in}) = E_w \qquad (4\text{-}8)$$

where

m_f = final mass of fluid in receiver

u_f = final specific internal energy of fluid in receiver

h_{in} = specific enthalpy of fluid entering the receiver.

E_w = energy removed from wall between initial and final conditions. Final wall temperature is assumed equal to final fluid temperature. E_w is determined from

$$m_w \int_{T_i}^{T_f} C_{p_w} \, dT$$

where

$$\int_{T_i}^{T_f} C_{p_w} \, dT$$

is the integrated specific heat of the wall material between T_i and T_f. Values are obtained from graphical integration of specific heat versus temperature curves.

Based on the NAR Space Shuttle design, receiver hardware involved in chilldown was taken to consist of the following.

Supercritical O_2 Tank

42.4 kg (93.3 lb) Inconel 718 Inner Tank
9.3 kg (20.4 lb) CRES Heater
2.5 kg (5.6 lb) Al Aly Transfer Line

Supercritical H_2 Tank

33.5 kg (73.8 lb) Al Aly Inner Tank
5.5 kg (12.2 lb) CRES Heater
2.5 kg (5.6 lb) Al Aly Transfer Line

High Pressure Helium Tank

50.8 kg (112 lb) Ti-6 Al-4V Liner
82.6 kg (182 lb) Kevlar-Epoxy Wrap

Calculations showed that the helium receivers were the only ones which could be filled to the desired capacity without venting. In this case the final pressure at the end of transfer was calculated to be 24800 kN/m^2 (3600 psia) at an equilibrium fluid/wall temperature of 236K (425R). A pump power of 42 MJ (11.7 kw-hr), which must also be absorbed by the transfer fluid, was included in the energy balance.

Since a completely non-vent chilldown and fill was not feasible for the H_2 and O_2 cases, calculations were made to determine the feasibility of accomplishing venting only during initial chilldown, followed by final chill and fill with the tank locked-up. Total energy absorption from the receivers then occurs by two modes; (1) vaporization and venting of incoming liquid, and (2) increase in fluid energy with increase in pressure of the tank fluid without venting.

The greater the energy which can be absorbed by the tank fluid without venting, the lower the quantity of fluid which must be vented. The energy absorption capacities of the H_2 and O_2, without venting, were determined as a function of final tank pressure. Data for single bottles are presented in Figures 4-29 and 4-30. Curves are shown

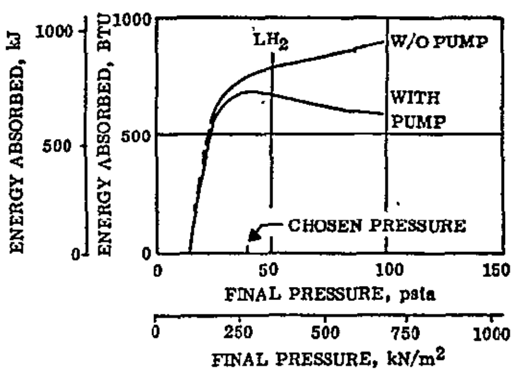


Figure 4-29. LH_2 Inlet Energy Absorption

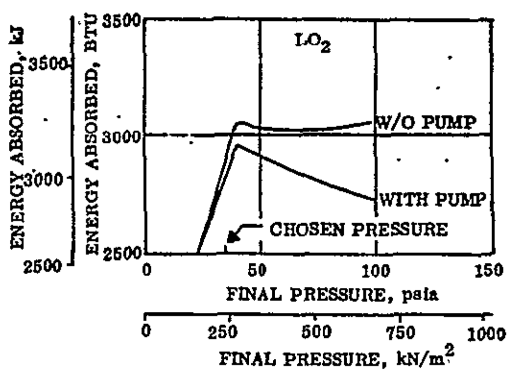


Figure 4-30. LO_2 Inlet Energy Absorption

with and without the energy required to pump the supply fluid from 138 kN/m^2 (20 psia) to the final locked-up pressure. The pump energy contribution is based on theoretical fluid pumping with the energy due to the various pump inefficiencies assumed to not enter the transferred fluid.

From Figures 4-29 and 4-30, it is seen that final pressures above 276 kN/m^2 (40 psia) are generally undesirable.

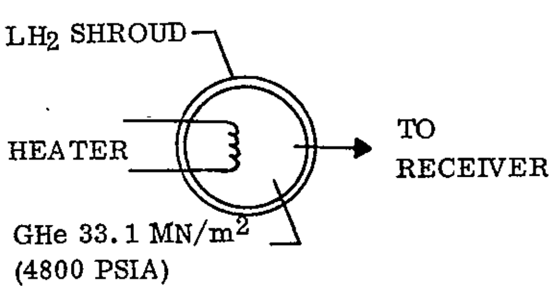
Weight trade-offs were made to determine final pressures and whether transfer should be provided by supply tank pressurization only or should include use of a pump. In the hydrogen case use of a pump represented the lowest weight system. For O_2 there was an insignificant difference and an all pressurant system was chosen for simplicity and safety.

In the final analysis, the optimum systems were determined to be where (1) the H_2 receiver is chilled to 94.4K (170R) by venting, at which time the tank is locked-up and filled to 276 kN/m^2 (40 psia), and (2) the O_2 receiver is chilled to 250K (450R) by venting, followed by lock-up and filling to a final pressure of 241 kN/m^2 (35 psia).

As discussed previously, helium transfer may be accomplished without venting. However, a high pressure 24,800 kN/m^2 (3600 psia) pump is required. To possibly eliminate the need for such pumping other methods of helium transfer were considered. These are compared with the above system in Table 4-27 on the basis of weight and power requirements.

From Table 4-27, the only reasonable alternate to the liquid storage system is cold storage at H_2 temperature and high pressure with supply bottle heating, as shown below. This system is further considered in Paragraph 4.2.3.3 for integration with pressurization of the other supply tankage. The LHe transfer with venting is only reasonable if some means could be developed to insure maximum efficiency venting during chilldown.

4.2.3.2 Vapor Return Supply Pressurization - It was stated in Reference 4-4 that significant weight savings could be achieved by circulating receiver vapor back to the supply. Variable system weights with and without vapor return were determined for the current N_2O_4 transfer. The vapor or gas return system is shown in Figure 4-31.



Transfer and return line, line residual, compressor and helium system weights are presented in Figure 4-32 for different transfer times as a function of supply pressure. For each time, an optimum transfer pressure was found. Weights at these optimum pressures are plotted in Figure 4-33 as a function of transfer time.

Table 4-27. Weights of Helium Transfer Concepts for Orbiter Supply

Concept	Power		Weight[1]	
	MJ	kw-hr	kg	lb
High pressure blowdown from ambient supply into receiver.[2]	-	-	2048	4512
Same as above, except helium residuals used to pressurize other supply tanks.			1927	4245
LHe transfer without receiver venting.	42	11.7	89.9	198
LHe transfer with receiver venting.	-	-	220[3]	484[3]
GHe transfer with compressor.	245	68	636	1400
Storage in LH_2 shroud[4] at 33070 kN/m^2 (4800 psia) with bottle heated to maintain pressure for transfer.	119	33	218	481
Storage in LN_2 shroud with bottle heated to maintain pressure for transfer	-	-	445	980

1. Weight includes supply tankage, insulation, zero-g vent, residual liquid, surface tension screens, pump and helium pressurization systems as applicable. A weight penalty for power requirements above 180 MJ (50 kw-hr) is also included.
2. Assumes maximum allowable initial supply pressure of 82.7 mN/m^2 (12,000 psi).
3. Assumes maximum venting efficiency (Ref. Para. 4.1). With minimum vent efficiency,weight would be approximately 4990 Kg (11,000 lb)
4. Storage is not in the LH_2 supply tank since this tank is not always carried (supply Case 1).

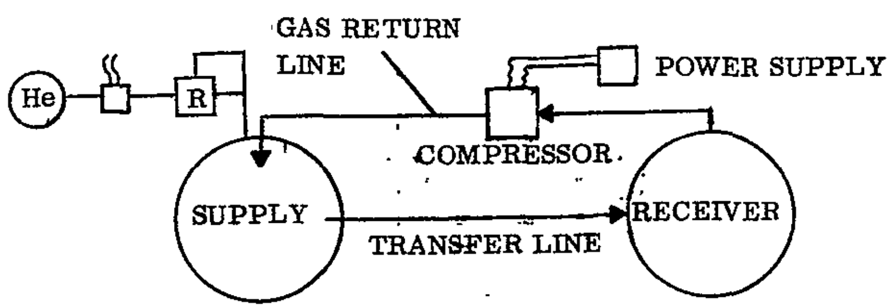


Figure 4-31. Vapor Return Supply System

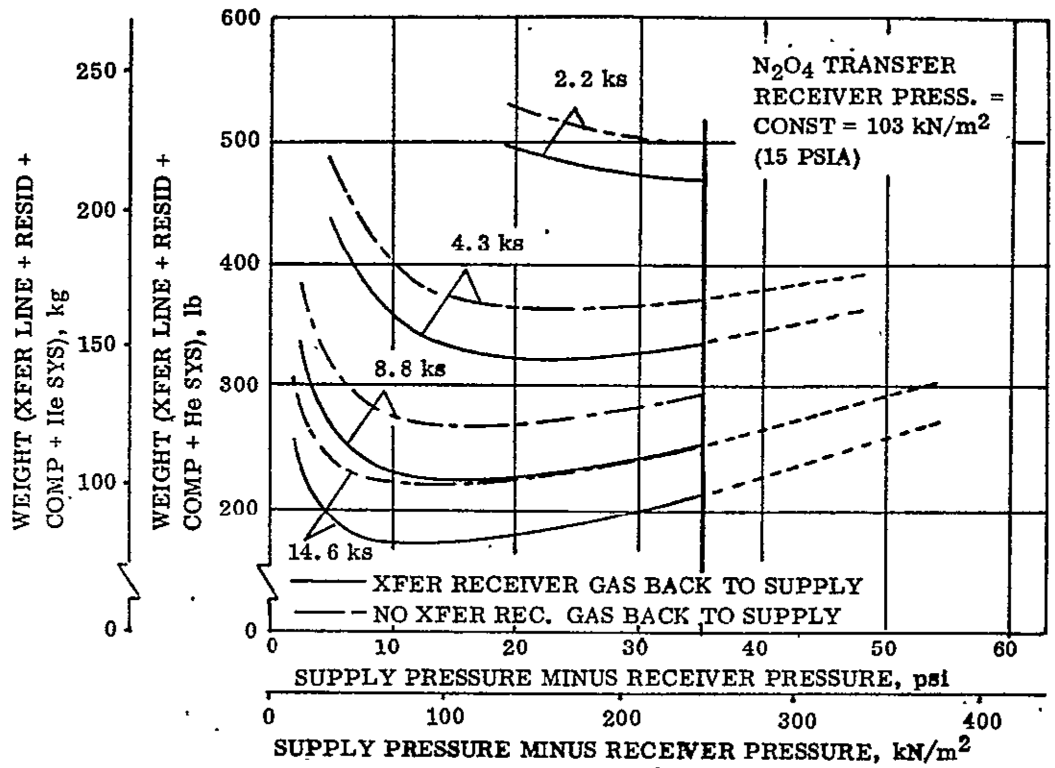


Figure 4-32. Transfer Weights vs Transfer Pressure

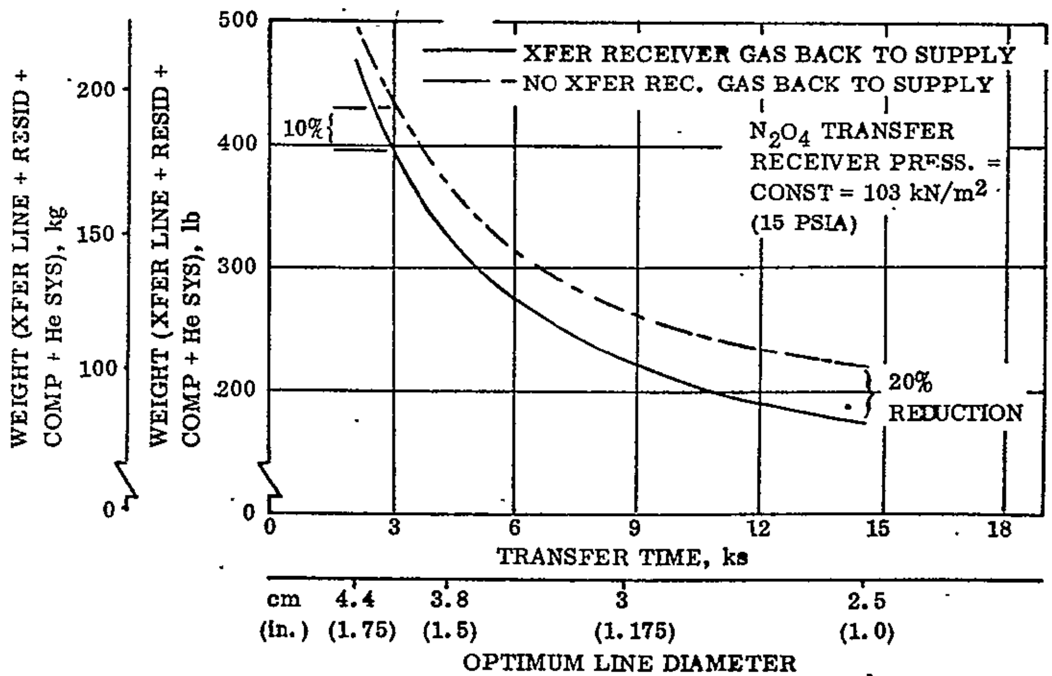


Figure 4-33. Transfer Weights vs Transfer Time

Depending on transfer time, the weight saving for a return system over a non-return system is only 10 to 20%.

This was not considered to be worth the added complexity of the compressor and additional requirements for liquid/vapor separation at the receiver vent. Also, the receiver may be empty of helium in certain cases where in-orbit supply is required.

Calculations were also made for the O_2 and H_2 cases. Since chilldown vent losses can be considerable it would first appear that the vapor so generated could be used to eliminate completely the need for any additional supply tank pressurization. In reality significant chilldown vapor generation only occurs at the start of transfer with only a relatively small volume of supply liquid used; i.e., the vapor generated during chilldown is not needed by the supply at the time generated. Even if this vapor could somehow be stored the total generated by chilldown is far in excess of that required for supply tank pressurization. Thus, very little saving in chilldown fluid loss could be realized.

4.2.3.3 Supply Tank Pressurization and Helium Transfer - This section discusses the trade-offs and analyses accomplished to define optimum supply tank pressurization systems. Consideration was also given to integration of the helium transfer system with supply tankage pressurization. Pressurant requirements and storage bottle weights are calculated using the methods presented in Appendix E.

Weight comparisons of the most promising concepts are presented in Table 4-28. The two basic systems considered are illustrated in Figure 4-34.

Data are presented for Case 1 (LH_2 and LO_2 supply tanks and associated pressurization not carried) and Case 2 (all tanks carried with N_2O_4, and MMH partially off-loaded).

Systems 1, 3, 4, and 7 assume that the surface tension screens will operate satisfactorily with warm pressurant. At the current time this is open to some question. Therefore, even though Systems 3 and 4 are slightly less weight, System 2 was chosen for detailed design. This system is also the simplest, requiring only pressurant heating for the helium transfer. Systems 5 and 6 meet the cold pressurant criteria, but are considerably heavier than System 2. Also, the use of Systems 6 and 7 would represent a problem for abort dumping, since heating would be a slow process and use of very cold helium would not provide sufficient expulsion pressure.

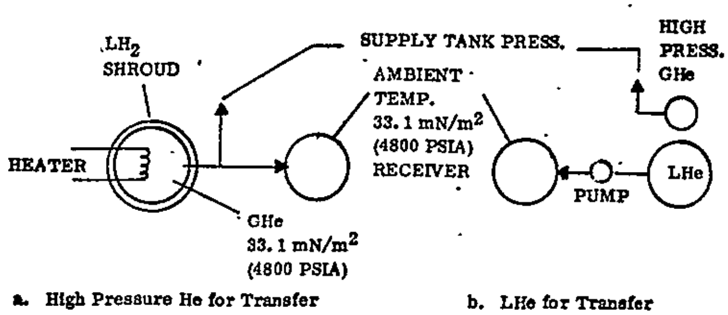


Figure 4-34. Tank Pressurization/Helium Transfer Concepts

4.2.3.4 Surface Tension Screens - Initial weight

Table 4-28. Tank Pressurization/Helium Supply Weight Comparisons

System	Case 1 Weight[1]		Case 2 Weight[1]	
	kg	lb	kg	lb
1. Helium for transfer stored as liquid in spherical tank containing surface tension screens. Helium at ambient temperature used to pressurize all supply tanks. He pressurant stored at ambient temp. and 33.1 mN/m² (4800 psia).	136	299	221	487
2. Helium for transfer stored as liquid. Pressurization of N_2O_4, MMH, LH_2 and LO_2 tanks using helium at the respective liquid temperatures. Helium pressurant storage also at liquid temperatures. LHe tank pressurization by external evaporation of helium stored as part of the LHe supply. Ambient He stored at 33.1 mN/m² (4800 psia) and cold He stored at 22.7 mN/m² (3300 psia).	151	333	205	452
3. Same as 2., except GH_2 stored as liquid and evaporated and heated to 167K (300R) used for LH_2 tank pressurization.	151	333	185	386
4. Same as 1., except helium pressurant stored at LH_2 temperature at 22.7 nM/m² (3300 psia) and heated to ambient temperature. A separate LH_2 pressurant storage shroud is required for the N_2O_4 and MMH pressurization system since the LH_2 tank is not always carried.	101	223	137	301
5. Helium for transfer stored in a LH_2 shroud at 33.1 mN/m² (4800 psia) and heated to maintain pressure for transfer. Pressurization for N_2O_4, MMH, LH_2 and LO_2 supply same as in 2.	299	658	353	778
6. Helium stored at 33.1 mN/m² (4800 psia) in LH_2 shroud and heated for transfer. Pressurization of other supply tankage is accomplished from this common bottle at corresponding liquid temperatures as the supply is warmed up during transfer.	301	663	301	663
7. Same as 6., except supply tank pressurization accomplished at ambient temperature using He supply bottle residuals.	234	515	234	515

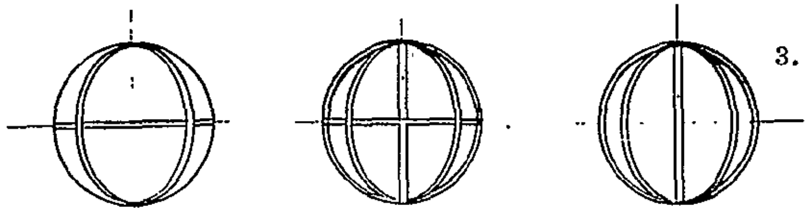
comparisons were made between simple single and double screen liners, single liner with center standpipe and pump, and channels. For the fairly large tanks of the current study, the channel system was found to have the lowest weight and was therefore chosen for detailed design.

Weight comparisons were then made between the three channel configurations listed below and illustrated in Figure 4-35.

1. Four vertical channels 1.57 rad (90°) apart running close to the tank wall between an upper and lower manifold with a horizontal channel connecting all vertical channels at the midsection of the tank.

2. Same as above, except with eight vertical channels 0.78 rad (45°) apart.

3. Same eight vertical channels as number 2 above but without the horizontal connecting channel.



a. 4 VERTICAL CHANNELS AND 1 EQUATOR RING b. 8 VERTICAL CHANNELS AND 1 EQUATOR RING c. 8 VERTICAL CHANNELS ONLY

Figure 4-35. Screen Channel Systems Considered

Pool residuals were computed by determining the largest spherical segment that could be located between channels. The channel residuals were calculated based on the assumption that all the channels remain full. An assumption was made for the channel construction and representative channel dry weights were computed. The sum of the three components, pool residuals, channel residuals and channel weights, were computed for each configuration for all the different fluids to be used. In all cases, the eight vertical channels without a horizontal connecting channel (Figure 4-35c) was the lightest and was chosen for detailed conceptual design. In the case of N_2O_4 the weight for configurations 1, 2 and 3 were respectively 338 kg (854 lb), 183 kg (404 lb), and 171 kg (377 lb). An increase in the number of vertical channels above 4 was considered but the additional channel residuals increased the weight more than the amount saved by the smaller pool residuals.

The channels utilize a rectangular rather than round cross section in order to minimize double curvature of the screen. The wide side of the channel was placed parallel to the tank wall in order to reduce pool residuals. The calculation procedures used are presented in Appendix G.

4.2.3.5 Orbiter Supply Final Conceptual Design Data - An overall schematic of the system and conceptual design drawings are presented respectively in Figures 4-36 and 4-37. System weights and a fluids inventory are presented respectively in Tables 4-29 and 4-30. Fluid quantities presented are based on supplying initially empty receivers. Transfer times and power usage are presented in Table 4-31. Additional hardware weight data are presented in Appendix F.

Two transfer cases are considered. Case 1 assumes all OMS fluids (N_2O_4, MMH and He) are to be transferred. Case 2 assumes all EPS/ECLSS, LH_2 and LO_2 are transferred, with the OMS N_2O_4 and MMH off-loaded as necessary to meet the Shuttle 29,510 kg (65,000 lb) payload limitation. In both cases, all the OMS He is carried. Even though a complete resupply of N_2O_4 and MMH fluids is not accomplished in Case 2, the entire OMS propellant tankage may need pressurization.

The N_2O_4, MMH, He, H_2, and O_2 are all stored as liquids. Low-g liquid acquisition for transfer is accomplished using eight screened channels in each tank. In all cases the basic channel designs are similar, except that for the cryogenics additional wicking screens are incorporated into the channels to prevent the channels from drying out from external heating. Fluid expulsion and/or NPSH is supplied by helium pressurant, and for simplicity and to eliminate screen drying during transfer, each system is pressurized with helium at the same temperature as the liquid being transferred. The helium pressurization systems are also used to expel fluids for emergency dumping, as required during Shuttle abort. Helium for N_2O_4 and MMH transer is stored at 33.1 mN/m^2 (4800 psia) and ambient temperature. Helium for LH_2 and LO_2 tank pressurization is stored at 22.7 mN/m^2 (3300 psia) within each liquid supply tank. LHe tank pressurization is by external pumping and vaporization of helium stored as part of the LHe supply.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

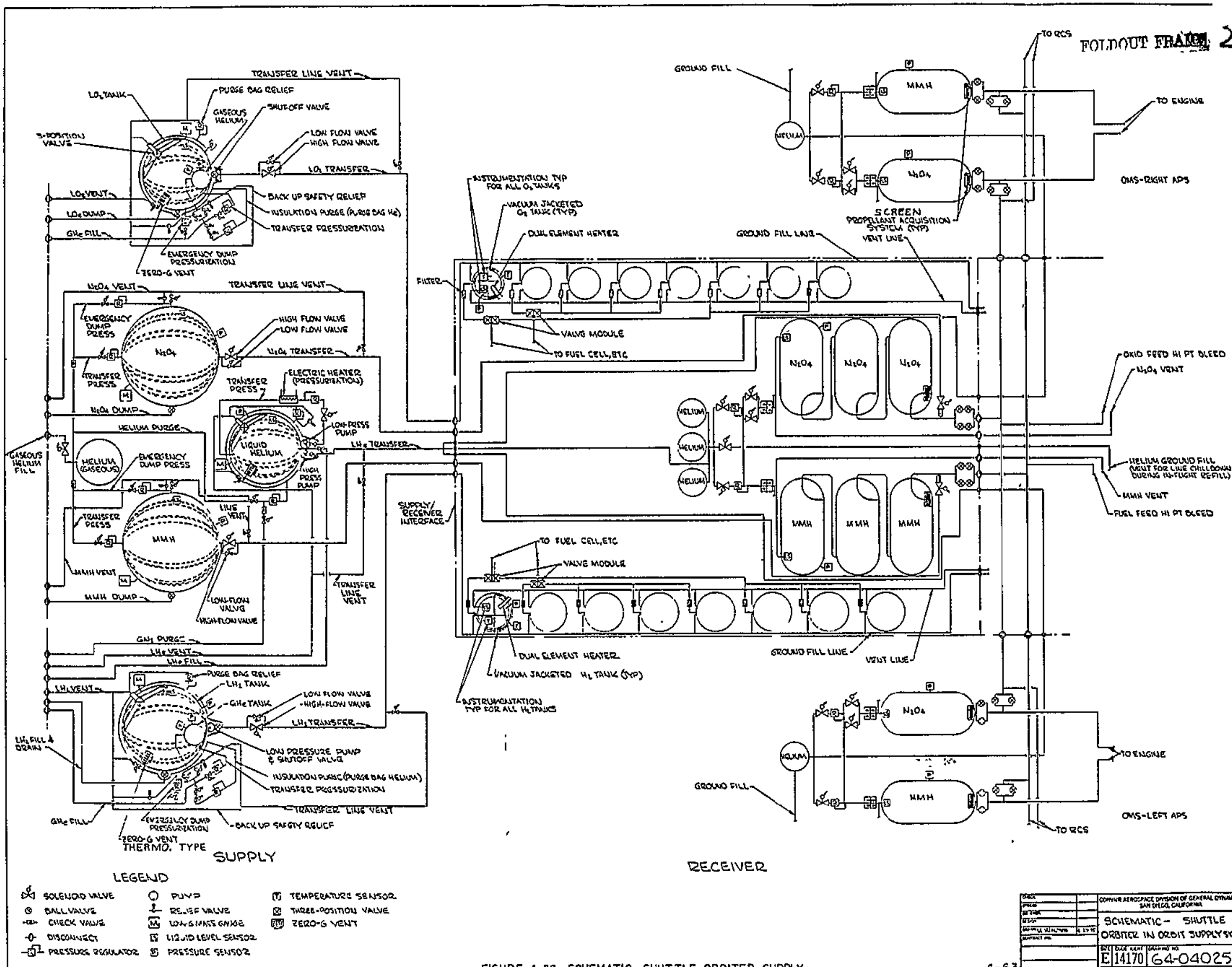


FIGURE 4-36. SCHEMATIC-SHUTTLE ORBITER SUPPLY

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FOLDOUT FRAME 4

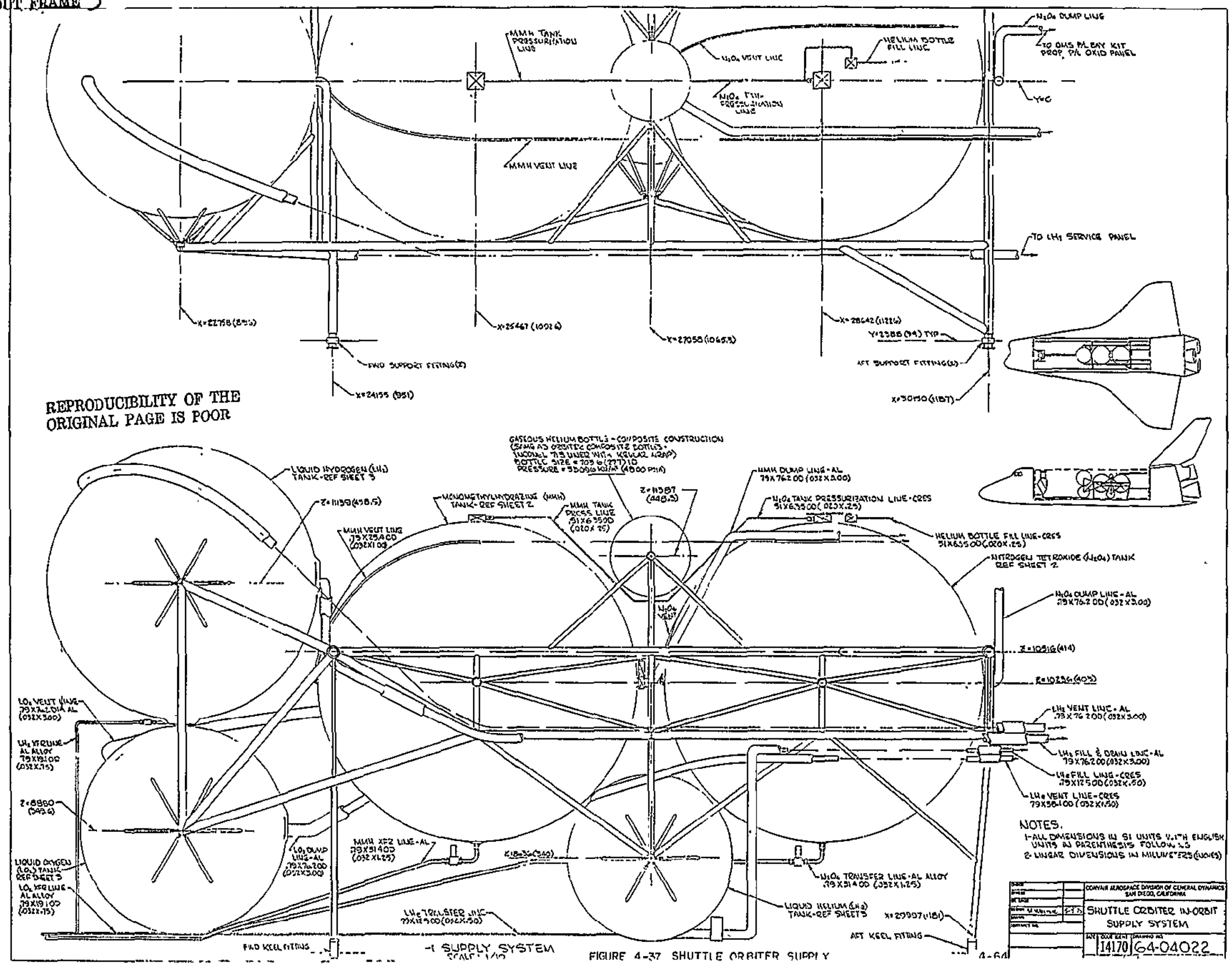

FIGURE 4-37 SHUTTLE ORBITER SUPPLY

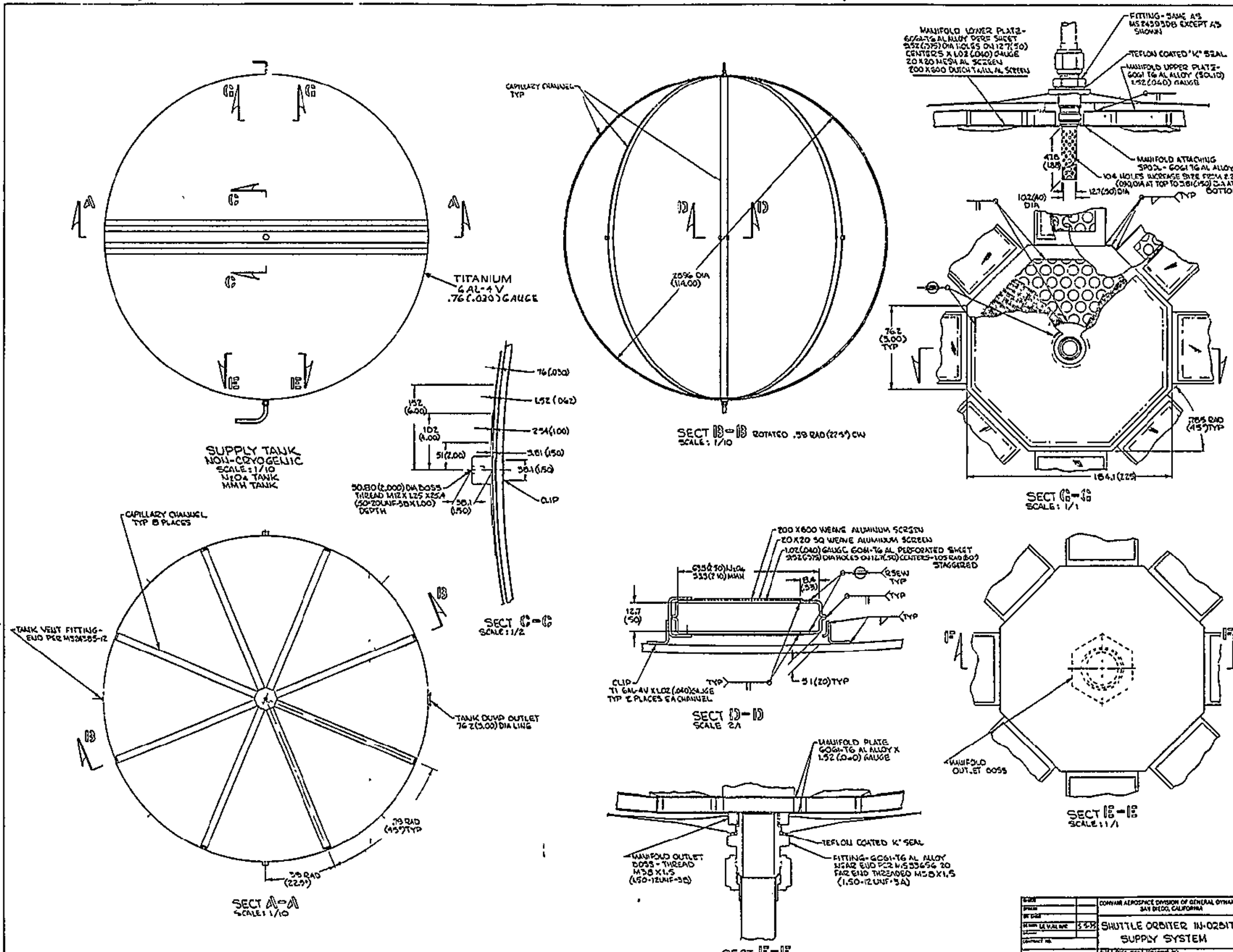

FIGURE 4-37. SHUTTLE ORBITER SUPPLY- CONT.

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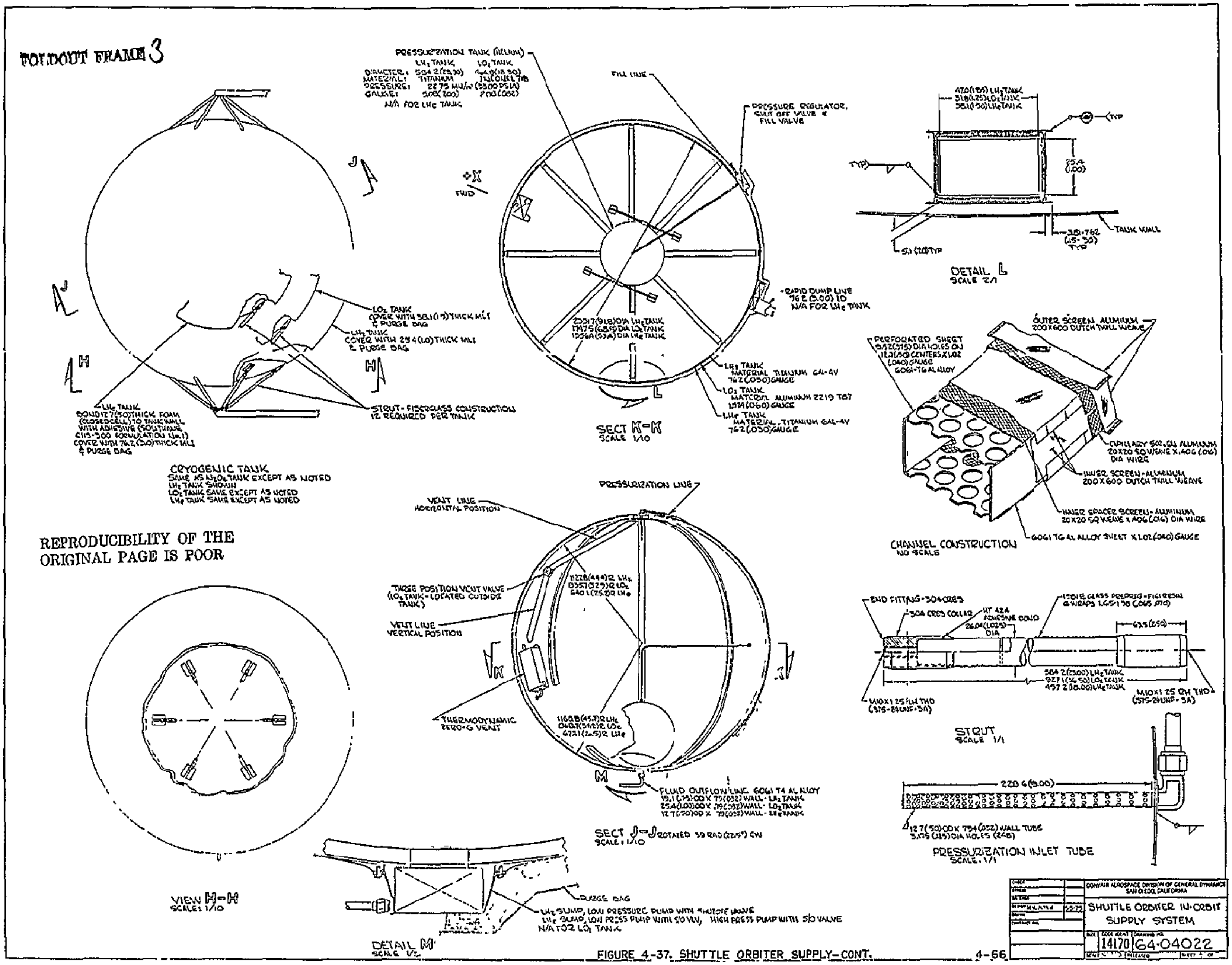

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FIGURE 4-37. SHUTTLE ORBITER SUPPLY-CONT.

Table 4-29. Shuttle Orbiter Supply System Weights

Item	Weight kg	Weight lb
Case 1 Transfer		
N_2O_4 System (Dry)	143	315
MMH System (Dry)	144	317
Ambient GHe System (Dry)	53	117
LHe System (Dry)	109	240
Overall Module Mounting Structure	183	404
Total Dry Weight	623	1393
Total Fluids (Ref. Table 4-30)	28780	63392
Lift-Off Weight	29412	64785
Total Fluids Supplied (Ref. Table 4-30)	28384	62520
Total Fluid Residuals (Ref. Table 4-30)	292	643
Total Return Weight	924	2036
Fluid Supplied/Lift-Off, %		96.5
Total Fluid Residuals, %		1.0
Case 2 Transfer		
N_2O_4 System (Dry)	143	315
MMH System (Dry)	144	317
Ambient GHe System (Dry)	53	117
LHe System (Dry)	109	240
LH_2 System (Dry)	176	387
LO_2 System (Dry)	139	307
Overall Module Mounting Structure	183	404
Total Dry Weight	947	2087
Total Fluids (Ref. Table 4-30)	28563	62913
Lift-Off Weight	29510	65000
Total Fluids Supplied (Ref. Table 4-30)	27894	61440
Total Fluid Residuals	382	841
Total Return Weight	1329	2928
Fluid Supplied/Lift-Off, %		94.5
Total Fluid Residuals, %		1.3

The N_2O_4 and MMH tanks are uninsulated, while the cryogenic tanks use high performance multilayer insulation with a rigid purge bag to prevent moisture condensation and freezing in the insulation during ground hold, boost and entry. Residual helium from the N_2O_4/MMH pressurant bottle is used for LHe tank purge bag pressurization during entry. Residuals from the LH_2 and LO_2 helium systems are used for purge bag pressurization of their respective tankage during entry.

During ground hold the helium tank insulation is purged with GN_2 to minimize heating. GN_2 has a lower effective conductivity than GHe and can be used with the LHe system without the fear of freezing because the LHe tank includes a 12.7 mm (0.5 in.) layer of closed-cell foam. MLI measuring 76 mm (3 in.) covers the foam. MLI measuring 25.4 mm (1 in.) and 38.1 mm (1.5 in.) without any foam is used respectively for the H_2 and O_2 tanks.

Helium at 207 kN/m^2 (30 psia) and 172 kN/m^2 (25 psia) is used respectively for N_2O_4 and MMH expulsion. These pressures are based on an optimization of pressurant system versus line and line residual liquid weights. The primary problem with the transfer of these fluids is in insuring that the screen start baskets in the receiver tanks are full at the end of transfer. In the present case, it is assumed that the receiver tanks can be vented to vacuum prior to transfer, or at least that all helium can be eliminated from the tanks such that the receivers can be locked-up during transfer and any fluid vapor trapped in the baskets will be condensed due to pressurization during and/or following fill. The vapor pressures of N_2O_4 and MMH at 294K (70F) are respectively 101 and 5.5 kN/m^2 (14.7 and 0.8 psia).

Even though the final He receiver condition required is ambient gas at 33.1 mN/m^2 (4800 psia), liquid helium is transferred to these receivers by a high pressure low flow rate pump. Initially cold helium flows to the receivers and heat is transferred from the initially warm lines and receiver bottles, thus increasing the temperature and pressure of the incoming helium. Some heat is also transferred from the ambient by radiation to the line and tank walls. However, even with uninsulated lines and receiver

Table 4-30. Fluids Inventory for Orbiter Supply

Case 1 Use, kg (lb)	Fluid							
	N_2O_4		MMH		LHe		GHe	
Required for Receiver Mission	17633	(38840)	10646	(23450)	104	(230)	-	
Transfer Line and Receiver Chilldown	0	0	0	0	0	0	-	
Transfer Line Residuals	55	(121)	33	(72)	—	(2)	-	
Total Transferred	17688	(38961)	10679	(23522)	105	(232)	-	
Supply Residuals	162	(356)	89	(195)	0.5	(10)	0.1	(0.3)
Supply Vented Prior to Transfer	0	0	0	0	15	(34)	0	0
Fluid Used for Pressurant, Purge and Safing	-	-	-	-	29	(63)	8	(18.7)
Total Initial Supply Load	17850	(39317)	10768	(23717)	157	(339)	7	(19.0)

Case 2 Use, kg (lb)	Fluid											
	N_2O_4		MMH		LHe		GHe		LH_2		LO_2	
Required for Receiver Mission	15350	(33812)	9268	(20414)	104	(230)	-		334	(736)	2837	(6248)
Transfer Line and Receiver Chilldown	0	0	0	0	0	0	-		84	(186)	69	(152)
Transfer Line Residuals	55	(121)	33	(72)	1	(2)	-		1	(2)	10	(22)
Total Transferred	15406	(33933)	9301	(20486	105	(232)	-		419	(924)	2916	(6422)
Supply Residuals	162	(356)	89	(195)	5	(10)	2	(5)	7	(15)	54	(120)
Supply Vented Prior to Transfer	0	0	0	0	15	(34)	0	0	12	(26)	7	(15)
Fluid Used for Pressurant, Purge and Safing	0	0	0	0	29	(63)	35	(77)	0	0	0	0
Total Initial Supply Load	15567	(34289)	9389	(20681)	154	(339)	37	(82)	438	(965)	2977	(6557)

tanks this heat is small compared to that generated by pumping and that absorbed from the receiver tank walls. Calculations show a final receiver charging pressure of 24.8 MN/m^2 (3600 psia) at an equilibrium fluid/wall temperature of 236K (425R). The receivers are then allowed to come to ambient conditions for their subsequent use. Maximum initial receiver tank temperatures were taken to be 311K (560R).

For H_2 transfer, liquid is transferred to supercritical receivers. Due to the high bottle masses involved, a non-vent transfer is unfeasible. The method chosen here to minimize vent chilldown losses is to vent the receiver tanks until the wall temperatures reach approximately 94.4K (170R), at which time the tanks are locked-up and filling continued to a final

Table 4-31. Transfer Times and Power Usage

	Fluid				
	N_2O_4	MMH	LHe	LO_2	LH_2
Transfer Time, hr	4	4	4	2	2
Power Required, MJ (kw-hr)					
Pumping	-	-	42.1 (11.7)	-	1.4 (0.4)
Pressurant Vaporization	-	-	0.7 (0.2)	-	-
Total Power	-	-	42.8 (11.9)	-	1.4 (0.4)

pressure of 276 kN/m^2 (40 psia). This procedure should result in a minimum of liquid venting. A liquid pump is used here for transfer rather than only helium pressurization due to the relatively high weight of the cold helium gas and associated storage bottle. Following the transfer, the receiver H_2 is heated to its operating pressure condition with electric heaters already located in the receiver tanks.

The transfer procedure for LO_2 is essentially the same as for the hydrogen except that supply tank pressurization is at 276 kN/m^2 (45 psia) to accomplish a final receiver pressure of 241 kN/m^2 (35 psia) without pumping. In the case of LO_2, pumping is more of a safety problem than with H_2 and the weight penalty for an all helium pressurized transfer is not as great. In this case the receivers are vented during chilldown,as necessary, until a wall temperature of 250K (450R) is reached. Maximum initial wall temperatures were taken to be 350K (630R) corresponding to the maximum supercritical fluid condition at normal depletion.

All of the liquid supply and pressurant storage tanks are mounted in a tubular framework that is in turn mounted in the shuttle cargo bay. The cryogenic tanks (LHe, LH_2, and LO_2) are attached to the framework by fiberglass struts while the non-cryogenic tanks are attached by bosses on the sides of the tanks. The LO_2 and LH_2 tanks are designed to be removed when the N_2O_4 and MMH tanks are carried full. This is necessary to meet the weight limitation associated with the shuttle cargo bay payload. All the other tanks remain with the basic supply module.

The tubular framework is constructed of welded aluminum alloy tubing with six support fittings to mount it in the shuttle cargo bay. The six support fittings are used in order to prevent large moments from warping the light framework and distorting the propellant tanks.

The non-cryogenic supply tanks (N_2O_4 and MMH) are of similar construction, differing only in the acquisition system channels. The tanks are 2896 mm (114 in.) in diameter with a 0.76 mm (0.030 in.) skin gauge and are made of titanium alloy. A pressurization fitting is installed in the top of the tank and a fluid outlet fitting is mounted directly opposite in the bottom of the tank. The tanks have a 76.2 mm (3.0 in.) diameter outlet in the side, located between two acquisition channels for dumping the propellants. The tanks are mounted in the cargo bay such that this dump out is facing the aft bulkhead of the cargo bay for rapid dump with acceleration on the shuttle (tank drain with the shuttle in a vertical position). The normal in-orbit transfer outlet allows tank drain with the shuttle in a horizontal position. A tank vent fitting is mounted in the tank wall directly opposite the tank dump outlet. The acquisition systems consist of an upper and a lower manifold connected by eight channels. The channels are attached to the tank wall at the midsection of the tank by two titanium clips on each channel. The upper and lower manifolds are attached to the tank by the pressurization fitting and the outlet fitting. Teflon coated aluminum "K" seals are used for sealing at these two outlets.

The channel construction is of aluminum alloy. The sides of the channel are solid sheetstock while the inner and outer surfaces are made up of a perforated backing sheet, 9.52 mm (0.375 in.) diameter holes on 12.7 mm (0.50 in.) centers, covered by an inner 20 × 20 square weave aluminum screen and an outer 200 × 600 Dutch twill weave aluminum screen. The outer 200 × 600 screen is seam welded to the solid aluminum channel sides to seal the channel. The manifold construction is similar for the inside surface of the upper manifold. All other manifold surfaces are solid aluminum sheet.

All of the cryogenic tanks are of different sizes and are supported by external fiber-glass struts rather than bosses on the tank wall. Each has an acquisition system somewhat similar to the non-cryogenic tanks.

The liquid helium tank is the smallest, 1356 mm (53.40 in.) in diameter and is made of titanium alloy 0.762 mm (0.030 in.) thick. A thermodynamic zero-g vent system is installed in the tank along with a normal ground vent system, consisting of a three-position valve and inlet lines to the valve to allow venting in either the vertical or horizontal position. The acquisition channels are constructed similar to the non-cryogenic channels in that the sides are solid, an inner and outer backing sheet of perforated aluminum is used and an inner spacer screen of 20 × 20 mesh aluminum is used between the 200 × 600 Dutch twill screen and the perforated sheet. The 200 × 600 screen is not welded to the inner solid channel, however, and another 20 × 20 aluminum spacer screen is used between this 200 × 600 Dutch twill screen and another (outer) 200 × 600 Dutch twill screen. This outer spacer screen is used as a capillary screen for wicking between the two layers of 200 × 600 screen so that the screens remain wet. The channel has an internal width of 38.1 mm (1.50 in.) and an internal thickness of 25.4 mm (1.00 inches). The clearance between the channel and the tank wall is 5.1 mm (0.20 inches).

The liquid hydrogen tank is similar to the liquid helium tank except for size and the addition of a rapid dump line in the tank wall (similar to the non-cryogenic tanks). Also, a gaseous helium pressurization bottle is installed in the bottom of the tank.

The hydrogen tank is made of titanium alloy with a skin gauge of 0.762 mm (0.030 inches). The tank diameter is 2332 mm (91.80 inches). The rapid dump line is 76.2 mm (3.00 inches) diameter.

The pressurization bottle is a conventional design titanium bottle, 584 mm (23.00 inches) in diameter with a skin gauge of 5.08 mm (0.200 inches). It is designed for 22.75 mN/m^2 (3300 psia) pressure.

The acquisition system channels are 47.0 mm (1.85 inches) in width and 25.4 mm (1.00 inches) in thickness.

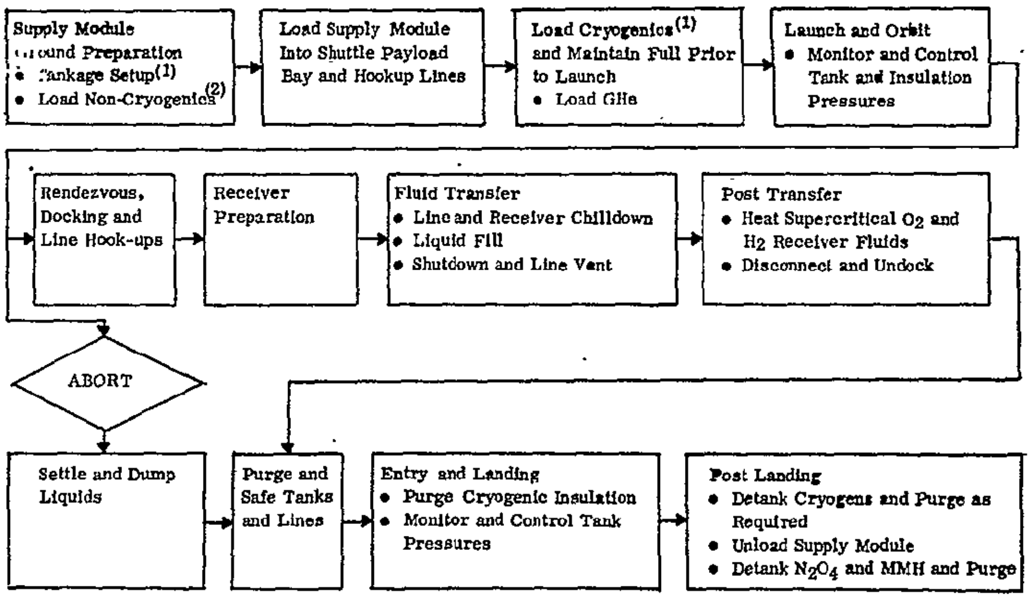
The liquid oxygen tank is similar to the hydrogen tank in design. It is an aluminum (2219) tank with a skin gauge of 1.524 mm (0.060 inches) and a diameter of 1748 mm (68.80 inches). The internal helium bottle is made of Inconel 718 and is 465 mm (18.30 inches) in diameter with a skin gauge of 2.08 mm (0.082 inches). The acquisition channels for the LO_2 tank are 31.8 mm (1.2 inches) wide by 25.4 mm (1.00 inches) thick. The LO_2 tank does not have an outlet sump and pump as does the liquid helium tank and the liquid hydrogen tank.

4.2.3.6 Shuttle Orbiter Supply Operating Procedure - Two transfer cases are considered.

1. Full load of OMS fluids (MMH, N_2O_4, LHe) with the LO_2 and LH_2 tanks not carried.

2. Full load of LO_2 and LH_2 with the MMH and N_2O_4 off-loaded in the amounts necessary to meet the Shuttle 29,510 kg (65,000 lb) payload limitation.

With respect to overall operations, the only difference between the two cases is that in Case 1 the LO_2 and LH_2 tankage are not carried and transfer of these two fluids is not accomplished. A functional flow diagram of the basic overall supply operation is presented in Figure 4-38. The various operations outlined in Figure 4-38 are expanded and discussed in the following paragraphs. A schematic of the complete transfer system is found in Figure 4-36.



(1) For Case 1, Full OMS Load, LO_2 and LH_2 Tankage Not Carried
(2) For Case 2, Only a Partial Load of N_2O_4 and MMH is Carried

Figure 4-38. Overall Shuttle Orbiter In-Orbit Supply Operations

Supply Module Ground Preparation - At this point the LO_2 and LH_2 supply tankage and associated pressurization and purge systems are either installed or removed from the supply module, depending on which supply case was last accomplished and which case is being prepared for.

Following hookup of the necessary lines, the non-cryogenic liquids, MMH and N_2O_4 are loaded into the supply module. The most critical factor here is to insure that the surface tension screened channels are completely filled with liquid, such that no non-condensible vapor is trapped within the channels. This is accomplished by providing a vent at the top of the channels and allowing liquid to flow out this vent before it is closed. Some ullage (approximately 3%) is provided in the remainder of the tank to allow for potential thermal expansion of the liquid.

Load Supply Module Into Shuttle - This simply consists of the installation of the supply module in the Shuttle Orbiter cargo bay and the hookup of all ground supply, abort dump and in-orbit transfer lines.

Load and Maintain Cryogenics - At this point the cryogenics, LHe, LO_2 (Case 2 only) and LH_2 (Case 2 only) are loaded into the supply module. This operation consists of initial line and tank chilldown and final loading. It is assumed that the maximum fluid loading is maintained by topping up to 180 seconds prior to launch. The high pressure helium storage bottles are also filled at this time. The cryogenic insulation for the LO_2 and LH_2 tanks is maintained in a purged condition with GHe from the ground prior to boost. The LHe tank insulation is purged with GN_2 to minimize liquid boiloff from external heating. As with the non-cryogenics, the surface tension screened channels must be filled with liquid to prevent the trapping of inert gases.

Launch and Orbit - During this phase of the resupply operation the main problems are to control tank and insulation pressures and to prevent the surface tension channels in the cryogenic tanks from drying out due to external heating or excessive liquid sloshing. Some venting of the LH_2, LO_2 and LHe tanks is allowed for during the boost phase. This occurs through the normal ground boiloff vents. Following entry into orbit, the zero-g vent valves are designed to accommodate any venting required. Tank over-pressure is a safety critical failure and provisions are made to allow the immediate dumping of liquids as well as emergency venting.

Abort (Only as Necessary) - If for some reason it becomes necessary to terminate the mission during boost or while in orbit, all liquids are dumped, with the exception of the LHe, which is considered to be a non-hazardous fluid. Sufficient vent capacity is, however, provided to prevent LHe tank overpressure during an abort or in case of an insulation or insulation purge failure. Liquid settling to the aft end of the Shuttle is provided by the normal Shuttle abort procedure. In the event that liquid dumping could not be accomplished, system design, with respect to c. g., is such that a satisfactory landing could still be accomplished.

Rendezvous and Docking - The main problem to consider here is maintenance of the screened surface tension channels in a full condition, ready for liquid transfer at low-g. The most critical case in this respect will be for the non-cryogenics where only a partial load of MMH and N_2O_4 are carried (Case 2). Even though accelerations are less than during boost the direction may be random and completely unpredictable.

Transfer line hookups will be accomplished upon docking or immediately thereafter. It is assumed that the lines are empty and vented to the atmosphere (vacuum) prior to hookup.

Receiver Preparation - Specific problems and operations peculiar to each of the receivers/fluids are presented below.

a. N_2O_4 and MMH - The main problem with these systems is to insure that the screen start baskets in the receiver tanks are full at the end of transfer. In the present case it is assumed that the receiver tanks can be vented to vacuum prior to transfer; or at least that helium can be eliminated from the tanks to the extent that the receiver can be locked-up during transfer so that any vapor trapped in the baskets will be condensed due to pressurization during and/or following fill.

b. High Pressure Helium and Supercritical O_2 and H_2 - The operation here is simply to vent the receiver tanks to be filled to approximately 0 pressure (vacuum).

Fluid Transfer - Specific fluid transfer operations are described separately below for each of the fluids involved. Depending on specific mission requirements each fluid could be transferred at a different time or at the same time as other fluids.

a. N_2O_4 and MMH - It is assumed that, for each fluid, one tank at a time is filled. The following sequence of events is proposed.

   (1) Actuate helium pressurization to pressurize supply tanks - 207 kN/m^2 (30 psia) N_2O_4, 172 kN/m^2 (25 psia) MMH.

   (2) Check that transfer line vents are closed.

   (3) Open supply tank low-flow outlet valves.

   (4) Open inlet valves of the particular receiver to be filled.

   (5) Following initial fill, designed to provide complete receiver surface tension start basket fill, open supply tank high-flow valves.

   (6) Following fill of each receiver tank, as indicated by low-g mass gauges, close the receiver inlet valves and the supply high-fill valves.

(7) During the transfer the supply and receiver mass gauges should be closely monitored to insure that the transfer is taking place as desired, without any excessive loss of liquid. If such loss occurs it may be necessary to terminate transfer of a particular receiver and go on to the next one.

(8) Repeat Steps (4) through (7) until all receivers are full.

(9) Close all supply valves and vent transfer lines to vacuum.

b. Helium - The following sequence of events is proposed.

(1) Activate the pressurant electric heater and low pressure pump and pressurize the supply tank to 138 kN/m^2 (20 psia).

(2) Open the chilldown vent valve located on the receiver and then the LHe supply tank outlet valve.

(3) Monitor pump and line temperatures and liquid at the chilldown vent outlet. When temperatures reach predetermined values or when liquid appears at the line vent, close the vent and activate the high pressure transfer pump.

(4) Open receiver tank inlet valves and allow filling to occur until receiver temperatures and pressures reach predetermined values indicating a full load. Nominal final receiver conditions are calculated to be 24.8 mN/m^2 (3600 psia) and 236K (425R)

(5) Close receiver inlet and supply outlet valves and terminate pump and pressurant heater operations.

(6) Vent transfer line to vacuum in preparation for disconnection and undocking.

c. LH_2 - Case 2 only

(1) Activate supply tank pressurization to accomplish pressurization to 138 kN/m^2 (20 psia).

(2) Open receiver vent.

(3) Actuate transfer pump and open supply tank low-flow valve and allow line and receiver tank chilldown to occur until the average receiver tank wall temperature reaches 94.4K (170R).

(4) Close receiver tank vent and open supply tank high-flow valve and continue fill until receiver is full; anticipated to be when the receiver pressure reaches 276 kN/m^2 (40 psia).

(5) Close receiver inlet valves and supply tank high-flow valve.

(6) Repeat Steps (2) through (5) until all receivers are full.

(7) Close all supply valves and vent the transfer line.

Note: During transfer to the last receiver, the high-flow transfer valve is not opened (entire transfer accomplished at low flow rate) in order to minimize supply tank residuals.

d. LO_2 - Case 2 only - The procedure here is identical to that for the LH_2, except that receiver tank chilldown while venting is now only to an average wall temperature of 250K (450R) and the final receiver pressure is anticipated to be 241 kN/m^2 (35 psia) rather than 276 kN/m^2 (40 psia). Also, in this case, transfer is by pressurization-only at a supply tank pressure of 310 kN/m^2 (45 psia).

Post Transfer - The primary operation here, other than normal system shutdown mentioned in the previous section, is to bring the supercritical O_2 and H_2 receivers to their final use conditions. This is accomplished by heaters, already installed, which are normally used for expulsion. The O_2 and H_2 use pressures are 6.5 and 1.96 MN/m^2 (950 and 285 psia) respectively.

Purging and Safing - Prior to entry and landing, LO_2 and LH_2 residuals are completely dumped and the tanks purged with helium. This is especially critical in the case of H_2 where venting cannot be accomplished immediately upon landing, as would likely be required due to entry heating if LH_2 were remaining.

Entry and Landing - The primary operations here are to (1) control pressure in the insulation purge bags to slightly above ambient to prevent entry of air with subsequent moisture condensation and insulation damage, and (2) maintain tank pressures above ambient to prevent implosion and below the design pressure to prevent bursting.

Post Landing - Following landing of the Shuttle Orbiter, any remaining cryogens are detanked and the tanks purged with helium gas and locked-up. The complete supply module is then unloaded from the Shuttle Orbiter and any remaining N_2O_4 and MMH drained. These tanks are then purged and locked-up with inert gas and the complete system stored. Prior to the next use of the supply module, the surface tension screens should be checked for cleanliness and cleaned if necessary, since contaminated screens could significantly reduce their available flow area.

4.2.4 MULTIPLE RECEIVERS IN-ORBIT SUPPLY SYSTEM - The supply module designed here is to be capable of supplying three different receivers. These are the Solar Electric Propulsion Stage (SEPS), Large High Energy Observatory B (HEAO-B) and Satellite Control Section (SCS). Hg, LHe and N_2H_4 are supplied respectively to each of these receivers in the quantities specified in Table 4-1. Different liquid acquisition schemes are employed for each supply case. These are, respectively, diaphragm, paddle vortex and surface tension screen channels.

An initial trade-off was made to determine the optimum packaging of the various supply tanks. Use of completely separate supply modules and structure for each receiver, integral module and framework with unused tanks not carried,and integral module with unused tanks carried empty were the three concepts considered. In each case, missions were considered requiring supply of any one, any two or all three receivers.

Comparative weight data are presented in Table 4-32. It was concluded that the small potential weight savings of separate modules does not warrant the added complexity and cost. Use of an integral module with unused tanks not carried was chosen as best. It is almost as light as the separate module system and has the advantage, over carrying all the tanks, of not having to control pressure in unused tanks during boost and re-entry.

The major trade-offs and analyses accomplished for each supply system are presented in the following paragraphs.

4.2.4.1 Solar Electric Propulsion Stage - The only problems with transfer of Hg are associated with its high density and its corrosiveness. Double wall CRES tanks and AF-E-332 diaphragms are used for their compatibility with Hg. Trade-offs were made between various supply tank pressurization schemes. A main consideration was the desire to maintain small ullage volumes prior to transfer to minimize liquid sloshing. Due to the small tank volumes involved it was determined to use GN_2 as the pressurant. He showed only an insignificant weight advantage and costs more than GN_2.

The three pressurization schemes listed below were considered.

a. Pressurant stored as the supply tank ullage and allowed to blowdown during transfer.

Table 4-32. Design Packaging Trade-offs

	(1) Transfer Module Weights, kg (lb)						
	Transfer Mission						
	(2)A+B+C	A+B	A+C	B+C	A Only	B Only	C Only
Supply Module Concept	kg (lb)	kg (lb)	kg (lb)	kg (lb)	kg (lb)	kg (lb)	kg (lb)
Integral Framework (Unused Fluid Tanks Carried Empty & Purged)	4059 (8940)	2520 (5550)	3375 (7435)	2515 (5540)	1836 (4043)	974 (2145)	1830 (4030)
Integral Framework (unused Tanks Not Carried)	4059 (8940)	2479 (5460)	3242 (7140)	2483 (5470)	1662 (3660)	903 (1990)	1668 (3675)
Separate Modules	4081 (8990)	2465 (5430)	3239 (7135)	2490 (5485)	1623 (3574)	874 (1925)	1648 (3630)

(1) Weights include tankage, structure and fluids.
(2) System A - Solar Electric, Hg
System B - Large HEAO B, LHe
System C - Satellite Control Section, N_2H_4

b. Pressurant stored in a separate tank and allowed to blowdown into the supply tank during transfer.

c. Pressurant stored in a separate tank and regulated to a constant supply tank pressure for transfer.

Final weight differences between the three schemes were found to be small. Therefore, for simplicity, along with maintaining a full supply tank to minimize sloshing, the blowdown system with pressurant stored in a separate bottle was chosen (System b). The initial storage pressure, for minimum system weight, was found to be 1378 kN/m^2 (200 psia) with blowdown to 345 kN/m^2 (50 psia) at the end of transfer.

4.2.4.2 Large High Energy Observatory B - The major problem with this system is that the receiver tank and superconducting magnet are relatively heavy and may require a large amount of fluid just for chilldown. The operating temperature of the magnet is such that helium saturated at approximately 103 kN/m^2 (15 psia) is required in the receiver. Transfer without significant receiver venting to maintain this pressure was determined to be unfeasible with current technology. The heat of vaporization of helium is very low, requiring a potentially large mass of helium for chilldown.

Comparisons were made between the use of helium for the total chilldown and use of LN_2 for initial chilldown with helium used only for final chilldown and fill. Weight data are presented in Table 4-33. As discussed in Paragraph 4.1 and illustrated in Table 4-33, there is a significant weight difference between chilldown where saturated vapor is vented versus venting of superheated vapor at a temperature corresponding to the tank wall as it chills. This is especially true for helium with its low heat of vaporization and high vapor specific heat. In addition to these two extreme cases, data are also presented in Table 4-33, assuming that saturated vapor venting chills the magnet while superheated vapor chills the wall. In this case it is assumed that wall cooling can be accomplished by passing the vented vapor through existing heat exchanger coils located within the receiver tank insulation. It is noted that in none of the cases considered here, is any liquid assumed to be lost directly through the vent,and means would need to be provided to insure this.

Table 4-33. Optimum Method of Chilldown of Large HEAO B

Chilldown Assumption	(1) System Weights: Liquid Helium Only, kg (lb)	Liquid Nitrogen Plus LHe, kg (lb)
1. With most efficient chilldown (only superheated vapor vented).	157 (345)	279 (614)
2. Chilldown with saturated vapor vented.	2260 (4977)	493 (1085)
3. Saturated vapor to chill magnet and superheated vapor to chill tank wall	959 (2112)	361 (795)

(1) Weights include storage tank and insulation, chilldown fluid, supply boil-off and receiver tank He purge (where applicable).

From Table 4-33, in general, the use of LN_2 as a pre-chill resulted in significantly lower total system weight than use of helium alone. The only way the use of helium alone would be weight competitive would be if the chilldown system could be designed such that helium vapor always left the receiver at the maximum temperature of the hardware as it was being chilled. This could possibly be accomplished for the tank wall using the existing vent cooling coils; however, it is not likely that cooling of the magnet in this manner could be assured unless special cooling coils and/or cooldown procedures were employed. This would need to be verified by considerable analysis and test.

Liquid acquisition for both the N_2 and He systems is assumed to be with a paddle. Calculation methods from Paragraph 4.1.2 and Reference 4-2 were used to design the paddle systems.

4.2.4.3 Satellite Control Section - Even though hydrazine is supplied to meet both Orbital Adjust (OAS) and Reaction Control (RCS) system requirements, transfer is assumed to be to a single receiver. The OAS tank incorporates screen channels and is designed to feed the RCS bladder tanks.

The screen channel system used for supply liquid acquisition is taken to be of the same basic design as that developed for N_2O_4 and MMH in Paragraph 4.2.3.

The major problem with the SCS system is to insure that the screen channels in the receiver are full and that the receiver is charged to its final use pressure of 2136 kN/m^2 (310 psia) at the end of transfer. The SCS system operates by blowdown of GN_2 stored as ullage in the main storage tank at an initial pressure of 2136 kN/m^2 (310 psia). Weight trade-offs showed that use of a pump in conjunction with GN_2 pressurization to provide NPSH was the best way to charge the SCS to its use pressure.

Final design characteristics of the overall multiple receivers supply system are presented in the following paragraph.

4.2.4.4 Multiple Receivers Conceptual Design Data - An overall schematic of the system and conceptual design drawings are presented respectively in Figures 4-39 and 4-40. System weights and an inventory of supply liquids are presented respectively in Tables 4-34 and 4-35. Transfer times and power usage are presented in Table 4-36. Additional hardware weights are presented in Appendix F. The long transfer times for the Large HEAOB include initial LN_2 chilldown plus purge and final chilldown with LHe. The overall supply system includes provisions to supply all the fluids illustrated in Figure 4-39 and Table 4-35 on a single mission. However, the supply module design is such that if less than all the receivers are to be supplied, only the supply tankage associated with the receiver(s) to be supplied are carried, except that, due to its mounting complexities, the mercury tankage is always carried.

For the Solar Electric stage two mercury supply tanks are employed. This is done for satisfactory control of the center of gravity as necessitated by the high concentration of weight associated with the mercury system. Double wall tanks are employed for safety to eliminate the chance of a spill of the highly corrosive mercury into the Shuttle payload bay.

Table 4-34. Multiple Receivers Supply System Weights

Item	Weight kg	Weight lb
SEPS Hg System (Dry)	69	153
HEAO B System (Dry)	181	398
LN_2 System, 30.4 kg (67 lb)		
LHe System, 136 kg (300 lb)		
GHe System, 14.1 kg (31 lb)		
SCS N_2H_4 System (Dry)	72	159
Docking Module	21	46
Overall Support Structure	77	170
Total Dry Weight	420	926
Total Fluids (Ref. Table 4-35)	3825	8426
Lift-Off Weight	4246	9352
Total Fluids Supplied (Ref. Table 4-35)	3438	7572
Total Fluid Residuals (Ref. Table 4-35)	71	156
Total Return Weight	491	1082
Fluids Supplied/Lift-Off, %		81.0
Total Fluid Residuals, %		1.9

Table 4-35. Multiple Receivers Fluids Inventory

Use, kg (lb)	Receiver/Fluid: Solar Electric, Hg	Solar Electric, GN_2	Large HEAOB, LHe	Large HEAOB, LN_2	Large HEAOB, GHe	Satellite Control, N_2H_4	Satellite Control, GN_2
Required for Receiver Mission	1498 (3300)	-	431 (950)	-	-	1508 (3322)	-
Transfer Chilldown, Purge and Boiloff	-	-	89 (197)	185 (407)	-	-	-
Transfer Line Residuals	2.2 (5)	-	0.5 (1)	1 (2)	-	1.4 (3)	-
Total Transferred	1500 (3305)	-	521 (1148)	186 (409)	-	1510 (3325)	-
Supply Residuals	7.7 (17)	-	16 (36)	7.2 (16)	-	23 (50)	0.05 (0.1)
Supply Vented Prior to Transfer	-	-	38 (83)	-	-	-	-
Fluid Used for Pressurant, Purge and Safing	-	1.4 (3)	11 (25)	-	1.4 (3)	-	2.7 (5.9)
Total Initial Supply Load	1508 (3322)	1.4 (3)	587 (1292)	193 (425)	1.4 (3)	1532 (3375)	2.7 (6.0)

Table 4-36. Transfer Times and Power Usage

	Receiver/Fluid: Solar Electric Hg	Large HEAOB LHe/LN_2	Satellite Control N_2H_4
Total Transfer Time, hr	0.5	28	1
Power Required, MJ [kw-hr (kw peak)]	-		
Pumping	-	0.072 [0.02 (0.04)]	6.1 [1.7 (2.0)]
Pressurant Vaporization	-	4.7 [1.3 (2)]	-
Liquid Orientation	-	2.5 [0.7 (0.12)]	-
Total Power	-	7.3 [2.02 (2.16)]	6.1 [1.7 (2.0)]

The supply is pressurized by N_2 stored in a separate bottle and allowed to blowdown from 1378 kN/m^2 (200 psia) to 345 kN/m^2 (50 psia). Storage of the N_2 pressurant in a separate tank, rather than as ullage in the supply tank, is done to allow essentially full liquid tanks, to eliminate sloshing problems prior to transfer. The transfer line is 6.4 mm (0.25 in.) diameter by 0.4 mm (0.016 in.) wall CRES with a total length of 589 cm (232 in.).

The Large High Energy Observatory B (Magnetic Spectrometer) receiver employs a superconducting magnet cooled with liquid helium. This magnet and the storage dewar have a relatively high mass that may need to be chilled at the initiation of transfer. Analysis showed that for reasonable weight, the magnet and tank should initially be chilled with LN_2. Liquid helium is then used for purge, final chilldown and fill. This is the same procedure presently proposed for ground fill of the operational HEAOB. As noted in Table 4-36, transfer times for this receiver are quite long. This is due primarily to the large masses to be chilled and the small vent lines existing on the HEAOB. Also, in order to increase the chilldown efficiency (minimum vent loss), a portion of the chilldown venting is accomplished at a low rate through the zero-g vent exchanger existing on the receiver tank. The time to chilldown to LN_2 temperature is estimated at 86.4 ks (24 hrs) with subsequent helium purging, chilldown and fill taking another 14.4 ks (4 hrs).

The LHe supply tank is protected with 7.6 cm (3 in.) of MLI (Superfloc) plus 1.3 cm (0.5 in.) of foam. The foam is necessary to prevent excessive boil-off during ground hold, after final topping, and during boost. A wall heat exchanger is employed for zero-g venting and the tank fluid is maintained in a mixed condition, as necessary, using the paddle. Using the paddle for complete liquid orientation could also provide back-up venting at low-g using the normal ground vent system.

The LN_2 tank employs 1.5 cm (0.6 in.) of MLI (Superfloc) and will not have to vent. This is insured by employing the paddle, as necessary, to mix the tank fluid to control the pressure rise rate.

The paddles are primarily employed for liquid orientation at the tank outlets during transfer. Nitrogen tank pressurization and LN_2 and LH_2 purge bag pressurization uses helium stored at ambient temperature and 33.1 MN/m^2 (4800 psia). LHe tank pressurization is by pumping, evaporation and heating to 78K (140R) of helium stored as liquid in the supply tank. This is the same pressurization concept as employed for System No. 3 (Paragraph 4.2.3.3), except that the pressurant temperature is higher here since screen drying was not a potential problem in the present case. Liquid expulsion is by pressure only, at 138 kN/m^2 (20 psia) in the LHe tank and 172 kN/m^2 (25 psia) in the LN_2 tank.

The Satellite Control Section Supply system employs screened channels for low-g liquid acquisition. Channels are also located in the receiver tank. The main transfer problem is insuring that the receiver screen channels are filled at low-g. Valving is provided,

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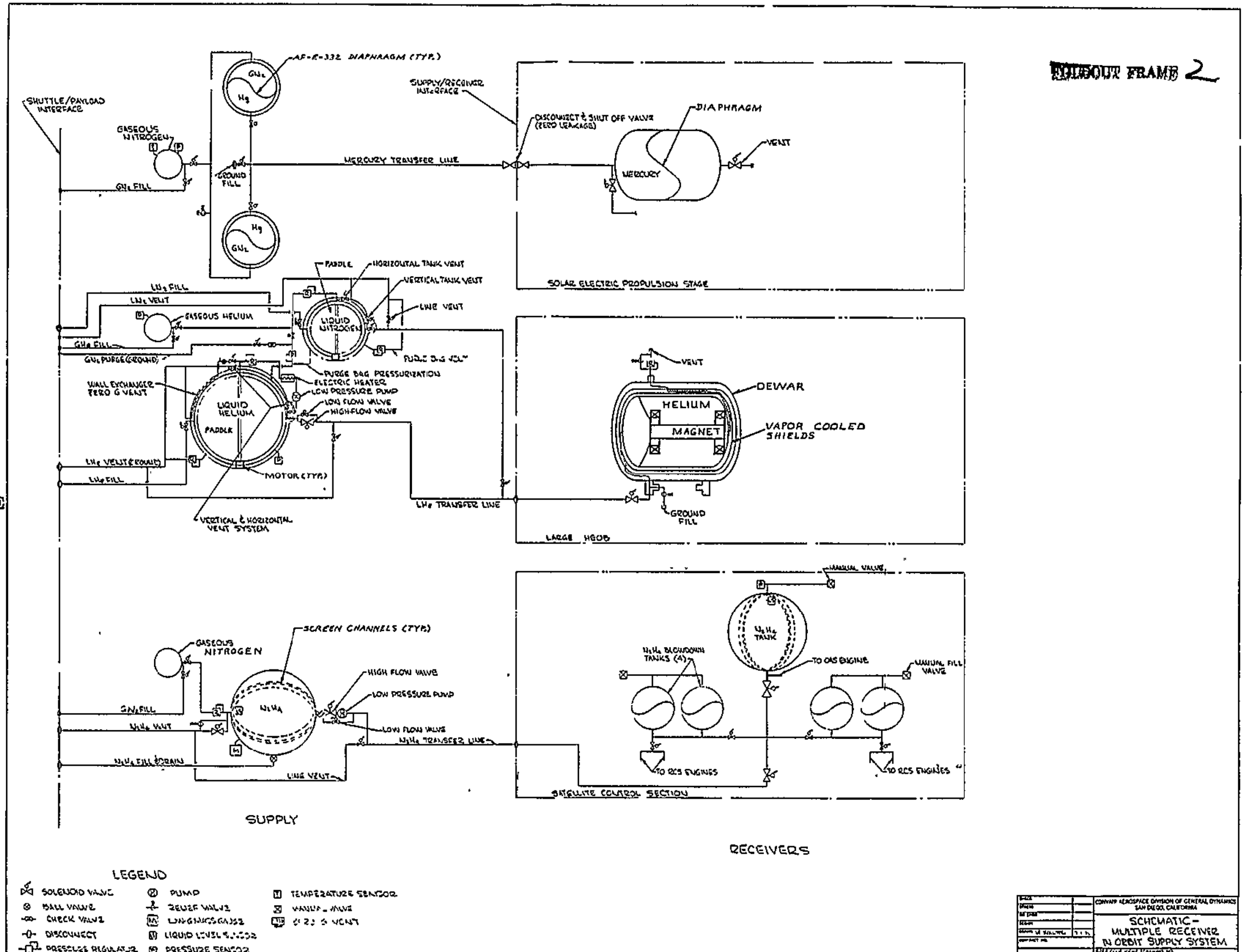

FIGURE 4-39. SCHEMATIC-MULTIPLE RECEIVER SUPPLY

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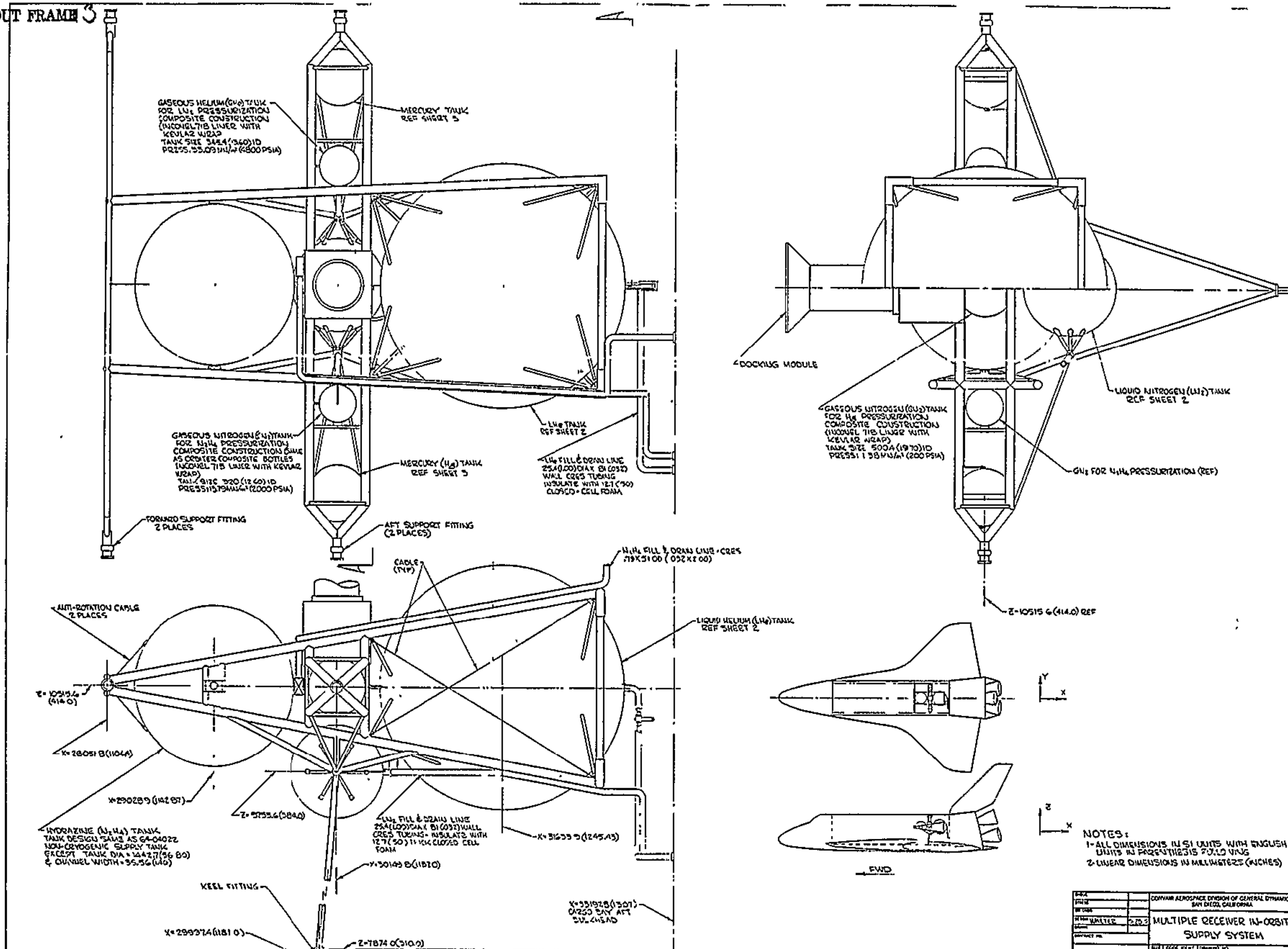

FIGURE 4-40. MULTIPLE RECEIVER SUPPLY

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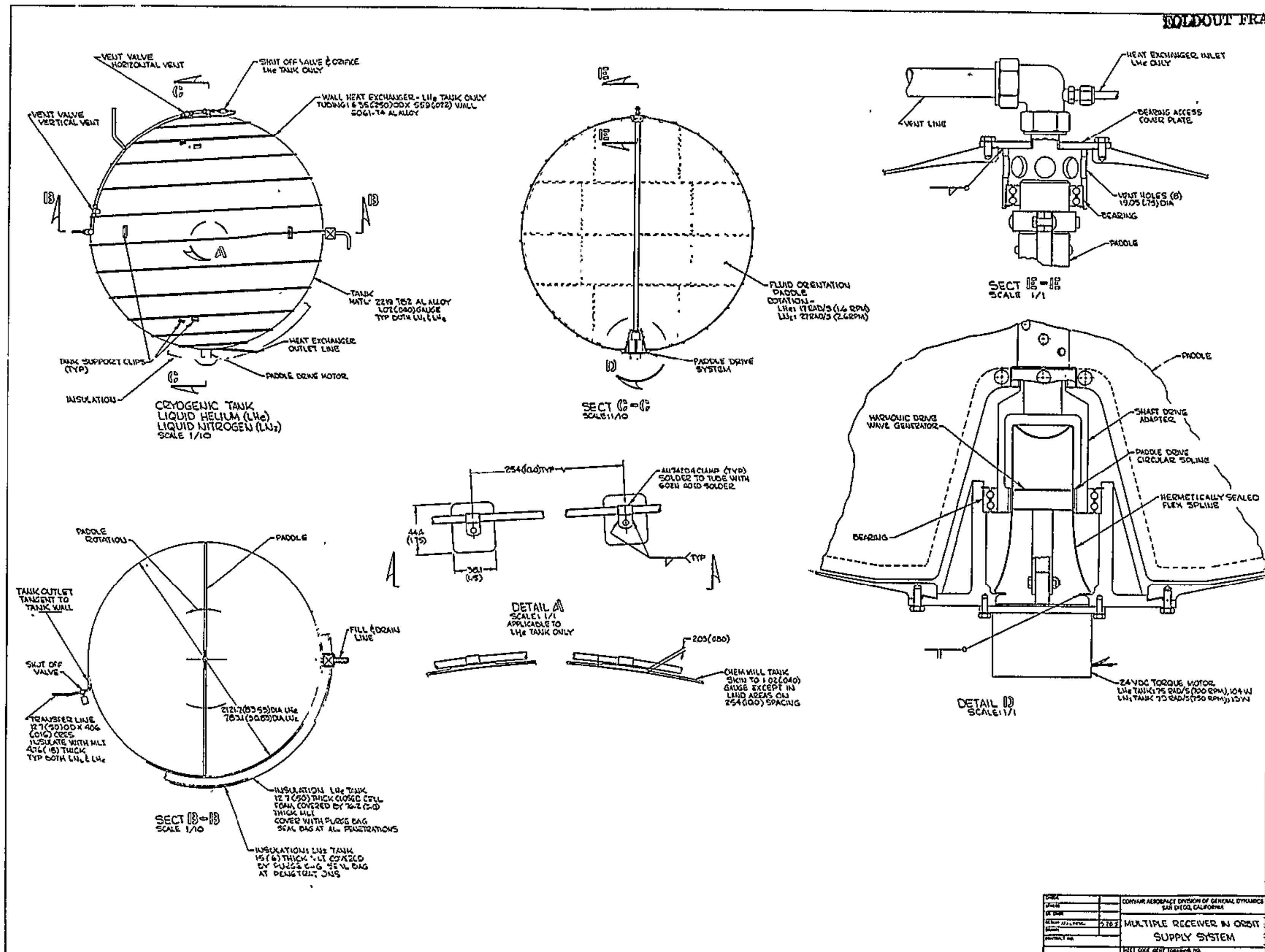

FIGURE 4-40. MULTIPLE RECEIVER SUPPLY-CONT.

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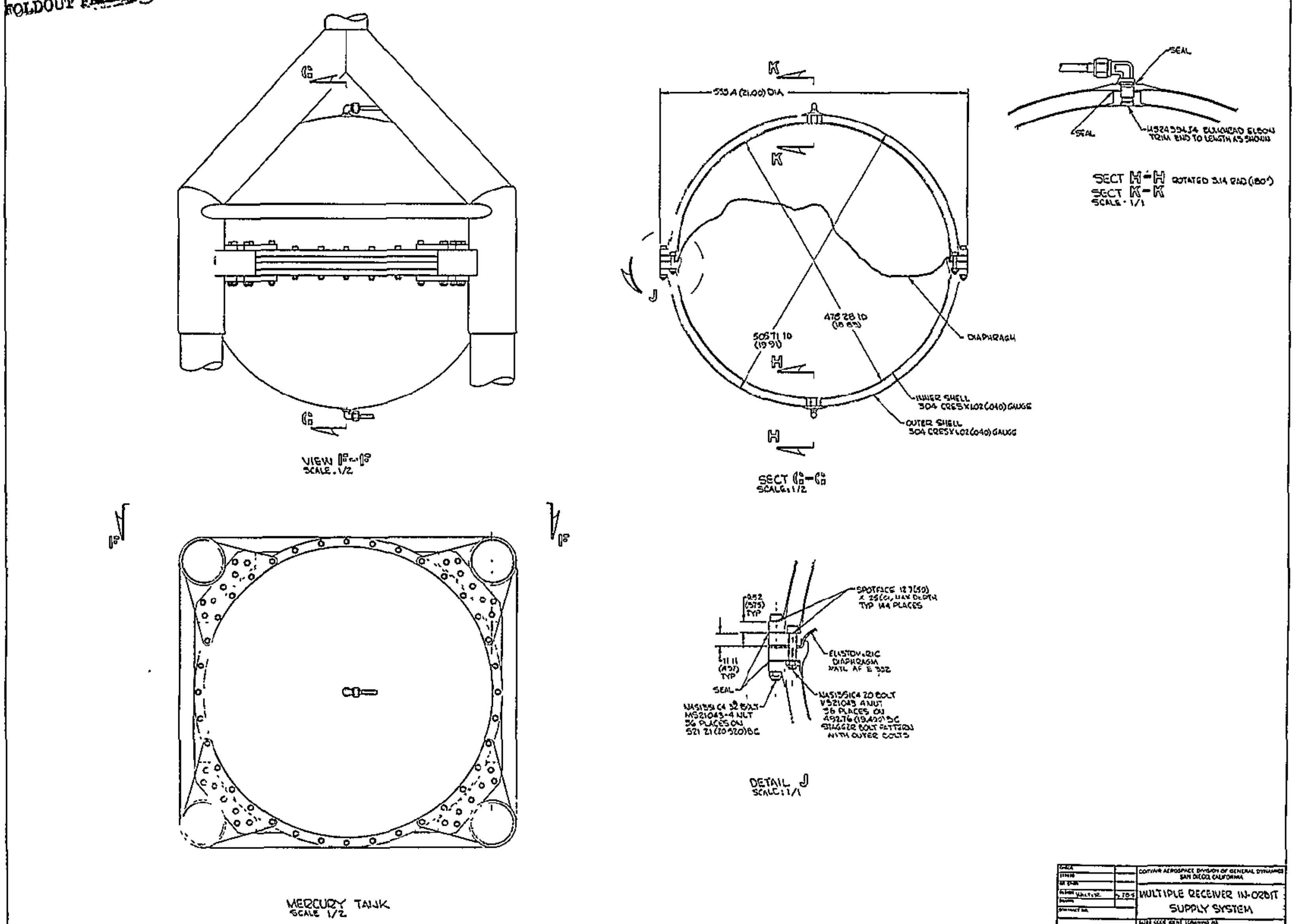

FIGURE 4-40. MULTIPLE RECEIVER SUPPLY-CONT.

as shown in Figure 4-39, to accomplish a slow transfer initially, until the receiver surface tension gallaries are full, following which, transfer is accomplished using a pump to charge the receiver tank to its final required pressure of 2136 kN/m^2 (310 psia).

GN_2 stored at 13.8 MN/m^2 (2000 psia) and used to 689 kN/m^2 (100 psia) resulted in a minimum weight system for supply tank pressurization.

The transfer line is 1.9 cm (0.75 in.) dia. × 0.08 cm (0.032 in.) wall × 560 cm (224 in.) long.

The overall supply module consists of a total of eight tanks supported in a welded aluminum tubular framework. The basic construction is of a main crossbeam of four tubes with five of the smaller tanks enclosed in the four-tube-beam. Two triangular longerons are attached to the main crossbeam with the two larger tanks, the liquid helium and the hydrazine, supported between the longerons. A smaller crossbeam attaches to the front of both longerons as a stabilizing beam. Support fittings for attachment to the Shuttle cargo bay are at the ends of both the main crossbeam and the stabilizing beam. A vertical triangular section, with a heel fitting is attached to the main crossbeam to absorb side loads imposed on the module. The liquid nitrogen tank is mounted in this triangular section by fiberglass struts. Fiberglass struts are also used to attach the liquid helium tank to the two longerons.

A docking module (for supply to receiver transfer) is attached to the center of the main crossbeam. A probe designed to mate with this docking module must be supplied on each receiver to be resupplied.

All of the tanks are removable directly from the supply module framework with the exception of the mercury tanks. These two tanks must be disassembled before they can be removed from the module. The aft members of the framework are removable so that the liquid helium tank can be removed.

The hydrazine tank is identical in construction to the MMH and N_2O_4 tanks in Paragraph 4.2.3.5 except here the tank diameter is 1443 mm (56.80 in.) and the channel width is 35.56 mm (1.40 in.). Also, two anti-rotation cables are attached to the tank to insure the tank does not rotate.

The cryogenic tanks, liquid helium and liquid nitrogen, both use a paddle system for fluid orientation. The LHe tank is 2122 mm (83.53 in.) in diameter and constructed of 1.02 mm (0.040 in.) gauge 2219 aluminum alloy. The internal paddle is also constructed of aluminum alloy. The paddle shaft is supported on both ends by bearings and is rotated by a 24 VDC torque motor. Power is transmitted from the motor to the paddle through a harmonic drive which reduces the angular velocity to 0.17 radians/second (1.6 RPM).

Fluid outflow is through a tangential outlet and shut-off valve. A fill and drain line is located in the tank wall opposite the tangential outlet. A vent line for horizontal operation is located at the upper end of the paddle. Vertical operation venting is provided by a vent on the tangential outlet line. The tank wall heat exchanger inlet is taken from the horizontal vent line at the outlet elbow.

The tank wall heat exchanger is spirally wound around the tank exterior with attach points located at 254 mm (10.0 in.) along the spiral and a spacing between spirals of approximately 254 mm (10.0 in). The heat exchanger attach points consist of raised lands [44.4 × 38.1 × 1.02 mm (1.75 × 1.5 × 0.4 in.], with tubing clips soldered to the heat exchanger tubing and welded to the wall lands.

The liquid nitrogen tank is similar to the LHe tank except it does not have a wall heat exchanger. The LN_2 tank is 783 mm (30.85 in.) in diameter and constructed of 1.02 mm (0.040 in.) gauge 2219 aluminum alloy. The paddle angular velocity is 0.27 radians/second (2.6 RPM).

The LHe tank is insulated with 12.7 mm (0.50 in.) thick closed-cell foam covered by 76.2 mm (3.0 in.) thick multilayer insulation with a semi-rigid purge bag enclosing the entire tank. The LN_2 tank is insulated by 15 mm (0.6 in.) thick multilayer insulation covered by a semi-rigid purge bag.

The mercury is stored in two tanks with each tank mounted as near to the support fitting on the end of the main crossbeam as practical to prevent excessive moment at 10g acceleration level. The tank construction is a double walled tank with a diaphragm for positive fluid displacement at zero-g. The diaphragm material is AF-E-332. The tank material is 304 CRES with a 1.02 mm (0.04 in.) skin gauge. Fittings are provided on each side for pressurization and for fluid outflow. These fittings also help hold the inner shell to the outer shell. Each of the inner and the outer tanks is made up of two hemispherical shells. The inner shells are bolted together with the diaphragm used as a seal between. The outer shells are placed over the inner shells and bolted together with additional seals to make the annulus between shells pressure tight. The addition of the inlet and the outlet fittings completes the attachment of the two tanks. These two tanks must be assembled in place inside the main crossbeam since the overall diameter of the assembled tank is larger than the spacing between tubes of the crossbeam.

The three ambient temperature gas bottles (two nitrogen and one helium) are all of composite construction; an Inconel 718 liner with a Kevlar overwrap. The helium bottle is 345 mm (13.60 in.) in diameter and is designed for 33.1 mN/m^2 (4800 psia) pressure. The largest of the two nitrogen bottles is 500 mm (19.70 in.) in diameter and is designed for 1.38 mN/m^2 (200 psia) pressure. The smallest nitrogen bottle is 320 mm (12.60 in.) in diameter and is designed for 13.8 mN/m^2 (2000 psia) pressure. These bottles all have mounting bosses for attachment to the main crossbeam.

4.2.4.5 Multiple Receivers Supply Operating Procedure - The supply module for this case has the capability to supply three different receivers; (1) Solar Electric

Propulsion stage, Hg, (2) Large High Energy Observatory B, LHe, and (3) Satellite Control Section, N_2H_4. All or any one or two of these receivers can be supplied on a single mission. For a given mission, only the supply tankage associated with the receiver(s) to be supplied are carried, except that, due to its mounting complexities, the Hg tankage is always carried. It is assumed that within each mission or Shuttle flight the receiver(s) are rendezvoused with and supplied one at a time.

A functional flow diagram of the basic overall supply operation is presented in Figure 4-41. Referring to Figure 4-41, it is noted that if more than one receiver is to be supplied, then the Rendezvous, Docking and Line Hookups, Fluid Transfer, and Disconnect and Undock operations will need to be repeated for each receiver. The various operations outlined in Figure 4-41 are expanded and discussed in the following paragraphs. A schematic of the complete transfer system is presented in Figure 4-39.

Supply Module Ground Preparation - At this point the LHe, LN_2 and N_2H_4 tankage and associated pressurization and purge systems are either installed or removed from the supply module, depending on which supply mission was last accomplished and which mission is being prepared for.

Following hookup of lines, the non-cryogenic liquids to be supplied are loaded into the supply module. A critical factor here is to insure that the surface tension screened channels in the N_2H_4 tank are completely filled with liquid, such that no non-condensible vapor is trapped within the channels. This is accomplished by providing a vent at the top of the channels and allowing liquid to flow out this vent before it is closed. Some ullage (approximately 3%) is provided in the remainder of the tank to allow for potential

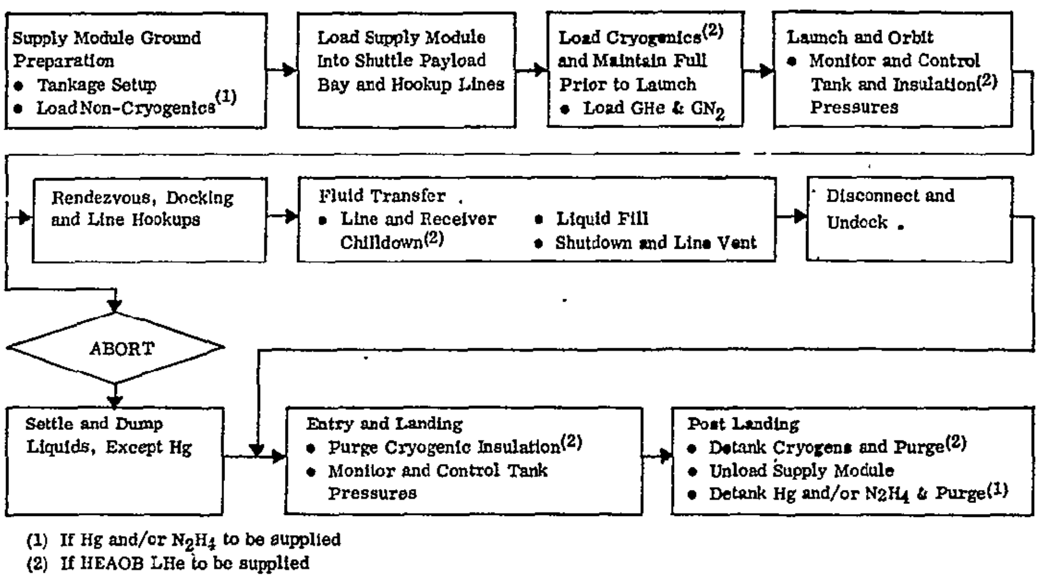

Figure 4-41. Overall Multiple Receiver In-Orbit Supply Operations

thermal expansion of the liquid. Also, mercury loading to avoid spills and/or leakage is critical from the standpoint of safety, since mercury is a highly corrosive fluid.

Load Supply Module Into Shuttle - This simply consists of the installation of the supply module in the Shuttle Orbiter cargo bay and the hookup of all ground supply, abort dump and in-orbit transfer lines.

Load and Maintain Cryogenics - At this point the cryogenics to be used, if any, are loaded into the supply module. This operation consists of initial line and tank chill-down and final loading. It is assumed that the maximum fluid loading is maintained by topping up to 180 seconds prior to launch. The high pressure helium and nitrogen storage bottles are also filled at this time. The cryogenic insulation for the LN_2 tank is maintained in a purged condition with GHe from the ground prior to boost. The LHe tank insulation is purged with GN_2 to minimize liquid boiloff from external heating.

Launch and Orbit - During this phase of the resupply operation the main problems are to control tank and insulation pressures and to prevent the surface tension channels in the N_2H_4 tank from drying out due to excessive liquid sloshing. Some venting of the LHe tank is allowed for during the boost phase. This occurs through the normal ground boiloff vent. Following entry into orbit, the zero-g vent is designed to accommodate any venting required. The LN_2 tank is designed to operate without venting. In both the LN_2 and LHe tanks pressure rise is minimized by use of the paddles to maintain mixed fluid conditions. Tank overpressure is a safety critical failure and provisions are made to allow immediate dumping of liquids as well as emergency venting; except that mercury is not dumped.

Abort (Only as Necessary) - If for some reason it becomes necessary to terminate the mission during boost or while in orbit, all liquids are dumped, with the exception of the mercury, which is an extremely hazardous fluid which would not be expected to dissipate completely in the Earth atmosphere. Liquid settling to the aft end of the Shuttle is provided by the normal Shuttle abort procedure. In the event that liquid dumping could not be accomplished, system design, with respect to c. g., is such that a satisfactory landing could still be accomplished.

Rendezvous and Docking - The main problem to consider here is maintenance of the N_2H_4 screened surface tension channels in a full condition, ready for liquid transfer at low-g. Even though accelerations are less than during boost, the direction may be random and completely unpredictable.

The receiver is docked directly into the supply module. The remote manipulator on the Shuttle is used for final mating and receiver stabilization during transfer.

Transfer line hookups will be accomplished upon docking or immediately thereafter. It is assumed that the lines are empty and vented to the atmosphere (vacuum) prior to hookup, except for the Hg.

Due to its extremely hazardous nature, the mercury is handled somewhat differently than the other fluids. In this case, both the receiver and supply lines are maintained full up to the supply/receiver disconnect. This disconnect is designed for zero leakage at all times, including periods of connection and uncoupling. The lines are not vented and for all practical purposes are part of the tankage.

Fluid Transfer - Specific transfer operations are described below for each of the three receivers involved.

a. Solar Electric Propulsion Stage (Hg) - Following connection of the supply and receiver lines some adjustment of mercury volumes in the supply and receiver tanks will occur automatically as the pressures in each tank equalize. Transfer is then accomplished by opening the GN_2 supply valves to pressurize the two Hg supply tanks causing transfer to occur. The receiver tank ullage is either vented or controlled to some desired back pressure below 345 kN/m^2 (50 psia).

b. Large High Energy Observatory B (LHe) - The following sequence of events is proposed:

   (1) Paddle rotation of the LN_2 is initiated.

   (2) The LN_2 tank is pressurized with helium gas.

   (3) The transfer line vent is closed.

   (4) LN_2 supply and receiver transfer line and receiver vent valves are opened and the receiver tank chilled to LN_2 temperature.

   (5) The LN_2 tank paddle is deactivated and the LN_2 supply valve closed.

   (6) Paddle rotation of the LHe is initiated, followed by start-up of the pressurant pump and heater, with subsequent helium tank pressurization.

   (7) Purging of the nitrogen from the receiver tank is then accomplished. This operation is performed by venting the receiver to 28 kN/m^2 (4 psia) [15.2 kN/m^2 (2.2 psi) above N_2 triple point] followed by pressurization to 83 kN/m^2 (12 psia) with cold helium for a total of 12 vent/pressurization cycles. LHe supply during this time is through the low-flow valve shown in Figure 4-39. It is noted that, even though LHe is supplied during the purging operation, it should vaporize in the relatively warm receiver.

   (8) Final chilldown of the HEAO B tankage to LHe temperature is accomplished at low flow while venting from the receiver through the zero-g vent located on the HEAO B tank.

(9) Following chilldown, the high-flow transfer valve is opened and the receiver filled. During this time, receiver venting is also accomplished through the zero-g vent located on the HEAO B tank. Such venting is necessary to maintain a minimum saturation pressure for maximum loading density.

(10) Following fill, supply and receiver valves are closed and the transfer line vented to vacuum.

c. Satellite Control Section (N_2H_4) - The primary problem here is to insure that the screen channels in the main receiver tank are full at the end of transfer. In the present case it is assumed that the channels are still full from their previous use or can be filled at the start of transfer. The following sequence of events is proposed.

(1) Actuate GN_2 pressurization of the supply tank.

(2) Open low-flow supply valve and receiver main tank inlet valves and fill receiver until liquid sensing instrumentation indicates that the receiver surface tension channels are full.

(3) Close low-flow valve, actuate transfer pump and open high-flow valve.

(4) Fill main receiver tank to desired amount while compressing the ullage to its desired use pressure of 2136 kN/m^2 (310 psia).

(5) Close inlet valves to receiver main tank and open inlet valves to the bladder tanks and fill to final use conditions.

(6) Close supply and receiver valves, deactivate transfer pump and vent transfer line.

Disconnect and Undock - The primary consideration here is the design of the transfer line disconnect for the mercury system for zero leakage under all conditions.

Entry and Landing - The primary operations here are to (1) control pressure in the insulation purge bags to slightly above ambient to prevent entry of air with subsequent moisture condensation and insulation damage, and (2) maintain tank pressures above ambient to prevent implosion and below the design pressure to prevent bursting.

Post Landing - Following landing of the Shuttle Orbiter, any remaining cryogens (LHe, LN_2) are detanked and the tanks purged with helium gas and locked up. The complete supply module is then unloaded from the Shuttle Orbiter and any remaining N_2H_4 and Hg drained. These tanks are then purged and locked-up with inert gas and the complete system stored. Prior to the next use of the supply module, the surface tension screens should be checked for cleanliness and cleaned if necessary, since contaminated screens could significantly reduce their available flow area.

4.2.5 OVERALL SYSTEMS WEIGHT SUMMARY - A weight summary of the various transfer systems is presented in Table 4-37. It is seen that the Tug supply system using shuttle drag is slightly lower in weight than the separated Tug supply using an auxiliary propulsion system. Thus, unless transfer in orbits higher than 296 km (160 n.mi.) were required, the drag system would be the likely choice for in-orbit supply of the Tug.

For both Tug supply cases, supply module weights are less than that of the baseline Space Tug being supplied. The dry weight of the Tug is 2338 kg (5150 lb) and the lift-off weight is 25778 kg (56779 lb). Thus, more payload can be accommodated with the transfer module than with the Tug. For example, a payload of 4089 kg (9006 lb) could be accommodated with resupply using Shuttle drag without RCS penalty, while the maximum payload with the Tug would be 3732 kg (8221 lb). Thus, with resupply, a basic payload increase of 9.5% is possible.

In all the cases considered the transfer efficiency (fluid supplied/lift-off weight) was quite high. The lowest efficiency was for the multiple receivers case and is due primarily to the low efficiency (38%) of the helium transfer. This low efficiency is primarily due to the high fluid losses associated with chilldown of the HEAO B receiver.

Table 4-37. Overall Weight Summary

	Space Tug Supply		Shuttle Orbiter Supply		Multiple Receivers (All Supplied)
	Shuttle Drag	Auxiliary Propulsion	Case 1	Case 2	
Dry Weight, kg (lb)	1600 (3524)	1641 (3615)	632 (1393)	947 (2087)	420 (926)
Supply Module Fluids, kg (lb)	23821 (52470)	23979 (52818)	28780 (63392)	28563 (62913)	3825 (8426)
RCS Propellant, kg (lb)	154 (340)				
Lift-off W/O RCS Penalty, kg (lb)	25421 (55994)	25620 (56433)	29412 (64785)	29510 (65000)	4246 (9352)
Lift-Off With RCS Penalty, kg (lb)	25576 (56334)	-	-	-	-
Fluid Supply/Lift-Off W/O RCS, %	92.1	91.3	96.5	94.5	81.0
Fluid Supply/Lift-Off With RCS, %	91.5	-	-	-	-
Total Fluid Residuals, %	0.86	1.1	1.0	1.3	1.9

5

SYSTEMS EVALUATION

This section describes the work performed to define technology requirements and programs necessary for final design and development of the specific transfer systems presented in Section 4. For each transfer system, identification is made of those components, characteristics, or operational procedures that are unique and/or critical to efficient and predictable in-orbit fluid transfer. Unique items are defined as those employed solely for low-gravity fluid transfer. Critical items are those vital to the functioning of the transfer system; i.e., having a direct effect on mission success or failure, transfer efficiency, weight, cost or reliability.

For each item identified, and using the results of the technology evaluation presented in Section 2, a determination is made of the need or desirability to perform further research and development studies. Included are brief descriptions of the specific work required.

Evaluations for each of the four systems conceptually defined in Section 4 are presented in the following paragraphs.

5.1 TUG SUPPLY USING SHUTTLE DRAG

This system utilizes maximum Shuttle Orbiter drag at 296 km (160 n.mi.) of 1.1×10^{-6} g's to orient LO_2 and LH_2 at tank outlets for transfer. The Orbiter vernier engines, providing 2.9×10^{-4} g's, are used for initial liquid settling and final scavenging of residuals. The LH_2 and LO_2 supply tanks are relatively long small diameter cylinders to minimize liquid residuals. The basic concept is illustrated in Figure 5-1. Detailed conceptual design data and operating procedures are presented respectively in Paragraphs 4.2.2.4 and 4.2.2.5. An examination of this data resulted in identification of the following items as being unique or critical to system design and performance.

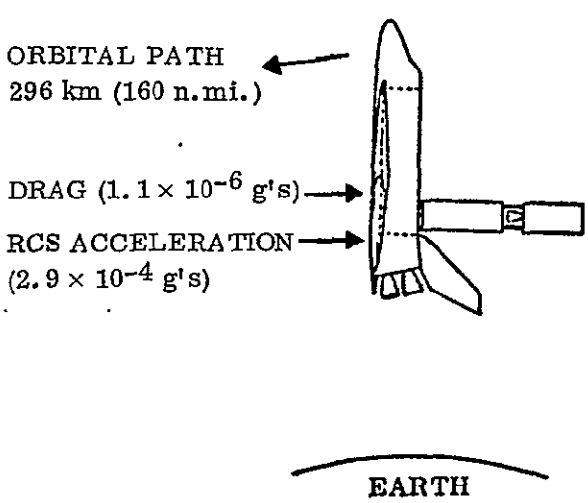

Figure 5-1. Tug Supply Using Shuttle Drag

a. Low-G Liquid Reorientation

b. Low-G Liquid Outflow

c. Receiver Childown and Fill

d. Tank Pressure Control Prior to Transfer

e. Tank Pressurization for Expulsion

f. Other Technology Items

The above items apply only to the transfer of LH_2 and LO_2. The quantity of hydrazine transferred is relatively small and a bladder, considered to be current state-of-the-art, is used. In the case of helium transfer, a simple blowdown system is used, and there are no anticipated problems. Before final use of this concept there would, however, need to be an evaluation of the Space Tug hardware to insure compatibility with the initially cold helium being transferred. If necessary, existing valves could be replaced with ones capable of operation at LH_2 temperatures.

Detailed evaluations of the technology items listed are presented in the following paragraphs. For weight evaluations, supply system weight savings and penalties are converted to potential payload increases or decreases based on a total Shuttle payload capability of 29,510 kg (65,000 lb) and a baseline supply module weight of 25,421 kg (55,994 lb). Assuming the 154 kg (340 lb) of RCS propellant, estimated to be required for initial settling, scavenging and station keeping, is payload chargeable, then the maximum baseline useable payload is 3934 kg (8666 lb).

5.1.1 LOW-G LIQUID REORIENTATION - For the current case the drag used to orient liquid for transfer may not be sufficient to initially settle the liquid (Bo_N less than 0.84, Paragraph 4.2.2). Calculations presented in Paragraph 4.2.2 indicate a maximum RCS propellant requirement of 66.7 kg (147 lb) for settling. This assumes that the RCS thrust is continuous over the entire settling time. Bond numbers with RCS thrusting, based on tank radius, are respectively 178 and 149 for the H_2 and O_2 tankage. Under these conditions, existing technology is adequate (Ref. Paragraph 2.1.4) and the current prediction of settling requirements should be close to the maximum which will occur in actual operation.

Assuming the RCS fluids to be payload chargeable these settling requirements translate to a payload penalty of 1.7%. This would be the maximum weight saving or payload increase, assuming technology in this area could be advanced to the state where settling propulsion requirements would be negligible.

This could possibly be accomplished by employing impulsive type settling. That is, the RCS engines would be operated for a short time to start the liquid moving toward the outlet, following which, drag would complete the settling.

Work has not been sufficient to allow a determination of settling requirements for the current in-orbit fluid transfer application.

Analytical modeling of impulsive settling phenomena has also been attempted using the Convair Marker and Cell (MAC) computer program. This work, also as yet unpublished, was accomplished under Contract NAS3-17839. It was shown that, with improvements, the MAC model should be capable of handling impulsive settling cases.

The work required to bring this technology to the point that impulsive settling requirements could be accurately predicted for the in-orbit fluid transfer case is outlined below.

<u>Task I - Drop Tower Testing</u>

Testing would be accomplished to develop empirical correlations to allow a determination of impulsive thrust times required for liquid settling sufficient to initiate outflow. The testing should cover thrusting Bond numbers of 100 to 200, followed by simulated drag at Bond numbers on the order of 0.2 to 0.8. As used here, Bo_N is based on tank radius. Tank outlets should be hemispherical and tank length to diameter ratios of 1 to 10 should be covered. Testing should also include various liquid quantities from 10 to 95% full.

<u>Task II - MAC Model Development</u>

The MAC computer technique would be useful to verify the design of full size transfer systems under specific operational conditions. To develop the model to this state would require further work to improve its computational stability at low-g. Computer runs should then be made and compared with drop tower results to verify the accuracy.

<u>Task III - Final Settling System Verification and/or Refinement</u>

This could be accomplished during the initial operational or demonstration flights of the in-orbit fluid transfer system. Sufficient excess settling propulsion fluids could be carried during these first flights to insure mission success, even if settling predictions were somewhat in error.

5.1.2 LOW-G LIQUID OUTFLOW - This technology is concerned with the determination of liquid residuals at the point where vapor breaks through into the outlet during draining at low-g. The basic process is illustrated in Figure 4-5.

For overall system definition, pull-through residuals must be determined at two basically different conditions. In the first case, outflow is at very low rates with liquid orientation by drag-only. Bond numbers are 0.7 and 0.6 for H_2 and O_2 respectively. Calculation of pull-through is required to determine when the RCS engines must be activated and for how long. There is, at present, no published data covering this Bo_N range. Some, as yet unpublished data, has been generated at the NASA/LeRC from drop tower tests and computer analyses. The data indicates that residuals in this intermediate Bo_N range can be made very low by proper contouring of tank outlets. The applicable correlating parameter for liquid pull-through height appears to involve both the Bo_N and the We_N [$h \approx We_N/(\text{const.} + Bo_N)$].

For determining final residuals, where the RCS engines are operating, less uncertainty exists, since Bond numbers are 178 and 149 for H_2 and O_2 respectively. More data are available in this Bond number range. In this case the liquid-vapor interface is fairly flat and tests at 1-g in small tanks with Froude number as the correlating parameter should be reasonably accurate. However, even in this case, further work is needed. For example, residuals could likely be reduced by improved outlet contouring and/or baffling, and criteria for the design of such systems is needed.

The overall importance of doing further work in this area is illustrated by the data presented in Paragraph 4.2.2.2. As shown in Figure 4-18, total system weights vary considerably, depending on whose data is used for calculating liquid residuals. For example, assuming transfer at 296 km (160 n.mi.) with scavenging and RCS propellant usage as a payload penalty, the minimum total system weight is 25,530 kg (56,240 lb) using the NAR residual calculation procedure. For the same conditions, using the LeRC data, minimum total systems weight is 25,430 kg (56,020 lb). This represents a payload difference of 2.5%. In addition, the optimum transfer time is only 15 hours with the LeRC data versus 30 hours using the NAR data. At a common transfer time of 20 hours the weight difference is 114 kg (250 lb) or 2.9% of payload. At this same transfer time, assuming elimination of use of the RCS engines, the weight difference is 145 kg (320 lb) or 3.7% of payload. Differences are even greater where shorter transfer times are desired.

It is noted that the uncertainties described above do not necessarily cover the full range of possibilities. There is no certainty that the NAR and LeRC data bracket the actual case. For example, liquid-vapor interface shape and stability under dynamic draining conditions have not been covered. The basic calculations made, assume that static interface configuration data apply.

It is estimated that further work in this area could result in payload increases on the order of 6% over designs based on current knowledge, as well as reduce transfer time and possibly eliminate the need to use the RCS system for scavenging. Following is an outline of recommended work.

Task I - Drop Tower Testing

Testing would be accomplished primarily in the intermediate Bond number range of 0.1 to 10.0 to provide basic data on the effects of tank outlet shapes, baffles, screens, pressurant inflow patterns, outflow rate and ratio of outflow line diameter to tank diameter. The effects of up and down flow throttling and changes in acceleration level would also be investigated as much as possible. However, the bulk of the testing would be to provide basic pull-through data for residual determination and for comparison with an analytical model designed to cover the full range of outflow conditions.

Task II - Analytical Model Development

This could be in the form of the Marker and Cell (MAC) technique or other model. The model should be capable of handling variations in tank outlet shape, baffle and screen configuration, outflow rate, and up and down throttling of outflow and settling acceleration. The capability to handle different ratios of outflow line diameter to tank diameter should be included. Computer runs would be made and compared with drop tower results to verify operation at specific conditions.

Task III - Ground Tests

These tests would be designed to obtain data required for the design of tank outlet baffling or contouring or other pull-through suppression devices when Bond numbers are on the order of 100 to 300, such as when the RCS engines are operating. Froude number correlations should be applicable here.

Task IV - Orbital Demonstration Testing

An orbital experiment would be required for final system demonstration. This could be a subscale system designed to verify the basic design calculations and pull-through method(s) developed in Tasks I, II and III.

5.1.3 RECEIVER CHILLDOWN AND FILL - Control of receiver tank pressure without excessive loss of the transfer fluid is critical to transfer system success. In the present case, some settling is provided, which may, with proper inlet flow control, be sufficient to prevent the direct loss of liquid. In addition to unknowns associated with direct liquid loss there can be a significant variation in the fluid required for chilldown of the H_2 and O_2 receiver tanks just due to differences in the thermodynamic efficiency of the fluid vented. This is discussed in Paragraph 4.1.

For the present case, assuming no liquid loss, the difference in supply system weight between minimum thermodynamic vent efficiency (venting saturated vapor) and maximum vent efficiency (venting superheated vapor at tank wall temperature) is 66 kg (145 lb) and 21 kg (46 lb) for H_2 and O_2 systems respectively. The total difference in terms of payload is 2.2%. The main uncertainty is with the H_2.

Assuming use of the existing Tug inlet lines, the Bond numbers for receiver filling, based on inlet line radius, are respectively 0.002 and 0.004 for H_2 and O_2 with drag-only applied and 0.54 and 1.1 with the RCS engines operating. Corresponding inflow Weber and Reynolds numbers are presented in Table 5-1 for inflow rates over the full 10:1 throttling range.

Table 5-1. Tug Receiver Inflow Parameters

	Bond Number		Weber Number		Reynolds Number	
	Max.	Min.	Max.	Min.	Max.	Min.
Hydrogen	0.54	0.002	5	0.05	36,850	3,685
Oxygen	1.1	0.004	2	0.02	16,200	1,620

Technology work accomplished to date on low-g receiver tank pressure control and/or vent systems has been limited,and consists of the following.

a. Development of an analytical model applicable to cryogenic fluids and settled liquid inflow patterns.

b. Some verification of the above model using LN_2 and LH_2 at one-g. However, testing was performed only over the first few seconds of receiver tank chilldown and only for locked-up tank conditions. Also, correlation with the model was poor when inlet baffling was used.

c. Investigation of mechanical and heat exchanger type systems for separating liquid from vapor at the vent. Prototype one-g test hardware have been built for application to cryogenic storage tank pressure control. However, since vent rates for these systems are much lower than desired for receiver chilldown, the specific hardware fabricated would not be directly applicable unless long chilldown times could be tolerated.

d. Drop tower testing at NASA/LeRC. Most of this testing was at near zero-g ($Bo \cong 0$) and designed to determine the conditions for stable inflow to prevent liquid from geysering to the top of the tank. Investigations were also accomplished on various inlet baffle configurations and vent locations designed to prevent liquid from exiting the tank while venting during inflow.

   Both the 2.2 and 5 sec drop facilities have been used. All published data are for non-cryogenic testing. These tests indicated a potential for specific inlet design and vent location to minimize liquid loss.

   Most data indicated that Weber numbers below about 1.3 to 1.5 should result in stable inflow, depending somewhat on the initial liquid loading. The criteria for stability is, however, somewhat in question. Tests have been with both cylindrical and spherical tankage.

   Only one series of tests were accomplished where a controlled amount of acceleration was applied (Reference 5-1). In this case,flow was into an initially empty cylinder with hemispherical bottom and no baffling. Jet height during inflow was correlated as a function of Weber and Bond numbers, respectively, ranging from 3.4 to 27 and 0.06 to 2.8. These values are based on inlet line radius. Inlet Reynolds numbers ranged from 1415 to 9870. Comparing with the Table 5-1 data shows that the drop tower tests did not cover a sufficiently wide range of parameters. Extrapolating the correlations to the lower Bond numbers of the current transfer case indicated that liquid could impinge on the top of the receiver tanks. Comparing the baffled versus non-baffled inlets where testing was at $B \cong 0$ indicated that by use of baffles, impinging of liquid at the vent could be eliminated. However, this data is only qualitative and further work would be needed for verification.

Cryogenic tests, including heat transfer effects, are currently being accomplished in a 0.6 m (2 ft) diameter CRES spherical tank with LN_2, using the 5 second drop facility.

In summary, no practical system has currently been developed and/or demonstrated to insure efficient receiver tank pressure control during chilldown and/or fill at low-g.

One way which has been proposed to eliminate the liquid/vapor separation problem of venting is to accomplish chilldown and fill with a locked-up receiver tank. However, analysis shows that tank design pressures would be exceeded during chilldown if venting of the Space Tug H_2 tank was not accomplished.

Even under conditions where non-vent filling can theoretically be accomplished for all or part of the transfer, further work is needed to define inflow systems which promote liquid/vapor mixing and which minimize wall to fluid heat transfer in relation to inflow rate, as required to control receiver tank pressure rise.

In the case of combination vent and lock-up systems, as proposed in the current study, means must be developed to determine when the receiver has cooled sufficiently to allow lock-up.

An additional problem which must be considered in filling subcritical tanks, such as are aboard the Space Tug, is that by allowing the saturation pressure to rise during locked-up fill the effective liquid density is reduced, resulting in the loading of less liquid than on the ground or if venting were accomplished. One approach is to use the zero-g vent system which is assumed to exist on the receiver to maintain a low saturation pressure near the end of fill. This must be further investigated with respect to fill times and specific receiver tank vent systems available. Also, use of this type vent tends to insure a low thermodynamic efficiency; i.e., saturated rather than superheated vapor venting.

Following is an outline of the tasks considered necessary to develop the required technology for efficient receiver tank pressure control for the Tug.

Task I - Analytical Model Development

Model(s) should be designed to allow calculation of fluid flow conditions and pressures within a receiver tank during low-g chilldown and fill of cryogenics (O_2, H_2) under both vent and non-vent conditions. The model(s) should be capable of handling cases where a heat exchanger type vent exists in the receiver tank. Flow and heat transfer modeling should be such as to allow use of the cryogenic and noncryogenic data generated from drop tower testing at the NASA/LeRC.

Task II - Drop Tower Testing

It is recommended to complete the LN_2 tests presently in progress at the LeRC to gain an insight into the low-g thermodynamic effects of chilldown. Further inflow tests should also be accomplished with cylindrical tanks with hemispherical bottoms at low-g (Bo_N = 0.002 to 2.0) to determine the quantitative effects of baffles on jet rise during inflow. This data should then be used to refine the analytical model(s) developed in Task I.

Task III - One-g Thermodynamic Testing

One-g testing should include inflow testing to at least three different size tanks to determine the effect of tank L/D on chilldown efficiency. Various inflow nozzle configurations should also be tested to determine the optimum method(s) for both locked-up and vented receiver chilldown. Existing receiver tank hardware must also be considered here.

Task IV - Systems Definition

Using the above model(s) and one-g and drop tower test data, analyses should be performed to define weight and performance of the most promising concepts for accomplishing low-g receiver tank pressure control. Both passive (inlet flow control) and active (liquid/vapor separation at the vent) type systems should be considered. The actual hardware which may exist at the Tug outlet (inlet for transfer) must also be considered in any final system design. Weight, reliability and performance comparisons would then be made between the various systems to determine the one(s) worthy of further development.

Task V - Prototype Demonstration Testing

Detailed design, fabrication and one-g testing of the most promising concept(s) would be accomplished under this task. This system should be capable of final testing at low-g in an orbital experiment.

Task VI - Orbital Experimentation

Final demonstration testing of the most promising receiver pressure control concept(s) must be accomplished by orbital experimentation. For proper scaling to the Tug, test tank sizes up to 1.8 m (6 ft) diameter by 3.7 m (12 ft) long should be employed. Due to the large expense involved in orbital experimentation, preliminary ground testing should be accomplished to develop reliable instrumentation and measurement techniques.

Other Technology Work

Other associated basic technologies are listed below in which further work would advance the overall state-of-the-art of low-g receiver tank chilldown and fill.

a. Low-g Boiling Heat Transfer in Receiver Tank. Both film and nucleate boiling are important, with film boiling being the most important. Heat transfer to an advancing liquid front (quenching) as well as droplet boiling phenomenon should be considered.

b. Transfer Line Chilldown and Two-Phase Flow. A knowledge of transfer line chilldown time and the condition of the fluid entering the receiver tank is required to improve predictions of receiver tank conditions during chilldown.

c. Low-g Condensation Heat Transfer. Information on this technology would be useful for complete definition of temperatures and pressures in the receiver tank during both chilldown and fill.

5.1.4 TANK PRESSURE CONTROL PRIOR TO TRANSFER - The normal method of pressure control for cryogenic tankage is to vent the excess vapor generated by external heating. The primary problem here is that prior to transfer, at low-g, the orientation of liquid and vapor is unknown and special means are required to insure that only vapor is vented. Also, the need and/or time schedule for venting is uncertain due to inadequate means of predicting pressure rise rates at low-g.

There are a number of analytical models available to predict pressure rise in cryogenic tanks. However, they are generally limited to settled or flat liquid-vapor interface shapes, and most all of the testing done to-date has been at one-g. The low-g data available is mostly qualitative and attempts to correlate existing analytical models with this data have not been conclusive. Using what low-g data is available, a statistical correlation was developed for pressure rise rate in a locked-up tank as a function of fluid mass, ullage volume and external heating (Ref. 5-2). Comparing the correlating equation with the actual data used in the correlation results in a scatter between predicted and actual rise rates of + 89/-35%.

With respect to hardware, a prototype bulk heat exchanger system (Figure 2-5) has been demonstrated at one-g to control H_2 and O_2 tank pressures while venting 100% vapor, even though surrounded by 100% liquid. This system should be capable of operating at low-g without any major problems. The main disadvantage is that a mechanical mixer is required in addition to the exchanger to insure thermal mixing of the fluid at the exchanger with that in the ullage. In order to simplify the concept, some analysis and one-g testing has been accomplished on a distributed wall exchanger type system designed to eliminate the mixer. The problem with this system is that operation is dependent on natural convection type heat transfer and feasibility demonstration would require orbital testing.

The Tug supply module illustrated in Figure 4-25 employs a bulk heat exchanger type vent system which should be capable of meeting all mission requirements. Further technology work in this area should be designed to simplify the overall low-g pressure control system. Several possibilities are listed below.

a. Venting without mixing.

b. Mixing without venting.

c. No mixing or venting.

Advantages and technology requirements for each of the above possibilities are discussed in the following paragraphs.

5.1.4.1 Venting Without Mixing - The advantage here would be to simplify the vent hardware. The next step in the development of a distributed wall type exchanger system without a mixer would be orbital experimentation to demonstrate feasibility. The next step in the development of the bulk type system with mixer would be orbital demonstration of a flight operational system.

An additional possibility for eliminating the mixer is to vent when the liquid is known to be away from the vent. This would be at low rates, as required to prevent bulk liquid boiling and liquid entrainment in the vent. Work in this area is reported in Reference 2-12. Further technology work required is discussed in Paragraph 2.1.11.

5.1.4.2 Mixing Without Venting - Calculations show that with complete liquid-vapor mixing, no venting of the Tug supply module should be required. Accurate calculation of mixed pressure rise is straightforward and current state-of-the-art.

Based on nominal pressure rise calculations without mixing, the present baseline supply system vents 31 kg (68 lb) of H_2 and 0 lb of O_2 prior to transfer. Taking account of uncertainties in present pressure rise calculations, a boiloff allowance of at least 51 kg (113 lb) of H_2 and 21 kg (47 lb) of O_2 would be required. Thus, eliminating venting could save 73 kg (160 lb) of boiloff or 83 kg (182 lb) total, including storage weight. This translates to 2.1% of payload.

There has been a significant amount of one-g testing accomplished to-date for the case where a jet type mixer is located in the liquid, as shown in Figure 5-2.

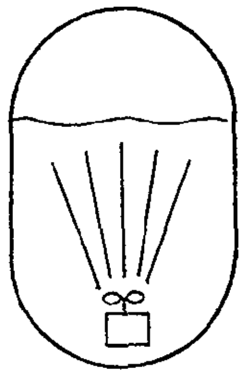

Figure 5-2. Jet Mixing With Mixer in Liquid

However, a comprehensive correlation of the data with an analytical model has not been accomplished. Test data using LH_2, LO_2, water and Freon 113 are available. Some drop tower tests were also accomplished to measure the jet flow criteria for providing complete liquid circulation within a spherical tank.

Where it is desired to minimize the pressure rise in a tank by mixing, it is also important to determine the mixing requirements when the mixer is encapsulated in the vapor. One proposed approach to the problem has been to locate a mixer at each end of the tank such that one is always in the liquid. Such a solution does not cover all liquid orientations and adds hardware to the system. In general, the case where the mixer is located in vapor has not been well explored. Current one-g data show that with the mixer in vapor, pressure rise rates are not reduced over that of the unmixed case. Some drop tower data (References 5-3 and 5-4) are available on gas jet impingement requirements for breaking up a liquid surface at low-g. This data indicates that velocities from jet mixers which have been tested in liquid at one-g are on the order of that required to break up the liquid surface at low-g. A significant extrapolation of the test data is however required since the maximum test jet Weber and Reynolds numbers are respectively on the order of 6 and 2500 while a typical mixer jet would have Weber and Reynolds numbers on the order of 300 and 150,000.

The information required with the mixer in vapor and in liquid is concerned with power and flow rates required to mix or destroy temperature stratification as well as that required to break the liquid/vapor interface. Also, the time to accomplish mixing as a function of power and flow rate is important. A program to provide design data necessary prior to orbital testing of a flight system is outlined below.

a. Develop an analytical mixing model and correlate with all existing data; both one-g and drop tower. The basic data available is with the mixer located in the liquid at one-g, however the model should have the basic capability to handle both liquid and vapor mixing cases at low-g.

b. Perform thermal tests simulating a mixer in vapor to determine the degree of liquid surface agitation required to accomplish destratification and reduce pressure rise rates as a function of jet/interface orientation and ullage volume. Such tests could be accomplished at one-g.

c. Perform drop tower tests to determine flow patterns and conditions for liquid breakup and/or mixing for conditions simulating a mixer located in the vapor.

d. Using the data generated in b. and c., incorporate vapor mixing capability into the analytical model developed in a.

5.1.4.3 No Mixing or Venting at Low-G - This of course would be the simplest system. As mentioned in the previous paragraph, for the nominal case, no O_2 venting is required even without mixing. In the H_2 case, low-g venting could be eliminated by venting during boost. The main problem, however, is that the accuracy of current pressure rise predictions are not now good enough to allow the elimination of both mixing and zero-g vent systems.

Further work in this area would be to accomplish orbital testing sufficient to significantly improve the accuracy of low-g pressure rise predictions.

5.1.4.4 Other Associated Technologies - Other basic technologies are listed below in which further work would advance the overall state-of-the-art of low-g tank pressure control.

a. Low-g convection, boiling and condensation heat transfer knowledge would enhance the prediction of pressure rise in a locked-up tank at low-g. Such information would also be useful in predicting the performance of vent systems designed to operate without forced mixing.

b. Data on forced flow boiling and condensation at low-g would be useful to the design of heat exchanger vent systems. This information would allow further weight optimization of such systems and increased confidence in operation at low flow rates.

5.1.5 TANK PRESSURIZATION FOR EXPULSION - There has been a significant amount of work done on this subject for application to propellant feed of space vehicle main engines. In this case the liquid is settled and operation is similar to expulsion of liquids at one-g. Based on a flat liquid surface during outflow, existing analytical models can predict pressurant requirements within ± 10%. It is not expected that further work in this area would result in any significant improvement in this.

The Tug supply case using Shuttle drag is, however, unique when compared to the engine start cases, in that much lower accelerations are involved in the transfer. During drag, which exists over the major portion of the transfer, the settling Bond numbers, based on tank radius, are only 0.68 for the H_2 and 0.57 for the O_2. This results in a significantly curved liquid-vapor interface during draining. Also, under these conditions, instabilities of the liquid surface could possibly occur due to non-steady draining of liquid from the wall. As an example of the potential effect of liquid sloshing and/or a non-flat liquid surface, test data has shown that mechanically shaking a tank to cause liquid sloshing can result in up to a 100% increase in pressurant requirements.

Another unique characteristic of the Tug transfer operation is that transfer times are significantly longer than for engine feed. This increases the possibility of heat transfer between the pressurant and the liquid, tank walls, and any internal hardware which may be present.

The importance of obtaining data on pressurant requirements for low-g draining over long expulsion times is basically a weight consideration. The storage of high pressure helium as used in the current system requires fairly heavy storage bottles and thus total system weight is quite sensitive to anticipated helium requirements. An estimate must also be made of the amount of liquid expected to be vaporized during transfer since this must be included in the initial fluid loading and tankage sizing.

Based on current uncertainties, calculations showed a total potential system weight penalty of 136 kg (300 lb) of helium system and 59 kg (130 lb) for vaporized liquid. This represents a fairly high payload differential of 5%.

In order to improve predictions the following work is recommended.

<u>Task I - Analysis and Model Development</u>

Under this task an analytical model would be developed to predict pressurant requirements over long transfer times when the liquid-vapor interface is significantly curved, as at the low Bo_N's involved here. Calculations would then be made to bracket the potential helium and liquid vaporization requirements more closely than can presently be accomplished. It is likely that modification of existing model(s) would be satisfactory.

<u>Task II - Orbital Testing</u>

In order to reach a final conclusion as to pressurant requirements, orbital testing will be required. This testing may be accomplished during the first demonstration flights of the operational supply module by carrying extra helium bottles to meet worst case contingencies. Following these demonstration flights, some of the helium bottles could likely be removed to allow an increase in payload.

<u>Other Technology Work</u>

Other basic technologies are listed below in which further work would advance the overall state-of-the-art of tank pressurization.

a. Low-g convection, boiling and condensation heat transfer data would improve the prediction of pressurant requirements and liquid vaporization during low-g transfer.

b. Further data on interface stability at low Bond numbers (below 1.0) during draining would be useful to gain insight into possible surface disturbances which could effect pressurant requirements and liquid vaporization.

5.1.6 OTHER TECHNOLOGY ITEMS - Other items which are unique and/or critical to efficient and predictable in-orbit fluid supply of the Tug are discussed below.

Low-g Mass Gauging

Mass gauging is required in both the Tug receiver and supply tankage. Accuracies similar to those obtainable on the ground are desired for the receiver, since the Tug flight performance is based on an optimized propellant loading. Any compromise in accuracy would result in a corresponding reduction in potential Tug performance. In the case of the supply module, low-g gauging is required for determining if sufficient liquids have been dumped for emergency re-entry, initiating and controlling flow throttling and auxiliary acceleration during transfer, and determination of fluid depletion.

The state-of-the-art associated with this technology and some further work required are discussed in Paragraph 2.3.2.1

One factor peculiar to the Tug supply case is that some settling acceleration exists during transfer. Thus the use of systems normally applicable to completely settled liquid could be considered. According to Reference 4-4, those such as point level sensors and capacitance gauges would not be satisfactory due to distortion of the liquid meniscus at the g-levels involved here. However, a system described in Reference 4-4 could possibly be used. This concept consists of wire elements (0.3 to 0.9M, 1 to 3 ft long) strung vertically between two end supports. A small electrical current is applied and the resistances of the wires are a direct analog of the liquid level. No meaningful development work has been accomplished on this system to-date. Further work would be required to choose the optimum system for the current application and to then develop and demonstrate the system at low-g in orbital tests.

Liquid-Vapor Sensing

The two primary applications for liquid-vapor sensing are listed below.

a. Detection of liquid at the receiver vent. Assuming liquid is being lost it may be possible to modify fill and/or vent conditions to eliminate this loss. This could possibly be by reducing the inlet flow rate or by momentarily shuting-off the inflow and/or closing the vent to allow liquid build-up at the inlet to retard geysering.

b. Detection of vapor at the supply outlets. Assuming premature ingestion of vapor, corrective action could possibly be taken by reducing outflow or actuating the

RCS engines ahead of schedule.

As discussed in Paragraph 2.3.2.2, further work is needed to develop and demonstrate satisfactory liquid/vapor sensing where low-g is involved.

Low-G Acceleration Measurement

The need for further work in this area would be associated with system testing prior to operational flights and in determining if settling accelerations during flight are of the magnitudes and directions required for satisfactory transfer. For the low accelerations involved here (below 10^{-5} g's) it is anticipated that further development work will be required (Reference Paragraph 2.3.2.6).

Purging and Safing

As described in Paragraph 4.2.2.5 it is necessary to expel all cryogenics, especially H_2, prior to landing. This is assumed to be accomplished following liquid transfer and prior to Shuttle re-entry. Due to the low-g environment the complete expulsion of liquid and purging of tanks and lines will require further investigation. For example, care must be taken to insure that freezing of O_2 and/or H_2 does not occur while venting the lines to vacuum.

Transfer Line Childown

The overall transfer time will be influenced by fluid conditions in the transfer lines and the time for the lines to childown to liquid temperatures. Based on the state-of-the-art as discussed in Paragraph 2.3.1, orbital testing is needed to determine flow regimes and how they relate to heat transfer and line childown at low-g.

5.2 SEPARATED SPACE TUG SUPPLY

The basic supply module configuration here is similar to that of the Shuttle drag system. The difference is that in the present case the supply module and Tug are separated from the Shuttle and liquid settling and orientation is with auxiliary thrusters located on the supply module. Hydrazine monopropellant engines are used providing a continuous thrust of 10^{-4} g's. The concept is illustrated in Figure 5-3. Detailed conceptual design data and operating procedures are presented in Paragraphs 4.2.1.5 and 4.2.1.6.

The same technology items are identified here as were identified for the drag system in Paragraph 5.1. The only addition is that in the present case long life low thrust hydrazine engines are required to provide liquid settling and orientation. Also, over the major portion of the transfer, accelerations and thus Bond numbers are significantly greater than for the drag case. The liquid-vapor interface will thus be flatter and one-g test data will be more applicable.

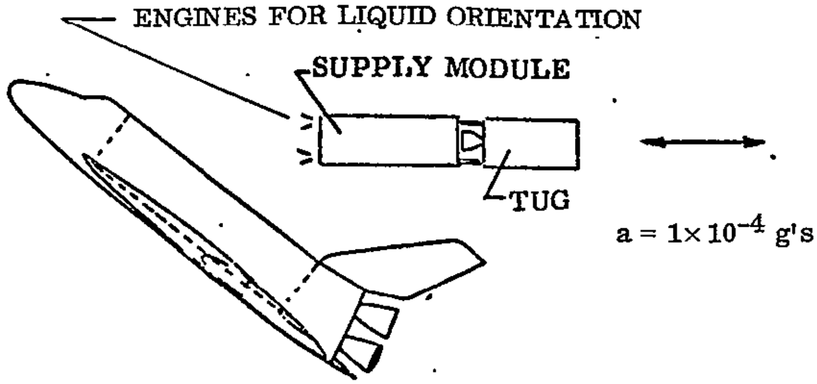

Figure 5-3. Separated Tug Supply

In the case of tank pressure control prior to transfer, exactly the same technology considerations apply here as for the drag system (Reference Paragraph 5.1.4). Other technology areas are discussed in the following paragraphs.

5.2.1 LOW-G LIQUID REORIENTATION - Based on the 10^{-4} g's settling acceleration and tank radii, settling Bond numbers are 62 for LH_2 and 52 for LO_2. For this Bond number range, per Paragraph 2.1.4, predictions of settling time can be adequately handled using NASA/LeRC drop tower correlations, such as found in References 5-5 and 5-6. Also, for the current application the settling time is very short and required impulse low in relation to the overall transfer. It is thus recommended that no further technology work be accomplished in this area for the separated Tug supply.

5.2.2 LOW-G LIQUID OUTFLOW - Bond numbers here are significantly higher than for the drag system (62 for LH_2 and 52 for LO_2 versus 0.7 and 0.6). At these higher Bond numbers the liquid-vapor interface is fairly flat and Froude number correlations should be reasonably accurate (Reference Paragraph 2.1.7). However, as indicated by differences between the flat interface residuals calculated from Reference 5-7 and those calculated from Reference 5-8, there is still considerable uncertainty involved.

Residuals for the baseline design were calculated using data from Reference 5-7. Total LO_2 plus LH_2 residuals were 212 kg (466 lb) or 5.4% of payload.

Work in this area should be extended to include the use of contoured outlets and various pullthrough suppression devices such as screens and baffles. Currently published studies do not provide sufficient data or criteria necessary to design optimum tank outlet configurations. Unpublished work by performed at the NASA/LeRC indicates that tank residuals can be essentially eliminated by proper outlet contouring. If this could be accomplished a significant payload increase would be possible.

For the current range of conditions, testing could be done at one-g, with different tank sizes, fluids and outlet configurations, as a function of outflow rate. Also, the effects of various flow throttling ranges should be studied, including the advantages and hardware availability for increasing throttling ratios beyond 10:1.

5.2.3 RECEIVER CHILLDOWN AND FILL - Based on inlet line conditions, the Bond number, Weber number and Reynolds number ranges of interest here are presented in Table 5-2.

Table 5-2. Separated Tug Receiver Inflow Parameters

	Bo_{N_R}	We_{N_R}		Re_{N_D}	
		Max.	Min.	Max.	Min.
Hydrogen	0.18	320	3.2	294,800	29,480
Oxygen	0.36	128	1.3	129,600	12,960

As mentioned in Paragraph 5.1.3, the only directly applicable low-g test data are found in Reference 5-1. This testing was accomplished in the NASA-LeRC 5 sec drop tower and consisted of flow into an unbaffled cylindrical tank with hemispherical bottom. Inlet Bond, Weber and Reynolds numbers were respectively 0.06 to 2.8, 3.4 to 27 and 1415 to 9870. This covers the Table 5-2 Bond number range, but not the range of Weber number or Reynolds number. Also, analysis in Paragraph 4.2.1.3 showed that baffles would likely be required to prevent liquid from exiting the vent during childown. Even though settling accelerations are greater here than for the drag case the inflow rates are also much higher due to the shorter transfer time. Thus prevention of direct liquid loss may still be a problem.

The technology work recommended here is essentially the same as that for the drag system in Paragraph 5.1.3, including the range of conditions presented in Table 5-2.

In defining receiver pressure control system(s) for the separated Tug supply it is of interest that a H_2 tank zero-g vent system which is likely to exist on the Tug would not have the capacity to vent the 66 kg (146 lb) required for chilldown, even if accomplished over the entire 2.5 hour transfer time. That is, a normal space storage type heat exchanger vent system would have a vent capacity of about 0.38 g/s (3 lb/hr) and weigh about 4.5 kg (10 lb). To vent 66 kg (146 lb) of H_2 in 2.5 hours would require a vent system weighing on the order of 61 kg (135 lb).

5.2.4 TANK PRESSURIZATION FOR EXPULSION - The technology requirements here are essentially the same as for the drag system (Paragraph 5.1.5), except that transfer times are shorter and the liquid-vapor interface considerably flatter. The analytical model development described in Paragraph 5.1.5 should allow application to the separated case where Bond numbers during transfer are on the order of 50 to 60. It is expected that analysis predictions would be more accurate for the separated case and actual pressurant usage should also be somewhat less.

5.2.5 LOW THRUST LONG LIFE ENGINES - Referring to Paragraph 4.2.1.5, it is seen that the separated Tug supply system requires four 0.9 to 1.4 kg (2 to 3 lb) thrust hydrazine engines which must operate continuously for a minimum of 2.5 hours.

These engines are required for the overall transfer system to operate as designed. Hydrazine engines in this thrust range are presently available. Their normal use is for pulse type firings where any one firing is for only a relatively short time.

However, it is not anticipated that firing durations of 2.5 hours will impose any severe penalties on the engines as presently designed. The technology program proposed here would be simply to demonstrate the long term continuous firing capability of an existing design.

5.2.6 OTHER TECHNOLOGY ITEMS - Technology requirements are basically the same as discussed in Paragraph 5.1.6 for the drag supply system, except that here acceleration levels are considerably higher. In general, the problems associated with the low-g environment will thus be lessened.

The effect on mass gauging is that even further consideration should be given to the use of more conventional systems such as point level sensors, capacitance gauges and the Reference 4-4 vertical resistance wire system.

In the case of low-g acceleration measurement it is likely that existing technology will be adequate.

5.3 SHUTTLE ORBITER IN-ORBIT SUPPLY

This system employs surface tension screen channels mounted inside spherical tankage to collect liquid for transfer (Figure 5-4). The nominal acceleration during transfer is 0, however, the screens are designed to retain liquid during disturbing accelerations in any direction up to 10^{-4} g's. Both cryogenic (LH_2, LO_2, LHe) and non-cryogenic (N_2O_4, MMH) fluids are transferred. Detailed conceptual design data and operating procedures are presented in Paragraphs 4.2.3.5 and 4.2.3.6. An examination of the data resulted in identification of the following items as being unique or critical to system design and performance.

a. Surface Tension Screens

b. LO_2 and LH_2 Cryogenic Receivers Childown and Fill

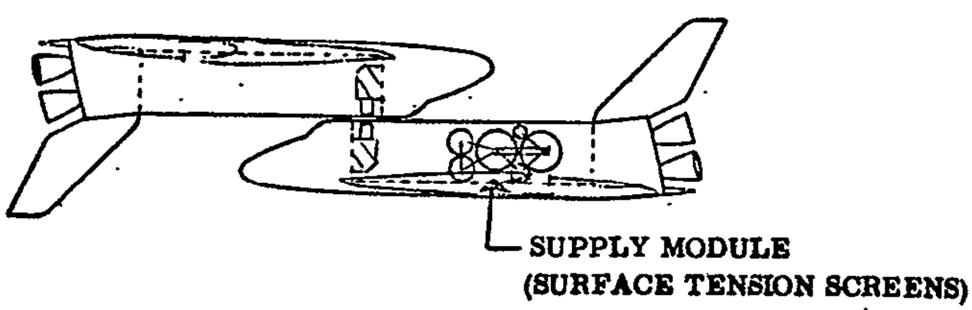

Figure 5-4. Shuttle Orbiter Supply

c. Non-Cryogenic Receivers Fill

d. LO_2 and LH_2 Tank Pressure Control Prior to Transfer

e. LO_2 and LH_2 Pressurization For Expulsion

f. Special Problems of He Transfer

g. Other Technology Items

Evaluation of the above items is covered in the following paragraphs.

5.3.1 SURFACE TENSION SCREENS - The basic screen system proposed for the Shuttle supply is illustrated in Figure 5-5, along with the various size and fluid applications. This screen system is composed of eight semi-circular channels with rectangular cross-section, attached between an upper and lower manifold and spaced evenly around the tank 0.79 rad (45°) apart.

Over 500 flight tests have been accomplished on screen type acquisition devices for use with non-cryogenic fluids. Most of these systems are of the local "basket" type for low-g engine start. For this application only a relatively small amount of liquid needs to be transferred at low-g; i.e., until engine thrust can accomplish liquid orientation. There is, however one propulsion system, the Satellite Control Section (SCS) which has been successfully flown and which utilizes a 1.5 m (5 ft) diameter channel system to provide continuous engine feed of hydrazine at low-g. This system is similar to that proposed for the current Shuttle in-orbit fluid transfer.

One of the main problems yet needing resolution is to insure that the screen channels are full at the start of transfer and that they remain full during transfer. Screen disturbances and/or vibration during loading, boost, orbital injection, rendezvous, docking and fluid transfer may cause the screen(s) to become unwetted, resulting in vapor formation within the channels. This could result in screen retention breakdown and subsequent failure to complete the fluid transfer. This is especially critical for the

N_2O_4 - 2.9 M (9.5 FT) DIA
MMH - 2.9 M (9.5 FT) DIA
LO_2 - 1.4 M (4.5 FT) DIA
LH_2 - 2.3 M (7.7 FT) DIA
LHe - 1.7 M (5.7 FT) DIA

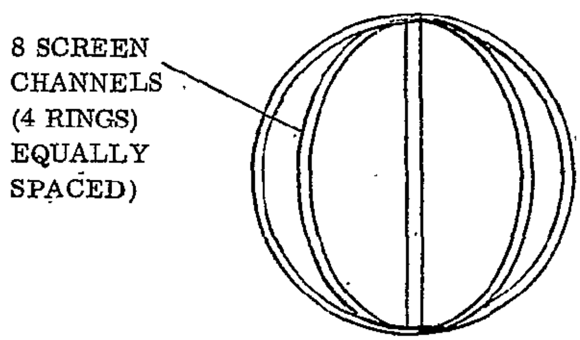

Figure 5-5. Surface Tension Screens for Shuttle Supply

Case 2 transfer (Paragraph 4.2.3.5) where the N_2O_4 and MMH tanks are not carried full. In the SCS system, liquid motion occurring at orbit injection was relied upon to rewet the top portion of the screens which could become dry during boost.

Basic low-g screen wicking and screen vibration retention characteristics need to be determined in order to design systems to insure full channels under current Shuttle supply conditions.

Another approach to the problem, which should receive attention, is the development of means to refill surface tension devices at low-g in case unwetting does occur. This technology would also be applicable to receiver tank filling where the receiver may contain surface tension devices. It is noted that start basket type screen systems are located in the Shuttle Orbiter OMS tanks which may need to be filled in orbit.

In the case of cryogenic systems, none to-date have been flown. A 1.8 m (6 ft) diameter screen liner system has been built to demonstrate fabricability. Also, there has been a significant amount of analysis and small scale testing accomplished covering transient and steady state outflow through screens, basic screen retention capabilities, thermal conditioning and wicking to prevent screen drying and determination of the effects of pressurant and vibration environments on screen retention. It is noted that the vibration work was very limited and final design data was not obtained. Recent data from Reference 5-9 indicates that the use of warm pressurant is not a problem. It was found that screens normally used for retention are also good wicks and that heating at screen surfaces of up to 9450 W/m^2 (3000 $Btu/hr\text{-}ft^2$) can be tolerated. Work on the transient effects of flow start-up and line chilldown is presently being accomplished under Contract NAS3-19719. Nothing has been done for specific application to the helium case.

The problems with cryogenics are the same as for non-cryogenics with the addition that thermal control is required to prevent external and/or internal heating and/or tank pressure excursions from causing screen drying and/or vapor formation within the channels.

For the current Shuttle in-orbit transfer application it is proposed to utilize wicking to maintain the screen surfaces wet at all times. Work is currently underway at Convair, as part of Contract NAS3-19693, to generate basic wicking data on various screen and screen backup sandwich type configurations. This data is also being correlated with a general analytical model which should be of use to the design of the Shuttle supply type systems. Further work will, however, be needed to demonstrate fabrication of seams, corners, supports and outlets such that wicking is satisfactorily accomplished in these areas. Any testing should include configurations representative of full scale potentially operational systems.

Prior to final system design, investigations should also be made into the overall effects on retention capabilities of fluid mixing, such as could occur with a heat exchanger type zero-g vent system.

A summary is presented in Table 5-3 of recommended work for the final development of screen transfer systems. It is noted that orbital testing would be required for final demonstration of overall system performance.

Other basic technologies are discussed below in which further work would advance the overall state-of-the-art of screen surface tension liquid orientation.

a. Low-G Convection, Boiling and Condensation Heat Transfer. This information would improve prediction of the expected screen system thermal environment and the potential for screen drying and/or vapor formation within the screen device. Requirements for thermal conditioning could also be more accurately determined.

b. Vapor Formation in Bulk Liquid. Tank venting as well as heating has the potential to form vapor within a screen system. This can cause reduced transfer efficiency or even complete breakdown of the liquid transfer process. Further work is needed to determine under just what conditions such vapor can be formed and how much. The effects of helium gas in the ullage should be included in the investigation.

c. Vapor Collapse in Bulk Liquid. This work would primarily be applicable to low-g refilling. Data suitable for design is needed on bubble collapse times as function of cooling and pressurization (subcooling). The effects of helium in the tank should be included.

5.3.2 O_2 AND H_2 CRYOGENIC RECEIVERS CHILLDOWN AND FILL - A significant difference between the Shuttle fill and the Tug fill is that filling of the Shuttle is accomplished without any settling acceleration tending to keep liquid away from the vent. Also, the supercritical H_2 and O_2 receivers aboard the Shuttle are relatively small and heavy and significant venting must be accomplished during chilldown, or allowable receiver pressures will be exceeded.

As described in Paragraph 4.2.3.1 the chilldown/fill operation is accomplished in two steps:

1. Initial chilldown to predetermined wall temperatures while venting.

2. Tank lock-up and final chilldown and fill without venting.

C. 3

Table 5-3. Screen Supply Technology Work

Recommended Technology Programs
Refilling of Capillary Devices - Analytically and experimentally evaluate and demonstrate the use of refilling techniques such as inflow baffles, refill valves, thermodynamic venting of capillary device contents and condensation of trapped vapor. Consider normal gravity filling prior to launch, refilling in orbit from supply tanks and repositioning of displaced liquid.
Passive Cooling of Capillary Devices - Experimentally evaluate candidate wicking configurations for preventing screen drying. Develop fabrication methods for controlling wick dimensions including close tolerance spacing between parallel screens. Develop methods of wicking around corners and across welds.
Effect of Disturbance on Final Draining of Capillary Devices - Analytically determine the extent of liquid displacement from channel contact. Design and test baffles or other means of preventing vapor pullthrough due to liquid displacement.
Capillary Device Thermal Environment - Determine the effect of tank pressure changes and incident heat transfer on capillary device passive cooling requirements. Investigate methods of intercepting feedline related heat flux and preventing pressure buildup in the feedline from interferring with liquid retention. Develop a prototype passively cooled channel type capillary device design. Experimentally evaluate a model of this device at normal gravity. Overall integration of the low-g vent system with the screen liquid acquisition system should be included.
Vibration - Determine the vibration spectrum to be experienced by the capillary device. Determine, experimentally, the effect of vibration on capillary device retention.
Impact Loading - Experimentally determine the effect of impulsive liquid loading on capillary device deflections, loads and retention.
Layered Screen Pressure Drop - Determine the flow/pressure drop characteristics of layered screen candidates designed to reduce the pressure drop of screen and perforated plate. Test several combinations of fine mesh screen and perforated plate; and fine mesh screen, coarse mesh screen and perforated plate.
Screen Repair Techniques - Determine screen repair techniques that are compatible with oxidizers (N_2O_4 and LO_2).
Film Bubble Point - Develop a reliable, and relatively simple film bubble point technique for evaluating the retention capability of full scale capillary devices

The general state-of-the-art of low-g receiver chilldown and fill was covered in Paragraph 5.1.3. Drop tower testing with non-cryogenic fluids indicated that by proper inlet flow baffling and vent location the direct loss of liquid could be minimized. Testing is presently being accomplished at the NASA/LeRC to determine the effect of using cryogenics. In this case LN_2 is transferred into an 0.6 m (2 ft) diameter CRES sphere.

Work to-date has not been sufficient to demonstrate a generally applicable system for insuring efficient receiver tank pressure control during chilldown and fill. A significant amount of further work, with specific application to the current Shuttle fill, is needed to bring such transfer to an operational status. The existing inlet, tank internal and vent hardware would need to be included in any final system evaluation and design.

The areas requiring further technology work are listed below.

a. Venting With a Minimum of Fluid Loss

b. Defining Criteria for Vent Termination or Tank Lock-up

c. Controlling Pressure Rise Following Tank Lock-up

To illustrate the importance of the above items, some quantitative data are presented in the following paragraphs.

Efficient Venting

The difference in H_2 supply system weight between venting saturated vapor and venting superheated vapor at the tank wall temperature was determined to be 123 kg (272 lb). This represents a 25% weight increase for the saturated case over the superheated case. For O_2 the difference is only 23 kg (50 lb). This assumes no direct loss of liquid. Based on data from Reference 4-2, assuming a mixed fluid is vented during chilldown, the Shuttle H_2 supply system weight was calculated to be 844 kg (1859 lb). This represents a 68% increase in weight over a system with superheated venting (supply weight = 502 kg, 1105 lb). The actual quantity of fluid vented will depend on liquid inflow patterns, vent location and inflow rates in relation to wall cooling rates.

Vent Termination

Vent termination or tank lock-up is based on receiver temperature measurements designed to verify the existence of a predetermined level of energy remaining to be removed from the receiver. If the tank is locked up too soon then a full fluid loading cannot be accomplished. Therefore, some tolerance must be maintained on the low side of the receiver temperature at lock-up; i.e., some excess in vented fluid must be tolerated to insure a full receiver tank. As an example of this effect, if the H_2 tank is locked up at an average temperature of 89K (160R) rather than 94K (170R) a 10% increase will occur in the quantity of vented fluid. In the O_2 case a temperature change from 250 to 236K (450 to 425R) would result in a 10% increase in vent fluid. An investigation would thus be required of the accuracies to be expected in relating receiver temperature measurements to receiver energy. This could be accomplished primarily by tests of the actual hardware at one-g.

Locked-up Pressure Rise Control

The minimum pressure rise in a locked-up tank occurs when the liquid and vapor are thermally mixed. For a given allowable pressure rise, this condition also results in the maximum energy absorption of the incoming fluid. Assuming the maximum receiver pressure to be fixed, any anticipated non-equilibrium must be made up by increased venting. As an example, in the O_2 case, if the degree of non-equilibrium is such as to result in a total excess pressure rise of 34.5 kN/m^2 (5 psi) then an additional vent requirement of 5% would result. If the excess is 69 kN/m^2 (10 psi) then the vent increase would be 20%.

Work to be Accomplished

An outline is presented below of the tasks considered necessary to develop the required technology for overall pressure control of the Shuttle O_2 and H_2 receivers.

a. Task I - Analytical Model Development. This would be basically the same as presented in Paragraph 5.1.3, only oriented to the Shuttle low-g fill requirements. Provisions should be incorporated into the model for including in the analyses the specific vent hardware existing in the Shuttle receiver tanks. Also, the capability is needed to determine when to stop venting and lock up the receivers.

b. Task II - Drop Tower Testing. It is recommended to complete the LN_2 testing program presently being performed at the NASA/LeRC. Depending on the outcome of the current program, some further tests with internal tank hardware representative of the Shuttle receivers may be warranted.

c. Task III - One-g Thermodynamic Testing - This testing would mainly be designed to develop reliable and efficient concepts for controlling the termination of venting and tank lock-up and to determine the accuracies to be expected in relating receiver temperature measurements to receiver energy. Various inflow configurations should also be tested to define optimum methods of promoting mixed fluid conditions following lock-up. Compatibility with the existing receiver tank hardware and the desire to minimize fluid loss while venting must be considered.

d. Task IV - Systems Definition. Using the above model(s) and one-g and drop tower test data, analyses should be performed to define weight and performance of the most promising concepts for accomplishing low-g receiver tank pressure control. Both passive (inlet flow control) and active (liquid/vapor separation at the vent) type systems should be considered. Weight, reliability and performance comparisons would then be made between the various systems to determine the one(s) worthy of further development.

e. Task V - Prototype Demonstration Testing - Detailed design, fabrication and one-g testing of the most promising concept(s) would be accomplished under this task. This system should be capable of final testing at low-g in an orbital experiment.

f. Task VI - Orbital Experimentation. Final demonstration testing of the most promising receiver pressure control concept(s) must be accomplished by orbital experimentation. Test tank sizes on the order of the full scale systems (0.9 m, 3 ft dia) should be used. Due to the large expense involved in orbital experimentation, preliminary ground testing should be accomplished to develop reliable instrumentation and measurement techniques.

Other associated basic technologies in which further work would advance the overall state-of-the-art of low-g receiver chilldown and fill are the same as presented in Paragraph 5.1.3.

5.3.3 NON-CRYOGENIC RECEIVERS FILL - As illustrated in Figure 4-36, the N_2O_4 and MMH receivers contain surface tension screen type low-g engine start systems. Insuring that these start systems are full at the completion of transfer is the main problem associated with in-orbit supply of these receivers.

The baseline transfer system design assumes that the tanks are or can be vented to vacuum in order to be completely empty of GHe. GHe is the pressurant used in flight and if present could become trapped within the screen devices during low-g fill. With the tanks initially empty, filling is in a locked-up or non-vented condition. Any vapor formed within the baskets would then be condensed upon final filling or subsequent tank pressurization prior to use. The vapor pressures at ambient temperature (294K, 70F) of N_2O_4 and MMH are respectively 101 kN/m^2 (14.7 psia) and 5.5 kN/m^2 (0.8 psia).

There has been essentially no work accomplished to-date on this specific concept. Further work is needed to demonstrate feasibility.

One task is to verify that no fluid problems, such as dissociation, result when expanding the entering liquid into a vacuum (0 psia). Discussions with the Convair fluid properties laboratory indicated this should not be a problem but must be verified by test. Another area for investigation is to insure that the baskets do become wetted following fill and that liquid is available at the basket to allow filling as vapor within the basket condenses.

Due to the importance of this filling problem, work should also be accomplished on back-up systems in case unforeseen problems occur or the receiver cannot be sufficiently emptied.

One approach would be to develop low-g screen refilling systems or procedures, as discussed in Paragraph 5.3.3. Possibilities include providing baffling, valving, and/or flow rates such that inflow expels any vapor from the basket into the main tank, or insuring baskets are full at the start of transfer and filling is in a manner to maintain them full.

Also, there is some chance that emptying the receivers prior to transfer is not practical such that a non-vent transfer is not feasible. In this case, passive and/or active vent systems as discussed in Paragraphs 5.1.3 and 5.3.2 would be needed to prevent excessive liquid loss during fill. Lock-up of the receivers near the end of fill should be considered in conjunction with venting to minimize the possibility of liquid loss which is more critical as the tanks become full.

In all the above cases, basic screen wicking data and low-g refilling of screen systems, as discussed in Paragraph 5.3.3, would be applicable, as would basic data on vapor formation and bubble collapse.

For any of the above concepts, final verification of performance would need to be accomplished in orbital testing. However, assuming satisfactory demonstration at one-g, locked-up flow into an empty tank would have the greatest chance for success.

5.3.4 LO_2 AND LH_2 TANK PRESSURE CONTROL PRIOR TO TRANSFER - This technology was discussed extensively in Paragraph 5.1.4. Only the major differences associated with the current Shuttle supply will be discussed here.

The basic difference is that in the present case the supply tankage is spherical and of much smaller size than for the Tug case. Any specific analytical model developments and/or testing must of course take this into account. Also, demonstration of compatibility with operation of the surface tension screens must be included for the Shuttle supply. This is discussed in Paragraph 5.3.1. Final demonstration of pressure-control-surface-tension system compatibility would need to be accomplished through orbital experimentation.

It was determined in the Shuttle supply case, as it was for the Tug supply, that venting could be eliminated altogether by tank fluid mixing or by a combination of venting during boost and improvements in low-g pressure rise predictions.

The estimated savings in total supply system weights by not venting were determined to be only 18 kg (40 lb) and 13.6 (30 lb) respectively for the LH_2 and LO_2 cases. Therefore, the desirability of further work in this area would be primarily for system simplification.

In the case of mixing without venting, as discussed in Paragraph 5.1.4, further work considering that the mixer may be in the vapor would be important, since for some missions, such as discussed in Section 6, it may be desirable to carry less than a full load of liquid.

5.3.5 LO_2 and LH_2 PRESSURIZATION FOR EXPULSION - This technology was also discussed in Paragraph 5.1.5. The primary consideration for the current Shuttle supply case is that orientation of most of the liquid may be completely unknown and a variable during the transfer. This mangifies the problems of a non-flat interface as discussed in Paragraph 5.1.5.

The current baseline system assumes that helium stored in the liquid tankage is used for pressurization at the temperature of the liquid. This eliminates the unknown associated with mixing and heat transfer between a warm pressurant and the liquid. Considering just pressurization of the O_2 and H_2 supply tanks with He, a minimum weight system was determined to be with He stored at LH_2 temperature and heated to 287K (520R). Using maximum pressurant collapse factors from Reference D-1, it was calculated that this system would weigh 44 (97 lb) less than the all cold system. This is only a small percentage of the total supply module weight of 29,500 kg (65,000 lb) and whether or not this reduction could actually be realized is open to question.

A significant amount of orbital testing would be required to determine how much, if any, of this weight reduction could be realized. Due to the relatively small weight involved, it is recommended that further work in this area not be pursued for the present Shuttle supply application.

5.3.6 SPECIAL PROBLEMS OF HELIUM TRANSFER - Essentially no work has been done in connection with transferring LHe at low-g. As discussed in Paragraphs 4.2.3.1 and 4.2.3.3 the baseline helium supply system chosen utilizes storage as a liquid with surface tension screen channels for collection. Helium is somewhat unique as a transfer fluid, as compared to LO_2 and LH_2, in that it has a very low surface tension and heat of vaporization. Its heat of vaporization is about one-tenth that of LO_2 and one-twentieth that of LH_2. Coupled with its low temperature, this results in a high potential for vapor formation. The surface tension is less than one-one hundredth that of LO_2 and one-twentieth that of LH_2.

Recommended He technology work for the various key transfer areas is discussed in the following paragraphs.

5.3.6.1 Surface Tension Screens - Use of surface tension screens to collect LHe at low-g needs further investigation. Due to its low surface tension, temperature and heat of vaporization the basic problems of screen retention with cryogenics are magnified; especially those dealing with the prevention of vapor formation within the screen channels.

Wicking, thermal control, and vent systems integration work, as discussed in Paragraph 5.3.1, should be extended to specifically cover the LHe case.

5.3.6.2 Tank Pressure Control Prior to Transfer - The baseline system assumes use of a bulk heat exchanger type vent system as shown in Figure 2-5. Such a system has not been demonstrated for use with helium, however, a design similar to that used for H_2 should be applicable. A further task would be to design, fabricate and test at one-g a system specifically for use with liquid helium.

As to system simplification, as discussed for O_2 and H_2 in Paragraphs 5.1.4 and 5.3.4; sufficient data is not presently available to make an accurate determination of the potential for eliminating low-g tank venting by mixing and/or venting during boost. Further work would be needed to determine helium mixing requirements and non-mixed pressure rise rates at low-g. The work required would be similar to that discussed in Paragraphs 5.1.4 and 5.3.4, except that here the starting point is with even less initial data than for H_2 and O_2.

5.3.6.3 Pressurization For Expulsion - In the baseline system, pressurization is by external pumping and vaporization of helium stored as part of the LHe supply.

As was found for the O_2 and H_2 cases (Paragraph 5.3.5), weight savings in relation to unknowns in going to higher pressurant temperatures would not warrant further work for the Shuttle.

The baseline system should present no major problems in development; however, work should be accomplished to demonstrate compatibility with the surface tension screen system. Since vaporized liquid is used as pressurant, the control of pressure excursions which may affect screen vapor formation is unique, as compared to O_2 and H_2 cases.

5.3.6.4 Tank Thermal Control - The helium storage system is unique, as compared to the O_2 and H_2 systems, in that a 1.3 cm (0.5 in.) layer of foam is used under the MLI in the helium case. No major problems are anticipated; however testing is needed to demonstrate reusability of the foam system for the present application.

5.3.6.5 Receivers Chilldown and Fill - This is likely the major problem connected with designing an efficient and reliable in-orbit supply system for the helium case. The receivers are 33.1 MN/m^2 (4800 psia) ambient temperature bottles constructed of an Inconel liner wrapped with Kevlar. The baseline system employs transfer of liquid helium into locked-up receivers. A high pressure transfer pump is required, since the final receiver pressure at termination of inflow is estimated to be 24.8 MN/m^2 (3600 psia). Even though this system has the lowest weight and/or greatest potential for operating as predicted, when compared to the other systems considered, there is a significant amount of development work required. The major tasks are listed below.

a. Compatibility Demonstration of the Receiver Systems With the Initially Cold Helium. Valves, plumbing and receiver bottles existing on the Shuttle would need to be investigated. Valves and plumbing could be replaced if needed. In the case of the high pressure bottles, there is some chance that during chilldown the Inconel liner could pull away from the Kevlar wrap. Testing could be done at one-g.

b. Detailed Thermal Analysis of Receiver Filling. The maximum pressure which will be reached at final fill is sensitive to the bottle, inlet fluid and pump energy levels; as well as inflow rates in relation to heat transfer from the bottle to the fluid. Further analytical work and possibly some one-g testing are needed to allow a final specification of maximum and minimum pressures to be expected.

c. Pump Survey to Determine the Availability of Pumping Systems to Accomplish the Required Transfer.

Due to the many uncertainties involved with the baseline transfer system it is also recommended to further explore the possibility of alternate systems. Calculations show that by accepting a weight penalty on the order of 136 kg (300 lb), that the system in Figure 4-34 could be used for the helium supply. This would eliminate liquid storage and high pressure pumping problems.

Further analysis and testing would however be needed to verify the practicality of this system for the present application, including detailed thermal analysis of receiver filling and demonstration of receivers compatibility with initially cold helium.

The same basic technology work as outlined at the end of Paragraph 5.1.3, oriented to the use of helium, would also be applicable here.

5.3.7 OTHER TECHNOLOGY ITEMS - Other items which are unique and/or critical to efficient and predictable in-orbit fluid supply of the Shuttle Orbiter are discussed below.

Low-G Mass Gauging

Mass gauging would be required in both supply and receiver tankage. Accuracies similar to those obtainable on the ground are desired for the receivers. Anything less will tend to compromise subsequent fluid use and reduce the advantages of in-orbit supply. Mass measurements in the supply are needed to determine if sufficient liquids have been dumped for re-entry and to determine if the transfer is proceeding as expected. A knowledge of fluid masses in both the supply and receivers would uncover whether significant quantities of liquid were being lost during the transfer. In the case of the storable fluids, existing state-of-the-art pressure-volume-temperature (PVT) systems should be adequate for both supply and receiver tankage. This concept is used in the current Shuttle design. For the supercritical H_2 and O_2 and high pressure He receivers PVT measurements would also be adequate since single phase fluids are involved.

For the LH_2 and LO_2 supply systems some further work would be required, as discussed in Paragraph 2.3.2.1. LHe has not been investigated to-date.

Liquid-Vapor Sensing

Where venting is accomplished, liquid-vapor sensing is needed to monitor any possible breakdown in the system. If liquid is detected it may be possible to take corrective action.

Detection of vapor at supply tank outlets is important to determine if screen breakdown has occurred, so that corrective action can be taken. Assuming the capability for low-g screen refilling, the transfer could be stopped and refill accomplished. Other possibilities may be to reduce flow or to momentarily stop the flow until adverse disturbing accelerations which may have caused the problem are terminated.

As discussed in Paragraph 2.3.2.2, further work is needed to develop and demonstrate satisfactory liquid-vapor sensing where low-g is involved. In the present case the problems are likely to be reduced since a forced flow will exist at the sensor.

Low-G Acceleration Measurement

The need for further work in this area would be associated with system testing prior to operational flights and in determining if disturbing accelerations during flight are of magnitudes and directions sufficient to hinder satisfactory transfer. For accelerations below 10^{-5} g's, it is anticipated that further development work will be required, (Reference Paragraph 2.3.2.6).

Purging And Safing

Following transfer or abort dump, all tanks and lines are to be purged of liquid. In the case of the tanks this liquid is replaced with helium. Complete dumping is especially important in the case of H_2. Due to the low-g environment and the existence of the screen systems, complete expulsion of liquid and purging of tanks and lines will require further investigation.

Transfer Line Childown

For the cryogenics, overall transfer will be influenced by fluid conditions in the transfer lines and the time for the lines to childown to liquid temperatures. Based on the state-of-the-art as discussed in Paragraph 2.3.1, orbital testing is needed to determine flow regimes and how they relate to heat transfer and line childown at low-g.

5.4 MULTIPLE RECEIVERS SUPPLY

The overall transfer system discussed here includes provisions for supplying three different receivers. These are the Solar Electric Propulsion Stage (SEPS), Large High Energy Observatory B (HEAO-B) and Satellite Control Section (SCS). Hg, LHe and N_2H_4 are supplied respectively to each of these receivers and different liquid acquisition schemes are employed for each case. These are, respectively, diaphragm, paddle rotation and surface tension channels. Detailed conceptual design data and operating procedures are presented in Paragraphs 4.2.4.4 and 4.2.4.5.

Technology requirements for each of the three receivers are discussed in the following paragraphs.

5.4.1 SOLAR ELECTRIC PRCPULSION STAGE (SEPS) - The fluid transferred here is mercury, which is highly corrosive. The only items which were identified as being unique or critical to the design of this system were the positive expulsion diaphragm and zero leak connectors.

In the case of the diaphragm, the main problem is compatibility with the mercury. However, an AF-E-332 diaphragm has been satisfactorily tested with Hg and no particular problems are anticipated in its use with the proposed transfer system.

Due to the high corrosiveness of the Hg, the baseline transfer system is designed to prevent leakage or venting of mercury at all times. To accomplish this, the receiver and supply lines are maintained full and the supply/receiver disconnect must be designed for zero leakage at all times, including periods of connection and disconnection. Such connectors are considered to be within the current state-of-the-art and would just need to be demonstrated for the current application.

5.4.2 LARGE HIGH ENERGY OBSERVATORY B (HEAO-B) - In this system, fairly large amounts of liquid helium (431 kg, 950 lb) are to be supplied. The two major technology items are receiver tank childown and development of the paddle rotation liquid orientation concept. Discussions are presented in the following paragraphs.

5.4.2.1 Receiver Childown and Fill - A major problem is that the receiver tank and superconducting magnet (Figure 4-39) are relatively heavy and require a significant amount of fluid just for childown. The operating temperature of the magnet is such that helium saturated at approximately 103 kN/m^2 (15 psia) is required in the receiver. Receiver venting is required to maintain this pressure. As mentioned in Paragraph 5.3.6, the heat of vaporization of LHe is very low and thus chilldown with LHe can require large fluid quantities. Weight comparisons for various chilldown cases are found in Table 4-33. It is seen that significant weight differences exist between the various chilldown assumptions and systems considered. Under the current state-of-the-art the actual conditions to be expected are quite uncertain, since very little work along these lines has been accomplished to-date. Use of LN_2 for pre-chill was chosen for the baseline design.

The state-of-the-art and further work required would be along the same lines as discussed in Paragraphs 5.1.3 and 5.3.2, with emphasis here on efficient venting at essentially zero-g. Some specific tasks are outlined below.

a. Task I - Analytical Model Development. This would be basically the same as presented in Paragraph 5.1.3, only oriented to the HEAO-B requirements. Provisions should be incorporated into the model for including in the analyses the specific hardware and vent provisions existing in the HEAO-B receiver.

b. Task II - Drop Tower Testing. It is recommended to complete the LN_2 testing program presently being performed at the NASA/LeRC. Depending on the outcome of the current program, some further tests with internal tank hardware representative of the HEAO-B may be warranted.

c. Task III - One-g Thermodynamic Testing. This testing would be designed to determine accuracies to be expected in relating receiver temperature measurements to receiver energy and when, if at all, the receiver could be locked-up for final fill. Also, relative energy removal between LN_2 and LHe and required tolerances would need to be determined. Intermittent venting to increase vent efficiency should also be investigated here; i.e., transfer of small amount of liquid to locked-up tank, warm up of transferred fluid to wall temperature and then vent overboard. Preliminary analysis indicated that such a system would be feasible, but time consuming. Further work is needed to include the possibility of using a mixer to reduce warm up time between vents. Basic information on low-g convection heat transfer would be applicable here.

   If high chilldown efficiency could be demonstrated with LHe alone, the LN_2 prechill system could be eliminated (Reference Table 4-33), thus considerably simplifying the overall transfer system.

d. Task IV - Systems Definition. Using the above model(s) and one-g and drop tower test data, analyses should be performed to define weight and performance of the most promising concepts for accomplishing low-g receiver tank pressure control. Both passive (inlet flow control or intermittent vent) and active (liquid/vapor separation at the vent) type systems should be considered. Weight, reliability and performance comparisons would then be made between the various systems to determine the one(s) worthy of further development.

e. Task V - Prototype Demonstration Testing - Detailed design, fabrication and one-g testing of the most promising concept(s) would be accomplished under this task. This system should be capable of final testing at low-g in an orbital experiment.

f. Task VI - Orbital Experimentation. Final demonstration testing of the most promising receiver pressure control concept(s) must be accomplished by orbital experimentation. Due to the large expense involved in orbital experimentation, preliminary ground testing should be accomplished to develop reliable instrumentation and measurement techniques.

Other associated basic technologies in which further work would advance the overall state-of-the-art of low-g receiver chilldown and fill are the same as presented in Paragraph 5.1.3.

5.4.2.2 Paddle Rotation System Development. This system was chosen for the HEAO-B baseline due to its positive orientation feature and minimization of heat transfer and liquid retention problems, as compared to a surface tension screen device. These advantages of the paddle system are magnified when dealing with LHe. Even for other fluids, including the Shuttle N_2O_4, MMH, LH_2 and LO_2, the paddle system appears to be a good back-up to surface tension screens. Other potential advantages of this system are listed below.

a. It may be possible to determine liquid mass during transfer simply by measuring pressure at the outer wall and knowing rotation rates (Reference 5-10).

b. Low-g venting may be easily accomplished by liquid orientation using the paddle. In any case, the paddle should be capable of fluid mixing to minimizing venting or to allow use of a wall exchanger without a separate mixer.

c. Since the liquid is oriented during transfer, warm pressurant gas could be used with a possible minimization of pressure collapse due to interaction of the vapor and liquid.

There has been very little work done to-date on such systems. What has been done has been restricted to small systems on the order ot 15.2 cm (6 in.) diameter.

The following work is recommended.

a. Small scale testing at one-g to determine the basic vortex flow and energy dissipation characteristics of the system. An insight would be obtained into power requirements and expected residuals as needed to determine the basic feasibility of the concept for the current application. Use of two-dimensional tests and liquid/liquid models should be considered.

b. Definition of optimum low-heat-leak method(s) of driving the paddle. Consideration should be given to locating the motor inside the tank, outside the tank and inside the insulation, and outside the tank and insulation.

c. Determination of optimum integration of paddle system with auxiliary systems, such as tank pressure control prior to transfer, tank pressurization for expulsion and mass gauging.

d. The desirability of further development work would depend on the results of a., b. and c. above.

5.4.2.3 Other Technology Items - As presented in Paragraph 5.3.6.4, some testing would be needed to demonstrate reusability of the 1.3 cm (0.5 in.) foam system incorporated into the LHe supply tankage.

Transfer line chilldown would also influence overall transfer time and initial conditions at the receiver inlet. The associated state-of-the-art and further work required are discussed in Paragraph 2.3.1.

Also, where LN_2 is used for pre-chill, work will be needed to demonstrate that the LN_2 can be completely purged from the system without formation of harmful solids.

5.4.3 SATELLITE CONTROL SECTION (SCS) - This supply system employs surface tension channels of the same basic design as used for the Shuttle N_2O_4 and MMH supply. The same technology items as discussed in Paragraph 5.3.1 would apply here, oriented to use with N_2H_4.

The main problem which is unique to supplying the SCS is that surface tension channels located in the SCS must be filled at low-g. The baseline approach is to assume these channels are full at the start of transfer and to then transfer in a manner to maintain them full. The technology required to insure that surface tension systems are full at the completion of low-g resupply has not been developed.

Basic development work is required to define systems and/or procedures which can be relied on to insure low-g liquid fill of surface tension systems such as are located in the SCS.

5.5 SUMMARY OF RECOMMENDED TECHNOLOGY WORK

A summary listing is presented below of the most pertinent work required to develop the four transfer systems defined in Paragraph 4.2.

1. Receiver Chilldown and Fill (Cryogenic and Non-Cryogenic)
   a. Analytical Model Development
   b. One-g Thermodynamic Testing
   c. Drop Tower Testing
   d. One-g Prototype Demonstration of Practical System(s)
   e. Orbital Demonstration

2. Surface Tension Screen Systems (Cryogenic and Non-Cryogenic)
   a. Develop Low-G Refill Capability for Supply Channels and Receiver Channels and Baskets
   b. Demonstrate Compatibility With Realistic Vibration And Thermal Environments, Including Integration With Operational Type Tank Pressure Control Systems
   c. Orbital Demonstration of Complete Supply System Concept

3. Paddle Vortex Liquid Orientation
   a. Demonstrate Feasibility and Generate Basic Design Data in Subscale One-g Tests
   b. Overall System Analysis and Design
   c. One-g Prototype Testing and Orbital Demonstration

4. Low-g Pressure Control - Orbital Demonstration of Bulk Heat Exchanger Type Vent System

5. Low-g Outflow to Improve Prediction and Minimization of Liquid Residuals
   a. Analytical Model Development
   b. One-g and Drop Tower Testing
   c. Orbital Demonstration

6. Investigation of Special Problems of Helium Transfer
   a. Demonstrate Compatibility With Screen System
   b. Investigate Practicality of Other Than Liquid Transfer
   c. Develop Methods for Improved Thermal Chilldown Efficiency

7. Low-G Boiling, Condensation, Convection, and Two-Phase Flow Heat Transfer - Orbital Experimentation Required

8. Orbital Demonstration of Low-g Mass Gauging

9. Orbital Demonstration Test of Overall Transfer Concept

   a. Prototype Hardware One-g Tests
   b. Instrumentation/Observation Demonstration
   c. Development of Orbital Test Techniques

6

ANALYSIS OF SHUTTLE/TUG FLUID TRANSFER BENEFITS

This section presents the results of work to quantify some of the benefits of employing transfer systems 1, 2, and 3 which were defined in Section 4.0. The study is designed to determine the benefits of in-orbit fluid supply of the Shuttle Orbiter and/or the Space Tug from tanks located in the cargo bay of a supply Orbiter. The Orbiter to be supplied is called the receiver Orbiter, while the tanker is called the supply Orbiter.

The study is divided into three parts; (1) performance improvements, (2) economic benefits and (3) supply module design considerations with respect to realizing items 1 and 2.

6.1 PERFORMANCE IMPROVEMENTS

Potential performance gains of in-orbit fluid supply to the Space Tug and/or the Shuttle Orbiter are described in the following paragraphs.

6.1.1 SHUTTLE ORBITER - Figures 6-1 and 6-2 show the potential gains in performance to circular orbit altitudes of in-orbit supply of OMS fluids (N_2O_4, MMH, He). Shuttle performance is based on the data presented in Table 6-1 (Ref. 6-1 & 6-2), and Reference 6-3. It is assumed that the entire receiver Orbiter payload is returned to the ground. Whenever payloads are carried to altitudes above 833 K_m(450 n.mi.) the Orbiter returns to 833 K_m (450 n.mi.) apogee and then deorbits directly. A typical sequence of events is presented in Table 6-2 and illustrated in Figure 6-3.

The supply module used , is presented in Paragraph 4.2.3.5 as Case 1. The dry weight is 632 Kg (1393 lb) and the auxiliary fluids required, other than the N_2O_4 and MMH propellants actually transferred, are 500 Kg (1102 lb).

In order to maximize the receiver Orbiter payload to a given final circular orbit, it was determined to be optimum to supply the maximum amount of OMS propellants at a maximum fluid transfer orbit. Optimum fluid transfer orbits, as a function of receiver Orbiter payload deployment altitude, are given in Figure 6-4. For maximum fluid supply, it was found best to carry empty kits on the receiver orbiter and to allow depletion of the main OMS tanks to the point where only enough propellants are left to allow re-entry of the receiver Orbiter in case something happened to prevent the fluid supply from taking place. The payload steps in Figures 6-1 and 6-2 represent the weights of the empty OMS kits, which are payload chargeable.

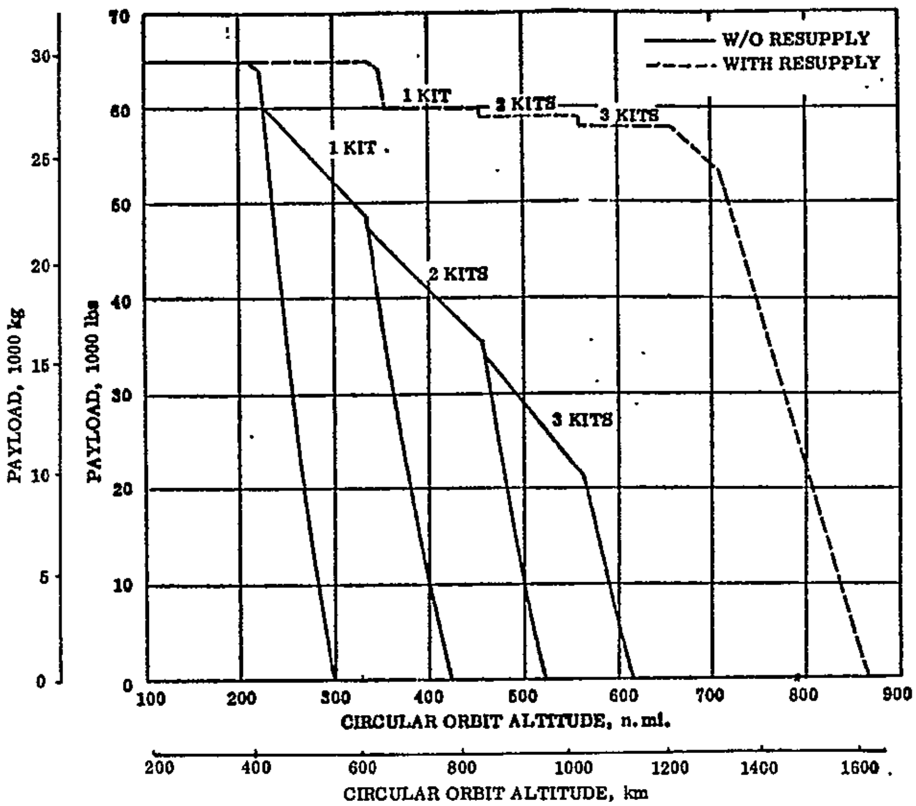

Figure 6-1. Shuttle Orbiter Performance With and Without In-Orbit Fluid Supply (Launch From ETR)

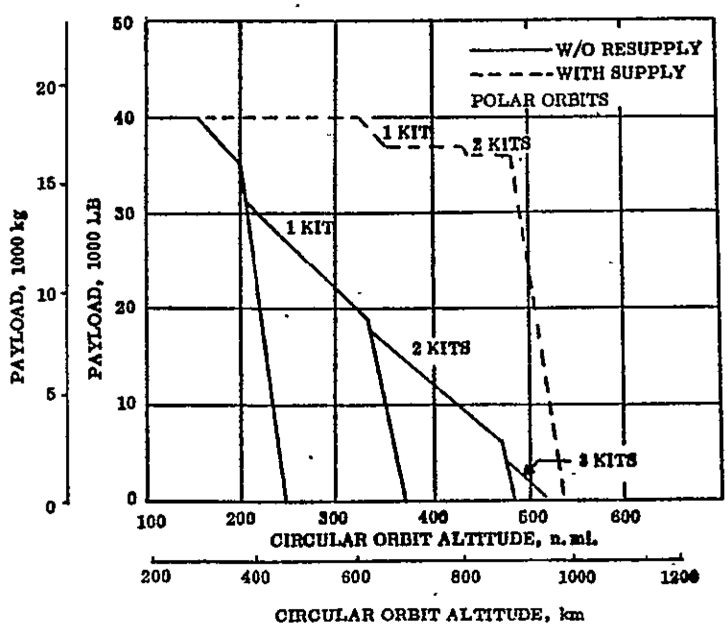

Figure 6-2. Shuttle Orbiter Performance With and Without In-Orbit Fluid Supply (Launch From WTR)

Table 6-1. Shuttle Orbiter Data (Ref. 6-1 and 6-2)

	Weights, kg (lb)			
Inert Weight	68,625	(151,156)		
Personnel	1,200	(2,644)		
Payload Accommodations	204	(450)		
Residual and Unusable Fluids	682	(1,503)		
Reserve Fluids	667	(1,469)		
Return Weight			71,379	(157,222)
Inflight Losses	1,002	(2,208)		
RCS Loading	1,946	(4,286)		
Orbiter OMS Ignition Weight			74,327	(163,716)
Main OMS Propellant (Max)	10,962	(24,146)		
First Kit Propellant	5,684	(12,520)		
Second and Third Kit Propellant (each)	5,618	(12,375)		
First Kit Weight	1,428	(3,145)		
Second and Third Kit Weight (each)	454	(1,000)		
Performance				
OMS Specific Impulse, sec			313.2	
Suborbit Velocity Rqmts, m/s (ft/s)		ETR	30.5	(100)
		WTR	107.0	(350)
Transfer Orbit to 185 km (100 n.mi.) Circular Velocity, m/s (ft/s)			27.8	(91)
Flight Performance Reserve, m/s (ft/s)			6.7	(22)

It was also determined to be optimum for the supply Orbiter to not carry any kits. However, for fluid transfer orbits above 389 km (210 n.mi.) from ETR and 296 km (160 n.mi.) from WTR some propellants from the supply module are used to supply the supply Orbiter OMS engines. The total fluid used in this manner is 4540 to 9080 kg (10,000 to 20,000 lb), depending on the final fluid transfer orbit. A discussion of modifications required to the basic supply module to accomplish this is presented in Paragraph 6.3.1.1.

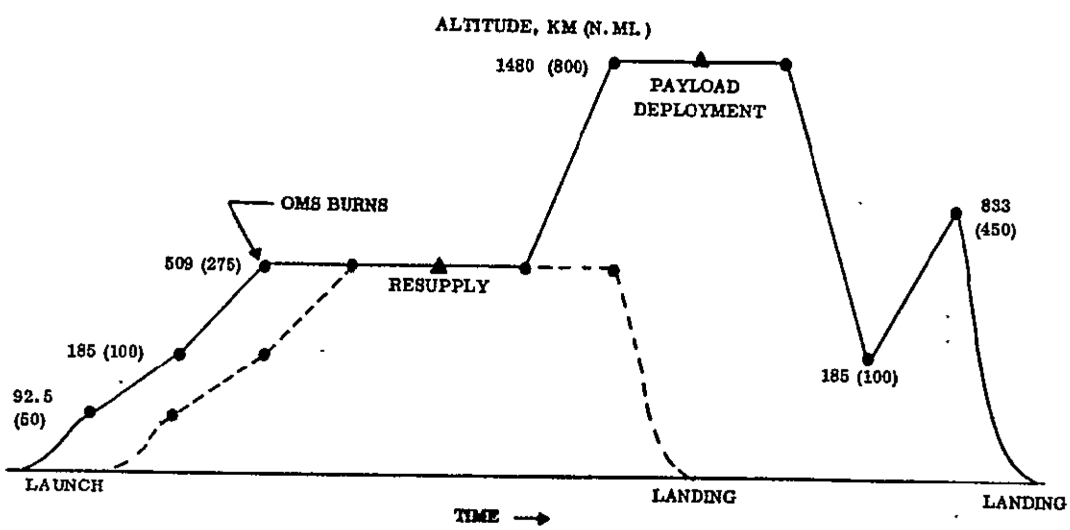

Figure 6-3. Typical Orbiter Resupply Sequence of Events

Table 6-2. Orbiter Resupply Sequence of Events (Payload Orbit - 1480 km, 800 n.mi.; Resupply Orbit - 509 km, 275 n.mi.)

Event	Supplier Orbiter Altitude		Receiver Orbiter Altitude	
	km	n.mi.	km	n.mi.
Launch	0	0	0	0
Orbiter Ascent Ellipse	93/185	50/100	93/185	50/100
Resupply Transfer Ellipse	185/509	100/275	185/509	100/275
Resupply Orbit	509/509	275/275	509/509	275/275
Payload Transfer Orbit	-	-	509/1480	275/800
Payload Deployment Orbit	-	-	1480/1480	800/800
Transfer Down - Ø1	-	-	1480/185	800/100
Transfer Down - Ø2	-	-	833/185	450/100
Deorbit From	509	275	833	450

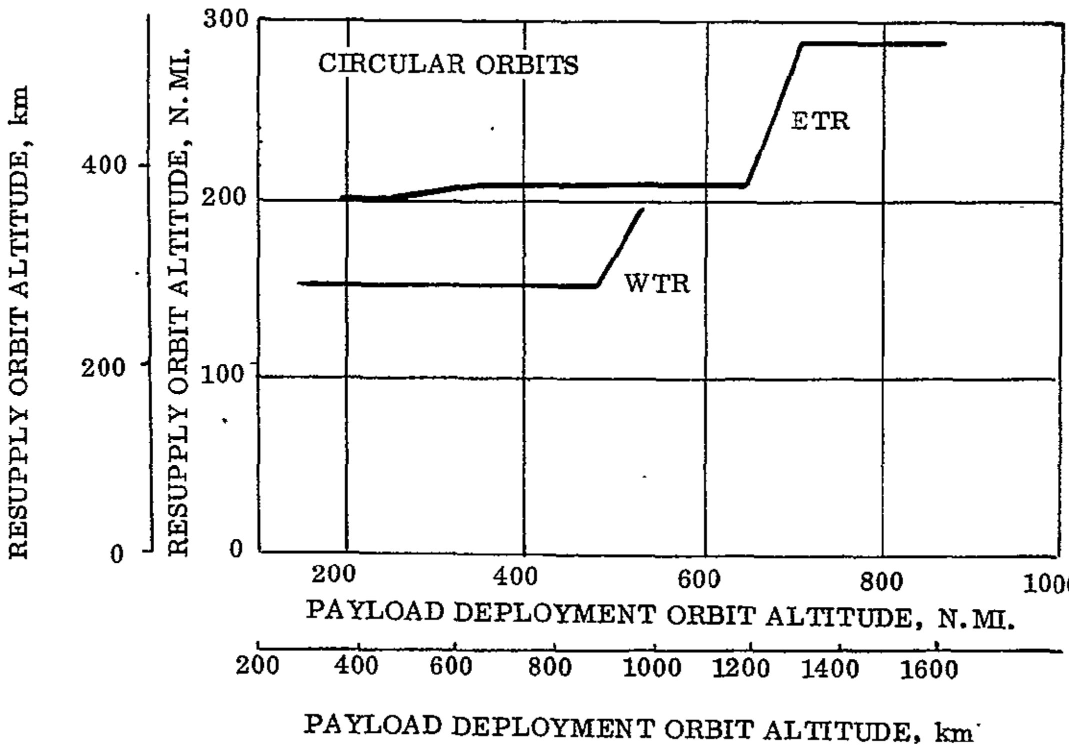

Figure 6-4. Supply Orbit for Maximum Payload

6.1.2 SPACE TUG - Tug performance may be improved by in-orbit fluid transfer to the extent shown in Figure 6-5. Fluid supply to the Tug is assumed to take place in a 296 km (160 n. mi.) circular orbit, from which the Tug leaves for whatever mission is to be performed. As shown in Figure 4-23, the payload capability of the Tug can be significantly increased by fluid transfer at orbits higher than 296 km (160 n. mi.) and the Figure 6-5 curves could possibly be shifted somewhat. However, the 296 km (160 n. mi.) orbit was chosen as a basic reference or standard operating altitude which is consistent with the current Shuttle operating philosophy.

Tug and kickstage (Tandem TE-364-4) weights and engine characteristics, used in this study, are presented in Tables 6-3 and 6-4. The kickstage data presented in Figure 6-5 is based on its use with a reusable Tug to increase overall performance. The basic fluid supply module is described in Paragraph 4.2.2.4. The module dry weight is 160 kg (3524 lb) and has the capability to supply all the Tug main and APS propellants. Non-transferred and auxiliary fluids total 424 kg (934 lb).

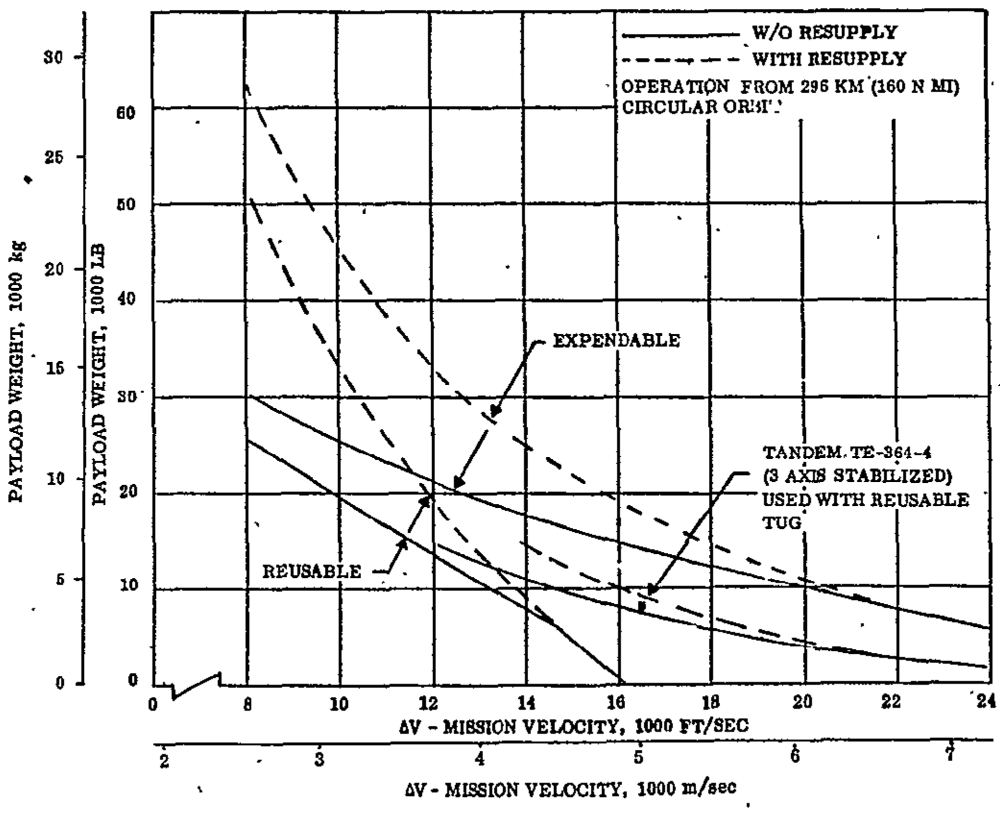

Figure 6-5. Tug Performance With and Without In-Orbit Fluid Supply (Launch from ETR)

Typical Tug missions for which the Figure 6-5 data would apply are payload delivery to synchronous equatorial orbit (ΔV = 4,300 m/s, 14,000 ft/sec), Mars sample return (ΔV = 3,500 m/s, 12,500 ft/sec) and Lunar operations with ΔV's on the order of 3,200 m/s (10,500 ft/sec). Velocities quoted and presented in Figure 6-5 are basic mission velocities assuming a one-way trip. Vehicle velocities required for Tug return in the reusable cases are only reflected in Figure 6-5 by reduced payload capability.

A typical ΔV and APS usage schedule for payload delivery to synchronous Equatorial orbit and for Tug return is presented in Table 6-5.

Table 6-6 presents a comparison of Tug payload capability with and without in-orbit fluid supply for the Mars Sample Return mission. Data are shown in order of

Table 6-3. Tug Characteristics

	Mass Characteristics (Ref. 6-4) kg (lb)			
Ignition Weight				25,778 (56,779)
Propellants - Usable			22,650 (49,889)	
APS			131 (288)	
Inert Expendables			248 (547)	
Burn-Out Weight			2,749 (6,055)	
Flight Performance Reserve	136	(300)		
Unusable Residuals	275	(605)		
Dry Weight	2,338	(5,150)		

Engine Characteristics (Ref. 6-5)

Mode	Thrust, kg (sec)	Specific Impulse, sec
Full Thrust	6,810.0 (15,000)	456.5
Pump Idle	1,703.0 (3,750)	439.5
Tank Head Idle	71.3 (157)	384

Table 6-4. Tandem TE-364-4 Performance Data

	Lower Stage		Upper Stage	
Propellant, kg (lbs)	1,056	(2,327)	1,056	(2,327)
Jettison Weight, kg (lbs)	136	(300)	186	(410)
Specific Impulse, sec	281		281	
Average Thrust, kg (lbs)	7,309	(16,100)	7,309	(16,100)
Stabilization	3 axis		3 axis	

Table 6-5. Typical Mission Events, Payload Delivery - Synchronous Equatorial Orbit

	Main Engine Velocity		APS Usage	
	m/sec	(ft/sec)	kg	(lbs)
Separate From Orbiter, 296 km (160 n mi)			4.1	9
Coast			9.5	21
Inject Into Phasing Orbit	1,369	4,489		
Coast			8.2	18
Inject Into Transfer Orbit	1,120	3,672		
Coast			6.4	14
Midcourse Correction	15.3	50		
Coast			6.4	14
Circularize at Geosynchronous	1,781	5,839		
Coast and Orbit Trim			46.8	103
Deploy Payload			14.5	32
Total for Basic Mission	4,285	14,050	95.8	211
Coast			6.4	14
Deboost	1,783	5,847		
Coast			3.6	8
Midcourse Correction	10.7	35		
Coast			3.6	8
Inject Into Phasing Orbit	1,157	3,794		
Coast			3.6	8
Circularize in Final Orbit, 296 kg (160 n mi)	1,295	4,246		
Rendezvous With Orbiter			17.7	39

Table 6-6. Mars Sample Return Tug Payload Capability, Tug Operations From 296 km (160 n mi) ΔV = 3813 m/s, (12,500 ft/s)

Tug	Payload		Resupply	Total Shuttle Flights
	kg	lb		
Reusable	5,448	12,000	None	1
Reusable	7,355	16,200	Tug	2
Expendable	9,080	20,000	None	1
Reusable	11,350	25,000	Tug + Orbiter	2
Expendable	13,892	30,600	Tug	2
Expendable	16,571	36,500	Tug + Orbiter	2

increasing payload capability. It is noted that data are also shown for two cases where a Shuttle Orbiter carrying a Tug and the Tug are both supplied from another Shuttle while in orbit. A significant increase in payload capability is shown for this mode of operation.

By supplying the Orbiter as well as the Tug the Orbiter can increase its orbital energy by going into an elliptic or higher circular orbit from which the Tug can depart, reducing overall Tug energy requirements. This makes full use of the two Shuttle flights (one to carry the Tug and one for in-orbit fluid supply). For example, assuming only the reusable Tug is to be supplied in orbit (Case 2, Table 6-6), the payload is 7355 kg (16,200 lb) and the empty Tug weight, from Table 6-3, 2338 kg (5,150 lb). This allows 1981 kg (43,650 lb) of fluid to be taken up with the Tug while meeting the 29,510 kg (65,000 lb) Shuttle payload limitation. Then from Table 6-3 the maximum amount of fluid needing to be supplied to the Tug in-orbit would be only 3622 kg (7979 lb). This leaves a considerable excess capability for the supply Shuttle. Utilizing this excess capacity to also supply the Tug-carrying-Orbiter results in the data presented in Table 6-6.

The example given above only illustrates the concept. In the final analysis an increase in payload capability reduces the amount of fluid to be carried into orbit with the Tug and thus reduces that available for transfer to the Shuttle, reducing the additional Shuttle energy. Thus an iteration process was employed to arrive at the final numbers presented in Table 6-6.

Supply module(s) have not been designed for the specific case of combination Tug and Shuttle in-orbit supply. A discussion of what would be required is presented in Paragraph 6.3.

6.2 ECONOMIC BENEFITS

Described here are potential economic benefits of in-orbit fluid supply of the Tug and/or Shuttle Orbiter. The cost of development and production of supply modules was not considered. The development of such cost data is beyond the scope of the present study. The following areas are covered.

a. Low Cost Payload Design

b. Extended Duration Shuttle Missions

c. Recovery of Disabled Orbiter

d. Increased Mission Capabilities

Details are presented in the following paragraphs.

6.2.1 LOW COST PAYLOAD DESIGN - Considered here are the cost savings achievable through relaxed constraints on payload weight which may be brought about by employing in-orbit fluid transfer. To provide an example of such potential cost savings, an analysis was made of the particular cost savings possible from applying low cost payload design concepts to the Mars Sample Return mission. The allowable payloads for this mission, with and without in-orbit fluid supply, are presented in Table 6-6. A reusable Tug with Tug-only in-orbit fluid supply is used for the present example. Allowable payloads are then respectively 7355 kg (16,200 lb) and 5448 kg (12,000 lb) for cases with and without in-orbit fluid supply.

From Reference 6-6, the current Mars Sample Return payload design was derived to have a weight of 4994 kg (11,000 lb), a nonrecurring cost of \$598 M and a recurring cost of \$108M. From Reference 6-7, the maximum cost savings which could be realized by employing low cost design concepts was determined to be 29% of the \$598 M nonrecurring cost or \$173M nonrecurring savings and 25% of the \$108M recurring cost or \$27M recurring savings. In order to realize these maximum savings the basic payload weight must be allowed to increase by a specific amount. The required increase depends on the basic (before low cost design) payload weight. This is illustrated in Figure 6-6. From Figure 6-6 it is seen that for the current case (payload = 4994 kg; 11,000 lb) an allowable payload of [1.5×4994 kg (11,000 lb)= 7491 kg (16,500 lb)] or payload growth of 2497 kg (5,500 lb) would be required to realize the maximum low cost design cost savings specified above. The actual weight growths allowable for in-orbit supply and no in-orbit supply cases are respectively 2361 kg (5,200 lb) and 454 kg (1,000 lb). This represents 95% and 18% of the weight growth of 2497 kg (5,500 lb) required for minimum cost design. Figure 6-7 shows the actual achievable cost reduction in % of maximum as a function of the actual payload growth in % of maximum. For the present case of 95% and 18% payload growth, cost reductions from Figure 6-7 are correspondingly 90% and 25% of maximum. That is, final cost reductions possible by the use of low cost payload design concepts are;

	Nonrecurring	Recurring
With In-Orbit Supply	0.9 × \$173M = \$156M	0.9 × \$27M = \$24M
Without In-Orbit Supply	0.25 × \$173M = \$43M	0.25 × \$27M = \$7M

The above cost savings differential between the two cases applied to two Mars Sample Return Missions, minus the added cost of the two Shuttle launches for in-orbit fluid supply, results in a final savings of \$120M for in-orbit supply versus no in-orbit supply. The cost of each Shuttle flight was taken to be \$13.6 M and is based on the \$10.5 M of Reference 6-10 escalated 30% to 1975 dollars.

The overall savings derivation is summarized in Table 6-7.

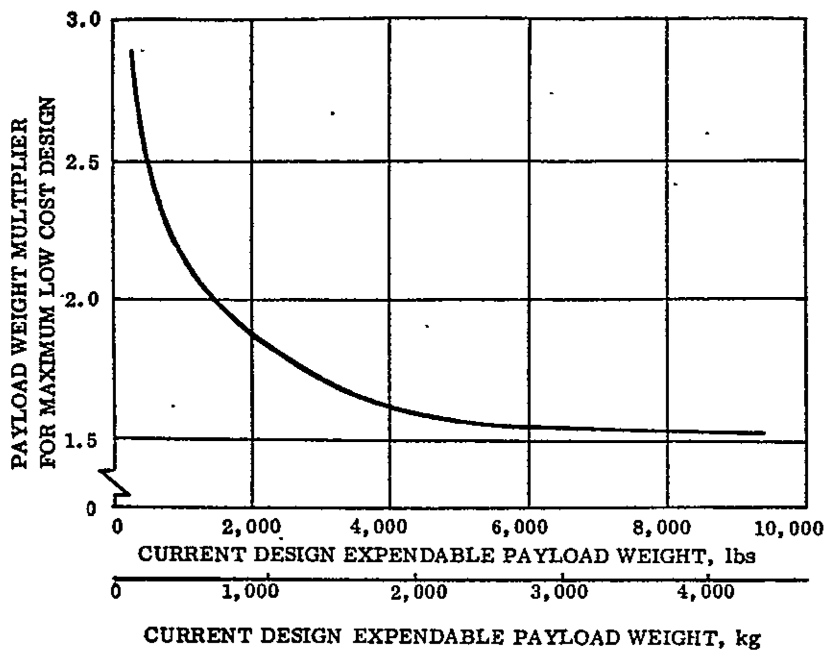

Figure 6-6. Low Cost Payload Design Weight Growth Relationships (Ref. 6-8)

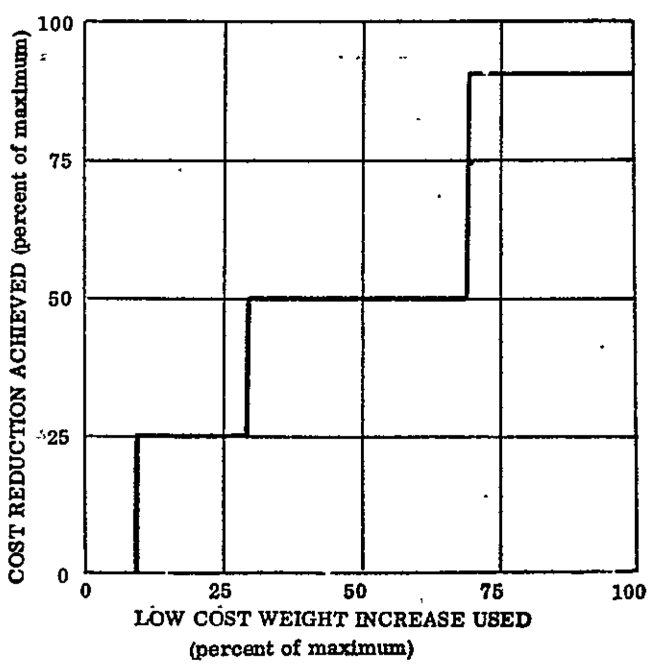

Figure 6-7. Nonrecurring and Unit Cost Payload Effects Relationship (Reference 6-9).

Table 6-7. Mars Sample Return Mission Cost Savings

Current Design		
Payload Weight, kg (lb)	4994 (11,000)	
Costs		
Non-Recurring	$598M	
Recurring	$108M per unit	
Low Cost Design		
Payload Weight (Optimum), kg (lb), Fig. 6-6	7491 (16,500)	
Savings (maximum)		
Non-Recurring (29% of basic)	$173M	
Recurring (25% of basic)	$27M per unit	
Achievable Savings with Shuttle/Tug		
	Without Tug Resupply	With Resupply
Payload Capability, kg (lb)	5448 (12,000)	7354.8 (16,200)
Payload Growth-Allowable, kg(lb)	454 (1,000)	2360.8 (5,200)
Payload Req'd Growth for Max. Cost Reduction kg(lb)	2497 (5,500)	2497.0 (5,500)
Achievable Wt. Growth, % of Optimum	18%	95%
Achievable Cost Reduction, % Max. Savings, (Figure 6-7)	25%	90%
Achievable $ Savings		
Non-Recurring	$43M	$156M
Recurring	$7M per unit	$24M per unit
Total (2 Units)	$57M	$204M
Added Shuttle Flights (2)*		$27M
Net Savings	$57M	$177M
Difference		$120M Savings

*$10.45M (1971$) Escalated 30% to $13.6M (1975$)

This brief review of just one mission with marginal Shuttle/Tug performance capability was done to illustrate a potential benefit of in-orbit fluid supply. Savings of a similar nature can be accommodated on all missions where Shuttle/Tug performance in the nominal mode is taxed.

6.2.2 EXTENDED DURATION SHUTTLE MISSIONS - There is a strong desire to extend the shuttle sortie on-orbit operations beyond the standard 7 day orbital time period. An identification of the benefits of extended duration and the rationale for times longer than the seven days are presented in Table 6-8 from Reference 6-11. The number of experiments which desire extended duration missions fall into the 75% to 100% range for most payload disciplines.

Extended duration sortie missions require the basic space lab equipment, spacelab mission dependent equipment, transfer tunnel, and payload chargeable orbiter equipment to all be returned from orbit. These support weights are a significant portion of the 14530 kg (32,000 lbs) orbiter landing constraint weight, leaving less than 7,260 kg (16,000 lbs) for available scientific payload weight. As the mission is extended beyond the standard seven day mission, additional life support expendables are required for the crew, and additional power supply is needed for the experiments. At a maximum 4 KW power usage for the payload (in addition to 12.5 KW and 3.0 KW needed by the orbiter and spacelab respectively) additional Electrical Power Supply (EPS) kits are required after seven days at the rate of one additional kit each 1.7 days. These kits in turn reduce the useful payload weight down to zero at a mission duration of thirty days as shown in Figure 6-8.

Also shown in Figure 6-8 are the minimum experiment costs in $/Kg/day ($/lb/day) for different experiment durations. These data are obtained by dividing the cost of a Shuttle flight ($13.6 M, Ref. Paragraph 6.2.1) by the experiment weight and maximum days in orbit as determined from Figure 6-8. It is seen that as the time in orbit

Table 6-8. Payload Benefits From Extended Sortie Missions
(Ref. 6-11)

Extended Duration Mission Benefit	Rationale for Extended Mission	Payload Applicability
Mapping/ Surveys	More observables/coverage per mission	Astronomy, High Energy, Earth Obs, Ocean Physics, Space Tech, Comm/Nav
Solar Related	Can observe total sun in 15 days; 30 days desired to follow sun spots.	Solar Physics
	15-30 days reasonably maximizes solar effects on Earth's Atmosphere	Atmos, Magnet, and Space Physics
Low Frequency Events	Additional Time to	
	- Collect low flux particles with limited intercept area	High Energy
	- View solar flares, cosmic and gamma rays, neutrons	Solar Physics
	- Observe meteor trails	Space Technology
Observable Dynamics	Cloud physics and atmosphere pollution benefits from changing conditions over longer missions	Earth Observation, Space Technology
	Dynamics of earth/ocean processes more completely observed	Ocean Physics
Truth Site Interaction	Additional viewing opportunities over limited no. available in 7-day mission	Ocean Physics, Space/ Technology, Comm/Nav
Experiment Cycle	Specimen imperfection reduced by lower growth rate over longer cycle	Space Processes
Product Yield	Later mission payloads want to maximize yield of production	Space Processes
Cardiovascular De-conditioning	Long-duration data required	Life Sciences
Fluids and Electrolyte Compartment-alization	Need to investigate long-term effects	
Circadian Rhythm	Need long duration to study rhythmic biological cycles to separate diurnal, lunar, solar cycle effects	
Growth and Genetics	Need several parent - progency cycles for plants and vertebrates	
Man Psycho-motor Perf.	Need long-duration zero-g to evaluate changes in man's psychomotor performance	

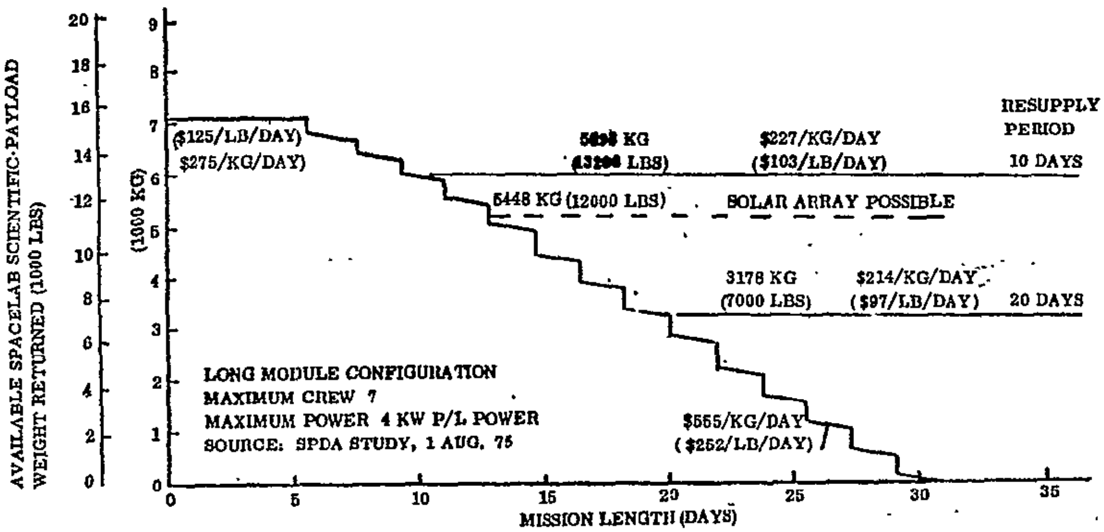

Figure 6-8. Extended Duration Spacelab Mission

increases, the experiment cost per unit weight and time decreases to a point where it then begins to increase.

The useful payload may be retained through fluid resupply of the supercritical oxygen and hydrogen EPS and ECLSS expendables. Resupply of these expendables every ten days will allow 5993 kg (13,200 lbs) of experiment to be carried, while resupply every 20 days will allow 3178 kg (7000 lbs) of experiments to be carried as shown on Figure 6-8. The experiment cost per unit weight and time remains the same for resupply as for no resupply, the advantage of resupply being that the experiment time can be extended indefinitely without intervention or additional unit cost. This assumes that the cost of a second Shuttle flight for resupply is the same as returning the spacelab to the ground, refurbishing, and relaunching. Another advantage of resupply is that all the payload capacity of the resupply Shuttle is not required so that other payloads could possibly be accommodated at the same time.

The Shuttle Orbiter supply module Case 2 (Paragraph 4.2.3.5) could be used for this application, except that only a small quantity of OMS fluids would likely be carried, possibly to provide additional RCS propellants to the experiment Orbiter.

Several solutions (other than fluid transfer) to the problem of extended duration mission and orbiter landing weight constraint were addressed in Reference 6-11, including throw away EPS bottles and deployable solar arrays. Fluid transfer with throw away EPS bottles would also be useful for long duration missions as the frequency of the resupply flights could then be reduced. For experiment weights less than 5448 kg (12,000 lbs) use of a large solar array would eliminate the need for resupplying the EPS fluids; however, RCS propellants would have to be resupplied after about 30 days.

There are also other potential problems with the solar array approach, as listed below:

a. Due to their large size they might; (1) block the payload view, (2) block the thermal radiation panels such that orientation of the Shuttle would be critical, (3) restrict experiment deployment, and (4) cause higher total orbiter drag and thus RCS usage.
b. RCS exhaust may contaminate solar panels.
c. Due to size and deployment requirements, storage of panels may be a problem.

6.2.3 RECOVERY OF DISABLED ORBITER - Should the Orbiter through some unfortunate circumstance not be able to return and land at the end of a planned mission, fluid resupply of the Orbiter may be necessary to (1) resupply the RCS and OMS propellants needed for control and de-orbit, and (2) resupply of the supercritical hydrogen and oxygen for electrical power and life support. This could be accomplished using the Shuttle Orbiter supply module Case 2 (Paragraph 4.2.3.5). Fluid resupply for Orbiter recovery may be needed due to unscheduled time extension in orbit causing depletion of reserves of RCS/OMS, EPS or ECLSS fluids. Fluids may also be lost due to leakage and or repair operations.

Rescue of the astronauts, if needed, would be accomplished by another Shuttle launch; however, return of the Orbiter must be accomplished by in-orbit repair of the malfunction and replenishment of expended fluids. An economic measure of the value of fluid resupply to the Orbiter may be equated to the costs incurred due to the loss of an Orbiter should fluid transfer not be available.

The first cost incurred due to an Orbiter loss is the replacement of the Orbiter itself. Replacement of the Orbiter will be \$450 to \$500 million (1975 dollars). These "Orbiter Replacement" costs are based on Reference 6-12 data with production Rate Effects Ratios (RERs) applied to reflect the cost impact of not having the benefit of concurrent production of other orbiters. The RERs were based upon prior extensive analyses of launch vehicle production rate effects.

Additional unplanned Shuttle launches would be required to provide personnel transfer, repair capability, payload transfer, salvage, and/or additional fluids. The most significant secondary effect would be the realignment of launch schedules caused by the failure of the Orbiter and the unplanned Shuttle launches required to support orbital repair and/or recovery. "Launch Schedule Realignment" costs are estimated at about \$0.6 million per payload per year, and reflect spacecraft/experiment related costs for maintaining capability as well as cost associated with storage and refurbishment for launch. In developing the "Launch Schedule Realignment" costs, it was assumed that the launch realignment occurred subsequent to payload development and fabrication. An analysis of Centaur minimum manpower requirements was the source for capability maintenance while Centaur inventory analysis gave an index of storage and refurbishment costs.

Table 6-9. Costs Incurred Due to Disabled Orbiter in Space

Cost Item	Fluid Resupply Not Avail.	Fluid Resupply Available
Orbiter Replacement	$450M	-
Rescue Flight	13.6	-
Repair Flight	-	$13.6M
Resupply Flight	-	13.6
Rescheduled Flights:		
Repair & Resupply Flights (2)	-	1.2
Rescue Flight	0.6	-
Flights Reschedule Due to Lost Orbiter (20 Flts/Yr for 3 Yrs)	36	-
Total	$500M	$28M
Savings Due to Resupply		$472M

If it is assumed that one of three Orbiters is not available for a replacement time period of three years, then a total of 60 flights (1/3×3 yrs × 60 fls/yr) will not be flown (delayed) for a total cost of $36 million.

A summary of the overall costs incurred due to a disabled Orbiter in space and potential savings of in-orbit fluid supply are presented in Table 6-9.

6.2.4 INCREASED MISSION CAPABILITIES

Mission requirements were reviewed (References 6-13 through 6-17) to identify missions where fluid resupply of either the Orbiter or the Tug would be useful. It was generally found that the missions were within the basic Orbiter/Tug performance envelopes illustrated in Figure 6-5 without resupply. This is, however, to be expected since the mission designers would have been aware of, and would have designed their missions to be compatible with the anticipated transportation systems.

However, there have been several missions which have undergone reduced mission requirements when it was found that they could not meet transportation capability. Typical of the automated spacecraft missions is the Mars Surface Sample Return mission which in June 1973 was listed in Reference 6-13 as a 1100 kg (24,000 lb) mission requirement, but was reduced to 3300 kg (7300 lb) by October, Reference 6-14, through using direct Mars entry and elimination of a rover vehicle. This mission is now listed as requiring 5000 kg (11,000 lb), Reference 6-15. Referring to Table 6-6, it is seen that the original mission requirement could be met by either a reusable Tug with in-orbit supply of both Tug and Orbiter fluids or an expendable Tug with in-orbit supply of only the Tug fluids. In no case can the original requirement be met with the Tug without in-orbit fluid supply.

Typical of Sortie missions which have been adjusted to meet existing payload limitations is the 30 m IR Interferometer (AS-09) payload with a length of 16.5 m (54 ft) and listed with a desired 740 km (400 n.mi.) circular orbit altitude per Reference 6-16 in 1974. But in 1975, Reference 6-17, the desired altitude is reduced to 400 km (215 n.mi.). The initial requirements in 1974 for payload AS-09 caused a conflict; the altitude requirement necessitated the Orbiter use of a single OMS kit, but the remaining 15.3 m (50 ft) of cargo bay availability was too short for the payload requirement of 16.5 m (54 ft). The OMS kit length is 3.1 m (10 ft). The experiment weight was less than 4540 kg (10,000 lb) and was thus not a problem. In-orbit fluid supply would allow the Orbiter to fly to the 1974 higher desired altitude without the use of OMS kits (see Figure 6-1).

It is anticipated that once the performance envelope of the Shuttle and Tug are expanded, some planned missions will grow to the new capability and new missions will be conceived which will fully utilize the new performance capability.

6.3 SUPPLY MODULE DESIGN CONSIDERATIONS

In the two preceding paragraphs it was found that in order to reap maximum benefits, the supply modules defined in Section 4.0 would need to be modified and/or operated under off-design conditions. For example, to realize maximum Tug performance, in-orbit supply of both the Tug and Shuttle Orbiter from a single supply Orbiter would be required. Also, in certain cases, it was found desirable to increase the fluid transfer altitude by using some of the supply module fluids to supply the supply Shuttle OMS.

Supply module design implications of the various supply operations required to realize Shuttle and Tug performance and economic benefits are discussed in the following paragraphs.

6.3.1 SHUTTLE ORBITER SUPPLY SYSTEM - As described in Paragraph 4.2.3.5, there are two basic supply cases considered. Case 1 assumes all OMS fluids (N_2O_4, MMH and He) are to be transferred, including that required to fill three add-on kits. Case 2 assumes all EPS/ECLSS cryogenic H_2 and O_2 are supplied, including six kits, with the OMS N_2O_4 and MMH off-loaded as necessary to meet the Shuttle 29,510 kg (65,000 lb) payload limitation. In both cases it was assumed that all the OMS He is supplied. A single module support structure is employed for both cases, the only difference being that the H_2 and O_2 supply tankage are not carried in Case 1.

6.3.1.1 OMS Supply (Case 1) - To realize the increased performance of the Shuttle as described in Paragraph 6.1.1, the Case 1 Orbiter supply module would be used to supply OMS fluids. The maximum N_2O_4 plus MMH fluid supply capability of this module is 28,280 kg (62,290 lb). In most of the actual cases considered here - somewhat less than the maximum is needed. Potential supply module fluid ranges are from near empty to near full.

The main problem would be associated with the near empty cases, where special attention would be required to insure that the surface tension channels are full at the initiation of transfer. With only a partial load of liquid the chances of spilling some liquid from the channels along with possible screen drying during boost, orbit injection, rendezvous, and docking are magnified. Potential solutions are to; (1) design the system to maintain full lquid channels under all fluid loading and environment conditions, (2) provide means to refill the channels at low-g just prior to transfer and (3) carry more than the needed quantities of fluids to increase the chances of the channels remaining full. The technology requirements to maintain the channels full or to refill them at low-g are discussed in Section 5.0.

In addition to the basic supply operation it was found that under certain conditions it is desirable to increase burn time or altitude of the supply Orbiter by using some OMS propellants directly from the supply module. Such fluid use is accomplished when fluid transfer orbits above 389 km (210 n. mi.) for ETR and 296 km (160 n. mi.) for WTR are desired (Reference Paragraph 6.1.1). 389 km (210 n. mi.) from ETR and 296 km (160 n. mi.) from WTR are the maximum circular orbits the basic Shuttle can attain without adding kits. It was, however, determined to be better to supply the OMS from the supply module rather than adding kits to the supply Orbiter.

This would require some modification to the basic supply module. The total amount of propellant to be supplied to the OMS engines from the supply module is 4540 to 9080 kg (10,000 to 20,000 lb), depending on the final orbit desired. Three different methods of accomplishing this OMS supply were considered, as described below:

1. Shutdown the OMS engines and add fluid to the main OMS supply tanks. In this case fluid transfer must be through the screened channels since fluid settling will not be available. Assuming no basic change to the fluid supply tanks, transfer will be at normal in-orbit transfer rates. A boost pump will be required in each supply system (oxid. & fuel) to boost the pressure above the OMS pod tank pressure. Additional plumbing to the aft bulkhead disconnects are required.

2. Transfer while OMS engines are firing, using boost pumps and leaving the supply tanks at their existing design pressure (345 kN/m^2, 50 psia). The existing fill and dump line can be used for the supply tank outlet since there will be thrust on the Orbiter and the fluids will be settled.

3. Transfer while OMS engines are firing without using boost pumps, by designing the supply module tankage to accommodate the OMS feed pressure required (1.76 MN/m^2, 255 psia).

Comparative weights of the three concepts are presented in Table 6-10. The weights presented are only those in addition to the basic supply module weights presented in Paragraph 4.2.3.5. In each case, 9080 kg (20,000 lb) of propellants were assumed to be transferred in this manner.

It is seen that concept number 3 requires a high additional weight, while concept number 1 has the lowest weight. The main disadvantage with 1 is that the OMS engines must discontinue firing while the main OMS tanks are being resupplied. Based on present information the choice of concept 1 appears to be the best compromise between weight and operational complexity.

6.3.1.2 OMS and/or EPS/ECLSS Supply (Case 2) - Shuttle Orbiter supply Case 2, where both OMS (N_2O_4, MMH, He) and EPS/ECLSS (H_2, O_2) fluids are supplied,

Table 6-10. Additional Weights to Supply OMS Engines from In-Orbit Supply Modules

	System Concept					
	1		2		3	
	kg	lb	kg	lb	kg	lb
Pumps (2 Units)	24	53	55	122	–	–
Tankage	–	–	–	–	547	1205
Plumbing	14	30	21	47	21	47
Total	38	83	76	169	568	1252

is applicable to extended duration Shuttle missions (Paragraph 6.2.2) and to recovery of a disabled Orbiter (Paragraph 6.2.3).

For extended duration Shuttle missions, which support Sortie experiments, the cryogenic H_2 and O_2 would be the primary fluids supplied. In a limited number of cases, small amounts of OMS/RCS fluids would also be required. Where only H_2 and O_2 are required, the supply module could be easily designed to allow removal of the N_2O_4, MMH and LHe supply tanks. Per Paragraph 4.2.3.5, this would result in a significant weight saving, which could be utilized for the transport of other payload. In any case, the quantities of fluids required would be a variable and the main problem would be that of insuring full screen channels at the start of transfer. This is further discussed in Paragraph 6.3.1.1.

For recovery of a disabled orbiter the Case 2 Orbiter supply system would likely be used as designed. Again, however, the required fluid quantities could be a variable.

6.3.2 TUG SUPPLY SYSTEM - Either the Shuttle drag or separated Tug supply systems could be used to obtain the performance improvements illustrated in Figure 6-5. These systems are described in detail, respectively, in Paragraphs 4.2.2.4 and 4.2.1.5. For transfer in the 296 km (160 n. mi.) baseline orbit the drag system is used, since it has a slightly lower weight and less operational complexity than the separated system.

As illustrated in Figure 4-23, the payload capability of the Tug could be increased if fluid transfer were accomplished at higher than 296 km (160 n. mi.). Further study would be required, based on specific mission requirements, to determine if transfer at higher altitudes would be warranted. In this case, due to reduced drag at higher altitudes, the separated supply system with auxiliary propulsion may prove to be best.

In either case the quantities of propellants required to be supplied may vary considerably. However, for neither of these linear acceleration settling systems should any special problems arise. Initial liquid settling times may be somewhat different, depending on the quantities of fluids involved, however, the basic system design was conservative to the point where this should not be a significant factor.

6.3.3 TUG AND SHUTTLE SUPPLY - As discussed in Paragraph 6.1.2, Tug performance can be increased, over that of Tug supply alone, by in-orbit supply of both the Tug and Orbiter. A system to accomplish this from a single supply Orbiter was not designed under the current contract.

As presently designed, the Tug and Shuttle supply modules would not both fit into the Orbiter cargo bay. By removal of the O_2 and H_2 tanks and associated support structure of the Shuttle supply module the two modules could however be made to fit.

Use of the separated Tug supply module in conjunction with the Orbiter supply module would require a complex operating procedure. The Tug would need to be separated from its Orbiter, docked with the supply module, separated for transfer, returned with the supply module to the supply Orbiter, and undocked and returned to its original Orbiter. In addition, the two Orbiters would need to be docked together for transfer of OMS fluids. Operations with the drag system would be slightly less complex, but would still require study to determine overall feasibility.

A more likely candidate for the combination transfer would be to use screen surface tension supply for all the fluids. The basic operation would then be essentially the same as presently proposed for supply of the OMS fluids. The required tankage would fit into the Orbiter cargo bay.

In any case, a system design would need to be specifically developed for the combination transfer. However, once the basic fluid technology work recommended in Section 5 is accomplished, the design of such a system should be within the state-of-the-art.

6.4 CONCLUSIONS

As a result of the foregoing studies, the following conclusions are made.

a. In-orbit fluid supply can increase the Shuttle performance envelope by 75%.

b. Tug performance can be significantly increased for most missions. For example, for in-orbit supply of the Tug-only, for the Mars Sample Return Mission, payload can be increased by 35% for a reusable Tug and by 53% for an expendable Tug. With in-orbit supply of both the Tug and Orbiter, payload increases of 108% for a reusable Tug and 83% for an expendable Tug are possible.

c. Applying low cost design concepts to two Mars Sample Return Missions results in an estimated saving of \$120M for resupply over no resupply.

d. Applying in-orbit fluid supply to recovery of a single disabled Orbiter can result in a savings of \$472M.

e. In-orbit supply of cryogenic H_2 and O_2 and some OMS/RCS fluids can extend uninterrupted spacelab missions indefinitely.

f. The basic transfer systems conceptually designed and described in Section 4.0 can be utilized to reap the benefits described above; except where both the Tug and Shuttle are to be supplied by a single supply Orbiter, significant design modifications would be required. However, assuming development of the individual Tug and Shuttle in-orbit transfer concepts per the recommendations of Section 5.0, design of a combination system should be within the state-of-the-art.

7

CONCLUSIONS AND RECOMMENDATIONS

This section presents the major conclusions and recommendations of a program to identify technology gaps, system characteristics, components, and operations critical to the design and performance of efficient and predictable in-orbit fluid transfer systems.

The primary problem of transferring fluids in space is the absence of unbalanced body forces to provide a natural orientation of the liquid and vapor in a tank. This results in requirements for systems to orient or collect the liquid to be transferred and for receiver tank vent systems that prevent excessive liquid loss.

For purposes of this study, transfer systems are defined in terms of the method used for liquid acquisition in the supply, however, a complete system consists of supply storage, transfer lines and up to three different receivers; as well as auxiliary support systems such as required for tank pressure control and venting. Both cryogenic and noncryogenic fluids are included and the supply module is assumed to be a payload of the Space Shuttle manned transportation system.

The overall study was made up of the individual tasks listed below.

a. Literature analysis to provide a sound base for all subsequent work.

b. Technology evaluation, in general terms, of the adequacy of existing technology to design cryogenic and noncryogenic in-orbit fluid transfer systems.

c. Receiver configurations and characteristics definitions to determine which receivers would need or could benefit from in-orbit fluid transfer, along with their relevant characteristics and specific transfer benefits.

d. Transfer systems studies to conceptually design overall transfer systems that appear most likely to provide efficient and predictable in-orbit supply of representative receivers determined in c. above. As a result of initial definition and screening, four different transfer systems were conceptually designed, as listed below.

System 1	Space Tug Supply (LH_2, LO_2, N_2H_4, He) with linear acceleration of supply module and Tug separated from the Shuttle Orbiter.
System 2	Space Tug supply (LH_2, LO_2, N_2H_4, He) with linear acceleration from Shuttle drag with the Tug attached to the Orbiter.
System 3	Space Shuttle Orbiter supply (N_2O_4, MMH, He, H_2, O_2) using surface tension screens for liquid acquisition.

System 4 Multiple Receivers supply of the Solar Electric Propulsion Stage (Hg) using a diaphragm, Large High Energy Observatory-B (LHe) using a paddle for liquid acquisition, and the Satellite Control Section (N_2H_4) using surface tension screens.

e. Systems evaluation to determine technology requirements and programs necessary for final design and development of the specific transfer systems defined in d. above.

f. Analysis of Shuttle/tug fluid transfer benefits as to specific performance improvements and potential cost savings of in-orbit fluid supply using supply systems 1, 2 and 3 defined in d. above.

Overall study conclusions and recommendations are presented in two parts; (1) general technical conclusions based on the work described in c, d and f above and (2) technology recommendations based on the work described in b. and e. The general technical conclusions are listed below.

a. There are a large number of existing and future space systems which would need or could benefit from in-orbit fluid transfer. Twenty-nine representative systems were so identified under the receiver configurations and characteristics task. In general, cost effectiveness (reduced cost, increased performance and/or mission capability) and safety are the benefits which can be realized. Other observations, based on the 29 receivers, indicated that liquid oxygen would be the predominant fluid, by mass, used in space; with hydrogen a close second. Hydrazine was used on the greatest number of different receivers. An average of three different fluids per spacecraft was used. Frequency of resupply was not considered. The number of applications of cryogenics and noncryogenics was about equal. In many cases, the resupply of one type of fluid without the other would limit the potential transfer benefits.

b. In-orbit fluid supply can increase the Shuttle performance envelope by 75%. Applying in-orbit fluid supply to recovery of a single disabled Orbiter can result in a savings of \$472M. In-orbit supply of cryogenic H_2 and O_2 and some OMS/RCS fluids to the Shuttle Orbiter can extend uninterrupted spacelab missions indefinitely.

c. Tug performance can be significantly increased for most missions. For example, for in-orbit supply of the Tug-only, for a Mars Sample Return Mission, payload can be increased by 35% for a reusable Tug and by 53% for an expendable Tug. With in-orbit supply of both the Tug and Orbiter, payload increases of 108% for a reusable Tug and 83% for an expendable Tug are possible. Applying low cost design concepts to two Mars Sample Return Missions, assuming in-orbit supply of a reusable Tug, results in an estimated savings of \$120M over no resupply.

d. Use of surface tension screens for low-g liquid supply has the best overall potential for low weight and simplicity for both cryogenics and noncryogenics, except for Hg. However, potential thermal problems with cryogenics still need final resolution.

e. The paddle rotation system appears to be a good back-up to the surface tension system. Advantages are a potential minimization of problems associated with heat transfer, mass gauging, low-g venting and vehicle disturbances, as compared to surface tension screens. Very little work has been done on the paddle system and work to demonstrate feasibility is needed.

f. For large systems such as the Space Tug, use of linear acceleration for liquid orientation has the advantage of being nearer to current state-of-the-art, since the liquid is settled during transfer. A Tug supply system using Shuttle drag was found to be slightly lower in weight than one with the Tug and supply module separated from the Shuttle and accelerated by a separate propulsion system incorporated into the supply module. Thus, unless transfer in orbits higher than 296 km (160 n. mi.) were required, the drag system would be the likely choice for in-orbit supply of the Tug. For both cases, supply module weights are less than that of the baseline Space Tug supplied. Thus, more payload can be accommodated with the transfer module than with the Tug. For example, a payload of 4089 kg (9006 lb) could be accommodated with resupply using Shuttle Drag, while the maximum payload with the Tug would be 3732 kg (8221 lb).

g. For linear acceleration systems it was determined to be optimum to use long, small diameter (high L/D) tankage rather than large diameter (small L/D) tankage characteristic of current launch and upper stage vehicles. Savings in liquid residuals more than offset the increased weights of the small diameter tanks. Further work on low-g outflow could likely reduce residuals in small diameter tanks even further.

h. A significant problem, for which final solutions have not yet been demonstrated, is receiver tank chilldown and filling. Due to the low-g environment, it may be a problem to prevent direct liquid loss at receiver tank vents. For most of the cryogenic receivers a non-vent chilldown is impractical. Also, since the Shuttle and Large HEAO-B receivers are quite heavy, the quantity of fluid required for chilldown, even without direct liquid loss, is quite sensitive to the thermodynamic condition of the vent fluid (saturated versus superheated vapor). This is especially critical where helium is the transfer fluid. In fact, due to uncertainties in expected chilldown efficiencies, LN_2, representing an additional fluid system, is used for pre-chill of the HEAO-B. Use of helium alone could result in excessive overall system weight.

   Another potential receiver problem is to insure that surface tension devices, such as exist in the Shuttle Orbiter N_2O_4 and MMH tanks and the Satellite Control Section N_2H_4 tank, are full at the completion of transfer. Premature screen wicking and trapping of non-condensible vapor are potential problems for which solutions have not yet been developed.

i. Due to its very low heat of vaporization and surface tension, as compared to other cryogenics, the transfer of helium represents potentially unique problems needing further investigation; primarily in relation to its use with surface tension screens and receiver tank chilldown.

A listing is presented below of the most pertinent technology work recommended to develop in-orbit fluid transfer capability.

a. Receiver Chilldown and Fill (Cryogenic and Noncryogenic)

   1. Analytical Model Development
   2. One-g Thermodynamic Testing
   3. Drop Tower Testing
   4. One-g Prototype Demonstration of Practical System(s)
   5. Orbital Demonstration

b. Surface Tension Screen Systems (Cryogenic and Noncryogenic)

   1. Develop Low-G Refill Capability for Supply Channels and Receiver Channels and Baskets
   2. Demonstrate Compatibility with Realistic Vibration and Thermal Environments, Including Integration With Operational Type Tank Pressure Control Systems
   3. Orbital Demonstration of Complete Supply System Concept

c. Paddle Vortex Liquid Orientation

   1. Demonstrate Feasibility and Generate Basic Design Data in Subscale One-g Tests
   2. Overall System Analysis and Design
   3. One-g Prototype Testing and Orbital Demonstration

d. Low-g Pressure Control - Orbital Demonstration of Bulk Heat Exchanger Type Vent System

e. Low-g Outflow to Improve Prediction and Minimization of Liquid Residuals

   1. Analytical Model Development
   2. One-g and Drop Tower Testing
   3. Orbital Demonstration

f. Investigation of Special Problems of Helium Transfer

   1. Demonstrate Compatibility with Screen System
   2. Investigate Practicality of Other Than Liquid Transfer
   3. Develop Methods for Improved Thermal Chilldown Efficiency

g. Low-g Boiling, Condensation, Convection, and Two-Phase Flow Heat Transfer-Orbital Experimentation Required

h. Orbital Demonstration of Low-G Mass Gauging

i. Orbital Demonstration Test of Overall Transfer Concept
   1. Prototype Hardware One-g Tests
   2. Instrumentation/Observation Demonstration
   3. Development of Orbital Test Techniques

APPENDIX A

RECEIVERS CHARACTERISTICS

Available data pertinent to in-orbit fluid supply are presented in this appendix for the following receivers.

Propulsion Vehicles - Manned

- Space Shuttle Orbiter, Table A-1
- Space Shuttle External Tank, Table A-2
- Reusable Space Based Tug, Table A-3
- Modular Nuclear Vehicle (Mars Landing), Table A-4

Propulsion Vehicles - Automated

- Centaur (Reusable Configuration), Table A-5
- Transtage (Reusable Configuration), Table A-6
- Agena (Reusable Configuration), Table A-7
- Satellite Control Section, Table A-8
- Space Tug, Table A-9
- Reusable Nuclear Shuttle, Table A-10
- Chemical Interorbital Shuttle (SII Stage), Table A-11
- Chemical Interorbital Shuttle, Table A-12
- Common Solar-Electric-Propulsion Upper Stage, Table A-13
- Solar Electric Propulsion Stage, Table A-14

Spacecraft - Manned

- Modular Space Station, Table A-15
- Large Space Station, Table A-16

Spacecraft - Large Observatory

- Large High Energy Observatory B (Magnetic Spectrometer), Table A-17
- Large X-Ray Telescope, Table A-18
- Geosynchronous Platform, Table A-19

Spacecraft - Automated

- Gravity and Relativity Satellite, Table A-20
- Upper Atmosphere Explorer, Table A-21

Storage

Orbital Propellant Depot, Table A-22

Maneuvering Units

Space Taxi, Table A-23
Remote Maneuvering Unit, Table A-24

Experiments

Liquid Xenon Compton Telescope, Table A-25
Remote Sensing Platform, Table A-26
Cloud Physics Laboratory, Table A-27
Automated Furnace/Levitation Power, Table A-28
Biochemical/Biophysical Analysis Unit, Table A-29
Biological Experiments, Table A-30

Table A-1. Space Shuttle Orbiter

Vehicle Category: Propulsion Vehicle, Manned
Orbit for Fluid Transfer: 160 n. mi.
Data References: 3-35, -36, -37, -38

Fluid	N_2O_4	MMH	He	Supercritical O_2	Supercritical H_2	N_2O_4	MMH	He	N_2H_4	Lube Oil
Fluid Function	OMS oxidizer	OMS fuel	OMS pressurization	EPS, fuel cell oxidizer, ECLSS	EPS, fuel cell fuel	RCS oxidizer	RCS fuel	RCS pressurization	APU monopropellant	APU hydraulics
Fluid Weight (full) (incl. trapped and line fluids) w/o kits	15,536 lbs	9380 lbs	92 lbs	1562 lbs	184 lbs	4084 lbs	2563 lbs	24 lbs	579 lbs (budget)	
Fluid Temperature (operating)			100F max	-300 <T< 170F	-425 < T < 170F	40 < T < 100F	40< T< 100F	-65 <T< 150F	45 < T < 55F	
Fluid Feed System (zero G start)	outlet screen retention	outlet screen retention				screen retention	screen retention		surface tension	
Fluid Mass Measurement System	point sensors (main tank only) helium PVT	point sensors (main tank only) helium PVT	PVT	PVT, quantity gaging	PVT, quantity gaging	PVT gaging system	PVT gaging system	PVT	PVT	
Tank Thermal Control				vacuum & radiation insulation	vacuum & radiation insulation					heater, water boiler, heat exch.
Tank Pressurization Method	He	He		internal heater	internal heater	He	He	self	helium blowdown	
Operating Tank Pressure	255 psi	255 psi	4800 psia	950 psia	285 psia	245 ± 3 psig	245 ± 3 psig	3600 ± 50 psig	400 psig	
Number of Tanks (kits)	2 (+3)	2 (+3)	2 (+3)	2 (+6)	2 (+6)	1 forward 2 aft	1 forward 2 aft	2 forward 4 aft	3	
Tank Weight (each)			136 lbs	654 lbs						
Tank Length	94.3 in.	94.3 in.	sphere	sphere	sphere	sphere	sphere	sphere	sphere	
Tank Diameter	49.1 in.	49.1 in.	40.2 in.			38 in.	38 in.	19.1 in.		
Tank Fill Line Dia				0.5 in.	0.5 in.				1/2 in. (N_2H_4) 1/4 in. (He)	
Tank Vent Line Dia				0.5 in.	0.5 in.					
Other characteristics affecting transfer system design	primary tanks located in aft pod. Kits located in cargo bay. Tanks crossfed to aft RCS subsystem and to each other.			primary and one kit located under cargo bay, other kits located fore and aft in cargo bay.		tanks located in nose and aft pods. Aft propellants crossfed with OMS tank.			located in aft	Closed system

Table A-1. Space Shuttle Orbiter (Continued)

Fluid	N_2	O_2	H_2O Potable	H_2O Waste	H_2O	Freon 21	NH_3	Freon 1301	N_2	N_2
Fluid Function	ECLSS, water tank pressurization	ECLSS (auxiliary)	ECLSS & APU water boilers	Ammonia boilers	ECLSS water coolant loop	Active thermal control subsystem	Ammonia boilers for re-entry cooling	Fire extinguisher for avionics	OMS engine gimbal pneumatic	APU gearbox pressurization
Fluid Weight (full) (incl. trapped and line fluids)		112 lbs	330 lbs	495 lbs			86 lbs	15 lbs		
Fluid Temperature					45 < T < 173F	35F < T < 120F				
Fluid Feed System (zero G start)			Inconel bellows	bellows						
Tank Pressurization Method	self	self	N_2				N_2			
Operating Tank Pressure	3000 psi	3000 psi	12 psig	2 psig		320 psia				250 psig
Tank Venting Pressure			15 ± 2 psig							50 psig
Number of Tanks	4	1	2	3	(accumulator) 2		2	3 fixed 2 portable	2	3
Tank Weight (each)	264 lbs	66 lbs	34 lbs							
Tank Length	sphere	sphere	35.5 in.	35.5 in.			sphere			
Tank Diameter	25 in.	25 in.	15.5 in.	15.5 in.			17 in.			
Tank Vent Line Dia	3/8 in.	3/8 in.								
Other Characteristics affecting transfer system design	forward of midfuselage		resupplied from fuel cells. Vented when full	returned to ground		closed system				mounted on gearbox

Table A-2. Space Shuttle External Tank

Vehicle Category: Propulsion Vehicle, Manned

Fluid	LO_2	LH_2
Fluid Function	MPS	MPS
Fluid Weight (full (incl. trapped and line fluids)	1,331,000 lbs	222,000 lbs
Fluid Mass Measurement System	liquid level point sensors	liquid level point sensors
Tank Thermal control		spray-on foam insulation
Tank Pressurization method	autogeneous + He pre-launch	autogeneous + He pre-launch
Tank Venting Pressure	22 psia	34 psia
No. of Tanks	1	1
Tank Material	aluminum	aluminum
Tank Weight (each)	78,000 lbs	
Tank Diameter	330 in.	330 in.
Other Characteristics Affecting Transfer System Design	not normally retained in orbit	

Table A-3. Reusable Space Based Tug

Vehicle Category: Propulsion Vehicle, Automated/Manned

Data Reference: 3-8

Fluid	LO_2	LH_2	O_2	H_2
Fluid Function	propulsion oxidizer	propulsion fuel	RCS, fuel cells, CM atmosphere	RCS fuel cells
Fluid Weight (full) (including trapped and line fluids)	66,900 lbs	11,100 lbs	50 lbs	12 lbs
Fluid Temperature (storage)			380R	200R
Fluid Feed System (zero G start)	capillary for RCS	capillary for RCS	-	-
Tank Thermal Control	HPI insulation	HPI insulation	HPI insulation	HPI insulation
Tank Pressurization Method			heat exchanger	heat exchanger
Allowable Tank Pressure	20 psia	20 psia	1000 psia	1000 psia
Number of Tanks	4	1	2	2
Tank Material	aluminum	aluminum	aluminum	aluminum
Tank Length			sphere	sphere
Tank Diameter	62 in.	164 in.	22 in.	27 in.
Other Characteristics Affecting Transfer System Design			pump feed replenishment from main tanks through heat exchanger	

Table A-4. Modular Nuclear Vehicle Mars Direct Landing Mission Vehicle

Vehicle Category: Propulsion, Manned
Orbit for Fluid Transfer: 267 n. mi.
Data Reference: 3-11

Fluid	LH_2	LH_2	LH_2
Fluid Function	propulsion	propulsion	propulsion
Fluid Weight (full) (including trapped and line fluids)	693,000 lbs	168,000 lbs	320,000 lbs
Fluid Temperature (operating)	40R	40R	40R
Tank Thermal Control	foam insulation	foam insulation	foam insulation
Tank Pressurization Method	accumulator, autogenous	accumulator autogenous	accumulator, autogenous
Allowable Tank Pressure	30 psia	30 psia	30 psia
Number of Tanks	3	1	2
Tank Material	aluminum	aluminum	aluminum
Tank Weight (each)	10,900 lbs		
Tank Length	943 in.	719 in.	691 in.
Tank Diameter	384 in.	384 in.	384 in.
Tank Fill Line Dia	12 in.	12 in.	12 in.
Tank Zero G Vent	bulk heat exchanger	bulk heat exchanger	bulk heat exchanger
Tank Zero G Vent Line Diameter	14 in.	14 in.	14 in.

Table A-5. Centaur (Reusable Configuration)

Vehicle Category: Propulsion, Automated (ground based design)
Data Ref.: 3-28 and 3-29

Fluid	LH_2	LO_2	H_2O_2	He
Fluid Function	MPS fuel	MPS oxidizer	boost pumps, auxiliary propulsion	MPS tank pressurization and engine controls
Fluid Weight (full) (incl. trapped and line fluids)	7371 lbs	39307 lbs	484 lbs	15.5 lbs
Fluid Temp. (operating)	38R	175R	$40 < T < 120F$	
Fluid Feed System (zero G start)			silicone rubber bladder	
Fluid Mass Measurement System	PU capacitance probes	PU capacitance probes		
Tank Thermal Control	multilayer insulation blanket	multilayer insulation blanket	heater, blanket	
Tank Pressurization Method	He	He	He	self
Allowable Tank Operating Pressure	23 psia	33.5 psia	290 psig	3300 psia
Number of Tanks	1	1	2	2
Tank Material	st stl 301	st stl 301	st stl	titanium
Tank Weight (each)	753 lbs	395 lbs		81 lbs
Tank Length	182 in.	119 in.	sphere	sphere
Tank Diameter	174 in.	120 in.	21 in.	24 in.
Tank Fill Line Dia	5.25 in.	4.25 in.	0.5 in.	0.5 in.
Tank Zero G Vent	bulk heat exchanger	bulk heat exchanger		
Tank Zero G Vent Line Diameter	2.5 in.	2.5 in.	0.5 in.	

Table A-6. Transtage (Reusable Configuration)

Vehicle Category: Propulsion, Automated (ground based design)
Data Reference: 3-30

Fluid	Aerozine-50	N_2O_4	N_2H_4	He	N_2
Fluid Function	MPS fuel	MPS oxidizer	ACS	pneumatics for MPS	pneumatics for MPS
Fluid Weight (full) (including trapped and line fluids)	10,700 lbs	21,300 lbs	267 lbs	61 lbs	8 lbs
Fluid Temperature (operating)	70F	70F			
Fluid Feed System (zero G start)	screen	screen	diaphragm blowdown		
Fluid Mass Measurement System	PU	PU			
Tank Thermal Control	multilayer insulation	multilayer insulation	multilayer insulation, heaters		
Tank Pressurization Method	He	He	N_2		
Max. Tank Operating Pressure	187 psia	207 psia	370 psia	3250 psia	
Number of Tanks	1	1	1	3	part of ACS tank
Tank Length				sphere	
Tank Diameter				34 in.	

Table A-7. Agena Reusable - "S
Configuration

Vehicle Category: Propulsion Vehicle - Au
Orbit for Fluid Transfer: 170 n. mi.
Data Ref.: 3-31, 3-32

Fluid	MMH mono-methyl hydrazine	N_2O_4
Fluid Function	MPS fuel	MPS oxidizer
Fluid Weight (full) (including trapped and line fluids) (kits)	4970 lbs (13518 lbs)	10070 lbs (27441 lbs)
Fluid Temperature (operating)	$<$75F	$<$75F
Fluid Feed System (zero G start)	start tank	start tank
Fluid Mass Measurement System		
Tank Thermal Control	MLI	MLI
Tank Pressurization Method (Kits)	He blow-down (He regulated)	He blow-down (He regulated)
Max. Tank Operating Pressure	40 psia	35 psia
No. of Tanks (Kits)	1 (+3)	1 (+3)
Tank Length (Kits)	82.3 in. (158 in.)	86.4 in. (158 in.)
Tank Diameter (Kits)	60 in. (41.5 in.)	60 in. (41.5 in.)

Table A-8. Satellite Control Section

Vehicle Category: Propulsion, Automated
Data Reference: 3-33, 3-34

Fluid	N_2H_4	N_2H_4	N_2	N_2
Fluid Function	OAS (orbital adjust system)	RCS	OAS pressurization	RCS pressurization
Fluid Weight (full) (including trapped & line fluids)	2910 lbs	412 lbs	37 lbs	8 lbs
Fluid Temperature (Operating)	70°<T< 100°F	40°<T< 140°F		
Fluid Feed System (Zero G Start)	surface tension device	white EPR rubber diaphragm		
Tank Thermal Control	heaters, insulation	heaters MLI		
Tank Pressurization Method	N_2 blowdown	N_2 blowdown		
Operating Tank Pressure	310 psia	264 psi		
Number of Tanks	1	4	part of OAS tank	part of RCS tank
Tank Length	sphere	sphere		
Tank Diameter	62 in	22 in		
Other Characteristics Affecting Transfer System Design		RCS may be fed from main OAS tank		

Table A-9. Space Tug

Vehicle Category: Propulsion Vehicle, Automated (Ground Based Design)
Orbit for Fluid Transfer: 160 n. mi.
Data Reference: 6-5

Fluid	LH_2	LO_2	N_2H_4	He
Fluid Function	Fuel, main propulsion & fuel cell	Oxidizer, main propulsion & fuel cell	ACS	ACS, propellant tank pressurization
Fluid Weight (full) (including trapped & line fluids)	7626 lbs	43574 lbs	336 lbs	9 lbs
Fluid Temperature (Operating)	-423°F	-297°F	45°<T< 105°F	ambient
Fluid Feed System (Zero G Start)			positive expulsion bladder	
Fluid Mass Measurement System	PLIS/PU capacitance probes	PLIS/PU capacitance probes	GSE metering	Pressure and temperature sensors
Tank Thermal Control	multilayer insulation	multilayer insulation	Heaters, multilayer insulation	None
Tank Pressurization Method	helium/GH_2	helium/GO_2	helium	None
Operating Tank Pressure	22 psia (max.)	20.5 psia(max)	331 ± 6 psia	3300 psia
Number of Tanks	1	1	3	3
Tank Material	2219-T87 aluminum alloy	2219-T87 aluminum alloy	titanium	CRES
Tank Weight (each	502 lbs	294 lbs		
Tank Capacity (each)	1748 ft^3	640 ft^3		
Tank Length	174.5 in	101.8 in		
Tank Fill Line Dia.	6 in.	5 in.	1.0 in.	.5 in.
Tank Zero G Vent	bulk heat exchanger	bulk heat exchanger		
Tank Zero G Vent Line Diameter	1.0 in.	.75 in.		

Table A-10. Reusable Nuclear Shuttle

Vehicle Category: Propulsion Vehicle, Automated
Data Ref.: 3-12

Fluid	LH_2	Super-critical O_2	Super-critical H_2
Fluid Function	MPS fuel	EPS & RCS	EPS & RCS
Fluid Weight (full) (including trapped and line fluids)	300,000 lbs	5560 lbs	1320 lbs
Fluid Temperature (operating)	-423F	160R	40R
Fluid Feed System (zero G start)	capillary barrier		
Fluid Mass Measurement System	point sensors capacitance probes		
Tank Pressurization Method	autogenous	heat exchanger	heat exchanger
Max. Tank Operating Pressure	27.5 psig	800 psia	500 psia
No. of Tanks	1	2	6
Tank Material	Al 2014-T6		
Tank Weight (each)	17470 lbs	155 lbs	155 lbs
Tank Length	152 ft	sphere	sphere
Tank Diameter	33 ft (max)		
Tank Fill LineDia (orbital)	6 in.		
Tank Zero G Vent	2 systems		
Other Characteristics Affecting Transfer System Design		Modular replacement of bottles planned	

Table A-11. Chemical Interorbital Shuttle (SII Stage) Configuration A

Vehicle Category: Propulsion Vehicle, Automated
Data Ref: 3-39

Fluid	LO_2	LH_2	LO_2	LH_2	He	Supercritical O_2	Supercritical H_2
Fluid Function	MPS oxidizer	MPS fuel	OMS oxidizer	OMS fuel	Propellant recovery	RCS oxidizer and pressuri-z·tion to main and OMS tanks, fuel cell	RCS fuel and pressurization to main & OMS tanks, fuel cell
Fluid Weight (full) (incl. trapped & line fluids)	880,000 lbs	160,000 lbs					
Fluid Feed Sys. (zero G start)	one retention screen canopy	two retention screen canopies					
Tank Thermal Control	high perf. insulation	high perf. insulation				heat exchanger	heat exchanger
Max. Operating Tank Pressure	28 psia	26 psia			3000 psia	800 psia	500 psia
No. of Tanks	1	1	1	4			
Tank Length	18.3 ft	52.3 ft	sphere	sphere			
Tank Diameter	33 ft	33 ft	7.4 ft	7.4 ft			

Table A-12. Chemical Interorbital Shuttle 1-1/2 Stage Configuration

Vehicle Category: Propulsion Vehicle, Automated
Data Ref.: 3-14

Fluid	LO_2	LH_2	LO_2	LH_2
Fluid Function	MPS oxidizer	MPS fuel	MPS oxidizer	MPS fuel
Fluid Weight (full) (including trapped and line fluids)	261,500 lbs	43,500 lbs	574,300 lbs	95,700 lbs
Fluid Feed System (zero G start)	acceleration	acceleration	acceleration	acceleration
Fluid Mass Measurement System	PU system	PU system	none	none
Tank Thermal Control	MLI	MLI	MLI	MLI
Tank Pressurization Method	He ground, GO_2 space	He ground, GH_2 space	He ground, GO_2 space	He ground, GH_2 space
Operating Tank Press.	27 psig	25 psig		
Number of Tanks	1	1	1	1
Tank Material	2014-T651 aluminum	2014-T651 aluminum		
Tank Length			46 ft	77 ft
Tank Diameter	20 ft	20 ft	23 ft	23 ft
Tank Zero G Vent		Thermodyn. Vent		
Tank Zero G Vent Line Diameter	1 in.	1 in.	1 in.	1 in.
Other Characteristics Affecting Transfer System Design	APS tanks are fed from main tank		Drop tank, modified shuttle external tank	

Table A-13. Common Solar-Electric-Propulsion Upper Stage Option B

Vehicle Category: Propulsion, Automated
Data Reference: 3-15

Fluid	Hg	N_2H_4	N_2
Fluid Function	MPS	terminal maneuvers	RCS
Fluid Weight (full) (including trapped) & line fluids)	2230 lbs	176 lbs	44 lbs
Fluid Feed System (Zero G Start)	bladder		
Tank Thermal Control	MLI insulation over propulsion module with louvers, heaters		
Tank Pressurization Method	regulated	blowdown	
Tank Pressure			4000 psia
Number of Tanks	1	1	2
Tank Material			titanium
Tank Weight (each)	55 lbs	13 lbs	7 lbs
Tank Length	sphere	approx. 40 in.	sphere
Tank Diameter	21 in.	approx. 22 in.	approx. 12 in.

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Table A-14. Solar Electric Propulsion Stage

Vehicle Category: Propulsion Automated
Data Reference: Solar Electric Propulsion Stage Characteristics of Rockwell Intl. design per B.C. Loman as of August 28, 1974 (Reference 3-16)

Fluid	Hg	N_2H_4	Freon 113	Kr Krypton
Fluid Function	primary propulsion RCS	RCS	Mercury pressurant	RCS
Fluid Weight (full) (including trapped) & line fluids)	3300 lbs	80 lbs	10 lbs	30 lbs
Fluid Temperature (Operating)	64-148 F	40-120 F	64-148 F	20-120 F
Fluid Feed System (Zero G Start)	diaphragm			
Fluid Mass Measurement System	flow rate	PVT		PVT
Tank Thermal Control	conduction	conduction/ radiation	conduction	conduction/ radiation
Tank Pressurization Method	Freon expulsion	blowdown		blowdown
Operating Tank Pressure	25 psia	500 psia	25 psia	3500 psia
Number of Tanks	4	2	Hg Tank	1
Tank Material	stainless steel	titanium		titanium
Tank Weight (each)	40 lbs	13 lbs		15 lbs
Tank Length	15 in.	sphere		sphere
Tank Diameter	15 in.	12 in.		12 in.

Table A-15. Modular Space Station

Vehicle Category: Spacecraft, Manned
Orbit for Fluid Transfer: 240 n. mi.
Data Ref: 3-17

Concept 8 - Closed O_2 OCLSS: 6 man crew, 90 day resupply - Initial Station (Cargo Module is normally serviced on the ground)

Fluid	GO2	GO_2	GO_2	GO_2	H_2O	GH_2	GH_2	GN_2	GN_2	Freon
Fluid Function	Emergency supply ECLSS & EPS	Repressurization ECLSS	EVA resupply	EPS, RCS buildup	Electrolysis resupply water. ECLSS, EPS RCS, PLSS recharge	Emergency Supply EPS	EPS, RCS buildup	Repressurization ECLSS	Leakage	Cooling Reservoir
Fluid Weight (total)	382 lbs	194 lbs	128 lbs Resupply = 9.6 lbs	606 lbs	993 lbs	20 lbs	76 lbs	380 lbs	985 lbs Resupply = 740 lbs	604 lbs
Fluid Temperature (operating)		70F			70F			70F	70F	
Initial Tank Pressure	3000 psia	3000 psia	3000 psia	3000 psia	300 psia	3000 psia	3000 psia	3000 psia	3000 psia	
Number of Tanks	1	1		3	3	1	3	3		2
Tank Weight (ea)		190 lbs		187 lbs			90 lbs	150 lbs		30 lbs
Tank Length		sphere		sphere			sphere	sphere		sphere
Tank Diameter		33 in.		33 in.			26 in.	31 in.		23 in.
Other Characteristics Affecting Transfer System Design	not normally resupplied	tanks are replaced & refilled on the ground	resupplied every 90 days	not normally resupplied	resupplied every 90 days	not normally resupplied	not normally resupplied	Tanks are replaced & refilled on the ground	resupplied every 90 days	not normally resupplied

Table A-16. Large Space Station

Vehicle Category: Spacecraft, Manned
Orbit for Fluid Transfer: 246 n. mi.
Data Reference: 3-18, 3-19

(Resupplied Fluids Only)
12 man station - 90 day resupply

Fluid	N_2H_4	H_2O	GN_2	GO_2	GN_2
Fluid Function	high thrust RCS	low thrust RCS	pressurize high thrust RCS	ECLSS	ECLSS
Fluid Weight (full) (including trapped & line fluids)	7000 lbs	200 lbs/90 days 450 lbs	225 lbs	102 lbs re-supply 405 lbs	239 lbs re-supply 1336 lbs
Fluid Temp. (operating)	50-120 F		50 - 120 F		
Fluid Feed System (Zero G Start)	stainless steel bellows	stainless steel bellows			
Tank Pressurization Method	N_2	CO_2			
Tank Pressure	250 psia		3000 psia	3000 psia	3000 psia
Number of Tanks	14	2	4	4	8
Tank Material	titanium		6 AL-4V titanium		
Tank Weight (each)	125 lbs	18 lbs	105 lbs	112 lbs	160 lbs
Tank Length	60 in.	sphere	sphere	sphere	sphere
Tank Diameter	22 in.	24 in.	26 in.	< 2.3 ft	< 2.8 ft
Other Characteristics Affecting Transfer System Design		stored as liquid, used as gas		Assumes bottle transferred manually	Assumes bottle transferred manually

Table A-18. Large X-Ray Telescope

Vehicle Category: Spacecraft - Large Observatory
Data Reference: 3-4

Fluid	N_2H_4	N_2	Cryogenic
Fluid Function	propellant	cold gas RCS	cool detector
Fluid Weight (full) (including trapped & line fluids)	1000 lbs (454 kg)	220 lbs (100 kg)	
Fluid Temperature (operating)			20 < T < 78K
Number of Tanks	4		

Table A-17. Large High Energy Observatory B (Magnetic Spectrometer)

Vehicle Category: Spacecraft - Large Observatory
Orbit for Fluid Transfer: 200 n. mi.
Data Reference: 3-41, 3-44

Fluid	LHe	Xe	N_2
Fluid Function	Magnet Cryostat	Detection Gas	RCS
Fluid Weight (full) (including trapped & line fluids)	950 lbs	370 lbs	221 lbs
Fluid Temperature (Operating)	3 ± 1.2K		
Fluid Mass Measurement System	Mass sensing or integrating flow meters		
Tank Thermal Control	multilayer		
Tank Pressurization Method	self	self	self
Max. Operating Pressure	16 psia	700 psia	
Number of Tanks	1	4	
Tank Material	aluminum 2219-T6		
Tank Weight (each)	981 lbs		
Tank Length	86.2 in.	sphere	
Tank Diameter	62.2 in.	24.6 in.	
Tank Fill Line Dia.	5/16 in.		
Tank Zero G Vent	wall heat exchanger		
Tank Zero G Vent Line Diameter	3/8 in. and 1/4 in.		
Other Characteristics Affecting Transfer System Design	one year resupply would be required		

Table A-19. Geosynchronous Platform

Vehicle Category: Spacecraft, Large Observatory
Orbit for Fluid Transfer: 19,300 n.mi.
Data Reference: 3-25

Fluid	N_2H_4	GO_2
Fluid Function	hydrazine mono-propellant RCS	atmosphere for shirt sleeve maintainence
Fluid Weight (full) (including trapped & line fluids)	176 lbs	
Allowable Tank Pressure		3000 psia
Number of Tanks (Modules)	4	
Tank Length	sphere	
Tank Diameter	20 in	
Other Characteristics Affecting Transfer System Design	2.5 yr lifetime. Planned to replace total module	carried on-board tug for service mission

Table A-20. Gravity and Relativity Satellite

Vehicle Category: Spacecraft-Automated
Orbit for Fluid Transfer: 500 n.mi.
Data Reference: 3-4

Fluid	LHe
Fluid Function	precession gyroscope cooling
Fluid Weight (full) (including trapped & line fluids)	297 lbs
Fluid Temperature (Operating)	1.6 K
Dewar Thermal Control	multilayer insulation vapor cooled shields
Number of Dewars	1
Dewar Weight total	148 lbs
Dewar Length	65 in
Dewar Diameter	50 in
Tank Zero G Vent	super fluid plug
Other Characteristics Affecting Transfer System	no replenishment planned. One year supply provided

Table A-21. Upper Atmosphere Explorer

Vehicle Category: Spacecraft/Automated
Orbit for Fluid Transfer: 259 × 3510 km
Data Ref: 3-4

Fluid	N_2H_4
Fluid Function	propulsion, orbit adjustment, RCS
Fluid Weight (full) (including trapped and line fluids)	385 lbs (175 kg)
Number of Tanks	1

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Table A-22. Orbital Propellant Depot RNS Supportive Depot

Vehicle Category: Storage, Man Tended
Orbit for Fluid Transfer: 262 n. mi.
Data Ref: 3-40

Fluid	LO_2	LH_2	He	LN_2
Fluid Function	storage tank, RCS, EPS, ECLSS	storage tank, RCS, EPS		leakage nitrogen makeup
Fluid Weight (full) (including trapped and line fluids)	150,000 lbs	430,000 lbs		
Fluid Temperature (operating)	163R	37R		
Fluid Feed System artificial gravity - rotational	pump	pump		
Fluid Mass Measurement System	capacitance probes + discrete point sensors	capacitance probes + discrete point sensors		
Tank Thermal Control	high perf. insulation	high perf. insulation		
Tank Pressurization Method	gas generator	gas generator		
Tank Pressure	20 psia	20 psia		
Number of Tanks	1	2	4	4
Tank Material	aluminum	aluminum		
Tank Length	23 ft	69 ft	sphere	sphere
Tank Diameter	12 ft	33 ft		
Tank Fill Line Dia	1-1/2 in.	4 in.		
Tank Zero G Vent	bulk heat exchanger	bulk heat exchanger		

Table A-23. Space Taxi

Vehicle Category: Maneuvering Units
Data Ref.: 3-22

Fluid	N_2H_4	LN_2	LO_2	O_2	N_2	LH_2	LO_2	H_2O	Freon
Fluid Function	MPS	ECLSS	ECLSS	emergency ECLSS	emergency ECLSS	EPS fuel cell	EPS fuel cell	cooling system	cooling system
Fluid Weight (full) (including trapped and line fluids)	234 lbs	4 lbs	11 lbs			1 lb	6 lbs	12 lbs	9 lbs
Fluid Temperature (operating)								50-80F	37F
Fluid Feed System (zero G start)	bladder								
Fluid Mass Measurement System	acoustic gage								
Tank Pressurization Method	pumped N_2H_4 gas generator								
Tank Pressure	250 psia		150 psia			50 psia	150 psia		
Number of Tanks	1	1	1	1	1	1	1	1	1
Tank Weight (each)	20 lbs	1 lb	2 lbs	3 lbs	1 lb	1 lb	1 lb	2 lbs	2 lbs
Tank Length	sphere	sphere	sphere	sphere	sphere			8 in.	8 in.
Tank Diameter	24 in.	7 in.	9 in.	4 in.	5 in.			5 in.	5 in.
Tank Fill Line Dia		1 in.	1 in.						
Tank Vent Line Dia		2 in.	2 in.						
Other Characteristics Affecting Transfer System Design								resupplied from fuel cell	not planned for resupply

Table A-24. Remote Maneuvering Unit

Vehicle Category: Maneuvering Units, Automated
Data Reference: 3-23, 3-24

Fluid	N_2H_4	N_2
Fluid Function	MPS	propulsion tank pressurization
Fluid Weight (full) (including trapped & line fluids)	77 lbs	2.1 lbs
Fluid Temperature (Operating)	70°F	
Fluid Feed System (Zero G Start)	347 stainless steel bellows	
Fluid Mass Measurement System	propellant quantity gage	PVT
Tank Pressurization Method	N_2 blowdown	
Initial Tank Pressure	800 psig	
Number of Tanks	2	same tank as N_2H_4
Tank Material	6 Al-4V titanium	
Tank Weight (each)	18 lbs	
Tank Length	29 in.	
Tank Diameter	10 in.	

Table A-25. Liquid Xenon Compton Telescope

Vehicle Category: Experiment
Data Reference: 3-27

Fluid	LXe Xenon	Ar Argon
Fluid Function	determine celestial source of x-rays	cooling LXe
Fluid Temperature (Operating)	-110 ± 1°C	-110 ± 1°C
Tank Thermal Control	use Ar gas	heater

Table A-27. Cloud Physics Laboratory

Vehicle Category: Experiment
Data Reference: 3-6

Fluid	H_2O
Fluid Function	cloud and ice formation
Fluid Weight (full) (including trapped & line fluids)	232 lbs (105.5 kg)

Table A-26. Remote Sensing Platform

Vehicle Category: Experiment
Data Reference: 3-6

Sensor	Radiometer, IR	Spectrometer, IR	Spectrometer, High Resolution Fourier SWIR
Fluid	cryogenic	cryogenic	cryogenic
Fluid Function	cool detector	cool detector	cool detector
Fluid Weight (full) (including trapped & line fluids)	4 lbs (2 kg)	44 lbs (20 kg)	18 lbs (8 kg)

Table A-28. Automated Furnace/ Levitation Power

Vehicle Category: Experiment-Space Processing
Data Reference: 3-6

Fluid	LO_2	LH_2	H_2O
Fluid Function	power (fuel cell)	power (fuel cell)	Liquid waste
Fuel Weight (full) (including trapped & line fluids)	825 lbs	111 lbs	
Number of Tanks	2	2	9
Tank Weight (each)	97 lbs	138 lbs	11 lbs
Tank Length	sphere	sphere	27 in.
Tank Diameter	20 in.	23 in.	12 in.
Other Characteristics Affecting Transfer System Design		Product of LO_2 & LH_2 usage. Returned to earth.	

Table A-29. Biochemical/ Biophysical Analysis Unit

Vehicle Category: Experiment-Life Sciences
Data Reference: 3-6, 3-26

Fluid	LN_2
Fluid Function	freeze trap for biological specimens
Fluid Weight (full) (including trapped & line fluids)	90 lbs
Fluid Temperature (Operating)	-320°F
Tank Thermal Control	super insulation double wall construction
Tank Material	aluminum
Tank Weight (each)	14 lbs
Tank Length	21 in
Tank Diameter	10-1/4 in.

Table A-30. Biological Experiments

Vehicle Category: Experiments
Data Reference: 3-6

Fluid	Electrolyte	Liquid Waste
Fluid Function	isolation of specific species of proteins, enzymes, or cells	resulting from experiments
Fluid Weight (full) Iincluding trapped and line fluids)	33 lgs (15 kg)	
Number of tanks	4	1
Tank Weight (each)	1.36 kg	4.5 kg
Tank Length	sphere	
Tank Diameter	0.15 m	

APPENDIX B

GROUND RULE DATA FOR TRANSFER SYSTEMS STUDIES

Ground rules covering use of the Shuttle as the supply vehicle are found in Reference B-1. Some of the more pertinent are presented below.

1. 50KWH of nominal 28 VDC electrical power is available to the payload at a maximum average rate of 7 KW with an allowable peak of 12 KW for 900 sec. Any additional requirement is chargeable to the payload. Assuming expansion of the existing Shuttle fuel cell system, additional power needs would be charged as follows.

   lb weight = 29 lb/kw + 1. 14 lb/kw-hr

   kg Weight = 13. 2 kg/kw + 0. 52 kg/kw-hr

2. The allowable cargo weight into orbit for 185 to 389 km (100 to 210 n. mi.) circular orbits (28. 5° inclination) is 29, 510 kg (65, 000 lb). The baseline orbit for fluid transfer is assumed to be a 296 km (160 n. mi.) circular orbit with 28. 5° inclination.

3. The on-orbit Orbiter weight without payload is 71, 300 kg (157, 000 lb).

4. Atmospheric drag on the Shuttle is presented in Figure B-1.

5. The Orbiter normal mission duration is seven days. However, single pass missions can also be provided. The Orbiter has the capability to rendezvous with orbiting payloads that are either cooperative or passive.

6. Simultaneous LH_2 and LO_2 propellant dump is feasible at altitudes above 33. 6 km (110, 000 ft) if the propellant dump exits on the Orbiter are sufficiently separated. At present the Orbiter has provision for dumping LO_2 by providing a forward longitudinal acceleration force from the main propulsion or orbit maneuvering system to settle the liquid. A 12. 7 cm (5 in.) dump line is provided on the Orbiter for this purpose. The actual pressure for expulsion must be provided by the payload.

7. The Orbiter payload bay is vented during the launch and entry phases, and operates unpressurized during the orbital phase of the mission.

8. The Orbiter provides a manipulator 15. 3 m (50 ft) in length on the left side of the vehicle. In orbit the manipulator is capable of removing and installing a 4. 6 m (15 ft) diameter, 18. 3 m (60 ft) long, 29, 510 kg (65, 000 lb) payload. A second

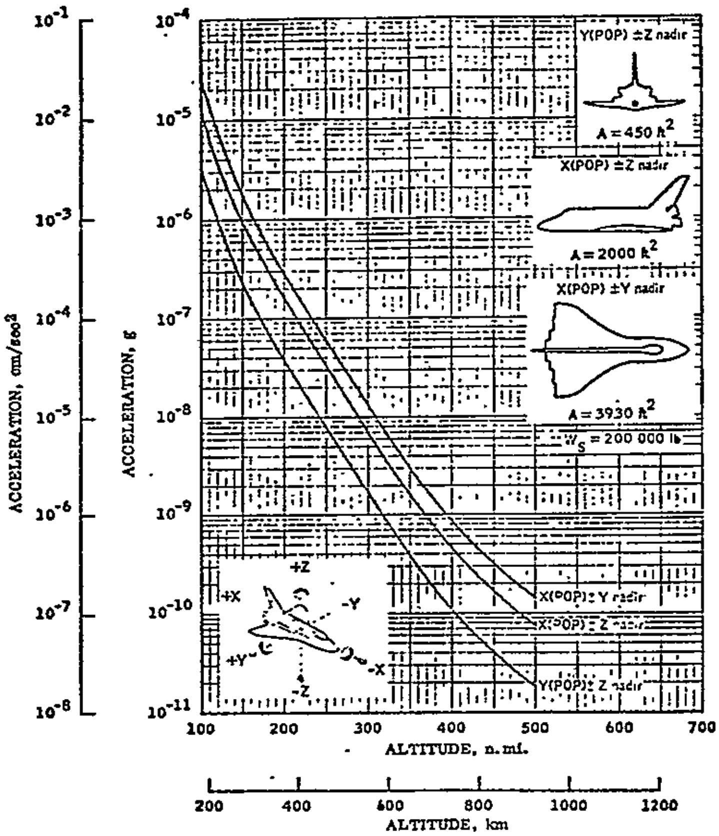


Figure B-1. Effects of Atmospheric Drag on the Orbiter (Reference B-1)

manipulator arm can be installed if required. The weight of the second manipulator is chargeable to the payload.

Basic fluid properties used in the study are presented in Table B-1.

A normal dedicated supply mission is illustrated in Figure B-2. For a worst case, where launch would be required at any specific time, such as for a space rescue, a significant phasing operation could be required and the time from lift-off to fluid transfer could be as great as 43.2 ks (12 hrs).

Additional ground rules of significance to the overall program, from Reference B-2, are presented below.

1. Fluid loading into the Shuttle cargo bay shall be with the Shuttle in the vertical position.

Table B-1. Fluid Properties Data Used in Transfer Analyses

	Supply Loaded Liquid Density		Surface Tension		Vapor Press. at 294K (70F)		Freezing Temp.		Heat of Vaporization	
Fluid	kg/m^3	lb/ft^3	Dynes/cm	lb/ft	kN/m^2	psia	K	F	J/G	Btu/lb
N_2H_4	1004	62.7	91.5	6.27×10^{-3}	1.58	0.23	275	35.0	-	-
N_2O_4	1453	90.7	27.5	1.88×10^{-3}	101	14.70	262	11.0	-	-
MMH	870	54.3	34.3	2.35×10^{-3}	5.5	0.8	221	-63	-	-
Hg	13542	845.3	465	3.19×10^{-2}	-	-	-	-	-	-
LHe	123	7.7	0.088	6.03×10^{-6}	-	-	-	-	21	9.02
LH_2	70.5	4.4	2.0	1.37×10^{-4}	-	-	-	-	442	190
LO_2	1137	71	13.0	8.9×10^{-4}	-	-	-	-	213	91.5
LN_2	801	50	8.3	5.7×10^{-4}	-	-	-	-	200	86

2. Venting and emergency detanking may be required with Orbiter in horizontal or vertical position until launch commit with payload bay doors open or closed.

3. Selective orientation in orbit for thermal control shall not be required.

4. Only GN_2 may be dumped into the payload bay and then only under controlled temperature and flow.

5. Transfer lines shall be purged after the transfer of hazardous fluids.

6. Mercury when used shall be double contained.

7. Fluid sloshing must be controlled to Orbit guidance and navigation control capability.

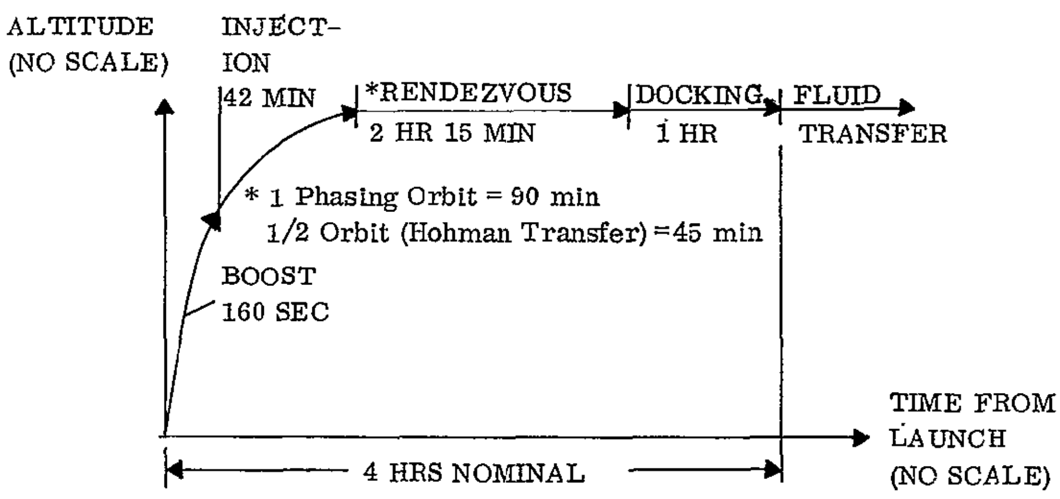


Figure B-2. Nominal Dedicated Fluid Supply Mission

APPENDIX C

CALCULATION OF PULL-THROUGH RESIDUALS AT LOW-G

Liquid residuals were calculated using the basic method described in Reference 4-4. The procedure is to calculate residuals at vapor pull-through assuming a flat liquid interface and then to add to these the residuals due to low-g curvature of the interface at pull-through (Figure 4-5).

Two different methods were employed for determining the flat interface residuals. The first method utilizes data from North American Rockwell (NAR), Reference 4-4, for outflow from a baffled hemispherically bottomed tank. The applicable equation for flat interface residual volume is

$$V_R = 4.89 \times 10^{-3} D_t^3 (F_{N_\ell})^{0.352}$$

The Froude no., F_{N_ℓ}, is based on conditions in the outflow line. A boundary condition which must be applied to the above equation is that V_R is never greater than 0.33 times D_t^3. Also, the diameter of the outflow line should never be greater than 10% of that of the tank.

The second method of calculating flat interface residuals utilizes NASA/LeRC data from Reference C-1. The applicable equation is presented below.

$$\begin{matrix}\text{Pull-Through} \\ \text{Height, h, ft}\end{matrix} = 8.74 \times 10^{-2} (D_\ell, \text{ in.})^{0.714} (D_t, \text{ ft})^{0.286} (F_{N_\ell})^{0.143}$$

$$h, \text{ m} = 1.92 \times 10^{-2} (D_\ell, \text{ cm})^{0.714} (D_t, \text{ m})^{0.286} (F_{N_\ell})^{0.143}$$

The above equation is also for hemispherically bottomed tanks and in general gives lower residuals than the NAR data. Flat interface residuals are calculated from h and the tank geometry.

In both the NAR and LeRC cases, the curved interface residuals are calculated using equations from Reference 4-5, as presented below.

$$\text{Curved Interface Residual Volume} = \frac{\pi}{3} \frac{1}{\xi} r^3 \qquad \text{(C-1)}$$

where

$$\xi^3 + \xi = 2 + Bo_r$$

and r is the local tank radius where pull-through occurs (Figure 4-5). r is determined from tank geometry and the flat interface residual volume (NAR) or liquid height at the tank centerline (LeRC).

It is noted that residuals due to low-g interface curvature, as calculated from the above equation, are based on a cylinder of radius r. A graphical comparison of the actual and assumed cases is presented in Figure C-1.

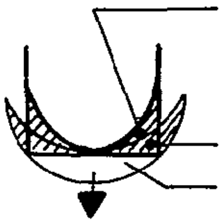


Figure C-1. Cylindrical Approximation of Curved Interface Residuals

By using the cylindrical approximation, reasonably simple equations can be used to estimate residuals. Computer programs do exist for calculating low-g interface shape in a hemisphere, however, time did not permit their acquisition and application to the current study. References C-2 and C-3 present some graphical data of interface shape which was used to compare with the cylindrical approximation. In this case, actual volumes were determined from the Reference C-2 and C-3 data by graphical integration. Comparative data are presented in Table C-1 for several conditions.

Table C-1. Actual Versus Approximated Curved Interface Residuals (D_t = 1.5 m, 5 ft)

Bo_r	Pull-Through Height, h		$\frac{\text{(Cylindrical Volume)}}{\text{(Actual Volume)}}$
	m	ft	
5 (Reference C-2)	0.094	0.3075	1.11
5 (Reference C-3)	0.23	0.75	0.99
1 (Reference C-3)	0.17	0.55	0.72

Table C-1 shows very good agreement at Bond numbers on the order of 5, with less agreement at lower Bo_r. In general, the accuracies were considered adequate for the current pre-design investigations.

The overall calculation procedures were programmed for the Hewlett-Packard Model 9100 computer and calculations made. Capability for throttling the outflow, both up and down, was included. This throttling is accomplished in any desired number of discrete steps. In the present cases it was found that for 10:1 throttling the final residuals were insensitive to any number of steps above five.

APPENDIX D

GENERAL THERMAL ANALYSIS

Initially, calculations were made to determine the optimum insulation concepts to be used for the various LH_2, LO_2 and LHe supply tankage. Comparisons were made between the use of closed-cell foam, purged multilayer insulation (MLI), purged MLI with foam substrate and vacuum jacketed MLI. The MLI was assumed to be aluminized Kapton with an effective conductivity in space of 9.03×10^{-5} W/m-°K (5.22×10^{-5} BTU/hr-ft-°F) and a density of 21 kg/m^3 (1.31 lb/ft^3). This data is for an installed system, including penetrations, and is based on information from Reference D-1.. For the closed-cell foam, conductivity and density were taken to be respectively 0.026 W/m-°K (0.015 BTU/hr-ft-°F) and 32 kg/m^3 (2 lb/ft^3).

Overall heating was calculated through the following steps.

1. Ground heating for 180 sec.

2. Linear transition from ground heating to space heating. Six hundred sec used for purged MLI cases and 120 sec for foam and vacuum jacketed MLI cases.

3. Space heating for 43.2 ks (12 hours) prior to fluid transfer.

The maximum environmental temperature during ground hold and boost was taken to be 322K (580R). Total integrated heating over the ground and boost phases was determined by adjustment of the data from Reference 4-4 to meet the current conditions. For in-orbit heating it was assumed that the supply tankage sees 238K (428R) space with a 40% view factor and 294K (530R) cargo bay walls with a 60% view factor.

For the LH_2 and LO_2 cases the purged MLI system was by far the lowest weight when compared to foam and vacuum jacketed systems. The foam substrate system was only slightly heavier, but was not chosen, due to its added complexity over the MLI-only system.

Final equations for calculating heat inputs to the cryogenic tankage when using MLI purged with GHe are presented below.

Ground Hold

$$\dot{Q}_G\text{, BTU/hr-ft}^2 = \frac{(530-T_x, °R)}{493}\left[118 + 82\left(\frac{1.5}{t_x,\text{ in.}}\right)\right] \tag{D-1}$$

$$\dot{Q}_G,\ W/m^2 = \frac{(294 - T_x, °K)}{274}\left[372 + 258 \quad \frac{3.81}{t_x,\ cm}\right]$$

where

T_x = tank liquid temperature

t_x = insulation thickness

Both convection and conduction of the helium purge gas is included per Reference D-2.

Space Heating

$$\dot{Q}_S,\ BTU/hr\text{-}ft^2 = \frac{1.134 \times 10^{-3}\ (428 - T_x,\ °R)}{t_x,\ in.} \qquad \text{(D-2)}$$

$$\dot{Q}_S,\ W/m^2 = 1.63 \times 10^{-2}\ \frac{(238 - T_x,\ °K)}{t_x,\ cm}$$

The total heat input prior to transfer is then determined from the following.

$$Q,\ BTU = \left[\left(\dot{Q}_G,\ BTU/hr\text{-}ft^2\right)\left(\frac{3}{60}\right) + \left(\frac{\dot{Q}_G,\ BTU/hr\text{-}ft^2 + \dot{Q}_S,\ BTU/hr\text{-}ft^2}{2}\right)\right.$$

$$\left.\times \left(\frac{10}{60}\right) + \left(\dot{Q}_S,\ BTU/hr\text{-}ft^2\right)\left(12\right)\right]\left[A_S,\ ft^2\right] \qquad \text{(D-3)}$$

$$Q,\ KJ = \left[\left(\dot{Q}_G,\ W/m^2\right)\left(0.18\right) + \left(\frac{\dot{Q}_G,\ W/m^2 + Q_S,\ W/m^2}{2}\right)\left(0.6\right)\right.$$

$$\left. + \left(\dot{Q}_S,\ W/m^2\right)\left(43.2\right)\right]\left[A_S,\ m^2\right]$$

where A_S is the total tank surface area.

Using the above equations,optimum insulation thicknesses were determined for typical Space Tug and Shuttle Orbiter LH_2 and LO_2 supply tankage. Trade-offs were made between insulation weight and fluid vented as a function of insulation thickness, as illustrated in Figure D-1. In each case the tanks were assumed to be spherical in shape and 95% filled with liquid on the ground. Based on previous experience (Ref. 4-4), 345 kN/m^2 (50 psia) was chosen as a reasonable maximum tank design pressure for minimum overall weight.

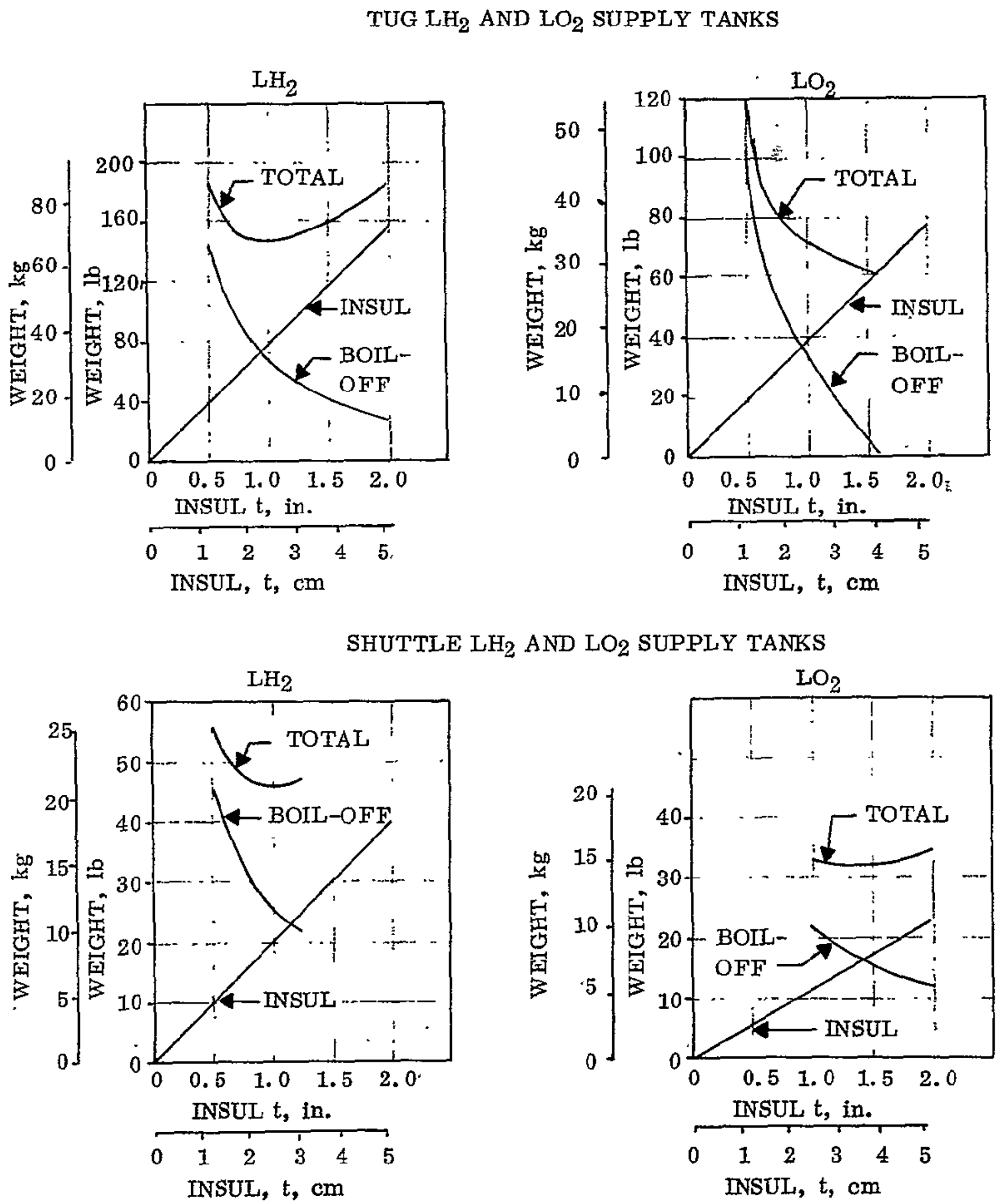


Figure D-1. LO_2 and LH_2 Insulation Optimization

Pressure rise rates were determined from the following equations taken from Ref. D-3.

LH_2 (Zero-g)

$$\frac{\Delta P}{\Delta t}, \frac{psi}{hr} = 86 \left[\frac{\dot{Q}, Btu/hr}{(M, lb)\ S} \right]^{0.975}$$

$$\frac{\Delta P}{\Delta t}, \frac{N}{m^2\ s} = 252 \left[\frac{\dot{Q}, W}{(M, kg)\ S} \right]^{0.975}$$

LO_2 (Zero-g)

$$\frac{\Delta P}{\Delta t}, \frac{psi}{hr} = 1450 \left[\frac{\dot{Q}, Btu/hr}{(M, lb)\ S} \right]^{1.14}$$

$$\frac{\Delta P}{\Delta t}, \frac{N}{m^2 s} = 4575 \left[\frac{\dot{Q}, W}{(M, kg)\ S} \right]^{1.14}$$

where

S = ullage volume, %

M = total fluid mass

$\dot{Q}$ = total heating rate to fluid

Any heat in addition to that required to raise the supply pressure from 103 kN/m^2 (15 psia) to 345 kN/m^2 (50 psia) was assumed to go into boiloff which was vented overboard.

From Figure D-1 it is seen that even though the Shuttle and Tug supply tanks are of significantly different size the optimum insulation thicknesses are approximately the same; 2.5 cm (1 in.) for LH_2 and 3.8 cm (1.5 in.) for LO_2.

In the case of LHe storage the use of purged MLI without a foam substrate or vacuum jacketing resulted in excessive boil-off weight penalties under ground and boost heating conditions. For the Large HEAO B system, trade-offs were made between use of purged MLI of different thicknesses with a 1.3 cm (0.5 in.) foam substrate and vacuum jacketing with a vapor cooled shield. Comparative weights are presented in Table D-1. The MLI plus foam data assume the use of a GN_2 ground purge to minimize the total heat leakage. The conductivity of GN_2 is less than that of helium. The 1.3 cm (0.5 in.) foam layer was determined to be sufficient to maintain the minimum insulation temperature above that which would liquify the GN_2. From Table D-1 it is seen that the MLI plus foam system is significantly lighter than the vacuum jacketed system and was therefore chosen here for LHe storage. The optimum MLI thickness from Table D-1 is 7.6 cm (3 in.).

Table D-1. HEAO-B Insulation System Weights[1], (Storage of 1097 lb LHe)

	MLI + 0.5 in. foam				Vacuum Jacketed With Vapor Cooled Shield
	2.5 cm (1 in.) MLI	5.1 cm (2 in.) MLI	7.6 cm (3 in.) MLI	10.2 cm (4 in.) MLI	
Weight, lb	264	226	198	208	716
Weight, kg	120	103	90	94	325

(1) Weights include tankage, boil-off and insulation

In the case of the non-cryogenic fluids the only thermal control problem is to maintain liquid temperatures above the freezing point (Ref. Table B-1). The most critical fluid is N_2H_4, due to its relatively high freezing temperature. Calculations were made for the SCS transfer case, showing a maximum of 6.1K (11R) temperature drop between lift-off and fluid transfer without any insulation. The tank was assumed to be coated with an aluminized enamel with an emittance of 0.35. The initial temperature of the N_2H_4 needs to be maintained only above 26K (46F) prior to lift-off to prevent freezing.

APPENDIX E

TANK PRESSURIZATION CALCULATION PROCEDURES

Pressurant requirements for the cryogenic cases were calculated using the equations presented in Reference E-1. These equations were developed empirically from computer calculations and test data.

Only pressurant temperatures up to 289K (520R) were considered in the current analyses. Previous work, such as reported in Reference E-2, indicated the optimum temperature for He and GH_2 pressurization of LH_2 was 167K (300R). For He pressurization of LO_2 the optimum was 278K (500R). In both cases, total weights were relatively insensitive to temperature changes between 167K (300R) and 289K (520R). It was also shown that expulsion while allowing liquid to vaporize in a self pressurizing mode was not weight competitive.

The storable fluids are always assumed to be pressurized with GHe or GN_2 at the same temperature as the liquids.

Once the quantity of pressurant required is determined, pressurant storage bottle volumes are found from; (mass of pressurant required)/(initial pressurant storage density - final density).

For pressurant storage at ambient temperatures, composite bottles are used consisting of Inconel 718 liners with a Kevlar overwrap. These bottles are lightweight and have been developed for the Space Shuttle. The parametric weight equation used is presented below.

$$\text{Bottle Wt., lb} = 3.66 \times 10^{-3} \, (V_B, \text{ ft}^3) \, (P_s, \text{ psia})$$

$$\text{Bottle Wt., kg} = 8.52 \times 10^{-3} \, (V_B, \text{ m}^3) \, (P_s, \text{ kN/m}^2)$$

where

V_B = bottle volume

P_S = initial storage pressure

For all cold pressurant storage (lower than ambient), 6 AL - 4 VTi bottles, as described in Reference 4-2, are used. The applicable weight equation for these bottles is given below.

$$\text{Bottle Wt., lb} = 4.33 \times 10^{-3} \, (V_B, \text{ ft}^3) \, (P_s, \text{ psia})$$

$$\text{Bottle Wt., kg} = 1.01 \times 10^{-2} \, (V_B, \text{ m}^3) \, (P_s, \text{ kN/m}^2)$$

APPENDIX F

SUPPLY MODULES DETAILED WEIGHT DATA

Weight data are presented in Tables F-1 through F-4 for the individual hardware which makes up the supply modules presented in Paragrah 4.2.

Table F-1. Detailed Weights for Tug Supply Using Shuttle Drag

Item	Qty.	Weight, ea		Total Weight	
		kg	lb	kg	lb
LH_2 System					
Tank, Including Baffles and Supports	1	316	695	316	695
Insulation (MLI)	1	26	57	26	57
Purge Bag	1	67	148	67	148
Zero-G Vent System (6 lb/hr)	1	5	10	5	10
Vent Line	1	5	10	5	10
Vent/Relief Valve (3-way)	1	5	12	5	12
Vent Disconnect	1	1.8	4	1.8	4
Fill/Drain/Emergency Dump Line	1	5.4	12	5.4	12
Vertical Fill/Drain Valve	2	3	7	6	14
Horizontal Fill/Drain Valve	1	3	7	3	7
Fill/Drain Disconnect	1	3	7	3	7
Transfer Line	1	8	18	8	18
Transfer Flow Control Valve	1	2.7	6	2.7	6
Transfer Disconnect	1	1.8	4	1.8	4
Transfer Line Vent	1	1.8	4	1.8	4
Tank Pressure Regulator and Check Valve	2	3.6	8	7	16
Insulation Purge Bag Relief	1	2.3	5	2.3	5
Mass Gauging System	1	5	10	5	10
LO_2 System					
Tank, Including Baffles and Supports	1	400	882	400	882
Insulation	1	24	53	24	53
Purge Bag	1	40	89	40	89
Zero-G Vent System (20 lb/hr)	1	5	10	5	10
Vent Line	1	3.6	8	3.6	8
Vent/Relief Valve (3-way)	1	5	12	5	12
Vent Disconnect	1	1.8	4	1.8	4
Fill/Drain/Emergency Dump Line	1	5.4	12	5.4	12
Vertical Fill/Drain Valve	2	5	10	9	20

Table F-1. Detailed Weights for Tug-Supply Using Shuttle Drag (Cont.)

Item	Qty	Weight, ea		Total Weight	
		kg	lb	kg	lb
Horizontal Fill/Drain Valve	1	3	7	3	7
Fill/Drain Disconnect	1	5	10	5	10
Transfer Line	1	6	13	6	13
Transfer Flow Control Valve	1	2.7	6	2.7	6
Transfer Disconnect	1	1.8	4	1.8	4
Transfer Line Vent	1	1.8	4	1.8	4
Tank Pressure Regulator and Check Valve	2	3.6	8	7	16
Insulation Purge Bag Relief	1	2.3	5	2.3	5
Mass Gauging System	1	4	9	4	9
<u>N_2H_4 System</u>					
Tank, Including Bladder and Supports	1	20	43	20	43
Vent Line	-	-	-	.9	2
Vent/Relief Valve	1	1.3	3	1.3	3
Vent Disconnect	1	.9	2	.9	2
Fill/Drain Line	1	.9	2	.9	2
Fill/Drain Valve	1	.9	2	.9	2
Fill/Drain Disconnect	1	.5	1	.5	1
Transfer Line	1	2.7	6	2.7	6
Transfer Line Shut-Off Valve	1	.9	2	.9	2
Transfer Line Disconnect	1	.5	1	.5	1
Transfer Line Vent	1	.9	2	.9	2
Tank Pressure Regulator	1	.5	1	.5	1
<u>Ambient Gaseous Helium System</u>					
Bottle, Including Supports	1	29	63	29	63
Relief Valve	1	.5	1	.5	1
Fill Line and Disconnect	1	.9	2	.9	2
Fill and Pressurization Shut Off Valves	2	.5	1	1	2
Insulation Purge Shut-Off Valve	1	.5	1	.5	1
Insulation Purge Regulator	1	.9	2	.9	2
Insulation Purge Line	1	2.3	5	2.3	5
GHe Purge Line, Valve and Disconnect	-	-	-	2.3	5
<u>Cold He Pressurization and Transfer</u>					
Storage Bottle, Including Supports	1	82	180	82	180
Pressurization Lines	-	-	-	6	13
Shut-Off Valves	2	.9	2	1.8	4
Transfer Disconnect	1	.9	2	.9	2
Fill Line	-	-	-	2.7	6

Table F-1. Detailed Weights for Tug-Supply Using Shuttle Drag (Cont.)

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
Fill Disconnect	1	.9	2	.9	2
Heat Exchanger	1	5	10	5	10
Relief Valve	1	.5	1	.5	1
N_2H_4 Gas Generator	1	3.6	8	3.6	8
Tug Attach Fittings and Legs	-	-	-	11	24
Swing-Out Ring	-	-	-	244	537
Overall Module Support Structure	-	-	-	175	386
Total Dry Weight				1600	3524

Table F-2. Detailed Weights for Separated Tug Supply

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
LH_2 System					
Tank, Including Baffles and Supports	1	316	695	316	695
Insulation (MLI	1	26	57	26	57
Purge Bag	1	67	148	67	148
Zero-G Vent System (6 lb/hr)	1	5	10	5	10
Vent Line	1	5	10	5	10
Vent/Relief Valve (3-way)	1	5	12	5	12
Vent Disconnect	1	1.8	4	1.8	4
Fill/Drain/Emergency Dump Line	1	5	12	5	12
Vertical Fill/Drain Valve	2	3	7	6	14
Horizontal Fill/Drain Valve	1	3	7	3	7
Fill/Drain Disconnect	1	3	7	3	7
Tranfer Line	1	8	18	8	18
Transfer Flow Control Valve	1	2.7	6	2.7	6
Transfer Disconnect	1	1.8	4	1.8	4
Transfer Line Vent	1	1.8	4	1.8	4
Tank Pressure Regulator and Check Valve	2	3.6	8	7	16
Insulation Purge Bag Relief	1	2.3	5	2.3	5
Mass Gauging System	1	5	10	5	10
LO_2 System					
Tank, Including Baffles and Supports	1	400	882	400	882
Insulation	1	24	53	24	53
Purge Bag	1	40	89	40	89

Table F-2. Detailed Weights for Separated Tug Supply (Cont.)

Item	Qty.	Weight, ea.		Total Weight	
		kg	lb	kg	lb
Zero-G Vent System (20 lb/hr)	1	5	10	5	10
Vent Line	1	3.6	8	3.6	8
Vent/Relief Valve (3-Way)	1	5	12	5	12
Vent Disconnect	1	1.8	4	1.8	4
Fill/Drain/Emergency Dump Line	1	5.4	12	5.4	12
Vertical Fill/Drain Valve	2	5	10	9	20
Horizontal Fill/Drain Valve	1	3	7	3	7
Fill/Drain Disconnect	1	5	10	5	10
Transfer Line	1	6	13	6	13
Transfer Flow Control Valve	1	2.7	6	2.7	6
Transfer Disconnect	1	1.8	4	1.8	4
Transfer Line Vent	1	1.8	4	1.8	4
Tank Pressure Regulator and Check Valve	2	3.6	8	7	16
Insulation Purge Bag Relief	1	2.3	5	2.3	5
Mass Gauging System	1	4	9	4	9
<u>N_2H_4 System</u>					
Tank, Including Bladder and Supports	1	30	65	30	65
Vent Line	-	-	-	.9	2
Vent/Relief Valve	1	1.3	3	1.3	3
Vent Disconnect	1	.9	2	.9	2
Fill/Drain Line	1	.9	2	.9	2
Fill/Drain Valve	1	.9	2	.9	2
Fill/Drain Disconnect	1	.5	1	.5	1
Transfer Line	1	2.7	6	2.7	6
Transfer Line Shut-Off Valve	1	.9	2	.9	2
Transfer Line Disconnect	1	.5	1	.5	1
Transfer Line Vent	1	.9	2	.9	2
Tank Pressure Regulator	1	.5	1	.5	1
<u>Ambient Gaseous Helium System</u>					
Bottle, Including Supports	1	29	63	29	63
Relief Valve	1	.5	1	.5	1
Fill Line and Disconnect	1	.9	2	.9	2
Fill and Pressurization Shut-Off Valves	2	.5	1	.9	2
Insulation Purge Shut-Off Valve	1	.5	1	.5	1
Insulation Purge Regulator	1	.9	2	.9	2
Insulation Purge Line	1	2.3	5	2.3	5
GHe Purge Line, Valve and Disconnect	-	-	-	2.3	5

Table F-2. Detailed Weights for Separated Tug Supply (Cont.)

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
Cold He Pressurization and Transfer					
Storage Bottle, Including Supports	1	82	180	82	180
Pressurization Lines	-	-	-	6	13
Shut-Off Valves	2	.9	2	2	4
Transfer Disconnect	1	.9	2	.9	2
Fill Line	-	-	-	3	6
Fill Disconnect	1	.9	2	.9	2
Heat Exchanger	1	5	10	5	10
Relief Valve	1	.5	1	.5	1
N_2H_4 Gas Generator	1	3.6	8	3.6	8
N_2H_4 Settling Thrusters (2 lb)	4	.9	2	3.6	8
N_2H_4 Settling Thrusters Feed Lines and Valves	-	-	-	2.7	6
Tug Attach Fittings and Legs	-	-	-	11	24
Swing-Out Ring	-	-	-	132	290
Overall Module Support Structure	-	-	-	312	688
Total Dry Weight				1641	3615

Table F-3. Detailed Weights for Shuttle Orbiter Supply System

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
N_2O_4 System				143	315
Tank, Including Supports	1	102	224	102	224
Surface Tension Channels	1	6	21	6	21
Fill/Drain Line				2.7	6
Fill/Drain Valve				3.6	8
Fill/Drain Disconnect				2.7	6
Vent Line				1.8	4
Vent/Relief Valve				1.3	3
Vent Disconnect				.9	2
Tank Pressurization Lines				.5	1
Tank Pressurization Shut-Off Valve				.5	1
Tank Pressurization Shut-Off Valve				.9	2
Tank Pressure Regulator				1.3	3
Tank Pressure Regulator				2.3	5

Table F-3. Detailed Weights for Shuttle Orbiter Supply (Cont.)

Item	Qty	Weight, ea.		Total Weight	
		kg	lb	kg	lb
Check Valve				.5	1
Transfer Line				5	11
Transfer Valves	2	1	2.5	2.3	5
Transfer Disconnect				1.3	3
Transfer Line Vent System				1.8	4
Mass Gauging System				2.3	5
MMH System				144	317
Tank, Including Supports	1	102	224	143	224
Surface Tension Channels				8	18
Fill/Drain Line				5	10
Fill/Drain Valve				3.6	8
Fill/Drain Disconnect				2.7	6
Vent Line				2.3	5
Vent/Relief Valve				1.3	3
Vent Disconnect				.9	2
Tank Pressurization Lines				.5	1
Tank Pressurization Shut-Off Valve				.5	1
Tank Pressurization Shut-Off Valve				.9	2
Tank Pressure Regulator				1.3	3
Tank Pressure Regulator				2.3	5
Check Valve				.5	1
Transfer Line				5	11
Transfer Valves	2	1	2.5	2.3	5
Transfer Disconnect				1.3	3
Transfer Line Vent System				1.8	4
Mass Gauging System				2.3	5
Ambient He System				53	117
Bottle, Including Supports				51	113
Fill Line				.5	1
Fill Valve				.9	2
Fill Disconnect				.5	1
LHe System				109	240
Tank, Including Supports				25	56
Surface Tension Channels				12	26
Foam Insulation				5	10
MLI				6	13
Purge Bag				5	12

Table F-3. Detailed Weights for Shuttle Orbiter Supply (Cont.)

Item	Qty.	Weight,ea.		Total Weight	
		kg	lb	kg	lb
Insulation Purge Lines				.9	2
Insulation Purge Shut-Off Valves	2	.9	2	1.8	4
Insulation Purge Regulator				1.3	3
Insulation Purge Disconnect				.5	1
Insulation Purge GN_2 Line				.5	1
Insulation Purge Check Valve				.5	1
Insulation Purge Relief Valve				.9	2
Fill/Drain Line (Vacuum Jacketed)				11	24
Fill/Drain Valve				.9	2
Fill/Drain Disconnect				.9	2
Vent Line				3	7
Vent/Relief Valve (3-Way)				1.8	4
Vent Disconnect				1.3	3
Tank Pressurization				.5	1
Tank Pressurization Shut-Off				.5	1
Tank Pressurization Regulator				1.3	3
Tank Pressurization Heater (100W)				1.3	3
Tank Pressurization Pump (20W)				.9	2
Transfer Line				7	15
Transfer Valve				.9	2
Transfer Disconnect				.9	2
Transfer Line Vent System				1.3	3
High Pressure Transfer Pump (3 KW)				11	25
Zero-G Vent System				2.3	5
Mass Gauging System				2.3	5
Overall Supply Module Mounting Structure				183	404
Case 1 Total Dry Weight				632	1393
LH_2 System				176	387
Tank Including Supports				64	141
Surface Tension Channels				13	29
MLI				5	11
Purge Bag				13	29
Pressurant Storage Bottle				25	56
Pressurization Lines				.9	2
Pressurization Shut-Off Valves	3	.3	.7	.9	2
Pressurization Shut-Off Valve				.5	1
Pressurization Regulator				.5	1
Pressurization Regulator				.9	2

Table F-3. Detailed Weights for Shuttle Orbiter Supply (Cont.)

Item	Qty.	Weight, ea.		Total Weight	
		kg	lb	kg	lb
Storage Bottle Relief Vavle				.5	1
GHe Fill Line				.9	2
GHe Fill Disconnect				.5	1
Insulation Purge Lines				1.3	3
Insulation Purge Shut-Off	2	.7	1.5	1.3	3
Insulation Purge Regulator				.9	2
Insulation Purge Relief Valve				1.3	3
Insulation Purge Ground Line				.9	2
Insulation Purge Ground Disconnect				.9	2
Fill/Drain Line (Vacuum Jacketed)				11	24
Fill/Drain Valve				3.6	8
Fill/Drain Disconnect				3.6	8
Vent Line				5	11
Vent/Relief Valve (3-Way)				2.3	5
Vent Disconnect				1.3	3
Zero-G Vent System				4	8
Transfer Line				2.3	5
Transfer Valves	2	.9	2	1.8	4
Transfer Disconnect				.9	2
Transfer Line Vent System				1.3	3
Transfer In-Tank Shut-Off				.5	1
Transfer Pump (250W)				3	7
Mass Gauging System				2.3	5
LO_2 System				139	307
Tank, Including Supports				49	107
Surface Tension Channels				11	24
MLI				5	10
Purge Bag				8	17
Pressurant Storage Bottle				16	36
Pressurization Lines				.5	1
Pressurization Shut-Off Valves	3	.3	.7	.9	2
Pressurization Shut-Off Valve				.5	1
Pressurization Regulator				.5	1
Pressurization Regulator				.9	2
Storage Bottle Relief Valve				.5	1
GHe Fill Line				.9	2
GHe Fill Disconnect				.5	1
Insulation Purge Lines				1.3	3

Table F-3. Detailed Weights for Shuttle Orbiter Supply (Cont.)

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
Insulation Purge Shut-Off	2	.5	1	.9	2
Insulation Purge Regulator				.9	2
Insulation Purge Relief Valve				.9	2
Insulation Purge Ground Line				.9	2
Insulation Purge Ground Disconnect				.9	2
Fill/Drain Line				10	23
Fill/Drain Valve				3.6	8
Fill/Drain Disconnect				3.6	8
Vent Line				5	10
Vent/Relief Valve (3-Way)				2.3	5
Vent Disconnect				1.3	3
Zero-G Vent System				3.6	8
Transfer Line				3	7
Transfer Valves (Outside Tank)	2	.9	2	1.8	4
Transfer Valve (In-Tank)				1.3	3
Transfer Disconnect				.9	2
Transfer Line Vent System				1.3	3
Mass Gauging System				2.3	5
Case 2 Total Dry Weight				947	2087

Table F-4. Detailed Weights for Multiple Receivers Supply System

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
Hg System (Solar Electric Propulsion)				69	153
Tank, Including Supports	2	26	54	49	108
Diaphragm	2	1	1	2.3	5
Transfer Line				.5	1
Transfer Valves	2	1	2.5	.5	1
Transfer Disconnect				.5	1
Tank Pressurization Line				.5	1
Tank Pressurization Shut-Off				.2	.5
Tank Fill Valves (Manual)	2	.2	.5	.5	1
Vent/Relief Valve				.5	1
GN_2 Storage Tank				12	26
GN_2 Storage Tank Relief				.2	.5
GN_2 Fill Line				.5	1
GN_2 Fill Valve				.5	1

Table F-4. Detailed Weights for Multiple Receivers Supply System (Cont.)

Item	Qty.	Weight, ea. kg	Weight, ea. lb	Total Weight kg	Total Weight lb
GN_2 Fill Disconnect				.5	1
Mass Gauging				1.8	4
Large HEO B System				181	398
LN_2 System					
Tank, Including Supports				11	25
Paddle and Paddle Drive System				5	11
MLI				.5	1
Purge Bag				1.8	4
Fill/Drain Line				1.3	3
Fill/Drain Valve				.9	2
Fill/Drain Disconnect				.9	2
Vent Line				.9	2
Vent Valves	2	.7	1.5	1.3	3
Vent Disconnect				.9	2
Tank Pressurization Lines				.5	1
Tank Pressurization Shut-Off Valve	2	.2	.5	.5	1
Tank Pressurization Regulator				.5	1
Insulation Purge Lines				.5	1
Insulation Purge Regulator				.5	1
Insulation Vent				.5	1
Chilldown Line				.5	1
Chilldown Shut-Off Valves	2	.7	1.5	1.3	3
Chilldown Line Vent System				.9	2
LHe System					
Tank, Including Supports				54	120
Paddle and Drive System				22	48
Foam Insulation				6	13
MLI				14	30
Purge Bag				13	28
Zero-G Vent (Wall Type)				2.3	5
Fill/Drain Line (Vacuum Jacketed)				7	16
Fill/Drain Valve				.9	2
Fill/Drain Disconnect				.9	2
Vent Line				1.8	4
Vent/Relief Valves	2	.9	2	1.8	4
Vent Disconnect				.9	2
Tank Pressurization Line				.5	1

Table F-4. Detailed Weights for Multiple Receivers Supply System (Cont.)

Item	Qty.	Weight, ea kg	Weight, ea lb	Total Weight kg	Total Weight lb
Tank Pressurization Shut-Off				.5	1
Tank Pressurization Regulator				.5	1
Tank Pressurization Heater				1.3	3
Tank Pressurization Pump				.5	1
Insulation Purge Lines				.5	1
Insulation Purge Check Valve				.5	1
Insulation Purge Shut-Off				.5	1
Insulation Purge Disconnect				.5	1
Insulation Vent				.9	2
Transfer Line				.5	1
Transfer Valves	2	.7	1.5	1.3	3
Transfer Disconnect				.5	1
Transfer Line Vent System				1.3	3
Mass Gauging				2.3	5
GHe Storage					
Bottle, Including Supports				13	28
Ground Fill Line				.5	1
Ground Fill Valve				.5	1
Ground Fill Disconnect				.5	1
N_2H_4 System (Satellite Control Section)				72	159
Tank, Including Supports				29	64
Surface Tension Channels				2.7	6
Fill/Drain Line				5	12
Fill/Drain Valve				1.8	4
Fill/Drain Disconnect				1.3	3
Vent Line				1.3	3
Vent/Relief Valves	2	.7	1.5	1.3	3
Vent Disconnect				.5	1
Tank Pressurization Line				.5	1
Tank Pressurization Shut-Off Valves	2	.2	.5	.5	1
Tank Pressurization Regulator				.5	1
Transfer Line				.9	2
Transfer Valve				1.3	3
Transfer Valves	2	.7	1.5	1.3	3
Transfer Pump				8	18
Transfer Disconnect				.9	2
Transfer Line Vent System				.9	2
GN_2 Storage Bottle				11	24

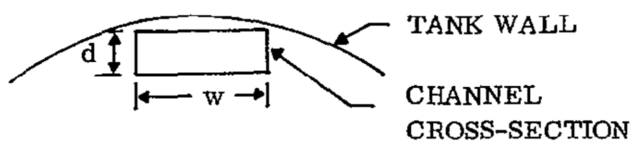
Table F-4. Detailed Weights for Multiple Receivers Supply System (Cont.)

Item	Qty.	Weight, ea.		Total Weight	
		kg	lb	kg	lb
GN_2 Fill Line				.5	1
GN_2 Fill Disconnect				.5	1
Mass Gauging				1.8	4
Docking Module				21	46
Overall Support Structure				77	170
Total Dry Weight				420	926

APPENDIX G

SCREEN CHANNELS DESIGN CALCULATION PROCEDURES

Described here are the methods used to determine the optimum rectangular dimensions of the individual screen channel systems. In each case, there were assumed to be eight semicircular channels (0.78 inch or 45° apart) connecting an upper and a lower manifold. Flow into the channel from the tank was assumed to be through the wide side (both inner and outer) of the channel only. The liquid pool residual was assumed to flow through one channel only at the midsection of the tank (to give maximum distance between channels). The basic cross-section is illustrated below.



Preliminary assumptions were made for screen mesh, channel dimensions, clearance between the channel and the tank wall, and the flow rate from the tank (assumed to be flowing in one channel only). The screen retention capability was then computed from the following relationship:

$$\Delta P_T = 4 \times \sigma / S.F. \times D_{BP}$$

where σ = fluid surface tension

D_{BP} = fluid bubble point diameter through screen

S.F. = safety factor (2 used here)

The screen retention capability was further related to the individual pressure losses as:

$$\Delta P_T = \Delta P_f + \Delta P_b + \Delta P_h + \Delta P_s$$

where ΔP_f = friction loss

ΔP_b = bend loss

ΔP_h = hydrostatic head

ΔP_s = screen pressure loss

$$\Delta P_f = f\frac{L}{D} \cdot \rho \frac{V_e^2}{2g_c}$$

$$\Delta P_b = KEC\, \rho \frac{V_e^2}{2g_c}$$

$$\Delta P_h = \rho h\ a/g_c$$

$$\Delta P_s = A\mu\ V_{e_s} + B_\rho\ V_{e_s}^2$$

where f = coefficient of friction, calculated from a roughness factor of 0.04

L = length of channel from midsection to outlet

D = hydraulic diameter = $2 \times w \times d/(w + d)$

w = channel width

d = channel depth

ρ = fluid density

V_e = fluid velocity in the channels

a = local acceleration (assumed to be 10^{-4} g's)

g_c = acceleration of gravity at sea level

h = one-half the diameter of the tank

K, E, C = constants relating aspect ratio (channel width versus depth) to velocity loss in bends, aspect ratio factor and total angle of bends

V_{e_s} = fluid velocity through the screen

μ = fluid viscosity

$$\left.\begin{array}{l} A = \alpha\,[Q b a^2/\epsilon^2\, g_c] \\ B = \beta\,[Q b/\epsilon^2\, g_c\, D] \end{array}\right\}\ \text{Screen Properties}$$

The actual screen mesh used is determined from the best combination of head retention and pressure drop for minimizing residuals and maximizing the allowable outflow rate. Previous studies (Reference G-1) showed that a 200 × 600 Dutch twill screen would be a good choice.

For the 200 × 600 screen:

$$\alpha = 7.0$$

$$\beta = 0.52$$

$$b = 4.92 \times 10^{-4} \text{ ft}$$

$$a = 10852 \text{ ft}^{-1}$$

$$D = 13.1 \times 10^{-5} \text{ ft}$$

$$\epsilon = 0.562$$

$$Q = 1.3$$

$$A = 5.189 \times 10^{4} \text{ sec}^{2}/\text{ft}^{2}$$

$$B = 0.250 \text{ sec}^{2}/\text{ft}$$

$$D_{BP} = 30 \mu\text{m}$$

For a given channel configuration, the screen retention capability (ΔP_T) was calculated and the individual pressure losses (ΔP_f, ΔP_b and ΔP_h) were calculated and subtracted from it. This left the screen pressure loss as the only outstanding amount. The fluid velocity through the screen was then calculated. The fluid flow rate divided by the fluid velocity through the screen gave the area of screen required to maintain the flow. Based on the geometry of the screen, and the wetted area required, the volume of the residual pool of fluid was calculated. The channels were assumed to be full (i.e., no screen breakdown has occurred), and the total amount of fluid in the channels was calculated and added to the residual pool for a total amount of fluid residuals. The above procedure was programed for a Hewlett-Packard 9110 computer.

The calculations were repeated for different channel cross-sections until a minimum weight configuration was found.

APPENDIX H

SYMBOLS

A	area
ACS	attitude control system
APS	auxiliary propulsion system
APU	auxiliary power unit
a	local acceleration
Bo_{NR}	Bond number = ρ a R^2/σ
CM	crew module
CF	collapse factor for pressurant calculation
C_P	specific heat at constant pressure
c.g.	center of gravity
D	diameter
E	modulus of elasticity, energy
ECLSS	environmental control life support system
EPS	electrical power system
ETR	Eastern test range
EVA	extra vehicular activity
FPR	flight performance reserve
F_{NR}	Froude no. = $V_e^2/g_c R$
GSE	ground support equipment
g_c	gravitational constant (32.2 ft/sec^2)
HEAO	High Energy Observatory
HPI	high performance insulation
h	specific enthalpy of fluid
h_c	pull-through height for liquid outflow
L	length
LeRC	NASA-Lewis Research Center

LMSC	Lockheed Missiles & Space Corp.
MLI	multilayer insulation
MPS	main propulsion system
m	mass
$\dot{m}$	mass flow rate
NAR	North American Rockwell
NPSH	net positive suction head
OAS	orbit adjust system
OMS	orbit maneuvering system
P	absolute pressure
$\dot{P}$	power
POP	perpendicular to Earth orbit plane
PPO	polyphenylene oxide
Q	heat transfer
$\dot{Q}$	heat transfer rate
R, r	radius
Re_{ND}	$\rho V_e D/\mu$
RCS	reaction control system
S	ullage volume, %
S_{TU}	ultimate material tensile stress
SCS	Satellite Control Section
SEPS	Solar Electric Propulsion Stage
S/C	spacecraft
S/O	shut-off
T	absolute temperature
t	thickness
u	specific internal energy of fluid
V	volume
$\dot{V}$	volume flow rate
V_e	velocity

W	weight
We_{NR}	Weber no. $= \rho R V_e^2/\mu$
WTR	Western test range
Δ	differential change
θ	time
λ	latent heat of vaporization
μ	dynamic viscosity
ρ	density
σ	surface tension
ϕ	phase
ω	angular speed

<u>Subcripts</u>

BP	bubble point
f	final
i	initial, insulation
L	liquid
l	line
T	total
t	tank
w	wall

APPENDIX I

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