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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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LOW-G FLUID TRANSFER TECHNOLOGY STUDY

EXECUTIVE SUMMARY

May 1976

By
J. A. Stark

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FOREWORD

This summary report was prepared by the Convair Division of General Dynamics Corporation in partial fulfillment of Contract NAS3-17814. The complete technical report is published as NASA CR-134911. 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 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.

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."

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. Drowns	-	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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INTRODUCTION

This report presents a summary of 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 efficient 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 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.

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.

CONCLUSIONS AND RECOMMENDATIONS

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. 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

STUDY RESULTS AND DISCUSSION

RECEIVER CONFIGURATIONS AND CHARACTERISTICS

Work performed under this task is illustrated in Figure 1. Only non-DoD missions were considered. Benefits can be derived for DoD missions, but these missions

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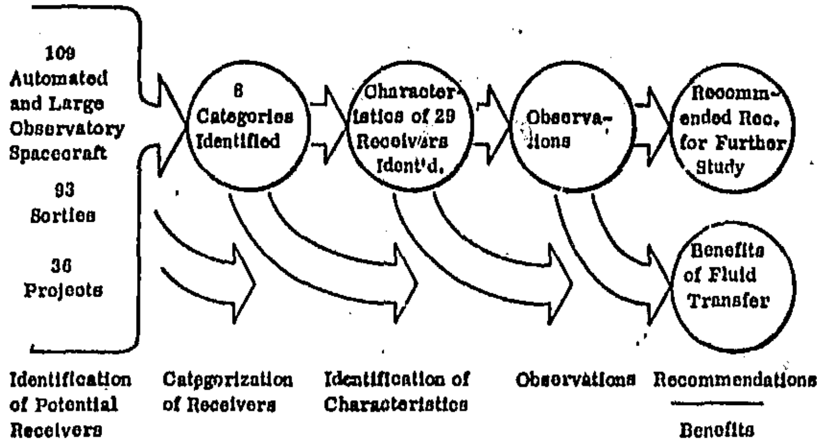
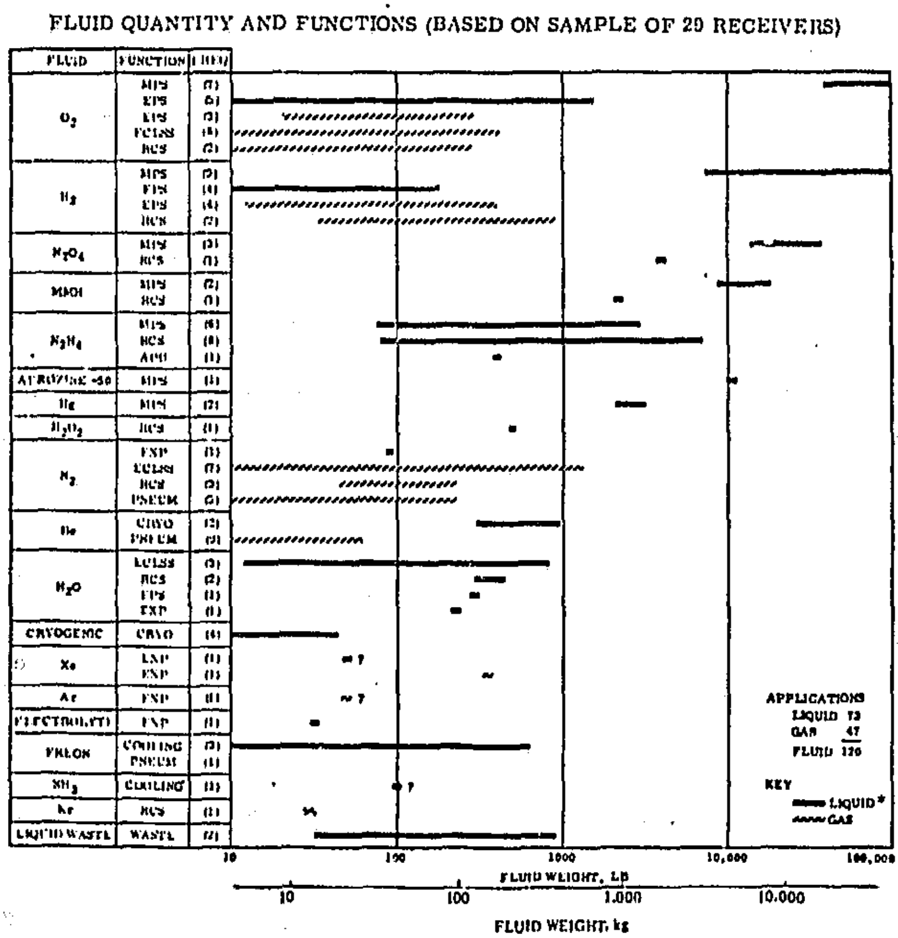


Figure 1. Receivers Configurations and Characteristics Work

were beyond the scope of this study. Spacelab was 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, experiments were included because they are potential receiver candidates for future spacecraft.

Compilation of fluid characteristics for a selected sampling of receivers is presented in Figure 2. The sample was selected to provide a cross-section of different fluid types, different fluid quantities, and examples from different receiver categories for



* Includes supercritical fluids

Figure 2. Sample Fluid Quantity and Function

potentially viable receivers. The frequency of resupply was not considered.

Planned and potential future space operations cover a wide range of activities as illustrated in Figure 3. The potential benefits of fluid transfer which are identified are found to be in the general category of cost effectiveness or safety and are summarized in Figure 4.

TRANSFER SYSTEMS DEFINITION

This task was to conceptually define overall in-orbit fluid transfer systems. The supply modules maximum allowable weight is 29.510 kg (65,000 lb). The maximum size is 12.2 m (40 ft) long by 4.6 m (15 ft) diameter. This allows a 6.1 m (20 ft) length for other payload. The baseline orbit for fluid transfer is 296 km (160 n.mi.).

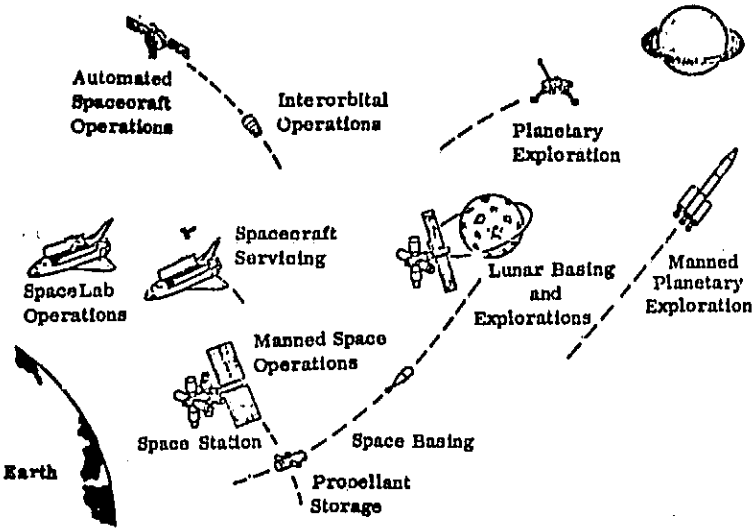


Figure 3. Planned and Potential Future Space Operations

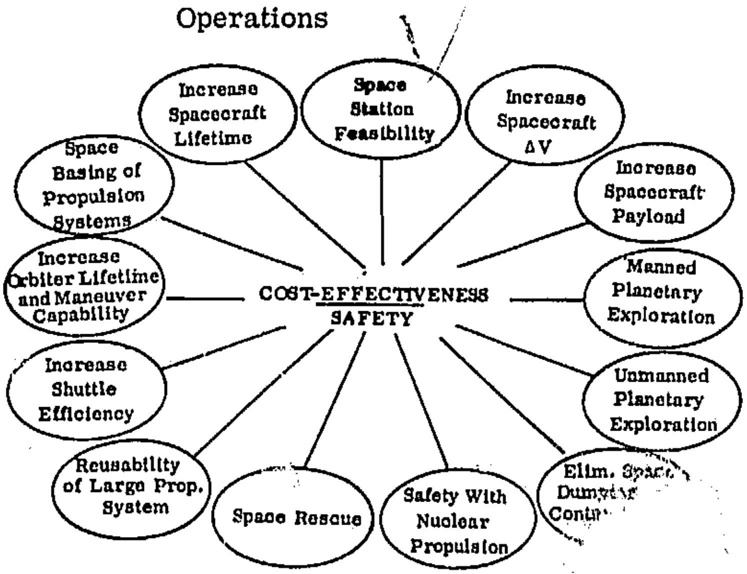


Figure 4. Potential Benefits of Fluid Tr

Based on results of the previous task, the three receiver systems listed in Table 1 were chosen for supply system design. Also presented in Table 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 1 to the Orbiter in a single mission, two different transfer cases were considered for supplying

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Table 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.[(3)] °K	Initial Wall Temp.[(3)] °R
Space Tug	Main Propulsion [(2)]	LH_2	3462	7626	49.5	1748	1	Al Aly	228	502	152	22	256	460
	Main Propulsion [(2)]	LO_2	1978	43574	18.1	640	1	Al Aly	133	294	141	20.5	256	460
	Auxiliary Propulsion	N_2H_4	153	336	-	-	1	-	-	-	-	-	256	460
	Tank Pressurization	He	4.1	9	-	-	1	CRES	-	-	22754	3300	256	460
Space Shuttle	OMS	N_2O_4	17633	38840	-	-	5	Ti	-	-	-	-	-	-
Orbiter With Kits	OMS	MMH	10640	23450	-	-	5	Ti	-	-	-	-	-	-
	OMS Pressurization	He	104	230	0.48	16.8	5	Kevlar Wrap'd Ti	133	294	33096	4800	311	560
	EPS & ECLSS	O_2	2837	6248	0.32	11.3	8	Inc. 718	42.2	93	6560	950	350	630
	EPS	H_2	334	736	0.61	21.7	8	Al Aly	33.6	74	1965	285	350	630
Mult. 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	446	981	110	16.0	267	480
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 childown requirements

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

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

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 was 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.

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 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.

Capillary acquisition, fluid rotation, positive expulsion (bladder, bellows, diaphragm), and linear acceleration methods of acquisition as shown in Figure 5, were considered.

Capillary acquisition uses the surface tension retention capability of screen channels to position liquid within a tank. Bellows are thin walled convoluted tubes composed of circumferential corrugated elements. Fluid to be transferred is 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 were not included due to their combination of high weight and moving seal problems, especially with cryogenics. The fluid rotation 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. The linear acceleration concept utilizes external forces to orient liquid at one end of a tank for transfer. 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.

The surface tension, fluid rotation and linear acceleration systems were determined to be 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

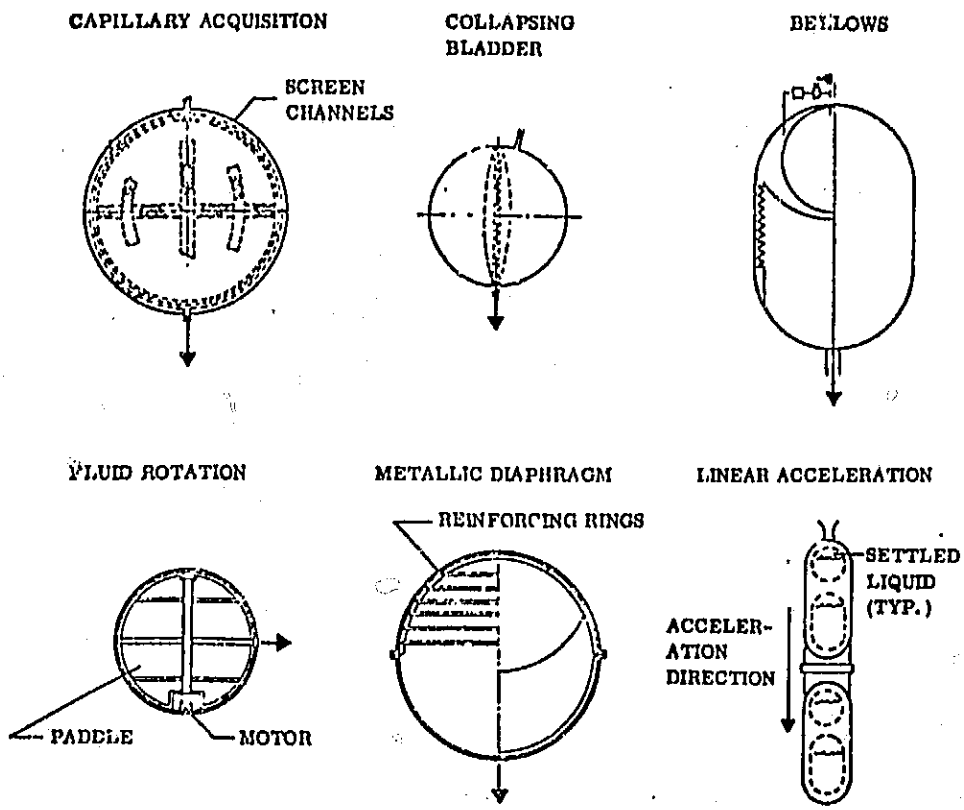

Figure 5. Interface Control and Liquid Acquisition Systems

sensitive to many unknowns which could not be resolved by preliminary analysis, thus detailed conceptual designs were developed for both the drag and auxiliary propulsion concepts.

In the case of Shuttle supply (all fluids) the surface tension and fluid rotation concepts have the best potential. In this supply case, weight is critical and the surface tension concept was chosen for its slightly lower weight over the fluid rotation concept.

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

In the case of the Large High Energy Observatory B the fluid rotation concept was chosen, primarily due to its lower development risk based on the fact that it is a positive force system.

The screen device was chosen for the Satellite Control Section (SCS) supply due to low weight and reusability. Also a channel type surface tension screen system for low-g engine feed is currently employed in the SCS.

Tug supply with supply module and Tug separated from the Orbiter, is shown operationally in Figure 6 and schematically in Figure 7. Trade studies were made to optimize supply tank pressurization/helium transfer, receiver pressure control, supply tankage geometry and thermal control, transfer time, and orientation acceleration level and propulsion. In the resultant system an acceleration of 10^{-4} g's 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 part of the supply module and uses N_2H_4 stored in the same tank used to supply the Tug and the gas generator for pressurant heating. 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 blow-down from 2070 kN/m^2 (300 psia) to approximately 689 kN/m^2 (100 psia). 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.

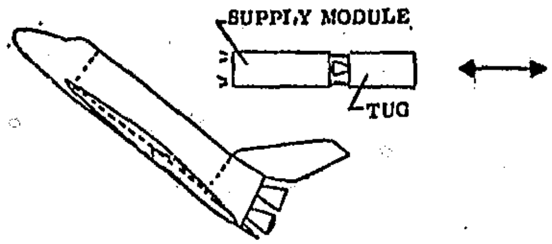

Figure 6. Separated Tug Supply

Helium is transferred to the Tug from a high pressure 33120 kN/m^2 (4800 psia) bottle stored in the LH_2 tank and which is also

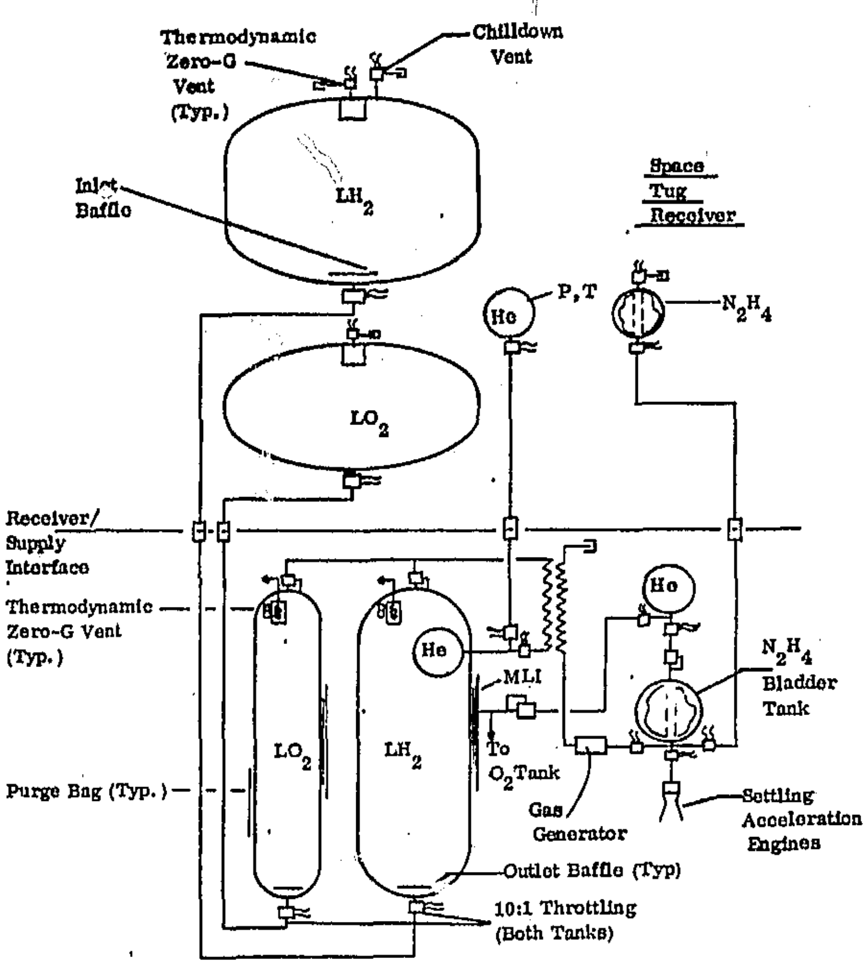

Figure 7. Transfer System No. 1 Schematic

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 zero-g 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 zero-g vent which will operate to maintain the required liquid vapor pressure for maximum loading.

Illustrated in Figure 8 is the Tug supply system 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. Otherwise, the system is the same as for the separated supply as shown in Figure 7. The Figure 8 Orbiter orientation provides maximum drag and 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. Optimum firing times are 180 seconds for settling and 660 seconds for scavenging. 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. The overall transfer time following settling is 72 ks (20 hours).

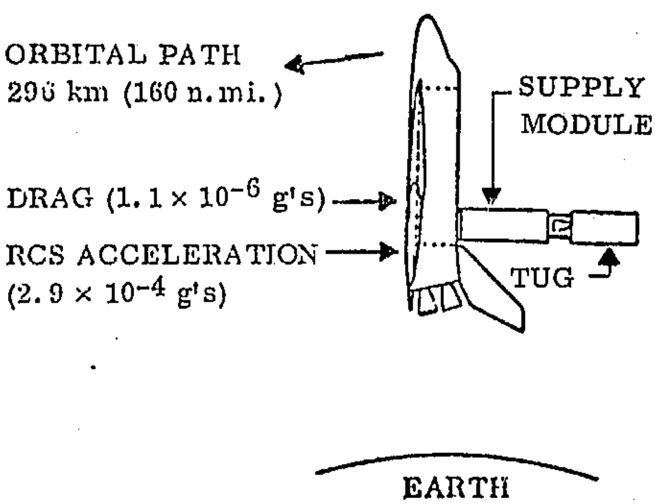

Figure 8. Space Tug Supply Using Shuttle Drag

Calculations were performed for transfer at both 185 km (100 n.mi.) and 296 km (160 n.mi.) orbits. Considering both Tug payload placement capability and fluid transfer optimization, a 296 km (160 n.mi.) orbit was determined to be best.

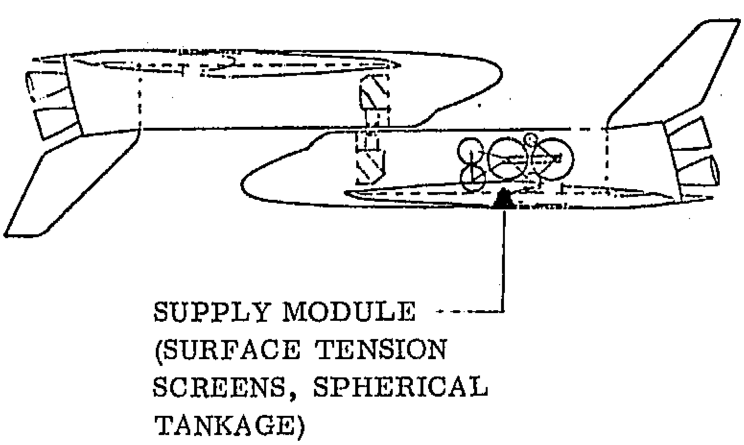

Figure 9. Shuttle Orbiter Supply

The Shuttle Orbiter supply concept is illustrated in Figures 9 and 10. Trade studies were accomplished to optimize tankage geometry and packaging, receivers pressure control and filling, helium transfer, supply tank pressurization, and screen system configuration. A major problem was to define an

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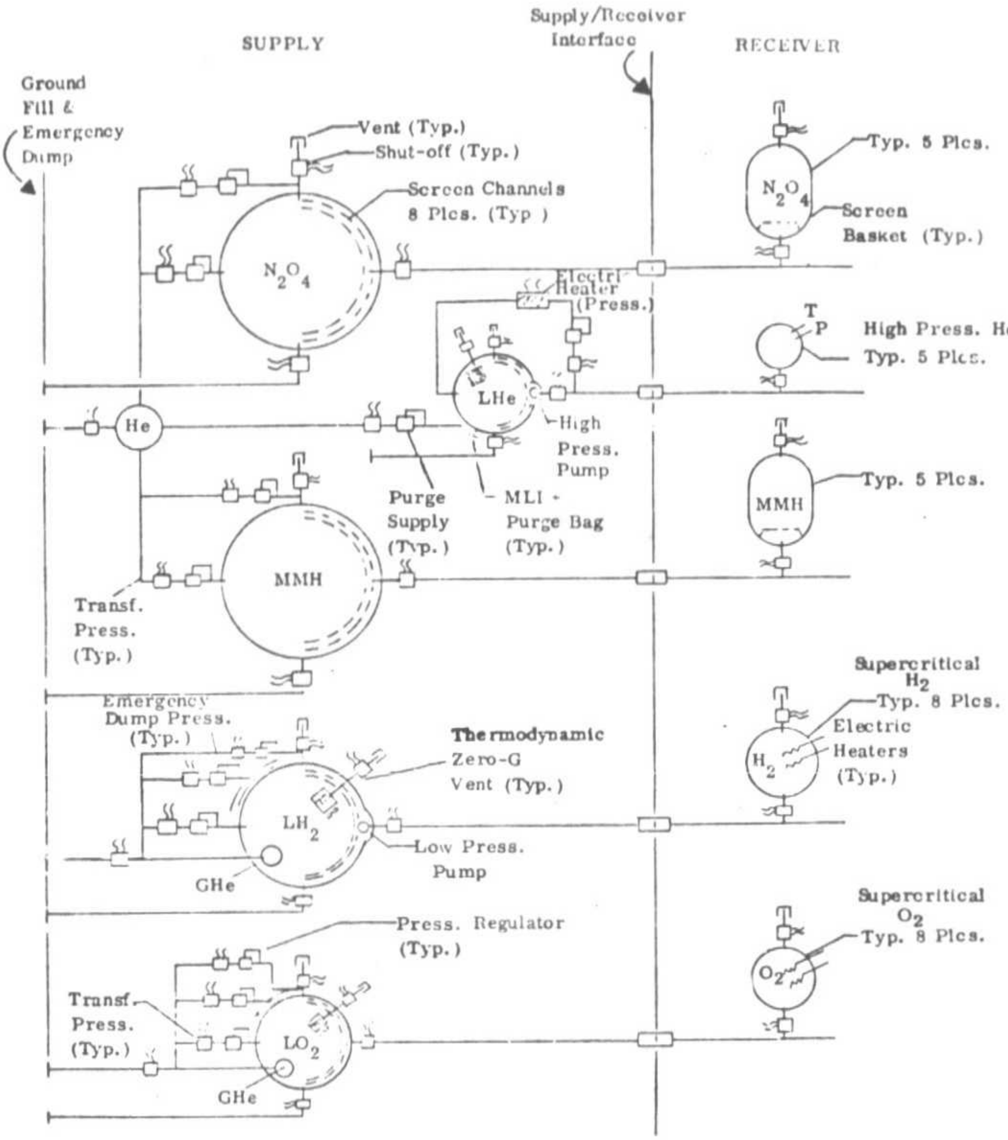

Figure 10. Shuttle Orbiter Supply Schematic

efficient method of filling the high pressure 33.1 MN/m^2 (4800 psia) ambient temperature helium bottles located on the Shuttle receiver. It was determined to transfer LHe to these receivers by a high pressure low flow 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. Calculations show a final 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 temperature for their subsequent use.

For H_2, 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 pressure of 276 kN/m^2 (40 psia). 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 O_2 is essentially the same except that in this case the receivers are vented until a wall temperature of 250K (450R) is reached with lock-up and final filling to 241 kN/m^2 (35 psia).

Overall power requirements and transfer times are presented in Table 2.

Table 2. Orbiter Supply 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)

In all cases low-g liquid acquisition for transfer is accomplished using eight screened channels in each tank. 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. Helium for N_2O_4 and MMH transfer 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.

The multiple receivers transfer system is presented schematically in Figure 11. The overall supply system is designed to supply all three receivers on a single mission, however, 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 for control of the center of gravity, necessitated by the high concentration of weight of the mercury. Double wall tanks are employed for safety to eliminate the chance of a spill of the highly corrosive mercury into the Shuttle payload bay.

The major problem is with the HEAO-B, where 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 receiver venting to maintain this pressure was determined to be unfeasible.

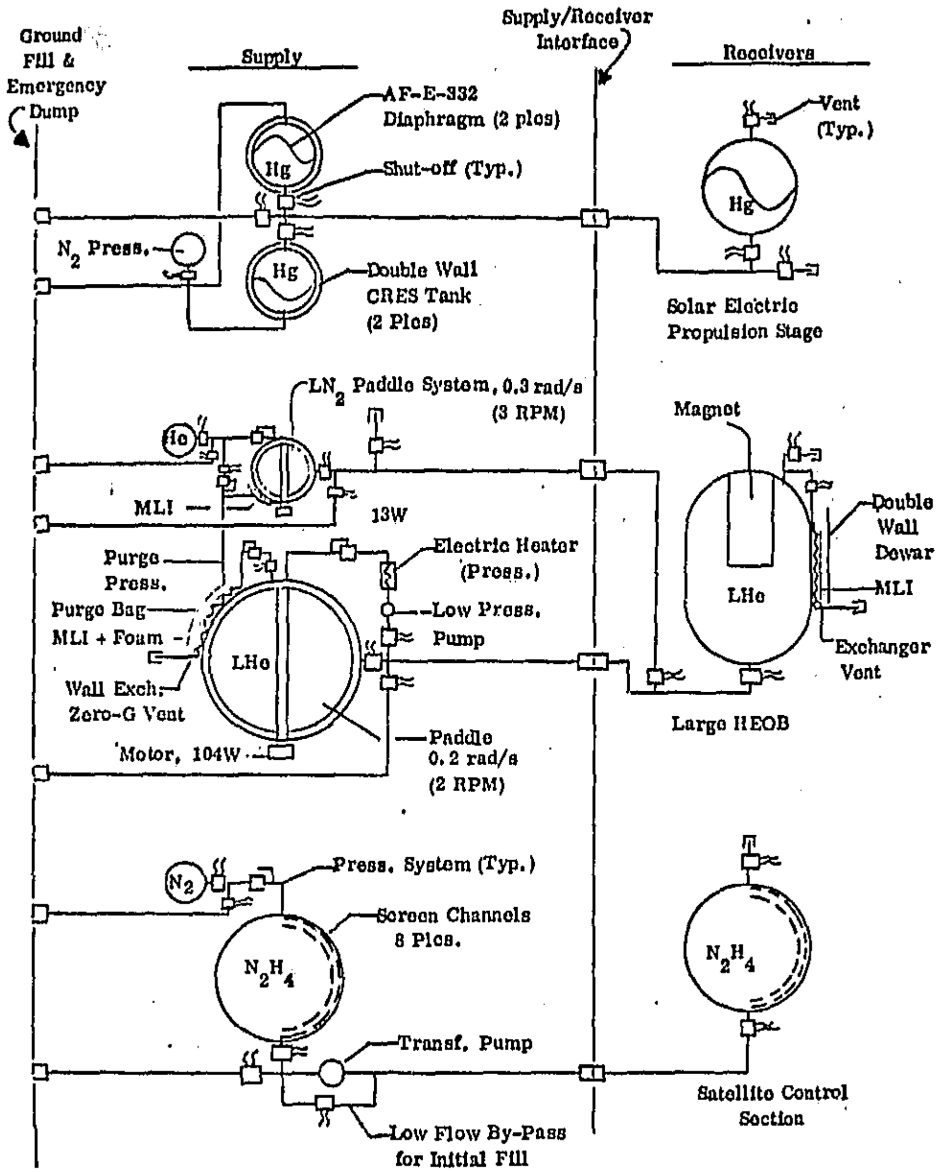

Figure 11. Multiple Receivers Supply Schematic

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 3. As shown, there is a significant weight difference between chill-down where saturated vapor is vented versus venting of superheated vapor at a temperature corresponding to the tank wall as it chills; especially for helium with its low heat of vaporization and high vapor specific heat. Data are also presented in Table 3, assuming that saturated vapor venting chills the magnet while superheated vapor chills the wall. It is assumed that wall cooling may 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 Table 3 cases is any liquid assumed to be lost directly through the vent. Special means would be needed to insure this.

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Table 3. Methods of Chilldown of Large HEAO B

Chilldown Assumption	(1) System Weights			
	Liquid Helium Only		Liquid Nitrogen Plus LHe	
	kg	(lb)	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).

In general, the use of LN_2 as a pre-chill resulted in significantly lower total system weight than use of helium alone. Use of helium alone would be weight competitive only if the chilldown system could be designed such that helium vapor left the receiver at close to the temperature of the hardware as it was being chilled. This possibility would need to be investigated by considerable analysis and test.

Due to the low-g environment, the major problem with the SCS system is to insure that the screen channels in the receiver are full at the end of transfer.

A weight summary of the various transfer systems is presented in Table 4. 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.

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 associated with chilldown of the HEAO B receiver.

Table 4. 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)
Lift-off Weight, kg (lb)	25421 (55994)	25620 (56433)	29412 (64785)	29510 (65000)	4246 (9352)
Fluid Supply/Lift-Off Weight, %	92.1	91.3	96.5	94.5	81.0
Total Fluid Residuals, %	0.86	1.1	1.0	1.3	1.9

ANALYSIS OF SHUTTLE/TUG FLUID TRANSFER BENEFITS

Presented here are the results of work to quantify some of the benefits of employing the Tug and Shuttle transfer systems defined in the previous task.

Figure 12 shows the potential gains in performance to circular orbit altitudes from the Eastern Test Range (ETR) of in-orbit supply of Orbiter OMS fluids (N_2O_4, MMH, He). Similar gains were found for launch from the Western Test Range (WTR). A typical sequence of events is illustrated in Figure 13. It was determined to be optimum 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 Figure 12 represent the weights of the empty OMS kits, which are payload chargeable. 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 fuel the supply Orbiter OMS engines.

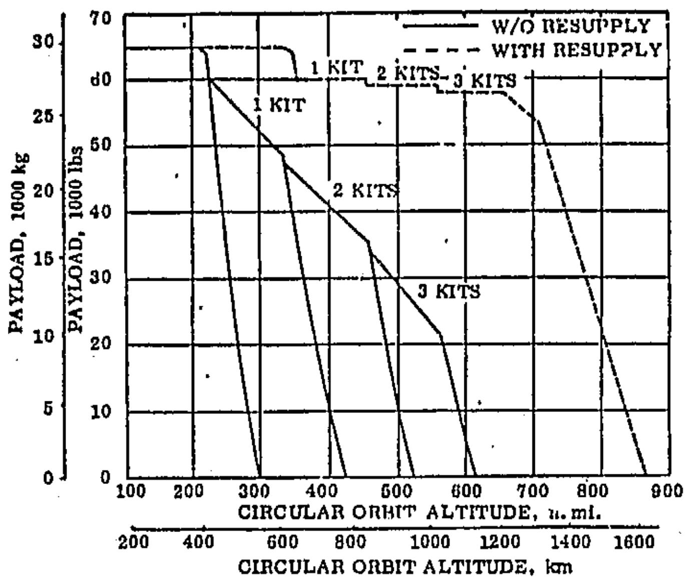

Figure 12. Shuttle Orbiter Performance With and Without In-Orbit Fluid Supply (Launch From ETR)

Space Tug performance may be improved by in-orbit fluid transfer to the extent shown in Figure 14. 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. The 296 km (160 n.mi.) orbit was chosen as a basic reference or standard consistent with the current Shuttle operating philosophy. The kickstage data presented in Figure 14 is based on use with a reusable Tug to increase overall performance

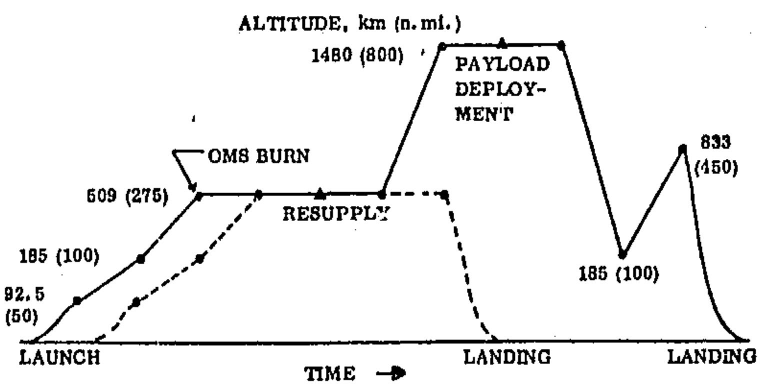

Figure 13. Typical Orbiter Resupply Sequence of Events

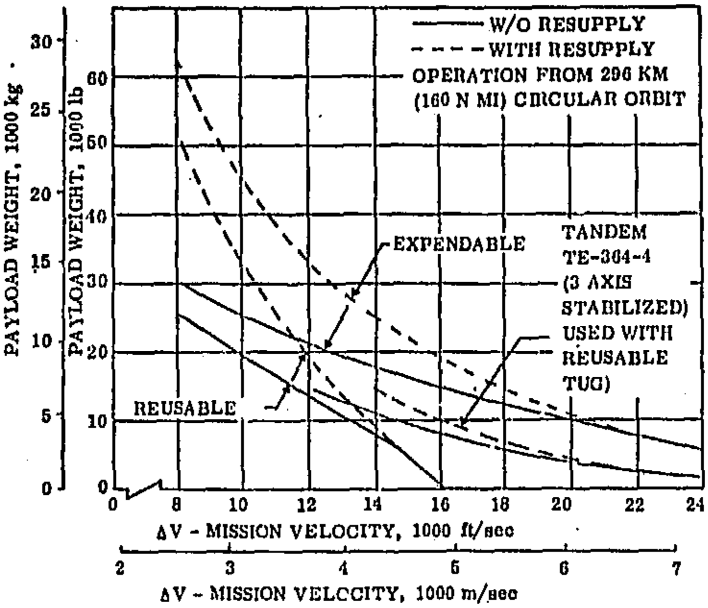

Figure 14. Tug Performance With and Without In-Orbit Fluid Supply (Launch From ETR)

Typical Tug missions for which the Figure 14 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 14 are basic mission velocities assuming a one-way trip. Vehicle velocities required for Tug return in the reusable cases are only reflected in Figure 14 by reduced payload capability. Table 5 presents a comparison of Tug payload capability with and without in-orbit fluid supply for the Mars Sample Return mission. It is noted that data are 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 partially full and one for in-orbit fluid supply). Supply module(s) have not been designed for the specific case of combination Tug and Shuttle in-orbit supply, however, assuming development of the technology required for the individual Tug and Shuttle in-orbit transfer concepts, design of a combination system should be within the state-of-the-art.

Table 5. Mars Sample Return Tug Payload Capability, Tug Operations From 296 km (160 n mi) ΔV = 3813 m/s, (12,500 ft/s)

Tug	Payload kg	Payload lb	Re-supply	Total Shuttle Flights
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

With respect to economic benefits of in-orbit transfer, the following areas are covered.

a. Low Cost Payload Design

b. Extended Duration Shuttle Missions

c. Recovery of Disabled Orbiter

d. Increased Mission Capabilities

The cost of development and production of supply modules was beyond the scope of the present study and was therefore not included in the following cost data.

Low Cost Payload Design - This is concerned with 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 savings possible from applying low cost payload design to the Mars Sample Return mission (Table 5). A reusable Tug with Tug-only in-orbit fluid supply is used. 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. The current Mars Sample Return payload design was derived to have a weight of 4994 kg (11,000 lb), a nonrecurring cost of $598M and a recurring cost of $108M. The maximum cost savings which could be realized by employing low cost design concepts was determined to be 29% of the $598M 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 15. From Figure 15 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)] 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 95% and 18%. This results in corresponding cost reductions of 90% and 25% of maximum. 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 $13.6M and is based on $10.5M from J. C. Fletcher 1974 Senate hearings, escalated 30% to 1975 dollars. This overall cost savings is summarized in Table 6.

This illustrates a potential benefit of in-orbit fluid supply for one mission. Savings of a similar nature can be accommodated on all missions where Shuttle/Tug performance in the nominal mode is taxed.

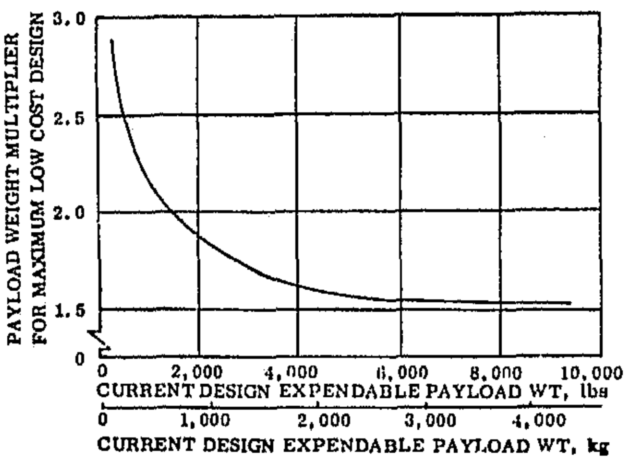

Figure 15. Low Cost Payload Design Weight Growth Relationships

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. The number of experiments which desire extended duration missions fall into the 75% to 100% range for most payload disciplines. As the mission is extended beyond the standard seven days, additional life support expendables are required for the crew, and additional power supply is

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Table 6. 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. 15	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,	25%	90%
Achievable $ Savings		
Non-Recurring	$43M	$150M
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 $)

needed for the experiments. The useful payload weight is reduced to zero typically at a mission duration of thirty days as shown in Figure 16. The useful payload may be retained through fluid resupply of the supercritical oxygen and hydrogen EPS and ECLSS expendables. As shown in Figure 16, resupply of these expendables every ten days will allow 5993 kg (13,200 lbs) of experiments to be carried. 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. 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.

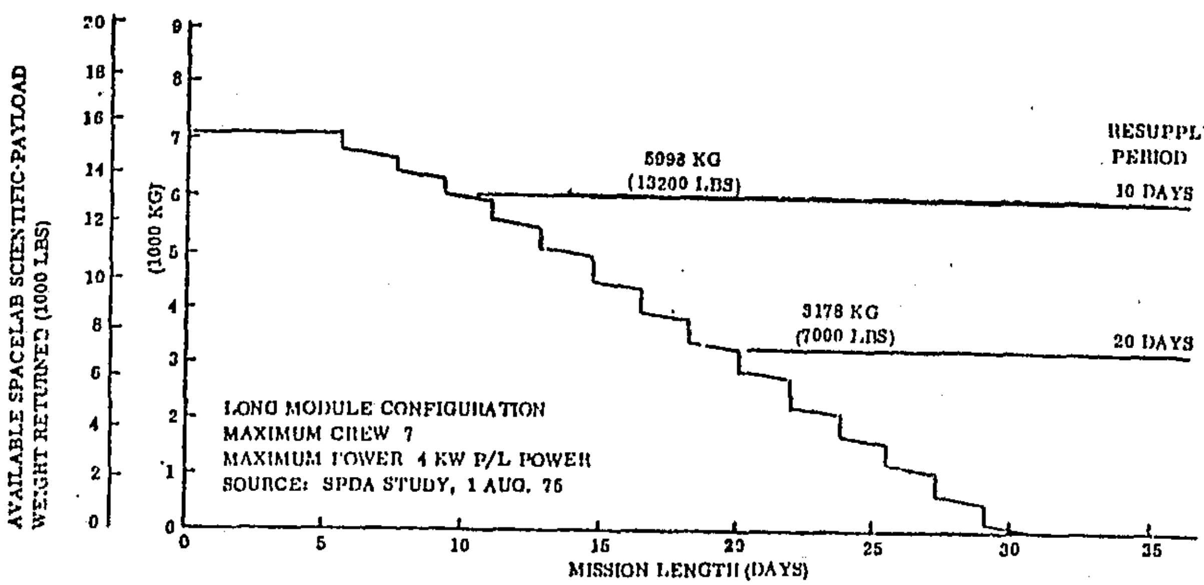

Figure 16. Extended Duration Spacelab Mission

Recovery of Disabled Orbiter - 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, could 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 savings from in-orbit fluid supply could be on the order of $472M as shown in Table 7.

Increased Mission Capabilities - Mission requirements were reviewed 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 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 as a 1100 kg (24,000 lb) mission requirement, but was reduced to 3300 kg (7300 lb) by October through using direct Mars entry and elimination of a rover vehicle. This mission is now listed as requiring 5000 kg (11,000 lb). Referring to Table 5, 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.

Table 7. 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

Typical of Sortie missions which have been adjusted to meet existing payload limitation is the 30 m IR Interferometer (AS-09) payload with a length of 16.5 m (54 ft) and listed in 1974 with a desired 740 km (400 n.mi.) circular orbit altitude. But in 1975 the desired altitude was reduced to 400 km (215 n.mi.). The initial requirements in 1974 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 (Figure 12).

It is anticipated that once the performance envelope of the Shuttle and Tug are expanded by in-orbit fluid supply capability, some planned missions will grow and new missions will be conceived which will require the new performance capability.