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SPACE PLATFORM EXPENDABLES RESUPPLY

CONCEPT DEFINITION STUDY

STS 85-0174

VOLUME II

TECHNICAL REPORT

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This final report of the Space Platform Expendables Resupply Concept Definition study was prepared by the Advanced Engineering organization of the Space Transportation Systems Division of Rockwell International Corporation for the Marshall Space Flight Center (MSFC) of NASA in accordance with Contract NAS8-35618. The study was conducted under the direction of the MSFC Contracting Officer Representative (COR), Mr. Wilbur Thompson, during the period from March, 1984 through December, 1984. The final report is organized into four documents:

- Volume I - Executive Summary
- Volume II - Study Results
- Volume III - WBS and Dictionary
- Volume IV - Cost Estimate

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The study was also supported by a subcontract to Science Applications, Inc. personnel, under the leadership of Dr. Brian O'Leary. Useful information was also received from SPAR Incorporated, Seton-Wilson Corporation, and MBB-ERNO.

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Acronym Dictionary

Ah	Ampere Hour
AXAF	Advanced X-Ray Astrophysics Facility
BBQ	Barbecue (Rotation of Spacecraft for Thermal Management
BOL	Beginning of Life
CREP	Cosmic Ray Experiment Program
DDT&E	Design, Develop, Test and Engineering
DMSP	Defense Meteorlogical Satellite Program
DRM	Design Reference Mission
V	Delta Velocity
DOD	Department of Defense
DOMSAT	Domestic Satellite
DPP	Deployable Photovoltaic Panel
DPS	Data Processing Subsystem
EOS	Earth Observing System
EPHP	Externally Pumped Heat Pipe
ERM	Expendables Resupply Module
ETR	Eastern Test Range
EUVE	Extreme Ultraviolet Explorer
EVA	Extra Vehicular Activity
GEO	Geosynchronous Orbit
GN&C	Guidance Navigation and Control
GP-B	Gravity Probe-B
GPM	Gallons Per Minute
GRO	Gamma Ray Observatory
GSE	Ground Support Equipment
GSTDN	Ground Space Flight Tracking and Data Network
He	Helium
IP	Integral Propulsion
IRR	Internal Rate of Return
Isp	Specific Impulse
JSC	Johnson Space Center
L	Life of Satellite In Years
LaRC	Langley Research Center
LDR	Large Deployable Reflector
LEO	Low Earth Orbit
Li	Lithium
MIL	Man-in-Loop
MMD	Mean Mission Duration
MMH	Monomethylhydrazine
MPS	Materials Processing In Space
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NTO	Nitrogentetroxide (N_2O_4)
N_2	Nitrogen
N_2H_4	Hydrazine
OMPS	Orbiter Main Propulsion System
OMS	Orbital Maneuvering System (Shuttle Orbiter)
OMV	Orbital Maneuvering Vehicle
ORCS	Orbiter Reaction Control Subsystem
OTV	Orbit Transfer Vehicle
PVT	Pressure, Volume, and Temperature
RCS	Reaction Control Subsystem
RF	Radio Frequency

Acronym Dictionary (Cont'd.)

ROM	Rough Order of Mangitude
S	Sulfur
SAI	Science Applications Inc.
S/C	Spacecraft
SDI	Strategic Defense Initiative
SIRTF	Shuttle Infrared Telescope Facility
SPER	Space Platform Expendables Resupply
STS	Space Transportation System
TFU	Theoretical First Unit
TDRSS	Tracking and Data Relay Satellite System
Ti	Titanium
USAF SD	United States Air Force Space Division
WTR	Western Test Range

Vol 2

1.0 INTRODUCTION

@A 85- NASA has recognized that the capability for remote resupply of space platform expendable fluids will help transition space utilization into a new era of operational efficiency and cost/effectiveness. The emerging Orbital Maneuvering System (OMV) in conjunction with an expendables resupply module will introduce the capability for fluid resupply enabling satellite lifetime extension at locations beyond the range of the Orbiter. This report summarizes a Phase A study of a remote resupply module for the OMV. Volume II represents study results @ABA Author

~~1.1~~ Background

Numerous studies have been performed in recent years concerning the transfer of fluids to satellites within the Orbiter payload bay. These studies have mainly focused on the transfer of hydrazine (N_2H_4). On a recent Shuttle flight, transfer of hydrazine using man-in-the-loop was demonstrated within the Orbiter payload bay. Also the Air Force is considering incorporation of interface requirements allowing on-orbit fluid transfer in future satellites (with notable applications to SDI). It is apparent that the era of fluid transfer on-orbit is emerging and that remote expendables resupply will become a recognized requirement for future satellites and/or propulsion modules.

1.2 Objectives and Scope

The overall objectives of the study are summarized as follows:

1.	Develop Performance & Operational Requirements associated with critical liquids, gases, and lubricants resupply	Task 1 = 60% of study
2.	Develop Design Requirements for the resupply module (RM) & equipment needed in conjunction with OMV to perform remote fluid resupply functions	
3.	Develop resupply module Concept Designs (including tankage, transfer systems & supporting mechanisms) & associated spacecraft adaptation (user interface emphasis)	Task 2 = 30% of study
4.	Define a Flight Demonstration Program for automated remote fluid resupply	
5.	Develop Program Planning Data, including cost, schedule & preliminary supporting development program	Task 3 = 10% of study

The study scope was initially limited to the definition of remote fluid resupply concepts for satellites/space platforms. This scope includes fluid transfer to propulsion modules for LEO and higher energy orbit satellites/platforms (expendable or reusable), but not to the space station. This assessment of resupply benefits, with and without other types of servicing was intended to confirm a comprehensive justification for future remote resupply operations.

1.0 INTRODUCTION

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1.3 Guidelines and Assumptions

The guidelines and assumptions established at the start of and during the study are summarized below:

- o Utilize available data from related study/development activities, including propellant transfer experiments, upper stage design, operations studies, and space station/platform studies.
- o Incorporate results of In-Bay Fluid Transfer Experiment from STS 41-G.
- o Insure coordination with space station/OTV fluid resupply studies.
- o Propellant supply facilities will be assumed to exist as part of the Shuttle, space station, or separate depot facilities.
- o The study scope was initially limited to investigation of fluid resupply only, but the requirements were later expanded to include satellite servicing concurrent with refueling.
- o Assuming that fluid resupply may be required prior to full-capability spacecraft servicing, the impact of remote spacecraft servicing on nominal fluid resupply modes will be considered.
- o The proposed system must meet all manned safety criteria during initial transport to low earth orbit and if returned to the orbiter or manned station after a resupply mission.

1.4 Study Flow and Methodology

The study tasks and their relationship to each others are shown on Figure 2.

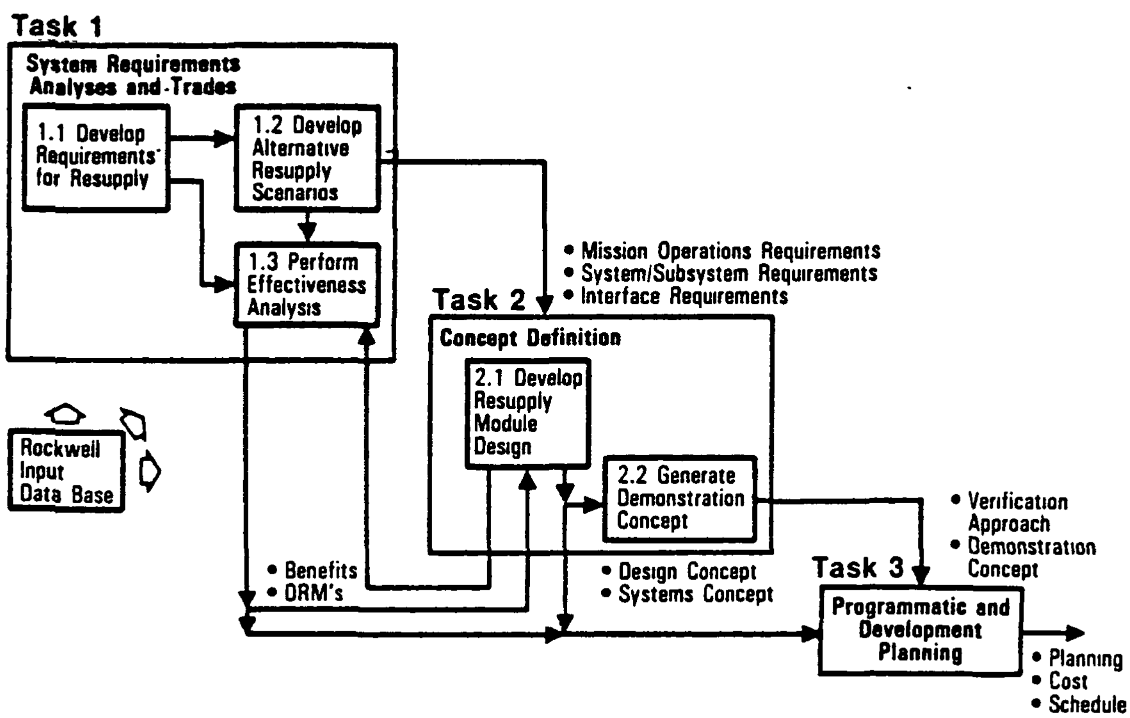


Figure 2. Study Flow

Task 1, System Requirements, Analysis and Trades, was the focus of the mid-term review. In subtask 1.1, resupply requirements were generated in the form of a mission model. Next, fluid parametric requirements were generated for this mission model to determine fluid types, quantities and usage rates for resupply. This data was then examined to allow selection of candidate spacecraft and platforms for resupply. In Subtask 1.2, alternative scenarios were developed and screened for effectiveness. Mission operational and system/subsystem requirements were next generated for these selected scenarios, to provide data in support of the effectiveness analysis (and for the Task 2 design analysis after selection of the Design Reference Missions).

The study effort after the mid-term review concentrated on the development of resupply module concept designs to meet the approved DRM's resulting from this review (Task 2). Continued effectiveness analysis (subtask 1.3) resulted in the definition of specific requirements for these DRM's in LEO and GEO as well as expanded benefits assessments. In addition the definition of a flight demonstration program and a supporting technology program was accomplished.

Finally Task 3 involved programmatic planning and cost/schedule data development for the technology and flight demonstrations as well as the initial operational segments of the total program.

2.0 Summay of Results

2.1 Major Conclusions

The remote resupply alone of LEO satellites is found to be of potential economic benefit but resupply combined with servicing is much more advantageous. The latter case can be most effectively accomplished at the Orbiter. Hence, remote expendables resupply alone was not recommended for most of these satellites. Certain DoD satellites in high inclination LEO orbits will require relatively frequent expendables resupply to avoid costly replacement. Since it will likely be desirable to accomplish these resupply operations without dependence on the Orbiter, the need for a LEO propellant storage depot with a space-based OMV/RM is emphasized.

The major commercial industry drivers for GEO resupply relate to communications satellite revenues. They are a function of the total number of operating transponders and the number of transponders per satellite, transportation costs, satellite production costs, insurance costs, and technological obsolescence.

Lifetime extension of communications satellites through Attitude Control Subsystem (ACS) resupply is of marginal benefit without concurrent servicing of the satellite electronics. This is due to the relatively rapid rate (12% annual) of technological obsolescence. Planned lifetime extension where the satellite's reliability is increased (and also the initial satellite cost) turns out to be the least attractive scenario because of the increased up-front costs. The contingency resupply of satellites which have lost their fuel from malfunctions is always cost effective, if the problem can be isolated and resolved early in the satellite's life when interruption of the revenue stream can have the greatest impact.

Though adding additional transponders at the expense of ACS fuel weight increases the toal revenue from a satellite, it is not cost effective primarily because of the risk involved in assuring future operation. Less onboard fuel will require more frequent (Optimally 2 in a 10 year period) resupply missions, thus increasing the risk to the satellite revenue stream if the resupply fails. Insurance premiums of less than 15% are required to make this and other resupply cases viable.

Servicing large satellites is generally more cost effective than smaller satellites because large satellites house more transponders which may be serviced on only one mission. Multiple manifesting of missions achieves the same benefit when the transponders are spread over several smaller satellites.

If a Comsat undergoes a low rate of technological obsolesence there is less incentive to resupply and upgrade it. If this rate can be held to 6% or lower it is unlikely that resupply/servicing will be beneficial unless a revenue restoration level greater than 100% can be realized. At a 12% obsolesence rate a revenue restoration of 70% is required to assure a non-negative impact to the satellite revenue stream.

The resupply system can be configured for reasonably low impact on the receiver spacecraft. The impact is primarily required interface provisions. The ullage exchange propellant transfer process has the inherent versatility to adapt to different receiver propulsion systems; regulated, pressure fed, blowdown, or pump fed without severe weight impacts to the receiver spacecraft.

The DDT&E costs for the resupply module program could be reduced by exploiting the synergism with the NASA-JSC N_2H_4 in-bay tanker program. Although the JSC program is being configured for different functional requirements and missions, it will potentially result in certain common technology development or modifiable hardware. Such hardware could include interface components, Mission Peculiar Equipment Support Structure (MPESS) (as tankage support for in-bay tanker and resupply module flight demonstration), and propellant pumps.

2.2 Selected Options

Ullage exchange was selected as the best option for the N_2O_4/MMH fluid transfer process. This approach is applicable to all potential receiver propulsion subsystem and acquisition types through appropriate modifications. It minimizes pressurant resupply requirements, involves no adiabatic compression (explosion hazard), requires no waste or hazardous effluent scavenging, and provides constant pressure resupply.

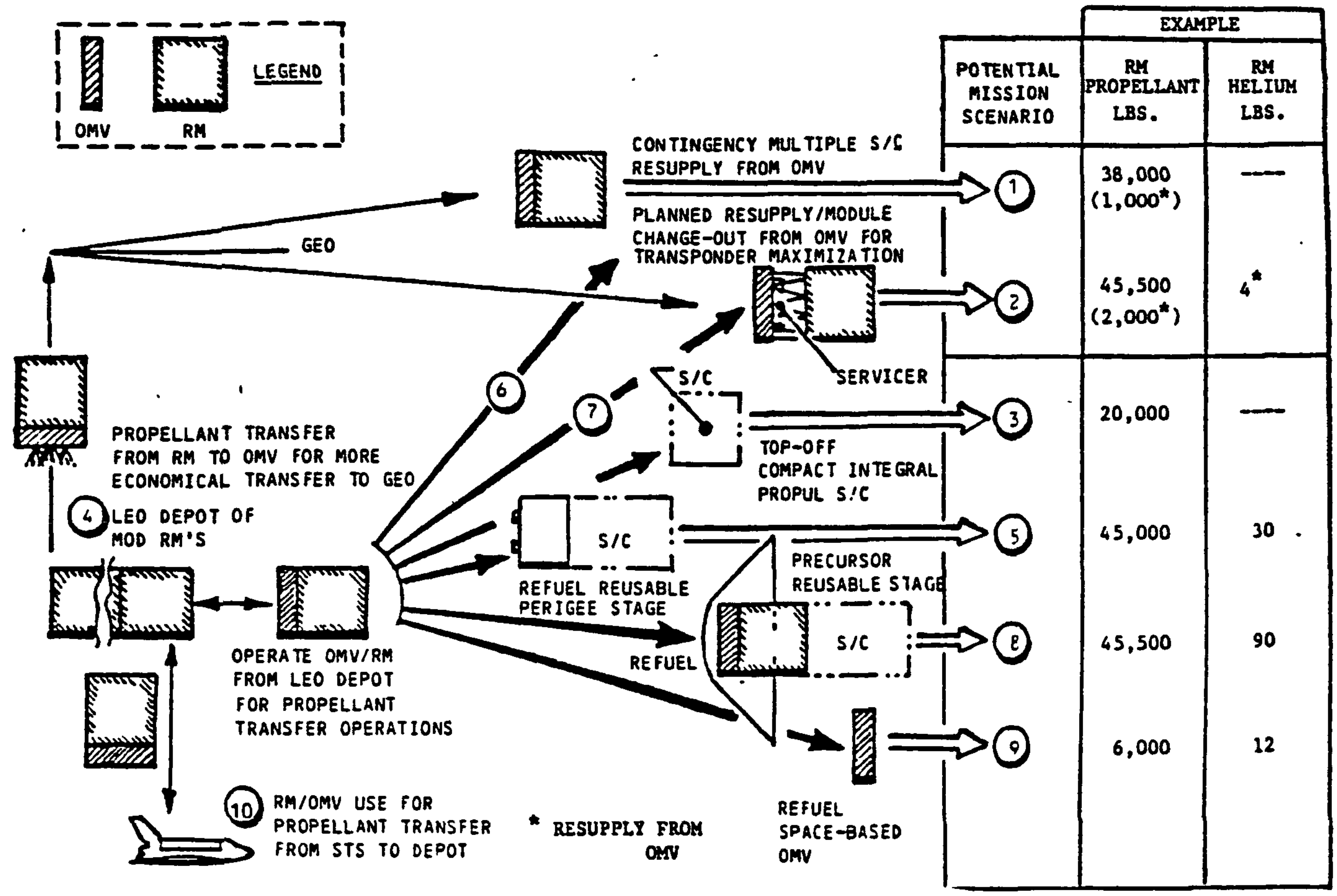
Bi-propellant N_2O_4/MMH fluids were chosen for the resupply module since they are compatible with some GEO propulsion systems, with the OMV propulsion subsystem, and they are scavengable from orbiter OMS tanks.

Figure 3 presents ten scenarios of bi-propellant use (some without helium) which were selected as design reference missions. Scenarios 1 and 2 are respectively contingency and planned resupply in GEO. Scenarios 6 and 7 are similar operations to 1 and 2 but in LEO. Scenario three is top-off of a satellite integral propulsion subsystem. Five is refueling of a perigee stage from the OMV/RM; eight is similar to five but also includes an apogee burn, Resupply Module staging, and OMV aerobraking maneuver on return. In scenario nine only the OMV is being resupplied. Four is a coupling of resupply modules in LEO which function as a depot. Finally, in ten propellant is being transferred from the orbiter to the depot through the OMV.

A propellant depot becomes most attractive if one uses the basic resupply module as the "core" element of the depot. Several of these modules could be used to achieve the desired storable propellant capacity. This could reduce development costs for the depot.

Propellant top-off capability in LEO via remote resupply would allow launch in the STS of multiple satellites with one or more in an off-loaded condition. Near term use of propellant top-off capability may also be applicable to currently projected larger DOD satellites which if launched fully loaded, would exceed STS lift capacity.

An additional application of the RM might be to fuel a high performance reusable perigee stage, as is being studied at McDonnel Douglas. This stage would be capable of launching a 12K lb payload into GEO after fuelling by a resupply module.



POTENTIAL MISSION SCENARIO	EXAMPLE RM PROPELLANT LBS.	EXAMPLE RM HELIUM LBS.
1	38,000 (1,000*)	—
2	45,500 (2,000*)	4*
3	20,000	—
5	45,000	30
8	45,500	90
9	6,000	12

Figure 3. Selected Mission Scenarios

2.3 Recommended Approach

The selected resupply module configuration is shown in Figures 4 and 5 and is presented in two configurations; without and with the satellite servicer. This concept was selected from among several competing approaches primarily on the basis of overall structural efficiency and for it's use of existing hardware providng low development cost. The resupply module is supported at its forward end by an existing IUS upper stage forward cradle. This cradle includes load equalization capability which reduces the structural redundancy between the resupply module and the Orbiter. It also allows a minimum of weight impact on the resupply module for attachment to the Orbiter payload bay longerons and keel at its forward end. The resupply module uses six stretched OMS tanks with a modified ullage positioning Propellant Management Device (PMD) for ullage bubble position control.

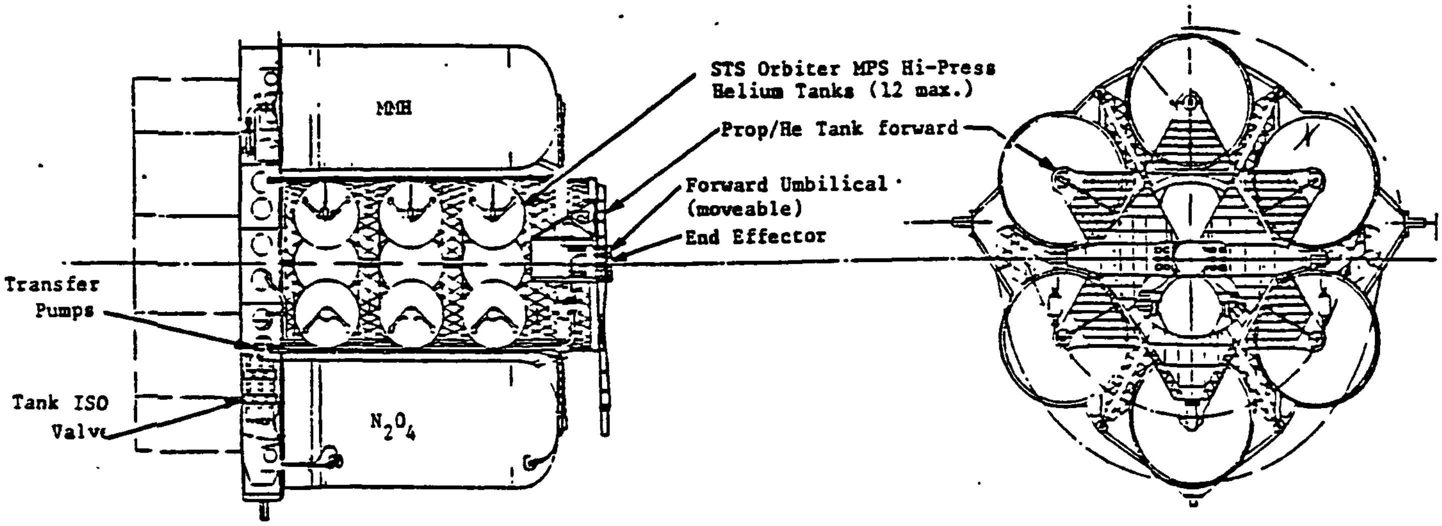


Figure 4. Resupply Module General Configuration (Without Servicer)

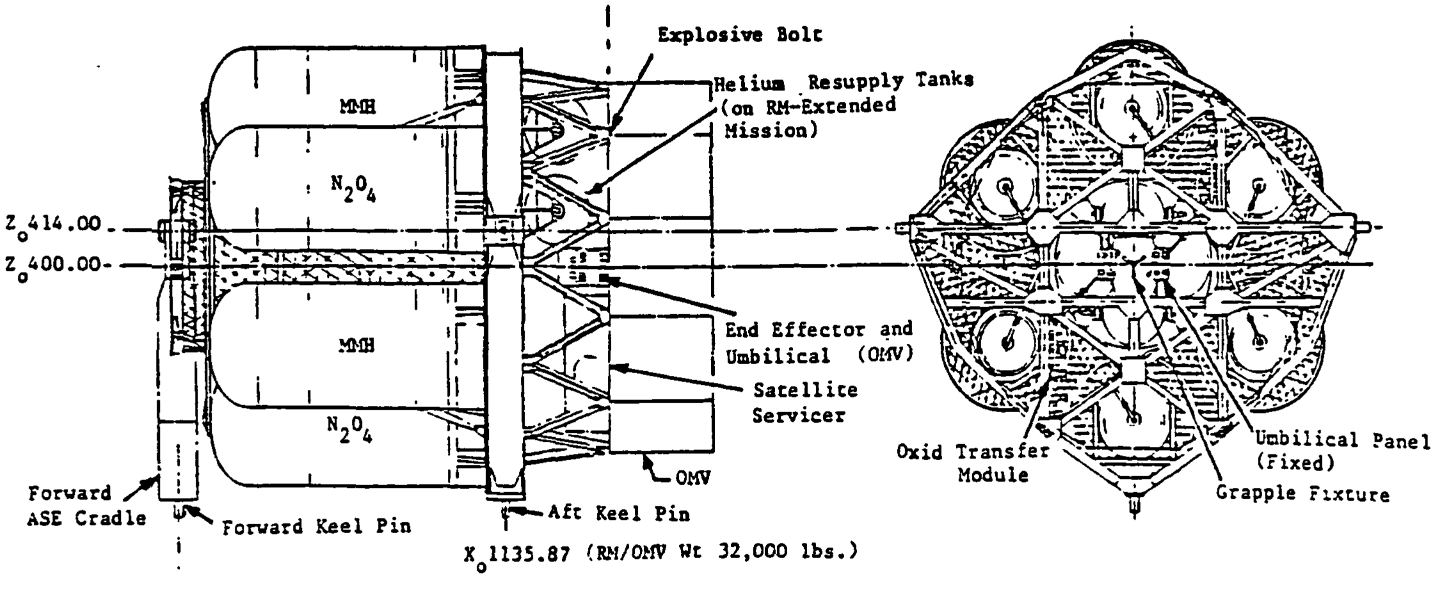


Figure 5. Resupply Module General Configuration (With Servicer)

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The selected flight demonstration approach uses hardware developed through a low-cost Technology Validation Program and many off-the-shelf components in conjunction with a low-cost structural framework. Two modules, both of nearly identical configuration, would be docked together in the payload bay using the RMS. Subsequently, storable bipropellants plus helium transfer would be demonstrated without man-in-the-loop. One module would represent the receiver spacecraft and the other the resupply module. Most of the components, except the main tankage (because of its small size) would later be used for operational resupply units to allow significant program cost savings. The selected flight demonstration concept is depicted in Figure 6.

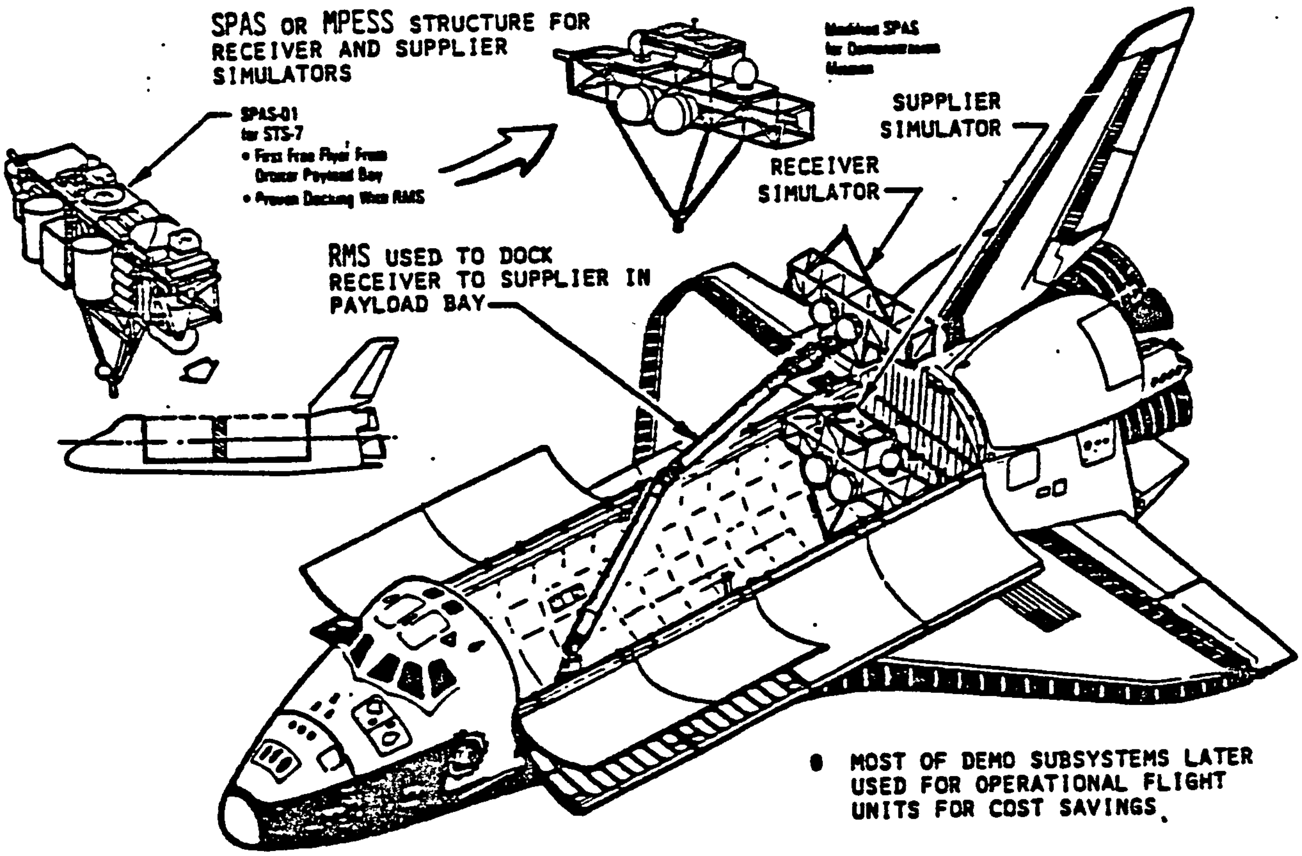

Figure 6. Bi-Propellant/Helium Transfer Flight Demonstration Concept

3.0 STUDY RESULTS

This section presents the analyses associated with the Space Platform Expendables Resupply Concept Definition Study.

3.1 System Requirements Analysis and Trades (Task 1)

The overall activities of the system requirements analysis and trades of the study are shown in Figure 7, followed by a description of the main study accomplishments for Task 1.

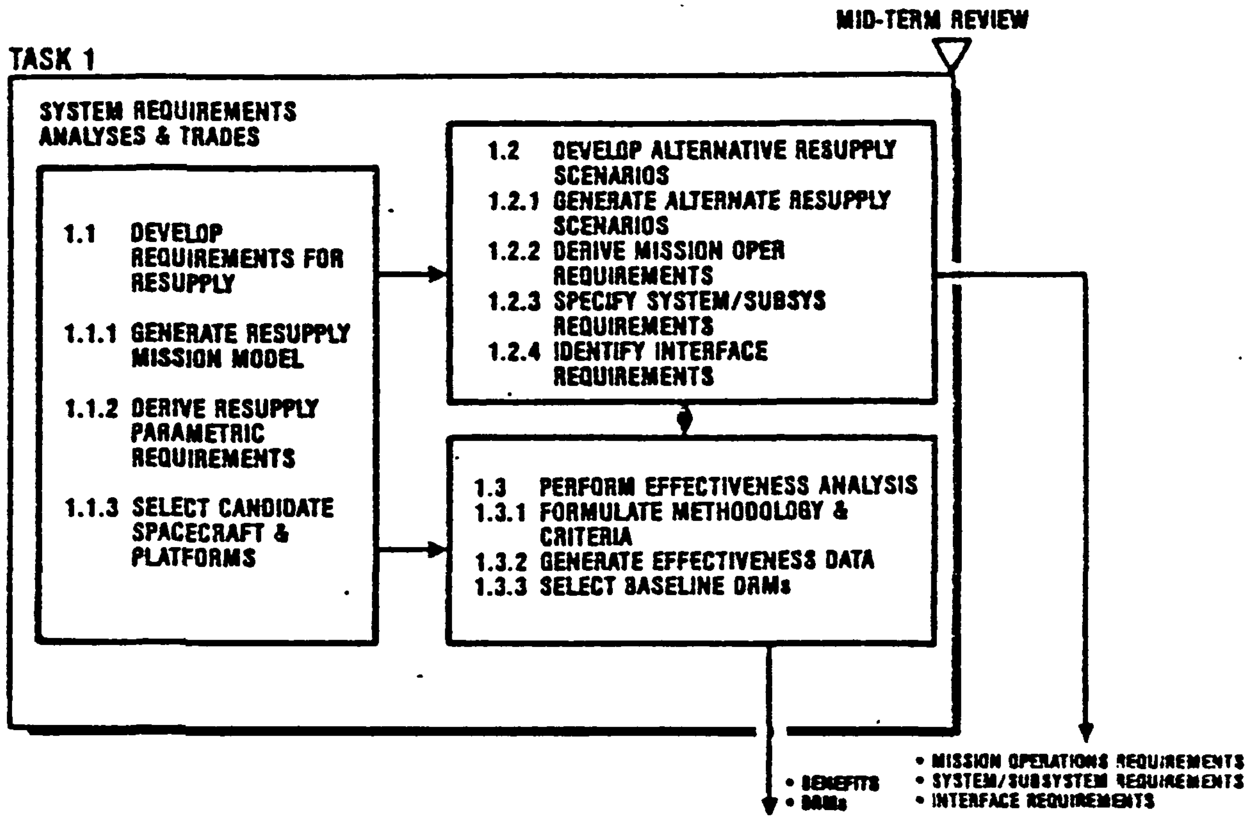

Figure 7. Task 1 Activities

3.1.1 Mission Model and Fluid Parametric Requirements

A detailed mission model of fluids resupply engagement was developed in Activity 1.1.1. The model was developed from earlier projections of OMV servicing tasks and estimates of future STS launch rates and is shown in Table 1. All spacecraft that could take advantage of fluid resupply were included in this engagement model.

Those missions which would actually be recommended for resupply are discussed later in this report.

Estimates of the number of resupply engagements for each spacecraft/platform were based on discussions with individual program offices. Such discussions covered estimates of spacecraft technological obsolescence and value of extended time on-orbit. Classified data on the make up of the DoD segment is not available under this contract. The potential line of Table 1 includes engagements which were considered speculative and beyond the nominal categorization of the other fluid transfer engagements. They included additional DoD users and engagements in GEO.

This mission model assumed the resupply capability would be incorporated in follow-ons to existing large communications satellite programs. Estimates of the satellites propellant requirements were made since the commercial users were considered only a potential category. The propellant requirements are for on-orbit station keeping and pointing. But these vary with satellite size and pointing accuracy specifications.

MISSION	LOCATION NMI	LOCATION DEG	1989	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	2001	COMMENT
GRO	227	28.5	D		1		1		1							3 AT ORBITER
AXAF	320	28.5			D			1			1			1		3
EUVE	324	28.5	D	1												1 AT ORBITER
CREP	216?	28.5			D		1									1
PROTEUS (1)	216	28.5		D	D	1	1	1	1	1	1	1	1	1	1	10, PLATFORMS
LDR	486	28.5							D			1			1	2
SIRTF	486	98					D		1		1		1		1	4
GRAV PROBE-B	520	90						D	1							1
EOS	380	99.8					D			2			2			4, PLATFORMS
MPS (2)	320	28.5		1	3	5	6	6	5	3	1					30, PLATFORMS
GEO PLATFORM	19323	0							1							EXPT SERVICING
CONTINGENCY	TBD	TBD			1m			1			1			1		20% OF OMV CONTINGENCIES NEED FLUIDS
CIVILIAN TOTAL				2	5	6	9	9	10	6	5	2	4	3	3	64
DOD				2	3	2	4	5	7	1	7	2	4	7	3	47
NOMINAL TOTAL				4	8	8	13	14	17	7	12	4	8	10	6	111
CUMULATIVE				4	12	20	33	47	64	71	83	87	95	105	111	
POTENTIAL LINE									3	10	7	8	12	8	12	60
POTENTIAL TOTAL				4	8	8	13	14	20	17	19	12	20	18	18	
CUMULATIVE				4	12	20	33	47	67	84	103	115	135	153	171	171

(1) — FIRST MISSION MAN-TENDED AT ORBITER
(2) — DONE AT ORBITER MAN-TENDED
D — DEPLOYMENT

- 111 FLUID RESUPPLY ENGAGEMENTS FOR DOD, NASA AND COMMERCIAL USERS FROM 1989-2001 (NOT INCLUDING SPACE BASED OTV, OMV AND STATION)

Table 1. Fluid Resupply Mission Model

Based on the SAI subcontract study, DoD resupply requirements are at present predicted to be primarily propellants. Evasion and on-orbit maneuvering constitute a large percentage of the propellant requirements. The SAI study also indicated that the DoD favors the bi-propellant combinations of N_2O_4 and A-50 for future system designs, although MMH may eventually be substituted. A storable bi-propellant combination was used in the parametric requirements for those satellites not currently baselined for hydrazine propulsion.

Because of the confidential nature of DoD programs, parametric requirements are for this portion of the model limited in scope and detail. General propellant requirements are presented in the form of total impulse requirements for the various missions in Table 2.

On-orbit propellant consumption for a few of the nominal and potential missions was calculated by using Isp's of 230 and 290 seconds for hydrazine and N_2O_4/A-50 bi-propellant systems, respectively. These Isp's were assumed for existing engine capability when utilzed for both delta V and attitude control. Requirements for the other missions were also based primarily on the SAI results. The DoD resupply requirements are summarized in Table 3.

ALTITUDE	TOTAL IMPULSE REQUIRED ($\times 10^6$ SECONDS) INSERTION	ON-ORBIT *	COMMENTS
LOW EARTH ORBIT	2.0	~ 2.0	INSERTION AND ON-ORBIT REQ'TS ARE SIMILAR
HIGH EARTH ORBIT	10.0 - 20.0	0.3 - 1.0	10.0×10^6 SEC REQUIRED FOR 1000 NM INSERTION
GEOSYNCHRONOUS	20.0	0.3 - 1.0	10,000 LB TO GEO, MANEUVERING TO SERVICE 6 SATELLITES

*Includes Maneuver and Station-Keeping Capability

Table 2. Military Satellite Propulsion Requirements

CANDIDATE SATELLITE	RESUPPLY QUANTITY (LB)	RESUPPLY INTERVAL (YR)
NOMINAL		
DMSP (1)	70 (N_2H_4), 5 (NITROGEN)	2
DOD - 1 (2)	7000 (N_2O_4/A-50)	1
DOD - 2 (2) (3)	8700 (N_2H_4), 1200(HELIUM)	3
DOD - 3 (2)	7000 (N_2O_4/A-50)	3
POTENTIAL		
DOD - 6 (2)	7000 (N_2O_4/A-50)	3
DOD - 7	250 (N_2H_4)	5
DOD - 8	350 (N_2H_4)	7
DOD - 9	1000 (N_2O_4/A-50)	5

(1) Propellant quantity shown is for remote resupply. Prior to 1994 retrieval to the orbiter is necessary for resupply.
(2) Propellant quantity calculated based on military propulsion requirements (Table 2)
(3) Cryogenic helium quantity based on SIRTF (80% of SIRTF capacity, extrapolated for 3 year life)

Table 3. DoD Resupply Requirements

Resupply parametric requirements were developed for non-DoD missions. The candidate missions selected were those likely to require or benefit from consumables resupply during the fiscal years 1989-2001.

The NASA space systems technology model provided the starting point for developing the parametric requirements. The appropriate program and/or project managers were then contacted for more specific information. In many instances, recently published documents were available on the individul projects. One advantage in establishing personal contacts was the information gained with respect to realistic mission launch dates and their attitude toward utilizing resupply capabilities.

Results from the user contact efforts are summarized on the following pages for the missions listed below:

Advanced X-Ray Astrophysics Facility (AXAF) - Objectives of the 28.5° free-flyer are to determine the positions of X-ray sources, their physical properties as composition and structure, and the process involved in X-ray photon production.

Cosmic Ray Experiment Program (CREP) - Utilizing various NASA capabilities (such as balloons, Spacelab, LDEF, special spacecraft), the CREP will improve upon previous terrestrial and interplanetary measurements by an order of magnitude.

Earth Observing System (EOS) - The objective of the sun-synchronous polar platform is integrated scientific observation of the Earth and its environment including the biological, chemical, and physical characteristics of the land, oceans, and atmosphere.

Extreme Ultraviolet Explorer (EUVE) - The EUVE explorer will make the first all-sky map to discover, obtain accurate positions, and determine the spectral energy distribution for all detectable EUV sources.

Geostationary Platform (Experimental) - A permanent platform for co-locating compatible geostationary payloads such as communication and maritime systems.

Gravity Probe-B (GP-B) - High precision measurement of the relativistic spin-orbit and spin-spin coupling will provide a fundamental step in discriminating between viable competing theories of gravitation.

Gamma Ray Observatory (GRO) - Scientific objectives include: 1) Study of discrete objectives; 2) Search for evidence of nucleo-synthesis; 3) Survey of the Galaxy in gamma rays; 4) Search for cosmological effects and possible primordial black hole emissions; and 5) observation of gamma ray bursts.

Large Deployable Reflector (LDR) - A dedicated astronomical observatory operating in the spectral region between 30 micrometers and 1 milimeter.

Materials Processing in Space (MPS) - Free -flying MPS factory modules designed to satisfy microgravity processing requirements of the commerical sector.

Proteus - Long duration LEO system derived from Leasecraft concepts and designed to support consecutive short duration Explorer - class payloads.

Shuttle Infrared Telescope Facility (SIRTF) - The objective is to conduct definitive high sensitivity infrared photo-metric and spectroscopic of a wide range of astrophysical phenomena.

Data obtained for these programs is summarized on the following pages.

CONSUMABLES REQUIREMENTS

PROGRAM: AXAF

SIZE: DIA (ft) 14 LENGTH (ft) 43 MASS: 19,000 LB

S/C REPLACEMENT COST: $200 M LIFETIME: > 10 YR

CONSUMABLES: TYPE GAS MIXTURE (90% Xe, 10% METHANE) USAGE 85 LB OVER 3 YR

S/C STORAGE CONDITIONS: TITANIUM RESERVOIRS AT VARIOUS PRESSURES

RESUPPLY TRANSFER CONDITIONS: COULD BE ACCOMPLISHED BY A PRESSURE OR PUMP FED UMBILICAL SYSTEM, CURRENTLY 952 psia IS THE HIGHEST INSTRUMENT PRESSURE REQUIREMENT.

RESUPPLY TIMES: APPROXIMATELY 3 HR AFTER 3 YR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 320 (205 AFTER 3 YR) i(deg) - 28.5

Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: CURRENTLY NEEDS AN OMV, MAY USE STS, INTEGRAL PROPULSION WILL BE INVESTIGATED

COMMENTS: PROJECT IN PHASE A, PHASE B WILL BE FALL OF 1984, ATP EXPECTED IN 1987 WITH IOC IN 1991; DESIGN OPTION SUPPORTED BY SAO IS TO PLACE CONSUMMABLE REQ'T ON INDIVIDUAL DETECTORS (LEADS TO INSTRUMENT C/O)

ATTITUDE TOWARD RESUPPLY: CURRENTLY BEING DESIGNED FOR MAINTENANCE AND RESUPPLY, MAY INCLUDE EXPERIMENT CHANGE

CONTACT: CARROLL DAILY (MSFC) 205-453-2788
BILL McKINNEN (SAO) FTS-830-7358
ED McLAUGHLIN (SAO) FTS-830-7362

NOTE: RESUPPLY OF AXAF FROZEN COOLANTS IS CONSIDERED TOO HAZARDOUS, REQUIRES INSTRUMENT CHANGEOUT

CONSUMABLES REQUIREMENTS

PROGRAM: CREP

SIZE: DIA (ft) UNDETERMINED LENGTH (ft) UNDETERMINED MASS: UNDETERMINED

S/C REPLACEMENT COST: UNDETERMINED LIFETIME: 4 YR

CONSUMABLES: TYPE N_2H_4 (S/C BUS PROPELLANT) USAGE ≈ 550LB OVER 2 YR (N_2H_4)
He (SUPERCONDUCTING MAGNETS) ≈ 160LB OVER 2 YR (LHe)
Xe-METHANE (GAS COUNTERS) ≈ 30 lb over 2 YR

S/C STORAGE CONDITIONS: UNDETERMINED

RESUPPLY TRANSFER CONDITIONS: PROBABLY PRESSURE OR PUMP FED UMBILICAL FOR HYDRAZINE AND GAS MIXTURE, REQUIRES ADVANCED TECHNOLOGY FOR SUPERFLUID HELIUM

RESUPPLY TIMES: ROUGHLY 12 HR TO RESUPPLY AFTER 2 YR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 216 i(deg) 2.85
Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: WILL PROBABLY REQUIRE A BUS/PLATFORM SUCH AS PROTEUS FOR THE FINAL PHASES OF THE PROGRAM

COMMENTS: PROJECT IN PRELIMINARY DESIGN PHASE, MULTI-PHASE PROGRAM USING VARIOUS NASA CAPABILITIES

ATTITUDE TOWARD RESUPPLY: WILL PROBABLY DESIGN FINAL PHASE OF PROGRAM TO UTILIZE RESUPPLY CAPABILITIES

CONTACT: Dr. Jon Ormes (NASA HQ) 202-453-1462
(GSFC) 301-344-4793

CONSUMABLES REQUIREMENTS

PROGRAM: EOS

SIZE: DIA (ft) UNDETERMINED LENGTH (ft) UNDETERMINED MASS: UNDETERMINED

S/C REPLACEMENT COST: $50M LIFETIME: > 10YR

CONSUMABLES: TYPE N_2H_4 (PLATFORM PROPULSION) USAGE 2000LB OVER 3YR (FUNCTION OF SYSTEM DESIGN AND NUMBER OF PAYLOADS)

S/C STORAGE CONDITIONS: PRESSURE TANKS OR DIAPHRAGM TANKS

RESUPPLY TRANSFER CONDITIONS: MAY USE PRESSURE FED UMBILICALS

RESUPPLY TIMES: APPROXIMATELY 4 HR TO RESUPPLY AFTER 3 YR OPERATION

ORBITAL PARAMETERS: h (nmi) - 380 i(deg) - 99.8
Ω (deg) - UNDETEMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: INTEGRAL PROPULSION PROBABLE, OMV IS AN OPTION

COMMENTS: PROJECT IN CONCEPTUAL PHASE, CURRENTLY BASELINED WITH SPACE STATION POLAR PLATFORM, DEMONSTRATION SERVICING LIKELY EARLY IN OPERATIONAL PHASE

ATTITUDE TOWARD RESUPPLY: WILL BE DESIGNED FOR RESUPPLY OF EOS PROPELLANTS

CONTACT: CHUCK MacKENZIE (GSFC) 301-344-6008

CONSUMABLES REQUIREMENTS

PROGRAM: EUVE

SIZE: DIA (ft) UNDETERMINED LENGTH (ft) UNDETERMNED MASS: 882LB

S/C REPLACEMENT COST: $24M LIFETIME: 2YR

CONSUMABLES:: TYPE POSSIBLY N_2H_4 USAGE 400LB OVER 1YR

S/C STORAGE CONDITIONS: PROBABLY PRESSURE TANKS OR DIAPHRAGM TANKS

RESUPPLY TIMES: APPROXIMATELY ONE HOUR AFTER ONE YEAR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 324 i(deg) - 28.5
Ω(deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: MAY USE INTEGRAL PROPULSION FOR STS RENDEZVOUS

COMMENTS: PHASE A RFP DUE IN JULY 1984, OPERATES AT "NIGHT" ONLY, MIRROR CONTAMINATION CRITICAL DURING RESUPPLY OPERATION

ATTITUDE TOWARD RESUPPLY: NASA JPL WOULD LIKE INCREASED LIFETIME BY RESUPPLY/ SERVICING, CURRENT PROGRAM BASELINED FOR 1YR OF OPERATION

CONTACT: R. A. PIOZAJ (NASA JPL) 818-354-7447

PROGRAM: GEO PLATFORM (EXPERIMENTAL)

SIZE: DIA (ft) 110 (ON-ORBIT) LENGTH (ft) 164 (ON-ORBIT) MASS: 12,000LB

S/C REPLACEMENT COST: $175M LIFETIME: 7YR

CONSUMABLES:: TYPE N_2H_4 USAGE 1764 LB OVER 7 YR

S/C STORAGE CONDITIONS: MAY USE PRESSURE TANK OR DIAPHRAGM TANK

RESUPPLY TRANSFER CONDITONS: COULD BE ACCOMPLISHED BY PRESSURE OR PUMP FED UMBILICAL SYSTEM

RESUPPLY TIMES: APPROXIMATELY 4 HR AFTER 7 YR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 19323 @ 100° WEST i(deg) - 0

Ω (deg) - N/A ω (deg) - N/A

S/C PROPULSIVE CAPABILITY: OTV NECESSARY, ACS USING A SINGLE BLOWDOWN RCS

COMMENTS: PRELIMINARY ARCHITECTURE STUDIES COMPLETE, PAYLOAD STUDIES ON-GOING, SERVICING DEMONSTRATION PROBABLY BEFORE 7 YR OF OPERATION

ATTITUDE TOWARD RESUPPLY: RESUPPLY/SERVICING SUPPORT (i.e. BY SPACE STATION) WILL BE MANDATORY

CONTACT: GEORGE KNOUSE (NASA HQ) 202-453-1515

CONSUMABLES REQUIREMENTS

PROGRAM: GP-B

SIZE: DIA (ft) 15 (ON-ORBIT) LENGTH (ft) 8 (ON-ORBIT) MASS: 4000 LB

S/C REPLACEMENT COST: $72M LIFETIME: 2YR MAXIMUM

CONSUMABLES:: TYPE SUPERFLUID He USAGE 397LB OVER 1 YR

S/C STORAGE CONDITIONS: DEWAR CONFIGURATION @ 1.8K

RESUPPLY TRANSFER CONDITONS: WOULD REQUIRE SIRTF-TYPE ADVANCED TRANSFER TECHNOLOGY

RESUPPLY TIMES: APPROXIMATELY 6 HR AFTER 1 YR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 520 i(deg) - 90
Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: ACS ONLY (He BOILOFF), USED TO COUNTER ATMOSPHERIC DRAG AND SOLAR WIND EFFECTS

COMMENTS: ONLY 1 YR PROGRAM IF ADVANCED TECHNOLOGY GYROS "PROVE-OUT", STATE-OF-ART GYROS WOULD RESULT IN 2-3 YR PROGRAM, FIRST FLIGHT ON STS WILL BE TECHNOLOGY DEMONSTRATION

ATTITUDE TOWARD RESUPPLY: CURRENT DESIGN HAS BEEN DOWN-SIZED, IF STATE-OF-ART GYROS ARE USED GP-B WILL PROBABLY REQUIRE RESUPPLY

CONTACT: JOYCE NEIGHBORS (MSFC) 205-453-5584

CONSUMABLES REQUIREMENTS

PROGRAM: GRO

SIZE: DIA (ft) 15 (ON-ORBIT) LENGTH (ft) 17 (ON-ORBIT) MASS: 33,000 LB

S/C REPLACEMENT COST: $180M

LIFETIME: 27 MO. (>27MO DESIRABLE)

CONSUMABLES: TYPE N_2H_4 USAGE 3830 LB OVER 27 MO

S/C STORAGE CONDITIONS: PRESSURE TANKS AT 60°F, INITIAL PRESSURE OF 350 psia DECREASING TO 100 psia AT END-OF-CYCLE

RESUPPLY TRANSFER CONDITIONS: PRESSURE FED UMBILICAL SYSTEM, 200°F AND >400 psia

RESUPPLY TIMES: APPROXIMATELY 10 HR AFTER 27 MO OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 227 (216 AFTER @ YR) i(deg) - 28.5

Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: INTEGRAL PROPULSION MAY BE USED TO LOWER

COMMENTS: APPROXIMATELY 50 LB He FOR SINGLE BLOWDOWN SYSTEM; ALTERNATE RESUPPLY SCENARIO - BLEED PRESSURANT, REFILL N_2H_4, REFILL PRESSURANT; EARLIER DESIGN USED ARGON

ATTITUDE TOWARD RESUPPLY: RESUPPLY HIGHLY DESIRABLE TO EXTEND OPERATIONAL LIFE

CONTACT: RON McCULLAR (OSSA) 202-453-1428

CONSUMABLES REQUIREMENTS

PROGRAM: LDR

SIZE: DIA (ft) 66 LENGTH (ft) 100 MASS: 88,000 LB

S/C REPLACEMENT COST: $250M LIFETIME: 10 YR.

CONSUMABLES: TYPE SUPERFLUID He USAGE 1320 LB OVER 3 YR

S/C STORAGE CONDITIONS: DEWAR CONFIGURATION (CLOSED SYSTEM) AT 4 K

RESUPPLY TRANSFER CONDITONS: WOULD REQUIRE SIRTF-TYPE ADVANCED TRANSFER TECHNOLOGY

RESUPPLY TIMES: APPROXIMATELY 10 HR AFTER 3 YR OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 486 i(deg) - 28.5
Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: WILL BE INVESTIGATED,

COMMENTS: WOULD LIKE SUN-SYNC ORBIT, CRYO USAGE DERIVED FROM ESTIMATED THERMAL REQUIREMENTS (HEAT FLOW RATE) ASSUMING A CLOSED SYSTEM, ℓ_{HC} = .65 Cal/cc, Q = 1 WATT @ 4 K.

ATTITUDE TOWARD RESUPPLY: DESIGN TO INCLUDE RESUPPLY AND CHANGEOUT

CONTACT: BRUCE PITTMAN (NASA ARC) FTS 448-5692

PROGRAM: MPS

SIZE: DIA (ft) UNDETERMINED LENGTH (ft) UNDETERMINED MASS: 25,000 LB

S/C REPLACEMENT COST: $60M LIFETIME: 5 YR

CONSUMABLES: TYPE H_2O USAGE 5000 LB OVER 6 MO

S/C STORAGE CONDITIONS: ELECTROPHORESIS SYSTEM

RESUPPLY TRANSFER CONDITONS: PROBABLY CHANGEOUT OF ELECTROPHORESIS TANKS

RESUPPLY TIMES: APPROXIMATELY 5 HR AFTER 6 MO OF OPERATION

ORBITAL PARAMETERS: h (nmi) - 486 i(deg) - 28.5
Ω (deg) - UNDETERMINED ω (deg) - UNDETERMINED

S/C PROPULSIVE CAPABILITY: IF NECESSARY, MMS TYPE PROPULSIVE MODULE COULD BE USED FOR ORBIT CHANGES ONLY (PROBABLY NOT NECESSARY)

COMMENTS: PRIMARY EFFORTS ARE TO LOCATE FACTORY MODULES ON SPACE STATION. TIMING MAY BE SUCH THAT FREE-FLYER FACTORY MODULES WILL BE IN OPERATION PRIOR TO SPACE STATION.

ATTITUDE TOWARD RESUPPLY: IF DESIGNED AS FREE-FLYER, RESUPPLY/SERVICING WILL BE MANDATORY

CONTACT: DAVE RICHMAN (McDONNELL DOUGLAS) 314-232-6967

CONSUMABLES REQUIREMENTS

PROGRAM: PROTEUS

SIZE: DIA (ft) VARIABLE LENGTH (ft) VARIABLE MASS: 6000 LB AVG. (W/PAYLOAD)

S/C REPLACEMENT COST: $25M LIFETIME: 10 YR (2 YR MAX PER USER)

CONSUMABLES: TYPE N_2H_4 USAGE 1100 LB OVER 2 YR (ORBIT TRANSFER & STATION KEEPING)

S/C STORAGE CONDITIONS: PRESSURE TANKS

RESUPPLY TRANSFER CONDITONS: MAY USE PRESSURE OR PUMP FED UMBILICAL SYSTEM

RESUPPLY TIMES: APPROXIMATELY 2.5 HR AFTER 2 YR MISSIONS

ORBITAL PARAMETERS: h (nmi) - 216 i(deg) - 28.5

Ω (deg) - VARIABLE ω (deg) - VARIABLE

S/C PROPULSIVE CAPABILITY: PROBABLY INTEGRAL, MAY USE OMV

COMMENTS: PRELIMINARY DESIGN PHASE, MAY BE DERIVATIVE OF MMS, TWO S/C BASELINED

ATTITUDE TOWARD RESUPPLY: DESIGN WILL INCLUDE RESUPPLY AND SERVICING

CONTACT: W. HIBBARD (GSFC) 301-344-7510

CONSUMABLES REQUIREMENTS

PROGRAM: SIRTF

SIZE: DIA (ft) $\approx$12 LENGTH (ft) 27 MASS: 16,300 LB W/O PROPULSION

S/C REPLACEMENT COST: $300M LIFETIME: 10 YR

CONSUMABLES: TYPE SUPERFLUID He (SOLID H_2 WOULD BE USED IF OPEATING AT LOW INCL.) USAGE 992 LB He OVER 2 YR 235 He OVER 2 YR

S/C STORAGE CONDITIONS: DEWAR CONFIGURATION, He @ 1.8K, H @ 7-10K, He COLS WITHIN H COULD KEEP IT SOLID

RESUPPLY TRANSFER CONDITONS: UMBILICALS POSSIBLE, NEW TECHNOLOGY REQUIRED FOR TRANSFER, EXPECT 80-90% OF He CAN BE REPLENISHED AND 80% OF H_2

RESUPPLY TIMES: APPROXIMATELY 7 HR AFTER 2 YR OF OPERATION 10 HR FOR He AND H AT LOW INCLINATION)

ORBITAL PARAMETERS: h (nmi) - 378-486 (SECOND CHOICE-378 NMI @ 28.5°) i(deg) - 93

Ω (deg) - VARIABLE ω (deg) - VARIABLE

S/C PROPULSIVE CAPABILITY: NONE

COMMENTS: TECHNOLOGY DEVELOPMENT FOR CONSUMABLES TRANSFER IS INCLUDED IN RFP, POLAR ORBIT FIRST PREFERENCE, ONLY 600 LB SUPERFLUID He REQUIRED FOR LOW INCLINATION

ATTITUDE TOWARD RESUPPLY: RESUPPLY WOULD BE RISKY, MAXIMUM OPERATIONAL LIFETIME (2 YR) DESIRED BEFORE RESUPPLY

CONTACT: WALT BROOKS (NASA ARC) FTS 448-6530
JIM MURPHY (NASA ARC) FTS 448-6643

Due to the conceptual or preliminary state of many of the candidate missions, particularly those in the late 1990's, many assumptions had to be made to develop representative requirements. Those assumptions were:

- o Hydrazine propulsion using a single blowdown system was assumed for those missions requiring propulsion, but which had no spacecraft design baselined. Although no currently baselined candidate spacecraft designs utilize bi-propellants or require resupply of pressurants, future spacecraft designs may include these as an option for trades analyses.

- o Once rendezvous with candidate spacecraft has been accomplished, one hour is required for each 500 lb of hydrazine to be resupplied (includes actual transfer and associated operations).

- o The spacecraft replacement cost was estimated to be 40% of the total program cost when the actual replacement costs could not be determined.

- o Mission operational lifetime was extended beyond the program baseline if it would be technically or scientifically advantageous and consumables resupply was the primary constraint.

- o A hybrid cryogenic system was assumed for the large deployable reflector to determine consumable quantities (eg. a combination closed dewar and passive radiator system).

- o Candidate missions scheduled prior to an operational OMV were assumed to incorporate integral propulsion or use a spacecraft bus to rendezvous with the orbiter for resupply.

- o Materials processing in space conducted on free-flying factory modules would be primarily involved in electrophoretic operations.

Results from the user contact efforts are summarized on the following pages for the missions listed below:

Summary of Fluid Transfer Parametric Requirements Study Results

Parametric techniques were used to make estimates of the types, amounts and yearly fluid usage rates required by the resupply mission model. Small quantities of associated pressurants such as He and N_2 were also required, but their amounts were relatively minor in comparison to the primary fluids. In some cases, depending on the receiver system design, no additional pressurants are required for the transfer process.

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While water is a significantly required fluid, it is involved in a modular changeout in MPS factories and is thus not ideally suitable for transfer by the resupply module. As expected, hydrazine and storable bi-propellant are the dominant required fluids. In addition, another significant fluid of interest was found to be liquid (primarily superfluid state) helium. The original study scope excluded the consideration of OMV and OTV in the development of the fluid transfer parametric requirements. This was changed at the mid-term briefing but the requirements phase had already been completed. However, OMV and OTV requirements will be added during the follow-on. The results of the fluid transfer parametric requirements analysis are shown in Table 4.

USER AND CONSUMABLE TYPE	RESUPPLY AMOUNT (LB) 1989	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	2001
NASA													
HYDRAZINE	—	400	3,830	1,100	5,430	1,100	6,694	5,100	1,100	1,100	5,100	1,100	1,100
LIQUID HELIUM	—	—	—	—	160	—	1,389	—	992	1,320	992	—	2,312
WATER	—	5,000	15,000	25,000	30,000	30,000	25,000	15,000	5,000	—	—	—	—
Xe-METHANE MIXTURE	—	—	—	—	30	85	—	—	85	—	—	85	—
COMMERCIAL (POTENTIAL)													
BI-PROP (N_2O_4/MMH)	—	—	—	—	—	—	818	2,482	2,307	3,318	4,694	2,414	4,694
HYDRAZINE	—	—	—	—	—	—	—	500	1,000	500	500	500	500
DOD*													
BI-PROP (N_2O_4/A-50)	—	14,000	14,000	7,000	21,000	14,000	28,000	—	35,000	7,000	7,000	28,000	14,000
HYDRAZINE	—	—	70	70	70	17,470	17,470	70	8,770	70	17,470	17,470	70
LIQUID HELIUM	—	—	—	—	—	2,400	2,400	—	1,200	—	2,400	2,400	—
DOD* (POTENTIAL)													
BI-PROP (N_2O_4/A-50)	—	—	—	—	—	—	8,000	15,000	2,000	1,000	15,000	2,000	14,000
HYDRAZINE	—	—	—	—	—	—	—	950	250	600	950	600	950
CONTINGENCY**	—	—	1,500	—	—	1,500	—	—	1,500	—	—	1,500	—

*SMALL QUANTITIES OF PRESSURANTS ALSO REQUIRED FOR SOME SATELLITES — IMPORTANT FOR DESIGN REQT
**ESTIMATE OF PROPELLANT RESUPPLY REQUIRED IN CONTINGENCY SITUATIONS

- HYDRAZINE USED FOR THE MOST NUMBER OF UMBILICAL ENGAGEMENTS
- GREATEST MASS OF FLUID FOR UMBILICAL TRANSFER IS BI-PROP
- WATER & SPECIAL GASES ARE TRANSFERRED VIA MODULE CHANGEOUT

Table 4. Fluid Parametric Requirements Summary

3.1.2 Candidate Spacecraft and Resupply Scenarios

Activity 1.1.3 was concerned with screening all the missions from the mission model and selecting a smaller number for use in the development of alternate resupply scenarios and mission operations requirements. The selection of a representative set of missions occurred in two parts. The first was the selection of missions which included the technical performance requirements of all possible resupply candidates (e.g., LEO, polar, GEO, etc.). The second step was a screening for missions that are likely to benefit from resupply. The four general mission characteristics which were used are shown in Table 5.

Nine candidate spacecraft/platforms were selected. They occupied the three general resupply locations of low-inclination LEO, high-inclination LEO and GEO. Various single and multi-servicing scenario combinations were then examined for these candidates. All of the candidates either require resupply or benefit highly from its use and thus are promising missions to examine closer for developing alternative resupply scenarios and requirements.

Using as input data the candidate spacecraft and space platforms selected for Activity 1.1.3, a comprehensive set of resupply scenarios was generatd (Activity 1.2.1). These scenarios included non-remote resupply options. In addition, certain scenario trade issues, such as multiple versus single spacecraft resupply advantage and disadvantages, wet versus dry launch considerations, etc., were addressed as part of this activity. These data were used in the concurrent effectiveness analysis to select the most promising scenarios for further definition. Mission operational requirements were derived for these scenarios (in Activity 1.2.2) to develop system/subsystem/interface requirements (Activities 1.2.3 and 1.2.4) for both the effectiveness analysis and the Design Reference Missions (DRM) selected

GENERIC SPACECRAFT CLASS		MISSION OBJECTIVE DURATION	UNIT ON-ORBIT VALUE $M84 SPACECRAFT	DURATION CONSTRAINT (HISTORIC)	COST OF ACCESS (STS)
• SCIENTIFIC EXPLORERS	EUVE CREP	SHORT 6 MO-24 MO	VERY LOW $50-100	EXPERIMENT, FLUIDS	VERY LOW-MODERATE
• ASTRONOMIC OBSERVATORIES	GRO AXAF	CONTINUOUS	VERY HIGH $500-1200	EXPERIMENT, INSTRUMENT	LOW
• EARTH/WEATHER OBSERVATION	LANDSAT DMSP	CONTINUOUS	LOW-HIGH $80-300	SENSORS, POWER	MODERATE
• RECON/SURVEILLANCE		CONTINUOUS	HIGH-VERY HIGH		MODERATE
• NAVIGATION	GPS	CONTINUOUS	VERY LOW $50-100	UNKNOWN	HIGH
• COMMUNICATION-DELTA	H.S. 376	CONTINUOUS	LOW $75-125	TECHNOLOGY OBSOLETE	VERY HIGH
• ENVIRONMENTAL OBSERVATION	GOES	CONTINUOUS	?	?	VERY HIGH
• EARLY WARNING DOD		CONTINUOUS	?	?	VERY HIGH
• COMMUNICATION-IUS/TOS	TDRS INTEL VI	CONTINUOUS	MODERATE-HIGH $250-300	TECHNOLOGY OBSOLETE	VERY HIGH
• COMPLAT	PLANNED	CONTINUOUS	VERY HIGH	DESIGN REQUIREMENTS	VERY HIGH
• SMALL PLATFORMS	PROTEUS LEASE CRAFT	MEDIUM 120 MO	MODERATE $100-200	CHANGEOUT/INVENTORY	VERY LOW-MODERATE

LOW INCLINATION LEO

- AXAF
- MPS
- GRO
- PROTEUS

HIGH INCLINATION LEO

- SIRTF
- DMSP
- DOD-1

GEO

- GEO PLATFORM
- COMSATS

VARIOUS SINGLE & MULTI-SERVICING SCENARIO COMBINATIONS EXAMINED FOR THESE CANDIDATES

Table 5. Candidate Spacecraft/Platform Selections

from this analysis. The system/subsystem requirements data defined as part of 1.2 also enabled the development cost/risk benefits data within the effectiveness analysis to support justification of the DRM's.

Ten general resupply scenarios, shown in Figure 8, were developed for operations in LEO. These scenarios included not only remote resupply, but use of Orbiter-tended operations in order to insure a comprehensive examination of alternatives. A spacecraft in LEO can be resupplied using a resupply module on a ground-based OMV, a space-based OMV or an OMV based at a space station. These constitute the remote resupply options. The spacecraft can also be resupplied using in-bay equipment as are being developed at JSC. Getting the spacecraft to the Orbiter bay can be accomplished using the spacecraft's own integral propulsion systems (IP), the OMV, or the Orbiter propulsion capability. In each of the remote resupply options, the resupply module can perform multiple or single engagements.

Several trade studies were identified in the original proposal related to evaluating the advantages/disadvantages of the alternative resupply scenarios. These studies focused on key discriminators among the various resupply scenarios and were intended to develop data for the screening of these alternatives in the effectiveness analysis. There were six small studies conducted within Activity 1.2.1. The first examined the utility of nodal regression effects in multiple engagement mission planning. The second examined alternative space-basing options relative to ground-basing the resupply module. This task also used much of the material developed in the first study on multiple engagements. The third study looked at the pros and cons of dry and wet launches of the resupply module itself. This had to

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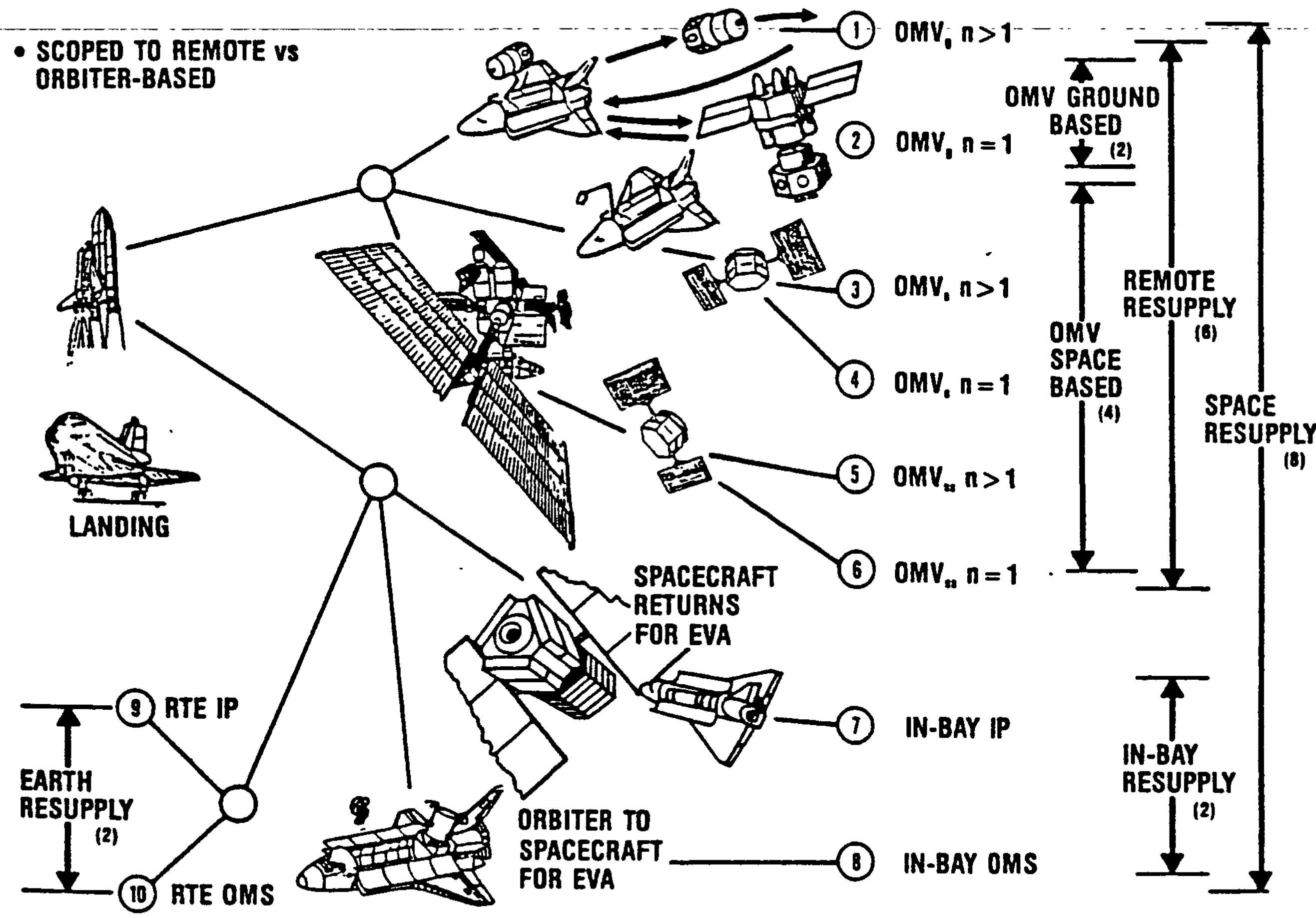


Figure 8. Low Earth Orbit Alternative Fluids Resupply Scenarios

include options for pumping from the OMV tankage, use in contingency operations where the required fluid may not be known well in advance, and the resulting on-orbit complexities involved if the resupply module arrives in space dry. Another study was oriented toward GEO operations and examined the weight sensitivities involved with expending versus reusing elements of a GEO resupply system (OTV, OMV, RM). Cost factors were also identified for use by the effectiveness analysis.

The final two studies supported development of system/subsystem/interface requirements. Issues as to the appropriate levels of autonomy and redundancy capabilities for the resupply module were addressed early in the study. We evaluated both the OMV's capabilities and the potential role of man-in-the-loop (M-I-L). Finally, we searched for critical issues in fluid transfer technologies relative to the identified and required fluids. Superfluid helium was found to have a variety of issues associated with its transfer and thus would require future study.

Multiple Engagement Planning

Prior work on use of the Orbital Maneuvering Vehicle (OMV) for satellite servicing raised the possibility of using nodal regression effects in multiple satellite engagements. Nodal regression is due to the non-sphericity of the earth and can result in changes of up to 9 degrees per day at low altitude and low inclination orbits. Such effects decrease by a cosine term as orbital inclinations increase and effectively cease for polar orbits.

An optimization algorithm was developed for planning multiple resupply engagements. For real world situations of 2-3 engagements, at most on a single resupply mission, closed form solutions are possible. These solutions specify what sequence of orbital engagements is optimum, depending on whether it is desired to minimize time or fuel usage. For a greater number of engagements, the particular number varies depending on the mission complexity, closed form optimal solutions are not possible and resort must be made to various search techniques.

Resupply Module Basing Options

Figure 9 depicts the range of basing options examined in Activity 1.2.1. Starting with filled resupply modules, we have two general ground-based options and three space-based options. These options serve the four general user groups denoted at the top of the page. Earlier work on space versus ground-basing studies with respect to the OMV identified many of the current issues related to the costs, risks and operational requirements of space-basing.

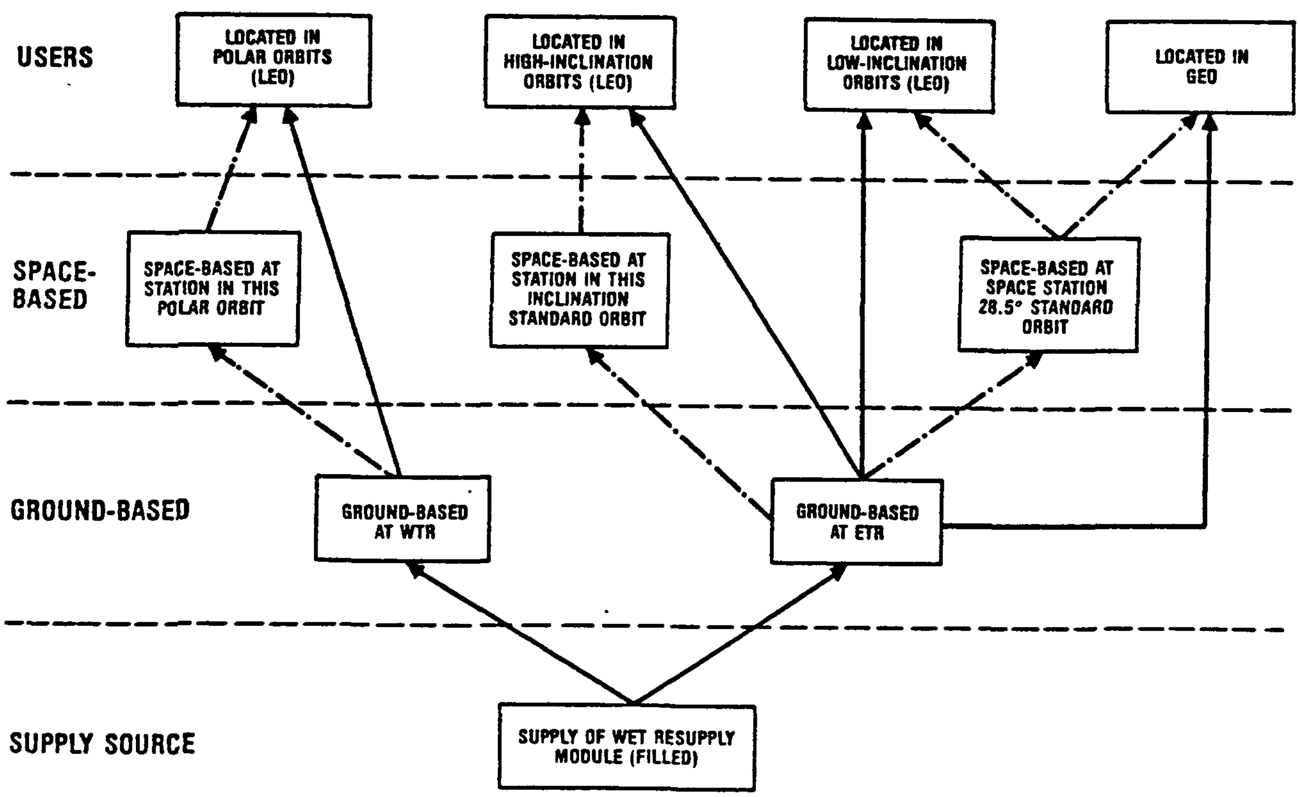


Figure 9. Resupply Module Basing Options

The specific conclusions drawn by the space vs. ground-basing study are shown in Figure 10. Essentially, neither system is completely preferred, but rather there needs to be a mix of both. The use of space-basing, particularly with a space-based OMV, has significant economic advantages in the saving of earth-to-space transport costs. Contingency missions, however, are likely to require the sending up of a ground-based resupply module in that the specific needs of the mission may not be available on-orbit within an acceptable time. The economics of space-basing argue for its implementation as operational space experience is built up. This increase in experience will allow later decisions on the exact number of vehicles that should be space-based and what their optimal locations should be. Aside from the choice of a 28.5 degree inclination, other parameters such as additional inclinations, altitudes and nodal crossing times of space-based assets are still in question.

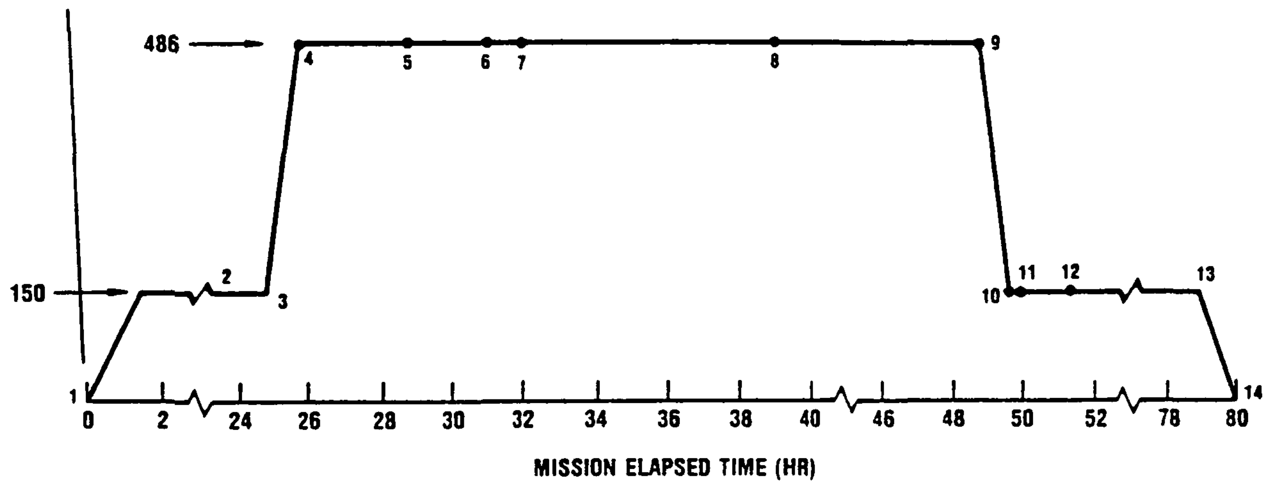
- RM BASING NEEDS TO BE CONSIDERED IN VIEW OF THE OPTIMALITY OF THE WHOLE SYSTEM EFFECTIVENESS, NOT IN SEPARATE PIECES
- SPACE- & GROUND-BASING MODES ARE COMPETING & COOPERATIVE CONCEPTS
- SPACE-BASING IS PREFERRED FOR MEETING DETERMINISTIC RESUPPLY DEMANDS; I.E., MISSIONS WHOSE RESUPPLY SCHEDULE IS KNOWN WELL IN ADVANCE
- GROUND-BASING IS PREFERRED FOR MEETING PROBABILISTIC DEMANDS, I.E., MISSIONS WHOSE SPECIFIC RESUPPLY REQUIREMENTS ARE NOT KNOWN WELL IN ADVANCE, AS WITH CONTINGENCIES; ALSO, DUE TO LACK OF EFFECTIVE NODAL REGRESSION EFFECTS, SPACE-BASING IN POLAR OR NEAR-POLAR ORBITS IS NOT PREFERRED
- NODAL COLOCATION OF SPACECRAFT ("BEADS ON A STRING") SHOULD BE ENCOURAGED FOR RESUPPLY EASE
- LONG-TERM SPACE EXPERIENCE WILL ACCURATELY DEFINE THE DETERMINISTIC DEMAND FOR RESUPPLY & THUS THE RELATIVE EMPHASIS NEEDED ON SPACE-BASING THE RM

Figure 10. Summary of Basing Mode Conclusions

Mission Timeline Analysis

Activity 1.2.2 focused on the development of mission operational requirements. Specific spacecraft were used to develop alternative scenarios to a greater level of detail. Starting with the functional descriptions developed in Activity 1.2.1, mission timelines, delta V requirements, and propellant weights were developed. The purpose of this was to take a closer look at what the various alternative scenarios actually involved and to create a base of data and experience for use in the effectiveness analysis. All of the remote resupply scenarios developed in Activity 1.2.1 for LEO and GEO operations were detailed. The candidate spacecraft/platforms selected in Activity 1.1.3 were used to derive specific timelines and propellant requirements. The use of the Orbiter in resupply operations was considered to be outside of contract scope for this task. Resupply with the Orbiter was examined in the effectiveness analysis as an option the remote resupply scenarios had to compete with.

Figure 11 shows an example of a resupply mission which uses a ground-based OMV in a single engagement with the Shuttle Infrared Telescope Facility.



PERFORMANCE EVENT	V	Wp
1. LAUNCH ORBITER, 28.5°	FT/SEC	LB
2. DEPLOY OMV/RM		
3. BOOST FROM ORBITER ALTITUDE	561	379
4. CIRCULARIZE (COPLANAR)	548	350
5. TERMINAL RENDEZVOUS		
6. PROXIMITY OPERATIONS		
7. RESUPPLY SIRTF		
8. DISENGAGE SIRTF		

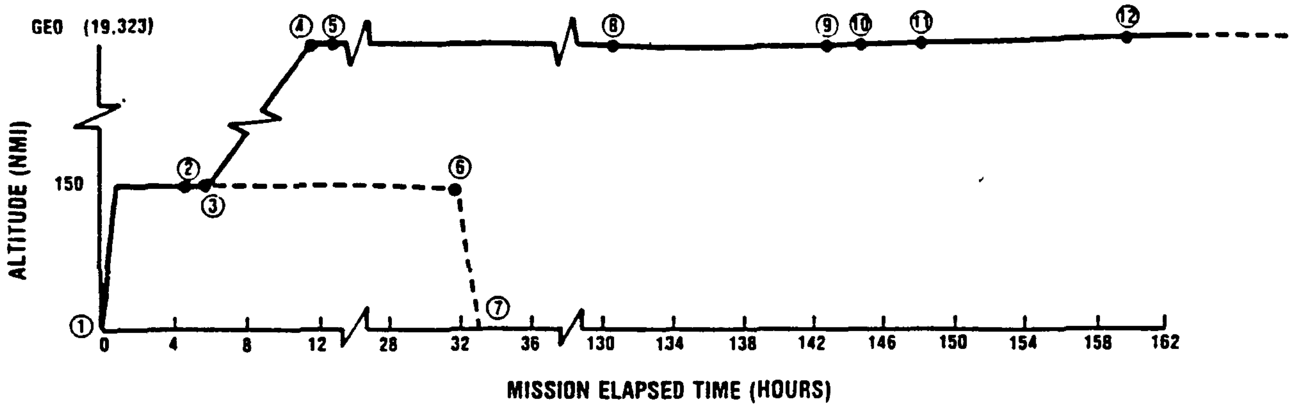
PERFORMANCE EVENT	V	Wp
9. RETROBURN W/.493° PLANE CHANGE	586	298
10. CIRCULARIZE AT 150 NMI W/437° PLANE CHANGE		
11. ORBITER/OMV RENDEZVOUS	10	5
12. RETRIEVE OMV/RM		
13. DEORBIT		
14. LAND		

- DEFINED 20 SCENARIOS TO GREATER LEVELS OF DETAIL INCLUDING TIMELINES, ΔV'S, PROPELLANT REQUIREMENTS, ETC

TOTAL OMV PROPELLANT	1423 LB
TOTAL RM WT (INCLUDE FLUIDS)	1742 LB
INITIAL MISSION WT (W/O ASE)	7126 LB

Figure 11.
SIRTF Servicing Mission Timeline (Ground Based OMV) 28.5° Orbit

Figure 13 shows a resupply mission to a large GEO platform using a Centaur G. The OMV used is ground-based and stored at GEO for later servicing opportunities. The RM may also be eventually retrieved by a reusable OTV.



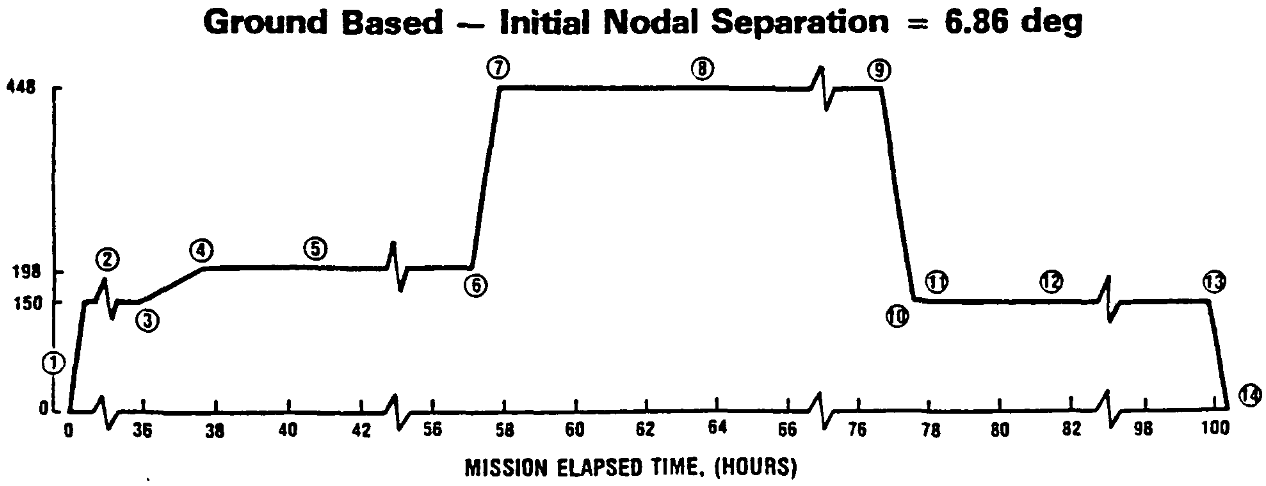
PERFORMANCE EVENT	ΔV ft/sec	Wp lb
1 LAUNCH ORBITER		
2 DEPLOY CENTAUR/OMV/RM		
3 BOOST FROM ORBITER ALTITUDE	8,134	20,370
4 CIRCULARIZE AT 19,300 NMI	5,852	8,578
5 DRIFT TO PLATFORM LONGIT	20	21
6 DEORBIT		

PERFORMANCE EVENT	ΔV ft/sec	Wp lb
7 LAND ORBITER		
8 RENDEZVOUS AT GEO ALT	5	5
9 PROXIMITY OPERATIONS		
10 RESUPPLY PLATFORM (1,764 lb)		
11 SEPARATE TO DORMANT ORBIT	22	23
12 STORE OMV AT Δh = 100 NMI FOR LATER SERVICING OPPORTUNITY		

TOTAL OMV PROPELLANT	2,350 lb
TOTAL RM WT (W/FLUIDS)	4,278 lb
INITIAL MISSION WT (W/O ASE)	46,796 lb

Figure 13 GEO Platform Resupply Using Centaur G.

Figure 14 shows a multiple satellite engagement in polar orbits with a ground based OMV. The DoD spacecraft being resupplied have not only different nodal crossing times, but slightly different inclinations as well.



PERFORMANCE EVENT	ΔV	Wp
① LAUNCH ORBITER		
② DEPLOY OMV/RM		
③ BOOST FROM ORBITER ALTITUDE	88	101
④ CIRCULARIZE (CO-PLANAR)	87	99
⑤ RENDEZVOUS & SERVICE DOD-1	10	11
⑥ BOOST TO DMSP W/2.38° PLANE CHANGE	1134	1104
⑦ CIRCULARIZE W/4.62° PLANE CHANGE	1992	1660

PERFORMANCE EVENT	ΔV	Wp
⑧ RENDEZVOUS & SERVICE DMSP	10	8
⑨ RETROBURN W/4.51° PLANE CHANGE	1965	1244
⑩ CIRCULARIZE AT 150 NM W/2.49° PLANE CHANGE	1221	658
⑪ ORBITER/OMV RENDEZVOUS		
⑫ RETRIEVE OMV/RM		
⑬ DEORBIT		
⑭ LAND		
TOTAL OMV PROPELLANT		5,085 lb
TOTAL RM WT (W/FLUIDS)		2,250 lb
INITIAL MISSION WT (W/O ASE)		11,306 lb

Figure 14. DoD-1 and DMSP Multi-Servicing Mission Timeline

This chart and the preceeding ones are meant as examples of the range of operational requirements that might be required of a resupply module and its carrier. The data developed in producing these specific scenarios was used in creating the functional allocation matrices for system/subsystem/interface requirements definition. A major purpose of the development of alternative scenarios was to establish the technical feasibility of the resupply options and provide data for the effectiveness analysis evaluation discussed next.

3.1.3 Effectiveness Analysis

The Statement of Work for the Spacecraft/Platform Expendables Resupply contract requires that "....one or more spacecraft or platforms will be jointly selected by the contractor and NASA for typical Design Reference Missions (DRM)."

The dual objectives of Subtask 1.3 were:

1. To select the "best" DRMs for subsequent use in Resupply Module (Task 2) design efforts, and,

2. To develop rationale and supporting data to justify authorization for remote resupply demonstration.

The two primary outputs of the Effectiveness Analysis were:

1. Recommended DRMs for NASA/MSFC approval/concurrence, and

2. Justification for Remote Fluids Resupply demonstration.

These outputs have been developed by an iterative, "rough order of magnitude" (ROM) parametric benefit/cost/risk assessment of resupply requirements for each of 12 spacecraft programs developed in Subtask 1.1; in combination with the mission profiles for each of 24 alternate fluids resupply scenarios developed in Subtask 1.2.

Completion of the Effectiveness Analysis subtask required three sequential activities:

1.3.1 Formulate Effectiveness Methodology
1.3.2 Generate Effectiveness Data
1.3.3 Recommend Design Reference Mission(s)

The methodology selected, and subsequently approved by NASA/MSFC, for use in the Expendables Resupply study was developed by Rockwell International. The methodology follows the (A-109 required) "justification for new start" process by defining user NEED as a function of benefits provided, and estimating concept AFFORDABILITY as a function of technical performance, cost, risk and other factors.

The NEED/AFFORDABILITY method has been used on several similar NASA concept evaluation study contracts (Alternate Thermal Protection System, NASA/LaRC; Shuttle Derived Launch Vehicle, NASA/JSC; and Teleoperator Maneuvering Systems Benefits, NASA/MSFC) and for internal Rockwell International Space Transportation Systems Division project evaluations in the past four years.

A formal review of the methodology, supplemented by computer demonstrations and NASA/MSFC participation in the prioritization of decision criteria for the Expendables Resupply study, was completed on 11 May 84.

Each Candidate Design Reference Mission (DRM) subjected to the Effectiveness Analysis consists of two major elements:

1. A candidate spacecraft/platform program in which the user's "NEED" for Expendables Resupply has been estimated as a function of the economic and/or operational benefits accruing to the program if fluids resupply were available, and

2. an alternate fluids resupply scenario, for which parametric cost, risk and complexity estimates are combined to determine that concept's AFFORDABILITY" to the user program.

As an example of the NEED/AFFORDABILITY evaluation method applied to the design reference mission selection, ROM parametric estimates of gross economic and operational benefits have been made for each candidate user spacecraft/platform defined in the potential resupply engagement model (from subtask 1.1). An initial list of 30-40 potential receiver spacecraft has been iteratively reduced by filtering out those programs in which significant economic/operational benefits could NOT be identified.

ROM cost, risk and complexity estimates were made for each of approximately 20 alternate fluids resupply scenarios for Low Earth Orbit (LEO) missions, and another four resupply scenarios for Geosynchronous Equatorial Orbit (GEO) missions.

Parametric estimates were refined with each iteration to produce a "PRIME CANDIDATE" list of approximately 12 receiver spacecraft programs for which substantial economic/operational benefits could be estimated and to which several of the 10 most cost/risk-effective resupply scenarios would be applicable.

Groundrules and Assumptions

The USAF Space Division's "Unmanned Spacecraft Cost Model" was used to develop many of the parametric cost estimates in the Expendables Resupply study, and ranges have been applied to the Cost Estimating Relationships (CERs) to indicate the degree of uncertainty associated with the estimates. Where alternative sources were felt to provide better CERs, they have been used; every reasonable effort has been made to reconcile CER estimates against available programmatic data on a spacecraft-by-spacecraft basis.

The "Analytical Hierarchy Process" (AHP) technique (after T.L. Saaty) has been used extensively throughout the study to convert subjective (engineering judgment or managerial preference) data into quantitative "proxy" form. The technique, with potential applications to the study, was discussed in considerable detail during the May, 1984 methodology review with NASA/MSFC.

Prime applications of AHP have been, (1) to establish decision goals, measurement criteria and the relative priority of each, (2) to estimate the risk and complexity preference parameters for each alternate fluids resupply scenario, and (3) to express the "contribution" of each alternative Candidate Design Reference Mission to each "best DRM" decision goal.

The groundrules and assumptions for the effectiveness analysis are summarized in Figure 15.

- ALL ECONOMIC & COST ESTIMATES IN CONSTANT 1984 $ MILLIONS
- PARAMETRIC ESTIMATES ARE ROUGH ORDER OF MAGNITUDE (ROM)
- USAF SD UNMANNED SPACECRAFT COST MODEL CERs USED
  - OMV ACQUISITION (MSFC BASELINE CONFIGURATION)
  - EXPENDABLES RESUPPLY MODULE (ERM) ACQUISITION
  - CANDIDATE SPACECRAFT/PLATFORM ACQUISITION
- ROCKWELL ORBITER DATA BASE CERs & INDEPENDENT ESTIMATES
  - LAUNCH (STS) & INSERTION STAGES
  - FLIGHT & MISSION OPERATIONS/SUPPORT
- RISK & COMPLEXITY ANALYTIC HIERARCHY PROCESS

BEST DESIGN REFERENCE MISSION DECISION RULE: SELECT CANDIDATE WITH HIGHEST OVERALL CONTRIBUTION TO CONFLICTING GOALS

Figure 15. Effectiveness Analysis Groundrules and Assumptions

LEO Benefits Summary - NASA LEO ETR Spacecraft

As shown in Figure 16, expendables resupply capability is a necessary condition for enabling satellite system availability through on-orbit maintenance (which offers substantial cost efficiencies compared to the traditional "proliferation" and/or "endurance" approaches).

Expendables Resupply as a stand-alone capability (without concurrent repair, technology upgrade or routine maintenance), may not be sufficient except in those cases where the only constraint on the spacecraft's function is depletion of its on-board consumables supply. Situations where this is the only constraint appear rare.

Rockwell's prior (1982-83) work with MSFC on the Teleoperator Maneuvering System (now, Orbital Maneuvering Vehicle) Benefits Assessment identified five general criteria which were useful in identifying the best candidates for generic on-orbit servicing:

- Continuous Mission Objectives
- High Unit-Value Spacecraft
- Unit Spacecraft Life Limited
- Low Cost Access
- Non-Competitive Program Sponsors

All five are applicable to Expendables Resupply (a subset of generic on-orbit servicing capability) and were used in the initial screening of the candidate spacecraft list.

The economic value of concurrent maintenance and fluids resupply is determined by the relationship between, (1) the spacecraft's remaining on-board fluids, and (2) the spacecraft unit's functional availability retention as described by a Weibull survival curve, for the nominally design spacecraft (Figure 17).

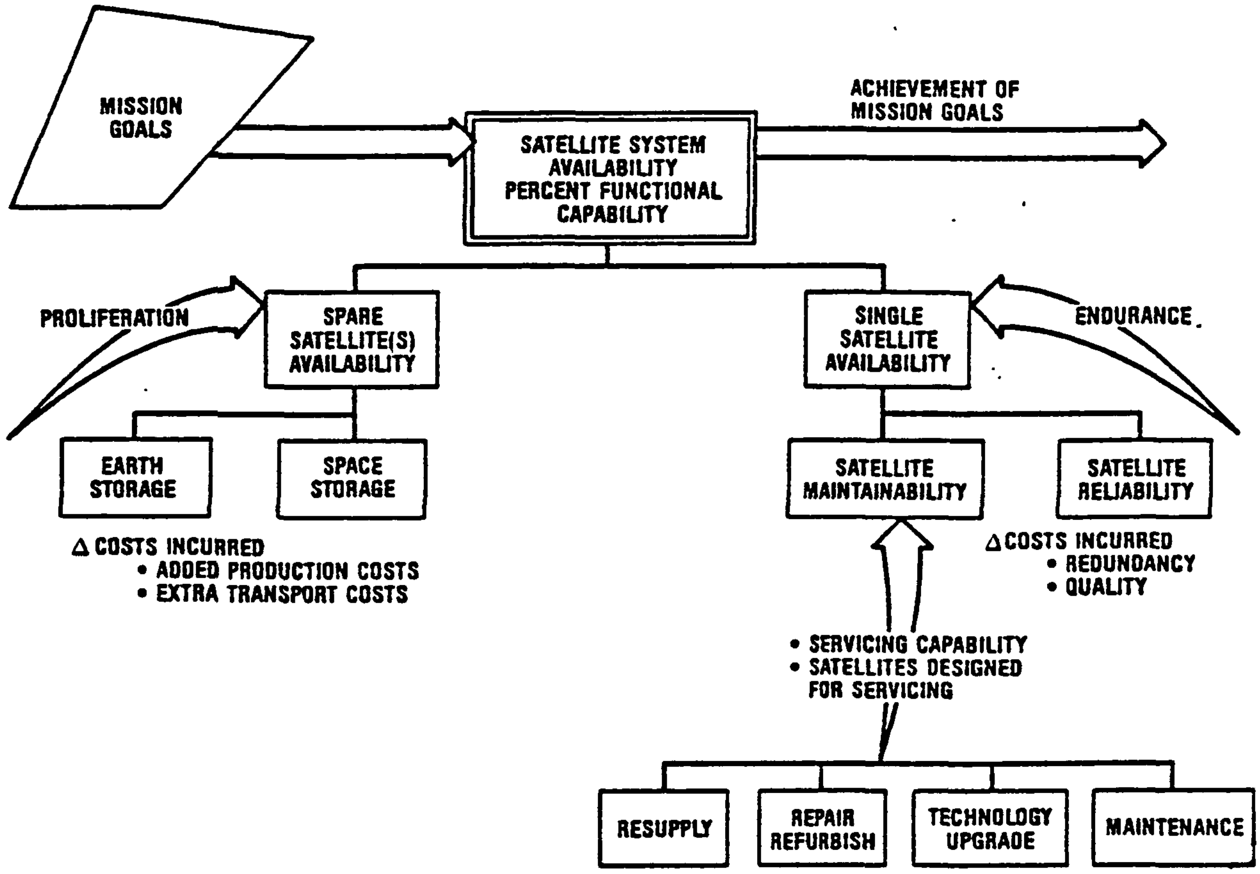


Figure 16. Satellite System Availability through Servicing

Only a small fraction of the spacecraft's initial functional capability can be expected to survive beyond its nominal design life (at mean mission duration, the probability of spacecraft function is approximately a/e = 37%). This effectively limits the expected residual economic value of resupply WITHOUT concurrent maintenance to the integrated area under the Weibull curve beyond t = mmd.

To receive greater benefit from resupply without concurrent maintenance, the spacecraft could be initially designed with a longer design life mmd, but at exponentially higher unit acquisition cost for internal redundancy and component quality.

Alternatively, the spacecraft might be designed for full serviceability with the nominal design life mmd for only a modest increase in acquisition cost, but with substantial increases in both spacecraft structural mass and volume to accommodate on-orbit servicing by either EVA or remotely with the OMV.

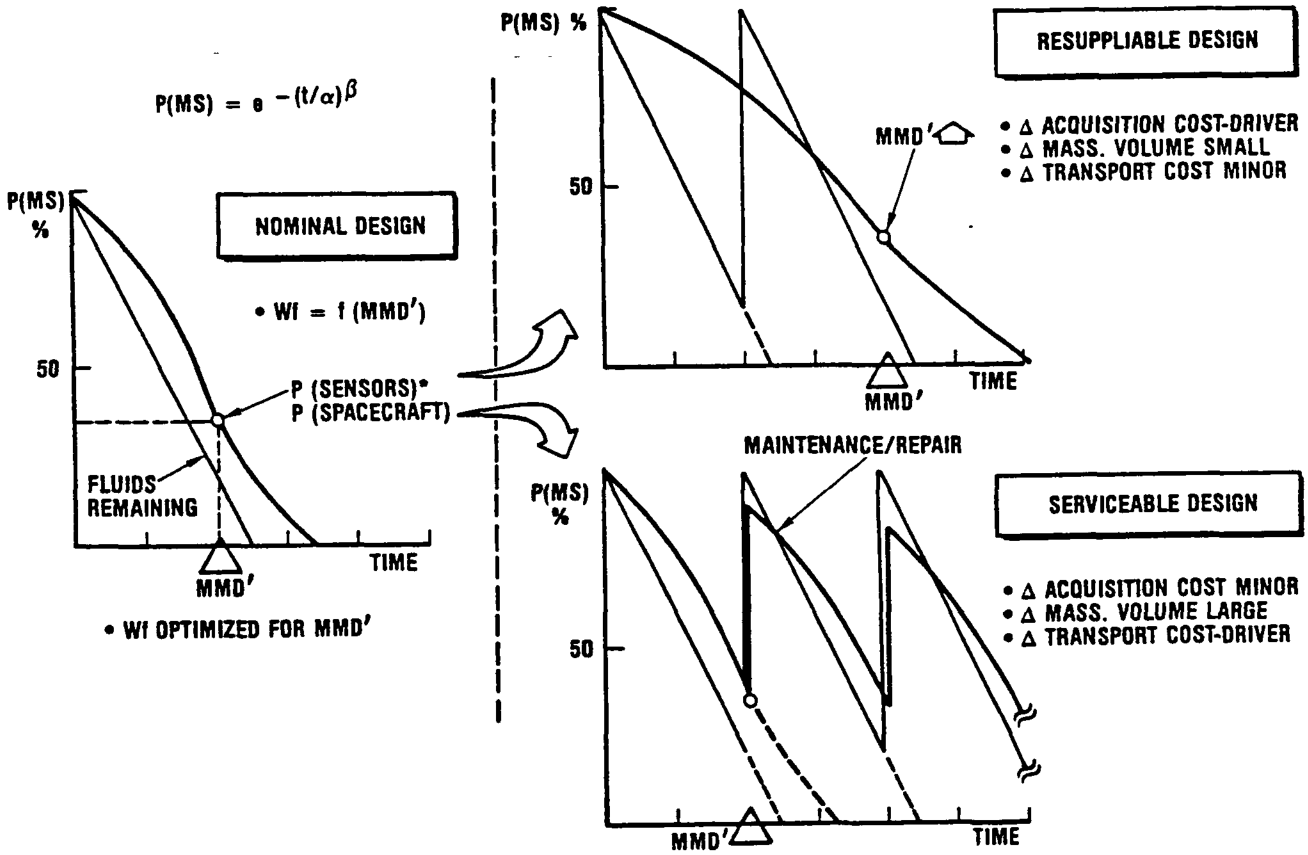


Figure 17. Expendable Resupply Relationship for Spacecraft Design Life

Gross economic benefit of Expendables Resupply without maintenance were estimated as the integrated area under Weibull survival curve past mmd for each of the 12 prime candidate spacecraft/platforms. (Figure 18).

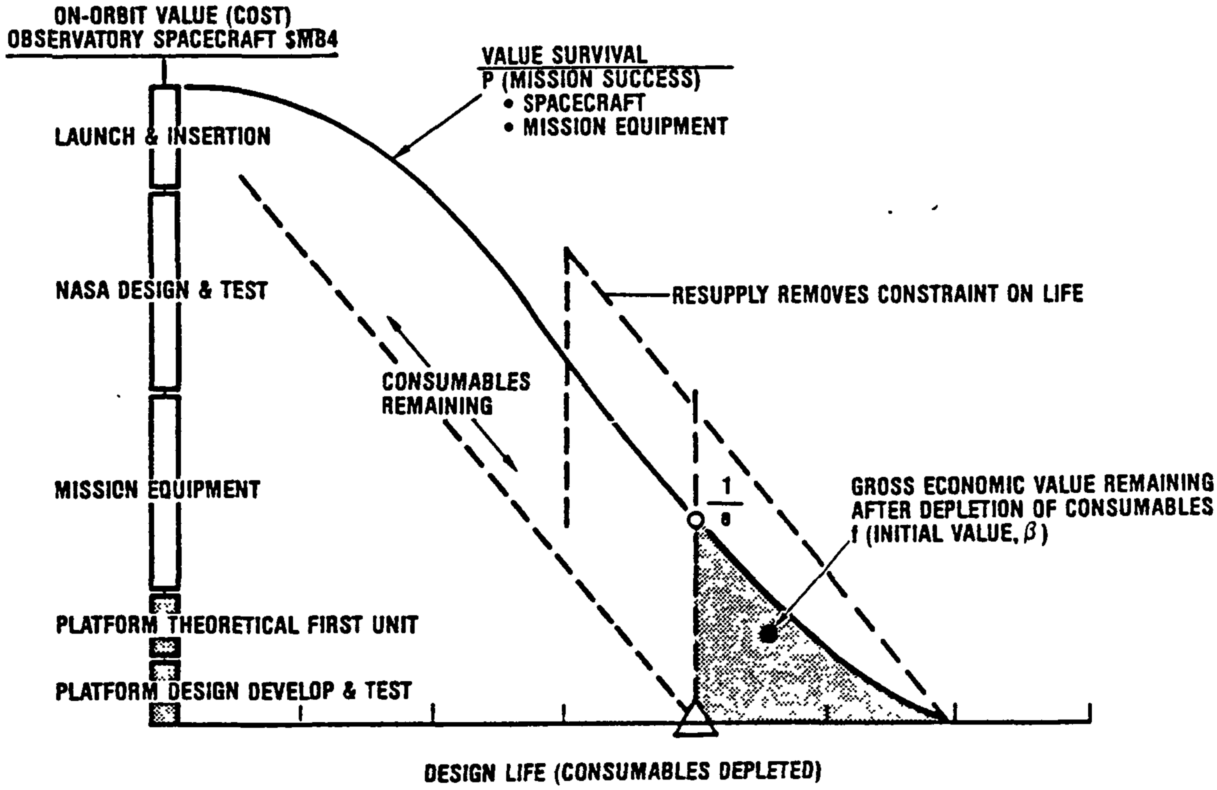


Figure 18. Design Life (CONSUMABLES DEPLETED)
Estimating Gross Economic Benefit of Resupply Without Maintenance

Results of the analysis of reestimated gross economic benefit of fluids resupply only (not including maintenance/refrain) for nine candidate spacecraft are presented in Table 6.

	RESUPPLY ONLY (No Maintenance)				REPLACE SPACECRAFT (Identical Technology)	
	(Beta)	ORIGINAL On Orbit Value $M	1st & (t)	Unit Area mmd $M	REPLACEMENT (TFU+Payload) Limit $ Benefit	(2nd) *CRC+L&I = Unit Cost of Service
AXAF	2.6	1,145	* 0.096 =	110	(142 + 223)	* .6 +110 = 329
EOS	3.0	728	* 0.066 =	48	(77 + 173)	* .6 +145 = 295
EUVE	1.6	195	* 0.173 =	34	(41 + ?)	* .8 + 10 = 43
GEOPLAT	1.2	686	* 0.205 =	141	(92 + ?)	* .8 +185 = 259
GrPr B	2.0	297	* 0.142 =	42	(43 + 93)	* .6 + 11 = 93
GRO	2.4	881	* 0.111 =	98	(94 + 272)	* .6 +110 = 330
LDR	1.8	1,369	* 0.158 =	216	(143 + 314)	* .6 +110 = 384
PROTEUS	1.8	254	* 0.158 =	40	(51 + ?)	* .8 + 13 = 54
SIRTF	2.8	693	* 0.081 =	56	(85 + 115)	* .8 +145 = 305

Table 6. Economic Comparison of Resupply only versus Replacement for Candidate Spacecraft

Initial on-orbit values for each spacecraft were estimated from available platform mass data and special mission equipment characteristic descriptions. An imputed "value-added" for NASA design and test efforts and launch/insertion costs were included. (Table 7) .

(estimates in $ 1984 million)

PROGRAM	PLATFORM dry mass (lbs)	++Contractor++ NR $M	TFT $M	DDT&E $M	NASA $M	Payload DDT&E lbs	$M	L&I $M	OnOrbit VALUE $M
AXAF	20,010	135	142	227	535	3010	223	110	1,145
EOS	9,660	75	77	152	258	11000?	173	145	728
EUVE	4,000	44	41	85	99	not	incl	10	195
GEOPLAT	12,000	89	92	181	320	not	incl	185	686
GrPr B	4,170	44	43	87	106		93	11	297
GRO	11,,835	84	94	178	321	12100	272	110	881
LDR	23,485	139	143	282	663	26115	314	110	1,369
PROTEUS	5,390	50	51	101	140	not	incl	13	254
SIRTF	10,625	79	85	164	269	7440	115	145	693
SP TEL	22,555	152	155	307	601	1800	?	110	1,310

* Platform Contract Value CER ($M84) = 36.7+.012 * dry mass lbs

Table 7. ROM ON-ORBIT VALUE OF NASA SPACECRAFT/PLATFORM PROGRAMS

Design life mmd were estimated from design provisions for consumables.

Residual value (beyond nominal design life) depends not only on estimated mmd and Beta parameters, but on initial spacecraft installed (on-orbit) value. From the wide variety of generic candidate spacecraft programs, the highest unit residual values--and the best bets for servicing and resupply--are found in the astronomic observatory, reconniassance/surveillance and earth observation satellite classes. (Figure 19).

Significant operational benefits can be expected to accrue to those programs where Expendables Resupply removes (or relaxes) an existing operational constraint, such as for surveillance programs (Figure 20).

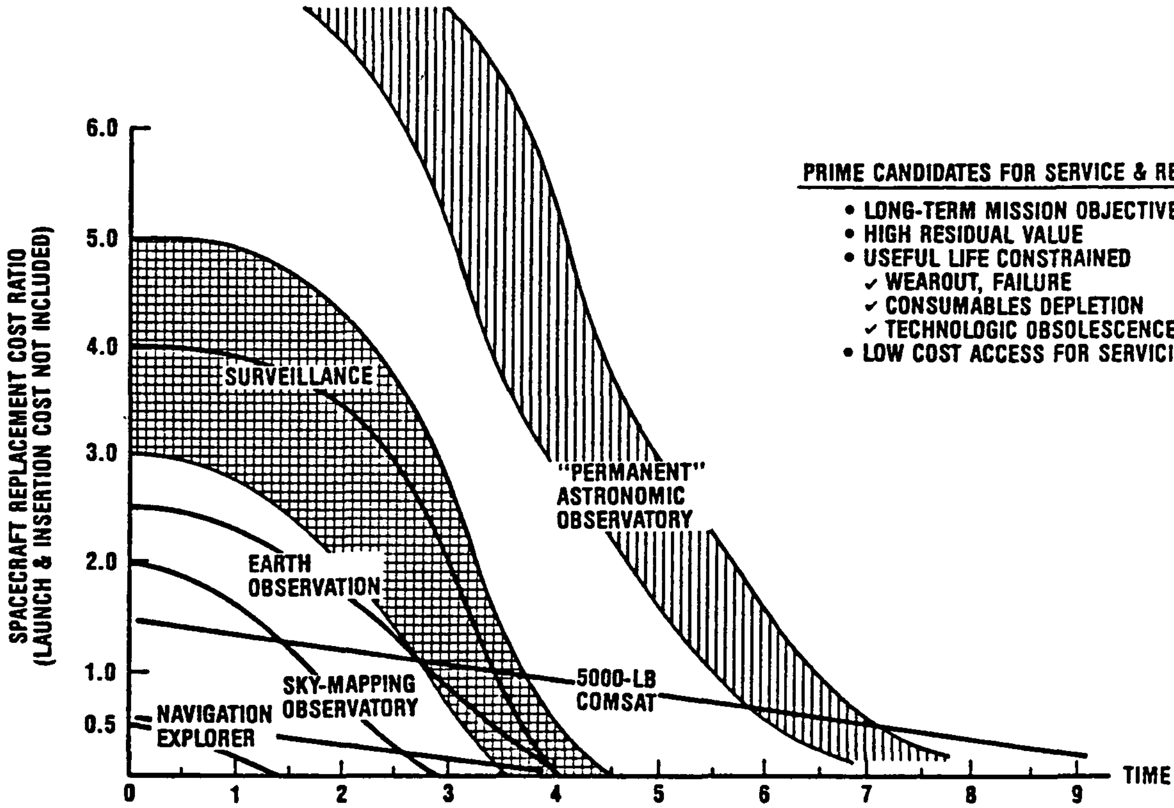


Figure 19. Relative Residual Value of Generic Satellites

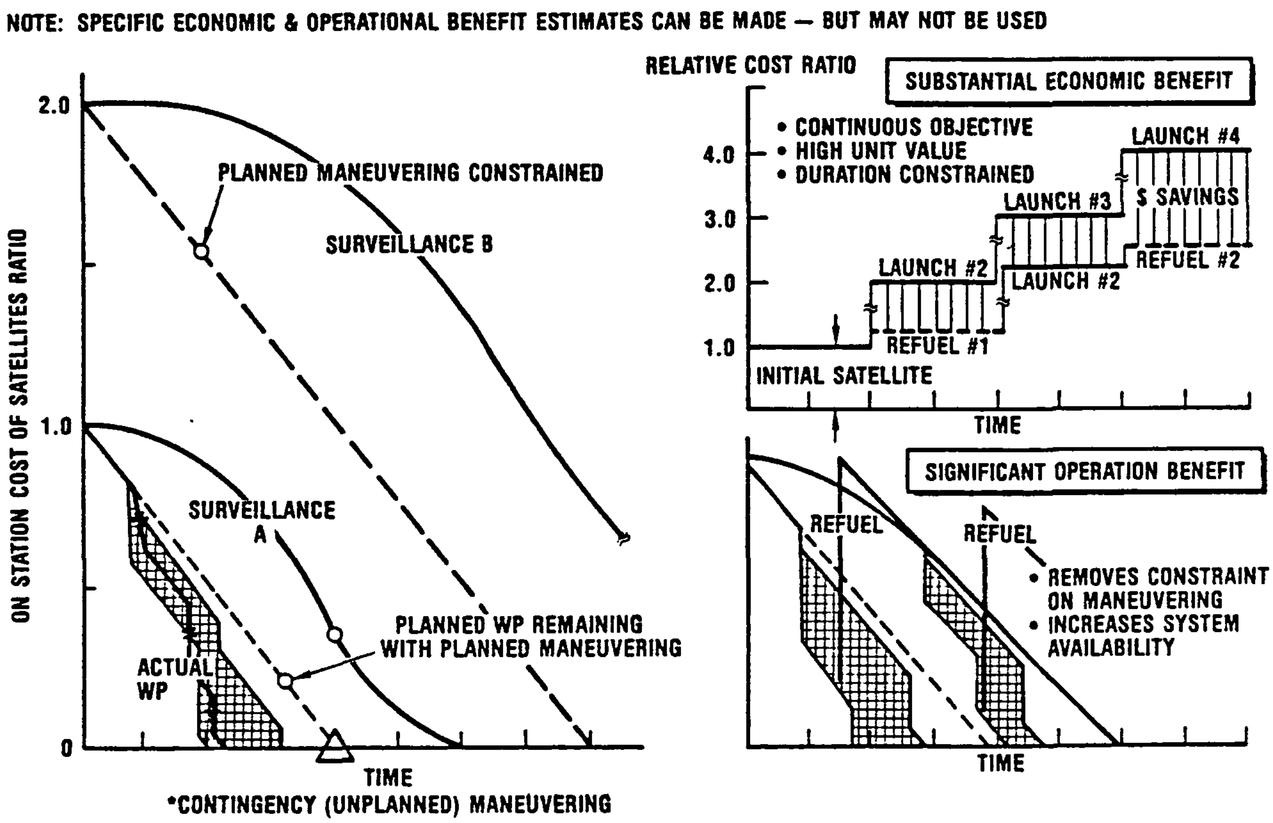


Figure 20. Potential Need of Surveillance Programs For On - Orbit Remote Consumables Resupply

Prime candidate spacecraft in low earth, high inclinastion orbits include selected DoD programs and, for superfluid helium transfer technology demonstration rather than for economic benefit motivations such as the SIRTF program as shown in Table 8.

PROGRAM (WF)	GROSS ECONOMIC BENEFIT	MAJOR OPERATIONAL BENEFIT	APPROXIMATE COST RESUPPLY SCENARIO				MAX NET ECONOMIC BENEFIT $84M
			1	2	7	8	
EOS 2,000 LB; N_2H_4	48		38	52	23	23	<28
DOD ? LB ?		MANEUVERING SURVIVABILITY					
GrPrB 400 LB; N_2H_4	42		28	44	12	12	<30
SIRTF 1000 LB; L He	56	TECHNOLOGY DEMONSTRATION	32	46	12	12	<44

NOTES:

(Estimates in $ 1984 Million)

1. SCENARIO 2 NEVER COST EFFECTIVE AT WTR
2. SCENARIOS 3, 4, 5, & 6 NOT FEASIBLE AT WTR
3. SCENARIOS 9, & 10 COST EFFECTIVE FOR GrPrB
4. DIFFERENT NODAL CROSSING TIMES MAY PRECLUDE MULTIPLE RESUPPLY UNLESS SPACECRAFT MANEUVERS

Table 8. Fluids Resupply (Without Maintenance) Benefits for Polar Orbits

Neither of the space-based OMV/RM alternate fluid resupply scenarios are thought to be technically feasible in high (polar) inclination orbits, due to the absence of high relative nodal regression rates which are required to achieve ascending node congruity between a space-based resupply carrier vehicle and the receiver spacecraft. Use of the ground-based OMV/RM mode for single resupply engagements is not cost-effective at WTR (Vandenburg AFB) due to the high transportation charge (approximately 250% of ETR) on the inert mass of the OMV/RM cargo unit and the technical feasibility of dual resupply engagements in high inclinations using a ground-based OMV is an unresolved issue.

Results of the parametric economic benefits analyses (Figure 21) indicate that NASA LEO ETR spacecraft programs could receive a net economic benefit by utilizing an Expendables Resupply capability, even without concurrent "full servicing". However, the marginal benefit to each of these NASA ETR representative programs which would be received by taking advantage of comprehensive repair, instrument changeout and fluids resupply is so substantial there will be a strong motivation to opt for Orbiter-based, man-in-loop repair until teleoperation maintenance provides routine remote servicing capability.

• RESUPPLY (W/O CONCURRENT MAINTENANCE & REPAIR) (+) NET BENEFIT

✓ OBSERVATORIES & EXPLORERS

- HIGH UNIT-VALUE SATELLITES
- LOW COST ACCESS FOR RESUPPLY

✓ COST-EFFECTIVE ONCE PER PROGRAM

SPACECRAFT	GROSS ECON BENEFIT $M84 RESUPPLY ONLY	COST TO RESUPPLY $M84	NET ECON BENEFIT $M84 RESUPPLY
AXAF	73-110	6-20	53-104
EUVE	23-34	7-21	2-27
GRO	65-98	15-29	36-83
LDR	144-216	9-43	101-207
PROTEUS	27-40	9-23	4-31
			196-452
			ONE RESUPPLY

• RESUPPLY WITH CONCURRENT MAINTENANCE & REPAIR (+) NET BENEFIT

✓ ORDER OF MAGNITUDE INCREASE IN BENEFITS

✓ "FULL SERVICE" WILL BE PREFERRED OPTION

SPACECRAFT	GROSS ECON BENEFIT $M84	COST TO RESUPPLY & REPAIR $M84	NET BENEFIT PER REPAIR MISSION $M84	NO ENGAGEMENTS PER PROGRAM	TOTAL PROGRAM NET ECONOMIC BENEFIT $M84
AXAF	219-329	23-80	139-306	4-6	558-1836
EUVE	29-43	8-14	15-35	1-2	15-70
GRO	220-330	32-87	133-298	2-4	267-1194
LDR	256-384	31-119	137-353	2-3	275-1060
PROTEUS	36-54	10-17	19-44	3-4	56-177
					1171-4337
					MULTIPLE ENGAGEMENTS

Figure 21. Net Economic Benefits Summary for NASA LEO ETR Spacecraft

Even though the value of full servicing--repair, resupply, technology changeout and routine maintenance--is limited to the spacecraft's replacement cost (second unit + replacement launch and insertion), the net economic benefits received from full servicing in major NASA ETR programs far exceed the benefits of Expendables Resupply alone. Since stand-alone resupply can be cost-effective only once during most programs, while on-orbit maintenance and repair can be cost-effective many times. Figure 22 presents the results of Figure 21 in the form of columns to more clearly illustrate the relative benefits of the two approaches (Resupply alone versus Resupply with Maintenance/Repair Servicing.)

GEO Satellite Resupply/Servicing Economics

Commercial operations in Earth orbit are primarily concentrated in Geosynchronous Earth Orbit (GEO), located approximately 19,500 nautical miles above the Earth's Equator. In this location, there are currently about 70 commercial communications satellites which provide television, telephone, telepgraph, data transmission, and other communications services world-wide. They originate from a vareity of countries and organizations, and represent an investment of several billions of dollars. Over the next decade the number of satellites in GEO will increase, until by the year 1990 it is estimated that there will be over 150 commercial satellites representing a direct investment of over 10 billion (thousand million) dollars in GEO.

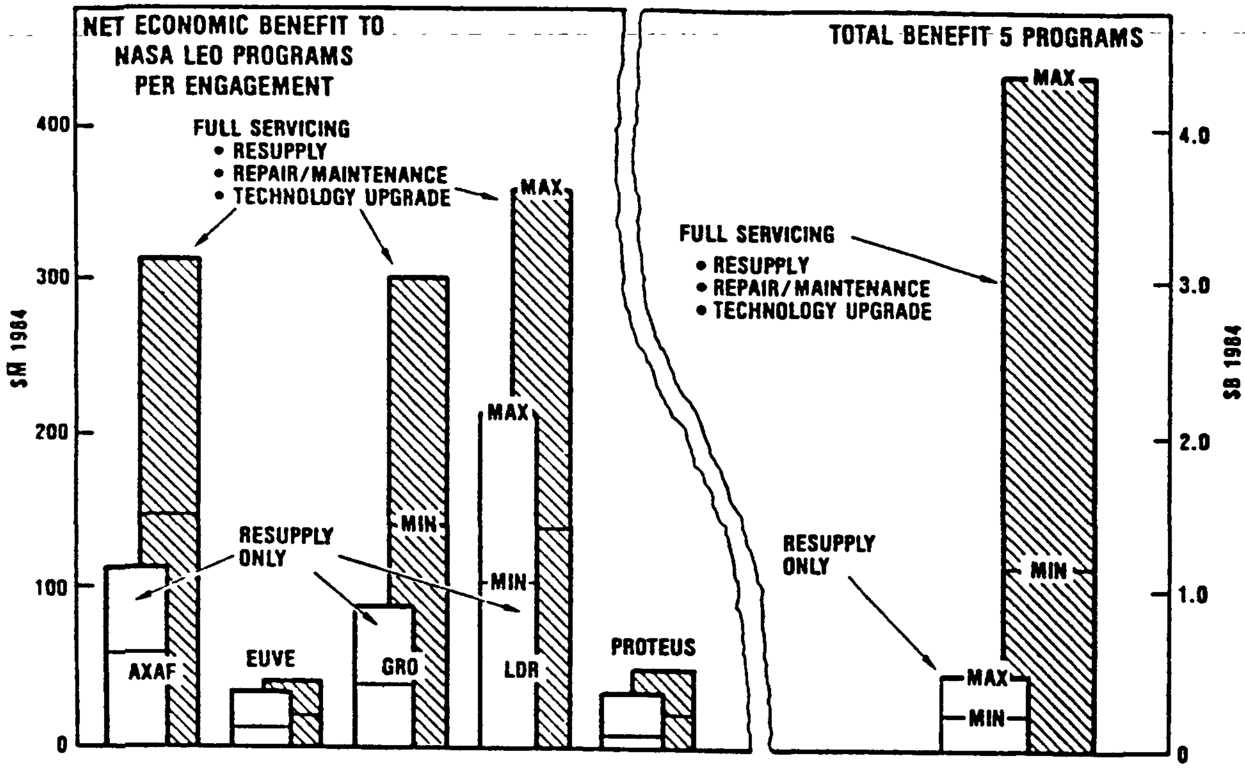


Figure 22. Economic Benefits of Full Servicing Relative to Resupply Only For Selected LEO Satellites.

A typical GEO satellite is designed to reliably provide service over a 7 to 10 year period, at which time it is retired from service by boosting its orbit several hundred miles into a "junk orbit" to free its valuable GEO location over a specific longitude, for a new user. Due to the high orbit, no effort are made to return the satellite for reburishment, nor are any methods currently available to repair a malfunctioning satellite.

The large amount of investment located in this orbit offers a unique opportunity, in that, within this one orbit is located the vast majority of current commercial investment in space. Interest has been expressed in the possibility of supplying services to GEO satellites; in particular to resupply them with attitude control system fuel (ACS) needed to maintain the orientation of the satellite and aim its antennas, and to replace or upgrade the electronics on board.

A commercial communications satellite venture makes its revenues from providing a service to customers; that is, it allows the usage of the GEO satellite's transponders for a fee. These service conracts may be for a very few hours, for several years, or the customer may buy an exclusive right to use the transponder for the expected life of the satellite.

To provide this service, the commercial satellite operator must first:

- establish financial backing for his venture,
- apply for and receive a permit to operate from the appropriate regulatory agency,
- book launch space on a launch vehicle,
- arrange financing for the launch of their satellite,
- arrange launch and operations insurance,
- establish an organization for monitoring and operating the satellite over its expected life.

Almost all of these milestones are performed in the private sector, and several private sector service or manufacturing operations are avalable to a new commercial satellite operator to accomplish these functions in each area. Only the regulatory function is being exclusively retained as a governmental responsibility. A GEO satellite is run as a business venture, and future operations will be governed by standard business decision analyses. The desirability of GEO servicing and resupply to a commercial user in the future will be determined by these decision methods.

GEO SATELLITE COSTS AND REVENUES

In order to assess the desirability of GEOsat servicing or resupply to a commercial satellite operator, it is first necessary to determine the expected costs and revenues they are likely to encounter. Parametric relationships were determined between fundamental characteristics of the GEOsats wherever possible in order to assess the sensitivity of these factors upon the user.

Satellite Production Costs

The initial cost of buying a GEO satellie is a function of several factors, the chief being the physical size of the satellite, whether or not it must be designed uniquely for a specific commercial customer, and the required lifetime of the GEO satellite. In this analysis, the basic satellite cost was taken to be:

$$\text{TFU cost} = 2.02 x m^{0.55} \quad (1)$$

where m = GEOsat BOL weight in pounds and TFU cost is in millions of 1984 $. This relationship is based upon a cost estmating relationship derived by D. E. Koelle in AIAA-84-0704, "GEO Space Platform Economics", and is in close agreement to ROM estimating relationships used internally. Application to Intelsat VI and DBSC satellite systems fitted quite well.

The actual cost to the user is further dependent upon the number of satellites built, and the development costs which went into constructing the first satellite of this type. A Design, Development, Test and Engineering (DDT&E) cost equal to 130% of the TFU cost was assumed to be incurred prior to the first satellite constructed, and then the DDT&E cost was spread evenly between all the satellites.

The experience of producing the GEO satellites will allow a reduction in the per satellite cost, dependent upon the number of satellites produced. Based upon quoted prices for Ford Aerospace's "Supersat" series, and the Hughes Satellite System's "Intelsat VI" series, an 80% learning curve was taken to be typical. Since orders for Hughes Satellite Systems "HS-376" series of satellites has reached 30, with Intelsat expected to buy from 5-14 of the Hughes "Intelsat VI" series satellites, and Ford Astronautics expected to build at least 5 of their "Supersat" series, a lot size of 10 satellites was taken to be typical, and the average cost for a 10 satellite batch was assumed.

Including DDT&E costs and learning curve effects into the satellite's procurement cost yields:

$$PC = (2.02\ m^{0.55})\ N^{(\ln(.8)/\ln(2))}\ [1+1.3/N] \qquad (2)$$

where N is the number of satellites over which the costs are averaged.

This relationship was primarily derived for current generation communications satellites, and does not include any sensitivity of the satellite cost to increased reliability for a longer satellite life. As the satellite's life on orbit increases, incorporating more reliable components and systems will increase the overall satellite costs. The relative variation in TFU cost as life increases was taken to be:

$$TFU' = TFU \times [0.15 + 0.85\ e^{(0.32L - 0.64)}] \qquad (3)$$

where L = the mean mission duration in years, and the 0.15 and 0.85 factors represent a 15% and 85% split between the non-electronic and electronic subsystems. This relationship was based upon a board and component level analysis of spacecraft subsystems during the TMS Benefit Assessment contract. It is documented in Voume II, of that contract report.

ACS Usage and Number of Transponders

Annual ACS usage was derived from the following data, obtained from several sources:

Satellite	ACS Fuel (lbs)	Lifetime (yrs)	S/C Wt (lbs)	Annual Usage
Galaxy (HS376)	300	9	1,145	33.3
HS299	139	10	725	5.2
ACTS	344	8	1,968	43.0
MSAT	494	10	2,909	49.4
TDRS	1,300	10	3,700	130.5
SBS	250	7	1,050	35.7
STC	305	7	1,105	44.0
BSGP Platform (GD Study)	1,764	7	7,092	252.0

From this data the trend line was determined to be

$$ACS = 15.6 + .036m \qquad (4)$$

where m = GEOsat BOL weight in lbs and ACS = annual usage of ACS fuel in lbs per year. See Figure 23.

However, the fraction of the GEO satellite that is occupied by the communications payload varies with the mass of the satellite. Using the estimating relation developed by Koelle, the following equation is best found to estimate the fraction of communications payload to total GEOsat BOL weight

$$x = -.012 + .032 \ln m \qquad (5)$$

where m is again the BOL weight in lbs. Thus the estimated weight of the communications payload is P, transponder plus antenna weight.

$$P = -.012m + .032m\ (\ln m) \qquad (6)$$

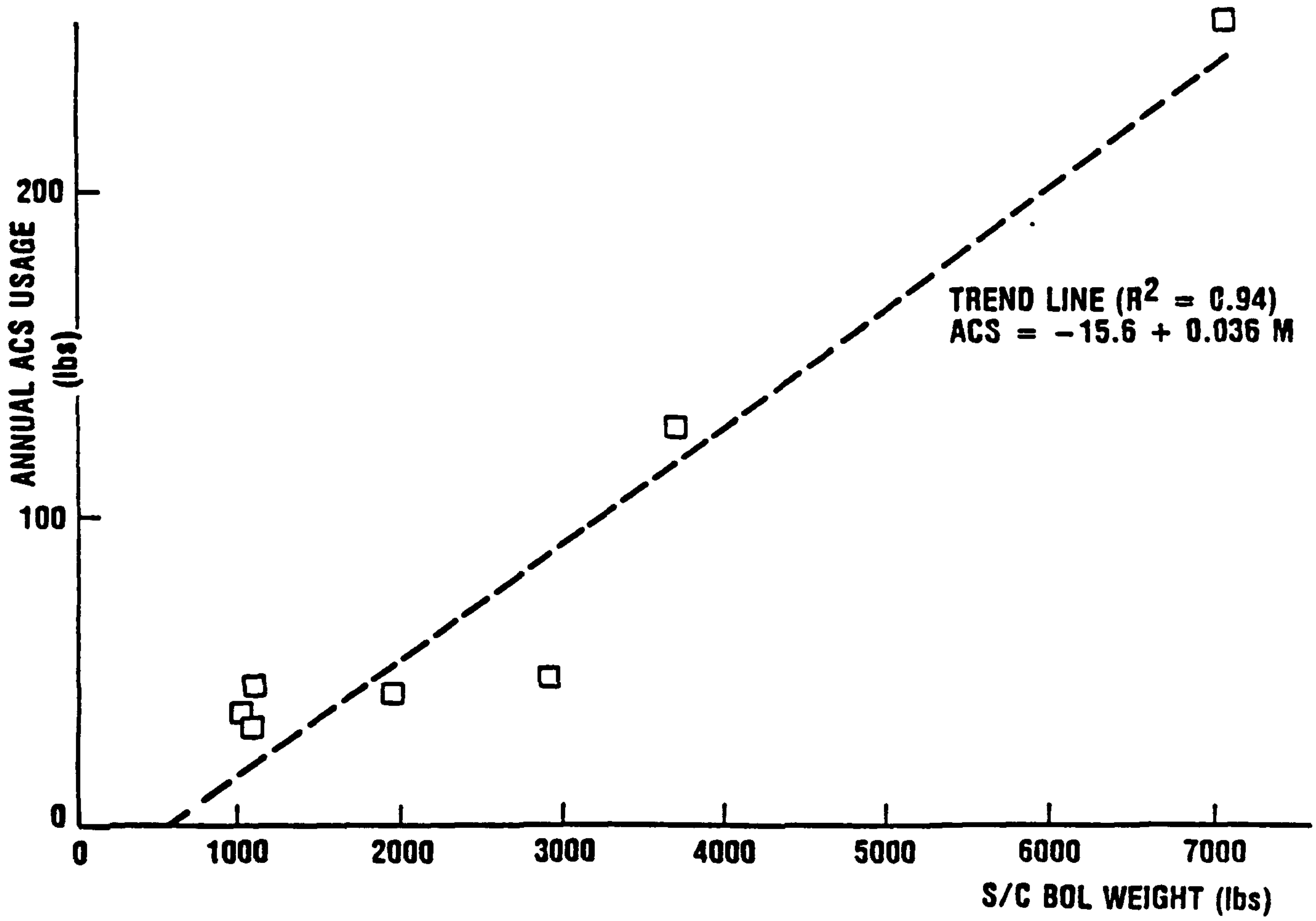


Figure 23. GEOSAT ACS ANNUAL USAGE

The actual number of transponders in the payload is again variable. However, to fully and precisely account for the number of transponders a very high level of detail is required including the frequency, number, type, bandwidth, and encoding scheme used on each channel. In the time available for this effort, the detailed data was not available and the "ROM" estimators of 4/6 GH_z (C bands) transponders = 1 equivalent, 12/14 GH_z (K_u Band) = 2.5 equivalent, and 30/20 GH_z (K_A Band) = 5 equivalent transponders were used. Thus we find

Sat	No. Transponders	Equivalent Transponders	BOL wt. (lbs)	Comm. P/L WT (lbs)
ABCS	$16K_u$	40	2,400	569
Amersat	$18C + 6K_u$	33	1,380	303
Fordsat	$24C + 24K_u$	84	3,300	816
Galaxy	24C	24	1,146	245
Gstar	$16K_u$	40	1,380	303
HS399	12C	12	725	144
SatcomK	$16K_u$	40	1,463	324
Spacenet	$18C + 6K_u$	33	1,260	273
Telstar	24C	24	1,435	317
ECS	$12K_u$	30	1,345	294
Telecom·	$5C + 6K_u$	20	1,150	246
Anik C	$16K_u$	40	1,250	270
Anik B	$12C + 6K_u$	27	1,014	212
Anik D	24C	24	1,350	295

which may be best approximated by the relationship

$$N = 4.197 + .089P$$

(7)

where N = number of equivalent transponders, and P = communicatons payload mass in lbs. (see Figure 24)

Satellite Revenues and Technological Obsolescence

Revenues from each GEOsat are dependent upon the number of transponders on the GEOsat, and the price charged per hour or usage. Based upon quoted rates for Satcom IV and Telstar 3 services, a rate of $200 per hour over a period of time is typical. For short term leases of transponder facilities, this rate will be much higher, but the proportion of time that the transponder is actually in use may be much lower, depending on market conditions.

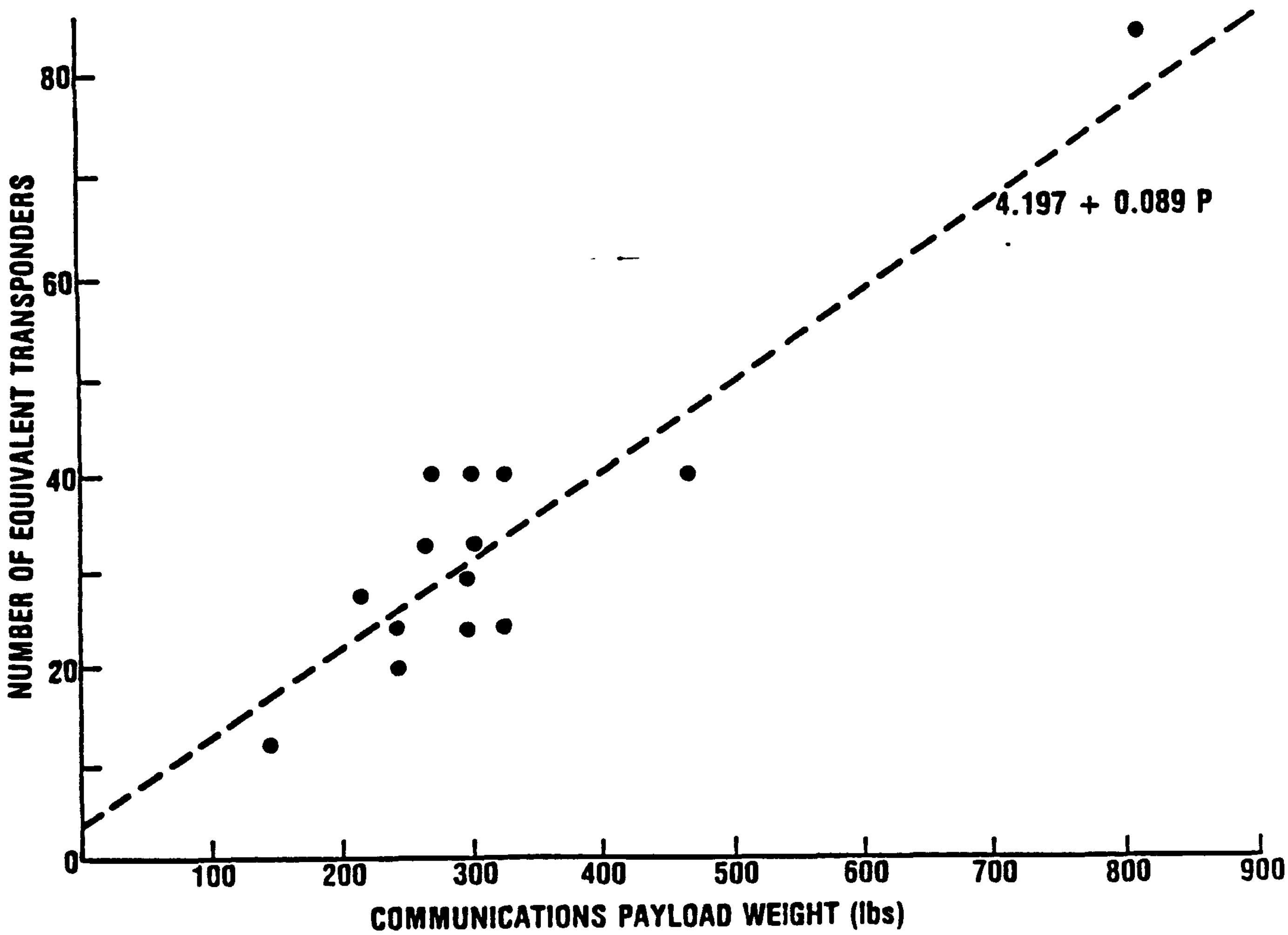


Figure 24. Equivalent GEOSAT Transponders as a Function of Communications Payload Weight

The actual level of reimbursement for usage of each transponder is determined by the supply and demand for transponder services in the maketplace, and the individual capabilities of each transponder. To get a workable answer without digressing into a treatise, the supply and demand levels for transponder services are assumed to be equal, maintaining the price per transponder hour at a constant level. Excess capabilitity is assumed to be filled by a growing demand for GEO satellite services. Thus, without changes in the capability of a transponder (a constant technology level), the revenues generate, and the competitive position of a satellite will remain constant in the market.

If an "improved" transponder, possessing a higher throughout capability than current technology transponders, is placed in GEO and its services are assumed to be offered at the market rate of $200 per hour, it will attract customers from existing systems. This will force a cut in the market price of the existing transponders to match capabilities to market price.

Historical changes in capability per transponder and future projections are shown in Figure 25. Overall, this represents about a 12% annual growth in transponder capability. Will this rate decline? Probably so; the most recent growth in capability has been at about 6% annually, but recent developments indicate there is still quite a bit of improvement achievable in GEOsats in the future. This analysis assumes the historical rate of improvement in transponder capabilities will continue.

If the asking price for transponders is assumed to remain constant, this rate of improvement may be taken to be a technological obsolescence factor. In other words, suppose a satellite operator launches a GEOsat in June, 1984 and charges $200 per hour for use of its transponders. If another GEOsat operator else puts June 1985 technology in their satellite and launches it, while charging the marke price of $200 per hour, the June 1985 GEOsat operator will be supplying (based on past trends) about 12% more capability for the same price. If the first operator doesn't take action, all their customers wil switch to the new service to achieve more capability at the same price. To keep them on board the first operator will have to discount their asking price by 12% to match offered capabilities to market price. If someone else launches a June 1986 satellite, it will have an additonal 12% of capability, and the June 1984 satellite operator will have to again discount their GEOsat services price by another 12% to match.

As time goes on, a satellite in GEO will continue to generate revenues, but the rate at which the revenues are generated must be discounted to account for more efficient competitors. Obviously, year by year changes are almost impossible to predice while the overall effect averages out with longer times.

Mathematically, this is represented as

$$\text{Total Revenues} = \sum_{n=1}^{L} \frac{R_o}{(1+r_1)^n} \qquad (8)$$

where R_o = initial revenues

= (# transponders) ($200/hr) (8750 hours/year)

r_T = annual obsolescence rate

L = life of satellite in years

since this process may be generalized to be a continuous process

$$\text{Total Revenues} = \int_{t=0}^{L} R_o e^{-r_c t} dt$$

where r_c = continuous discount rate

$r_c = \ln (1 + r_T)$

if $r_T = 12\%$ then $r_c = 11.33\%$

Solving,

$$\text{Total Revenues} = \frac{R_o}{R_c} (1 - e^{-r_c L}) \qquad (9)$$

Note that to derive total revenues, one must only know the BOL weight of the GEOsat. Knowing this, one may find

- the communications payload weight (eqn 6)
- the number of transponders (eqn 7)
- the total revenues over the satellite life (eqn 9)

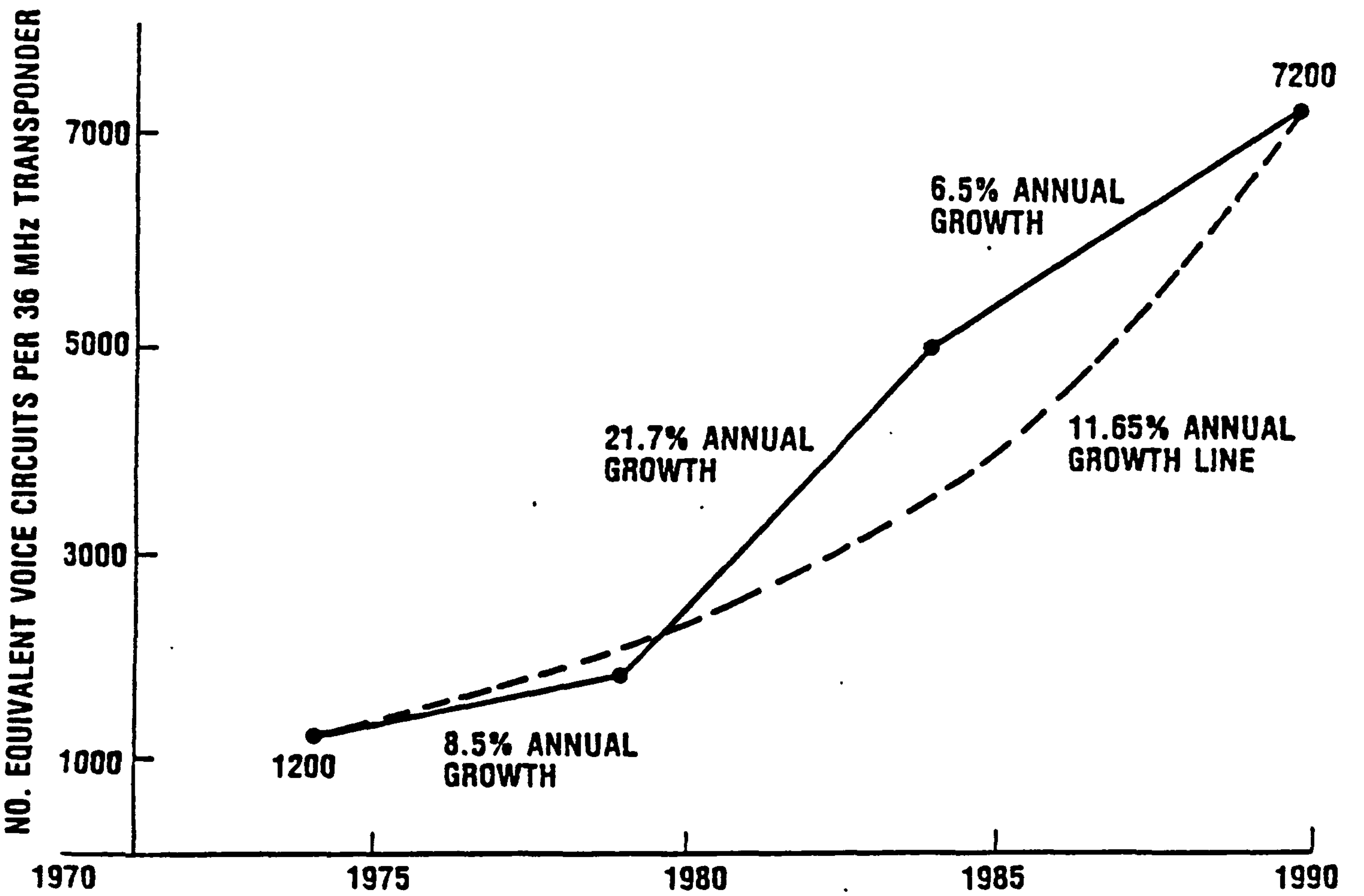


Figure 25. Transponder Capability Growth

Total Costs and Revenues

Total costs to operate the satellite over its lifetime are comprised of TFU (production cost), operations, insurance and transportation. An evaluation of TFU cost and its sensitivity to satellite lifetime is given in eqns 1 and 2. Operations cost may be taken to $2 million per year, (from typical current systems) so total operations cost is 2L. Insurace may be assumed to be

$$I = r_i \text{ (TFU + Transp.)} \qquad (10)$$

for just replacement value. The insurace rate, r_i, is around 10% (although current rates are higher due to recent upper stage failures).

An estimator for transportation price was obtained from the data in Figure 26 which summarizes the results of a previous analysis which compared the total transportation costs to the user (excluding insurance) for various space launch systems. This incorporates launch costs, payload integration and processing costs and the procurement of an upper stage system if required.

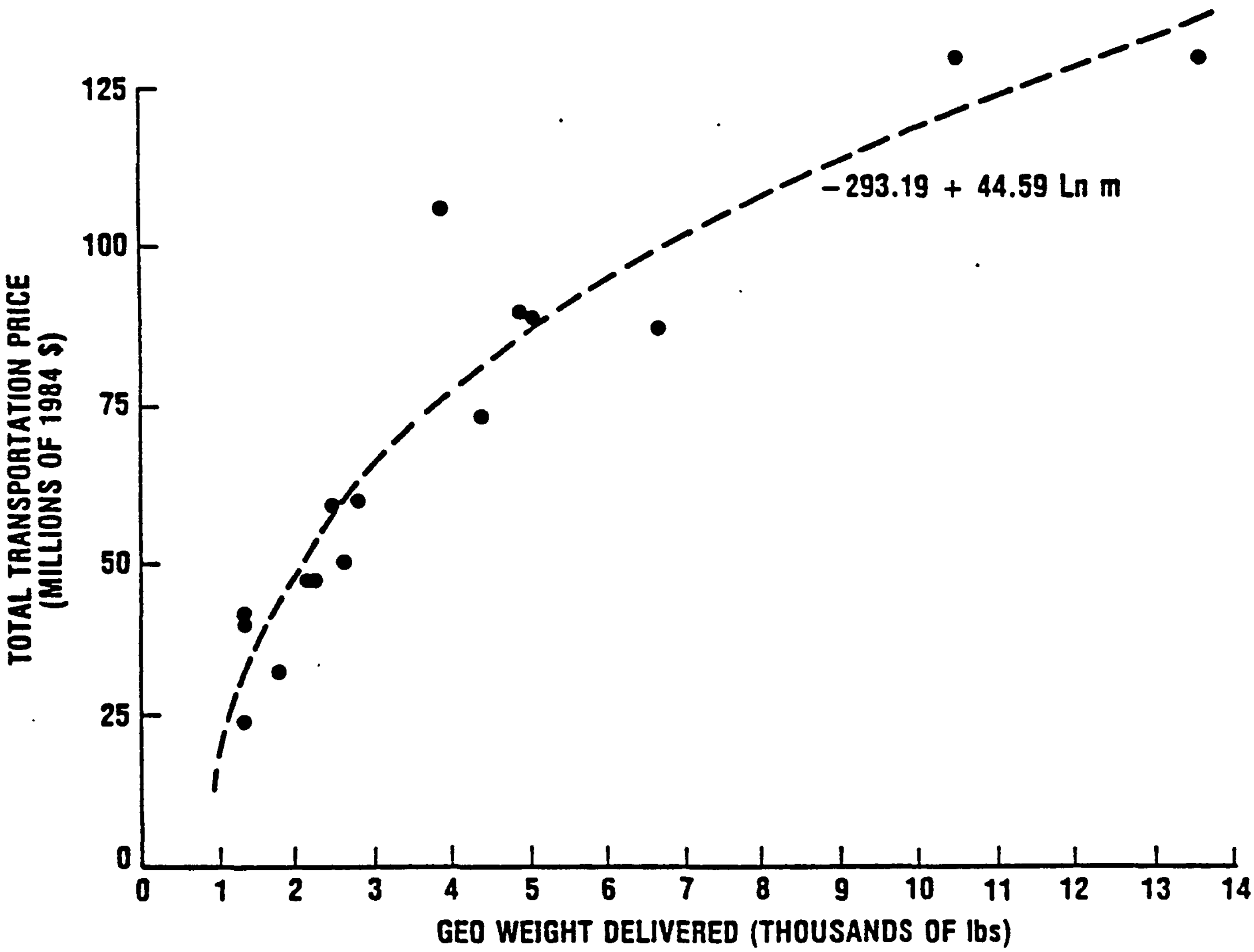


Figure 26. Transportation Price As A Function Of GEOSAT Weight

Transportation price was derived from the following data:

Systems	GEO Wt. (lbs)	Transportation Price (1984 $)
Ariane - 3 (single)	2,800	60.3
(dual)	1,350	35.5
Ariane - 4 (single)	5,130	85.2
(dual)	2.340	47.3
Atlas/Centaur	2,500	61.5
Atlas-G/Centaur	2,600	50.5
Atlas II/Centaur	6,700	88.7
Delta 3920	1,400	41.4
Titan 34D/Transtage	3,950	106.5
STS/Centuar F	13,600	129.4
STS/Centuar G	10,600	129.4
STS/IUS-1 (RI)	5,000	90.1
STS/IUS-1 (Hughes)	4,410	72.0
STS/PAM-A	2,200	45.9
STS/PAM-D	1,350	23.7
STS/PAM-D2	1,800	31.6

This data may be approximated by

$$\text{Transp} = -293.19 + 44.69 \ln m \qquad (11)$$

Thus, the toal cost of launching and operating a GEOsat is found to be:

$$\begin{aligned}\text{Cost} &= \text{TFU} + \text{Ops} + \text{Insur} + \text{Transp}.\\ &= 2.02m^{0.55} + 2L + .1(2.02m^{0.55} - 293.19 + 44.69 \ln m)\\ &\quad -293.19 + 44.69 \ln m\\ &= 2.22m^{0.55} + 2L + 49.16 \ln m - 322.51 \qquad (12)\end{aligned}$$

where m = GEO sat BOL weight in pounds, and L = lifetime in years. If we want to service a satellite at GEO, we need to also consider how the costs and revenues increase. Without technological updating of the GEOsat, the total revenues may be found using eqn 9, and just substiuting the new satellite lifetime L for the life of the satellite. The production costs change substantially, however. As before,

$$\text{Cost} = \text{TFU}' + \text{Ops}' + \text{Insur}' + \text{Transp}' \qquad (13)$$

TFU cost increases since there may be a longer life required for the satellite. This may be derived from Eqn (3). The increased TFU cost, TFU', may be expressed as

$$\begin{aligned}\text{TFU}' &= \text{TFU} \frac{\left[0.15 + 0.85e^{(0.32L' - 0.64)}\right]}{\left[0.15 + 0.85e^{(0.32(10) - 0.64)}\right]}\\ &= \text{TFU}\,(.0897)\left(0.15 + e^{(0.32L' - 0.64)}\right]\\ &= \text{TFU}\left[.0135 + .0763\, e^{(0.32L' - 0.64)}\right] \qquad (14)\end{aligned}$$

Operations cost is much simpler

$$\text{Ops}' = 2L'$$

Transportation cost for a resupply may be taken to be

$$\text{Transp}' = \text{Transp} + \text{Resupply} \qquad (15)$$

Where Transp is the transportaton cost from equation (11) and Resupply cost is

Resupply = (annual fuel usage)(Years resupplied)(Cost/lbs)

Since resupply may be made to seveal GEOsats on one mission, we cannot use equ (11) here, and shold use the minimum relasitic cost per pound to deliver fuel in GEO. This derivation is shown below:

	Weights (lbs)	Cost (1984 $M
Payload to GEO (full STS flight required)	13,800	84M
Centaur G'		40M
OMV (assumed 20 flights into 25 flight life)	(3,500)	10M
OMV fuel	(1,000)	
ERM	(1,000)	5M
Tankset	(1,000)	5M
Total GEO fuel	7,300 lbs	
Total cost		$154M

The minimum cost of resupplied fuel in GEO is then about $21,100/lb, and eqn (15) becomes (in millions)

$$\text{Transp}' = -293.19 + 44.69 \ln m + A(L' - L)\,(.021) \qquad (16)$$

where A is the annual fuel usage from eqn (3)

$$\text{Transp}' = 293.19 + 44.69 \ln m + (6.45 + 0.21m)\,(L' - L)\,(.021) \qquad (17)$$

The GEO fuel weight of 7,300 lbs is the total mass installed in a GEO satellite. The weight allocation may be used for some combination of fuel and replacement parts, rather then fuel alone. This needs to be considered when performing concurrent servicing with resupply as discussed later.

Insurance is a bit more complicated. The insurance will have to include a premimum on replacement cost, the resupply cost, and possibly expected future revenues. Ignoring future revenue insurance, the insurance cost would be

$$\begin{aligned}\text{Insur}' &= r_I\,(\text{TFU}' + \text{Transp}') \qquad (18)\\ &= .1\,(\text{TFU}' + \text{Transp}')\end{aligned}$$

where Transp' and TFU' are defined from eqns (17) and (14).
Combining all these equations, the resupplied satellited system's costs are

$$\begin{aligned}\text{Cost} &= (1.1)\text{TFU}\,[.0135 + .0763\,e^{(.32L' - .64)}]\\ &\quad + 2L' - 293.19 + 44.69 \ln m\\ &\quad + (6.45 + .021m)\,(L' - L)\,(.021)\,(1.1)\\ &= 2.22m^{.55}\,[.0135 + .0763e^{(.32L' - .64)}\\ &\quad + 2\,L' - 293.19 + 44.69 \ln m\\ &\quad + (.149 + .0005\,m)(L' - L) \qquad (19)\end{aligned}$$

This is a somewhat complicated equation, but it only needs GEOsat BOL mass m, extended lifetime of the satellite L', and normal satellie life = L (usually, L = 10 years), to determine GEOsat mission cost.

The total revenues achieved from this operation are found to be:

$$\text{Revenues} = \sum_{n=1}^{L} \frac{1.75(4.197 + .089[-.012 + .032m + .032m(\ln m)]}{(1 + r_T)^n} \qquad (20)$$

The total return from the satellite is

Total Return = Revenues - Cost (21)

and this may be calculated from the above equations. However, this formulation does not include any recognition of the time value of money. Net Present Value recognizes this effect, and may be found from

$$NPV = \sum_{n=1}^{L} Cn/(1+k)^n - I_o \quad (22)$$

where C_i = is the annual cash flow from the project (usually the revenues less expenses), I_0 is the initial investment, and k is the appropriate opportunity cost of capital (here taken to be 20%).

To compare projects on a percentage basis, the IRR or "Internal Rate of Return" on each project may be found from:

$$0 = \sum_{n=1}^{L} \frac{C_n}{(1+IRR)^n} - I_o \quad (23)$$

Projects with a higher IRR than their alternates should be chosen.

IMPACT OF GEO RESUPPLY AND SERVICING

When the GEOsat is merely resupplied, there will be no technological updating of the satellite's electronics, and the market position of the GEOsat will continue to decline after the satellite resupply. Revenues will be earned, but the stream will continue to decline with time. Extra cost may be incurred if increased reliability is added to the satellite before launch to ensure that it will continue to operate past its initial lifetime, through the period for which it is resupplied.

If GEO servicing is performed upon the GEOsat, then it may be assumed that the technological obsolescence in the satellite's communications payload wil be eliminated at that time, and the revenue stream earned by the GEOsat will return to the market position. There will be an increased cost of performing the the mission, as extra weight will be carried into GEO and the required electronics boxes will cost more than that of the fuel alone. However, the increased costs of including extra reliability to ensure a longer satellite lifetime will be avoided, as the GEO servicing capability should also allow replacement of any failed subsystems, returning the system to the same reliability as when it was launched.

Three basic methods of operating a GEO satellite system with GEO servicing and/or resupply were considered:

Lifetime Extension - was considered to be the extension of an exisiting GEO satellite's life via GEO resupply. This in turn may be broken down into several variations including:

- Planned Resupply: the GEOsat has been built with the extra redundancy and reliability needed for the longer service life.
- Opportunity Resupply: a GEOsat not built with the extra redundancy and reliability for extended service life is found to be operating satisfactorily; and is resupplied in GEO to extend its life.
- Resupply and Concurrent Servicing: besides resupply of ACS to extend the GEOsat's lifetime, the electronics in the communications payload are updated to remove the technological obsolescence which has occured since launch or the last servicing.

Graphically, these modes are shown in Figure 27. The Internal Rates of Return (IRR) for these modes are shown below and graphically in Figure 28.

Geosat (lbs)	Baseline	Planned Resupply	Opportunity Resupply	Resupply Service
1000	16.7%	2.5%	17.3%	20.5%
2000	24.8	8.8	25.3	27.8
5000	52.0	25.3	52.1	53.1
7000	67.5	40.0	67.5	68.1
10000	87.9	45.1	87.9	88.1
15000	118.1	61.1	118.0	118.1
20000	114.6	75.1	144.5	144.6

Since the favored project has the highest IRR, planned resupply is seen to be not preferred over the alternatives. The highest IRR is obtained from resupply and service.

IRR (Internal Rate of Return) calculations do not always give an accurate assessment for projects of different durations. IRR implicitly discounts cash flows occurring in the future and tends to underestimate a longer term project compared to a shorter term. To resolve this, a comparison of the "Equivalent Annual Annuity" for each project was made as shown on Figure 29.

This was done by first calculating the Net Present Value (NPV) of he project using

$$NPV = \sum_{t=0}^{L} \frac{C_t}{(1+r)^t} = C_0 + \frac{C_1}{(1+r)} + \frac{C_2}{(1+r)^2} + \ldots\ldots + \frac{C_L}{(1+r)^L}$$

where C_t = the total cash flow (revenues less costs) in each year, and the cash flows are discounted at r = Weighted Average Cost of Capital for the project (This may be thought of as the equivalent interest rate earned by this money in alternative investments plus the required profit margin in the company). A company should take only projects with a positive NPV (i.e., those that earn more than investing in riskless securities and meet minimum profitability levels to cover their riskiness).

However, NPV as such does not easily compare projects of unequal duration. Let us assume profitable projects will be repeated at the end of their life (it's reasonable to assume that a profitable commercial comsat will be replaced at the end of its lifetime).

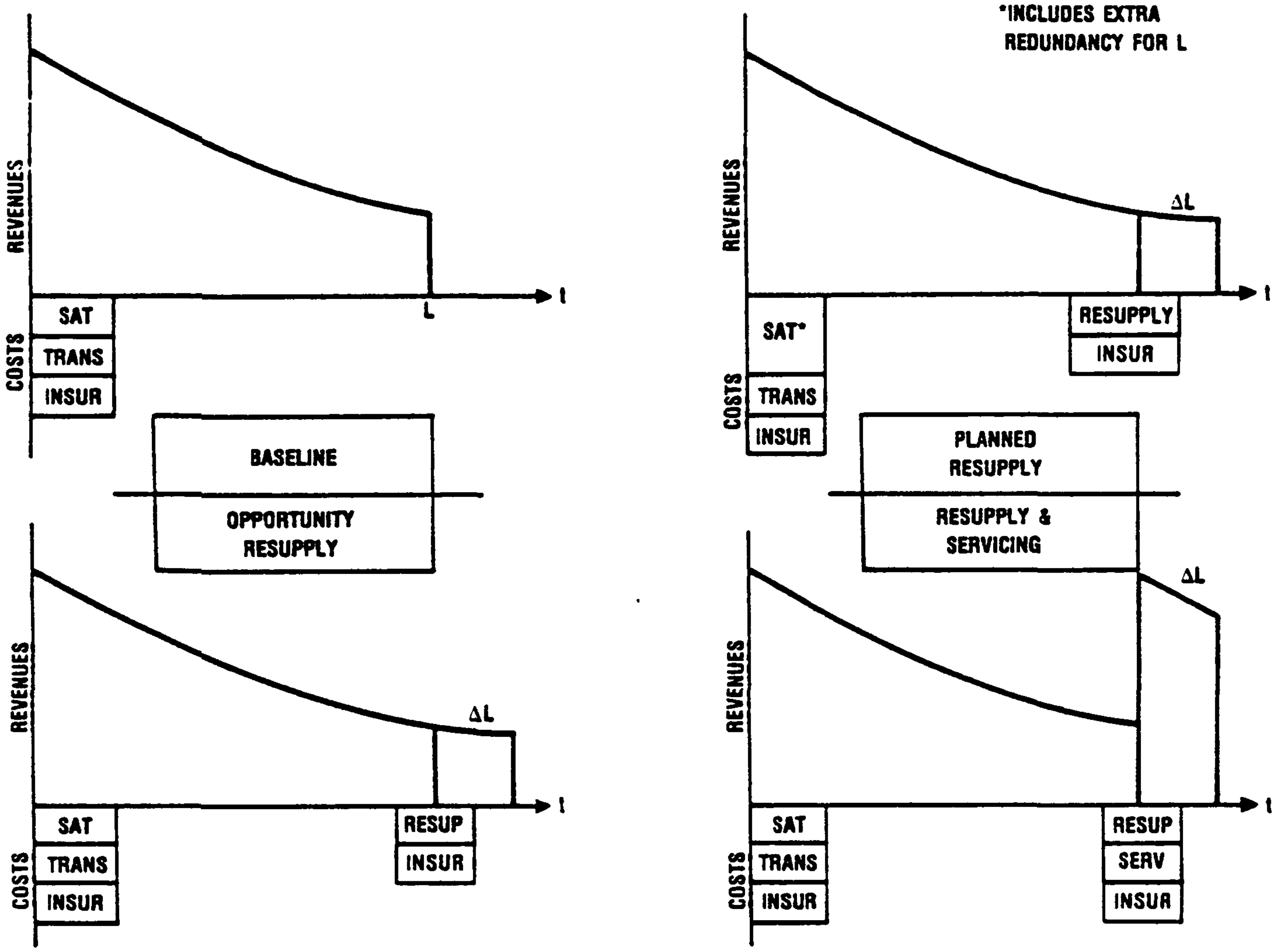


Figure 27. GEO Comsat Lifetime Extension

A NPV of say, "x" (NPV = X), may be considered to be equal to a constant annuity of A. That is, having an NPV = X is the same as earning A dollars per year each year the project is run. The equivalent annual anuity is found from

$$NPV = \sum_{t=1}^{L} \frac{A}{(1+r)^t} = A\sum_{t=1}^{L} \frac{1}{(1+r)^t} = A\left[\frac{1}{r}\left(1 \quad \frac{-1}{(1+r)^t}\right)\right]$$

$$A = NPV \Big/ \frac{1}{r}\left(1 - 1/(1+r)^t\right)$$

This allows the comparison of projects of different duration. Projects with a higher constant annual cash flow are preferred.

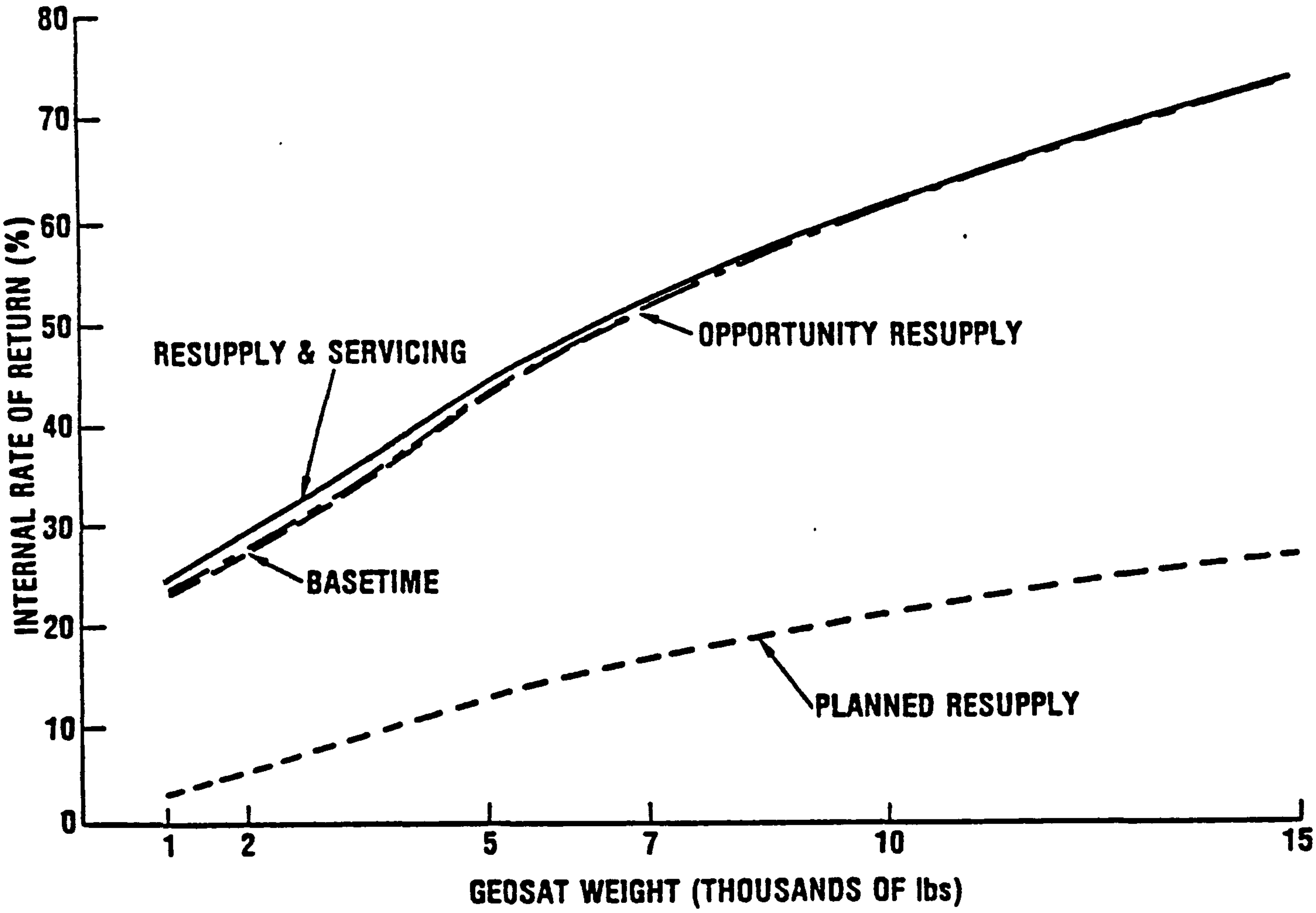


Figure 28. IRR of Lifetime Extension

Figure 30 is an extension of the material of Figure 29 modified to indicate an example of the impact of various penalties for including servicing/resupply capability into the satellite design. Three basic cases were examined in this analysis and the data was linearly extrapolated over the range of satellites examined.

The data used was determined by two data points in each model:

MODEL	Geosat Wt. (LBS)	Wt. Impact (%)	Geosat Wt. LBS	Wt. Impact (%)
Low	3,000	5	5,000	2.5
Medium	3,000	10	5,000	5
High	3,000	20	5,000	10

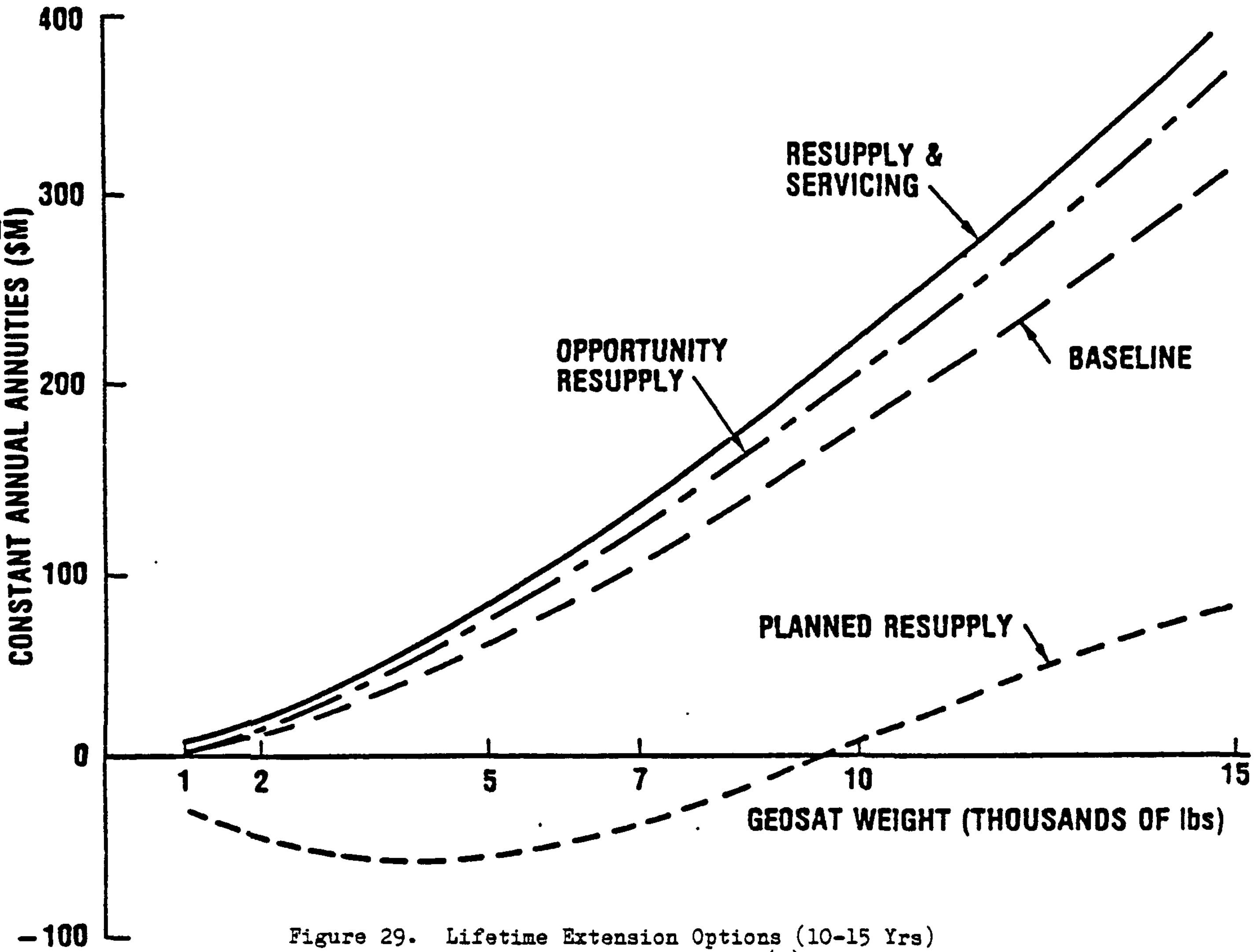


Figure 29. Lifetime Extension Options (10-15 Yrs)
Relative Annual Annuities (A)

Using the constant annual annuity method of comparison, the following data was generated, comparing the annuity of these three options to the two cases of the baseline 10 year non-serviced, non-resupplied GEOsat case and the 15 year nonserviced, but resupplied GEOsat (these satellites are designed with extra reliability to last 15 years).

WEIGHT	LOW CASE	MEDIUM CASE	HIGH CASE	BASELINE (10 YEARS)	RESUPPLY (15 YEARS)
1,000	1.5	-4.0	-14.2	-10.7	-158
2,000	46.7	41.6	32.3	12.6	-279
5,000	330.6	328.4	323.9	223.1	-334
7,000	567.8	567.8	567.8	407.3	-271
10,000	952.4	952.4	952.4	713.5	-62

What this analysis indicates is that the penalties of designing the satellite for GEO servicing is not a major consideration to the user in comparison to

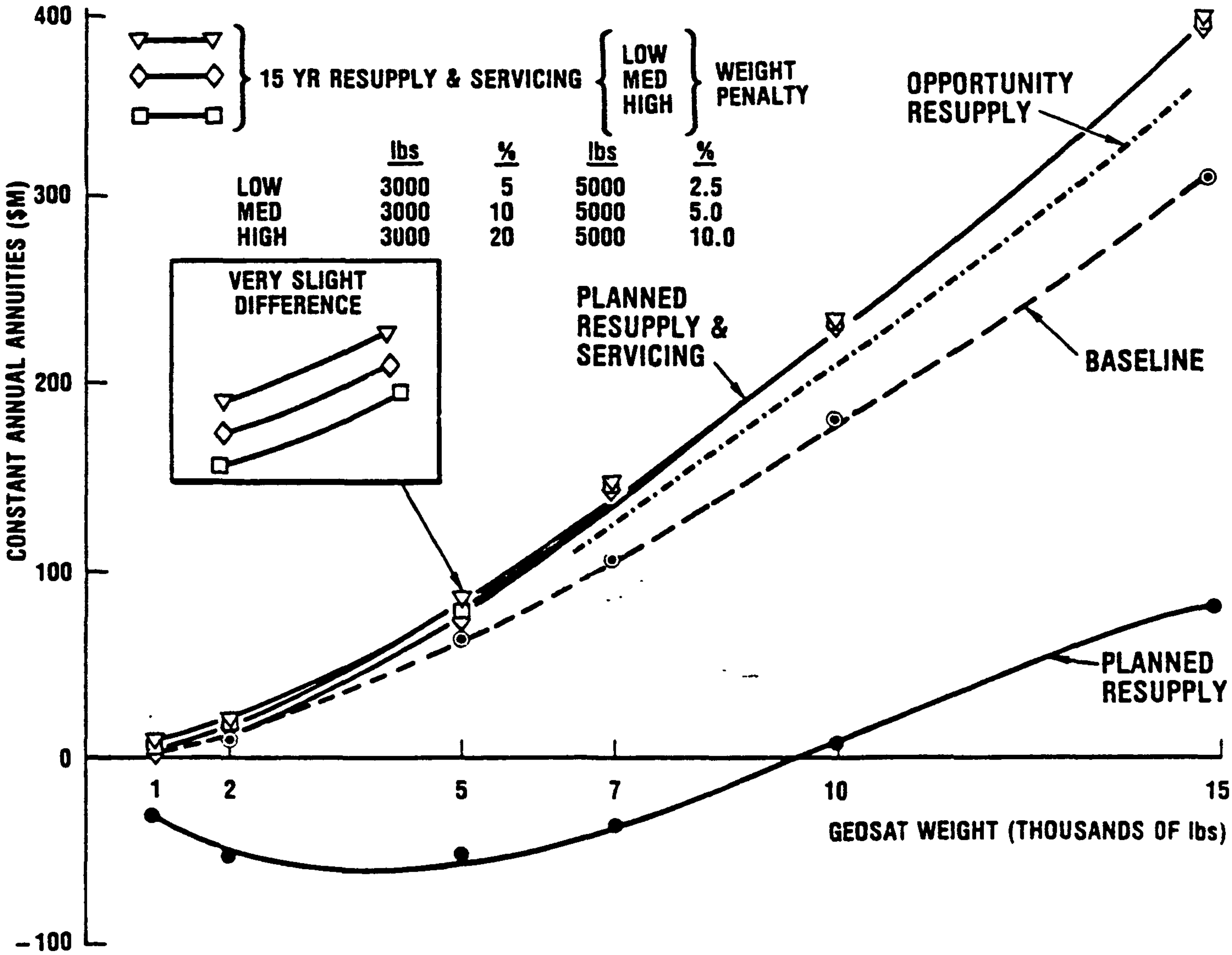


Figure 30. Lifetime Extension Options (10-15 Yrs)
Relative Annual Annuities (B)

the penalty of designing the exta redundancy into the satellite to increase its life by 50%. Under the assumptions used here, the most penalties are imposed upon the smaller GEO satellites in order to make them serviceable.

<u>Contingency Resupply</u> - was considered to be the resupply of ACS fuel to a GEOsat whose fuel has been deplete through a failure, in comparison to the replacement of this satellite from the ground. Assuming a rapid, unplanned depletion of fuel at the end of the first year, a GEO-based resupply system should be be able to rendezvous and resupply the depleted satellite within 90 days with enough fuel to complete its 10 year life. If the satellite was to be replaced, it was estimated it would take about 18 months to checkout, integrate, and launch a ground spare which would then have a 10 year operational life. This is shown in Figure 31.

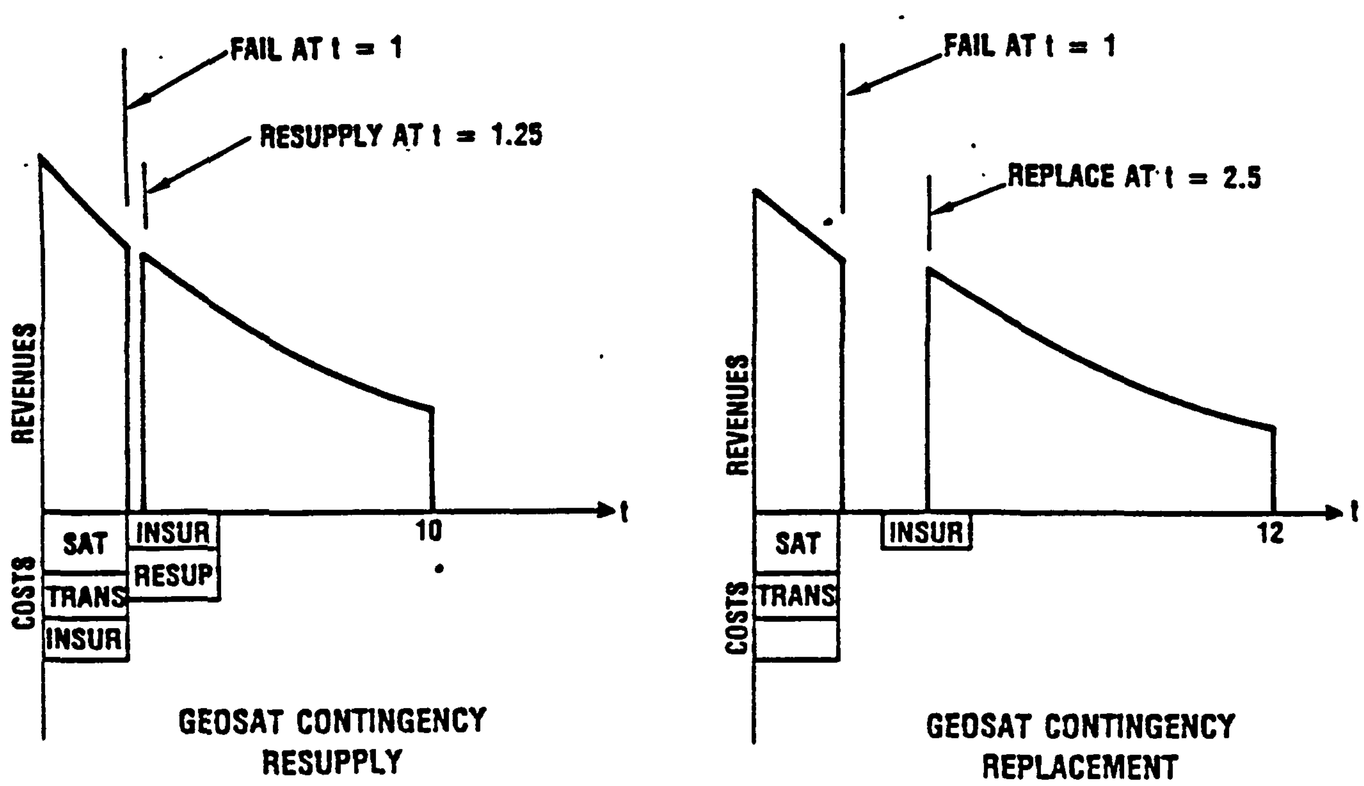


Figure 31. GEOSAT Contingency Resupply Versus Replacement

Comparing these two projects, the IRR is found to be:

GEOsat Weight	Contingency Resupply	Contingency Replace
1,000	12.3%	8.6%
2,000	17.6	13.4
5,000	31.0	24.7
7,000	37.6	30.2
10,000	45.6	36.9
15,000	56.0	45.7
20,000	64.2	52.9

This is plotted graphically in Figure 32.

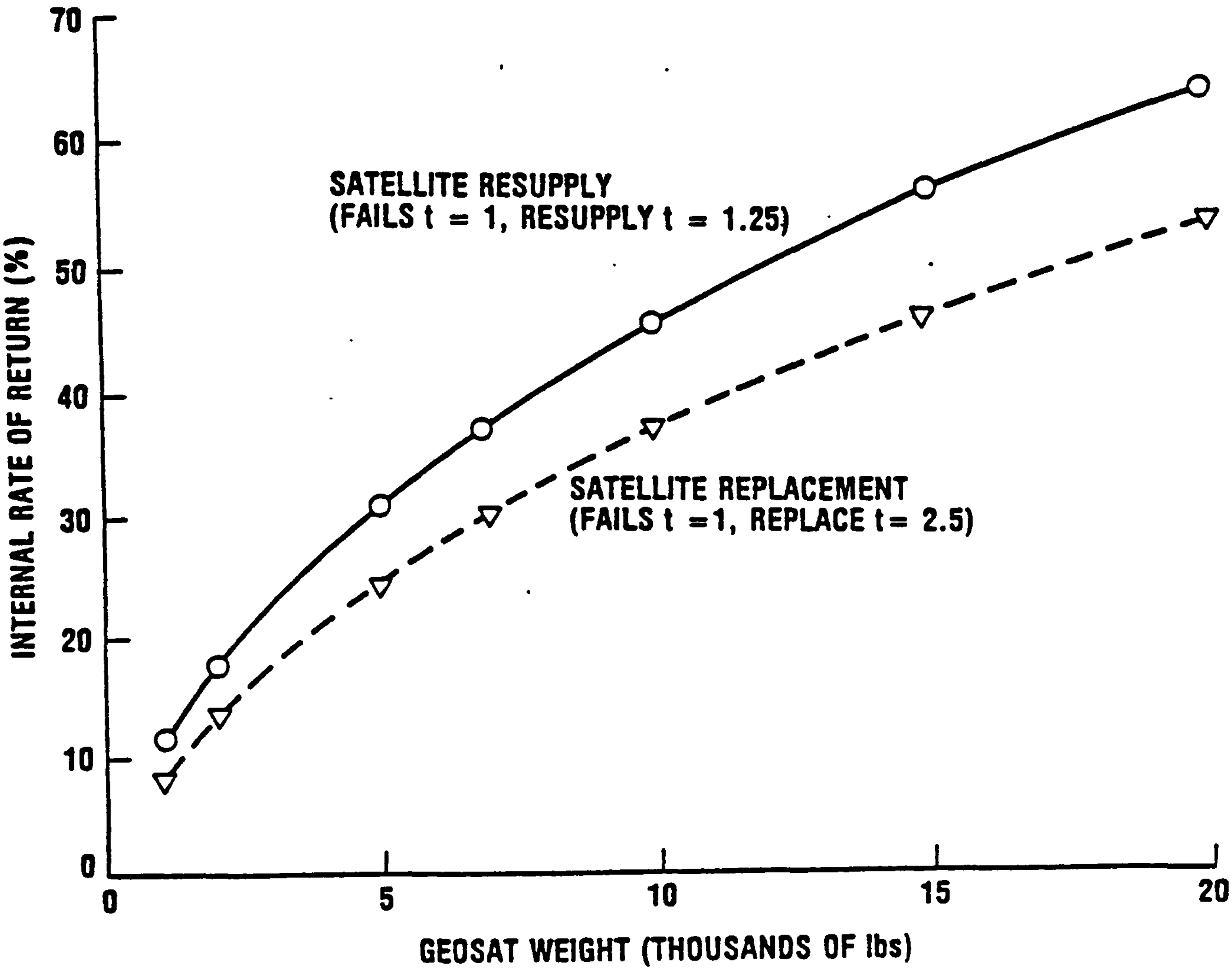


Figure 32. Internal Rate of Return GEOSAT Contingency Resupply Versus Replacement

Transponder Replacement

Transponder replacement refers to the potential of replacing some portion of the on-board ACS fuel with extra transponders. In this analysis, 60% of the ACS fuel (equivalent to 6 years life) was replaced with transponders. Resupply of the ACS took place at t = 3 and t = 6 years. The cost of the satellite was increased to reflect the increased number of transponders. If that 25 transponders are added to a 40 transponder satellite (and realizing 85% of a GEOsat typically arises from electronics costs), the GEOsat cost would increase to

$$\text{Sat cost} = (\text{orig. sat cost})\ (1 + .85(25/40))$$

Insurance coverage was increased to reflect this riskier operation mode. Besides reflecting TFU and transportation price, it must reflect the possibility that the revenues from years 4 - 10 may not be gained if the resupply operation fails. An insurance rate of 10% (reflecting a 90% overall probability of resupply success) was used. Each time a resupply operation was performed, this insurance factor was applied to recognize the risk to future revenues. Thus for the first resupply, insurance will equal:

$$\text{Insur}\ (t = 3) = .1\ (R_4 + R_5 + R_6 + R_7 + R_8 + R_9 + R_{10})$$

where R_i is the expected revenue in year i.

In comparison to the baseline (no service, no resupply) case, the following IRR's were obtained:

GEOsat Wt (lbs)	Baseline	Transponder Replacement
1,000	16.7%	15.9%
2,000	24.8	23.6
5,000	52.0	45.8
7,000	67.5	58.2
10,000	118.1	97.4
20,000	144.6	117.8

These are plotted graphically in Figure 34 (shown schematically in Figure 33).

RESULTS

Lifetime Extension

Figure 29 indicates the results of the analysis for the various lifetime extension options listed above. The comparison is done using the constant annuity method of comparison, due to the 15 year satellite life with lifetime extension, in comparison to the 10 year baseline satellite life. This indicates that:

- planned resupply does not offer increassed benefits to the user over the baseline case. This is due to the required extra cost of the GEOsat required to ensure it will survive the increased operational life.

- opportunity resupply offers a net benefit over the baseline case, but must be tempered by the realization that no assurance of the GEOsat surviving the increased lifetime can be given in this case.

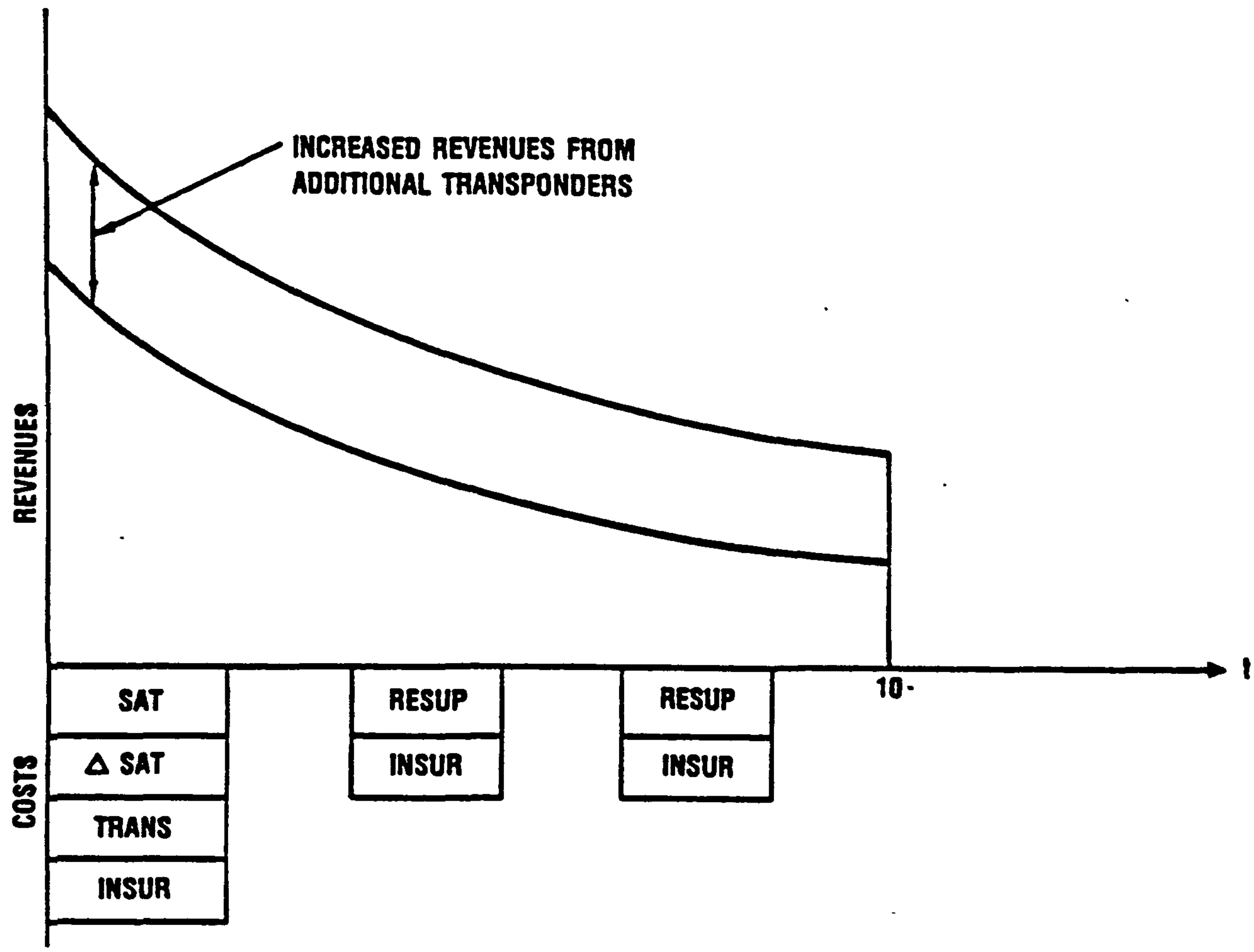


Figure 33. Transponder Replacement
Through Resupply

- a combination of GEO resupply and servicing offers the greatest benefit to the user over that of the baseline case. GEO resupply continues the revenue stream to the user by adding ACS propellant to increase satellite operational lifetime. GEO servicing capability removes the accrued technological obsolescence of the payload, thus increasing the total revenues, and avoids the requirement for an increased satellite cost for the extra lifetime reliability.

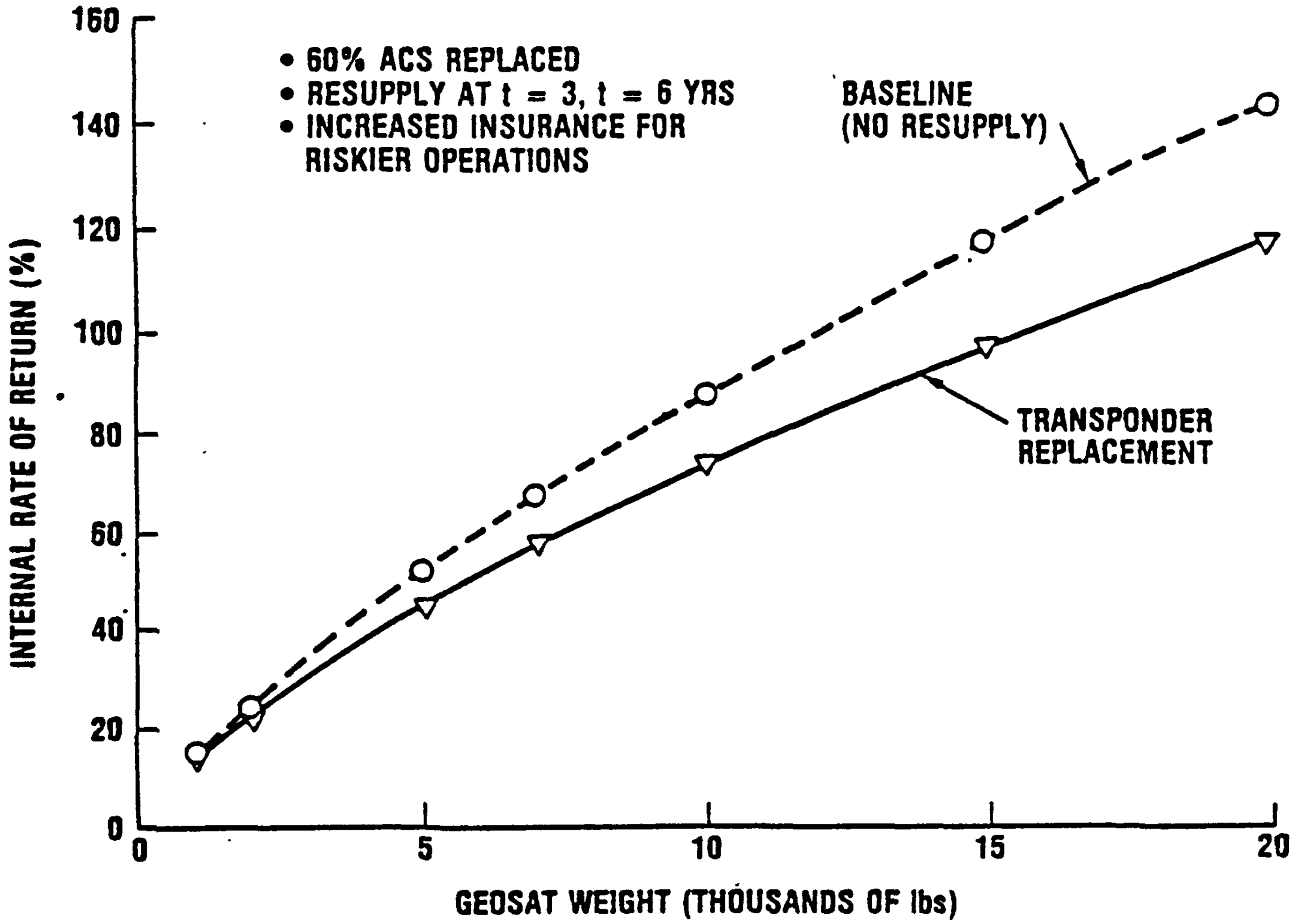


Figure 34. GEOSAT Transponder Replacement

Contingency Resupply

Figure 32 indicates the IRR comparison of the contingency GEOsat resupply case in comparison to the baseline replacement option. This implies:

- o GEO resupply without servicing offers a benefit to GEOsat operators over satellite replacement. This result may only be applicable in a few cases where ACS fuel has been depleted while the status of the communications payload remains unchanged.

Transponder Replacement

Figure 34 indicates the IRR comparison of the case where additional transponders are added to the GEOsat communications payload as a replacement for unneeded ACS fuel assuming a GEO resupply capability exists. This result indicates that:

- transponder addition to GEOsats in lieu of ACS fuel does not offer a net benefit over that of the baseline case. The increased revenues from the increased transponders is not sufficient to overcome the increased risks of this operation, which subordinates future profits to the success of a resupply operation.

It appears possible that the demonstration of a very highly reliable GEO resupply system could reverse this conclusion in some cases by reducing the risk to future profits from the GEO resupply.

Caveats

The assumption used that transponder usage price remain constant with time was chosen to simplify the analysis and is probably not reflective of actual future price performance. The actual price of a transponder is highly dependent upon a variety of factors - most important being supply and demand. Rather than digressing into a projection of GEO transponder supply versus demand, the results of this assumption shold be approximately true in most cases; particularly in the ranking of alternatives. What is important is the discounting of obsolete transponder value over new transponder value which models real world behavior. Discussions in-house and with communications satellite manufactures raised several questions over the assumption used in defining baseline numbers (e.g., cost/lb to GEO, insurance rates, transponder growth rates, etc). The major focus of questions, as expected, was the factor which produced the 12 percent per year decline in saellite revenue generation. There was general agreement that the revenue does decline with time, but there were differing opinions as to why the decline occurred and what its rate might actually be. Major decline factors suggested were:

- Technological obsolescence, as originally suggested.
- Over-supply of transponders leading to price cuts.
- Learning curve effects which allow the deployment of more cost-effective satellites which in turn allow price reductions, i.e., the economic obsolescence of older systems.
- Failure of satellite componenets resulting in a reduction of revenue earned.
- Changes in the competitive position of the communications ground segments.

Revenue decline can occur as a result of both lower prices and loss of revenue generating components. These declines are also not all caused by the space segment of the communications systems. Several observers pointed out the importance of segmenting revenue declines due to the space segment from changes occurring in the ground segment. What we really need to know is how much of the revenue decline is due to the space segment and how much of that consists of items we can actually do something about as part of satellite servicing.

The cost of making a satellite capable of accepting resupply or servicing in GEO has not been included in the satellite costs used here. Mostly, these costs are expected to add to the non-communications and non-electronics costs of the satellite where their impact is less than that of the electronics reliability costs or the changes in the revenue streams. New, larger satellite designs incorporating resupply and servicing options are believed to have a proportionally lower cost than retrofitting these options onto exisitng smaller satellite designs optimized for current operations.

CONCLUSION

The major benefit to GEO satellite operators seems to be the incorporation of both GEO servicing and resupply options in their satellite designs. This offers them:

- o a longer life, avoiding recurring satellite procurement and transportation costs,
- o a means to return their offerring to the current market position by replacing obsolete electronics for current technology electronics.
- o the possibility of avoiding contingency failures by allowing GEO repair in place of replacement of the entire satellite.

Replacing ACS tankage and assuming GEO resupply in order to add additional transponders does not appear to offer a benefit over current operations, since the increased revenues are accompanied by an increase in risk of operations. A very highly reliable GEO resupply system could possibly reverse this conclusion in some cases. This subject and further sensitivity analysis are discussed in the following sections.

Evaluation of Optimum GEO Servicing/Resupply Invervals

While the previous section concluded that servicing in conjunction with resupply was a desirable mode of operation, a key factor was left undefined That is, what is the optimum servicing interval as defined by economic requirements?

Methodology

Based on the spreadsheet analysis programs developed earlier, resupply/servicing intervals were varied to examine their affect on the economic value of the satcoms. Annual annuities were calculated for commercial satcoms in the range of 1 to 10 klb. BOL as a function of servicing intervals of 2 to 10 years. (The final case being equivalent to no servicing).

Two general cases were examined. The first assumed GEO resupply/servicing of a satcom with a standard transponder mix as appropriate to its BOL mass. The second case assumed a tradeoff of ACS fuel for additional transponders. That is, only enough ACS fuel was alloted to last to the next resupply time. The weight savings were then assumed to be used for additional revenue-generating transponders. In both cases, the result of performing a servicing mission was to restore the satcom's revenue to its year 1 value (and thereafter experience the 12% annual decline till the next service). The cost of performing the mission to GEO was assumed to consist of three general elements:

- o Transportation
- o Use of the OMV/RM
- o Cost of resupplied fuel and "black boxes"

The cost of resupplied fuel was assumed negligible and the cost of the replaced "black boxes" (technological upgrading of the communications payload and replacement of life-critical components) was included as a delta cost of production. Transportation and OMV/RM usage fees were combined as part of a net cost of getting to GEO; approximately $21,000/lb. in the cases examined.

Crucial assumptions are related to the reliability of GEO operations and the role of insurance. The reliability of satcom components was assumed to be the same as those used in typical 10 year life satellites. Some decrease in reliability may be allowable if routine visits are possible, but the cost impact of this factor is uncertain and thus a more conservative assumption was made. If a GEO mission fails, that failure may keep the satcom from earning revenue until the next flight can be performed or, if severe enough, the failure can result in the total loss of the satcom. While the cost of the satcom itself and its transportation are insured against at launch, the performance of GEO resupply/servicing will likely require insurance of the satcom's projected revenue stream. Since the revenue stream, except for near the end of the satellite's operational life, is much greater than the original cost of the satcom and its transport to GEO, assumptions on the appropriate insurance rate contribute directly to the cost of the mission. We assumed a 10% risk premium in our calculation and this was felt to be a conservative assumption. Actual operational experience may eventually allow the decrease of this rate to 5% or even 2%.

Results

Figure 35 is a plot of the expected annual annuities as a function of the servicing interval for a range of satcom BOL masses. As expected, with greater weight and more transponders, the economic value of the satellite increases. These satcoms have a standard transponder mix and contain generous safety margins of ACS fuel. Peak annual annuities for all satcom masses were found at 6 years. Servicing every 5 was so close in value as to be easily within the uncertainty range of the CERs we have used. As expected, servicing more infrequently did not carry a large penalty. Compared to baseline 10 year life cases for each satcom weight, servicing had positive advantages for all intervals of 3 years and up.

Figure 36 shows a similar plot for the case of maximizing onboard transponders through the off-loading of ACS fuel. This dramatically increases revenue-generating capability, but at the risk of no safety margin for the ACS fuel. The solid lines relate to the annual annuities scale to the left while the dashed lines (for the larger satcoms) relate to the scale at the right. This combination was necessary to avoid compressing the curves of the smaller satcoms. It is interesting to note that an optimum servicing interval exists in these cases as well, but that it occurs earlier at 4 years. The peak is also more sharply defined, as would be expected from satcoms with greater numbers of transponders to upgrade. Compared to baseline cases, this transponder substitution is superior at all servicing intervals greater than two years. A caution should be raised here, however, Adding a greater supply of transponder to a specific satellite does not mean they can be sold for the same price as the baseline situation. The number of satellite transponders to be emplaced is less a technial capabilities question and more a function of market forces. No attempt has been made in this analysis to account for the price elasticity of transponder demand (which itself is a variable with time). It is unlikely that all additional transponders would have the same revenue-generating capability.

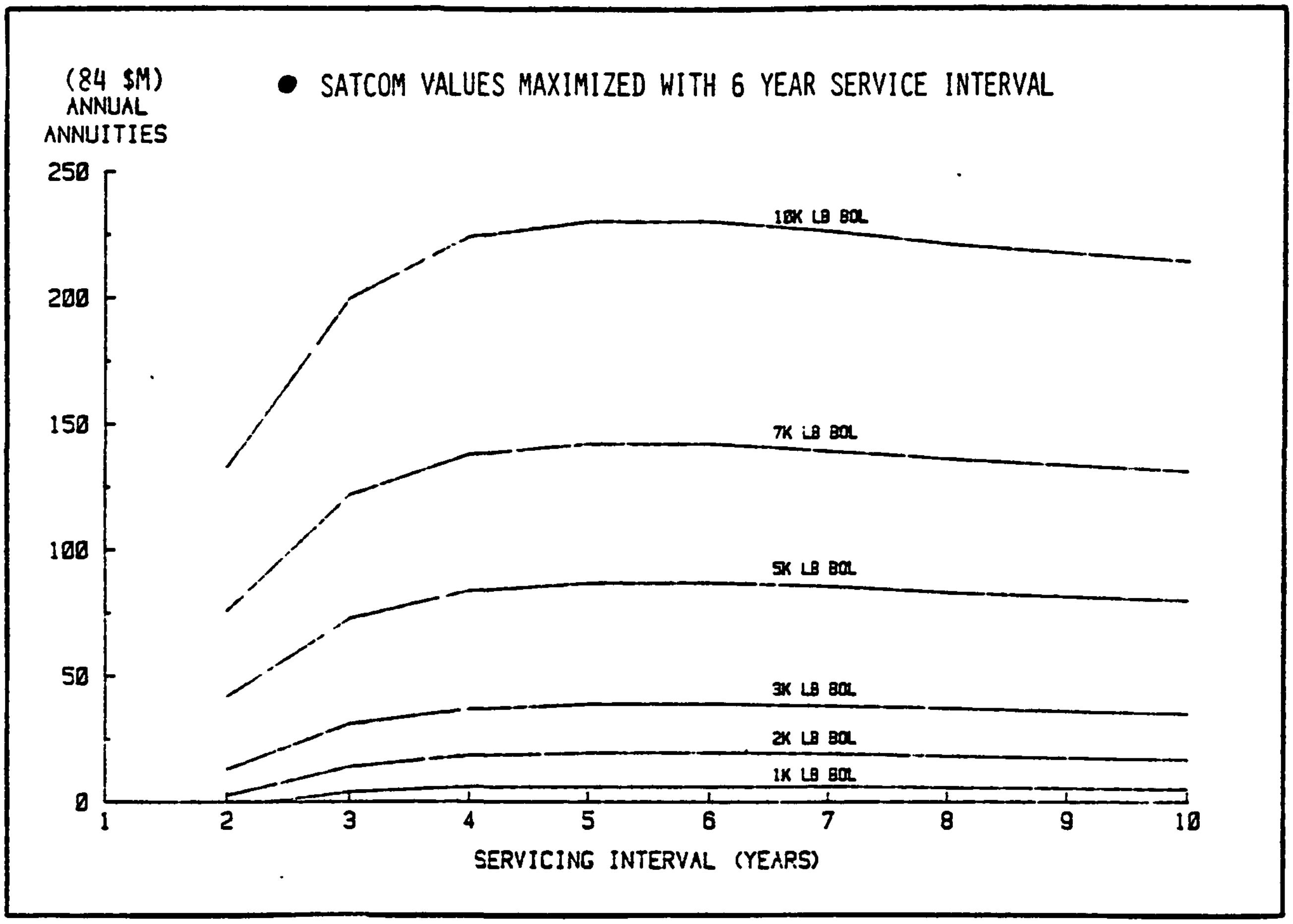


Figure 35. GEO Resupply & Servicing
Std. Transponder Mix

Conclusion

Single optimum servicing intervals exist for the case of resupply and servicing of commercial GEO communications satellites. Consistent acros the existing range of satcoms (1-10 klbs), satcoms with standard transponder mixes should be serviced every 5-6 years. In cases where transponders have been added to use the weight allowance of off-loaded ACS fuel, that optimum interval is every 4 years. The optimum points denote a peak in the equivalent annual annuity value of the satcom as a function of varying intervals. The performance of resupply/servicing enhanced the economic value of the satcom project in all cases.

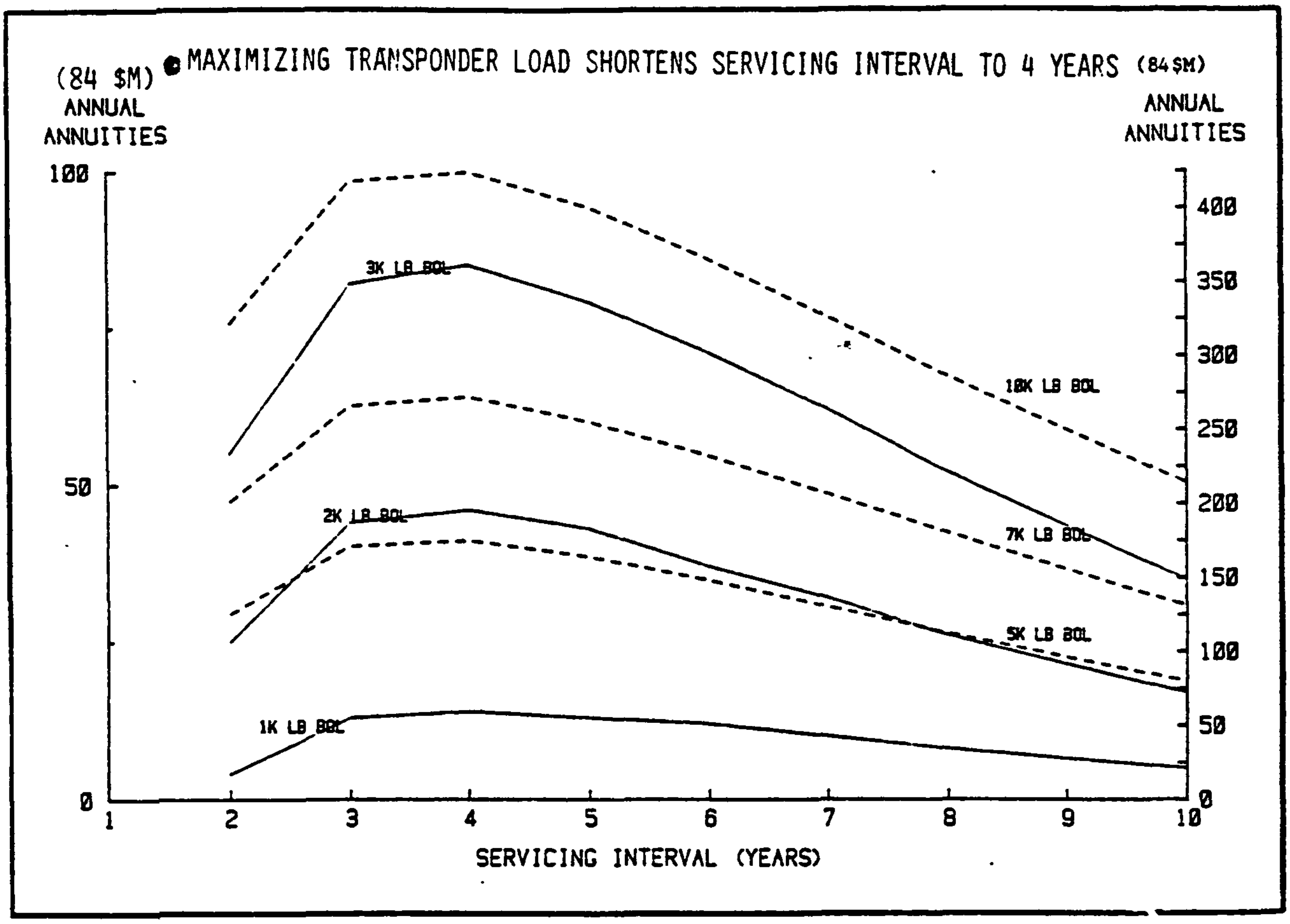


Figure 36. GEO Resupply and Servicing
Transponder/ACS Tradeoff

Sensitivity Analysis of Optimum GEO Servicing/Resupply Intervals

After the determination that such optimum servicing intervals existed, the next step was to examine the sensitivity of these results to changes in key baseline assumptions. Three key assumptions were examined:

- insurance rates for resupply/servicing operations
- the cost per lb. of performing operations in GEO
- the obsolesence rate for satcoms, reflecting the yearly decline in their revenue-generating capability.

It should be cautioned that these analyses and those in the preceeding references were performed on a brefore-tax basis. No attempt was made to simulate the effects of differing depreciation rates or changes in tax law.

Effect of Insurance Rates on Optimum Geo Service Intervals

A significant portion of the cost of performing resupply/servicing missions to GEO satcoms is the requisite insurance. In the event of damage to the satcoms, its future revenue stream will be reduced or eliminated and insurance will likely be needed against this possibility. Using the previously developed spreadsheet analysis programs, the baseline insurance rate of 10% was varied across the range of 2% to 15%. The resulting changes in the satcom's annual annuity value and associated optimum servicing interval were noted and plotted.

Figure 37 shows the optimum servicing intervals as a function of the insurance rates for the resupply/servicing operations. Insurance for the launch and deployment phase, in contrast, was maintained at the baseline 10% in all cases. As the insurance premium drops, performance of a GEO resupply/service mission decreases and it becomes advantageous to perform such missions more often. A key result here is that if insurance rates cannot be brought to 15% or less, GEO resupply/servicing does not carry a strong economic benefit.

Figure 37 also shows how differing insurance rates affect the satcom's annual annuity value as a function of the satellite's BOL Mass. Lower insurance rates increase the value of the satcom's annual annuities-with the greatest benefits occurring to the smaller satcoms. Conversely, the negative impact of higher rates is greatest for the smaller satcoms.

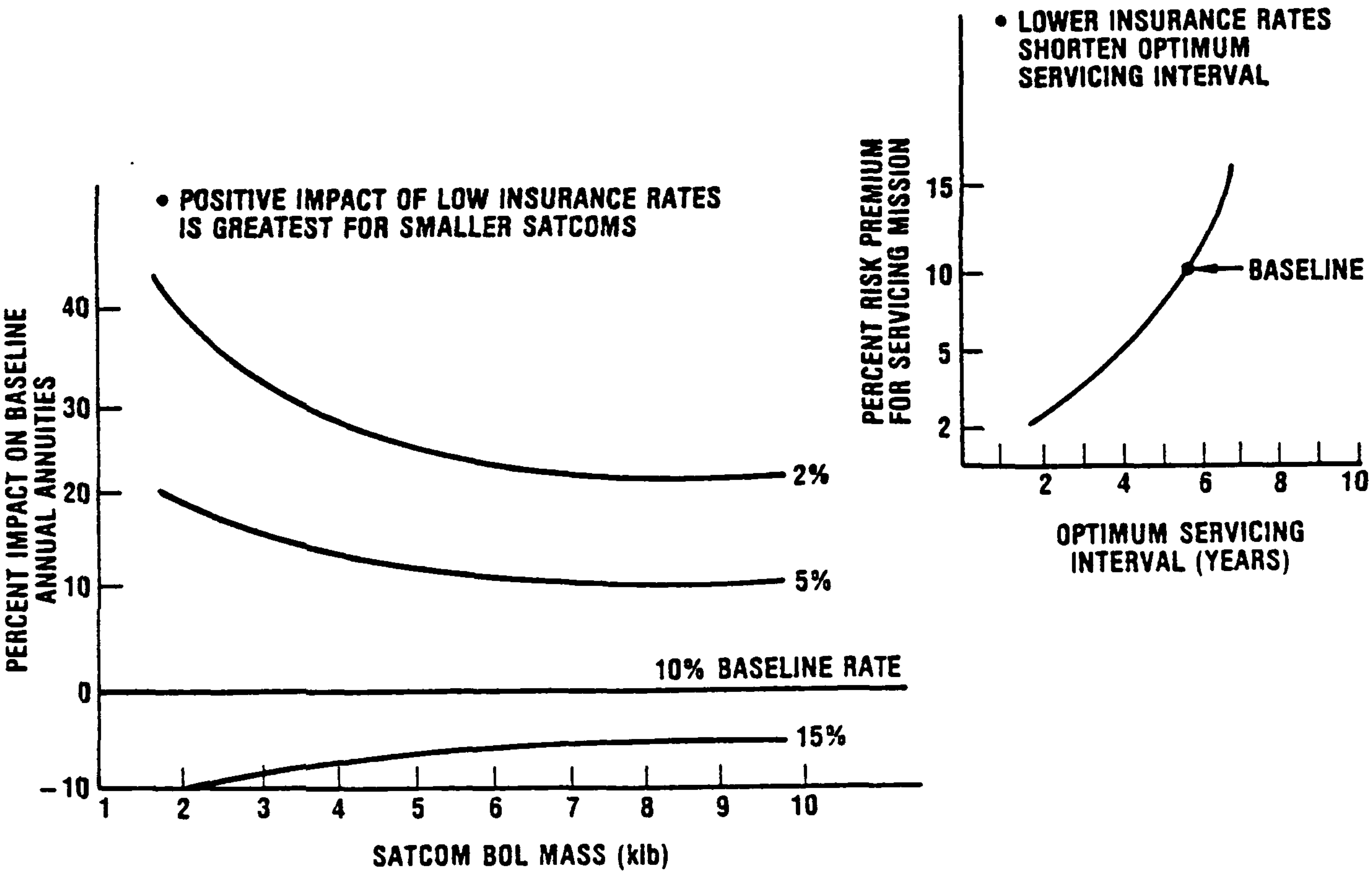


Figure 37. Insurance Has A Significant Impact On Geo Resupply/Servicing

Effect of the Cost Per Lb. for Geo Operations

The costs of transportation to GEO and the use of the OMV/RM were combined into a net cost of putting the resupplied fuel and "black boxes" on the satcom. A variety of scenarios exist for getting fuel and components to a GEO satcom; rather than attempting to use specific scenarios, parametric cost estimates were used as target figures. For a financial analysis, how the resupply and servicing is accomplished is of less importance than the associated risk (see the case above) and the net cost of getting the actual ACS fuel and new components on the satcom. In the baseline case, the net charge was placed at $21,100/lb.

The cost of performing GEO resupply/servicing was varied across the range of $10,000/lb. to $100,000/lb. The associated change in the optimum servicing interval is shown in Figure 38. A decrease in cost/lb. makes it optimal to service more often and increases in cost/lb. cause a lengthening in the optimal interval. While these conclusions seem rather obvious, it is important to note how insensitive the optimal servicing point is over a relatively broad range of cost figures. From about $20k/lb. to $60k/lb., the optimum interval stays at about 6 years. Only at the extreme figure of $100k/lb. does the optimal servicing interval stretch to a point where resupply/servicing is no longer desirable. On the other hand, dramatic decreases in cost/lb. do not result in significantly more frequent GEO operations.

The effect of variences in the cost/lb. assumption on the satcom's annual annuity value is shown in Figure 38. As expected, cutting the cost/lb. increases the satcom's value and raising the cost/lb. decreases the satcom's annual annuity. The effects are a relative insensitive function of satcom BOL mass with smaller satellites being more affected than larger ones. The benefit of cutting the baseline cost/lb. figure in half is, for example, less than 5%. The disbenefit of raising it an order of magnitude is mostly less than 15%. The original baseline figure of $21.1k/lb. to Geo was felt to be a conservative and defendable estimate. The key result from this is that the cost/lb. for GEO resupply/servicing has only a minor effect on the desirability and timing of these operations.

Effect of Satcom Obsolesence Rate on GEO Resupply/Servicing

A crucial assumption of the spreadsheet analyses we have been performing is that the constant dollar revenues from the satcoms decreases at a constant average annual rate. In the baseline case, this has been assumed to be 12% per year and is based on the historical growth rate of transponder demand. The major benefit of planned resupply and servicing is the performance of technological upgrading. The assumption being that such technical upgrading is not only feasible but that it results in restoring the satcom's revenue-generating capability to its year 1 value.

The 12% figure has been questioned as being overly ambitious. The contention is that such rates of decline are not occuring with current systems and that the decline rates that do occur are only partially related to the space segment-with ground services and market forces being the predominant influence. To test the sensitivity of this assumption, obsolesence rates of 6% to 12% were used across the range of 1-10 klbs. satellite BOL mass.

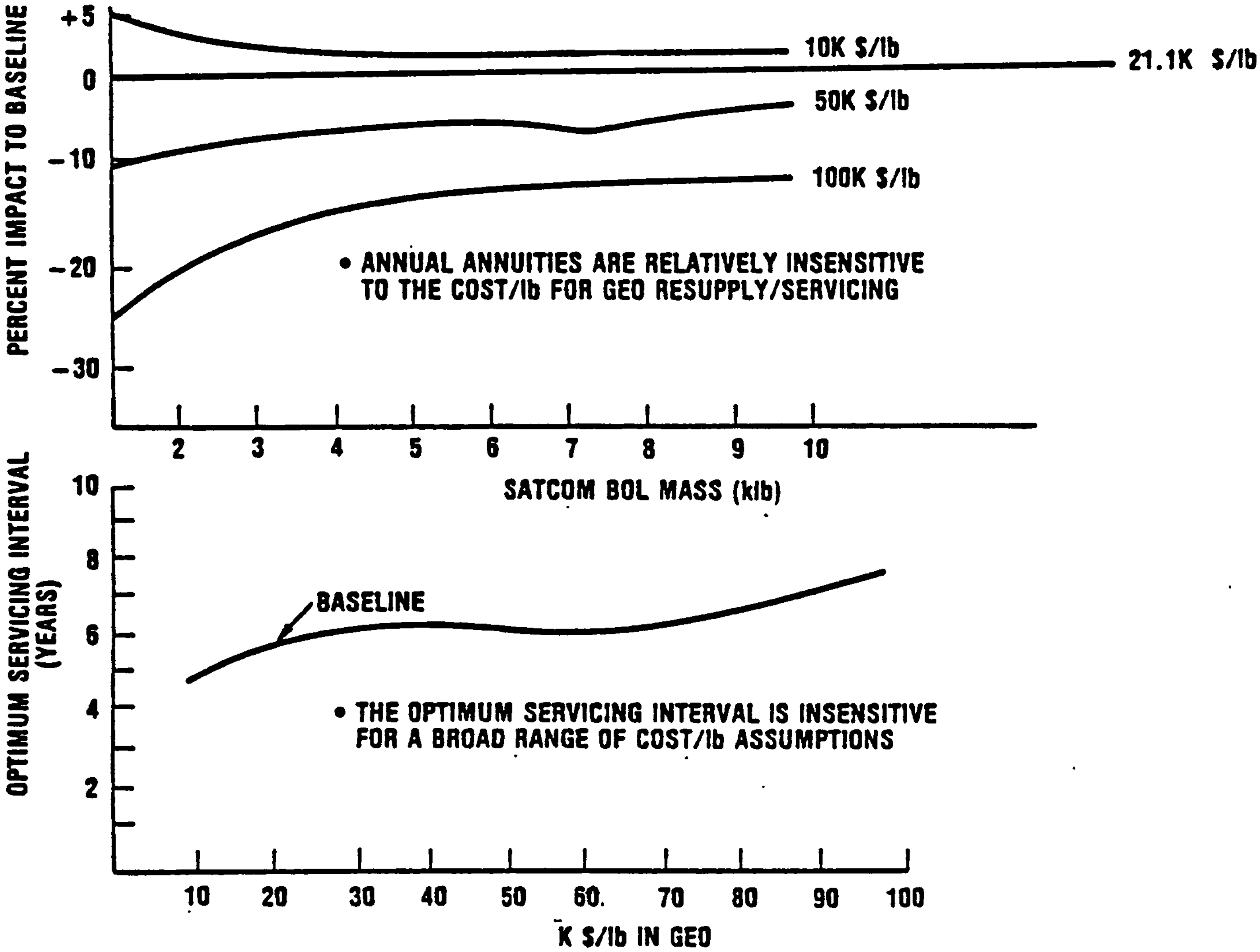


Figure 38. Effect of GEO Servicing Cost/Lb. On Satcom Values

The effect of different obsolesence rates on the optimum GEO resupply/servicing interval are shown in Figure 39. If the satcom suffers less obsolesence, there is less incentive to resupply and upgrade it. If a satcom's revenue-generation decline can be held to a 6% rate or lower, it is unlikely that resupply/servicing will present significant advantages. At about a 9% rate, the optimum servicing interval has moved out about a year. This move however is still well within the uncertainty range of our CER's and not highly significant.

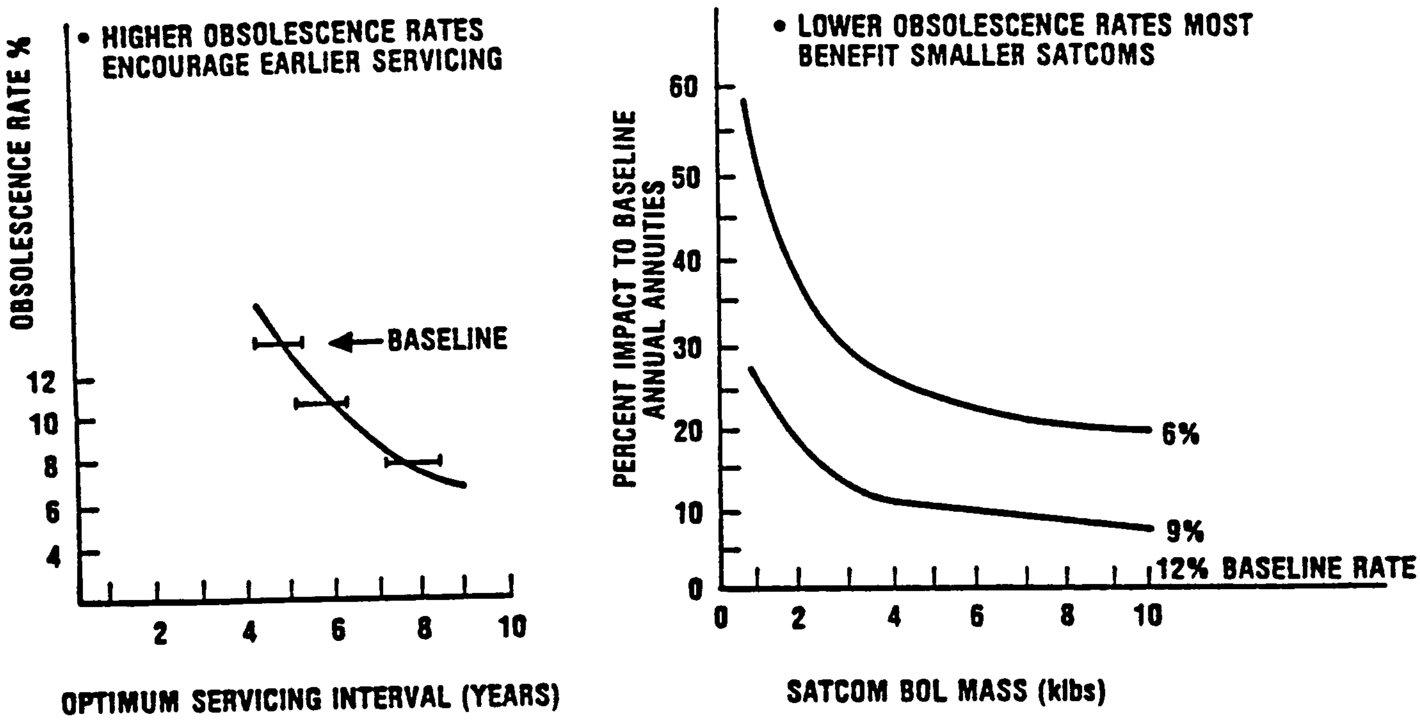


Figure 39. Effect of SATCOM Obsolesence Rates on GEO Resupply/Servicing

Figure 39 also shows the effect of two different obsolesence rates on a satcom's annual annuity value. Obviously the lower the rate of revenue decline, the higher the equivalent annual annuity. The significant point here, however, is the dramatic percentage increase in the annual annuity for satcoms below about 3klbs. This can be expected from the relatively lower mass devoted to communications payload in the smaller satellites and the higher leverage of the obsolesence rates. The key result here is not only that the obsolesence rate significantly affects the economic value of the satcom, but that it is surrounded by a large degree of uncertainty. It is not obvious that technological upgrading will be able to fully restore year 1 revenue-generating capability or that changes on the ground might lessen the need for such actions.

The financial model we have been using has been shown to produce stable results across a wide spectrum of potential changes in key baseline figures. No changes to earlier conclusions need to be made. Several key new conclusions from the sensitivity analysis can be drawn:

- o Insurance rates have a high impact on the economic viability of resupply/servicing operations for satcoms. Rates of 15% or lower are required with a consequent impact to the overall reliability requirements for GEO operations. Based on historical experience, an overall reliability level of 95% or better is required from launch through GEO operations.

- The cost/lb. figure for GEO resupply/servicing has a relatively minor influence on the timing of optimum servicing intervals and the satcom's annual annuity value. Resupply/servicing scenarios only need to keep their cost/lb. figures below about $50k/lb. to be economically viable. This includes not only the cost of getting to GEO, but associated usage fees for required elements (e.g. OMV, OTV, servicer kit, etc).

- If satcom revenue decline rates average less than 6% over their lifetime, resupply/servicing is not likely to be attractive. Decline rates of about 9% and above make technological upgrading potentially attractive. The key questions here are the technical feasibility of revenue restoration with upgrading and what combinations of space and ground segment modification are required.

Sensitivity Analysis of Satellite Industry Concerns

Based on discussions held with TRW, Hughes Aircraft and Ford Aerospace, several additional economic sensitivity analyses were suggested. These analyses were to further examine the effect of varying assumptions used in the spreadsheet modeling of communications satellite resupply and servicing. Three general situations were examined:

1. The first question addressed the assumption that satellite servicing would restore the satcom's revenue-generating capability to its Year 1 value. What happens if full restoration is not possible by satellite servicing, either due to some technical impossibility, or problems with the ground segment? What happens if the revenue rate is increased? The latter possibly would use coordinated changes in the ground and space segments.
   To examine this case, the level of revenue restoration was varied in the spreadsheets. Instead of 100%, values of 30%, 50%, 70%, 90%, and even 110% and 120% were used. Each time satellite servicing occurred, revenue levels were restored to some percentage of their Year 1 value.
   The results are displayed in Figure 40. For lower restoration rates, the optimum servicing interval lengthened. At restorations above 100%, the servicing interval did not decrease below the baseline case (about 5 years) while the annual annuity value of the satcom increased. As expected, annual annuity values decreased for less than 100% restoration. These results held across the examined range of satellite BOL masses (1-10K lb). The relation of restoration rate to the obsolescence rate (the yearly decline in satcom revenue-generating capability) was that satcoms with higher obsolescence rates could tolerate less restoration and still find servicing viable. If your earning power is dropping fast enough, you can be less picky about how much your revenues need to be enhanced.

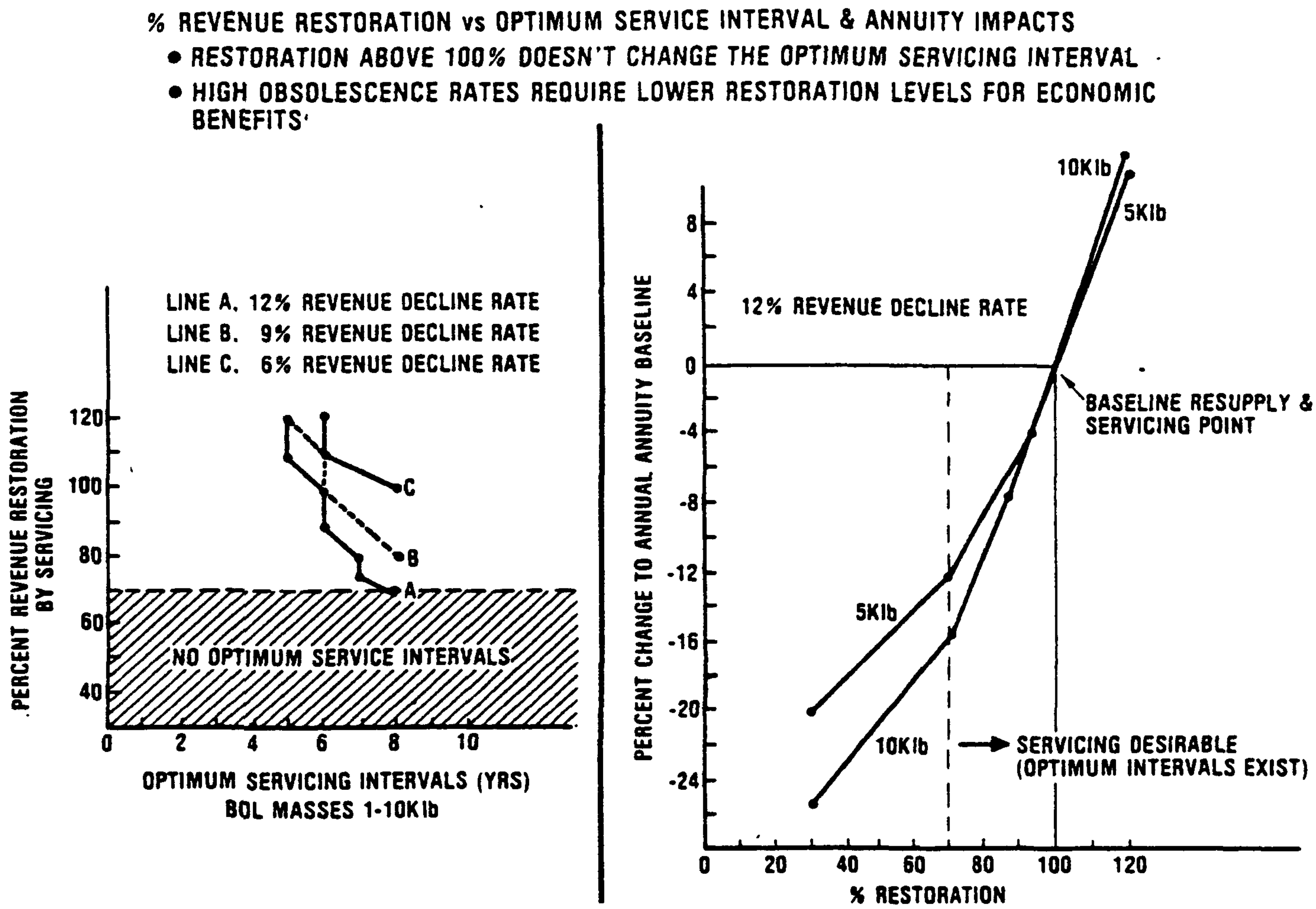


Figure 40. Impact of Varying Levels of Revenue Restoration from Servicing

Conclusions to be drawn from this case are:

- Greater than 100% restoration is beneficial, but doesn't change the servicing interval.

- Revenue restoration rates required for economic viability are not so much a function of the satellite mass as of the obsolescence rate. A baseline revenue decline of 12% per year requires a minimum of 70% revenue restoration from servicing. A 6% decline requires a greater than 100% restoration level. The less obsolescence suffered, the greater the required benefit from servicing.

2. The second question addressed the assumption of $200/hr as the price for a transponder. Industry representatives pointed out that transponder prices vary greatly depending on how they are leased, either long-term or on the spot market. If changed, how would this assumption affect the selection of servicing intervals?

The spreadsheets were modified to use transponder prices of $100 to $600/hr, reflecting current possible pricing extremes. The results were that the optimum servicing interval varied very little. Servicing intervals were every five years for up to $600/hr and every six years for as low as $100/hr. This is essentially the same result, within the uncertainty of the data, as the baseline cases. Also, as expected, satellie annual annuities rose with increases in the transponder price; the greatest percent benefits being for the smaller satcoms.

While the tansponder price greatly affects the annual annuity value of satcoms, see Figure 41, it does not significantly affect the timing of servicing operations. The earning power of the satcoms relative to the current market, as reflected in the obsolescence factor, is the more important issue to watch.

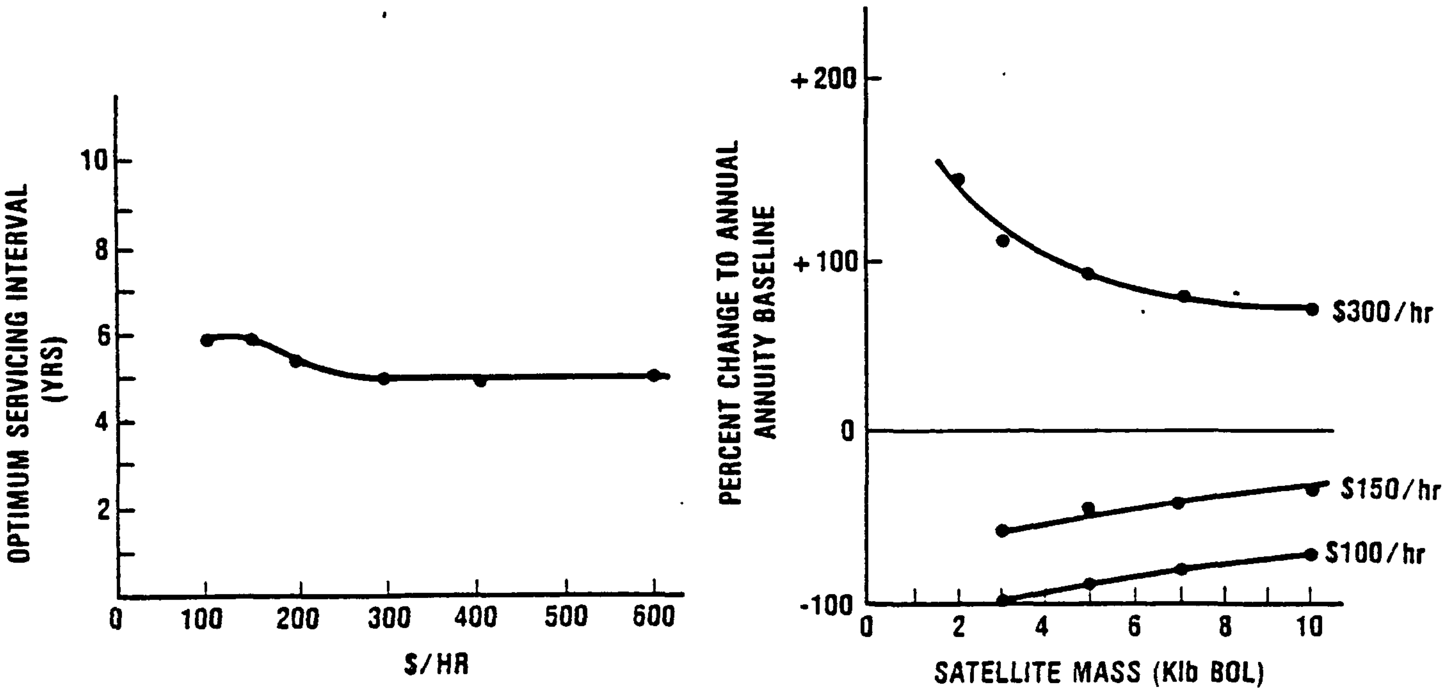


Figure 41. Impact of Changes in Transponder Prices

3. The third question addressed the assumption of the cost impact of producing longer life satcoms for the case of planned resupply. In the planned resupply case, 10-year satcoms were modified to have a 15-year life at the same level of reliability. At the 10-year point, a resupply operation occurs, providing ACS fuel for the remaining five years of life. No other servicing occurs at this time. This case contrasts with the opportunity resupply case in which a 10-year satcom was refueled for an additional five years, but no costs were incurred upfront to insure it would last the full 15 years. Essentially, since it was working at 10 years, it was assumed to continue to work if given sufficient ACS fuel.

Since the planned resupply case was not found to be economically viable, the question was raised as to the relative importance of the extra upfront costs incurred for the longer lifetime. The planned resupply case made some conservative assumptions about the cost penalty of extra life reliability, thus making it and opportunity resupply (no cost penalty) two extreme cases. To examine the significance of the penalty, it was reduced in percentages from the planned resupply case (100%) to that of opportunity resupply (0%) and the impacts to the annual annuity levels were noted.

The results of this case are shown in Figure 42. The baseline case of a standard comsat, 10-year life with no resupply or servicing, is superior in almost all cases to that of planned resupply. This holds until the cost penalty for extra life reliability is reduced to 20% of its original value. Thus, very optimistic assumptions on the cost penalty for extra satcom life are necessary to make planned resupply alone econmically attractive. This conclusion further emphasizes the importance of performing servicing in conjunction with resupply, particularly in the case of GEO operations.

The three cases examined here do not change any of our original conclusions on the viability of GEO resupply and servicing. They do, however, provide further data and insight into the particular financial model we have been using. The results emphasize the importance of minimizing the cost impact of making satcoms serviceable and of the need to restore/enhance their revenue-generating capabilities for GEO operations to be desirable.

Potentially Resupply Uses in GEO Operations

The economical remote resupply of satellites and platforms in GEO depends on the capability for technology upgrading and maintenance in GEO. An economically desirable way of accomplishing this uses the resupply module as a propellant tanker for the OMV. The primary alternative to use of the OMV would be a space-based reusable OTV, if available.

In addition to the remote resupply of assets in GEO, the resupply module can support some GEO operations while employed in LEO. A preliminary screening of possible concepts included:

- o Use the OMV/RM combination as a reusable perigee kick stage prior to the development of a fully reusable OTV. See Figure 43.

- o Develop a LEO propellant depot to support the space-based operation of the OMV/RM. Such operations would include propellant transfer to elements heading for GEO. Supplies of depot propellant may be economically available by scavenging orbiter OMS/RCS fluids. The OMV/RM could also be used for such scavenging. See Figure 44.

Figure 42. Lifetime Extension Options (10-15 Yrs)

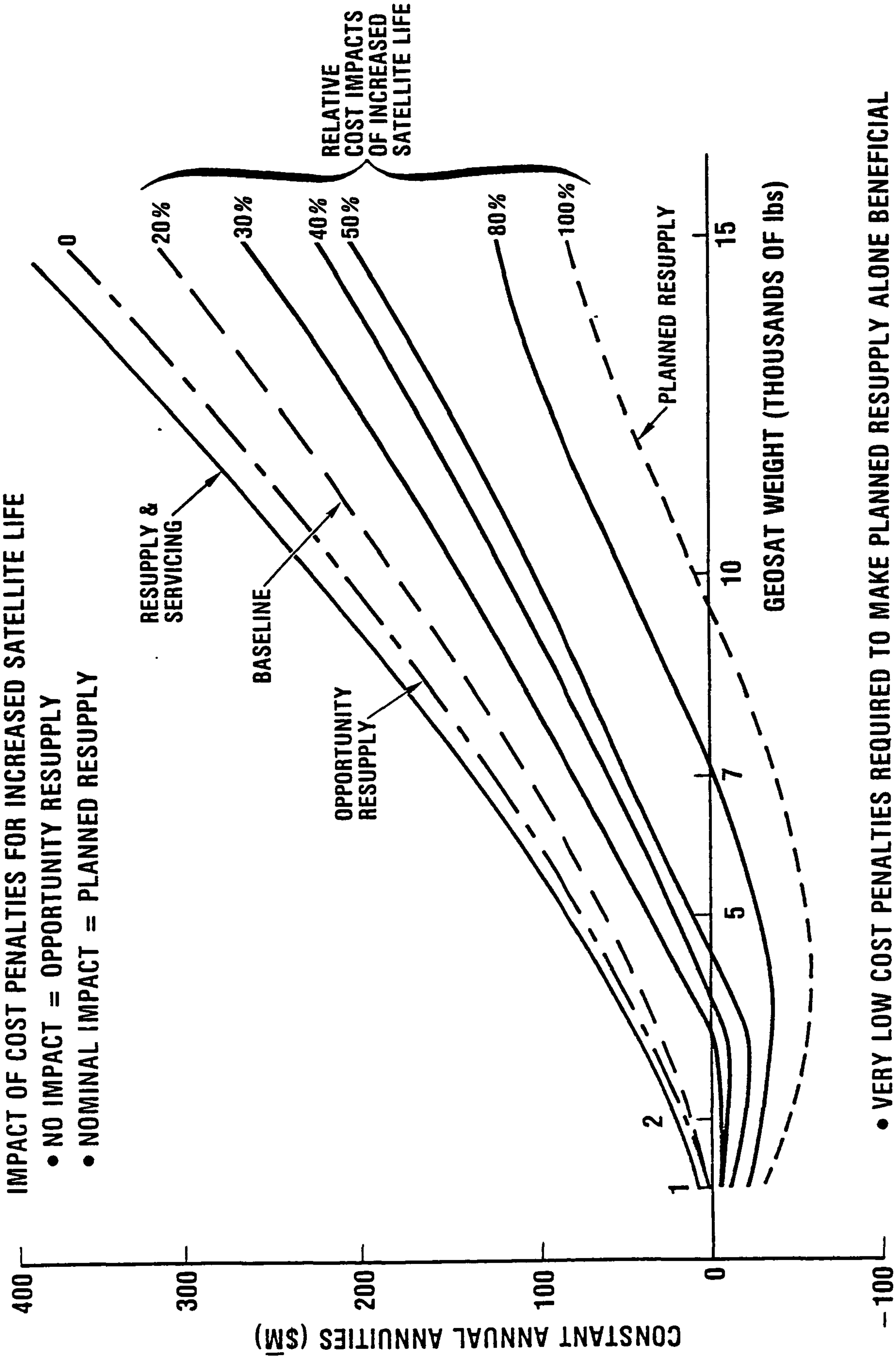
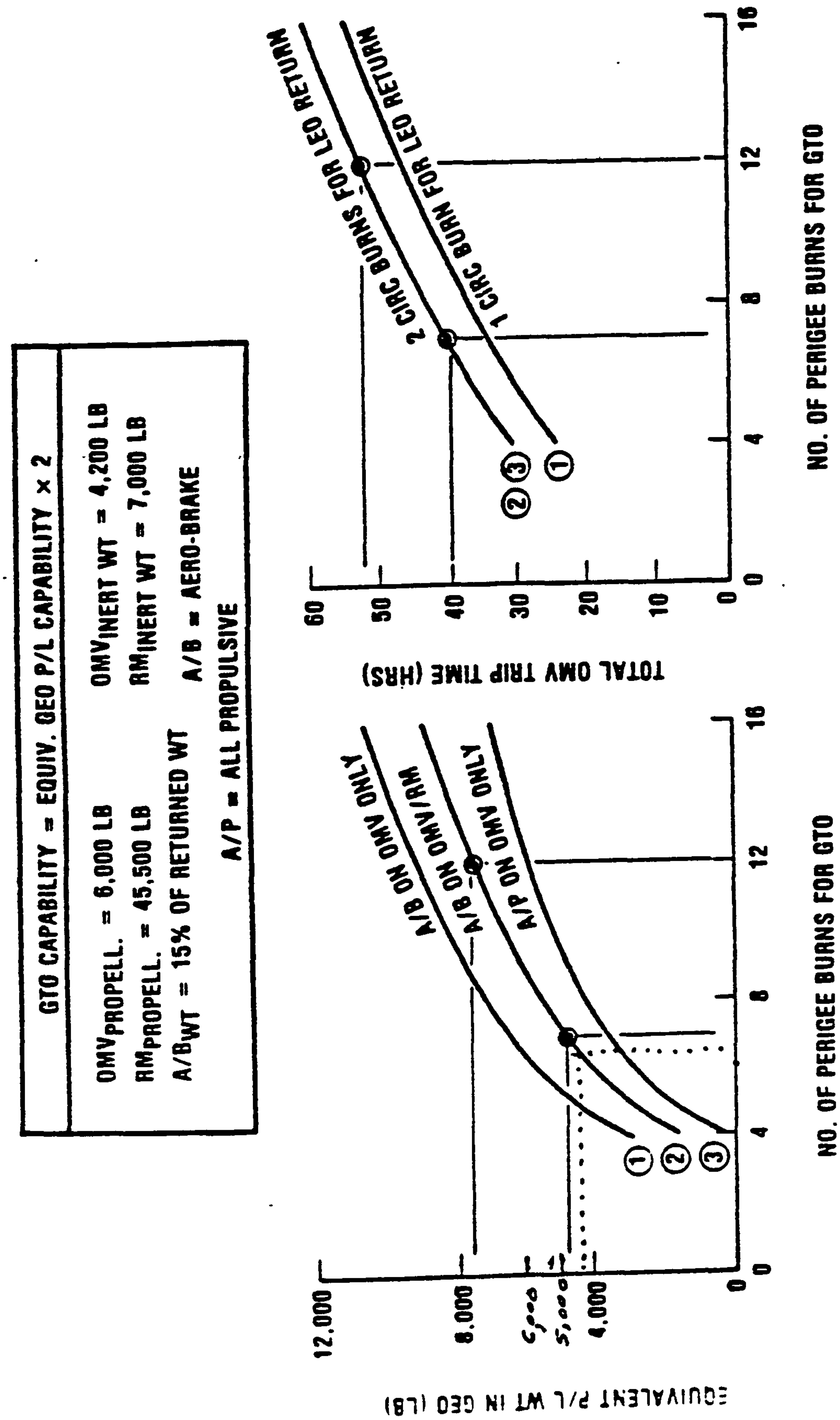


Figure 43. Equivalent GEO P/L Capability With
OMV/RM Reusable Perigee Stage



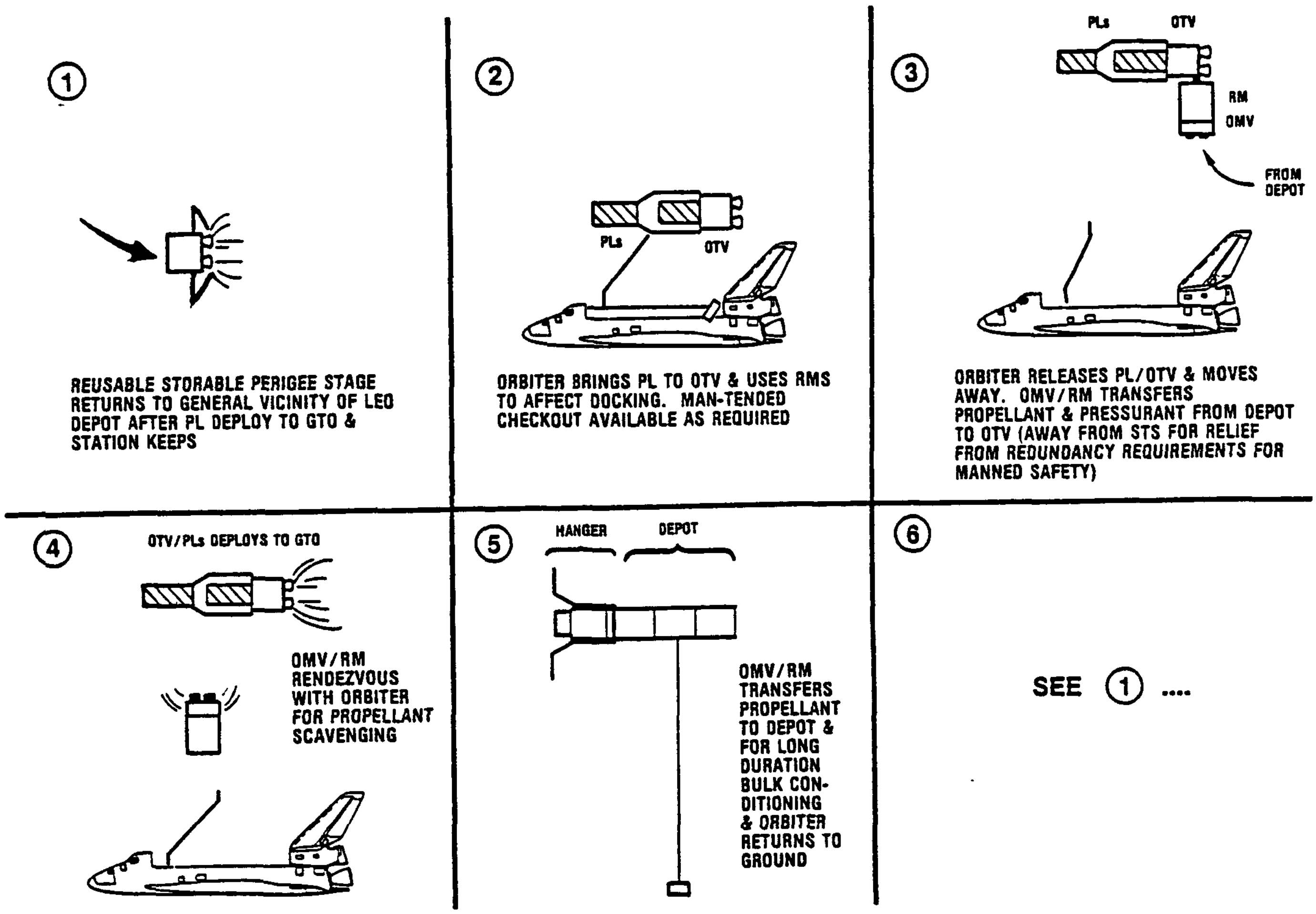


Figure 44. Representative Resupply Module/OMV Operations in Conjunction with LEO Depot and Reusable Perigee Stage

o Use the OMV/RM to top-off satellites that have reached LEO, but are still to be placed on station at higher altitudes (e.g., 12 hr orbits, GEO) such top-off operations could ease weight limits on an individual shuttle flight (i.e., the orbiter capacity is used to lift a greater amount of dry mass). This concept is related to earlier proposals to dry launch satellites and stages and consequently to reoptimize their design. Such reoptimization would presumably result in more efficient launch installations.

These concepts show that the resupply module can be used for more than the support of specific spacecraft and platforms. In its support of GEO operations, the RM becomes an important part of the developing space infrastructure. See Figure 45. LEO remote resupply can serve to increase the efficiency of space operations by introducing users to space-based operational modes prior to the IOC dates of space station and the reusable OTV. The OMV/RM combination is not only an effective one for GEO remote resupply and servicing, but it can support the economic deployment of those assets to GEO itself. Further examination of these possibilities in cooperation with potential users is desirable.

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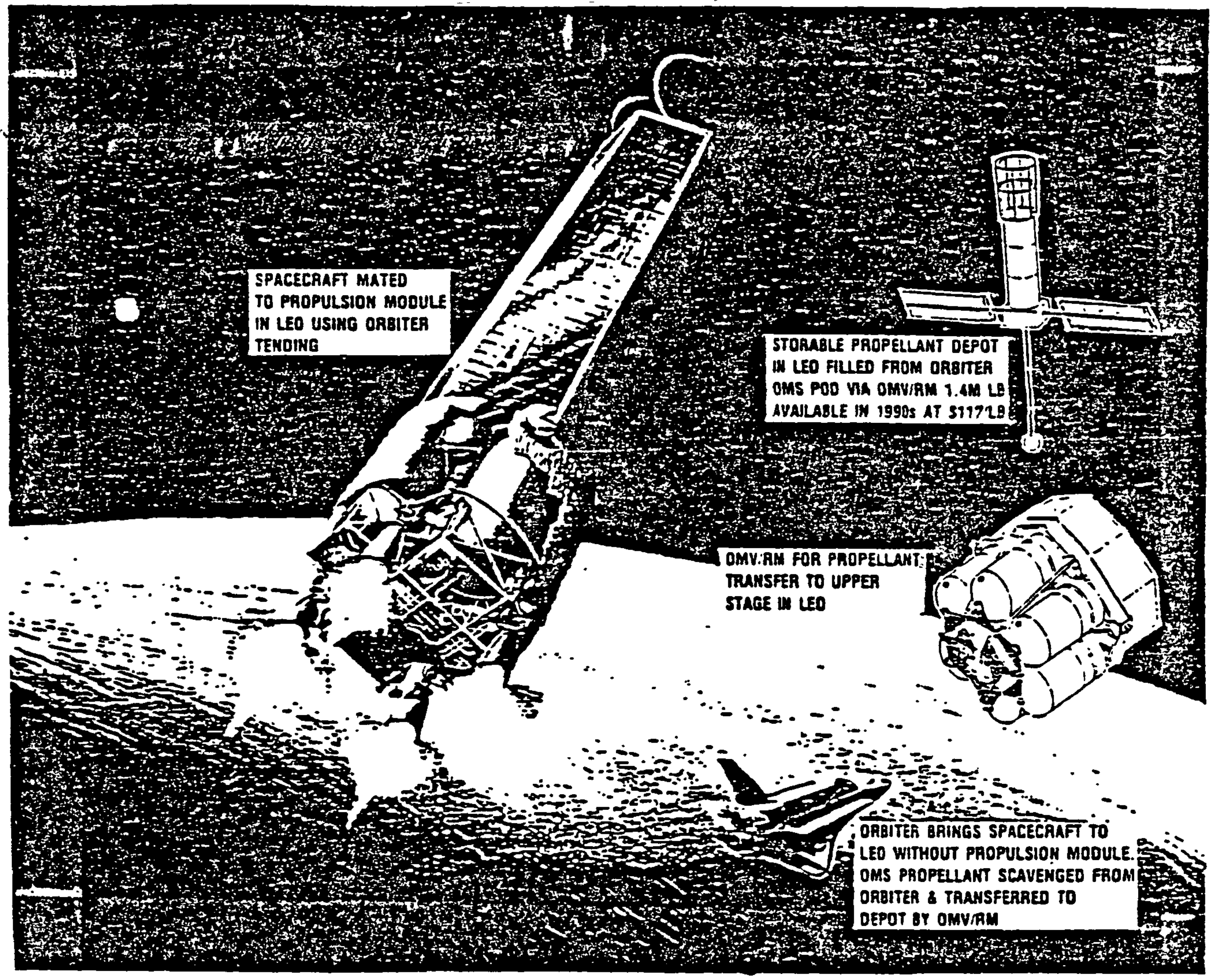


Figure 45. LEO Remote Refueling of Storable Stage Candidate Scenario

3.1.4 Design Reference Mission (DRM)/System Requirements Selections

Approximately 20 LEO fluids resupply scenarios and 4 GEO fluids resupply scenarios were initially developed, and mission profiles defined for each in Subtask 1.2. For ROM cost, risk and complexity estimating purposes, the scenario list was "scoped" to remote scenarios versus "Orbiter-based" scenarios.

The six remote resupply scenarios, two in-bay scenarios and two return-to-earth" scenarios indicated in Figure 46 were used to compile initial parametric affordability estimates. These were subsequently combined with the needs (gross economic or operational benefits) assessments for each of the twelve screened spacecraft programs to develop the semi-final list of candidate design reference misisons. This section address how cost, risk, complexity and resupply benefit factors were derived and used in selecting design reference missions (DRMs) for the remote resupply module.

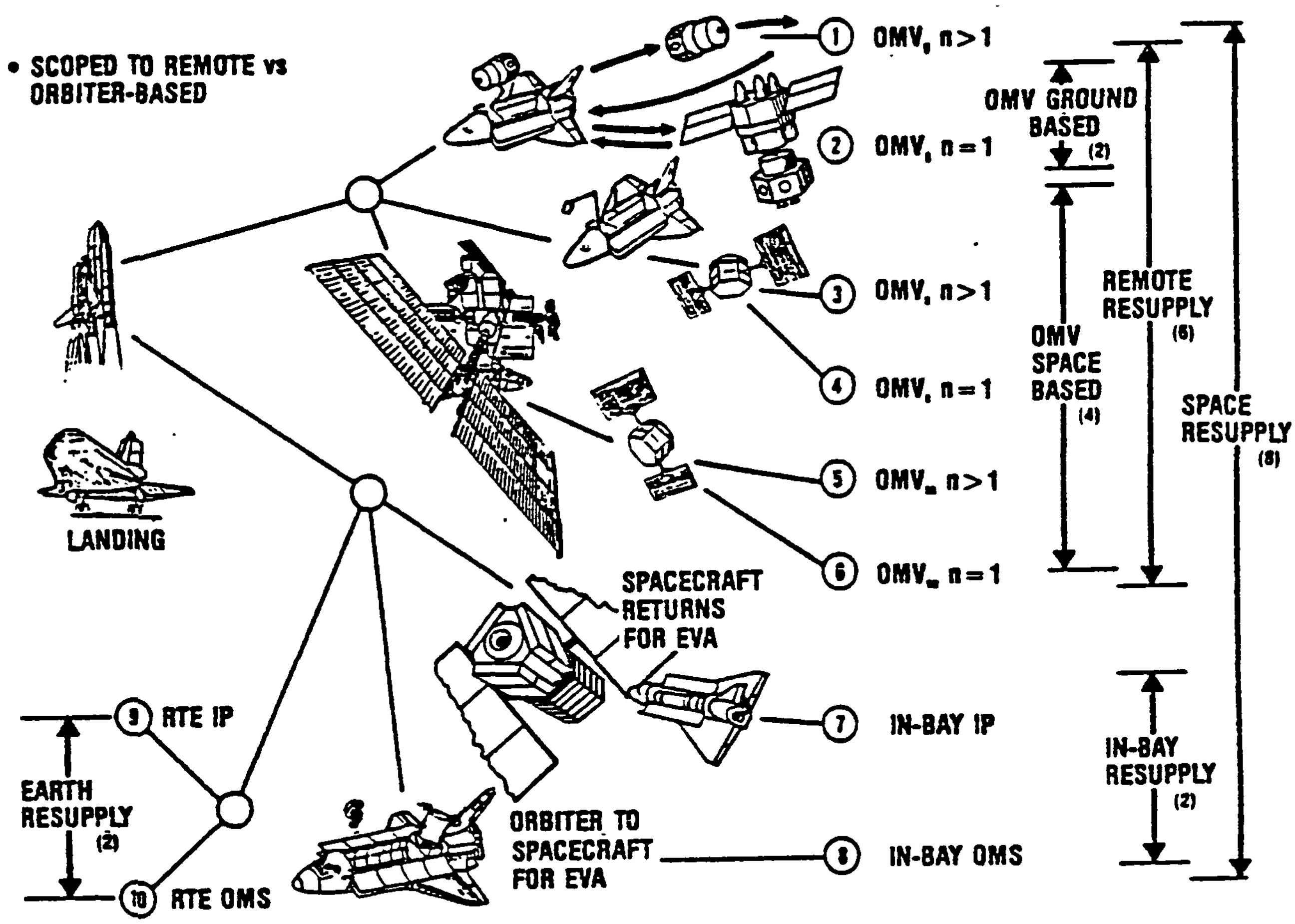


Figure 46. Low Earth Orbit Alternative Fluids Resupply Scenarios

Cost Factors

Major "Rough Order of Magnitude" (ROM) cost estimates were generated for each of 10 LEO and 4 GEO alternate fluids resupply scenarios, with minor adjustments made to reflect program differences at the candidate design reference mission level. Average total costs per resupply engagement (initially assuming 111 engagements) were netted against spacecraft gross economic benefits to produce net economic benefits for each candidate design reference mission. Figure 47 shows the relationship between the various factors contributing to the average total cost of a resupply mission.

Space Transportation System (STS) level transportation costs were estimated using average cost per flight (not "user fee") relationships for the 1990 - 1998 time frame, with specific estimates for each launch site:

- o ETR -- $110M cost per flight = $2255/pound *
- o WTR -- $145M cost per flight = $6050/pound *

It was assumed that LEO spacecraft designs would be Shuttle-optimized in the post-1990 period, therefore, "length" and "weight" transportation cost factors would be identical.

Objective — Minimize Resupply Cost to End User

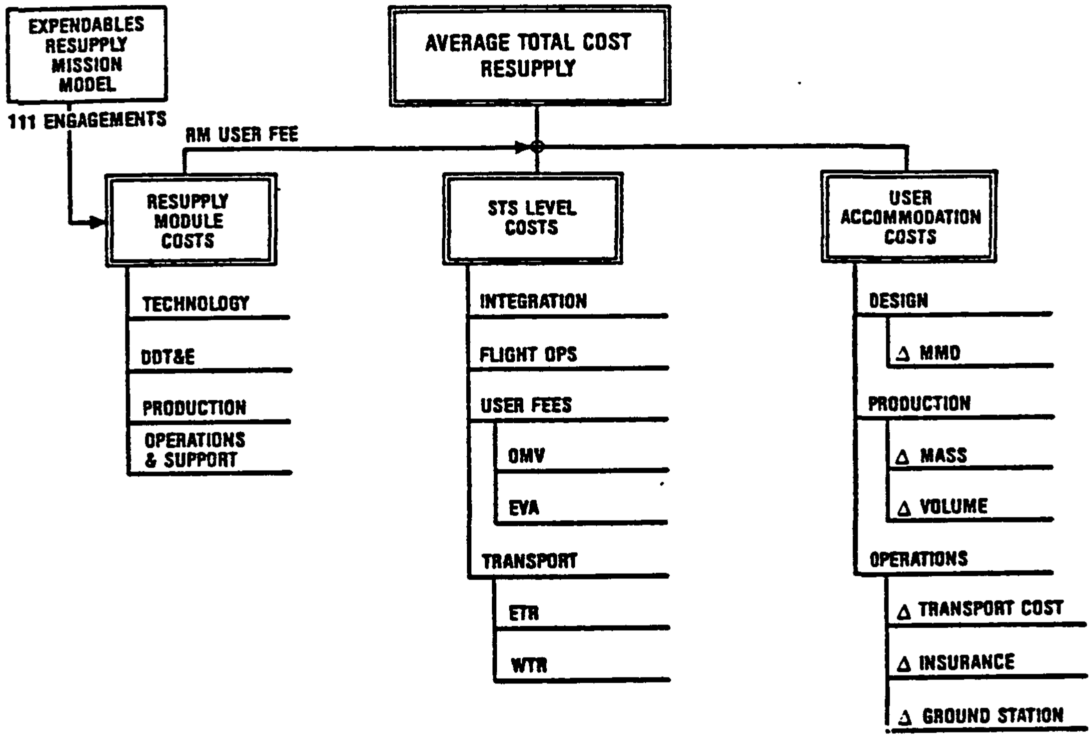


Figure 47. Relationship Between Considerations for Minimizing Resupply Cost to the Potential User

User "accommodation" costs (to benefit from resupply without concurrent maintenance) are known to be significant, but were not specifically estimated. It is quite unlikely that a program sponsor would deliberately increase the spacecraft design life (at very large marginal cost) to obtain the benefits of resupply alone, without accommodating for full-scale on-orbit servicing.

It was found that transportation and insertion costs dominate all other cost elements for the LEO alternate fluids resupply scenarios by a substantial margin. For the cost-minimization goal, those scenarios which rely on transporting the OMV or spacecraft itself to/from low earth orbit are very heavily penalized--particularly at WTR. Table 9 shows the LEO resupply operations cost drivers identified by this analysis.

Expendables Resupply Life-Cycle Cost Drivers ($ million 1984)					
Carrier (OMV) User Fee	=	OMV Acquisition / # OMV Engagements			
$1.0-$1.5 / Engagement		$300 (4 Units)		200-300	
Resupply User fee	=	ERM Acquisition / # ERM Engagements			
$0.5-$1.0 / Engagement		$55-$110		110	
STS-Level Transport		ETR	$2250/lb.	WTR	$6050/lb.
o OMV inert 3200#-4000#		$7.2	$9.0	$19.4	$24.2
o ERM inert 800#-1600#		1.8	3.6	4.8	9.7
o Integration Charge		1.0	1.0	1.0	1.0
o Launch & Flight Ops		1.2	1.4	1.2	1.4
o STS-Level Cost		11.2	15.0	26.4	36.3
* STS Cost Per Flight		$110M		$145M	
* Capacity * 0.75		48,750 lbs.		24,000 lbs.	
* STS Cost/Pound		$2250		$6050	

Table 9. LEO Resupply Cost Drivers

Rough Order of Magnitude (ROM) acquisition costs for the Resupply Module (RM) were estimated using parametric Cost-Estimating-Relationships (CERs) from the USAF Space Division's "Unmanned Spacecraft Cost Model" for a range of RM inert weights between 800 pounds and 1600 pounds. Pending design definition and trades in Task 2 of the Expendables Resupply study, no attempt was made to estimate software development at this time. Table 10 presents the results of this analysis.

Figure 48 presents the estimated total acquisition cost as a function of number of units acquired which was projected using the preliminary work breakdown structure also depicted in this figure. Further definition of RM development and acquisition costs were performed as part of the Programmatics effort, as found in Section 3.3.

An important "select the best DRM" affordability decision goal is to minimize the total cost of resupply to the user (receiver spacecraft) program. Except in those very few cases in which the satellite itself has an extremely low mass and/or low volume, STS transportation costs preclude either of the "return-to-earth" fluids resupply scenarios, unless concurrent full-servicing is also accomplished.

At ETR (due East launch), all of the remaining eight LEO on-orbit fluids resupply scenarios are sufficiently cost-effective to generate positive net economic benefits for each candidate spacecraft in the final list. Table 11 presents representative cost data for these scenarios.

Expendables Resupply Module ROM Acquisition Cost ($ million, 1984)						
	min (lbs)	max (lbs)	minimum $NR	minimum $TFU	maximum $NR	maximum $TFU
Structure, Thermal	700	1300	12.8	2.2	17.6	3.2
Tele, Track, Comm	50	150	3.7	1.8	9.0	4.8
Elec Power & Dist	50	150	0.7	1.0	2.2	2.1
Program - Level			8.0	2.3	13.4	4.7
DDT&E (incl TFU)	800	1600	32.5		57.1	

ROM Total Acquisition Cost
($ million, 1984)

Quantity:	1	2	4	8
Total Acquisition $:	32.5	39.1	51.6	75.4
Average Acquisition $:		19.5	12.9	9.4
Total Acquisition $:	57.1	70.5	95.9	144.3
Average Acquisition $:		35.3	24.0	18.0

Table 10. ROM Acquisition and Total Costs for Expendables Resupply Module

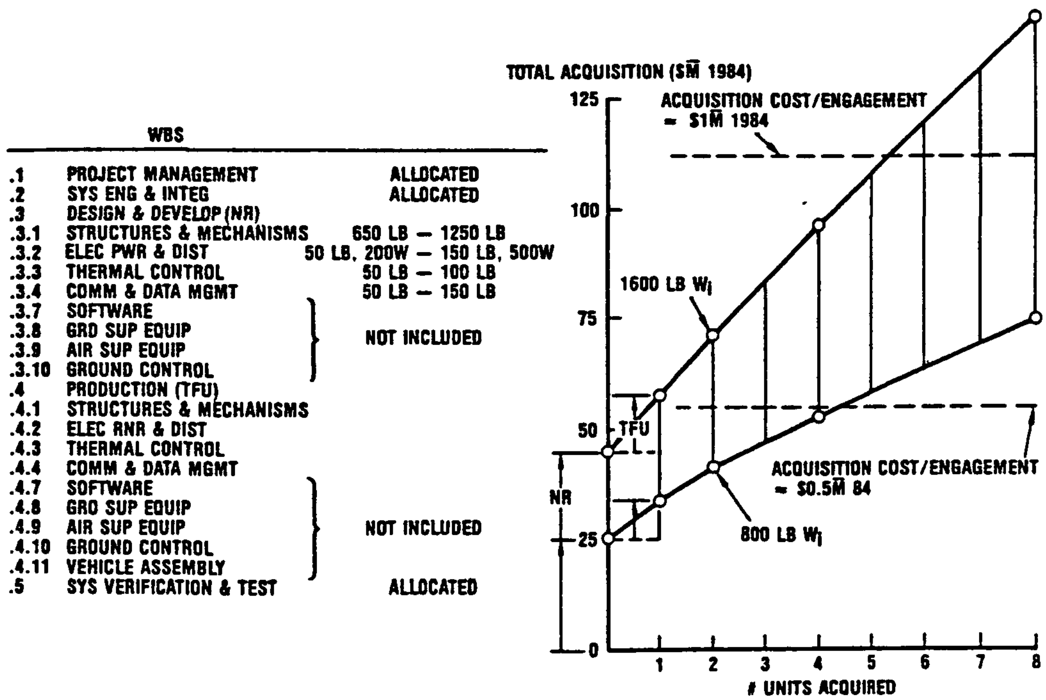


Figure 48. Expendables Resupply Module ROM Acquisition Cost

Cost (Per Resupply) to User -- ETR ($ million, 1984)								
Secnario (See Fig. 47)	1	2	3	4	5	6	7	8
OMV User Fee	1.2	1.2	1.5	1.5	1.5	1.5		
ERM User Fee	.8	.8	1.0	1.0	1.0	1.0		
OMV (Wi) Transport	4.0	8.1	.4	.4	.4	.4		
ERM (Wi) Transport	1.3	2.7	.1	.1	.1	.1	2.7	2.7
OMV (Wp) Transport	2.8	2.8	2.0	2.8	2.0	2.8		
STS Integration	.5	1.0	.5	1.0			1.0	1.0
STS Flight Ops	1.0	1.3	.8	1.4	.8	1.4	.4	.5
OMV Refuel Fee		.5	1.0	.5	1.0			
In-Bay EVA Fee						1.0	1.0	
ERM ASE Transport							.4	.4
Cost to ETR User	11.6	17.9	6.8	9.2	6.3	8.2	5.5	5.6

NOTES:
1. OMV Wp 1250 lb/engagement; 1775lb/2 engagements
2. Space-based OMV ($400M, 4 units)/300 engagements
3. ERM ASE 200 lb.

Table 11. Relative Cost Effectiveness of Selected LEO Resupply Scenarios

At WTR (sun-synchronous), only the "in-bay" fluids resupply scenarios are cost-effective across the entire polar candidate spacecraft list, and the "in-bay" scenarios may require the spacecraft to perform its own orbit transfer and/or planar/nodal maneuvering.

Figure 49 presents the relative cost effectiveness of all ten LEO resupply scenarios shown in Figure 47.

Risk and Complexity Factors

Avoidance of (and/or manageability of) risk is an important affordability goal, and each of the above 10 LEO fluids resupply scenarios was subjected to a comprehensive risk + complexity analysis using the Analytical Hierarchy Process technique accepted at the methodology review at NASA/MSFC early in the study.

Major risk + complexity issues were identified and then prioritized through several "modified Delphi" working sessions during the latter phases of Task 1. Three issues, which remain unresolved, dominated the combined risk + complexity analysis:

1) CREW SAFETY, particularly the potential exposure of astronauts and their life-support systems to toxins as a result of human involvement in propellant transfers, either in the Orbiter payload bay or at the Space Station.

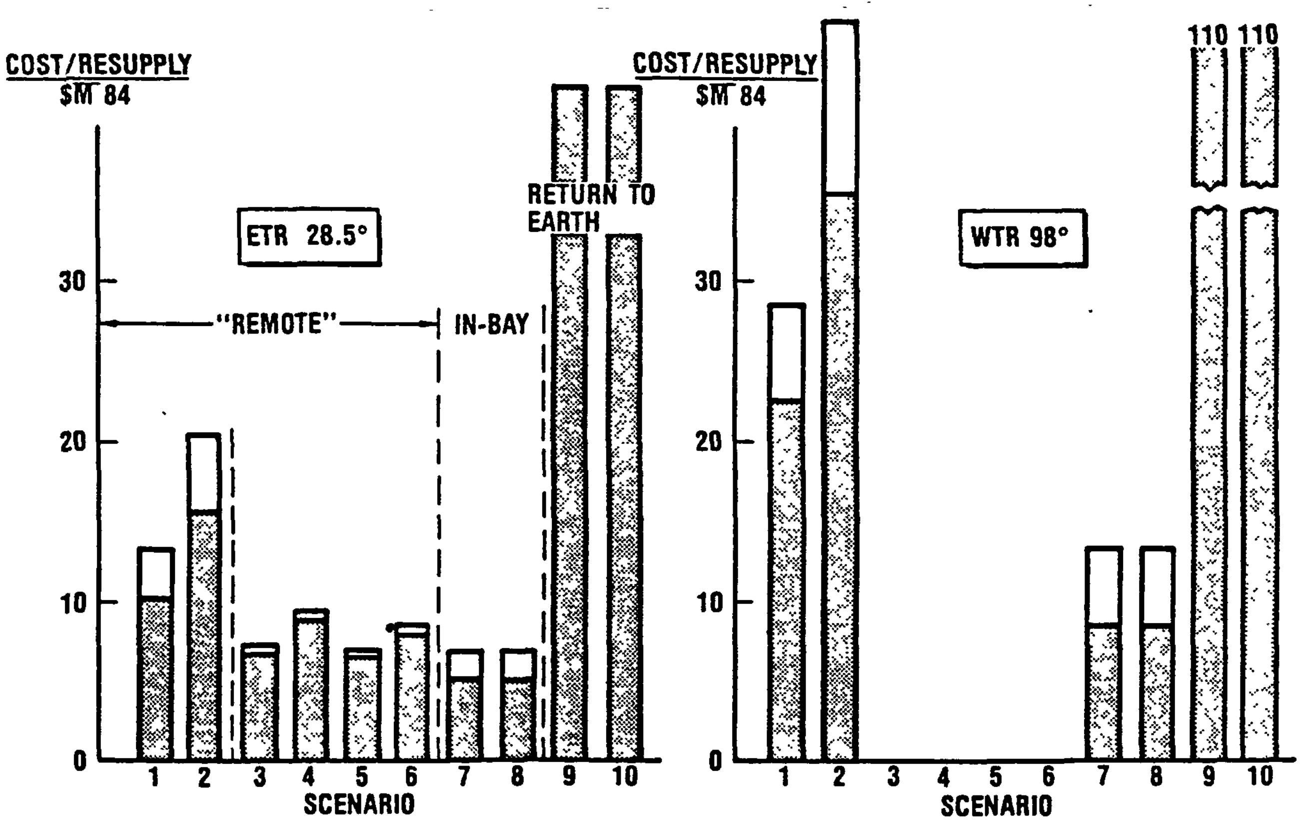


Figure 49. Relative Cost Effectiveness Estimates for Selected LEO Resupply Scenarios

2) CONFLAGRATION POTENTIAL, particularly the consequences of inadvertant combinations of bi-propellant reactants in the vicinity of either the Orbiter vehicle or the Space Station.

3) OPERATIONAL COMPLEXITY, particularly the number of orbital inclinations and ascending node crossing times involved in the multiple servicing fluids resupply scenarios.

Final resupply scenario risk evaluation results obtained using the Analytical Hierarchy Process are shown on Table 12.

C-2

g = ground based; s = space based; n = no. of engagements; ss = space station

	Crew Safety 0.61	Explosion Potential 0.26	Rendez /Dock 0.07	Contam-ination 0.06		RISK
OMV g n=2	.07	.02	.00	.00	0.10	moderate
OMV g n=1	.08	.03	.01	.01	0.12	moderate
OMV s n=2	.07	.02	.01	.00	0.10	moderate
OMV s n=1	.06	.03	.01	.00	0.10	moderate
OMV SS n=2	.04	.01	.01	.00	0.07	mod-high
OMV SS n=1	.04	.02	.01	.01	0.07	mod-high
IN-BAY IP	.01	.01	.00	.01	0.04	high
IN-BAY OMS	.01	.01	.00	.01	0.04	high
R-T-E IP	.10	.04	.00	.01	0.16	low
R-T-E OMS	.13	.07	.00	.01	0.21	very low

Table 12. Results of Risk Evaluation for Selected LEO Resupply Scenarios Using AHP

Considerations for risk include the factors shown in Figure 50.

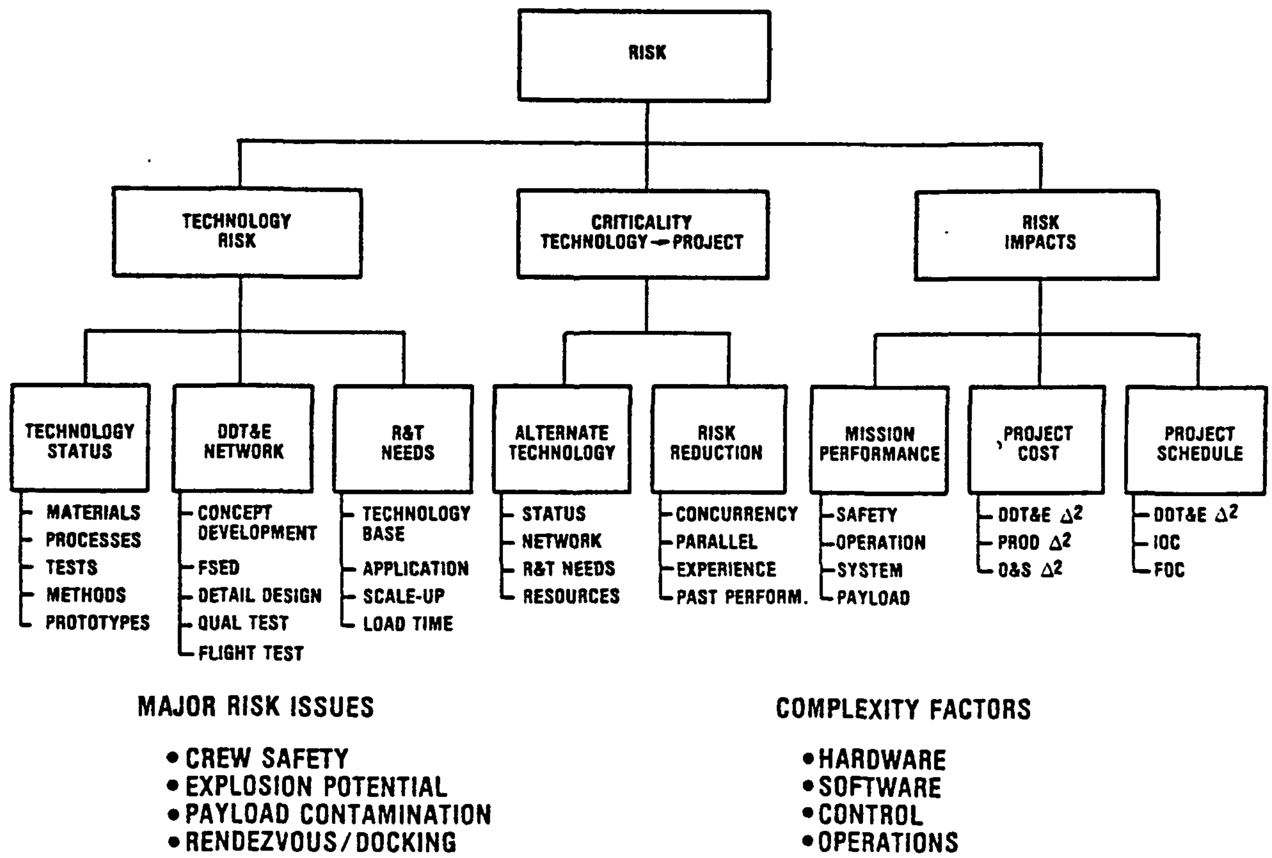


Figure 50. Risk Considerations for Scenario Evaluation

The combined risk plus complexity preference rankings for the 10 LEO alternate fluid resupply scenarios were developed by AHP specification of risk criteria importance and subsequent AHP evaluation of the contribution of each scenario to each risk goal. Mission complexity was incorporated as a candidate design reference mission level parameter, given the specifics of receiver spacecraft and fluids resupply scenario.

Combined risk plus complexity evaluation results using the Analytic Hierarchy Process as shown in Table 13.

	Crew Safety	Explosion Potential	On-Orbit Operations	RISK+CMPLX OVERALL	RANK
OMV g n=2	0.06	0.07	0.01	0.14	7
OMV g n=1	0.07	0.07	0.04	0.19	3
OMV s n=2	0.07	0.07	0.01	0.15	6
OMV s n=1	0.08	0.07	0.02	0.17	4
OMV SS n=2	0.01	0.01	0.01	0.03	10
OMV SS n=1	0.01	0.01	0.02	0.05	9
in-bay IP	0.01	0.01	0.09	0.11	8
in-bay OMS	0.01	0.02	0.13	0.16	5
R-T-E IP	eliminated--not cost-effective				2
R-T-E OMS	eliminated--not cost-effective				1

Table 13. AHP Evaluation of Combined Risk and Complexity Results

Significant risk avoidance preferences were indicated for the two "return-to-earth" resupply modes, which were clearly not cost effective for the majority of candidate spacecraft, and for the ground-based OMV single engagement scenario.

The risk evaluation criteria used for the pair-wise comparisons (which characterize the Analytic Hierarchy Process) are shown in Figure 51 along with their respective rankings as determined from the indicated results.

Considering the combination of risk and return enabled elimination of the poorest set of candidate design reference missions and retention of the most-efficient (highest return for any given level of risk, or lowest risk for any given level of return) set of candidate DRMs.

Combined estimates of return (net economic benefit) and risk for five NASA astronomic observatory and explorer programs in low inclination orbits were arrayed as shown in Figure 52 and "dominated" candidate DRMs were eliminated. For example, if the Gamma Ray Observatory program had been selected for a resupply design reference mission, either fluids resupply scenario #2 (ground-based OMV, single engagement) or scenario #4 (space-based OMV, single engagement) or scenario #8 (in-bay, using Orbiter maneuvering) would have been preferred on combined return/risk estimates.

			RANK ORDER					RANK ORDER					
NO.	SCENARIO		CREW SAFETY	EXPLOSION	CONTAMINATION	RENDEZ/DOCK	RISK/RANK	HARDWARE	SOFTWARE	CONTROL	OPERATIONS	COMPLEXITY	RISK COMPLEXITY
1	OMV g	n >1	4	5	6	8	MOD/6	6	9	7	8	MOD-HI/8	MOD-HI/7
2	OMV g	n = 1	3	3	5	5	MOD/3	6	6	7	5	MOD/5	LOW/3
3	OMV s	n >1	5	6	4	9	MOD/4	6	9	7	9	HIGH/10	MOD-HI/6
4	OMV s	n = 1	6	4	3	6	MOD/4	6	6	7	6	MOD/6	LOW-MOD/4
5	OMV ss	n >1	7	8	2	10	MOD-HI/7	9	9	7	9	HIGH/10	VERY HIGH/10
6	OMV ss	n = 1	8	7	1	7	MOD-HI/7	9	6	7	6	MOD/6	HIGH/9
7	IN BAY	IP	10	10	8	2	HIGH/10	3	4	3	4	LOW-MOD/4	HIGH/8
8	IN BAY	OMS	9	9	10	1	HIGH/9	3	2	3	3	LOW/3	MOD/5
9	RTE	IP	2	2	7	4	LOW/2	1	2	3	3	LOW/2	LOW/2
10	RTE	OMS	1	1	9	3	LOW/1	1	1	3	1	VERY LOW/1	VERY LOW/1

- 6 RESUPPLY SCENARIOS EXHIBIT "MANAGEABLE" RISK
- MULTIPLE ENGAGEMENTS/MISSION DIFFICULT

Figure 51. AHP Ranking of Risk Evaluation Criteria for Selected LEO Resupply Scenarios

Overall, the Advanced X-Ray Astronomic Facility program exhibited the lowest risk range for on-orbit resupply, since a Xenon/Methane transfer diminished the Crew Safety and Explosion Potential concerns. Conversely, the Large Deployable Reflector program exhibited the highest risk range, reflecting the unknowns involved with superfluid helium transfer technology.

An efficient set (return versus risk avoidance) of NASA low inclination orbit candidate design reference missions was developed by eliminating both the low payoff spacecraft and the high risk scenarios from further consideration as shown in Figure 53.

Efficient Return/Risk ∩ Complexity Eliminates Worst Selections

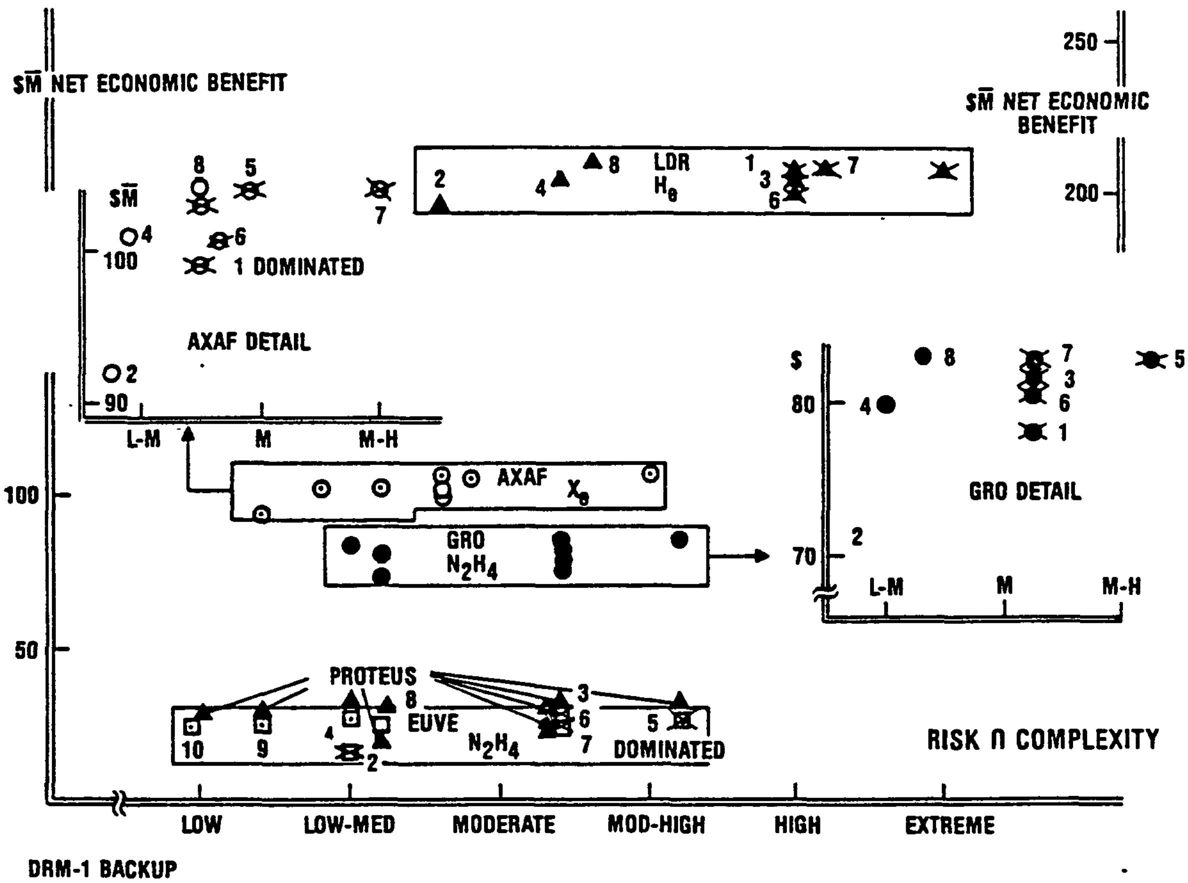


Figure 52. Risk/Economic Benefit Array for Candidate LEO Resupply Scenarios

The same return/risk methodology was applied to NASA and DoD candidates in polar inclinations and to NASA, commercial and DoD candidates in higher energy (12 hour and GEO) orbits to define a refined set of candidate design reference missions. For example, it was determined that the contingency resupply of consumables to GEO satellites in the event of their premature depletion (as opposed to the typical recovery plan of building and launching a replacement unit) was benefical.

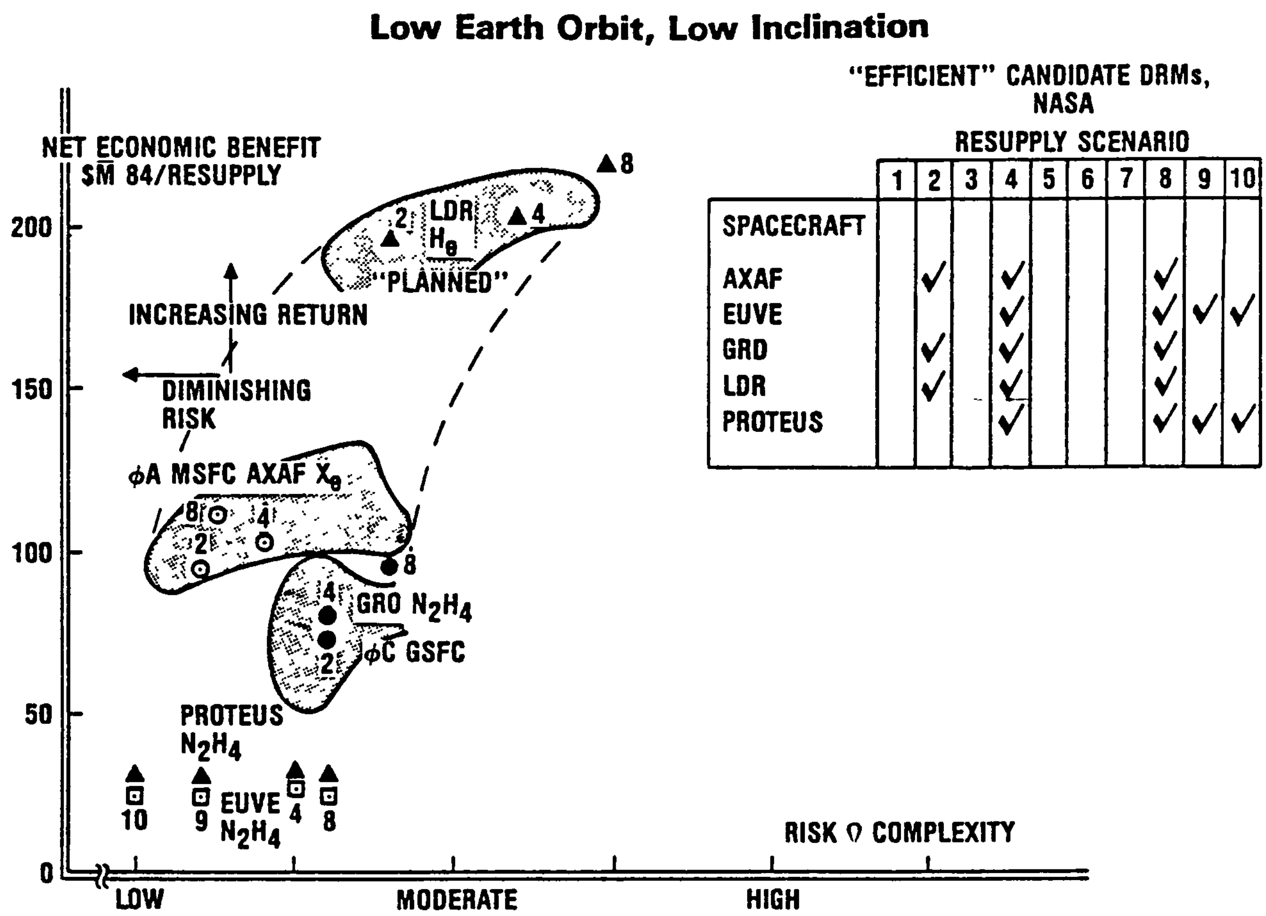


Figure 53. Efficient Set of Candidate LEO, Low Inclination Design Reference Missions

The refined set of candidate DRMs is shown in Table 14.

	Net Econ Benefit	Operation Benefit	Scenario Risk	Service Orbiter	Potential Sponsor
AXAF #2	92	special	low	YES	MSFC
AXAF #4	101	purpose	low-mid	YES	MSFC
EUVE #3	24	contingency	moderate	?	JPL
GRO #2	71	contingency	low-mod	YES	GSFC
GRO #4	80	contingency	moderate	YES	GSFC
LDR #2	195	contingency	high	?	ARC
LDR #4	204	contingency	high	?	ARC
PROTEUS #4	28	contingency	moderate	?	GSFC
DoD #2	?	operational	low-mod	?	DoD
EOS #2	10	contingency	low-mod	?	GSFC
IUS ComSat	60	contingency	moderate	NO	Commercial
GEO ComPlat	70	design rqmt	low-mod	NO	MSFC

Table 14. Efficient Set of Remote Fluids Resupply Candidate DRMs

This semi-final list of the most (return/risk) efficient candidate design reference missions for expendables resupply without concurrent servicing was comprised of nine individual spacecraft/platform programs and three alternate fluids resupply scenarios.

Three programs (AXAF, GRO and LDR) were eliminated from selection as the "best" DRM, since the extremely high marginal value of concurrent servicing to each of these programs is very likely to strongly motivate the program sponsors to choose an "in-bay" maintenance plus technology upgrade plus resupply scenario.

The AXAF spacecraft is typical of many large and complex LEO systems which will require resupply for an adequate design life. However, other types of servicing will also be accomplished concurrent with the resupply operations. (In fact, for AXAF, the resupply and module changeout will likely be accomplished without fluid transfer across an umbilical interface since its design uses experiment modules which are removeable/replaceable and contain specific fluids). Such spacecraft will likely be returned to the Orbiter for fluid transfer/modular exchange servicing since, as previously discussed, the resulting economic beneifts are much greater. Therefore, they are not recommended for remote resupply using the OMV with an add-on resupply module kit.

The final list of spacecraft candidate DRMs without concurrent maintenance all involve propellant transfer (either hydrazine or storable bi-propellant) using the remote (in-situ) resupply capability of the Orbital Maneuvering Vehicle.

The Extreme Ultra Violet Explorer (EUVE), Earth Observation System (EOS) and PROTEUS programs all exhibit limited economic payoff (relative to other candidates) and, given identical risk levels associated with on-orbit propellant transfer, were eliminated in favor of the two selected "best" design reference missions-- and Geosynchronous Equatorial Orbit Spacecraft programs in which consumables resupply is likely to be a specific design requirement to allow increased revenue from operations in GEO or reduced transportation costs to LEO through propulsion module propellant top-off or resupply of a reusable version of the satellite/propulsion module. These LEO operations could both be accomplished using an OMV/RM based at a LEO depot. Figure 54 indicates these "resupply only" DRM selections, along with their common fluids, NTO/MMH bi-propellant, on a plot of estimated risk versus net economic benefit.

Ample evidence currently exists (e.g., Insat, Intelsat VI, and NASA/MSFC's baseline OMV) to indicate an evolution from hydrazine to storable bi-propellant forms of energy storage on-orbit. The study mid-term review recommended the DRMs are shown in Figure 55.

Recommended DRMs for Remote Fluids Resupply Without Maintenance

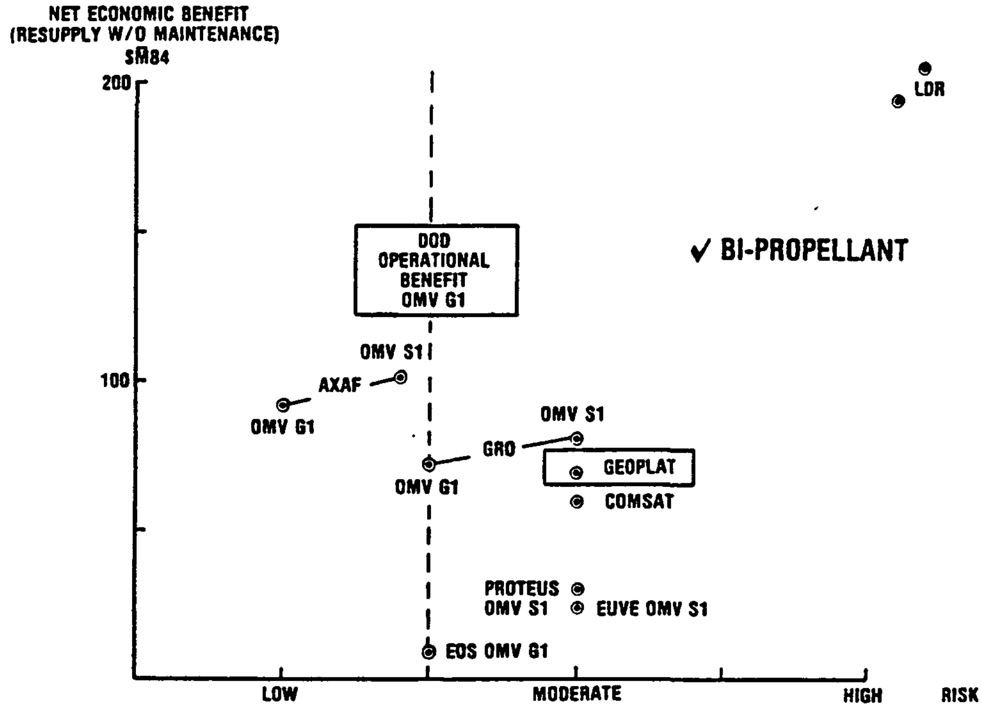


Figure 54. Estimated Risk versus Net Economic Benefit for Selected DRMs without Concurrent Servicing

Recommended Design Reference Missions

- **REF FLUIDS FOR RESUPPLY ARE BI-PROPELLANTS**
  - NTO & MMH
- **RATIONALE:**
  - LARGEST PROJECTED FUTURE QUANTITY
  - NATURAL EVOLUTION FROM N_2H_4 IN-BAY DEMONSTRATIONS
  - BASIC PROPELLANT FOR NUMEROUS FUTURE SPACECRAFT WHICH CAN BENEFIT FROM RESUPPLY
    - GEO { LARGE COMSATS (ALREADY USED ON INTELSAT VI) / FUTURE LARGE GEO PLATFORM
    - LEO { REFUELING OMV & HIGH PERFORMANCE STORABLE STAGE / POTENTIAL RESUPPLY DEPOT REQUIREMENT
- **REFERENCE QUANTITY TBD**
- **CANDIDATE MISSIONS**
  - ✓ LARGE GEO PLATFORM
  - ✓ FUTURE LARGE COMSATS
  - ✓ RESUPPLY IN LEO
    - DOD (MANEUVERING)
    - ORBITAL VEHICLES

Figure 55. Mid-Term Review DRM Recommendations

Based on these recommendations and comments at the mid-term review, additional work was performed to summarize the range of fluid requirements. The initial focus of the effectiveness analysis was on the end users of resupply services, such as particular satellites. In the course of this study, it became apparent that the resupply module was most effective when considered an integral part of the entire space infrastructure. The variety of potential uses for a resupply module, other than simple resupply, presorted in the preceeding pages are summarized in Figure 56.

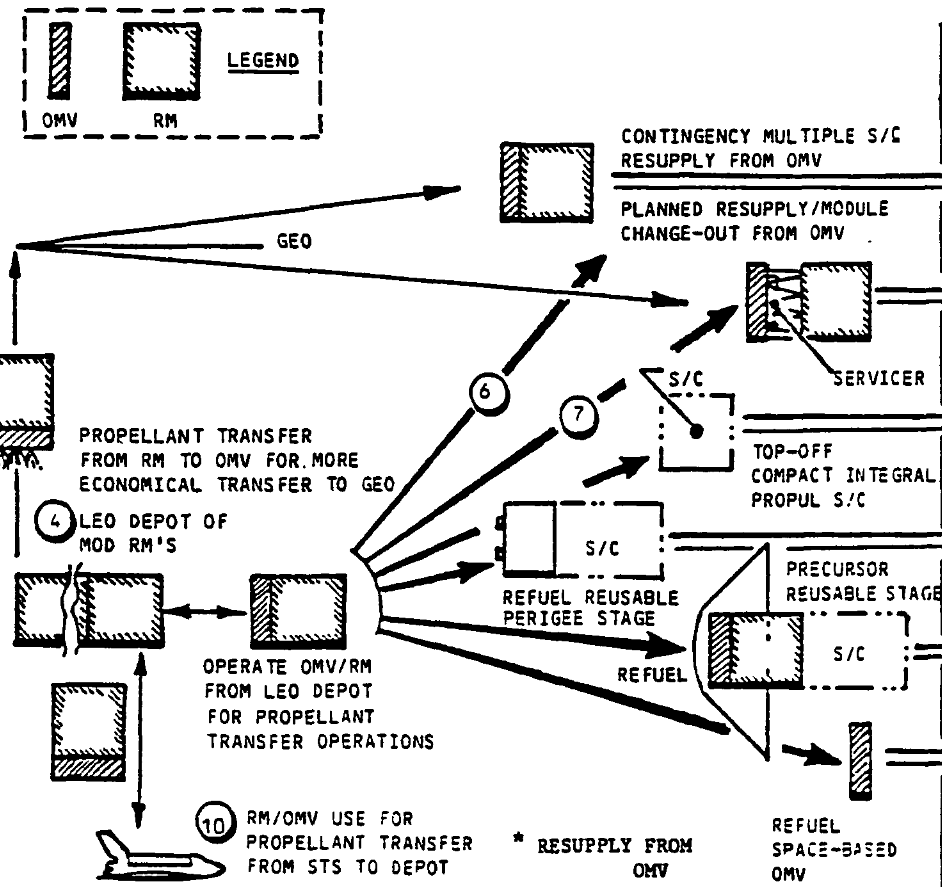
The resupply scenarios developed in the effectiveness analysis for LEO and GEO operations were used to create mission event timelines and to better understand the operational issues in remote resupply. They were the basis of the eventual final set of Design Reference Missions shown in Figure 56. This set of DRMs included a broader performance range of potential mission scenarios (with a consequent sizing impact for the resupply module). The broad operational range and complexity that might be required of a resupply module necessitated a large number of Design Reference Missions, without specifying particular satellite programs. We identified near and far-term resupply candidates with priority requirements, bounded the range of performance requirements, and proceeded with the concept Definition Task.

3.2 Concept Design (Task 2)

The Concept Definition task was divided into two subtasks: 2.1 - Develop Resupply Module Design, and 2.2 - Generate Demonstration Concept. These tasks incorporated inputs from Task 1 including: Design Reference Missions and Scenarios, Missions Operations Requirements, System/Subsystem interface requirements, and systems concepts. The objective of Subtask 2.1 - Concept Design - was to develop a resupply module concept design including provisions for: propellant tankage, pressurant storage bottles, propellant and pressurant transfer systems and supporting mechanisms which are consistant with the results of Task 1. In conjunction with this design definition those impacts to the user spacecraft, which are required to support on-orbit remote resupply, were identified. The objective of subtask 2.2 was to develop a flight demonstration concept which would provide sufficient validation of the resupply module concept defined in subtask 2.1. The design and test concepts defined in subtask 2 are intended to serve as a basis for the programmatics section of the study.

3.2.1 Operational System Configuration Selection

The ten basic mission scenarios/DRM's indicated in Figure 56 were evaluated for sizing requirements to the Expendables Resupply Module propellant requirements from 6,000 to 45,000 lbs. and helium requirements from zero to 90 lbs. Scenario two is the driving requirement for GEO operations. In this mission the full resupply module capacity is used to supply enough propellant to transfer an OMV plus its servicer kit to GEO. Resupply in this case would be directly from the OMV but the OMV would not be recommend from GEO. Scenario five is the driving requirement for LEO remote resupply operations. This scenario is the refueling of a high performance reusable storable OTV. Lesser levels of resupply module capability are provided by off-loading the module rather than removing tankage. The objective is to adapt one basic "core" resupply module design to many different uses.

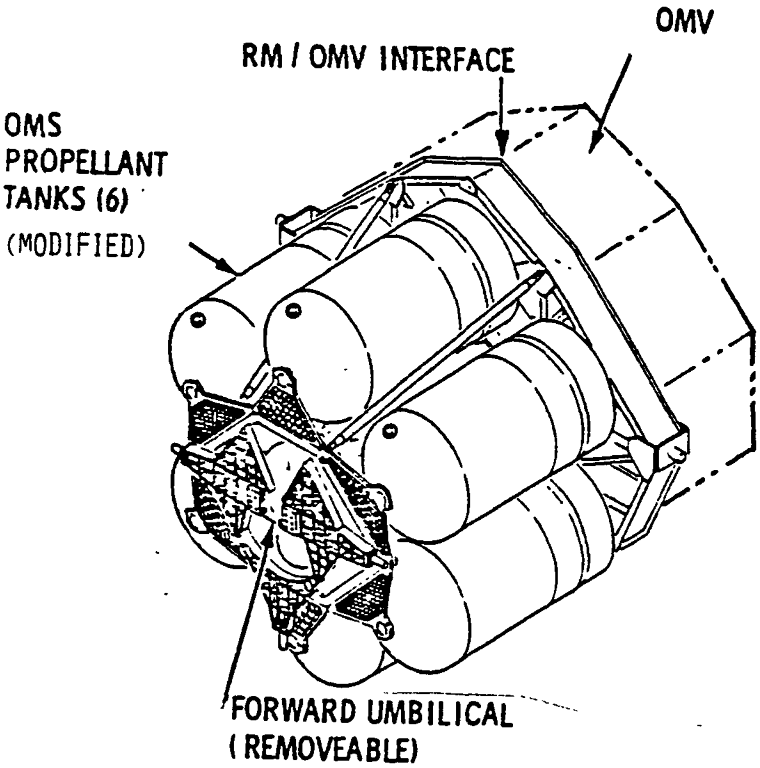


POTENTIAL MISSION SCENARIO	EXAMPLE RM PROPELLANT LBS.	EXAMPLE RM HELIUM LBS.
1	38,000 (1,000*)	——
2	45,500 (2,000*)	4*
3	20,000	——
5	45,000	30
8	45,500	90
9	6,000	12

Figure 56. Summary of Resupply Module Scenarios/Fluid Transfer Requirements

The ERM in the various scenarios served as an OMV extended mission kit, a LEO refueling ERM, a LEO propellant and helium storage depot, a precursor reusdable upper stage, and a LEO propellant depot for STS scavenged storable propellants.

A versatile, low-cost resupply module concept resulted from the trade analyses and design definition phases of the study. Some of the various trades and design characteristics included propellant transfer alternatives, weight trades for a pump-fed versus pressure regulated transfer systems, propellant system schematics, transient and stady state pressure drop characteristics, fluid interface panel design, thermal control provisions, electrical power requirements and distribution, avionics and data processing interfaces, major subsystem weight breakdowns, user spacecraft impact assessment, general configuration layouts and key features. To provide the required propellant capacity for the envelope of propellant quantity requirements associated with the selected resupply module scenarios, the resupply module configuration depicted on Figure 57 was selected. This concept was selected from among several competing approaches mainly on the basis of overall structural efficiency in combination with utilization of existing hardware for low development cost. The resupply module is supported at its forward end by an existing IUS upper stage forward cradle. This cradle includes load equalization capability which reduces the structural redundancy between the resupply module and the orbiter by one degree. This allows a minimum of structural weight on the forward end of the resupply module for attachment to the orbiter payload bay. The resupply module uses modified Orbiter OMS tanks. The modifications include a lengthened cyclinderical section and a zero "g" PMD.



KEY CHARACTERISTICS

- DRY WT FOR RESUPPLY MODULE ONLY (AS SHOWN) = 7960 LB
  - BI-PROPELLANT TRANSFER ONLY IN LEO (FROM DEPOT)
  - 45,000 LB USABLE PROP
  - ADD 1171 LB FOR HE RESUPPLY CAPABILITY
- (6) STRETCHED OMS TANKS WITH NEW PMD FOR 0-G PROP MGMT AND ULLAGE POSITIONING
- IUS FWD CRADLE USED IN CONJUNCTION WITH INTEGRAL AFT LONGERON ATTACH PTS FOR MINIMUM FLIGHT WEIGHT
- OMV SUPPORTED FROM RM IN PAYLOAD BAY
- DRY WT AS PROPELLANT KIT FOR OMV ONLY (INCLUDES SIMPLIFIED PMD'S AND ELIM OF COMPRESSORS/BATTERIES FOR) = 7200 LB

Figure 57. RM/OMV Configuration

As shown on Figure 58, the basic structural components of this Resupply Module are very simple, contributing to meeting the objective of low cost. Yet, the configuration is very efficient structurally. All fore and aft loads and part of the vertical loads are reacted at two payload bay longeron attachment points in the main structural bulkhead. As a representative attachment to the OMV, six bolts are provided. For the case where the Resupply Module is detached from the OMV in GEO, these bolts may be explosive. The main bulkhead also had a keel attachment for the Orbiter Payload Bay.

The tanks are supported by the main bulkhead for fore and aft, as well as lateral and vertical loads. They are stabilized at their forward ends by machined fittings which attach to the cylindrical center body of the Resupply Module. This body provides the main bending stiffness of the Resupply Module. NASTRAN finite element analysis showed that this concept meets the Orbiter fundamental response stiffness requrement of 7 Hz even when fully loaded with propellant. Most elements of this Resupply Module structure are machined aluminum plate, including the center body which assembled from quarter sections that are rolled to proper shape after machining.

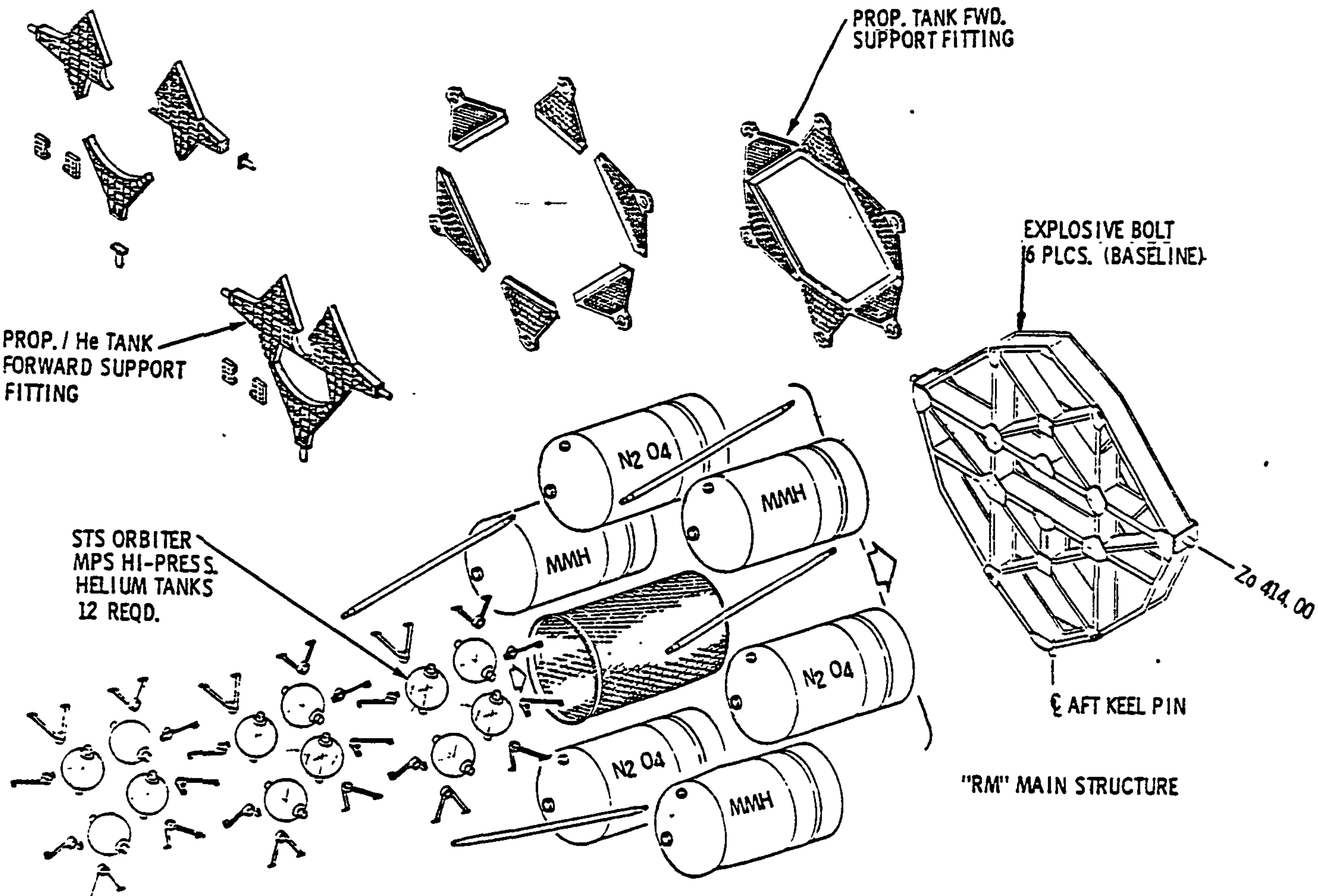


Figure 58. Resupply Module Structural Concept

A design layout (Figures 4 & 5) was produced of the selected adaptable Resupply Module configuration. These layouts were used to develop weight estimates for the resupply module structure. Two adaptations are depicted. In both cases the OMV is supported from the resupply module in the orbiter payload bay (OMV support fittings could be removed to reduce weight and cost). The combination OMV/ERM would be deployed from the payload bay using the orbiter RMS. The adaptation depicted with the truss interstage allows sufficient volume between the OMV and ERM for the OMV's servicer kit (for missions in which concurrent module change-out type servicing is required).

3.2.2 Safety and Contamination Control Requirements and Considerations

The general requirements presented in Chapter 2 of "Safety Policy & Requirements for Payloads Using the Space Transportation System, NHB 1700.7A," were used as a guideline in defining the Resupply Module subsystem's redundancy and safety requirements. Specifically, subsystem/functions that could result in a catastrophic hazard (i.e., potential for personal injury, loss of orbiter, space station, or other free flying vehicles) such as a failure in the propellant transfer or hazardous waste scavenging subsystems, should be controlled by a minimum of three independent mechanical inhibits. Also, since the pressurant system is considered a critical hazard (i.e., potential for damage to the orbiter, space station, or other free flying vehicles) and, as such, requires to be controlled by two independent inhibits.

The safety requirements presented in Chapter 2 of "Safety Policy and Requirements for Payloads Using the Space Transportation Systms," NHB 1200.7A, December 9, 1980 are used as the safety and redundancy design criteria for all orbiter payloads. Contact GSFC NAS5-26152 (Multi-Mission Modular Spacecraft In-Orbit Refueling Study) results, which included the Orbital Resupply System's (ORS) interpretations of the NHB 1700.7 safety requirements, was used as a guideline in explicating the Resupply Module (RM) System's redundancy and safety tolerances, defined by NHB 1700.7. Some of the more critical RM related items are listed as follows:

a) For hazardous functions, the Resupply Module must tolerate a minimum of credible failures and/or operator errors determined by the hazard level. This criterion applies when the loss of a function or the inadvertent occurrence of a function results in a hazardous event.

b) No single failure or operator error shall result in damage to STS equipment, space station, satellite, or resupply module; or in the use of contingency or emergency procedures.

c) No combination of two failures, operator errors, and RF (radio frequency) signals shall result in the potential for personal injury, loss of the orbiter, space station, satellites, resupply module, ground facilities, or STS equipment.

d) No single mechanical or electrical failure will endanger personnel, orbiter, space station, satellite, resupply module, or shuttle payload. As a possible design goal for the resupply module, no single mechanical or electrical failure will preclude the transfer of propellant.

e) A function that could result in a critical hazard must be controlled by two independent inhibits. For orbiter and possibly space station deployed payloads (e.g. Resupply Module), monitoring and safing of the two inhibits will not be required when both inhibit power and control circuits for the function are connected to a bus that is not energized until the payload reaches a safe distance from the orbiter or space station.

1. Pressurant Considerations - The pressurant system is considered a critical hazard and, as such, requires two independent inhibits (e.g. isolation valves, burst disks). A design goal of three independent inhibits was considered for the pressurant fill and ullage vent systems of the Orbital Resupply System. However, system cost and complexity may force the use of only two shutoffs for the pressurant fill and vent system for the resupply module.

2. Hazard Detection & Safing - The need for hazard detection and safing by the flight crew to control time - critical hazards shall be minimized and will be implemented only when an alternate means of reduction or control of hazardous conditions is not available.

f) A function that could result in a catastrophic hazard must be controlled by a minimum of three independent inhibits, whenever the hazard potential exists. Paragraph 202-2 of NHB 1700.7 states that one of the three inhibits must preclude operation by RF commands. When remotely controlled vehicles (e.g. OMV, Satellites, RM) are resupplied by the Orbiter and/or Space Station, paragraph 202-2 imposes the requirement for an electrical data/command interface to exist between the transfer and receiver vehicles; hence, establishing manual control(through the Orbiter and/or Space Station) over the propellant transfer process and eliminating the use of RF commands. However, when remotely controlled vehicles are resupplied by other remotely controlled vehicles (e.g. RM resupplying a satellite or OMV, OMV resupplying a satellite) the entire propellant transfer process would have to be controlled through RF commands. Under this situation, paragraph 202-2 may not be considered applicable and may be modified accordingly.

1. Propellant Transfer - The Expendables Resupply Module will rquire three independent mechanical inhibits (e.g. isolation valves, manifold valves, QD's) for its transfer system. Any residual propellant in the receiver tank that requires disposal prior to, during, or subsequent to transfer operation may be captured and stored in disposal tanks aboard the resupply module. However, whenever applicable, decompostion reactors could be used to disassociate the residual propellant into innocuous gases, which in turn would be vented overboard. Propellant contaminated ullage gases in the receiver tank that require disposal prior to transfer operations should be handled using the same techniques discussed for residual propellant disposal.

2. Deployment and/or Separation - The premature deployment or separation of the ERM to a condition where it cannot withstand STS-induced loads (including landing) or will prevent safe entry of the Orbiter is a catastrophic hazard. The inhibit mechanisms must be designed to preclude inadvertent mechanical operation in the induced environments.

3. Deployment/Extension Preventing Payload Bay Door Closure - If during planned ERM operations an element of the ERM or any ERM support equipment violates the payload bay door envelope, the hazard of preventing door closure must be controlled by independent primary and backup methods, and this combination must be two failure tolerant.

4. Retrieval of Expendables Resupply Module) - ERM should have the capability to return hazardous systems to a safe condition, that is, to meet the requirements under sections 2.1 and 2.2 of NHB 1700.7A. ERM shall provide verification to the Orbiter, Space Station, or the ground that safing has been accomplished prior to its retrieval, and while still at a safe distance from the Orbiter or Space Station.

5. RF Energy Radiation - ERM produced radiated fields from ERM transmitter antenna systems in excess of levels defined for payload bay door open operatioins in JSC 07700, Volume XIV, Attachment I is a catastrophic hazard. Three independent inhibits to radiation in excess of these levels must be provided.

Structural safety requirements listed in the GSFC Study, Multimission Modular Spacecraft In-Orbit Refueling NAS5-26152, as applicable to ERM, are given below.

a. The structural design will provide ultimate factors of safety no less than 1.4 for all mission phases except emergency landing. When failure of structure can result in a catastrophic event design will be based on fracture control procedures.

b. The structural design will comply with the ultimate design load factors for emergency landing loads are specified in JSC07700, Volume XIV, Attachment I.

c. The structural design will comply with the stress corrosion requirements of MSFC-SPEC-522.

d. All pressurant and propellant tanks will have a minimum burst-to-operating pressure ratio of 4:1 in the prescence of ground personnel. The other considerations described in NHB 1700.7 for pressure vessels also apply.

e. All lines and fittings will have an ultimate safety factor of no less than four. Components such as valves, regulators, etc., will have an ultimate safety factor of no less than 2.5.

A burst disk in series with a relief valve should be considered for thermal relief mechanisms. Each element should be designed to operate when the internal pressure level exceeds 1.1 times the maximum operating pressure. The relief valve should be located such that it can be checked without violtating the remainder of the system. The propellant relief assembly should be considered for connection to a disposal container.

To insure the integrity of the Shuttle Orbiter's Payload Bay environment and the local environment of the Space Station and other free-flying spacecraft, a disconnect/line purging system should be incorporated into the propellant transfer systems of space vehicles requiring on-orbit refueling.

Any propellant or contaminated ullage gas leaks, spills, and/or venting would create a tenuous atmosphere around the space vehicles involved. The contamination to the local environments of these spacecraft could obstruct the field-of-view of the optical telescopes and highly sensitive cryogenically cooled sensors on these vehicles.

In addtion to obstructing the field-of-view, if vented residual propellants and ullage gases were to come in contract with experiments on board the Orbiter, Space Station, or satellites, the caustic and toxic characteristics of monopropellants and hypergolic propellants could destroy the optical and/or the cryogenically cooled surfaces of these experiments.

Another area of concern, in conjunction with propellant spills and venting, are Extra Vehicular Activities (EVA's). EVA's may necessitate maneuvers in and/or near the vicinity of vented propellant clouds or spills. Astronauts could conceivably return to the Orbiter or Space Station in propellant contaminated space suits, creating a potentially hazardous condition within the interior environment of these vehicles.

Venting of residual propellant and ullage prior to propellant transfer operations within or in close proximity to the Orbiter payload bay is not acceptable. According to orbiter safety policies, hazardous materials shall not be released or ejected in or near the Orbiter.

Because of the numerous concerns associated with propellant/ullage leaks, spills, and venting, it may be in the best interest of both the supply and receiver vehicles to implement a hazardous waste storage/purge subsystem. Regulated pressure gas (which would be supplied by a resupply module, or any other resupply system), would flow through the propellant and ullage transfer purge circuits and clear all refueling disconnects and adjoining lines of residual propellants, prior to disengaging the resupply and receiver vehicles. A simplified disconnect/line purge circuit is illustrated in Figure 59. The safing of these propellant and ullage transfer circuits would help minimize the possibility of a leak or spill emanating from the propellant/ullage transfer interface panel.

3.2.3 Pump Versus Pressure Regulated Propellant Transfer

Two different transfer systems were studied to transfer 45,500 lbs of bipropellant from the Expendables Resupply Module. The first system is a pressure regulated system with propellant tanks at 250 psia. This system will be referred to as the pressure regulated system. The pressure in the pressurant tanks in this pressure regulated bipropellant transfer system was allowed to drop from the maximum operating pressure to 500 psia.

Two alternatives for the first system studied looked at pressure regulated system weights using existing pressurant tanks with a maximum operating pressure of 4,500 psia. A new 6,000 psia maximum operating pressure tank system was also evaluated.

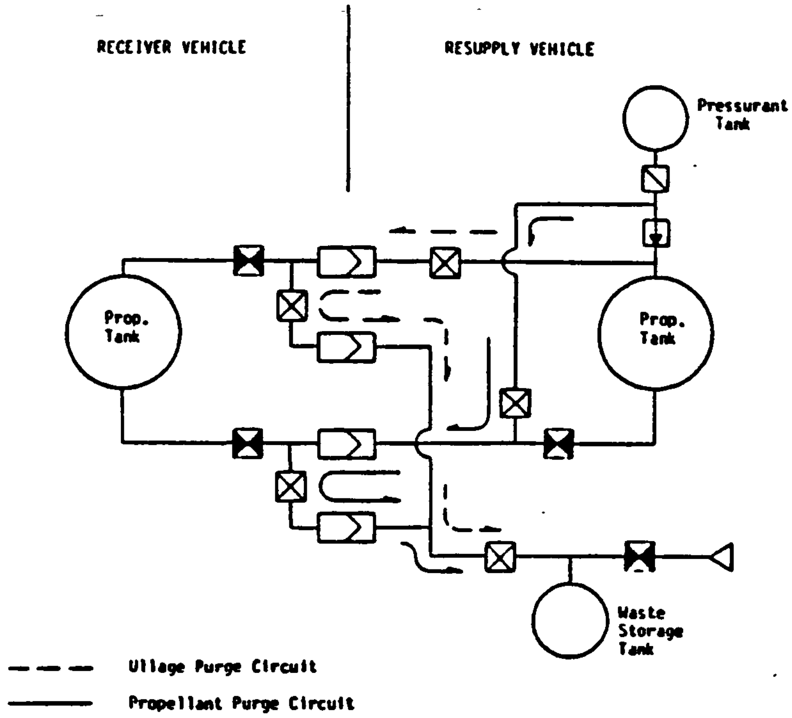


Figure 59. Example Disconnect/Line Purge and Waste Storage Subsystem

The second bipropellant transfer system used a bipropellant pump along with a pressurant system as shown in Table 15. This system will be referred to as the combination pump and pressurant system. In this system the propellant tanks are pressure regulated to 50 psia. The pressurant supply pressure is allowed to drop from the maximum operating pressure of the pressurant tank to 300 psia. In this system the propellant is transferred with a pump and a sufficient amount of helium must be carried to replace the volume vacated by the transferred bipropellant and also pressurize the propellant tank to 50 psia. This lower propellant tank pressure requires fewer pressurant storage tanks and allows a pressurant system weight savings. The combination pump and pressurant system weight has two major elements.

One element comprises the pump system and the other the pressurant tank system weight. The pump system weight depends heavily on the power source. The battery is the heaviest component of a pump system as can be seen in Table 16

	Types of Batteries		
Parameter	Ag/Z_n	$Li/T_i\ S_2$	$Li/T_i\ S_2$ (Developed)
Pump and Motors	30 lbs	30 lbs	30 lbs
Battery	540 lbs	378 lbs	330 lbs
Electronics	40 lbs	40 lbs	40 lbs
Heat Pipe	15 lbs	15 lbs	15 lbs
Total	625 lbs	463 lbs	415 lbs

Table 16. Propellant Transfer Pump System Weights

These pump system weights are sized to transfer 45,5000 lbs of bipropellant in approximately 7 hours.

Two alternatives for the combination pump and pressurant systems weight using existing pressurant tanks and new 6,000 psia pressurant tanks were also be evaluated. Alternative number one is to transfer 45,500 lbs of bipropellant with a pressure regulated systems using exisiting pressure tanks. To accomplish this, 10 Orbiter Main Propulsion System (OMPS) helium tanks with a total pressure regulated system weight of 925 lbs are required. (see Table 15) Alternative number two is to transfer 45,500 lbs of bipropellant with a pressure regulated system using 6,000 psia pressurant tanks. Four combinations of 6,000 psia tanks were considered and are presented in Table 17.

No. of Tanks	Volume of each tank, in^3	Total pressure regulated system weigt, lbm
2	25,170	609
3	16,648	608
4	12,508	602
5	10,034	608

Table 17. 6000 Psi Helium Tank Combinations

Alternative number three is to transfer 45,500 lbs of bipropellant using the combination pump and pressurant system with existing pressurant tanks. Two OMPS helium tanks with a weight (tanks + helium) of 185 lbs are required. The total weight of the combination pump and pressurant system would be the weight of the tanks plus the weight of the pump system. The weight of the developed Li/TiS_2 battery powered pump will be used since it should be attainable in the near future. Therefore, the total weight (tanks plus helium plus pump system) to do the above transfer wil be 600 lbs for the system with 2 OMPS tanks (See able 15),

Alternative number four is to transfer 45,500 lbs of bipropellant using the combination pump and pressurant system with new 6,000 psia pressurant tanks. The weights and volumes of the different 6,000 psi tank combinations can be seen below in Table 18.

No. of Tanks	Volume of each tank, in^3	Helium system weight, lbm
2	5,189	125
3	3,435	124
4	2,591	125
5	2,081	130

Table 18. Helium System Weight Using 6000 Psi Tank Combinations

The above pressurant supply went from an operating pressure of 6,000 pisa to a minimum pressure of 300 psia to displace 45,500 lbs of bipropellant. The toal combination pump and pressurant system weight (weight of tanks plus helium plus pump system) for the four different combination of tanks can be seen in Table 19.

No. of Tanks	Total, Combination Pump and Pressurant System Wt. (Lbs)
2	540
3	539
4	540
5	545

Table 19. Combination Pump and Pressurant System Weights

Using the combination pump and pressurant system to transfer 45,500 lbs of space storable bipropellants yields 54% savings in weight over a pressure regulated transfer system which uses existing MPS helium bottles. The new 6,000 psi composite helium bottles are used in the transfer process however the weight savings are not significant (Table 15). It is concluded that for systems utilizing existing helium bottles, a combination pump and low pressure regulated propellant tank is most weight efficient.

3.2.4 Propellant Transfer Process Selection

Numerous propellant transfer scenarios and timelines can be projected for the on-orbit resupply of propellants to satellites, the OMV, propellant tank farms, etc. The various scenarios depend upon the type of tank being resupplied (i.e. screen, vane, or diaphragm), the type of tank being used to supply propellant, and the type of propellant being resupplied (NTO, MMH or N_2H_4). All of the scenarios however can be placed in one of three general categories: ullage recompression, ullage vent, and ullage exchange. A brief discussion of all three methods of propellant transfer are presented in the following sections. A summary of their key advantages and disadvantages are tabulated in Table 20. The rationale for selecting the ullage exchange scenario as the principle propellant transfer process for the resupply module is also presented.

The ullage recompression method of propellant transfer will be used to resupply tank systems which operate in blowdown. This is assumed feasible for either screen, vane, or diaphragm tanks with either mono or bi-propellants. The resupply would ideally be performed from a system pressure regulated at a pressure equal to or greater than the beginning of life blowdown pressure in the receiver systems (typically about 350 psia). If this is not possible a liquid pump will be required in the resupply system to obtain the higher pressures in the receiver system. The major advantages to propellant transfer using this method are (1) its mechanical and operational simplicity and (2) resupply of the pressurant gas is not required, though it may be necessary to periodically make up the pressurant gas which is lost through propellant saturation. Figure 60 depicts a ullage recompression type of resupply.

A thermal analysis of the adiabatic recompression of propellant tank ullage gas reveals that the heat generated by this process can exceed the maximum operating temperature limits of existing propellant tanks, the auto-decomposition temperature of hydrazine (310°F), and the auto-decomposition temperature monomethyl-hydrazine (400°F). A representative adiabatic,

Table 20. On-Orbit Propellant Resupply Options

Transfer Options	Main Advantages	Main Disadvantages
Ullage Recompression	* Mechanical Simplicity; No provisions required for venting and repressurizing. * Operational Simplicity; Only Propellant is transfered.	* Recompression of the receiver tanks ullage bubble requires a large heat dissipation. * The pressure of the resupply propellant must be higher than the beginning-of-life receiver tank pressure.
Ullage Venting	* Lower heat dissipation requirements than than ullage recompression resupplies. * Low pressure propellant resupply.	* Mechanically more complex than ullage recompression resupplies. * Operationally more complex than ullage recompression resupplies. * Pressurant resupply required. * Overboard venting of propellant saturated ullage gas could be a contamination concern. Implementation of a waste scavenging subsystem for storage of ullage gas is not feasable. * Can utilize only those PAD's capable of positioning the prop. tank's ullage bubble. * Propellant metering required.
Ullage Exchange	* Constant pressure resupply. * Negligable compressve heating effects. * Receiver tank ullage is transfered over to the supply tank, and reused. A large waste scavenging subsystem would not be required. * Most versatile of all transfer options -usable in other resupply scenarios, depending on receiver vehicle requirements.	* Mechanically more complex than ullage recompression resupplies. * Operationally more complex than ullage recompression resupplies. * Pressurant resupply required. * Propellant transfer pumps necessary for this transfer option. (pumps are a possible option for ullage venting & recompresion) * Can utilize only those PAD's capable of positioning the prop. tank's ullage bubble. * Propellant metering required.

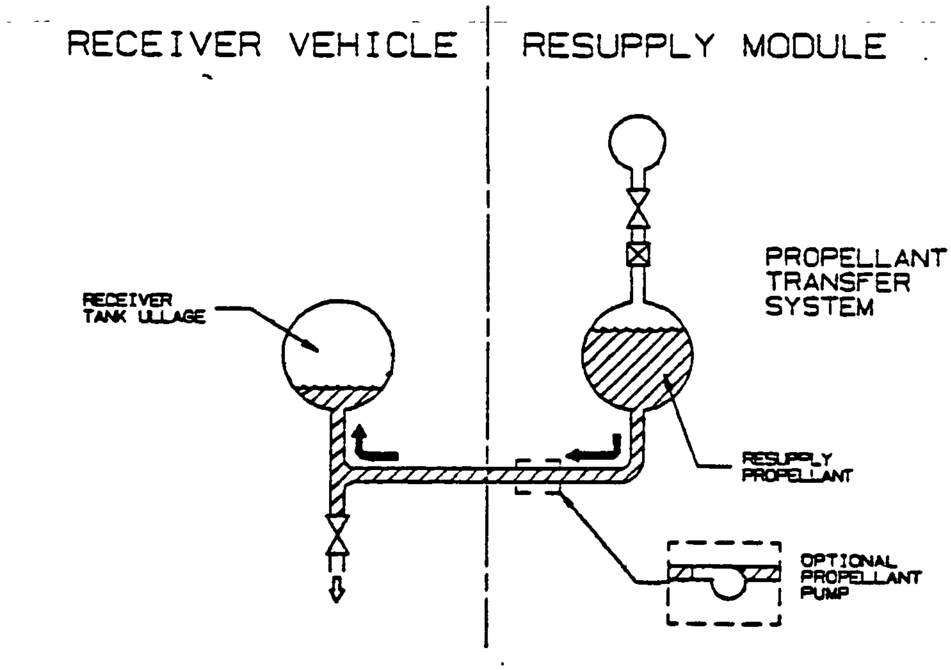


Figure 60. Ullagae Recompression Propellant Resupply

thermal loading profile for ullage recompression resupply is illustrated in Figure 61. Even though the extreme temperatures calculated by the adiabatic analysis are not an exact representation of real-life compressive heating, the trends illustrated by the thermal profiles show that excessive amounts of heat could be generated by a ullage recompression type of propellant resupply.

To transfer propellant using the ullage recompression method only one mechanical connection is required, that being the propellant fill line. Additional propellant disconnects would provide for system redundancy, purging, and safing. The propellant quantity transferred could be measured with flow meters or quantity gaging on the supplier and receiver.

The ullage exchange scenario is used for resupplying pressure regulated propulsion systems. A simple pictorial representation of this resupply scenario is depicted in Figure 62. Ullage gas is displaced out of the receiver vehicle's propellant tanks as the incoming resupply propellant fills the tank. The ullage gas is then transferred to the resupply module's propellant tanks, filling the volume left by the transferred propellant.

This resupply scenario is a constant pressure process, requiring the use of propellant pumps for the transfer of resupply propellant. However, current bipropellant seal technology may find it difficult to produce a seal which can endure the cycle life and endurance specifications required by the resupply module's propellant pumps. An advantage to having a constant pressure, propellant transfer process is the absence of extreme compressive heating effects. The receiver tank ullage gas is displaced out of the tank rather than recompressed; thus eliminating the need to dissipate any large quantities of heat (as would be generated if the ullage were recompressed).

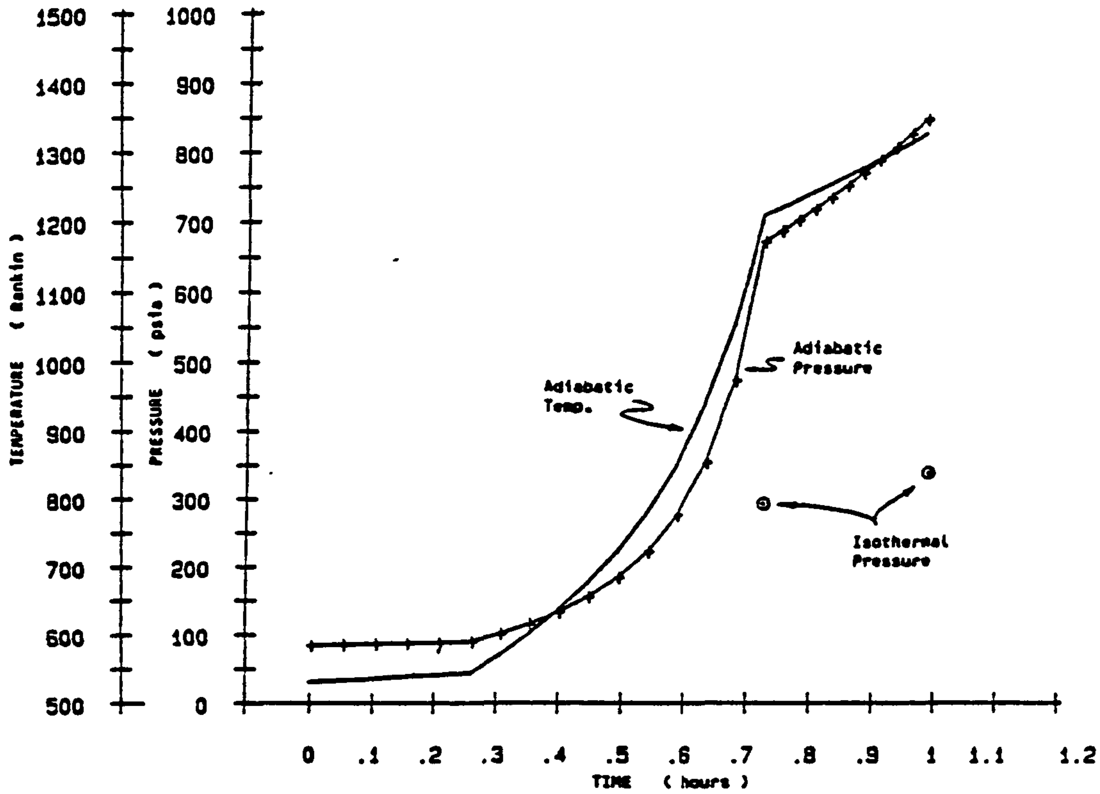


Figure 61. Ullage Recompression Thermal and Pressure Profiles (Resupply Volume = 144,500 cu.in.)

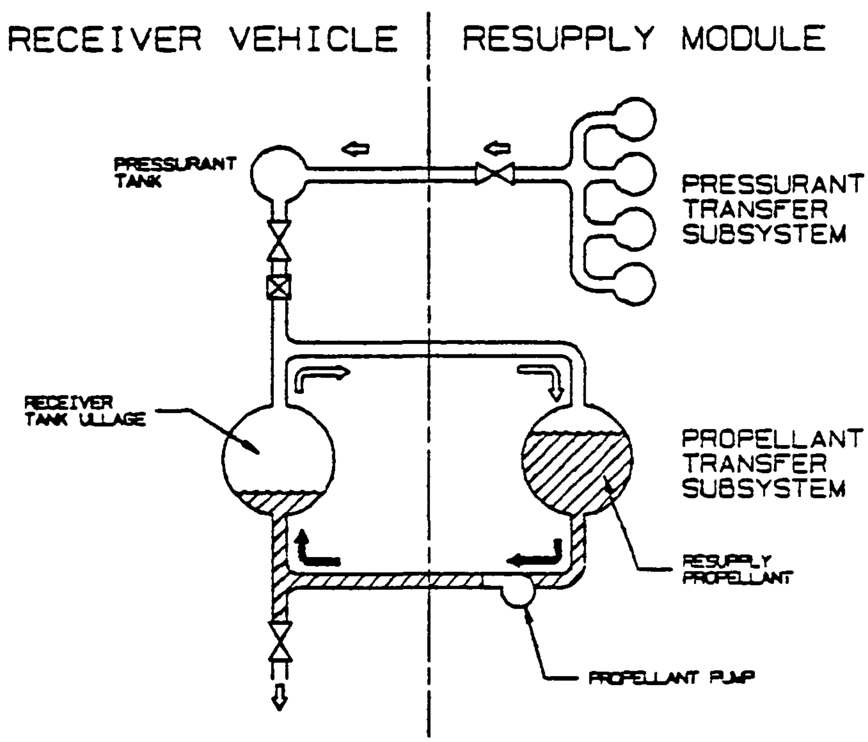


Figure 62. Ullage Exchange Propellant Resupply

Flowmeters would be used to calculate the quantity of propellant transferred to the receiver vehicle. A pressure, volume, and temperature (PVT) gaging system will not work on a resupply system of this type, due to the interconnections of the ullages.

Resupply modules designed for ullage exchange resupply scenarios are flexible in that they can also resupply propulsion systems operating in a blowdown node using a ullage recompression type process. However, unlike ullage recompression propellant transfers, the ullage exchange process must utilize only those propellant acquisition devices (PAD) capable of positioning the ullage bubble within the receiver propellant tank. If the PAD is unable to do so, resupply propellant could unknowingly be transferred (through the ullage exchange lines) back into the resupply module.

In addition to the transfer of propellant, a pressure regulated propulsion system requires the resupply of high-pressure helium gas. The development of a high-pressure gas compressor would be most beneficial in improving the volumetric efficiency of the resupply module's pressurant supply tanks. If a contamination-free gas compressor is not technically and/or economically feasible, the pressurant transfer procedure would be reduced to operating as high-pressure blow-down process.

Ullage Venting Resupply transfer can be used in resupply pressure regulated propulsion systems. With this method, the receiver tank is vented to near the propellant vapor pressure before transferring propellant. The required amount of propellant is then resupplied. The tank is then pressurized collapsing any propellant vapor bubbles trapped in the propellant acquisition device. Just as with the ullage exchange resupply process, ullage venting resupply also requires the transfer of high-pressure helium gas, which is used by the pressure regulated propulsion system.

It should be noted, that the overboard venting of propellant saturated ullage gas could cause the degradation of the local environment and/or damage the sensors and experiments of nearby spacecraft. It has been suggested that the implementation of a waste scavenging subsystem on the resupply module, could safely contain and store the vented ullage gas. However, an analysis on the feasibility of a wste scavenging subsystem reveals that the waste storage tank volume required for storage of the resupply receiver vehicle's propellant tank ullage gas could be substantial. Figure 63 presents the required waste storage tank volume as a function of the resupply receiver tank's ullage volume. For an empty shuttle RCS tank to be vented down to 8% of its operational pressure, slightly over two OMS tanks would be required to receive the vented ullage gas. Clearly the waste storage tank volume requirements make the implementation of a waste scavenging subsystem of this approach impracticable. Contamination sensitive spacecraft would have to make its own provisions in protecting itself during propellant resupply operations, or disallow ullage venting propellant resupplies.

An analysis of the various on-orbit propellant resupply scenarios suggest that the resupply module's propellant transfer subsystems be designed to carry out "ullage exchange" type of propellant transfers.

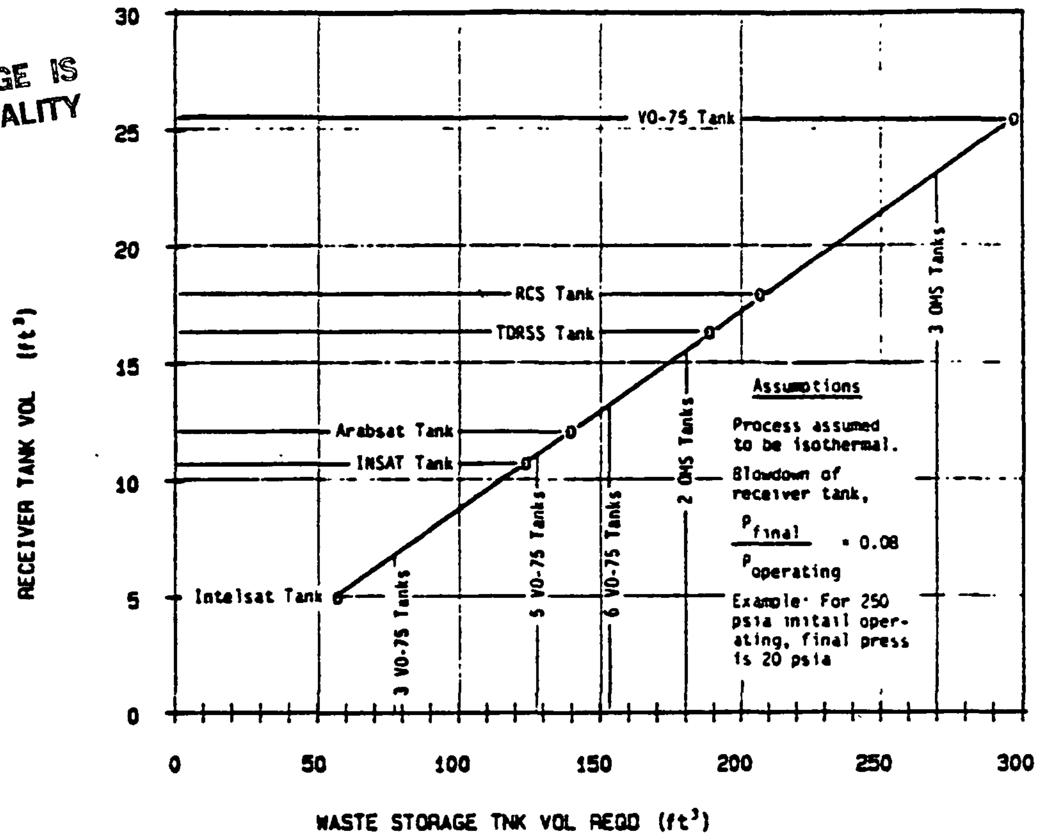


Figure 63. Waste Storage Subsystem Tank Volume Requirements

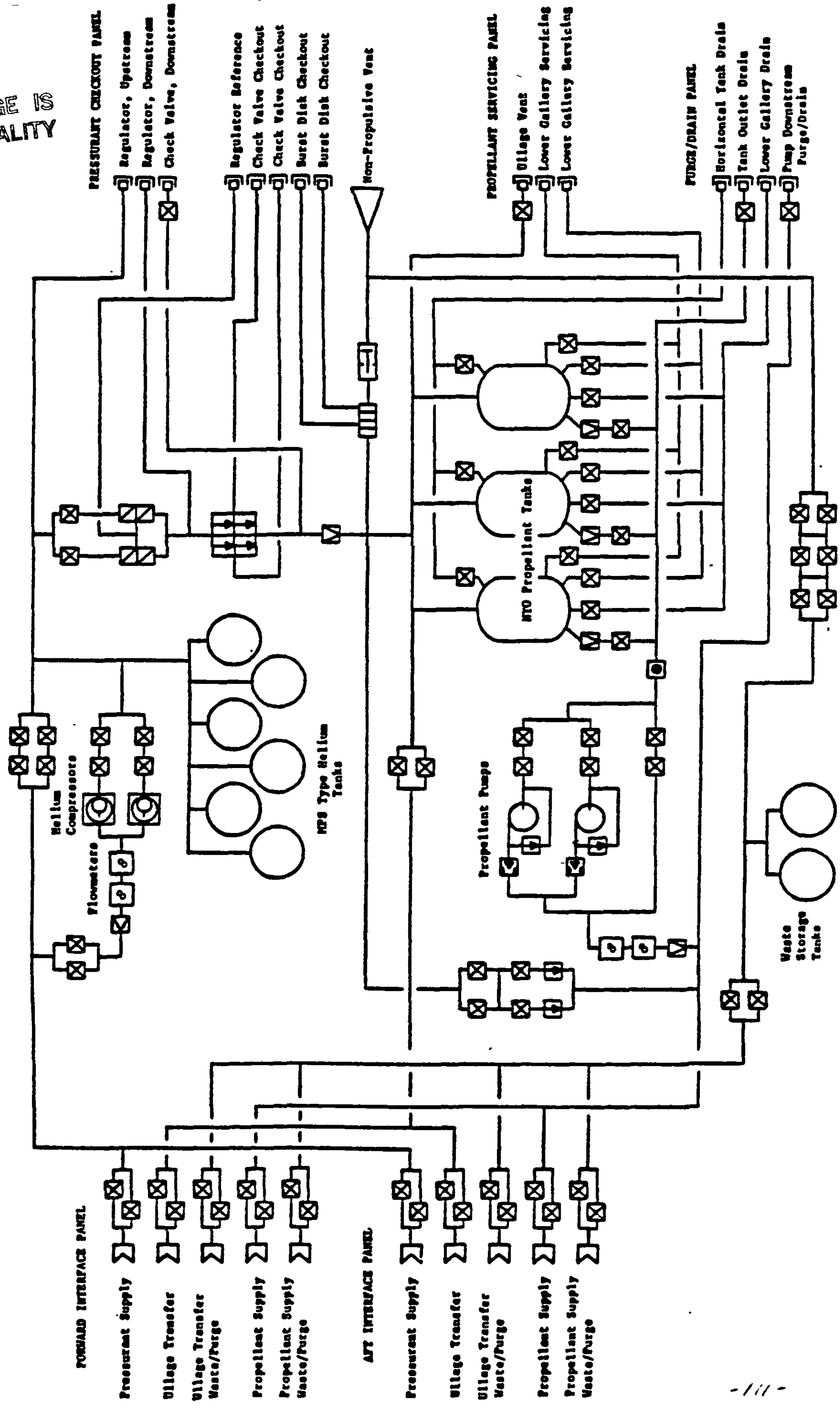
Ullage venting resupply scenarios were the least acceptable of the three transfer methods analyzed. The overboard venting of propellant saturated ullage gas could be a contamination concern. Implementation of a waste scavenging subsystem for storage of ullage gas was found to be impracticable. Since the mechanical and operational characteristics of ullage venting scenarios are similar to ullage exchange scenarios, it would be most beneficial for contamination sensitive spacecraft to convert from ullage venting to ullage exchange propellant resupplies.

Even though ullage exchange scenarios are mechanically and operationally more complex than ullage recompression scenarios, a resupply module designed for the ullage exchange process can also resupply vehicles' using the ullage recompression process. The versatility of the ullage exchange process is it biggest asset.

3.2.5 Conceptual Designs For Ullage Exchange Bi-Propellant and Pressurant Transfer Proces

Preliminary conceptual design system schematics for the on-orbit resupply of bi-propellants the using ullage exchange process and their accompanying pressurants were defined and are presented in Figures 64 and 65. The differences between the two conceptual resupply schematics, are in the design of the pressurant subsystems. The high-pressure pressurization/pressurant resupply subsystem defined as Option "A" (Figure 64.), stores and provides gaseous helium or nitrogen to the propellant transfer, waste storage/purge subsystem, and the pressurant transfer susystem. Pressurant for the propellant transfer and waste storage/purge subsystems is supplied via

Figure 64. Option "A: Expendable Resupply System Schematic, with the Helium Transfer



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parallel and series redundant regulators, while pressurant transfer is controlled by high-pressure gas compressors. The presurant subsystem defined as Option "B" (Figure 65), was not designed for on-orbit resupply of pressurant. The subsystem provides helium or nitrogen for propellant transfer and disconnect/line purging only.

Contingent on the type of propellant management device used in the resupply receiver vehicle's propellant tanks (capillary screen, surface tension vane, diaphragm), all three types of propellant resupply schemes: ullage transfer, ullage recompression, and ullage venting, could be resupplied by the Option "A" transfer systems. Without the addition of a pressurant resupply module, the Option "B" transfer system will be limited to ullage recompression resupplies, or to enhance an OMV's mission capabilities. The propellant transfer sysltem's propellant pumps, or a pressure regulated procedure could be used in the resupply of mono-propellant or space storable bi-propellant propulsion systems. Due to the hypergolic nature of these bi-propellants two independent propellant and waste storage subsystems (one for the transfer of fuel and the other for oxidizer) would be required. Two independent pressurant systems, one for the fuel system and the other for the oxidizer system, may be necessary since fuel and oxidizer vapors cannot be restrained from migrating into common lines.

The safety and redundancy requirements defined by "Safety Policy and Requirements for Payloads Using the Space Transportation System," NHB 1700.7A require two independent inhibits to control/restrain all presurant transfer paths leading to overboard outlets. In the case of propellant transfer and wase management functions, safety guidelines require three independent inhibis controlling propellant paths leading to overboard outlets.

The majority of the components in the preliminary resupply design schematic are existing qualified components or utilize existing component technology to manufacture new hardware. Table 21 of the enclosure presents the component legend for the resupply system schematics. However, there are three major components in the conceptual resupply system design that require advanced development. One of these new components is a high-pressure gas compressor used in pressurant transfer missions (e.g. ullage transfer, ullage venting). The main obstacles in the development of such a component are: (1) gas contamination by lubricants during the compression process, and (2) reducing the compressor weight.

A propellant transfer pump is another component requiring advanced development. With current propellant seal development it may be difficult to produce a seal which can endure the cycle life and endurance specifications required by the propellant transfer pumps.

If the development of a contamination-free gas compressor or propellant transfer pump are not technically and/or economically feasible, the resupply system design would be limited in its transfer capabilities; the pressurant system would be reduced to operating as high-pressure blow-down system, and the propellant transfer system would not be able to accommodate the ullage transfer resupply scenario.

Figure 65. Option "B" Expendable Resupply System Schematic, Without Helium Transfer

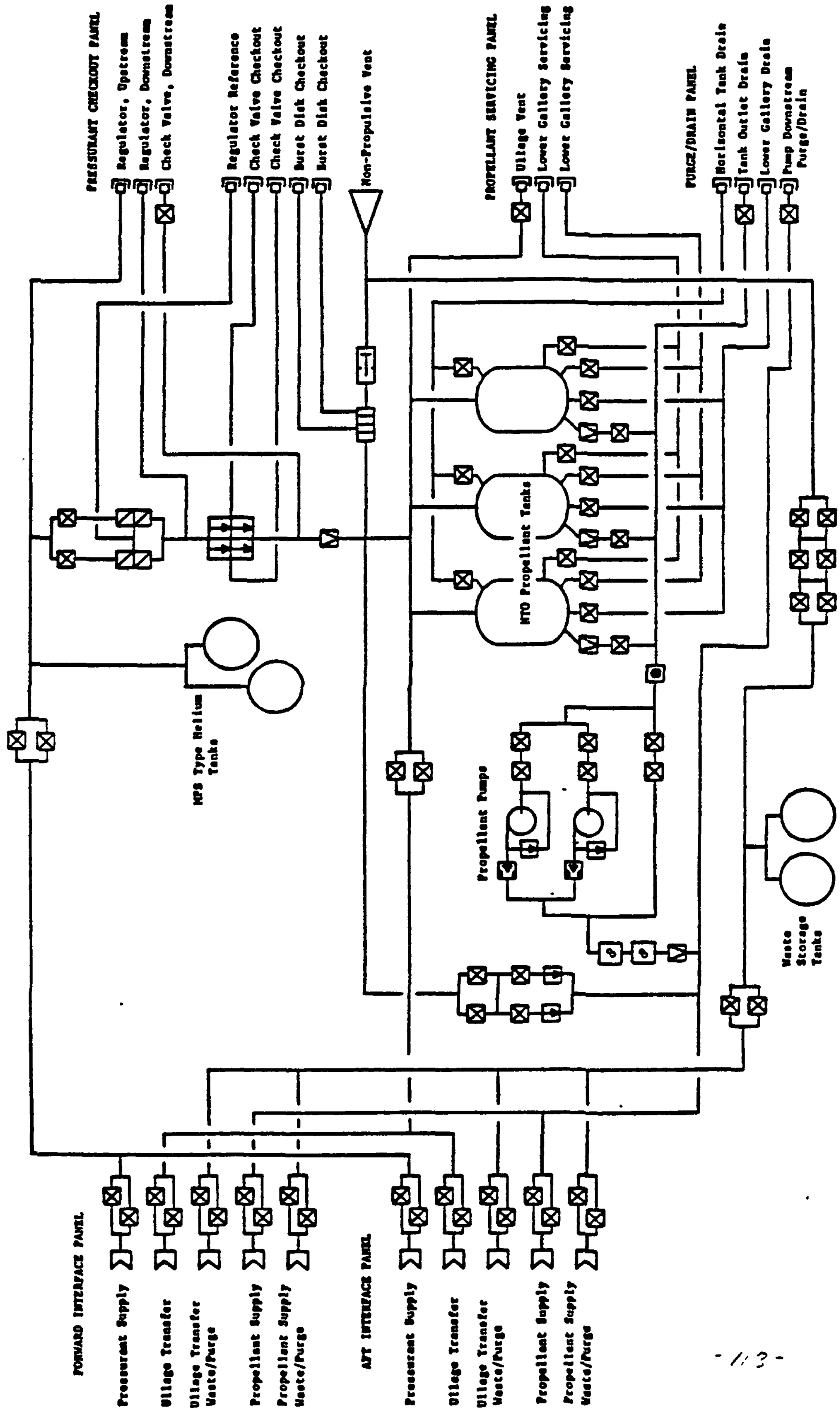

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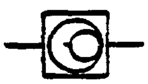
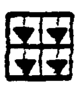
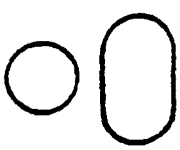
Table 21. Conceptual Resupply System Design, Component Legend

SYMBOL	COMPONET
	Check Valve
	Disconnect, Fluid Transfer
	Disconnect/Cap, Fluid Servicing & Check-Out
	Dual Burst Disk
	Dual Pressure Regulator
	Filter
	Flowmeter
	Gas Compressor
	Non-Propulsive Vent
	Propellant Pump
	Quad-Check Valve
	Relief Valve
	Tanks: i. Propellant, Stretched OMS ii. Pressurant, Shuttle MPS iii. Waste Storage
	Vapor Bubble Detector
	Valve

The third new component, fluid transfer disconnect. This disconnect is expected to be a remotely operated spin-off of the manual valve presently being developed by NASA JSC.

3.2.6 Propellant Transfer Pumps

Pump propellant trasnsfer may be utilized in any of the three general category scenarios of propellant resupply (ullage recompression, vent, and exchange). The general requirements of the pump are determined by the type of receiver subsystem (vane, screen, and diaphragm). Threfore, the power requirements in this report were determined for several head values at flow rates up to 10 GPM.

Figure 66, 67, and 68 show the pump power requirements versus the flow rate for MMH, NTO and the MMH-NTO propellant transfer with a head rise of 250 PSI. Two commercial pumps (Garrett and Marquardt) are compared. The Marquardt pump can be used up to a flow rate of 21 GPM. Marquardt is the preferred pump for flow rates greater than 10 GPM. The Garrett pump has a lower power requirement at flow rates between 1 and 10 GPM, therefore, it is the preferred

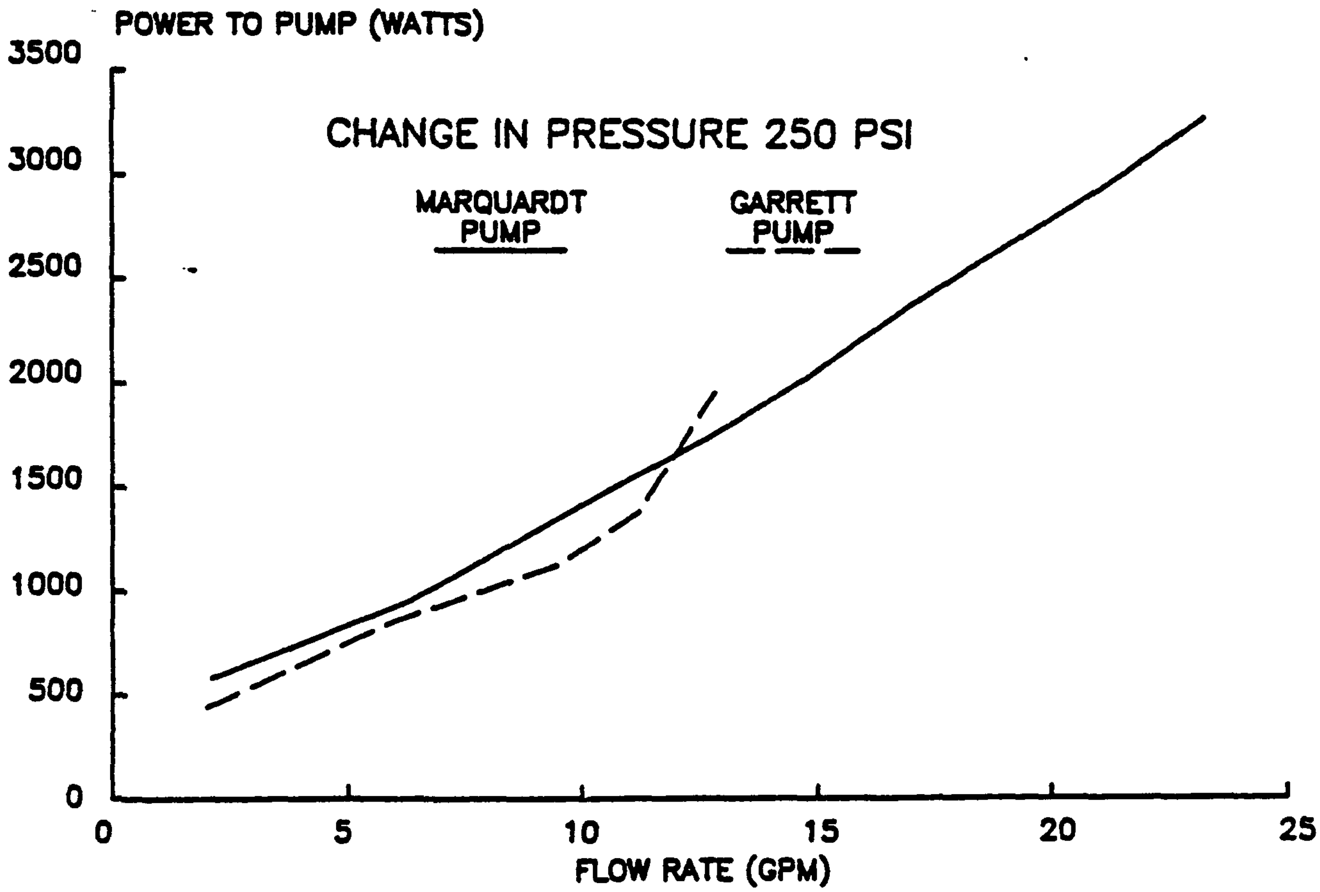


Figure 66. Pump Power For MMH Transfer

pump in the lower flow rate region. Similarity relations were used on the Garrett pump to redesign a pump with optimum flow rate at 1 GPM. The result is shown in Figure 67 (labeled similarity pump). The results of the similarity calculations indicate that the design of a 1 GPM optimum pump would not result in sufficient energy reductions compared to the Garrett pump.

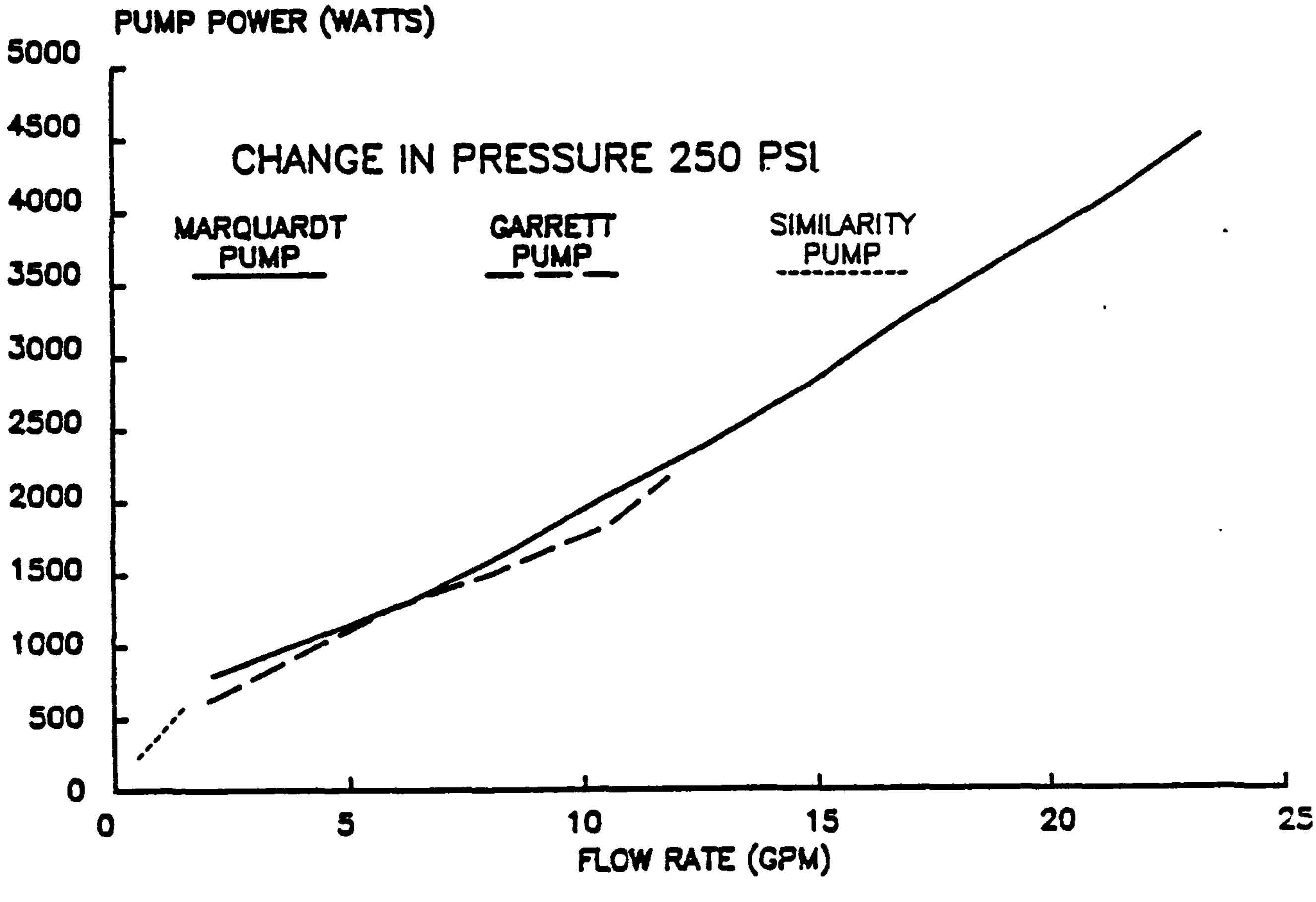


Figure 67. Pump Power For NTO Transfer

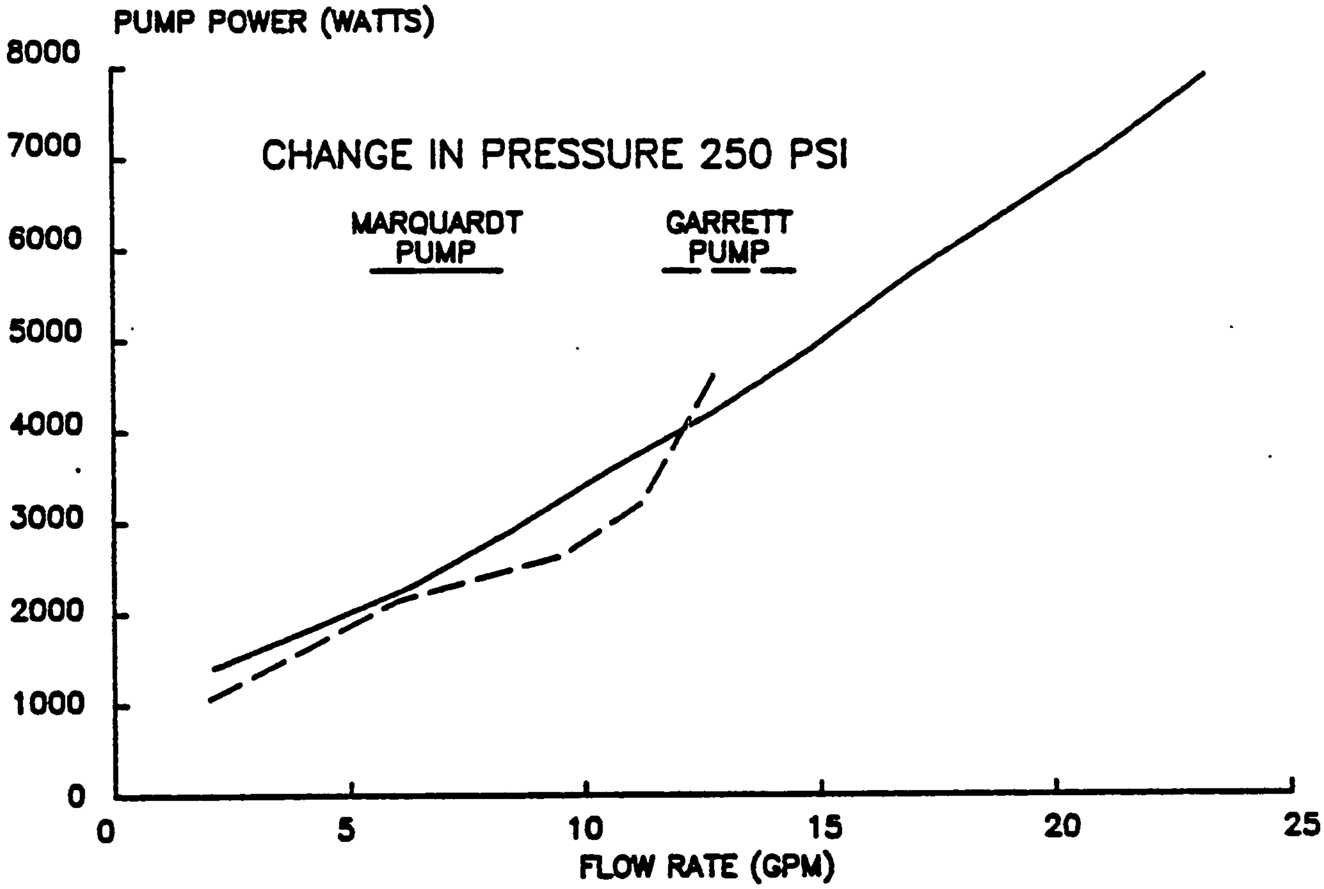


Figure 68. Pump Power For MMH and NTO Transfer

Figures 69, 70, 71, and 72 present the pump power requirements versus the flow rate at various head rise values for the propellants MMH, NTO, a system of MMH & NTO, and hydrazine. The Garrett pump data was extapolated to complete the family of head curves seen in Figures 69, 70, and 72. The data has an increasing variance with decreasing head and flowrates. The method of data extrapolation incorporated the available performance pump curves with similarity relations at lower pump speeds. The power requirements to pump hydrazine at different heads was determined by using the Garrett pump data and correcting for the higher density and viscosity of the hydrazine as compared to MMH.

Table 22 lists the power required by the pump system (including motor, efficiency = 0.90) for MMH, NTO, MMH & NTO, and hydrazine. The values in Table 22 differ from Figures 69, 70, 71, and 72 as a result of the inclusion of the motor efficiency. The parameters of the pump's motor are listed in Table 23. In particular it lists the input requirements of the motor, and total system weight of the pump and motor system.

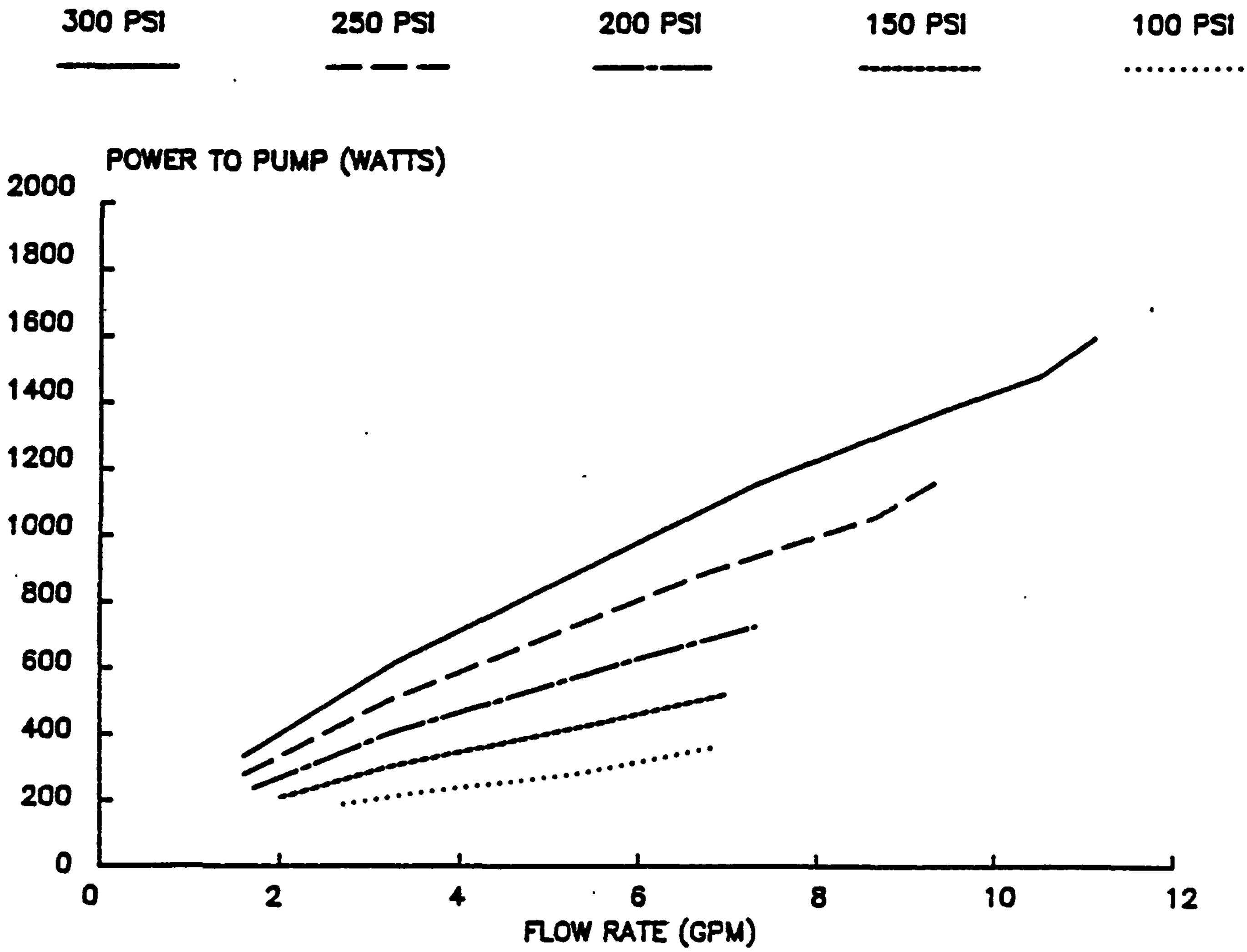


Figure 69. Power to Pump For MMH

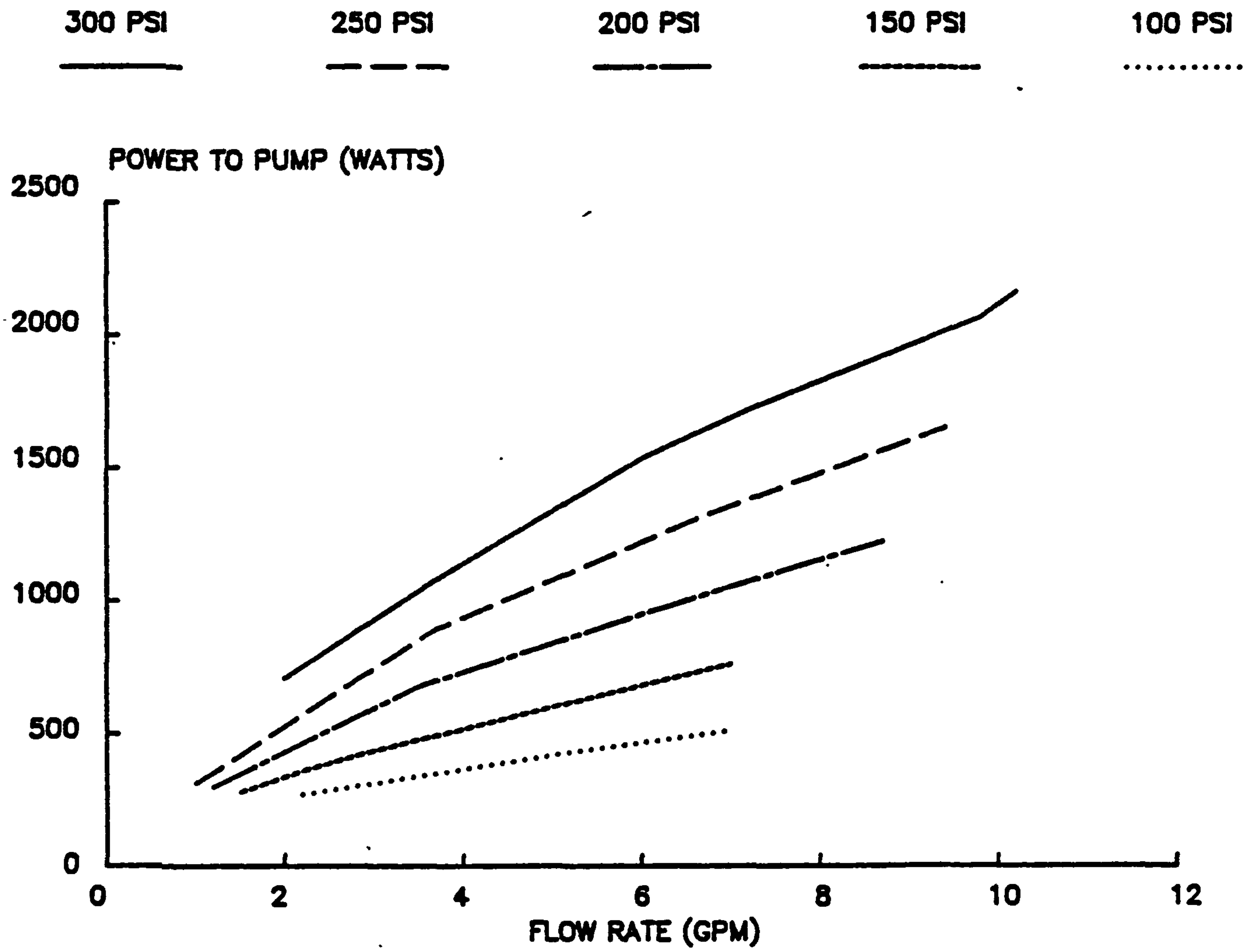


Figure 70. Power to Pump For NTO

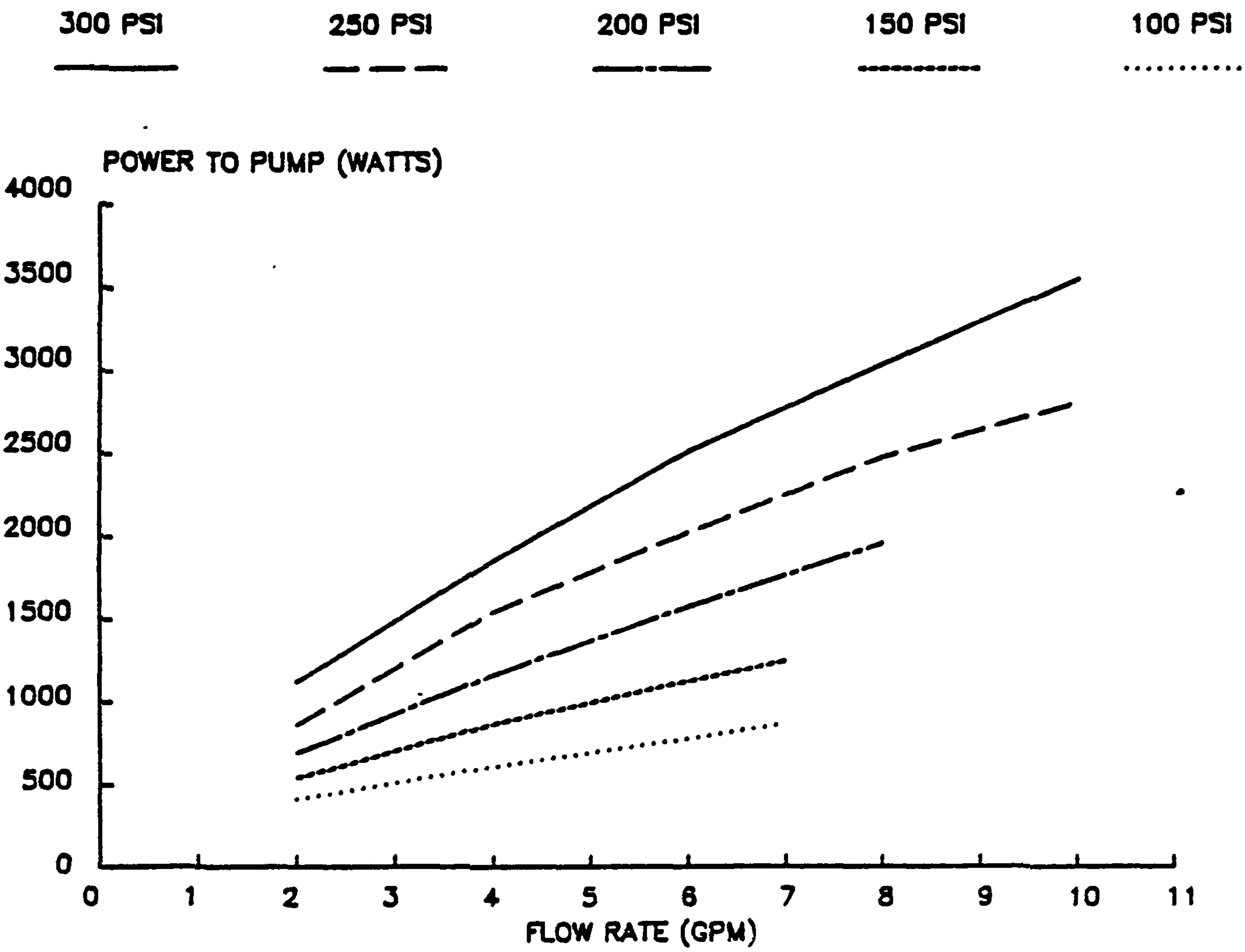


Figure 71. Power to Pump For NTO & MMH

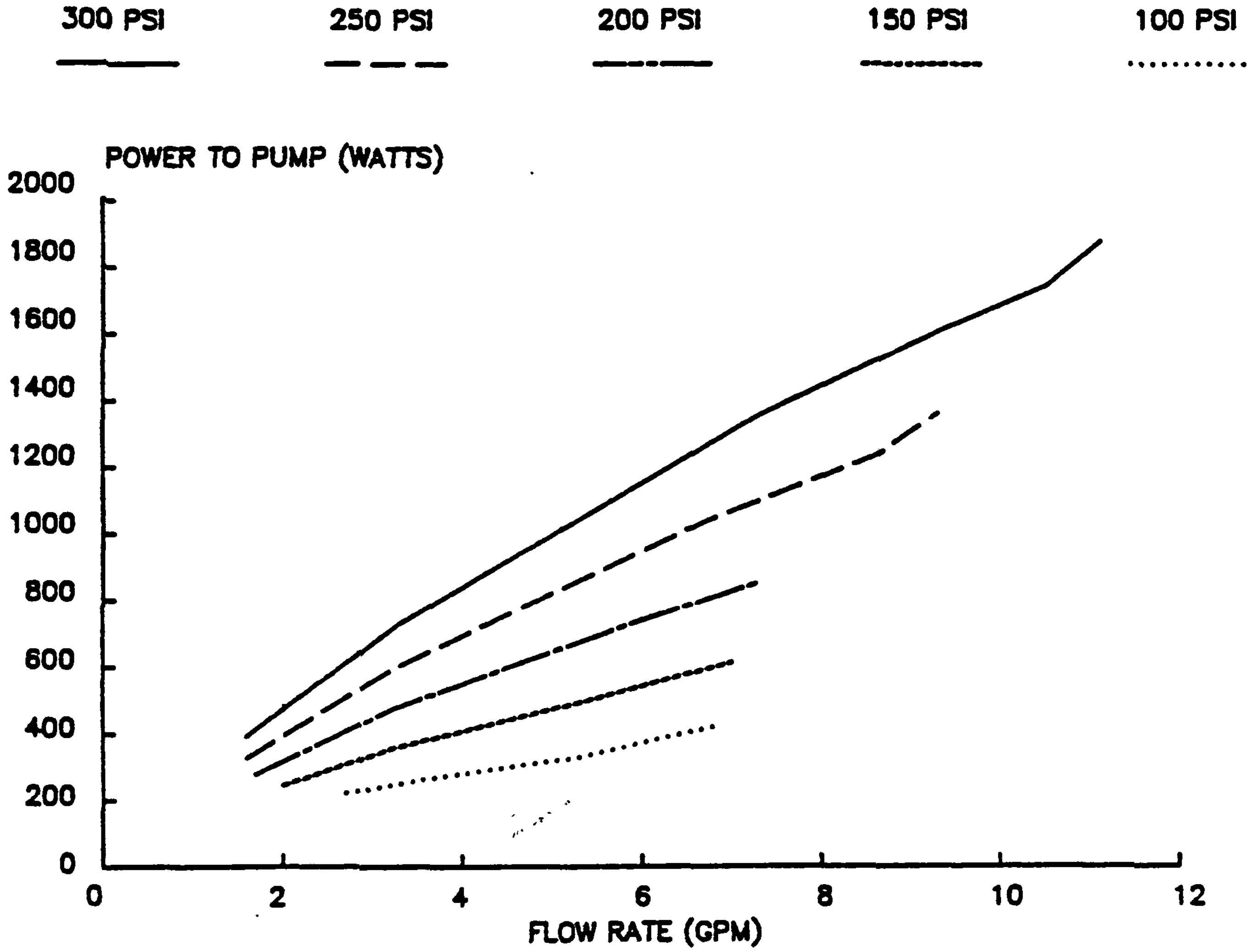


Figure 72. Power to Pump For Hydrazine

Head Rise (PSI)	Flow Rate (GPM)	Power (watts) MMH	NTO	MMH & NTO	HYDRAZINE
300	10	1620	2330	3950	1904
	8	1360	2030	3390	1590
	6	1090	1710	2800	1280
	4	800	1270	2070	940
	2	445	800	1245	522
250	10	1340	1780	3120	1580
	8	1110	1640	2750	1300
	6	940	1370	2310	1100
	4	655	1070	1725	770
	2	367	589	956	430
200	8	890	1290	2180	1040
	6	690	1070	1760	809
	4	528	800	1330	620
	2	289	480	769	339
150	7	556	833	1390	652
	6	500	756	1260	587
	4	378	589	967	444
	2	233	367	600	274
100	7	411	556	967	483
	6	352	533	885	414
	4	267	413	680	313
	2	178	278	456	209

* Pump system consists of one motor for both pumps.

Table 22. Power Required by Pump System for Various Propellants

Table 23. Parameters of Motors and Pumps

Parameters	Garrett	Marquardt
pump type	Centrifugal	Centrifugal
operating speed (RPM)	23,664	80,000
optimum flowrate (GPM)	10	21
suction specific speed (RPM)	MMH (11,089) NTO (16,013)	MMH (12,500) NTO (8,500)
motor type	AC Induction	AC Induction
Input Voltage	212 VAC	225 AC
frequency	420 Hz	1400 Hz
current	25 amp	35 amp
operating time	>15 hr	1 hr (Grease Pack) 20,000 hr (mist)
efficiency of motor	0.90	0.90
weight of pump and motor (lbs)	30	30

3.2.7 Pressurant Transfer

Two helium receiver volumes were investigated. The first receiver volume to be investigated was an OMS helium tank design case. The second receiver volume investigated was a volume equivalent to two RCS helium tanks. The volume of two RCS tanks is in the size range of required helium to expel 2,000 pounds of storable bipropellants.

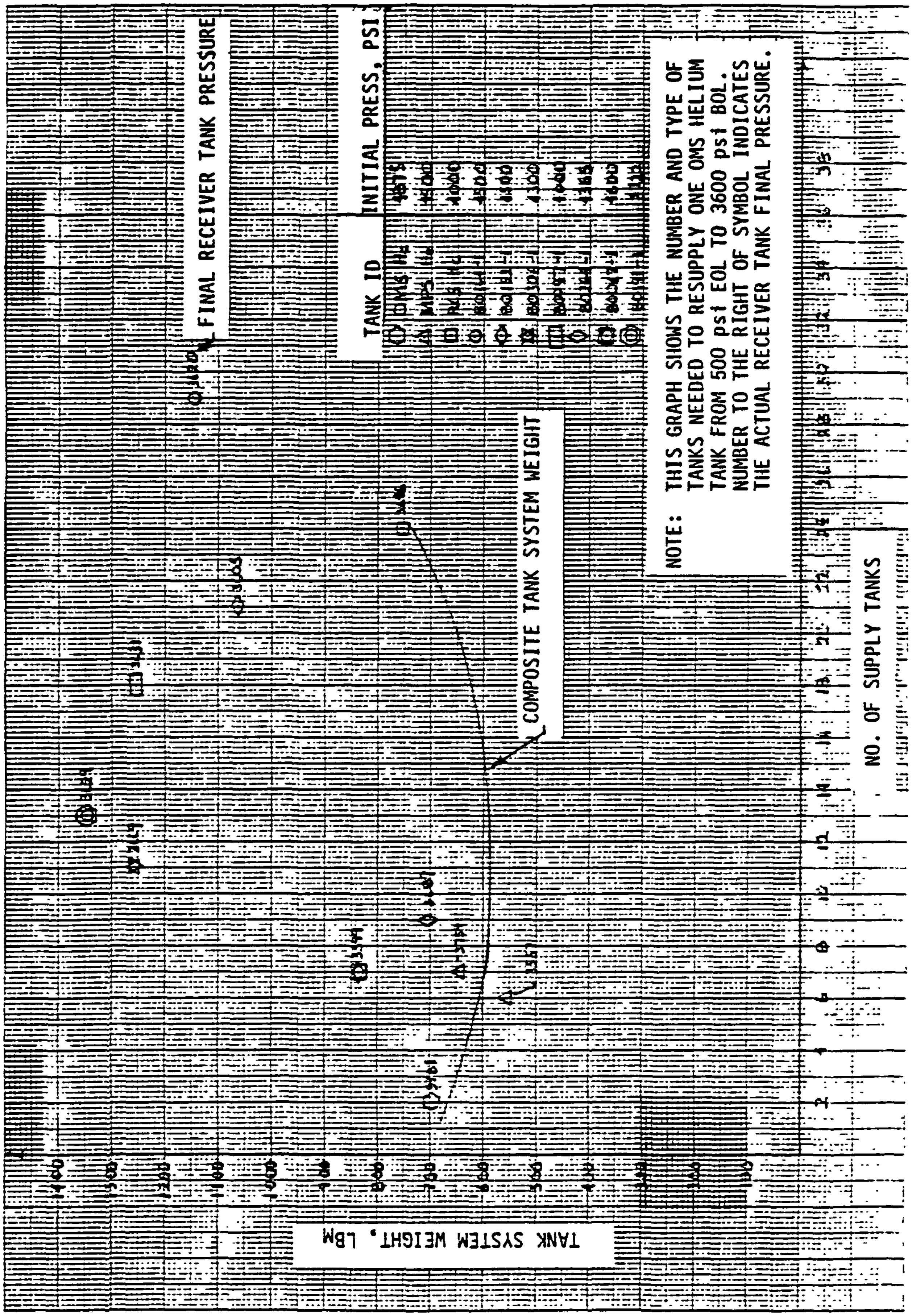
Different existing helium tanks were studied for resupplying one OMS helium tank from 500 psi and 70°F. end of life (EOL) to approximately 3600 psi and 70°F. beginning of life (BOL). Figure 73 shows the number and the type of tanks to resupply one OMS helium tank. When multiple supply tanks were used, one supply tank at a time was opened to the receiver tank. This process was found to increase the efficiency. In Figure 73 the number to the right of the ID symbol represents the final pressure in the receiver and in the last supply tank. The supply tanks' initial pressure was taken as the maximum operating pressure of the particular tank being used. Figure 73 also shows the wide variation in number and weight of the resupply systems using existing tanks. For this particular resupply requirement of resupplying one OMS He tank to 3600 psi and 70°F., a system of 6 MPS helium bottles proved to be optimum.

The space transfer of pressurants can be accomplished by two basic approaches when using a compressor:

1) Using a variable flow rate/inlet pressure, or
2) Using a set flow rate/inlet pressure.

A metal diaphragm compressor will accept a wide range of inlet pressures where as a 2-stage piston compressor will accept a specific inlet pressure or a specific flow rate. Flow rates, energy requrements and fill times for these compressors are presented in Figures 74, 75, and 76. Specific cases of gas transfer are presented in Tables 24 and 25. A summary of their major advantages and disadvantages are tabulated in Table 27. The rationale for the tentative selection of the metal diaphragm compressor in general pressurant

Figure 73. Number of Supply Tanks vs. Tank System Weight



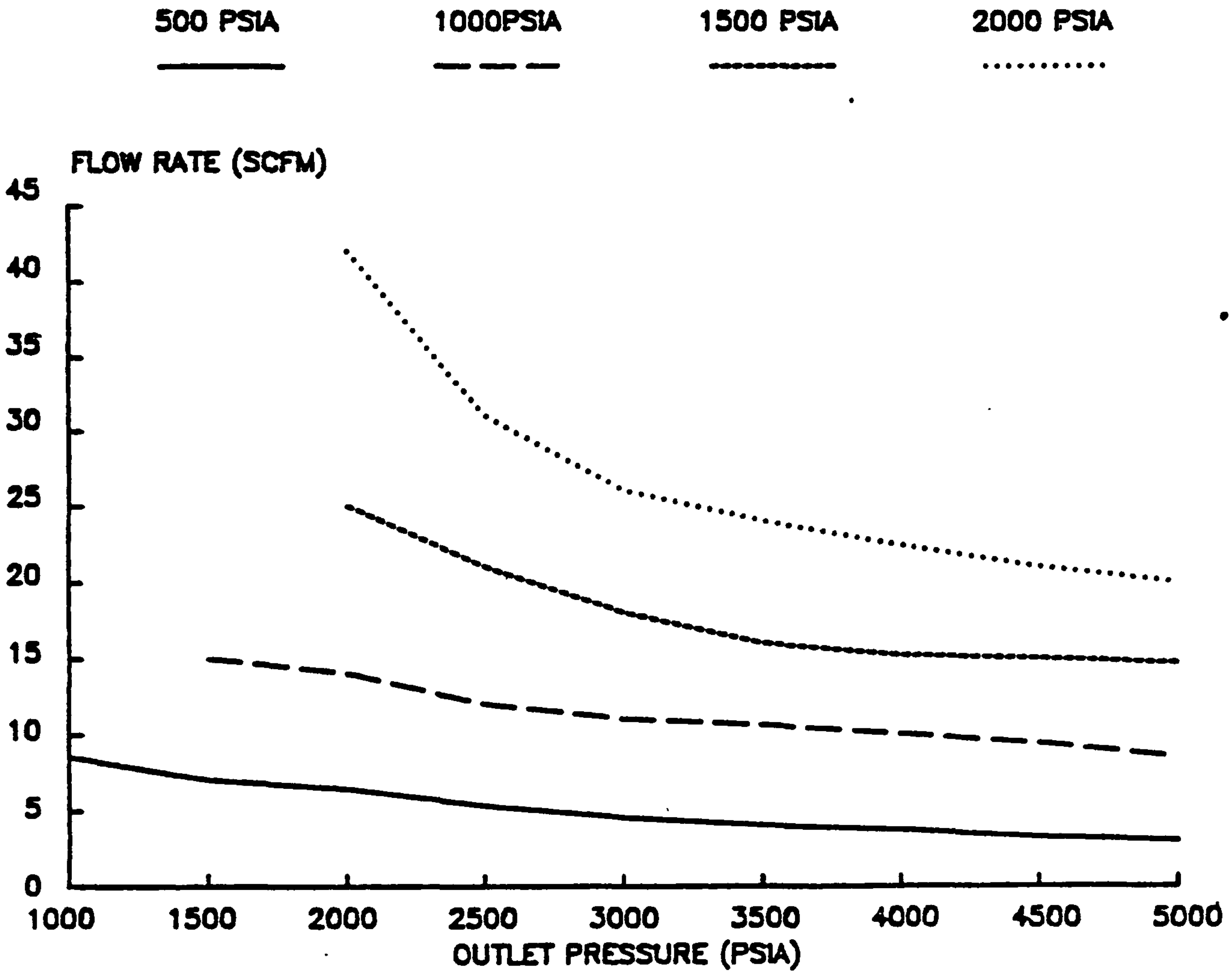


Figure 74. Flow Rates of a 1.5 HP Diaphragm Compressor at Various Inlet Pressures

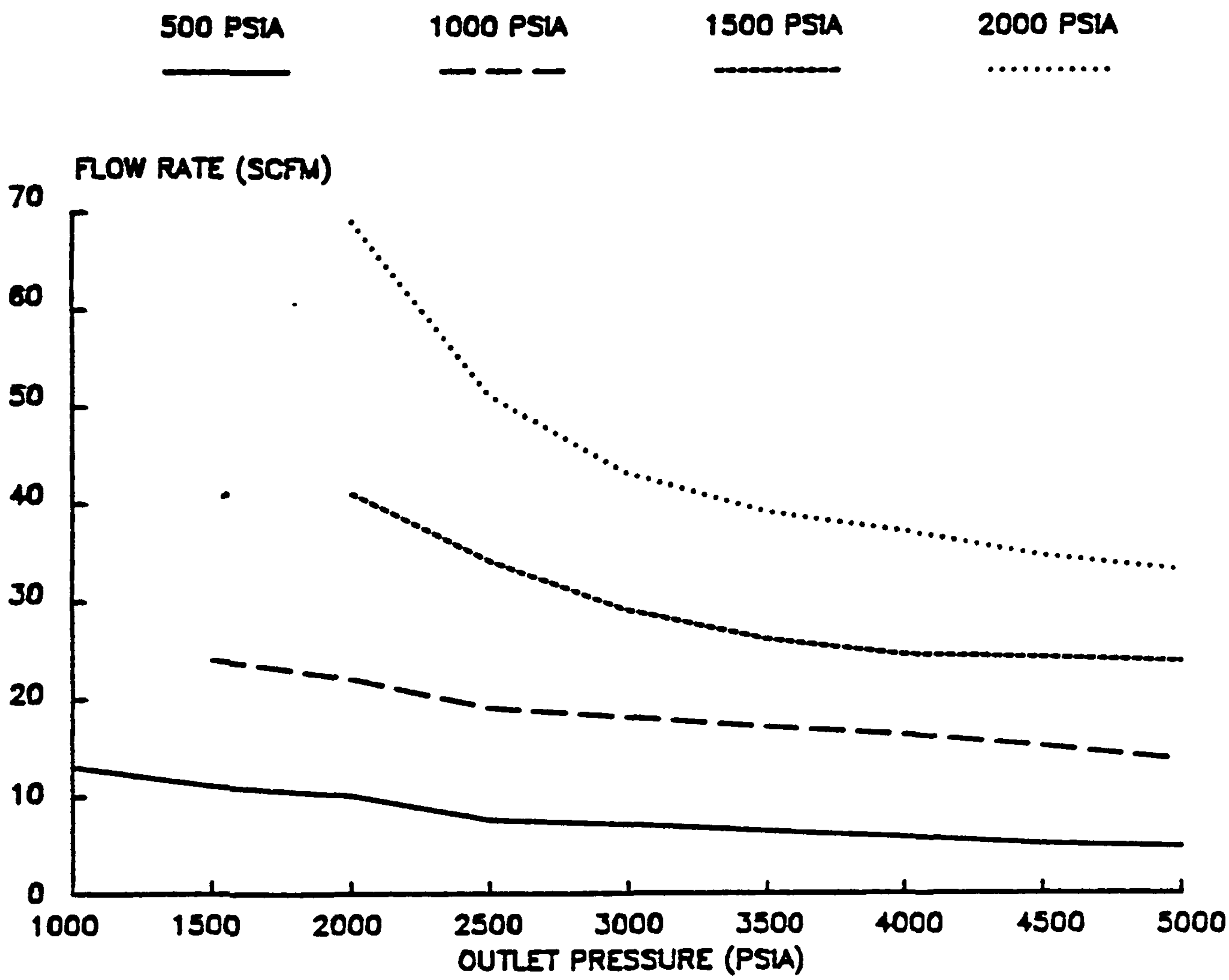


Figure 75. Flow Rates of a 3.0 HP Diaphragm Compressor at Various Inlet Pressures

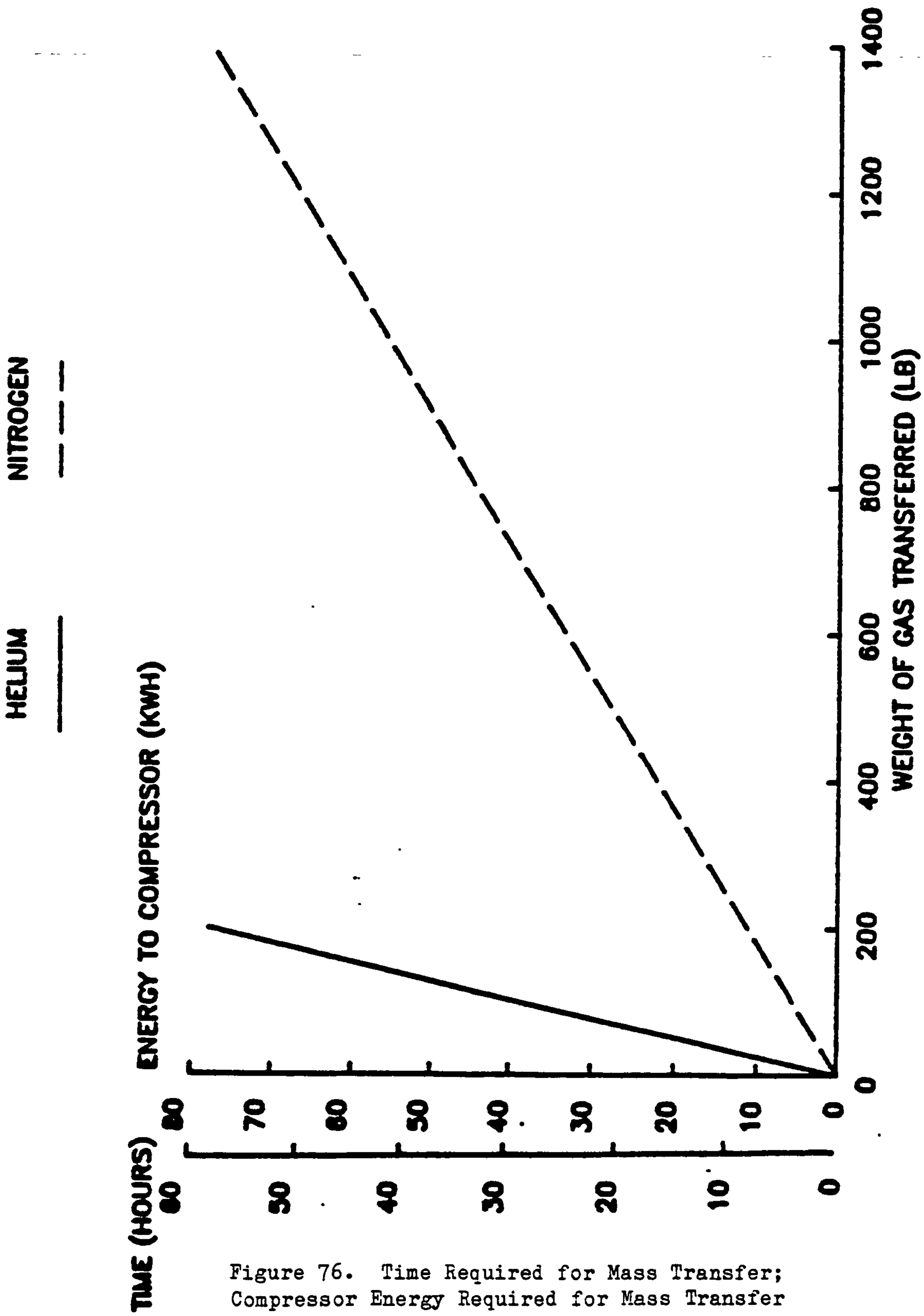


Figure 76. Time Required for Mass Transfer;
Compressor Energy Required for Mass Transfer
Using A 2-Stage Piston Compressor

Table 24. Metal Diaphragm Compressor Requirements to Fill Various Pressure Bottles

Type of Pressure Bottle	Power of Compressor	Fill Time	Total Energy to Compressor	Required Heat Rejection Rate
RCS	1.5 HP	0.50 hr	0.56 kwh	3300 BTU/hr
	3.0 HP	0.26 hr	0.58 kwh	6320 BTU/hr
MPS	1.5 HP	1.4 hr	1.55 kwh	3000 BTU/hr
	3.0 HP	0.92 hr	2.06 kwh	6410 BTU/hr
OMS	1.5 HP	5.5 hr	6.09 kwh	3000 BTU/hr
	3.0 HP	3.6 hr	8.05 kwh	6400 BTU/hr

Table 25. Two-Stage Piston Compressor Requirements to Fill Various Pressure Bottles from a 500 Psia Source

Type of Pressure Bottle	Power of Compressor	Fill Time	Total Energy to Compressor	Required Heat Rejection Rate
RCS	2.0 HP	0.70 hr	1.04 kwh	4551 BTU/hr
MPS	2.0 HP	2.52 hr	3.76 kwh	4552 BTU/hr
OMS	2.0 HP	8.4 hr	12.5 kwh	4557 BTU/hr

Table 26. Parameters of the Compressor/Motor

Parameter	2-Stage Piston	Metal Diaphragm Compressor	
Motor	2.0 HP	1.5 HP	3.0 HP
Flow Rate	6 SCFM	20→3.0 SCFM	33→4.7 SCFM
System Weight	175 lbs	150 lbs	200 lbs
System Volume	1.1 ft^3	3.75 ft^3	5.6 ft^3

Table 27. Compressor Options

Metal Diaphragm Compressor

Advantages	Disadvantages
- very low gas contamination	- higher compressor/motor weight
- variable inlet pressure/flow rates	- greater compressor/motor volume - lower operating period
- less energy storage required per lb. of pressurant transferred	- possible diaphragm failure
- shorter transfer times	

2-Stage Piston Compresor

Advantages	Disadvantages
- smaller system volume	- fixed flow rate/inlet pressure
- lower compressor/motor weight	- higher energy storage required per lbm of pressurant
- longer operating time	- longer fill times required - possible gas contamination

transfer is also presented. It is further recommended that prototypes of both compressors be built and tested. In particular note Table 26 which presents three compressors with 1.5, 2.0, and 3.0 HP motors and their respective weights of 150, 175, and 200 lbs.

A metal diaphragm compressor allows a variable inlet pressure and thus a variable flow rate. Figure 77 shows a typical metal diaphragm compressor. Operation starts with the diaphragm deflectd to the bottom of the cavity by the gas inlet pressure with the hydraulic piston at bottom dead-center position. As the crankshaft rotates and the hydraulic piston moves toward top dead-center, the hydraulic pressure increases. When the hydraulic pressure reaches the pressure level of the inlet gas in the cavity, the diaphragm moves up compressing the inlet gas.

When the pressure of the inlet gas in the cavity reaches the pressure level downstream of the discharge check valve, the valve opens and the inlet gas is discharged into the tank. Since the hydraulic system has slightly more displacement capacity than the gas system, all gas is displaced. Excess hydraulic fluid is then released through the hydraulic relief valve (pressure level is set at just above outlet pressure).

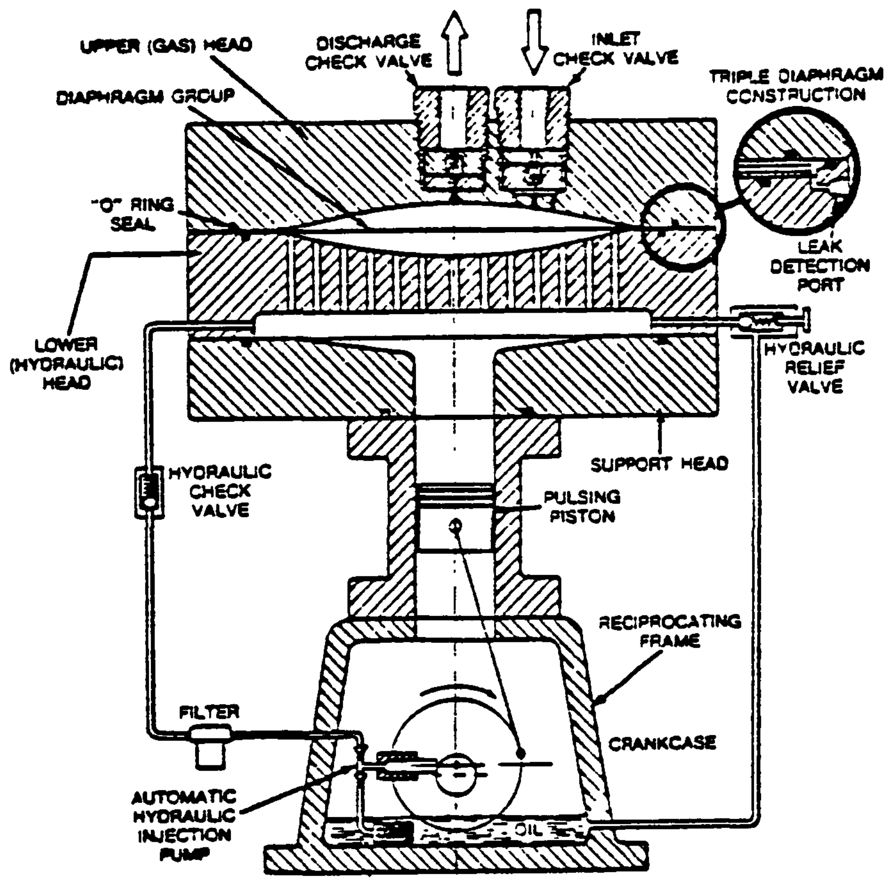


Figure 77. Typical Assembly of a Diaphragm Compressor

Figure 74, Figure 75 and Table 28 present expected flow rates at varying inlet and outlet pressures for a 1.5 HP and a 3.0 HP motor operating at 800 RPM. In the process of transferring pressurant from the transfer tank to the receiver tank the first step will be free transfer, until the two tanks reach a equilibrium pressure. When the equilibrium pressure is obtained the compressor is activated and pressurant is transferred until the required receiver pressure is obtained. Using the variable flow rates (at a variable inlet pressure) one can determine the time and electrical energy requirements to fill any tank(s).

Table 24 presents results based on the assumption that the transfer and receiver tanks are the same type tank. For example, one OMPS transfer tank transfers pressurant to one OMPS receiver tank. The energy required to run a 3.0 HP motor for 0.92 hr is 2.1 kwh with a required heat rejection of 6410 BTU/hr.

One of the disadvantages of a diaphragm compressor is its higher mass and volume required for respectable flow rates. Fluitron (a metal diaphragm compressor company) is testing compressors at higher operating speeds. This will result in higher flow rates with a small increase in weight. Another way to double capacity but only increasse the weight by about 22% is to use a duplex compressor. A duplex consists of two compressor heads driven by a single motor. the two compressor heads operate in parallel. Expected weights

Table 28. Flow Rates of a Diaphragm Compressor at Various Inlet and Outlet Pressures

Inlet (PSIA)	Outlet Pressures	Flow Rate (SCFM) 1.5 HP	3.0 HP Motor
500 →	1000	8.5	13
	1500	7.0	11
	2000	6.4	10
	2500	5.3	7.5
	3000	4.5	7.0
	3500	4.0	6.3
	4000	3.7	5.7
	4500	3.2	5.0
	5000	3.0	4.7
1000 →	1500	15	24
	2000	14	22
	2500	12	19
	3000	11	18
	3500	10.6	17
	4000	10.0	16.2
	4500	9.4	15.0
	5000	8.5	13.6
1500 →	2000	25	41
	2500	21	34
	3000	18	29
	3500	16	26
	4000	15.2	24.4
	4500	15.0	24.1
	5000	14.7	23.7
2000 →	2000	42	69
	2500	31	51
	3000	26	43
	3500	24	39
	4000	22.4	37
	4500	21	34.5
	5000	20	33

and volumes are presentd in Table 26 based on present technology and assuming a one-third earth usage weight using light weight.high strength materials and the removal of structually excess material.

A 2-stage piston compressor is confined to a narrow inlet pressure range and thus a fixed flow rate. Normally the inlet pressure varies during pressurant transfer, thus a pressure regulator will be required at the inlet of the piston compressor. The only advantage of a fixed flow rate is that the system is quickly in steady state so that energy, time, and heat transfer values are linear with respect to the weight of gas transfered (Figure 76). Figure 76 was created on the assumed values of a 2.0 HP motor with a flow rate of 6 SCFM. Note in Figure 76 that the mass of nitrogen transferred is greater than the mass of helium tranferred for the same stored energy. This can be understood if one realizes that the work done is pressure-volume work and not mass transfer at high pressures. Table 25 presents estimated time, enegy, heat rejection rates to fill one ORCS, OMPS, or OMS pressure bottles from a 500 psia source using a 2.0 HP motor at a 6 SCFM flow rate.

The major advantages of the 2-stage compressor are: 1) a smaller system volume, 2) a slightly lower weight, and 3) a longer operating time between servicing. The smaller volume exists due to the ability to build the compressor into a cylindrical package 10 inches in diameter and 2 feet long.

The diaphragm compressor is large at the diaphragm (1.2 feet in diameter) plus it requires a larger gear drive. The mtal diaphragm compressor has a lower expected operating life due to the metal diaphragm (1000 hours as compared to 2000 hours for the 2-stage piston). The diaphragm life can be increased by using a layered diaphragm of metal sheets or combind metal and elastomer layers (to 2000 hours).

The major diadvantages of the 2-stage piston compressor are: 1) longer fill times, 2) a higher energy requirement, 3) possible gas contamination, and 4) a fixed inlet pressure. The fixed inlet pressure will require a pressure regulator (adds about 5 pounds) and also leads to longer fill times and higher energy requirements. The longer fill times exist due to the fixed scfm that enters the piston chamber at a given pressure, where the diaphragm compressor accepts higher scfm at higher pressures. The 2-piston compressor also loses the higher pressure gas (a loss of stored energy) due to the fixed inlet pressure. Gas contamination will be greater problems with the 2-stage compressor because of the direct contact of the piston with the gas. The least case contamination will be teflon dust and a worst case will be hydrocarbon contamination from the grease packs.

Table 27 lists the advantages and disadvantages of the two candidate compressors. The advantages of the diaphragm compressor include; very low gas contamination, less energy required to transfer a given mass of gas, shorter tank transfer times, and a variable inlet pressure. the disadvantages of the diaphragm compressor include; a greater weight, a larger volume, and a lower operating life. The metal diaphragm compressor is recommended for use in cases of transfer between tanks of similar size. The advantages of the 2-stage piston compressor include; a small volume, a lesser weight and a longer operating life. The disdvantages of the 2-stage piston compressor include; possible gas contamination, longer fill times and greater energy requirements because of the fixed inlet pressure. The 2-stage piston compressor is recommended for usage when the transfer tank is large in comparison to the receiver tank.

3.2.8 Battery Power Sources for Pumps and Compressors

There exist two types of batteries, the primary battery and the secondary battery, that can be used as energy storage devices. The primary battery is to be used in cases where the system is used only once such as an expendable stage. A primary battery cannot be recharged. the secondary battery can be recharged and should be applied in cases of multiple use such as LEO where the resupply module can be recovered.

Table 29 Battery Parameters

Type	Name	Capacity (Ah)	Max Cont. Current (A)	Storage Wh/lb	Storage Wh/in^3	Shelf Life 70°F	Cycle Life
Primary	Lithium/.Thionyl Chloride $Li/SOCl_2$						
	Disk Cells (3.1 Volt)	500	7	109	15.0	1.5% Loss / Year	1
		1400	16	159	15.2		
		2000	25	175	14.9		
		8000	40	216	16.2		
	Prismatic Cells	2000	25	209	14.9		
		10000	45	218	15.6		
Primary	Zinc/Silver Oxide Zn/AgO Manually Activated Cell (1.55 Volt)	410	20	95	9.2	Dry 3% Loss / Year; Wet 16% Loss / Year	1
Secondary	Lithium/Titanium Disulfide+ Li/TiS_2 (1.5 Volt)	400	*	100	7.2	2% Loss / Year	200
Secondary	Nickel/Hydrogen Ni/H_2	100	*	20	1.0	2% Loss / 10 Hours	6000
	(1.2 Volt)	50	*	14	0.9	2% Loss / 10 Hours	6000

* Function of battery design but at least 20 amps

\+ Is still in development, simulated flight tests are in progress

Lithium/Thionyl Chloride has the highest energy density of the developed primary batteries, as can be seen in Table 29 The battery has a 90% depth of discharge and a 90% efficiency in chemical to electrical conversion. The battery has one disadvantage; it tends to overheat when physically abused. Physical abuse is clasifed as puncturing, opening, and mutilating. Battery overheating can also occur by exceeding current drain capabilities or fusing. However, this disadvantage is of concern only when batteries energy density exceeds a 5000 Ah capacity.

Zinc/Silver Oxide is another primary battery but at lower energy density than the Lithium/Thionyl Chloride battery. An advantage Zn/AgO has over $Li/SOCl_2$ is that it can dischage a massive amount of energy without the concern of an exothermic dischage. Shelf life is a major disadvantage of the battery where a 16% energy loss per year occurs as a wet cell. A solution to this problem is to introduce the electrolyte just before application, either manually or remotely.

Lithium/Titanium Disulfide has the highest energy density of the near-developed secondary batteries. Secondary batteries have a depth of discharge of 80% and a chemical to electrical conversion efficiency of about 85%. The battery can be recharged about 200 times before replacement. The battery is recommended for use when the resupply module can be recovered and then reused or in conjunction with a photovoltaic system. One disadvantage is that the battery has not been flight tested since it is still in the development stage. Simulated flight tests are in progress.

The Nickel-Hydrogen battery has an outstanding cycle life (about 6000 charges/discharges before replacement) for a secondary battery. This allows the battery to mate well with photovoltaic panels over long time periods. One drawback of the Ni/H_2 system is low energy density (20Wh/lb) as compared to the Li-Ti S_2 (100 Wh/lb). The Ni/H_2 system can last 20 years as long as is cylced less than 6000 times. The Ni/H_2 battery plus photovoltaic panels is best used for a continuous power source such as a tank heating system.

Table 30 presents energy weight requirements for the resupply module to transfer 45,500 lbs of fuel, fill 12 MPS bottles, run the heaters, etc. The deployable photovoltaic panel (DPP), used in all scenarios, is the same as presented in Table 31 using the Si 100 mm cell and a 100 mm cover. The batteries come from Table 29 and specific amp-hour batteries are used depending on the application. The system weight includes batteries, DPP, voltage regulators, charge/discharge unit, and power hardware.

In LEO (scenario 1 Table 30), it was assumed that the battery would be discharged 60 minutes and recharged 30 minutes and function for one year. The $Ni-H_2$ battery is preferred in this case over the Li/Ti S_2 because of the number of recharging cycles required in LEO (one every 90 minutes). One 50 Ah $Ni-N_2$ battery can be used for 5000 to 6000 recharging cycles and still supply the needed power. Whereas four 400 Ah Li/Ti S_2 batteries are required because they can only be recharged about 200 times this requires more batteries with a higher energy density to last 1 year of usage.

In GEO (scenario 2 Table 30), it was assumed that the battery would be cyclically charged for for 12 hours and discharged for 12 hours and be functional for 1 year. The Li/Ti S_2 battery is the preferred battery in this case. Five 100 Ah $Ni-H_2$ batteries are required to supply power during the eclipse period compared to two 400 an Li/Ti S_2 batteries (with power to spare). The cycle life of the battery is now of secondary consideration. What is of concern is to have a large energy density so that the battery system weight is a minimum during the 12 hour discharging. Using two 400 Ah Li/Ti S_2 gives an expected life of 373 days and weighs 460 lbs, but using the five 100 Ah Ni-H_2 batteries gives an expected life of about 14 years and weighs 936 lbs. Since the scenario only exists for one year the lower weight Li/Ti S_2 battery system is preferred.

Scenario 3 (Table 30) uses only primary batteries with an eight day mission. Since the Li/$SOCI_2$ cannot be recharged the continuous energy requirements sets the 8 day mission life. One 8000 Ah battery, weighing 1037 lbs satisfies the above requirement. This demonstrates the validity of using a rechargeable system for continuous energy requirements.

Table 30 Resupply Module Energy Weight Requirements To Transfer 45,500 lbs of Bipropellant and Fill 12 OMPS Bottles

Scenario 1 - LEO - 1 Year Life

a) Continuous Power - Use Ni-H_2 Batteries, and a 2.2KWe Deployable Photovoltaic Panel (DPP)
Peak Power - Use Li/Ti S_2 Batteries, and a 1.2 KWe DPP

System Weight 1208 lbs

b) Continuous Power - Use Li/Ti S_2 Batteries, and a 1.6 KWe DPP
Peak Power - Use Li/Ti S_2 Batteries, and a 1.1 KWe DPP

System Weight 1510 Lbs

Scenario 2 - GEO - 1 Year Life

a) Continuous Power - Use Ni-H_2 Batteries, and a 2.2 KWe DPP
Peak Power - Use Li/$SOCl_2$ Batteries

System Weight 1540 lbs

b) Continuous Power - Use Li/Ti S_2 Batteries, and a 2.2 KWe DPP
Peak Power - Use Li/$SOCl_2$ Batteries

System Weight 1064 lbs

Scenario 3 - LEO or GEO - 8 Day Mission

Use One 8000 Ah Li/$SOCl_2$ Battery

System Weight 1037 lbs

Table 31 Deployable Photovoltaic Panels

Name	Power (KWe)	Weight (lb)	BOL-W/lb	Deployed Area (ft^2)
Deployable Rigid Solar Arrays (DRSA)	1.0	76.4	13	110
Deployable Hybrid Solar Arrays (DHSA)	3.7	176	21	460
Solar Electric Propulsion System (SEPS)	12.5	416	30	1335
Power Extension Package (PEP)	26.8	1022	32	3020

BOL - Beginning of Life

3.2.9 Thermal Management of Pump and Compressor Systems

Heat from the bipropellant pumps can be rejected into the large heat reservoir available in the pumped MMH and NTO. The MMH is used to cool the pump. When 45,500 lbs of bipropellant is transferred at a flow rate of 6 GPM in seven hours, the pumped MMH temperature will increase about 3°F and the NTO temperature will increase about 2°F. As a result of the small temperature change of the space storable bipropellant, no further heat rejection is required.

Heat from the battery and the power conditioner can be removed by several heat pipes. Chemical to electrical conversion in the battery has an efficiency of at least 85 percent. By pumping 45,500 lbs of bipropellant and transferring helium from 12 MPS bottles, about 850 W of heat must be removed from the batteries and 530 W from the power conditioner (with a 87% efficiency). Table 32 presents two types of heat pipes that are candidates for use and the total weight of the requird heat tubes to remove 1380 watts.

Table 32. HEAT PIPE PARAMETERS

PARAMETERS	MONOGROOVE HEAT PIPE	AXIALLY GROOVED HEAT PIPES	
Length (M)	0.75	0.55	1
Weight (lbs)	5	3	6
Heat Transferred	215	200	150
T (evap - cond) (F°)	20	7.2	10.8
Tube material	Al	Al	Al
Number of grooves	1	20	27
Transport material	ammonia	ammonia	ammonia
Required tubes to remove 1380W	7	7	10
Total Weight (lbs)	35	21	60

Ammonia is selected as the coolant in the aluminum heat tubes. At room temperature ammonia has a good surface tension (20 dyne / cm), high thermal conductivity as a gas (0.0144 BTU/hr ft °F), and a high heat of vaporiztion (509 BTU/lbm). The aluminum heat tube is selected because it is compatible with ammonia, it is light weight, and it has a high thermal conductivity (118 BTU/hr ft °F).

Figures 78 and 79 present the monogroove heat pipe and the axially grooved heat pipe. The basic monogroove design contains two large axial channels, one for vapor and one for liquid. The small slot separating the channels creates a high capillary pressure difference which, coupled with a minimized flow resistance of the two separate phase channels, results in a high axial heat flux. The evaporation and condensation film coefficients are provided by circumferential grooves in the walls of the vapor channel. It is important that a continuous liquid flow path exists between the primary axial groove and the circumferential wall grooves in the evaporator sectin (otherwise dry out will occur).

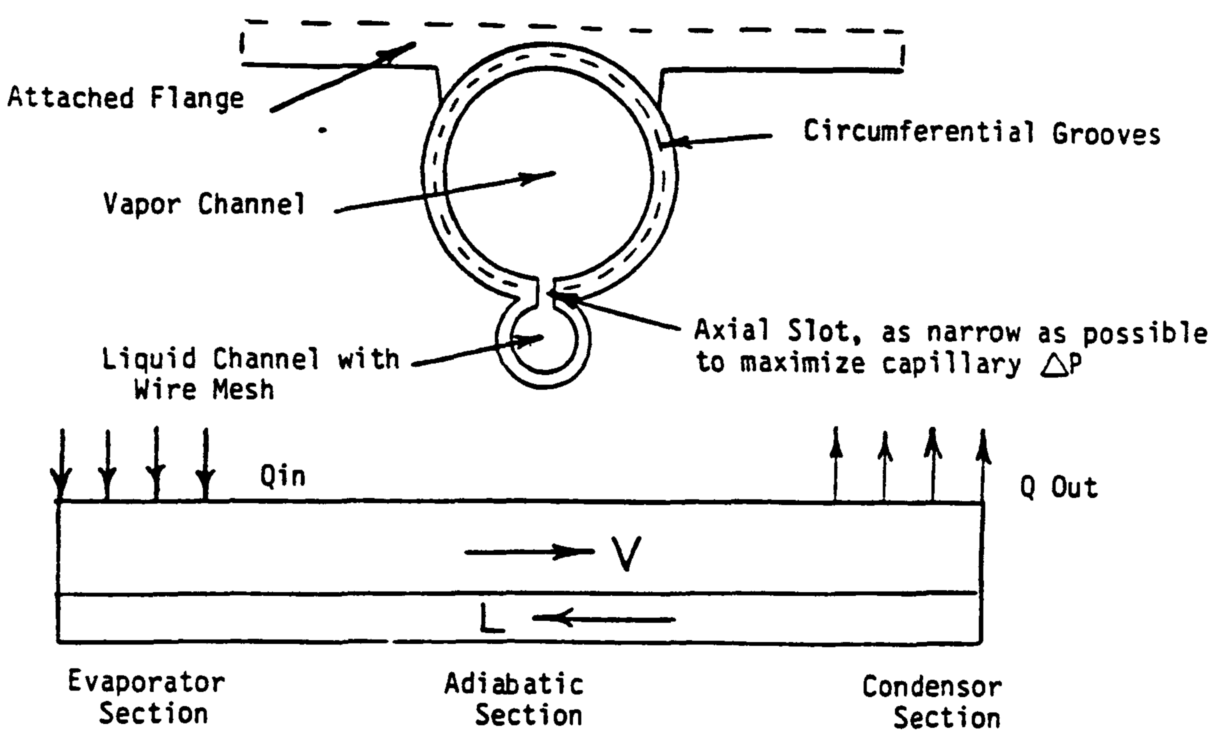


Figure 78. Monogroove Heat Pipe

The monogroove operating principle is characterized by two differential pressure relationships which must be satisfied simultaneously. The monogroove slot must develop enough capillary rise to overcome the vapor and liquid channel viscous losses.

ΔP (monogroove slot) $> \Delta P$ (vapor channel) $+ \Delta P$ (liquid channel)

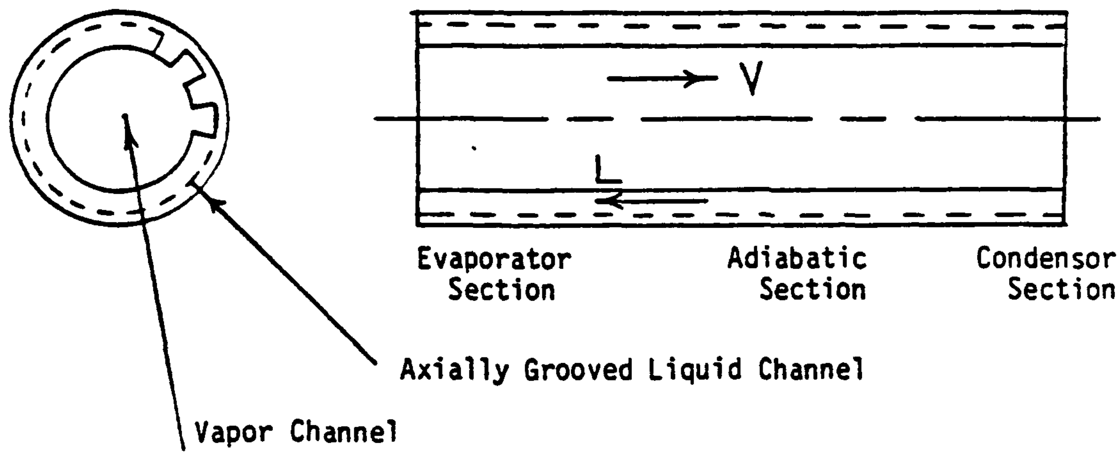


Figure 79. Axially Grooved Heat Pipe

The primary relationship requires the wall wick capillary pressure rise to offset the cumulative viscous pressure losses in the vapor channel, liquid channel, and circumferential wall grooves.

ΔP (wall cap) $= \Delta P$ (vapor channel) $+ \Delta P$ (liquid channel) $+ \Delta P$ (wall wick)

The internal configuration of an axially grooved heat pipe consists of a series of flow channels fabricated as a part of the tube wall and parallel to the longitudinal axis of the heat pipe. The large open-flow channel of such a design offers a low resistance to liquid flow. The large open-flow channels, however, are sensitive to liquid/vapor shear interaction.

Unlike the monogroove heat pipe, the axial groove heat pipe operating principle is characterized by one diffrential pressure relationship. The capillary pressure difference must equal the cumulative viscous pressure losses in the vapor channel, liquid channel, and the additional pressure drop in the liquid phase due to the countercurrent vapor flow.

ΔP (cap) $= \Delta P$ (vapor) $+ \Delta P$ (liquid) $+ \Delta P$ (shear)

A potential advantage of the monogroove heat pipe is the anticipated 2 KW heat flux. At this time, very few zero "g" heat tests have been performed on heat pipes. Zero "g" has been simulated, for 20 sec., by flying in a parabola. The test proved that the pipe would prime and the liquid would be drawn into the proper channel. A quickly designed test was performed on the eighth shuttle flight (when a payload was cancelled at the last minute). As a result of a shortage of design time (6 weeks) liquid crystal tapes that change colors at different temperature were used instead of installed thermocouples. The data was comparable to predictions and a heat flux of 70 watts was demonstrated.

Axially grooved heat-pipe technology has been developed extensively in the last 15 years for zero "g" work. An aluminum axially grooved heat pipe flow aboard the Orbiting Astronomical Observatory was still functioning after more than four years in orbit. A total of 55 aluminum axially grooved heat pipes were used on the Application Technology Satellite for almost three years of continuous flight operation. Since the technology of the axial grooved heat pipe is further developed than the technology on the monogroove heat pipe, the axial groove heat pipe is the suggested choice for the ERM.

There are three approaches that may be utilized to cool the compressor:

Approach A — The excess heat may be absorbed in either a mass storage device or a phase change material.

Approach B — The excess heat may be absorbed by a coolant loop and then rejected by either a pumped fluid radiator or a heat pipe radiator.

Approach C — The excess heat may be absorbed by a two-phase system and then rejected by either a pumped fluid radiator or a heat pipe radiator.

A short description of each of the three approaches follows.

APPROACH A: Figure 80 presents a heat transport loop using MMH as the mass storage device. An extended OMS tank holds 853 gal. of MMH. A 2 HP compressor running 30 hours will incrase the temperature of the extended OMS tank by 25°F. An advantage of using the MMH tank is that no radiator is required, which also indicates a disadvantage - you need a full MMH tank available when the helium is compressed. This approach can only be considered a viable approach when the helium can be transferred before the MMH.

APPROACH B: Figure 81 presents a heat transport loop using a pumped fluid as the coolant and either a pumped fluid radiator or a heat pipe radiator as the heat rejection device. An advantage of a pumped fluid system is the excellent thermal control that one has in the heat management. Flow rates and thus heat dissipation control is done by in-line valves. A disadvantage of a pumped fluid system is that it contains active components (such as pumps and valves) which will require periodic replacement, maintenance, and redundancy components.

Figures 82 and 83 present a pumped fluid radiator and a heat pipe radiator, respectively (both with an area of 53 F^2). The fluid radiator has the same advantages as a pumped fluid system; it is more reliable and better thermally controlled than a heat pipe system. A disadvantage of the pumped fluid system is if a leak occurs (due to micrometeoroid damage, etc.) complete system loss will occur; but, a leak in a heat pipe radiator will result in the loss of only the heat pipe that is damaged. This leaves the majority of the system intact and functional.

Freon-21 ($CHCl_2F$) was selected as the coolant fluid. Its normal boiling and freezing points are respectively 48°F and -211°F. The heat capacity of the liquid at 77°F is 0.256 BTU/lbm°R. Assuming a 2HP compressor is running for 30 ;hours and the maximum temperature of the compressor and compressed He gas is 110°F would require a flow rate of 1/2 GPM for the Freon-21 cooling loop. Another advantage of the Freon-21 is its excellent compatibility with other materials.

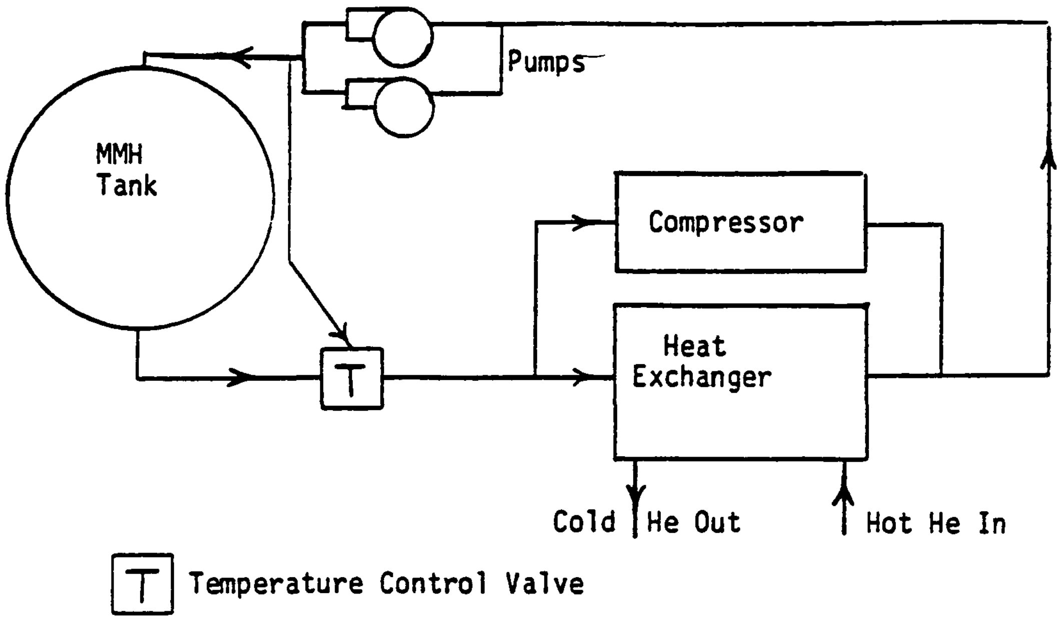


Figure 80. Heat Transport Loop Using Mass Storage

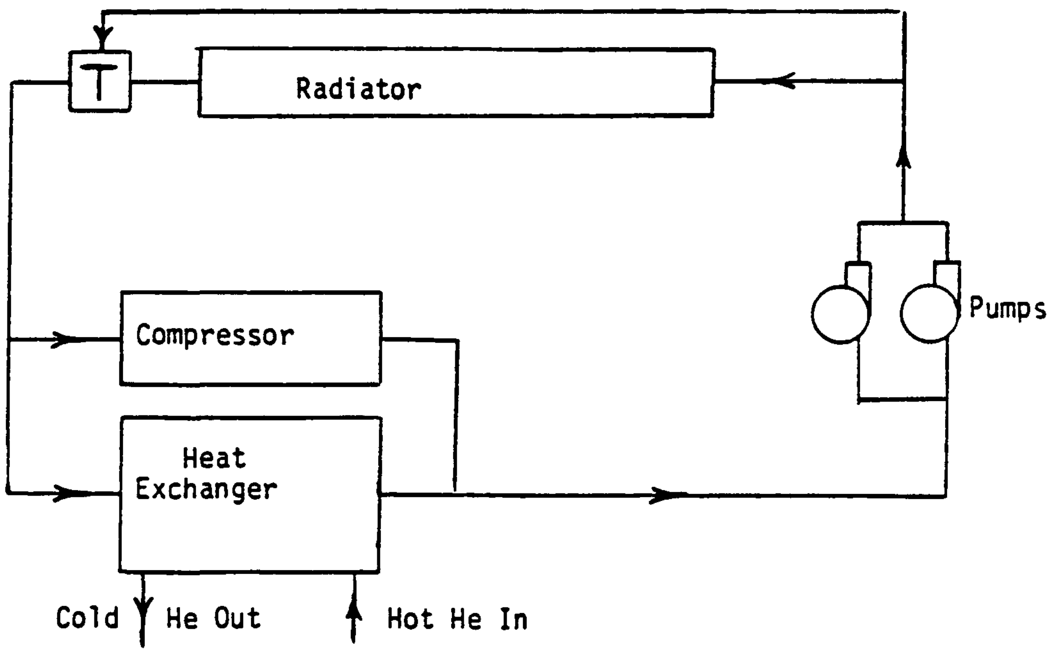


Figure 81. Heat Transport Loop Using Radiator

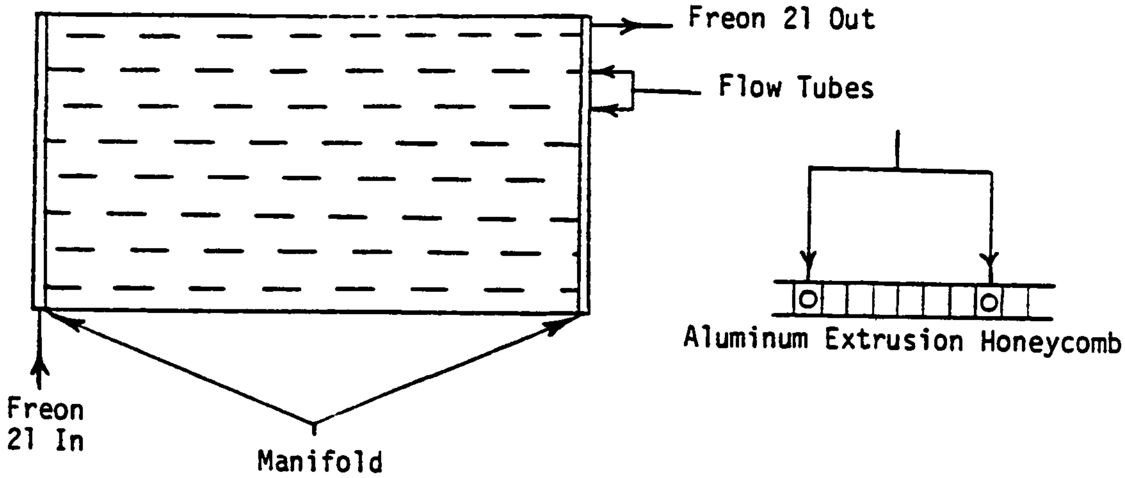


Figure 82. Pumped Fluid Radiator

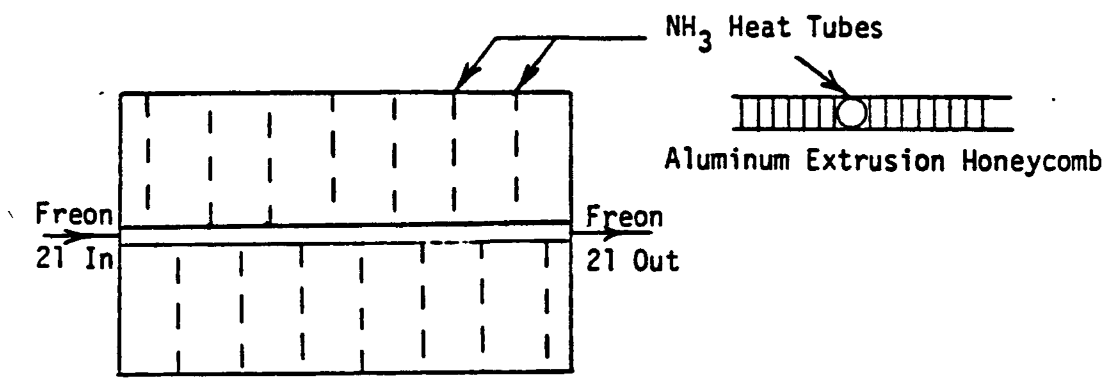


Figure 83. Heat Pipe Radiator

<u>APPROACH C</u>: Figure 84 presents a two-phase transport cooling loop. The primary function of an externally pumped heat pipe (EPHP) is to transport large amounts of heat across a small temperature gradient, thereby consuming a small amount of pumping power. The fluid circulation rate can be reduced by at least an order of magnitude because the heat of vaporization is much greater than the amount of heat the fluid can absorb just warming up. Cooling a 2-HP compressor operating for 30 hours will require a flow rate of 1/20 of a GPM and a 5-10 watt pump. A limitation or specific system requirement is needed by a capillary structure (porous wicking or machined grooves) near the exit of the radiator (condenser). Externally pumped heat pipes appear quite promising for the transport of large heat loads over long distances; but some analytical development, but no new technology, will be required before implementaion of the system.

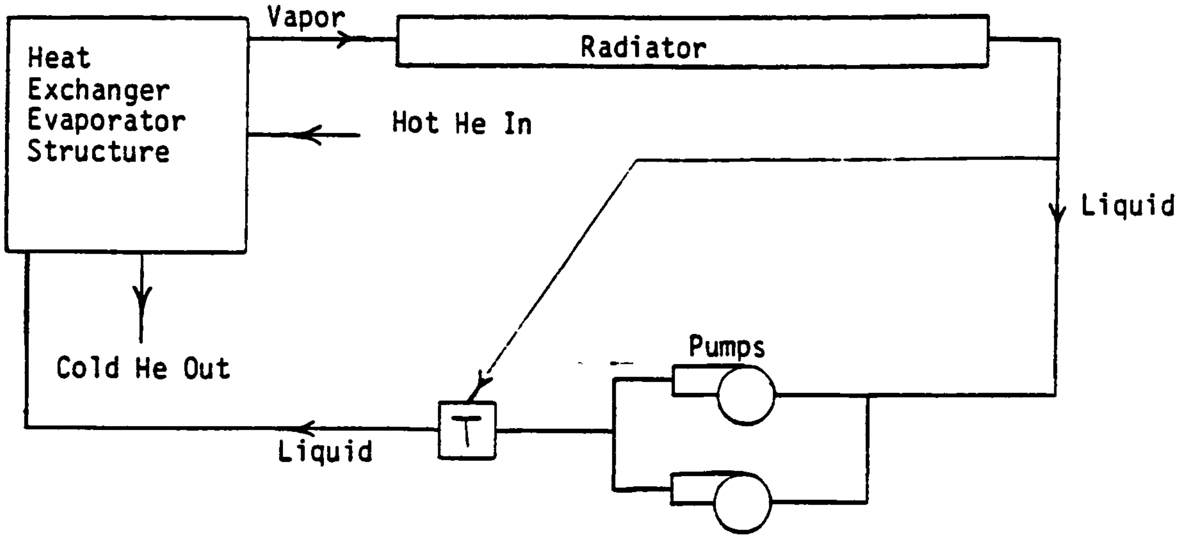


Figure 84. Heat Transport Loop Using a Two-Phase System

Ammonia is preferred over freon-21 in th EPHPs because of better surface tension, thermal conductivity, and a higher heat of vaporization. Since ammonia is to be used, the tube walls should be aluminum for compatibility.

Table 33 presents a weight comparison of each heat transport loop mentioned above. Approach A (heat storage in MMH) has the lowst system weight of the three aproaches. Approach A is also the only approach that does not have a 53 ft^2 radiator. Approach B (a liquid coolant loop) has listed both applicable radiators. As shown, there is very little weight difference between the two radiators, and thus very little weight difference between the complete systems. The approach C system (a vapor/liqud coolant loop) is of lower weight than the Approach B system because of the lower flow rates and power requirements.

PARAMETER	APPROACH A	APPROACH B		APPROACH C
		Pumped Fluid,	Heat Pipe	Heat Pipe Radiator
Radiator (lbs)	--	66	64	64
Heat Exchanger (lbs)	10	10	10	12
Temperature Control Valve (lbs)	5	5	5	5
Two pumps (lbs)	10	10	10	4
Tubing (lbs)	30	30	30	30
Extra battery weight (lbs)	15	15	15	2
Total (lbs)	70	136	134	117

Table 33. Weight Comparison of Heat Transport Loop

The key results of this analysis are as follows:

1. No heat dissipation devices are required for the space storable bipropellant pump. All heat will be absorbed in the transferred MMH and NTO. The MMH will cool the pump and increase in temperature of about 3°F and the NTO temperature will increase about 2°F.

2. Heat from the battery and the power conditioner can be removed by several ammonia heat pipes. In the case examined, the peak heat of 1380 watts can be removed by using 7 axially grooved heat pipes each transferring 200 watts to the structure.

3. Heat removal from the compressor system will occur at two separate areas. Heat must be removed from the compressor and the compressed gas. At this time, the best approach is to use a freon-21 coolant loop with an ammonia heat pipe radiator. The complete heat rejection system has a weigh of about 135 lbs. The externally pump heat pipe represents a potential 13% weight reduction over the above system but is not at present fully developed.

3.2.10 Pressure Drop Analysis

An analysis of the pressure drop occurring during a resupply module refueling of an OMV was accomplished. The resupply line schematic defined in 3.2.5 was used to describe the resupply design. The fill/drain lines for the OMV were taken from "OMV Alternative Systems Design Concepts" Proposal (February 14, 1985). Two different line sizes, 0.5 inch and .75 inch, were used for the resupply lines. Pressure drops were calculated for a resupply system using solenoid valves exclusively and then for a system using ball valves exclusively. Losses were calculated only for the propellant transfer lines and ullage return line. Purge and vent line losses were not examined in this study.

To begin with, the line lengths and number of bends were estimated for the resupply module. The resupply module was assumed to have 30 feet of line running from the supplier tank to the propellant quick disconnect (40 feet including lines to the propellant pumps) and a total number of line bends equivalent to twenty 90 degree bends. Using vendor supplied data the pressure loss across a solenoid valve (Parker P/N 5770021) for NTO and MMH flowing at 10 gpm was found. For NTO, a drop of 36 psid was found; for MMH a drop of 22 psid was found to occur across a solenoid valve. Ball valves have virtually no flow losses and hence were assumed have 0 psi pressure drop. Data on a Symetrics 0.5 inch quick disconnect give a pressure loss of 9.8 psid for NTO and 6.2 psid for MMH based upon a 10 gpm flowrate.

Tanks were assumed to have a 5 psid drop from the ullage gas to the tank outlet and flexline was assumed to have a five fold increase in pressure loss over an equivalent length of smooth pipe. Figure 85 presents a simplified resupply module schematic of all propellant lines and of the ullage return line used in the ullage transfer resupply configuration.

With 1/2 inch line and solenoid valves the pressure drop from the resupply tank through the propellant transfer disconnect was 252/163 psid for ox/fuel (this includes a 5 psid drop encountered in the tank). The pressure drop through the OMV's propellant transfer disconnect to the receiver tank's ullage was 196/126 psid for ox/fuel respectively. The pressure drop in the ullage

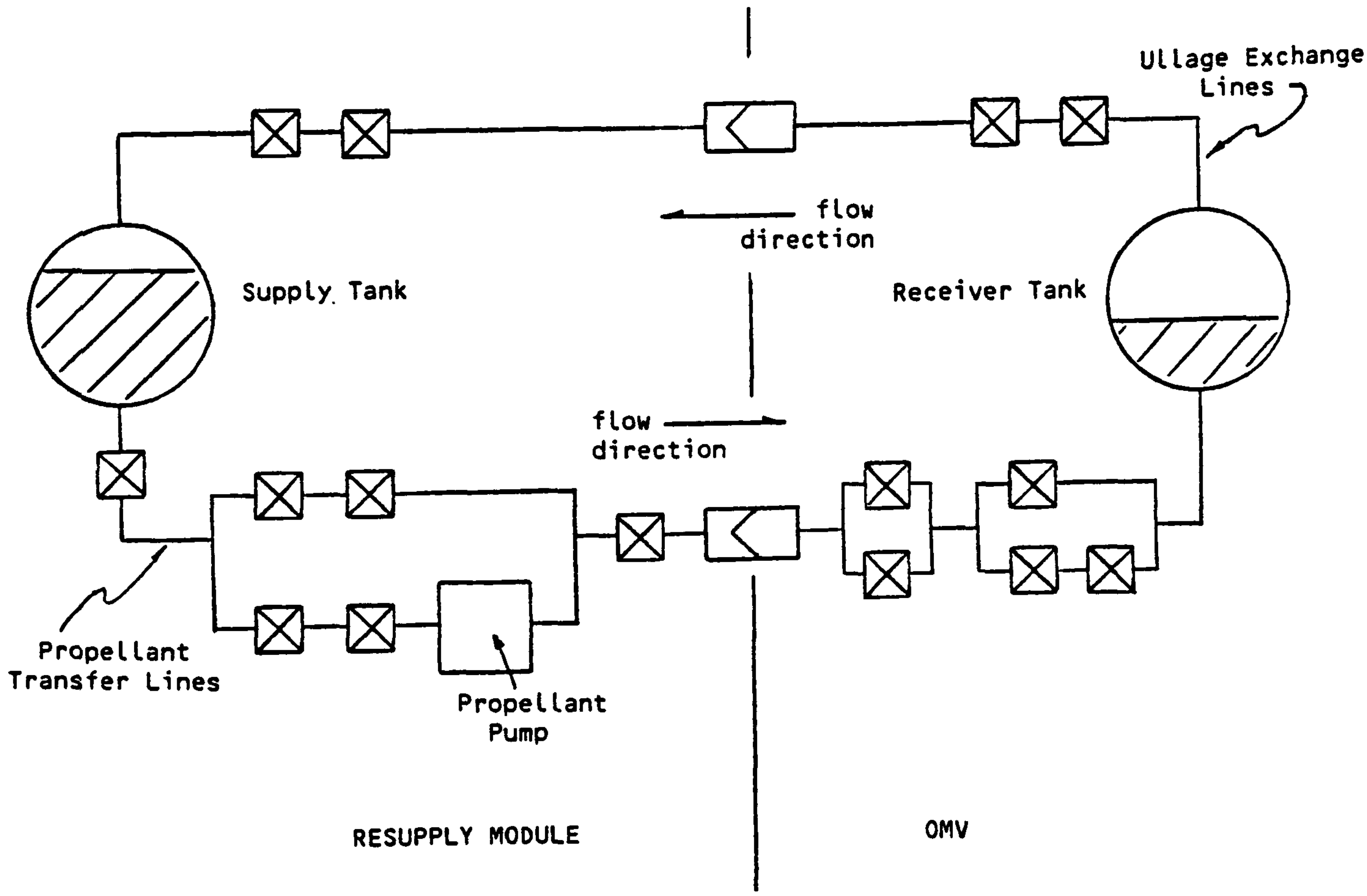


Figure 85. Simplified Resupply Module/Receiver Vehicle Fluid Line Schematic

return line was approximately 2 psid for 100% helium in the line. The total pressure loss was 450/291 psid for a ox/fuel system with 1/2" lines, solenoid valves, and 100% ullage gas in the ullage return lines. All of the pressure losses were at a flowrate of 10 gpm.

With 3/4 inch line and solenoid valves the pressure drop from the resupply tank through the propellant transfer disconnect was 175/110 psid for ox/fuel (this includes a 5 psid drop encountered in the tank). The pressure drop through the OMV's propellant transfer disconnect to the receiver tank's ullage was 136/86 psid for ox/fuel respectively. The pressure drop in the ullage return line was approximately 1 psid for 100% helium in the line. The total pressure loss was 312/197 psid for a ox/fuel system with 3/4" lines, solenoid valves, and 100% ullage gas in the ullage return lines. All of the pressure losses were at a flowrate of 10 gpm.

With 1/2 inch line and ball valves the pressure drop from the resupply tank through the propellant transfer disconnect was 108/75 psid for ox/fuel (this includes a 5 psid drop encountered in the tank). The pressure drop through the OMV's propellant transfer disconnect to the receiver tank's ullage was

88/60 psid for ox/fuel respectively. The pressure drop in the ullage return line was approximately 2 psid for 100% helium in the line. The total pressure loss was 198/137 psid for a ox/fuel system with 1/2" lines, ball valves, and 100% ullage gas in the ullage return lines. All of the pressure losses were at a flowrate of 10 gpm.

With 3/4 inch line and ball valves the pressure drop from the resupply tank through the propellant transfer disconnect was 31/22 psid for ox/fuel (this include a 5 psid drop encountered in the tank). The pressure drop through the OMV's propellant transfer disconnect to the receiver tank's ullage was 28/20 psid for ox/fuel respectively. The pressure drop in the ullage return line was approximately 1 psid for 100% helium in the line. The total pressure loss was 60/43 psid for a ox/fuel system with 3/4" lines, ball valves, and 100% ullage gas in the ullage return lines. All of the pressure losses were at a flowrate of 10 gpm.

A breakdown of the total pressure losses by component is listed in Table 34. As can be seen in this table, the solenoid valves dominate the pressure losses. Line losses and bend losses are of the same order followed by small losses in the flexlines, tanks, and quick disconnects. The total system pressure losses are listed at the bottom of the table.

The data for the 3/4" dia. line system pressure loss is plotted in Figure 86. This figure suggests the magnitude of pressure losses for propellant type and valve type. The maximum achievable pressure rise that can be achieved with near-term propellant pumps is in the range of 300 psid. This maximum achievable pressure rise limits the achievable flowrates to between 11 and 17 GPM for a system with solenoid valves and between 26 and 30 GPM for a system with ball valves.

At flowrates in the range of 1 to 7 gpm, adequate performance could be obtained with a system using solenoid valves. A major advantage in using solenoid valves rather than AC-motor operated ball valves, is in the components weight savings. A solenoid valve weighs 2.3 lbs, while a ball valve weighs 4.1 lbs. With the high number of valves required for the operation of a remotely operated resupply module, the use of solenoid valves rather than ball valves could mean a weight savings of as much as 250 lbs.

An analysis of the pressure transients experienced by the propellant transfer system during shutdown was conducted. To carry out the analysis a simplified propellant transfer system was simulated using an in-house fluid transient math model. The resupply module and receiver vehicle combination was simulated by connecting a constant pressure tank to a constant chamber pressure engine. The engine was used to simulate the receiver tank since the program that was used could not simulate propellant transfer between tanks. The isolation valve located upstream of the injector (i.e., receiver tank isolation valve) was closed at a specified rate while the engine chamber pressure and supply tank pressures were held constant. As the isolation valve closed a fluid transient was initiated, and the time history of the fluid pressure immediately upstream of the isolation valve was observed. Figure 87 compares the analytical model and the simplified fluid line schematic.

Table 34: TOTAL SYSTEM PRESSURE LOSS: COMPONENT BREAKDOWN (psid) AT 10 GPM FLOWRATE

Component	0.5" Line				0.75 " line			
	Solenoid Valve System		Ball Valve System		Solenoid Valve System		Ball Valve System	
	NTO	MMH	NTO	MMH	NTO	MMH	NTO	MMH
Tank Losses	10	10	10	10	10	10	10	10
Propellant Feed Line	65.1	49.9	65.1	49.9	9.4	7.4	9.4	7.4
Flexline	9.5	7.1	9.5	7.1	1.5	1.2	1.5	1.2
bends	91.8	55.6	91.8	55.6	18.5	11.0	18.5	11.0
quick disconnects	19.6	12.4	19.6	12.4	19.6	12.4	19.6	12.4
valves	252	154	0	0	252	154	0	0
Ullage Exchange Line (100% Helium in line)	2	2	2	2	1	1	1	1
Total System Loss	450	291	198	137	312	197	60	43

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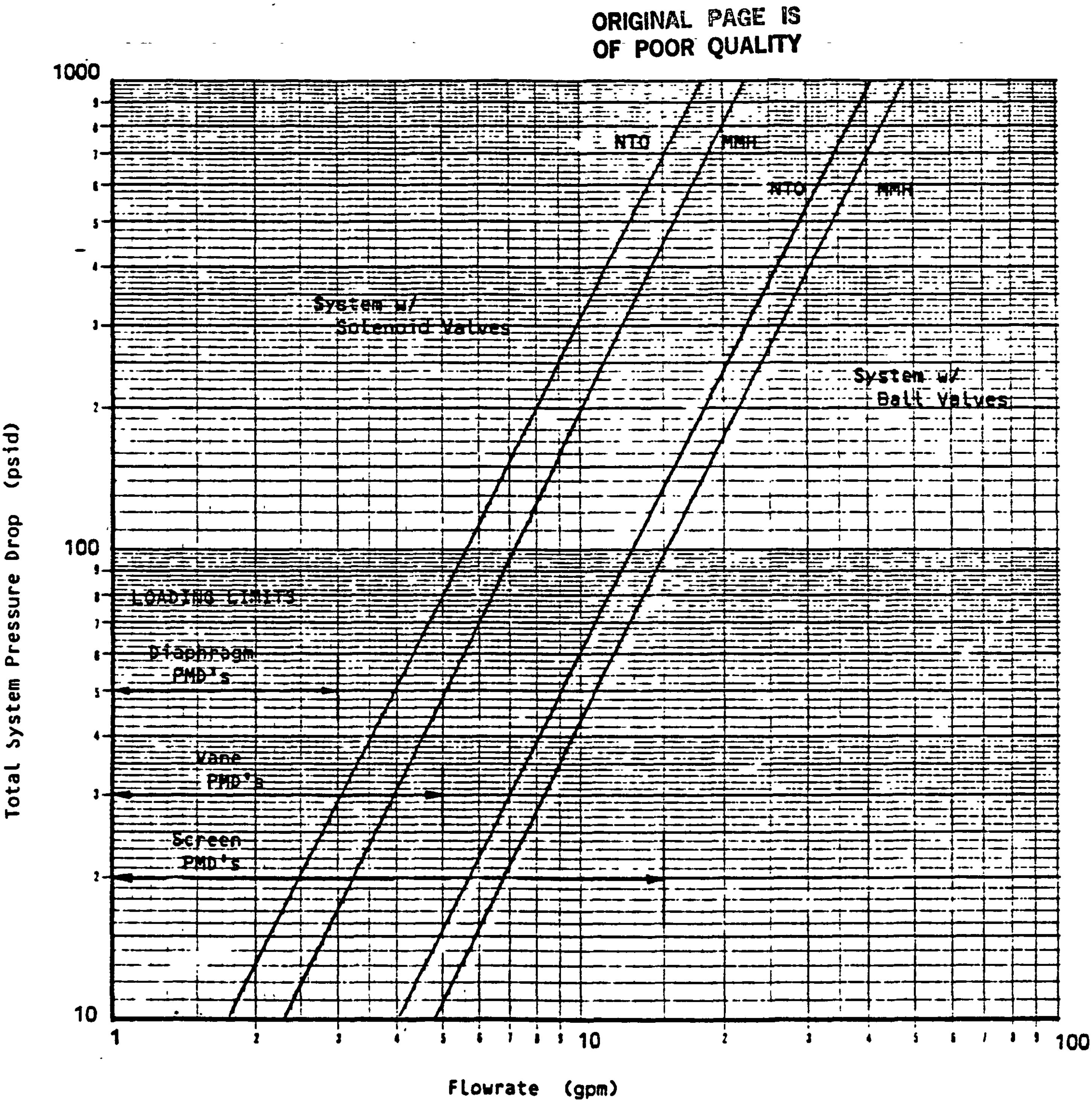

Figure 86. System Pressure Loss Vs. Flowrate

Figure 88 shows the maximum surge pressure as a function of initial flowrate. With a valve closing time of 8 msec, the surge pressure slope is 83 psid/GPM. While these values are preliminary, they do suggest the magnitude of the transients that can be expected in the resupply system. During a pumped propellant transfer, the shutdown transients can be minimized by tapering the pumping rate down to a low value and then shutting the system down. However, if the propellant transfer operations had to be terminated immediately an isolation valve would be closed. The sudden closure of a valve would initiate a pressure transient with a magnitude similar to the values predicted above, and the fluid system would have to be designed to withstand fluid transients of this magnitude.

From this analysis, it can be concluded that solenoid valves can be used in the resupply module for fluid management. Using a pump with a 300 psid pressure rise in a resupply module that used solenoid valves would yield maximum flowrates in the range of 10 to 15 gpm. Most fluid transfer operations will have maximum flowrates limited by propellant management devices and not by resupply system pressure losses. Fluid transients during shutdown were examined, and it was observed that for worst case conditions (high flowrate, fast acting valve) a peak surge pressure rise of up to 83 psid/GPM above steady state conditions was obtainable.

3.2.11 Propellant Management Considerations

The selection of propellant tanks is an important step in the design of any low-g propellant handling system. In many cases the tanks constrain the vehicle's performance capabilities, including constraining the number of attitude control thrusters allowed to fire, the duration of thruster burns, minimum time between burns and allowable maneuvers along some axes. The tanks will also define the vehicle's capability to either resupply or to be resupplied under low-g or zero-g acceleration fields. The selection tanks also helps determine the design of the rest of the system, including the need for hazardous waste scavenging system, liquid or gas pumps, ullage transfer systems, etc.

Propellant tank and propellant acquisition/management device requirements for tanks operating as either an OMV extended kit or propellant resupply module were determined and are summarized in Table 35. State-of-the-art proipellant tank concepts were then investigated to determine their potential for meeting the identified requirements. Finally, current tanks representative of each of the concepts were researched to determine estimate of the capabilities of each concept.

Three propellant acquisition concepts were investigated: elastomeric diaphragms, surface tension vane devices, and surface tension screen acquisition devices. A summary of each type's advantages and disadvantages is presented in Table 36. It was determined that the concept best suited to the needs identified in this study, including operating capabilities and the capability of being resupplied on-orbit, was elastomeric diaphragms. However, no suitable diaphragm has been flight qualified for multiple cycles when used with NTO. It was, therefore, recommended that the developments in this area be closely monitored. Of the remaining two concepts, vane device tanks have the very desirable capability of controlling the position of the ullage bubble in zero-g. This capability allows the venting and transfer of the ullage to the resupply tank during propellant transfers, eliminating the need for hazardous waste scavenging systems in the extended mission kit or resupply module. Vane devices may not be suitable for operation in acceleration fields such as will be required of OMV and OMV extended mission kit tanks. Satisfactory performance may be achieved by placing a propellant retention sponge type device over the tank outlet, although this would require

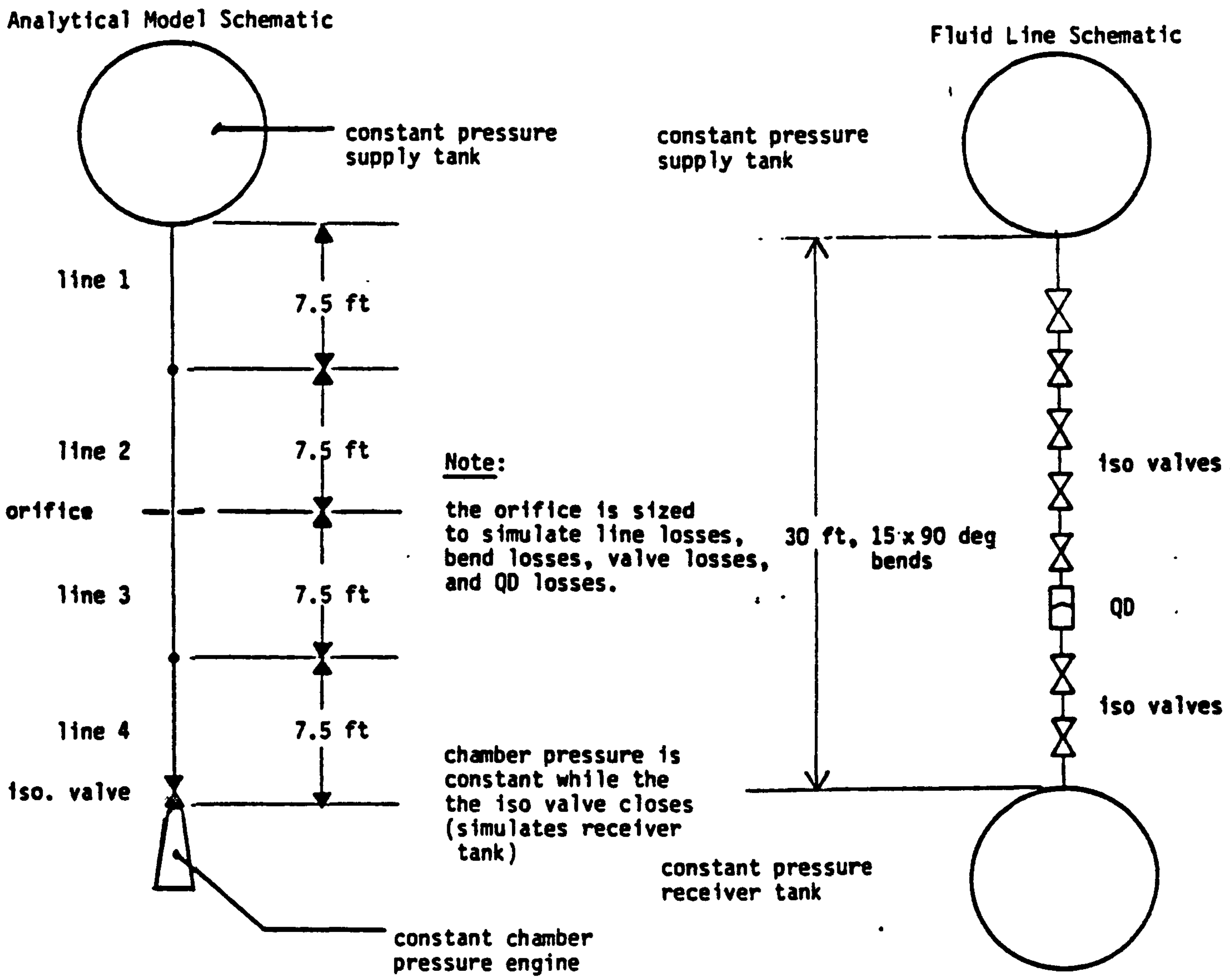

Figure 87. Analytic and Fluid Line Schematic Used in Fluid Transient Analysis
(0.5 inch line diameter)

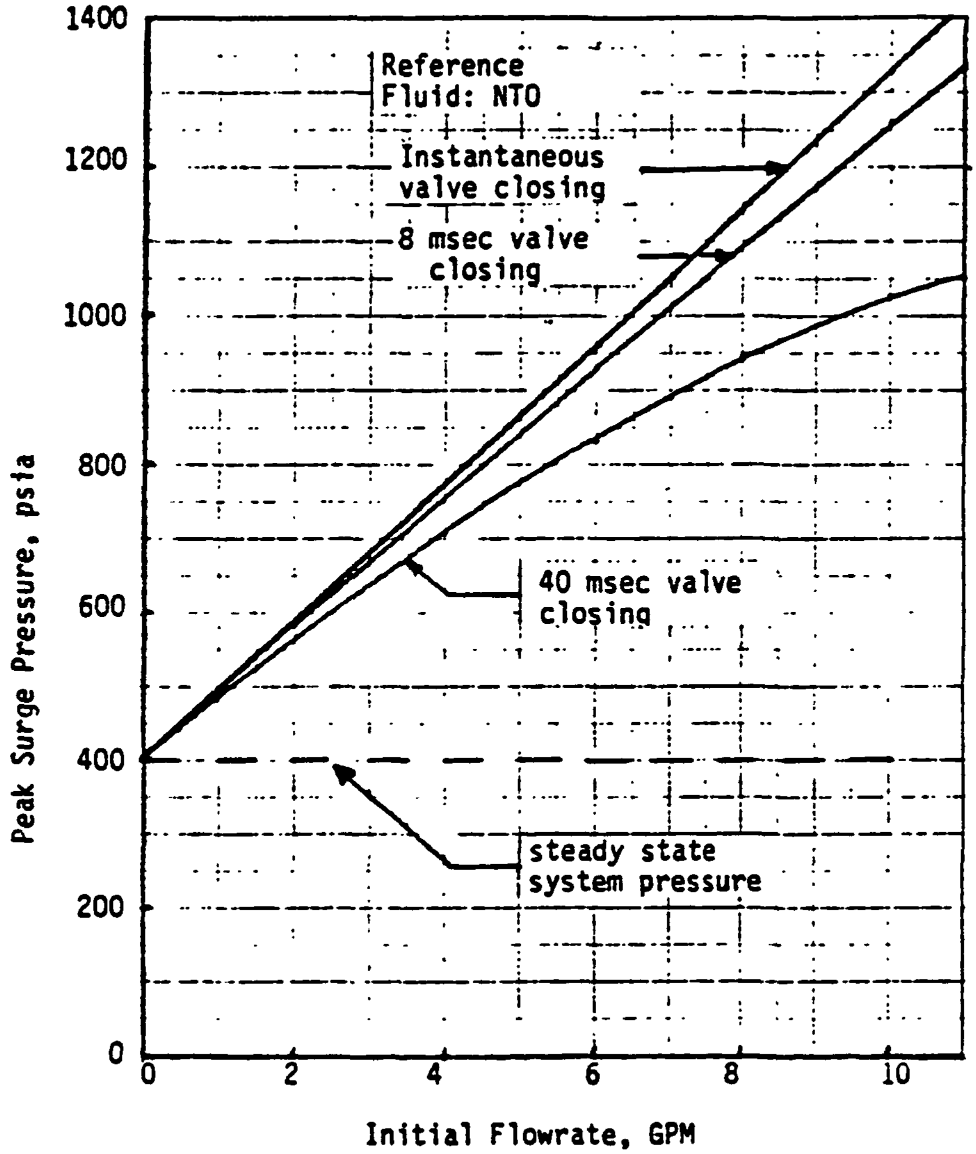

Figure 88. Peak Surge Pressure Vs. Initial Propellant Flowrate

constraints be placed on OMV maneuvers, such asduration of maneuvers producing adverse accelerations, and time between maneuvers to allow the sponge to refill. Surface tension screen acquisition device tanks perform very well under the environments required of an OMV or extended mission kit tank. The problem if there is no known screen device tank which has the capability of positioning the ullage bubble. It is felt however that a screen tank can be modified to obtain this capability.

Table 35. RESUPPLY MODULES AND OMV EXTENDED
MISSION KITS TANK REQUIREMENTS
SUMMARY

	Extended Mission Kit (Bi-Propellants)	Resupply Modules
o Acceleration Level		
o +X	0.16 g	N/A
o Omnidirectional	0.0048 g	N/A
o Steady Coast Resupply	1×10^{-7}g	1×10^{-7}g
o Expulsion Rate		
o +X	1.39 lb/sec NTO,0/85 lb/sec MMH	N/A
o Omnidirectional	0.048 lb/sec NOT,0.028 lb/sec MMH	N/A
o Steady Coast Resupply	1.0 lb/sec NOT,0.6 lb/sec MMH	0.6 lb/sec
o Propellant Temperature	40 - 100° F.	40 - 100°F.
o Operating Pressure	257 psia - TBD	TBD
o Volume	200,448 in^3*	N/A
o Vibration	100 Missions Shuttle Launch & Ascent	100 Missions Shuttle Launch & Ascent
o Shock	Shuttle Crash Landing Loads	Shuttle Crash Landing Loads
o Expulsion Cycles	100	100
o Pressure Cycles	200	200

TABLE 36. Summary of Propellant Acquisition Concepts

	DIAPHRAGM	VANE DEVICE	SCREEN DEVICE
Advantages	o Positive Ullage Positioning o Can be resupplied with or without venting. o Operates over wide range of acceleration levels. o No potential for gas ingestion. o High Expulsion Efficiency	o Positive Ullage Positioning o Can be resupplied with or without venting.	o Operates over wide range of acceleration levels. o Few Operational Constraints o Does not require propellant to be in specific location.
Disadvantage	o Diaphragm Life Limits o Not applicable to NTO.	o Operates only under extremely low accelerations. o Must allow time for propellant to resettle between maneuvers.	o No ullage positioning. o Must vent prior to refilling.
Examples of Users	o Shuttle APU o TDRSS	o Viking Orbiter o INSAT o ARABSAT	o Shuttle RCS o Shuttle OMS o INTELSAT

Before an examination of propellant tanks and propellant acquisition device concepts can be made, it is first necessary to estimate the propellant tanks operating environment and performance requirements. One of the most important parameters is the acceleration field under which the tank must supply propellants. The Rockwell OMV design consists of 6 main engines, 110 lb_f thrust each, all firing in the +X direction (+X translation), four 5 lb_f thrust RCS engines in both the plus and minus X directions, and two 5 lb_f thrust RCS engines in each of the $\pm$Y and $\pm$Z directions. The approximate dry weight of the OMV is 4200 lbs., including an estimate for residual propellants. Since sizes and weights of resupply modules and OMV extended mission kits have not been finalized to date, these weights will not be included in this analysis. With one of these kits attached to the OMV, it can be expected that the accelerations will be 25 to 50% less. The maximum expected linear acceleration then is along the x-axis and is approximately:

$$A_{x_{max}} = \frac{(6 * 110 \text{ lbf} + 4 * 5 \text{ lbf})}{4200 \text{ lbm}} \, 32.174 \frac{\text{lbm ft}}{\text{lbf s}^2} = 5.2 \frac{\text{ft}}{\text{s}^2} = 0.16 \text{ g's}$$

Any tank selected would be installed along this axis so that this acceleration accentuates the greatest capability of the tank. The largest adverse acceleration would then be in the -X direction due to the 4 RCS engines and would be approximately:

$$A_{adverse_{max}} = \frac{(4 * 5 \text{ lbf})}{4200 \text{ lbm}} * 32.174 \frac{\text{lbm ft}}{\text{lbf s}^2} = 0.15 \frac{\text{ft}}{\text{s}^2} = 0.0048 \text{ g's}$$

Any translation in any other direction would impose an acceleration of 0.075ft/s^2 (0.0024 g's) corresponding to 2 RCS engines firing. The maximum rotational rate about any axis is estimated to be 0.052 rad/sec (3°/sec). The radius of rotation (i.e. OMV - extended mission kit c.g. to the center of the tanks) is estimated to be 5 feet for x-axis rotations (i.e. roll) and 10 feet for pitch and yaw. This yields centripetal accelerations (rw^2) of 0.013, 0.027, and 0.027 ft/sec^2 for rotations about the roll, putch and yaw axes. Since it is expected that pitch and yaw rotations will not be sustained to settle propellant, especially in conjunction with the worst case translational accelerations, and since the translational acceleration requirements are much more severe, pitch and yaw maneuvers need not now be considered. Likewise, assuming roll maneuvers will not be performed in conjunction with translations (except possible along the x axis), and since the translational requirements are again much more severe (0.15 or 0.075 ft/s^2 vs. 0.013 ft/s^2) roll maneuvers will also not be considered at this time.

Another important design requirement is the tank's expulsion rate. This requirement can be determined by either the propellant flow rate required by the thrusters of the time available to resupply either the OMV or satellites. The propellant flow rate to the OMV's six main thrusters (I_{sp}=312 sec) simultaneously is:

$$\frac{(6 * 110)}{312} = 2.12 \frac{\text{lb}}{\text{sec}}$$

at a mixture ratio of NTO to MMH of 1.65. It is also desired to fire up to six 5 lb_f RCS engines (I_{sp} = 260 sec) in conjunction with the main engines resulting in an additional flow of:

$$\frac{(6 * 5)}{260} = 0.115 \ \frac{\text{lb}}{\text{sec}}$$

or a total flow of 2.24 lbs/sec (1.39 lbs/sec NTO, 0.85 lbs/sec MMH). Again it is assumed that the tanks will be installed such that the settling acceleration caused by the main engines requiring this flow rate causes the propellant to settle in the most advantageous location in the tank (i.e. probably over the tank outlet). The worst case flow rate required under an adverse acceleration would be that required to support either 2 or 4 RCS engines, or 0.038 (0.024 lbs/sec NTO, 0.014 lbs/sec MMH) or 0.076 (0.048 lbs/sec NTO, 0.028 lbs/sec MMH) respectively.

Propellant temperature is another important parameter in the performance of any surface tension driven propellant acquisition device. For this study it will be assumed that the temperature will be controlled to between 40 and 100°F.

The OMV extended mission kit will be required to supply propellants to the OMV. The operating pressure of the OMV is planned to be 257 psia, the same as the Space Shuttle OMS regulated pressure. It may be desired, however, to have different operating pressures for extended mission kits and resupply modules depending on the type of propellant transfer which is selected. This will be determined later and for now will not constrain the selection of any tank.

Since the OMV, SPER modules, and extended mission kits will be launched and returned to earth in the Space Shuttle, They should each be capable of sustaining a maximum of 100 missions worth of launch, ascent and random vibration. Also, any propellant tank which flies on the Orbiter is required to be certified to be able to withstand crash shock loads. Various tanks, including the Shuttle OMS and RCS screen acquisition device propellant tanks and the Shuttle APU diaphragm tanks, have been certified to these requirements. It will therefore be assumed that these requirements will not preclude the use of any tank, although it should be noted that they may require that off-the-shelf tanks be redesigned and requalified. Additionally, it was assumed that each tank will be required to be capable of the equivalent of a maximum of 200 expulsion cycles and 200 pressure cycles.

Three low-g propellant tank concepts exist which have the potential of meeting the above requirements. These include elastomeric diaphragms, surface tension screen acquisition devices, and surface tension vane devices. Each of these concepts will be discussed in the following sections with an attempt made to identify some of the advantages of each type. Elastomeric expulsion devices have received wide use throughout the industry in mono-propellant (hydrazine) systems. To date, although much effort is being expended, no diaphragm has been qualified for repeatable service with NTO. Examples of operating systems currently employing elastomeric diaphragms include the tracking and data relay satellites (TDRS) and the Space Shuttle Auxiliary Power Units (APU). The use of these diaphragms in the Shuttle APU tanks has demonstrated the ability of the tanks to meet the requirements of Table 35 for mono-propellant resupply module, specifically the vibration, shock, pressure cycles, expulsion cycles and expulsion rate requirements.

The major advantage of employing elastometric diaphragm tanks in a resupply module is the ease with which these tanks can be resupplied in low "g" environment. A system using these tanks can operate either in a blowdown mode or in a pressure regulated system. In both cases, either recompressing the ullage or venting the ullage during a refill is comparatively easy. Other advantages of elastomeric diaphragm tanks include: typically exhibiting expulsion capabilities of 99% and greater; the independence of expulsion on acceleration levels; the independence of performance on on-orbit propellant sloshing; and the improbability of pressurant gas ingestion to the thrusters.

The major disadvantage to elastomeric diaphragm tanks is the inability to qualify a diaphragm compatible with NTO. There is a good possibility, however, that a NTO diaphragm could be developed in the near future. Work in this area should be closely monitored. If a diaphragm is developed which meets all of the requirements of Table 35, it would be a leading candidate for any tank system developed in the future.

A second propellant tank concept capable of managing propellant in a low "g" environment utilizes surface tension vane devices. Vane devices, manufactured of sheet metal, rely on the very small surface tension forces of the propellant to orient the propellant over the outlet. A typical vane device is shown in Figure 89. (This device was developed for the Viking 75 Orbiter.)

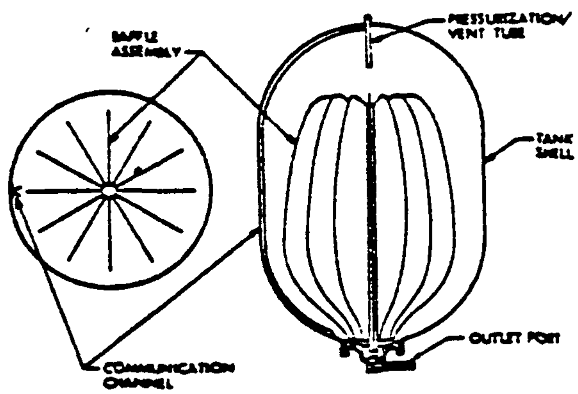

Figure 89. Viking Orbiter Baseline Propellant Management Device

The design of the vanes, including their height, number and shape (i.e., distance between the tank wall and the vane) determine the forces acting on an off-center ullage bubble and the stable location of the ullage. Because the surface tension forces are so low, these tanks are designed to operate under an extremely low acceleration field, corresponding to a bond number (ratio of acceleration forces to surface tension forces) of approximately 1, or in an acceleration field which causes the bulk propellant to settle over the outlet. For a 36-inch spherical or cylindrical tak, a bond number of 1 corresponds to an acceleration of approximately 1×10^{-5} g's. Because of this, the use of these tanks is normally confined to providing propellant under zero "g" to start an engine which will then provide thrust to settle the propellant over the outlet. The capability of a vane device can be enhanced by designing a propellant retention honeycomb type sponge over the tank outlet. If this sponge is designed large enough to supply propellant during any adverse acceleration and then allowed to refill before any subsequent thruster firings, satisfactory tank performance could be obtained.

In many cases this is impractical as the time necessary to resettle the propellant becomes unacceptable. If the propellant becomes reoriented such that it loses contact with the vane device, surface tension forces will be unable to act to resettle the propellant over the tank outlet. For this situation a communication channel is designed into the tank, extending from the top of the tank to the bottom. For propellants with approximately a zero contact angle (angle formed at the intersection of the propellant interface and a solid surface, measured within the propellant) such as NTO and MMH, their surface tension will force propellant into the channel and then wick along the channel to the tank's bottom. In near zero "g" this can take several hours. This can be acceptable in a system such as a satellite when thruster burns are required infrequently, but would be unacceptable to a maneuvering vehicle such as an OMV or Space Shuttle. For propellants with a high contact angle such as hydrazine, the performance of a communication channel is very much degraded. The alternative would be to perform a settling burn of approximately 10^{-3} g's, which could resettle the propellant in about 30 seconds.

The vane device used in the V075 tanks was certified to an expulsion rate capability of 0.64 lb/sec NTO, 0.42 lb/sec MMH, and 1.4 lb/sec N_2H_4. It is expected that these rates could be greater under a settling acceleration.

The only operating advantage of a vane device tank over a surface tension screen acquisition device tank is its ability to control the location of the ullage bubble. This is necessary if it is desired to vent the tank such as could be required in an emergency situation or during a resupply of the tank on-orbit. Low "g" reloading of tanks of this nature has been experimentally shown possible as long as the inertial forces of the incoming propellant do not exceed its surface tension forces. The dimensionless Weber number is used to define this critical range. The Weber number is defined as the ratio of the inertia forces to the surface tension forces by:

$$W_e = \frac{V^2 R}{2\beta} \quad or \quad \frac{Q^2}{2\pi^2 R^3 \beta}$$

Where: Q = Volumetric Flow Rate
R = Tank Inlet Radius
V = Incoming Liquid Velocity
β = Kinematic Surface Tension

For the V075 vane tank, it has been shown experimentally through drop tower testing that the propellant-gas interface remains stable (i.e., reloading is possible) for Weber number less than 7. This traslates into volumetric flow rates of 2.81 in^3/sec NTO, 4.13 in^3/sec MMH, or 5.37 in^3/sec N_2H_4. For the V075 tank (43,000 in^3 volume) this means a resupply of 4.25 hrs., 2.89 hrs. and 2.22 hrs. for NTO, MMH, and N_2H_2, respectively. If this design was to be revised to incorporate baffles over the tank inlet this time could be reduced by at least a factor of 2.

The major disadvantage of vane device propellant tanks is, as was discussed previously, the inability of the propellant management device to maintain propellant over the tank outlet under adverse accelerations greater than approximately 10^{-5} g's. This capability can be enhanced by incorporating a propellant retention sponge over the tank outlet although this will necessitate placing maneuvering constraints on the duration of adverse accelerations and time between maneuvers to allow propellant to resettle if a tank of this type is used on an OMV or an OMV extended mission kit. However, a vane device tank would work very well in a resupply module as long as the propellant resupply occurs in near zero gravity.

Another problem exists while outflowing vane device tanks in low "g" environments. That is the tanks tend to outflow the propellant nearest the tank outlet resulting in premature ingestion of gas while a significant amount of propellant remain in the tanks. For propellants with low contact angles (NTO and MMH) which have a low "g" configuration of approximately a column over the outlet in the PMD, outflow in low "g" causes a "necking" of this column at the outlet, allowing gas to enter the outlet. Drop tower tests have shown that outflow in this condition will result in two-phase flow with a series of bubbles with a diameter the size of the outlet line. For propellants with a high contact angle (hydrazine) which have a low "g" configuration of approximately a flat interface with propellant settled over the outlet, outflow will cause "suction dip", or a pull through of gas through the propellant. Either case can result in approximately a 10% residual trapped in the tank. For the Viking Orbiter missions, however, with a relatively high-g (0.1g) settling propellant over the outlet, the expulsion efficiency of the tanks was as near to 100% as the on-board instrumentation could measure. Other problems relating to premature gas ingestion have been experienced using vane device tanks. One problem was caused by sloshing of propellants upon engine start-up which causes propellants to leave the outlet area. For the Viking Orbiter missions this gas ingestion was estimted to be 20 in^3 per engine start.

The ability of vane devices to withstand the vibration and shock requirements is now know. During vibration testing of the V075 tanks "several baffle elements experienced localized yielding as a result of three corner folds induced by the slosh motion." Assuming the Viking Orbiter One Mission requirements were not as severe as 100 missions of Shuttle launch vibration, a serious problem could exist. Further, the V075 tests were conducted with water. It was discovered during the Shuttle RCS Tank Certification Program that NTO propellant loads during vibration can be an order of magnitude greater than those in water. Also, it may be impossible to support the vanes with stiffeners or struts without impacting the tanks' ability to move and position the ullage bubble. Therefore, much attention should be given to this area before a vane device tank is chosen for the ERM, OMV, or OMV extended mission kit.

Although there appears at this time to be several serious disadvantages to using vane device tanks for the applications of interest to this study, the advantage of being capable of positioning the ullage may be of such benefit that these type tanks should receive thorough study. Since there is an infinite number of possible vane designs, and since each vane design is chosen for a specific set of requirements, it is possible that an acceptable and effective vane design can be found for these applications.

Surface tension screen acquisition device propellant tanks again rely on the very low surface tension forces of the propellant to separate the pressurant gas from the propellant. In this case, barriers and flow channels made of very fine screens (pores measured in microns) allow passage of propellants but because of the surface tension forces, restrict the flow of gas. Propellant then flows beneath the screen barriers or through the channels to the tank outlet. Because of the very small pore size, a screen device tank is capable of operating in much higher acceleration fields than are the vane tanks.

There are an unlimited number of possible screen device designs, varying in complexity from a simple start basket over the outlet to a total low "g" propellant acquisition system. The Shuttle Orbiter OMS tank is designed to maintain propellant over the tank outlet during adverse accelerations such as would be caused by RCS Maneuvers, and then to provide propellant for OMS engine start, which then provides thrust to settle the propellant. To do this, the Shuttle Orbiter OMS tank propellant management device consists of a screen bulkhead isolating the outlet end of the tank which contains a screen gallery system. The screen bulkhead is designed to provide a barrier to hold propellants, sufficient for approximately 18 OMS engine starts, at the AFT end of the tank. The OMS tank propellant management system configuration is shown in Figure 90.

The greatest advantage of screen device tanks is their capability to operate in a wide range of acceleration levels, from essentially zero-g to greater than 0.07 g's. They can be designed to operate with propellant settled in a specified location due to engine thrust or under omni-directional accelerations which may settle propellant in any location in the tank. This makes screen tanks free from many operational constraints. There is also minimum time requirement between burns to allow propellant to settle or reorient. The Shuttle Orbiter OMS tank has shown some capability during past Shuttle flights for extremely low outflow rates (0.056 lb/sec NTO, 0.035 lb/sec MMH) in zero "g". With a settling acceperation it outflows 12 lb/sec NTO and 7.3 lb/sec MMH to the OMS engine.

The disadvantage of screen device tanks is their inability to position the ullage bubble in the tank. Without this ability, it is impossible to vent the ullage from the tanks while propellant is being resupplied to them. This makes it necessary to vent the tanks, both ullage and remaining propellant, to near vacuum before reloading, and then vacuum loading. This venting of ullage gases and propellant could necessitate the incorporation of a hazardous waste scavenging system in any resupply module or extended mission kit. It may be possible to eliminate this problem by designing a stand pipe in the tank which would locate the ullage in a desirable position under near zero gravity. Using a stand pipe, in conjunction with a liquid/vapor separating device,

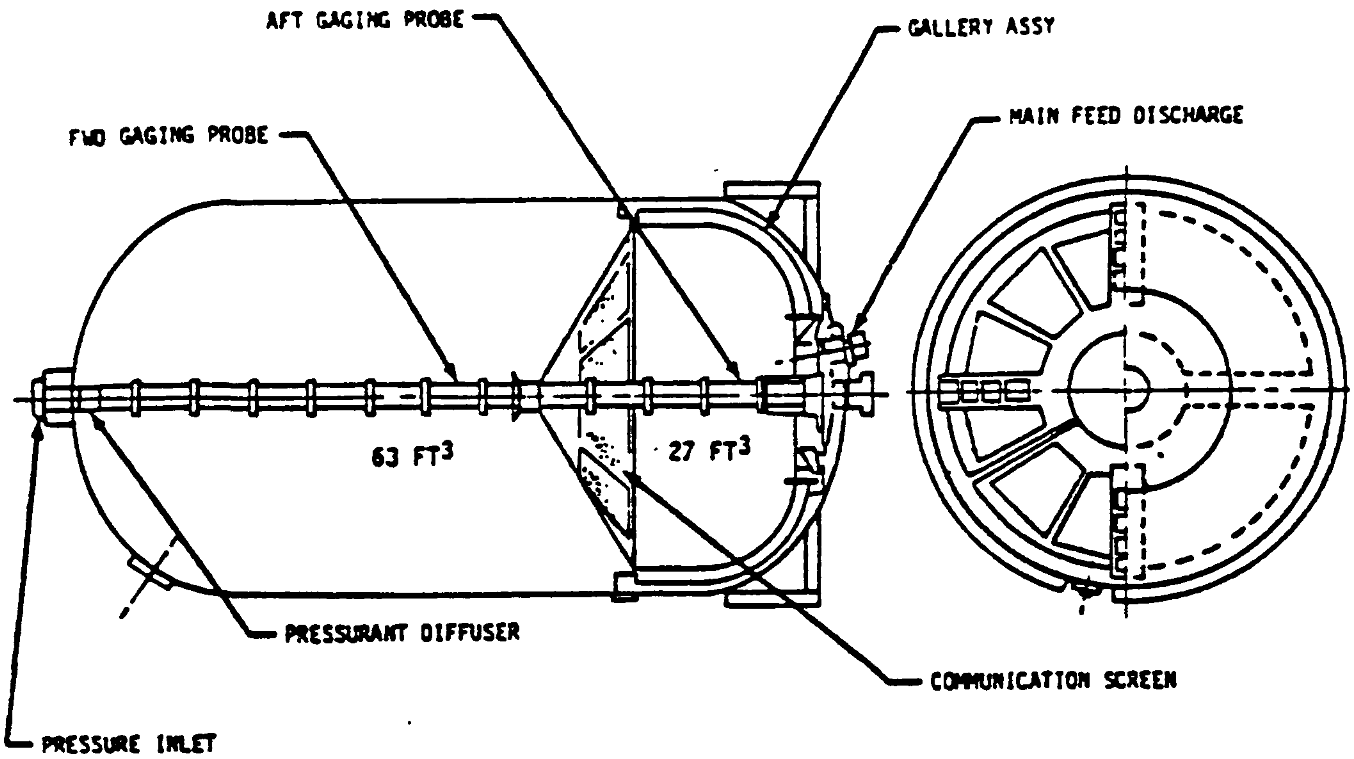

Figure 90. Shuttle OMS Tank Screen Configuration

could allow the tank to be vented during propellant loading. Although there is no known tank of this type available now, it may be a rather simple modification to existing tanks. This should be thoroughly investigated. If a screen tank can be made with this capability it would have many advantages over a vane device tank.

A matrix of resupply mission scenarios and propellant acquisitions devices are arranged in Table 37. The worst case resupply scenario (requiring ullage positioning and operational capability under varying acceleration levels), dictates that a propellant acquisition device integrating a vane arrangement and a capillary screen system would be necessary.

3.2.12 Propellant Quantity Gauging and System Health Monitoring

Tank quantity gauging and flowmeters are the two principal means of determining the quantity of propellant transferred during refueling. Although tank gaging does not directly measure the transferred amount (but infers it from tank quantity), it has the advantage of directly verifying total tank load, which is important for mission confidence. A flowmeter, by itself, cannot do this unless tank load is accurately known prior to transfer. In the case of spacecraft returning from its mission, this is not known unless a tank gauging system is provided, or the spacecraft tanks are off-loaded to zero quantity before transfer.

Positive displacement integrating fluid meters similar to commercial water meters or gas meters are not considered acceptable due to their close-tolerance moving parts and a general lack of development for withstanding launch vibrational environments. Turbine meters have been developed for rocket engine application and would be acceptable in terms of cost, accuracy and reliability. Consideration should also be given to the solid state type meters such as vortex-shedding flowmeters and swirl meters. The former consists simply of a stationary rod or bar placed across the diameter of the flow stream such that trailing edge (Von Karman) vortices are generated in the downstreams flow which can be detected by presssure sensors in the wall of the duct. The quantity of flow per vortex is repeatable over a 15:1 range of flowrate. Similar performance is provided by the swirlmeter (Fischer & Porter) design in which an axial vortex is generated by stationary radial vanes and caused to precess downstream by a sudden enlargement of the flow duct. These vortex precessions are sensed by a pressure pickup and electronically counted as an indication of total flow.

Spaceborn system accuracies realistically achievable by the above flowmeter types are +0.1% positive displacement meters, + 0.50% for small turbine meters and +.75% for the shedding-vortex and swirlmeter types. These values are stated as a percentage of volumetric flow rate over a 10:1 flow range; with the benefit of calibration vs. flowrate, temperature and pressure.

The principal zero "g" tank gaging methods are listed as follows:

- <u>PVT (Pressure, Volume, Temperature)</u> - Ullage volume is inferred from the perfect gas law, PV=MRT, where the mass of gas (M) and the gas constant (R) are known, and the gas temperature (T) and pressure (P) are directly measured. Propellant quantity is obtained by subtracting the inferred ullage volume from the total tank volume.

- <u>RF (radio frequency)</u> - An antenna mounted in the tank is used to determine the number of resonant RF frequencies inside the tank enclosure as an index of propellant mass (Bendix).

- <u>Acoustic Resonance</u> - An acoustic transponder is used to determine the fundamental resonant frequency of the ullage bubble as an indicator of ullage volume and, hence, propellant quantity. For accurate results, this ullage bubble should be positioned at the transponder by means of a diaphragm, capillary device, or a slight settling thrust.

- <u>Nuclear Gaging</u> - This technique uses a number of radioactive sources (typically gamma ray) mounted on the outside of the tank, and infers the mass of propellant in the tank by the amount of radiation absorbed.

- <u>Capacitance Probe Gaging</u> - The Shuttle OMS tanks use full length capacitance electrodes to sense the level of propellant. This is not a true zero "g" system since appreciable settling thrust, is required to prevent propellant from filling the narrow space between the electrodes by capillary action, and giving a false "full" signal. Furnishing this amount of settling force, by either engine thrust or vehicle spinning, may not be acceptable.

Table 37. Resupply Module Design Reference Missions Scenarios Vs. PMD

		Prop Acq Devices	
		Screen	Vanes
1.	ERM serves as an OMV extended mission kit to get the OMV to GEO with sufficient propellant to do 5 or 6 resupplies of GEO satellites (propellant resupply only). - The OMV will provide the resupply	req'd	req'd
2.	ERM serves as an OMV extended mission kit to place the OMV + Servicer + Helium Servicer into GEO to supply and service 1 GEO platform to extend its life by 5 years. This includes helium servicing. - The OMV will provide the resupply	req'd	req'd
3.	The ERM with the OMV as the propulsive stage provides top-off to an off-loaded, launched integral propulsion satellite requiring 30K lbs of propellant. The ERM + OMV take the propellant from a propellant depot at the Space Station and resupply the integrated propulsion system satellite.		req'd
4.	The ERM serves as the propellant depot attached to the Space Station. It accepts propellant from the orbiter by orbiter bi-propellant scavenging. It refuels other ERM's or OMV's or both.		req'd
5.	The ERM with the OMV as the propulsive stage refuels a GEO kick stage.		req'd
6.	The ERM + OMV load another ERM + OMV (which will do scenario 1) which have been launched off-loaded in LEO.		req'd
7.	The ERM + OMV load or top-off another ERM + OMV (which will do Scenario 2) which have been launched off-loaded in LEO.		req'd
8.	The ERM + OMV serve as a refuelable/reusable periges kick stage. The ERM + OMV are loaded in LEO and provide the kick for GEO, then use aerobrake to return to LEO to be refueled to do another GEO kick.		req'd

Of these techniques, PVT is the only zero "g" method wich has reached an acceptable state of hardware development and flight experience. (Apollo, GPS, Viking and Orbiter, etc.). In addition, the RF and nuclear systems are quite complex, with the latter requiring special care in the handling of radioactive materials.

The tank gaging achievable by PVT methods can be as good as 0.5% of full load, but more typically is around 3.5-5%, depending chiefly on dynamic temperature sensing errors and the accuracy of tank calibration. The principal source of error is the practical difficulty of sensing the true bulk temperature of the ullage gas under transient thermal conditions, which result from changing ambient conditions, or from isentropic compression or expansion of the ullage.

Another source of error in the PVT Method is solubility of the pressurizing gas in propellants, and its gradual loss through the engines. A nominal correction can be made for this effect in the PVT calculations.

The PVT system developed in the early 1970's and currently in use gaging the Shuttle Orbiter RCS propellant quantities has a worst case analytical gaging accuracy of 5.4%. This system includes two pressure transducers on both he pressurant tank and the propellant tanks with repeatability accuracy of 1.4% (+0.5% tansducer, +0.6% noise and impedence, +1.00% signal conditioner and +0.5% multiplexer demultiplexer encoding accuracy) and one temperature probe in the pressurant tank and one temperature sensor on the propellant tank, each with 1.5% repeatability accuracy (+0.75% transducer). Shuttle Orbiter flight experience has shown this systems actual gaging accuracy to be approximately 3.5%. This is a relatively inexpensive system using off-the-shelf transducers. Better accuracy can be gained although the price will begin to increase rapidly for the more accurate transducer. Approximately a 1% increase in accuracy can be gained without much increase in cost if the temperature sensor on the propellant tank skin is replaced with a temperature probe into the tank. This is not possible with diaphragm tanks and probably not desired with tanks with ullage positioning capability. Better accuracy can also be gained by adding additional redundant pressure and temperature transducers. It is believed, however, that a system similar to the Shuttle Orbiters RCS will be acceptable for resupply module applications. It will be necessary, however, to add redundant temperature transducers on the pressurant and propellant taks.

To measure propellant transferred quantities, a redundant integrating type flowmeter system should be incorporated into the propellant transfer lines. A turbine type of flowmeter system will probably be the least expensive in terms of overall cost and can provide good reliability, accuracy, flow rangeability, and the ability to indicate reverse flow in the event of off-loading the receiver tanks. The requirement to off-load could occur as a result of inadvertent overfilling of the receiver tanks or as a prelude to refueling the receiver tanks. The guard against flowmeter failure or disagreement between flowmeters, and since PVT may not be available as backup, it is desirable to include three flowmeters in series. The three can be averaged or a 2 out of 3 voting network can be utilized to determine actual propellant quantities transferred. The overall loading accuracy of this system is estimated to be approximately 1%. It will be necessary to place controlling valves and orifices downstream of the flowmeters to prevent propellant outgassing near the flowmeters which will significantly affect their accuracy.

System Health Monitoring system health status should be computer monitored during all phases of the ERM systems orbital operation. Pressures and temperatures of components, pressurant and propellant are verified to be within operational limits. Corrective action for environmental control is also available through system heaters. All temperature and pressure sensors incorporated in the resupply system must be operable during the general health monitoring mode.

During resupply operations, critical temperature and pressure measurements will be programmed by the computer for fluids and components of the resupply system to assure the resupply operation selected will not cause system opration outside established tolerance limits.

Fault isolation requires pressure and temperature sensors, placed at appropriate resupply system locations, to be operative. These may be the same sensors (or redundant to) used to monitor system health. These sensors have been positioned on (in) pressurant tanks, propellant tanks, upstream and downstream of pressurant and propellant isolation valves and on (in) all transfer quick disconnects. These sensors provide the capability to detect leakage, fault isolation and safe system operation through fault isolation. Using state-of-the-art microprocessors to integrate these pressure and temperature parameters, the system status is processed on-board. The status presents system health, fault isolation, and safe system configuration after fault isolation.

A computer provides control to the transfer of pressurants and propellants during resupply, assuring system limits are not violated. Computer control of the systems health and safety configuration is presently achievable as is the capability of the on-board computer to work in harmony with the computer controlling the vehicle being resupplied.

In summary, the PVT method was the only state-of-the-art zero "g" tank quantity gaging system identified. Also, an integrating flowmeter was the only method available to measure propellant transferred if a quantity gaging system was inoperative (i.e., during an ullage transfer type of resupply). A computer controlled Health Monitoring system should also be incorporated into the ERM to assess the system performance at all times.

3.2.13 Quick Disconnect Selection

The difference between the pressure balanced versus non pressure balanced fluid disconnects lies in the interface panel separation loads. Quick disconnects coupling designs utilizing pressure balanced fluid disconnects yield low interface panel separation loads. Therefore, latches may not be reqired and the end effector-grapple fixture mechanism may provide all of the connecting force requirements. For fluid disconnects that are non or partially pressure balanced the flow rates and pressures encountered by the disconnects, during fluid transfer operations, may produce sufficient forces to require latching mechanisms to secure the interface panels.

Representative propellant disconnect couplines are shown in Figures 91, 92, 93, 94, and 95. Figure 91 is a partial pressure pressure balanced configuration. This coupling has been designed with the entrance and exit fluid force to be almost counter-balanced. The design shown is a NASA-sponsored Fairchild design which features low connection forces and very low dribble volume upon disconnection. Figure 92 represents a design of a currently existing pressure balanced design by Purolator and has essentially zero pressure connection forces. The dribble volume may be greater than the NASA design but is expected to be low. In both designs figures 91 and 92, venting the front face of the male poppet to vacuum or local ambient is reuired. Figure 93 presents a conventional quick disconnect doupling with high connection forces and potentially excessive dribble volume. Figures 94 and 95 present existing quick disconnect couplings with high connection forces and small dribble volumes.

NASA-JSC is currently in the procurement cycle to develop a quick disconnect for manual and remote actuation operation. The coupling design concept, as specified by the NASA RFP, should exhibit versatility of use by a variety of different spacecraft and fluids. In addition, the coupling design should be mechanically simple, highly reliable, light weight, easy to engage and disengage, and readily adaptable to automatic operation. The quick disconnect which will be selected for the resupply module will be a derivation of this NASA design and development contract.

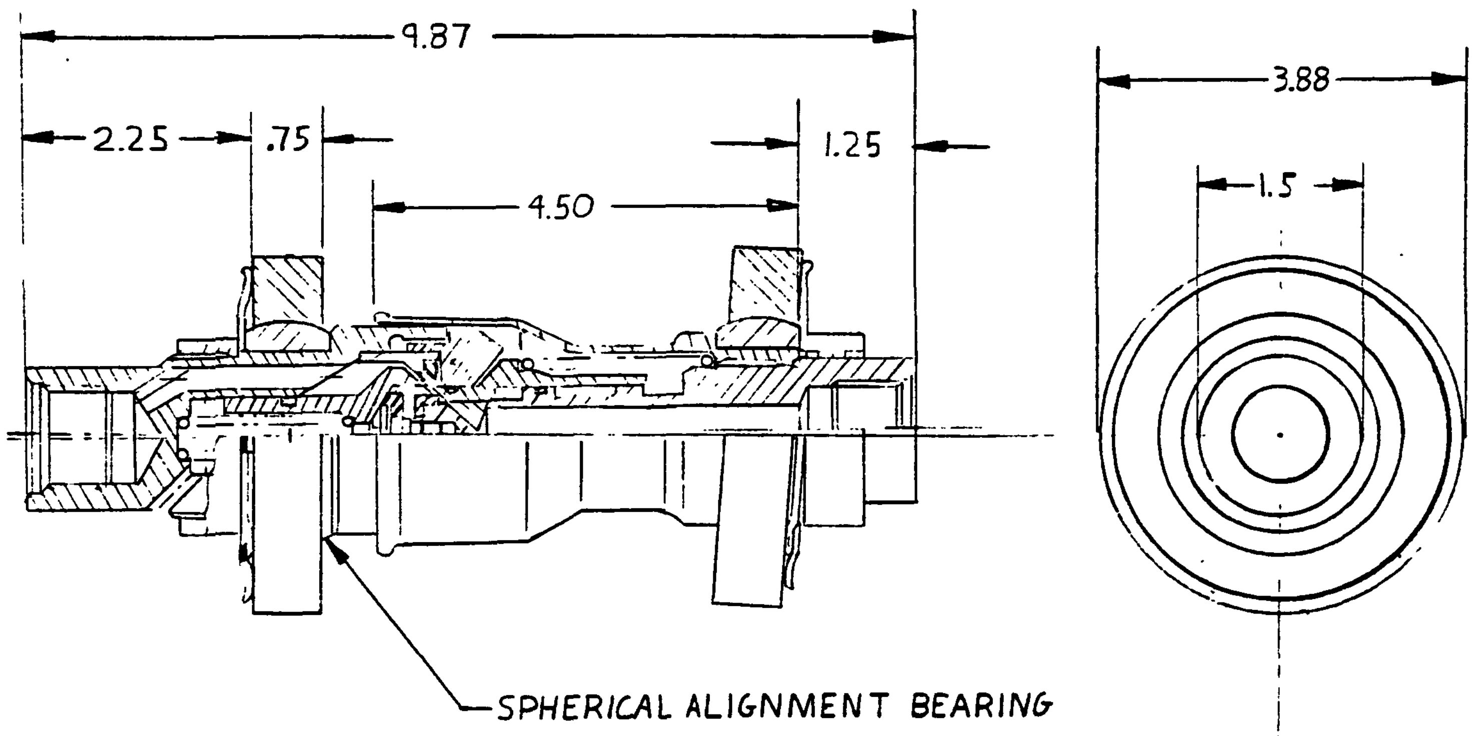

Figure 91. Partial Pressure Balanced Coupling

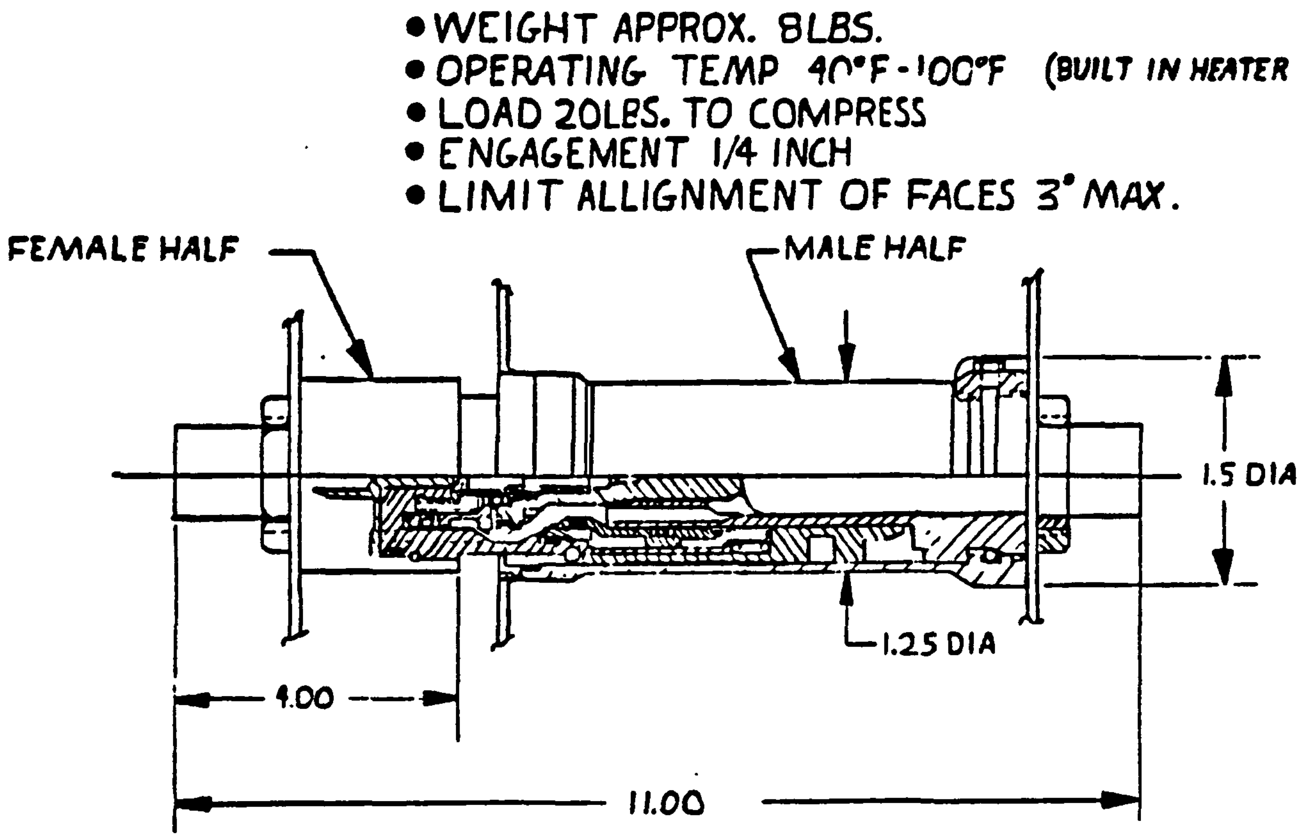

Figure 92. 1/2 Inch Pressure Balanced Coupling
(Balanced Spring Lockup and Release)

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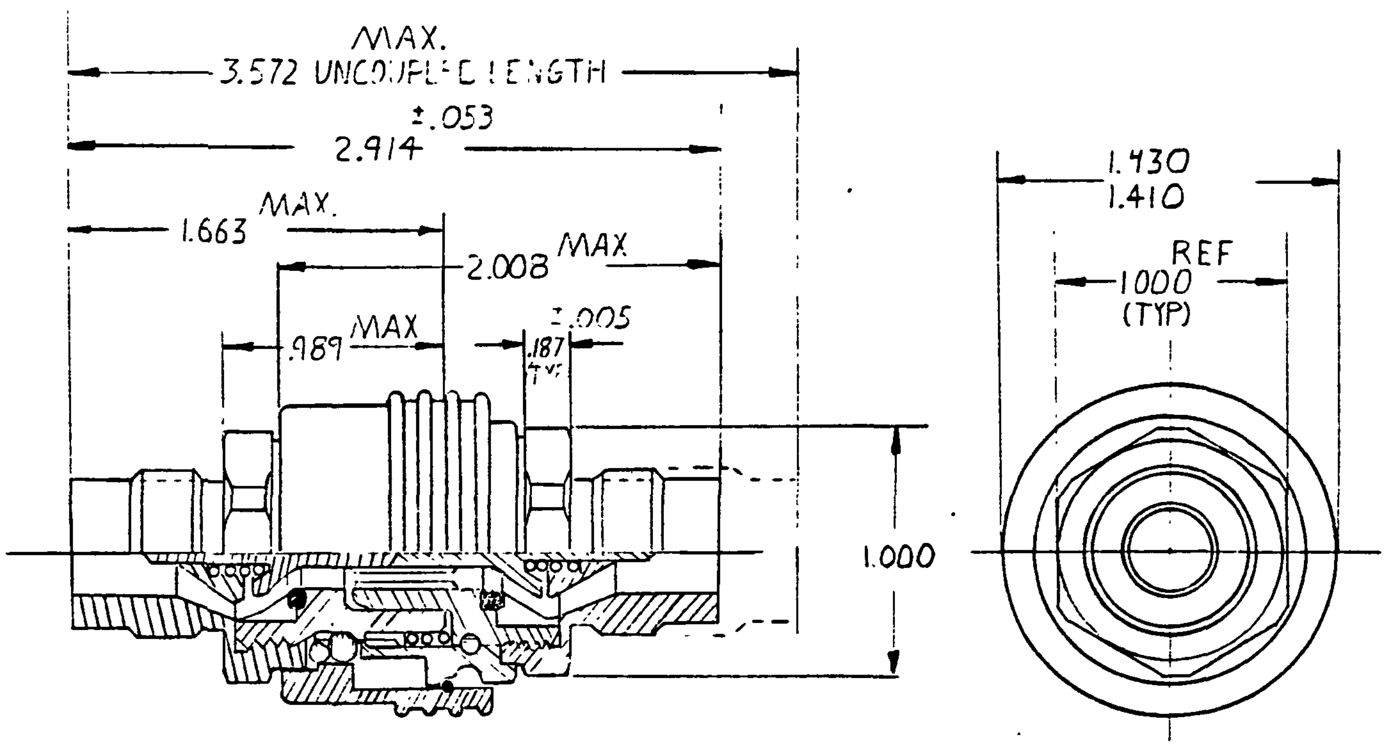

Figure 93. Conventional Q/D Coupling

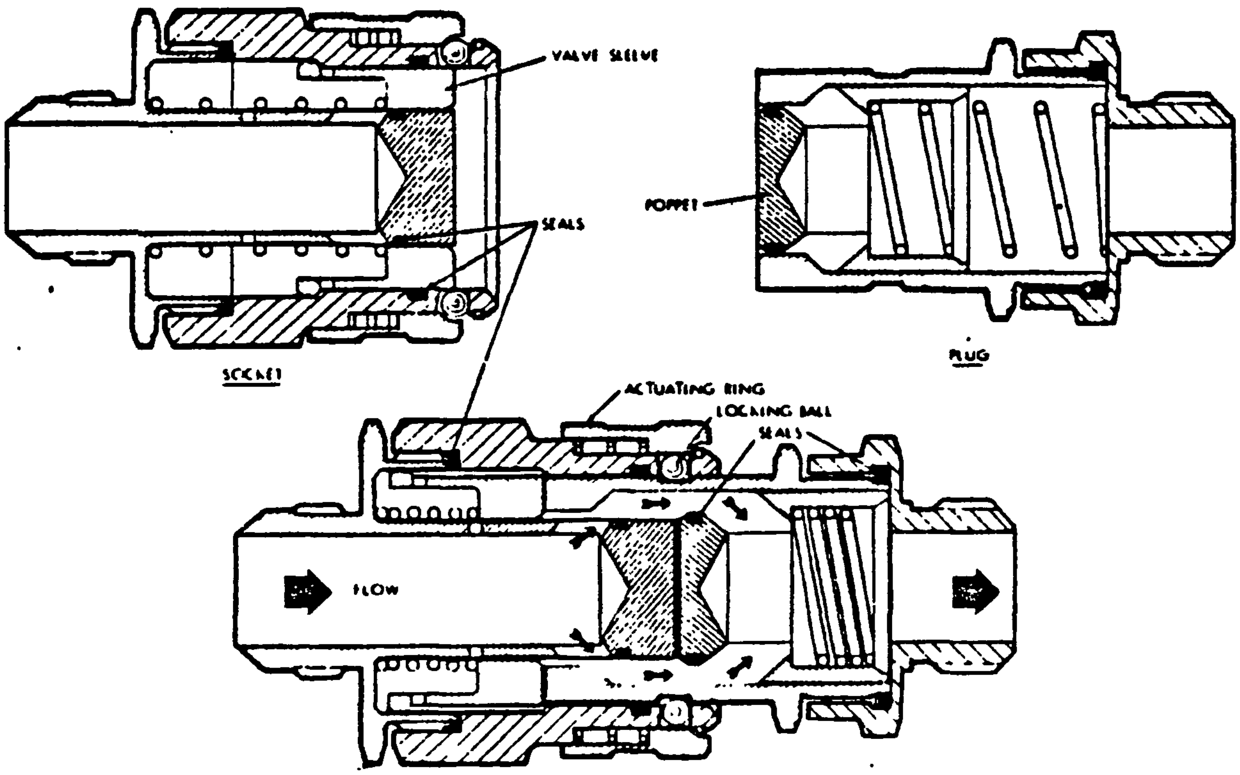

Figure 94. Tube Sleeve and Poppet Quick Disconnect Coupling

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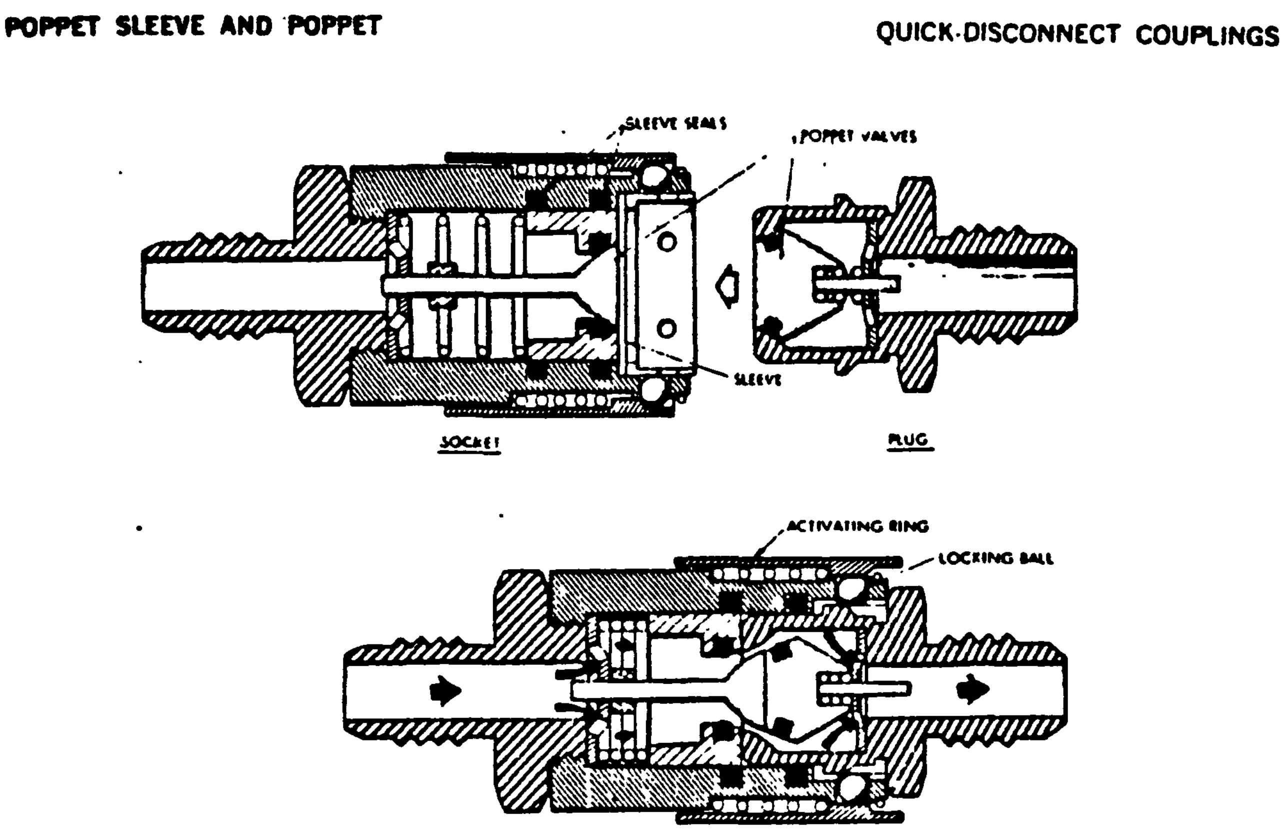

Figure 95. Poppet Sleeve and Poppet Quick-Disconnect Coupling

3.2.14 Avionics and Power Subsystems Definition

The ERM is baselined to operate with a smart host vehicle. Where the OMV acts as the host vehicle an extended capability OMV is baselined. This OMV is expected provide the following functions which drive ERM system requirements:

- o GN&C
- o Communication and tracking
- o Vehicle docking
- o Primary power supply (quiescent mode)
- o Primary data processing

These interfaces are derived from the ERM mission scenarios. The prime mission scenario driving these interface requirements is the ERM operating as an OMV extended mission kit enabling the OMV to be placed in GEO where, the OMV alone would provide the resupply and/or servicing operations. The second major scenario is the ERM acting as the Space Station propellant depot accepting Orbiter scavenged propellant.

GUIDANCE, NAVIGATIONS CONTROL

The GN&C activities are primarily an OMV function. This is due to the remote servicing/resupply operation requiring the OMV as the carrier vehicle.

DOCKING

The docking sybsystem is also provided by the OMV although some ERM interfaces will be required. The present configuration mounts the ERM to the forward structure of the OMV as shown in Figure 4. This requires the docking mechanism, an RMS end effector, to be mounted on the ERM to facilitate the umbilical connections. The video and lighting which are part of the OMV docking subsystem would be occluded in the present OMV configuration. Therefore some mounting hardware and signal links are required on the ERM to use the OMV docking subsystem.

COMMUNICATIONS & TRACKING

Communications and Tracking subsystem is provided by the OMV and provides compatibility with the TDRSS, Space Station, GSTDN and the STS.

POWER

The power subsystem design is driven by the electrical load profile. Two operational phases of the ERM mission include; a quiescent mode of status monitoring and propellant storage thermal conditioning, and the hi-operating mode during fluid transfer activities. During the 7 day ERM mission, 90% of the time spent is in the quiescent power mode.

During quiescent mode the ERM power source will be the OMV power system. The ERM thermal control system both active and passive will be designed to function within the OMV power subsystem capability.

During fluid transfer, the high power requirements must be provided by a dedicated ERM power supply. This power supply should be rechargeable batteries to provide multi-mission capability and on-orbit recharging. The power distribution sybsystem shall provide switching between both internal and external power generation. It will also provide for energy storage management to recharge the ERM batteries between missions. The batteries are sized to provide all the power requirements of the transfer mode which include the umbilical mating, system checkout, resupply operations, system purge, post-activity checkout, and demating activities. The electrical power distribution system as shown in Figure 96.

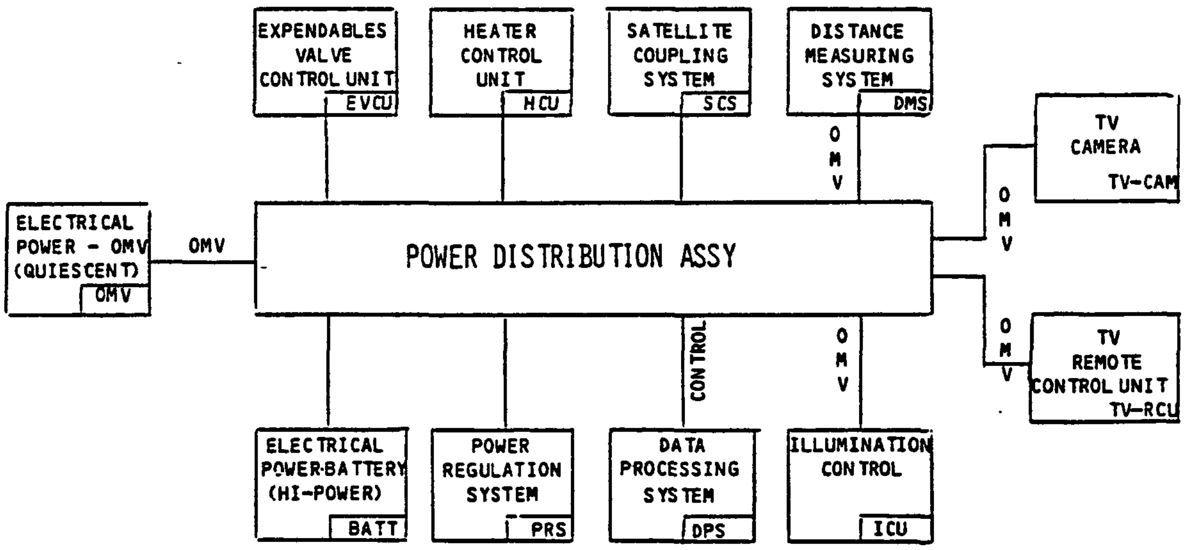

Figure 96. Electrical Power Distribution

Data Processing Subsystem

The data management function is performed by the Data Processing System (DPS) controlling all events and sequencing all operations on-board the ERM. The ERM DPS provides for all software necessary to perform dedicated ERM functions and is connected to the OMV's DPS via a 2-way data bus. This allows for maximum utilization of OMV software and processors for communications (uplink & downline) and rendezvous/docking operations.

The DPS architecture is modular in nature with a distributed, standardized processor arrangement integrated by a power and databus oriented structure shown in Figure 97. The DPS operates in 3-modes: as the executive DPS controlling all dedicated ERM functions; as a subsidiary DPS providing the OMV DPS with all critical information and as a relay for all man-in the loop control data uplinked via the TDRSS and DOMSAT systems.

The DPS provides for the following functions:

- o System checkout
- o Sequencing events
- o Automated malfunction procedures
- o purging & venting operations
- o power distribution and control
- o System monitoring
- o Fault detection & annunciation
- o Expendables resupply control
- o Environmental control
- o Satellite receiver system checkout

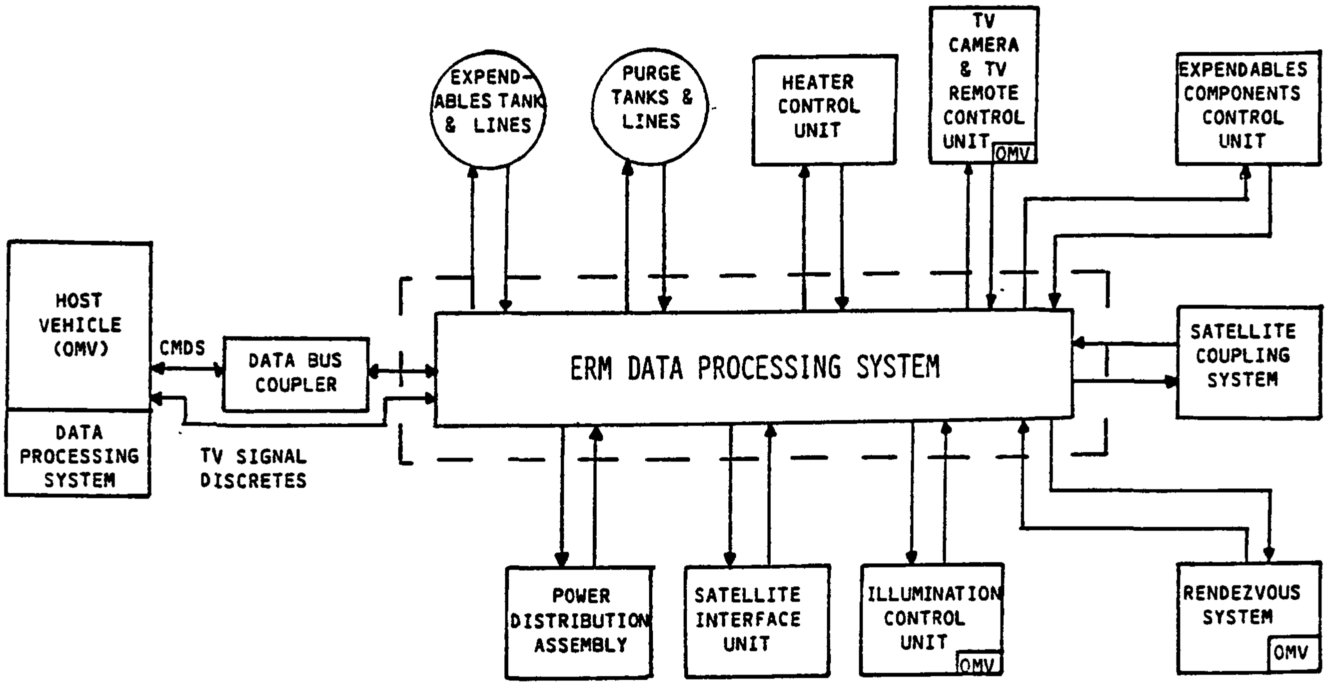

Figure 97. Avionics/Data Processing System

ERM/CARRIER VEHICLE AVIONICS INTERFACE

The ERM Avionics will be designed such that when it is employed operationally, subsequent to the demonstration experiment, minimum changes will be required. An exception will be the additional software required with the DPS to interface with the Orbiter for the demonstration phase.

The design of the Avionics system maximizes available carrier vehicle capability while maintaining key ERM functional autonomy. The extended OMV Avionics will provide the communications and tracking system and the rendezvous and docking system.

To mimimize ERM cost and weight the OMV power system will provide necessary system maintenance power during quiescent mode operations. The electrical interface between the ERM Avionics and the Orbiter (demonstration program) will be identical to that between the ERM Avionics and the carrier vehicle (operational program). The DPS on-board the ERM will provide software for all dedicated ERM functions and take advantage of the OMV's DPS where desirable via a two-way data bus. The OMV's DPS will provide for master data processing for the ERM acting as an extended kit to the OMV.

Figure 98 shows the ERM/Carrier vehicle avionics for the operational and demonstration Expendables Rsupply Module.

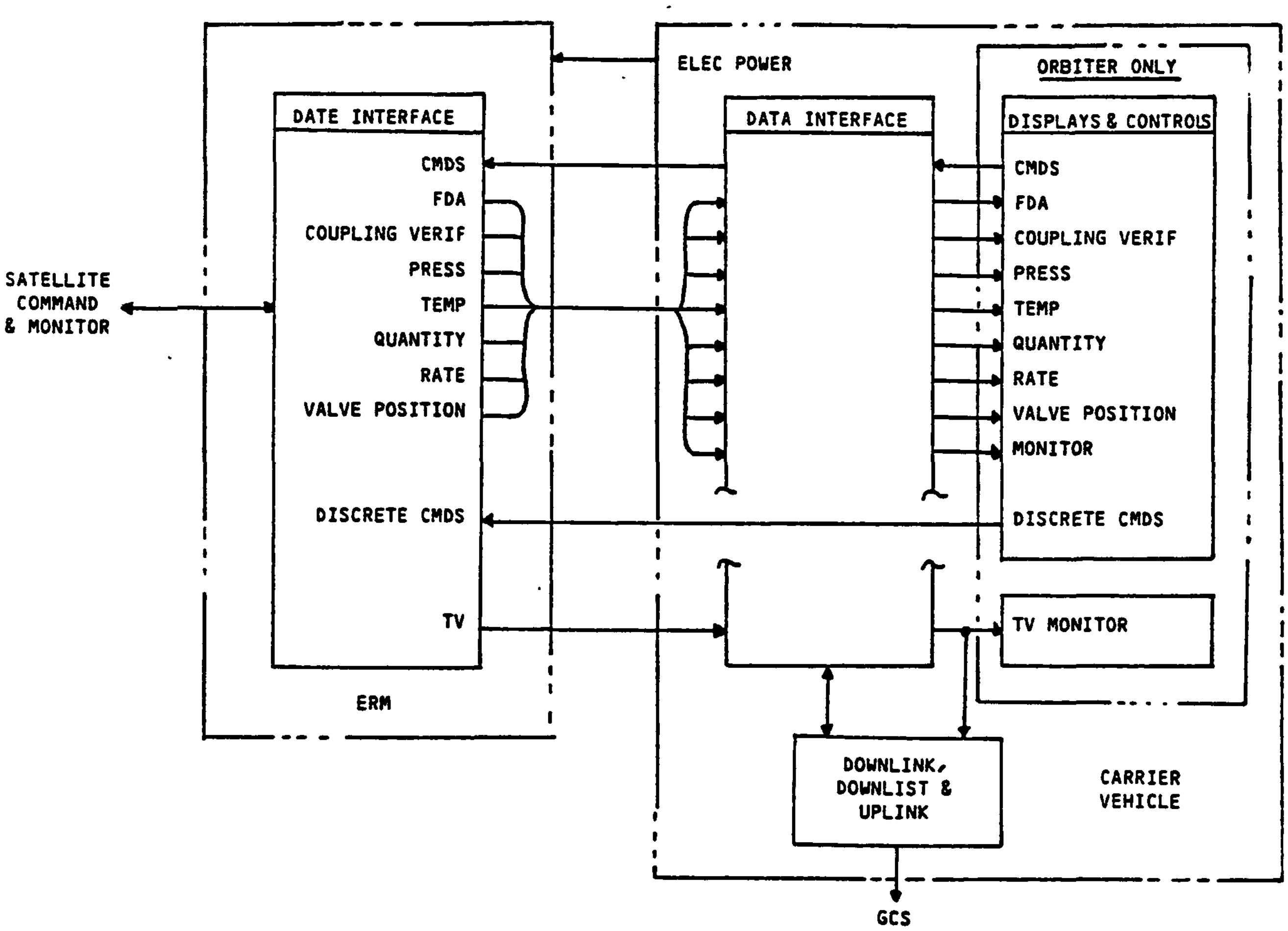

Figure 98. ERM/Carrier Vehicle Avionics Interface

Result of the Expendables Resupply Module power requirements analysis indicates that five (5) Li/TiS_2 batteries having a total capability of 43.4 kwh and weighing approximately 675 lbs. would provide power adequate for the helium compressor and the bipropellant pump for the mission duty cycle shown in Figure 99.

The mission selected represents a full STS duration OMV out-and-back excursion. The hosekeeping power level was determined to be 815 watts, based on the identified subsystem load requirements, as shown in Table 38. The power required for the heaters is high and would drive the battery requirements for a long 7-day mission even though proper thermal control with insulating blankets and coatings was taken into account.

Figure 99. Resupply Module Power Duty Cycle

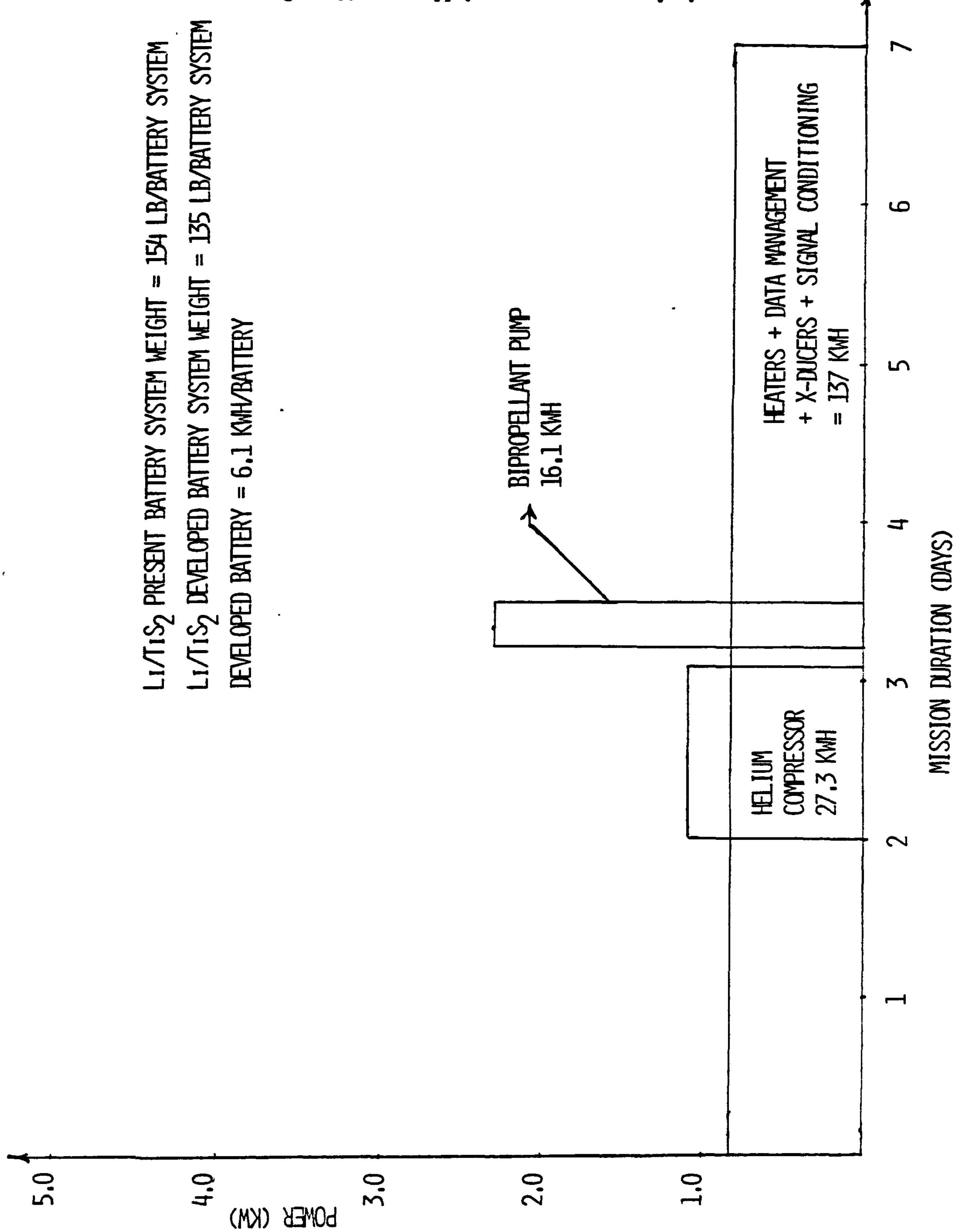

Table 38. Resupply Module Electrical Power Requirements

ERM EQUIPMENT	POWER (WATTS)	
	QUIESCENT	PEAK
DATA MANAGEMENT SYSTEM	50	75
THERMAL CONTROLS - HEATERS FOR TANKS, PUMPS, VALVES & OTHER COMPONENTS	670	2200
MECHANICAL ACTUATORS LATCHES, END EFFECTOR	-	300
FLUID TRANSFER SYSTEM	-	
PUMPS	-	2300
COMPRESSORS	-	1107
VALVES (5)	.	500
VENT/PURGE VALVES (4)	-	400
X-DUCERS/SIGNAL CONDITIONING	20	40
MARGIN	75	150
TOTAL POWER	815	7072

The heaters were sized for a side to the sun GEO mission with a 50% BBQ. With this orientation constraint, the power required was reduced to 670 watts required to hold proper thermal temperature limits versus a 1000 watt requirement with the resupply module/OMS end to the sun orientation. The heaters were sized for the worst case to the sun orientation at a 50% duty cycle or for a total of 2000 watts. With six tanks, this reduced to 350 watts required per tank.

The heater requirement of 670 watts for a 7-day mission would drive the battery sizing requirement; therefore, a power source user designation for each different mission scenario was constructed and shown in Table 39.

3.2.15 Docking Hardware Selection

Several fluid transfer umbilical plate concept designs have been generated. Each concept utilizes the standard RMS and effector (Figure 100) and grapple fixture (Figure 101) to perform the docking and final closure mating. In designs utilizing pressure balanced fluid disconnects to yield low interface separation loads, latches may not be required and the end effetor-grapple fixture mechanism may satisfy all of the connecting force requirements. It is established that the closing force realized from the end effector-grapple fixture is between 700 to 900 pounds. Fluid disconnects that are not or only partially pressure balanced would require panel latches (Figure 102) to meet the connecting force requirements.

Figure 103 represents a typical bipropellant fluid interface panel on a receiver vehicle. This panel would contain fluid disconnects for propellant transfer, electrical disconnects and a grapple fixture. The mating panel, to be used on the resupply vehicle is shown on Figure 104. It contains mating fluid and electrical disconnects and a standard end effector.

The sequence to be used for fluid interconnect would be the following:

1. Contact would be made with the end effector and grapple fixture.

2. The end effector would snare the grapple fixture and pull the satellite or payload to be refueled to the resupply vehicle.

3. The end effector would bring the two vehicles together and would engage the interface umbilical panel alignment pins.

4. The end effector would bottom out the grapple fixture plate to the end effector face properly aligned by the alignment guides or pins.

5. The latches (if used) would actuate and clamp both halves of the umbilical panel togeher.

6. The fluid disconnect would be actuated to engage to their final open positon.

Umbilical Interconnect and Latching Device

The orientation of the latches and an estimate for the actuation travel required for the fluid disconnects is shown in Figure 105. The engagement travel distance was taken from a representative fluid disconnect.

Table 39. Resupply Module Power Users Source Designation

DESIGN SCENARIO	POWER SOURCE								
	BATTERIES*			EXTENDED OMV W/SOLAR ARRAY			SPACE STATION		
	PUMP	COMP	HTRS	PUMP	COMP	HTRS	PUMP	COMP	HTRS
1	✓					✓			
2	✓					✓			
3	✓					✓			
4							✓	✓	✓
5	✓	✓				✓			
6	✓					✓			
7	✓					✓			
8	✓					✓			

*BATTERIES SIZING DESIGN CONDITION, TOTAL ENERGY REQMT = 43.4 KWH

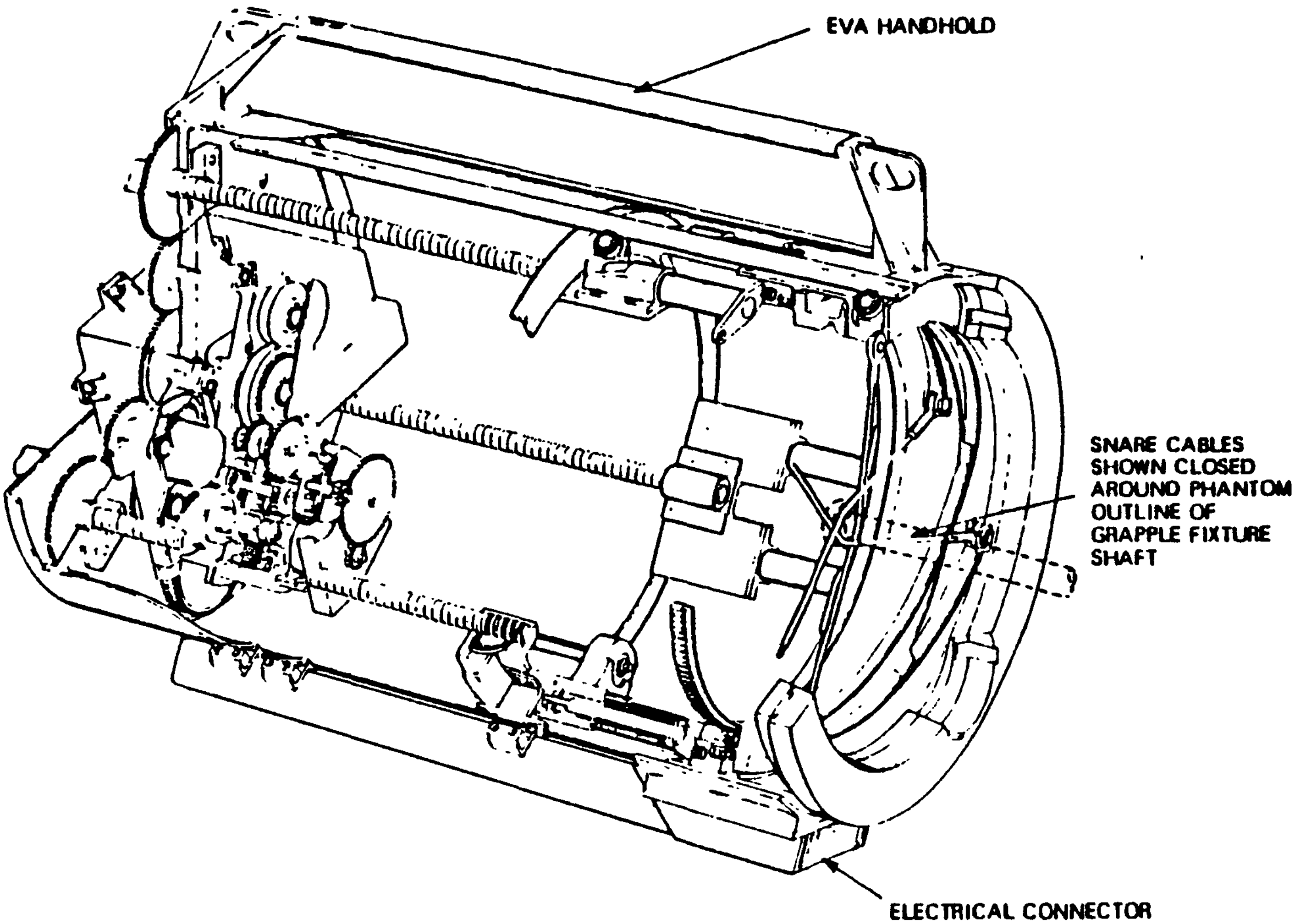

Figure 100. Standard Snare Type End Effector

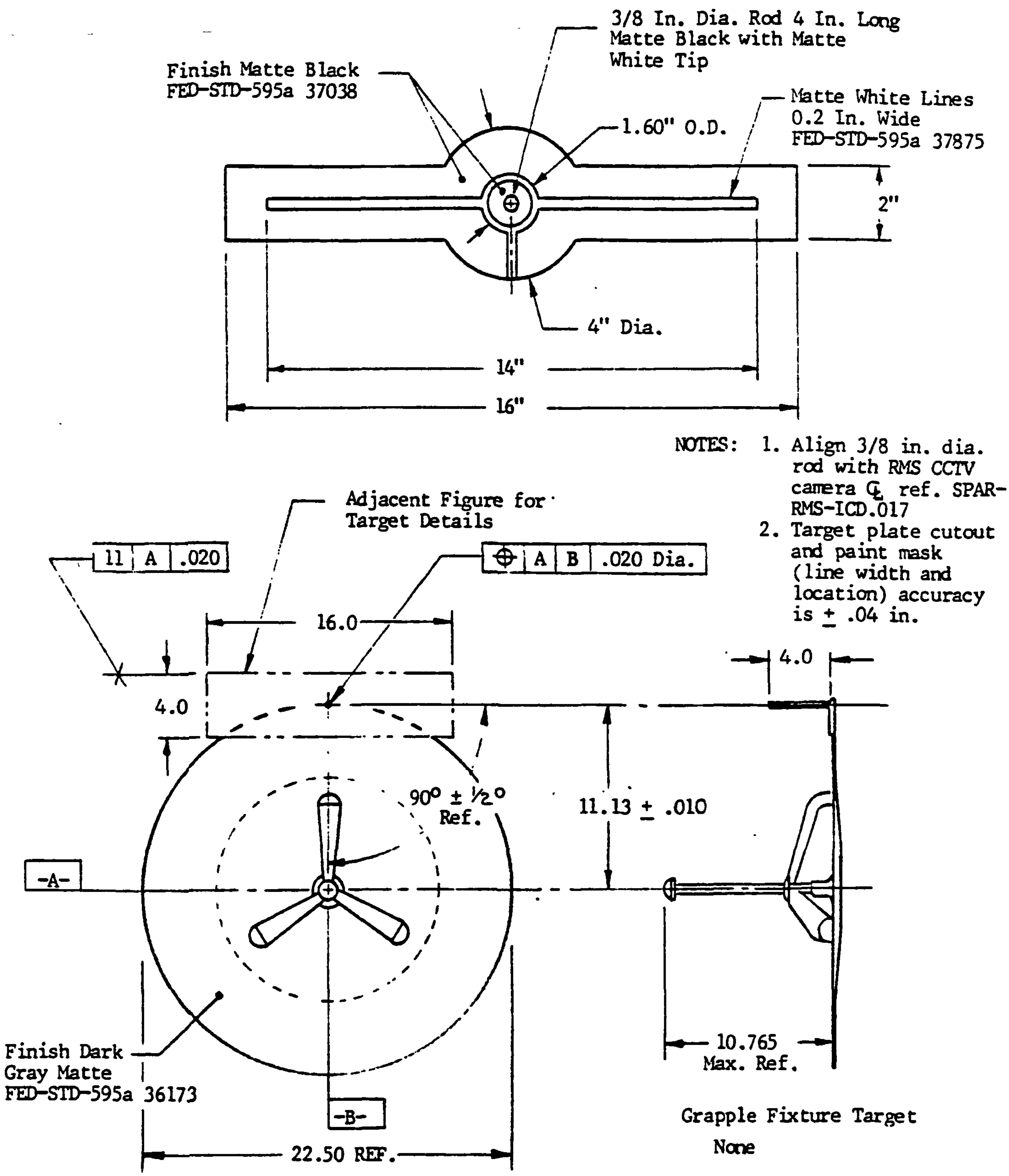

Figure 101. Grapple Fixture/Target Assembly

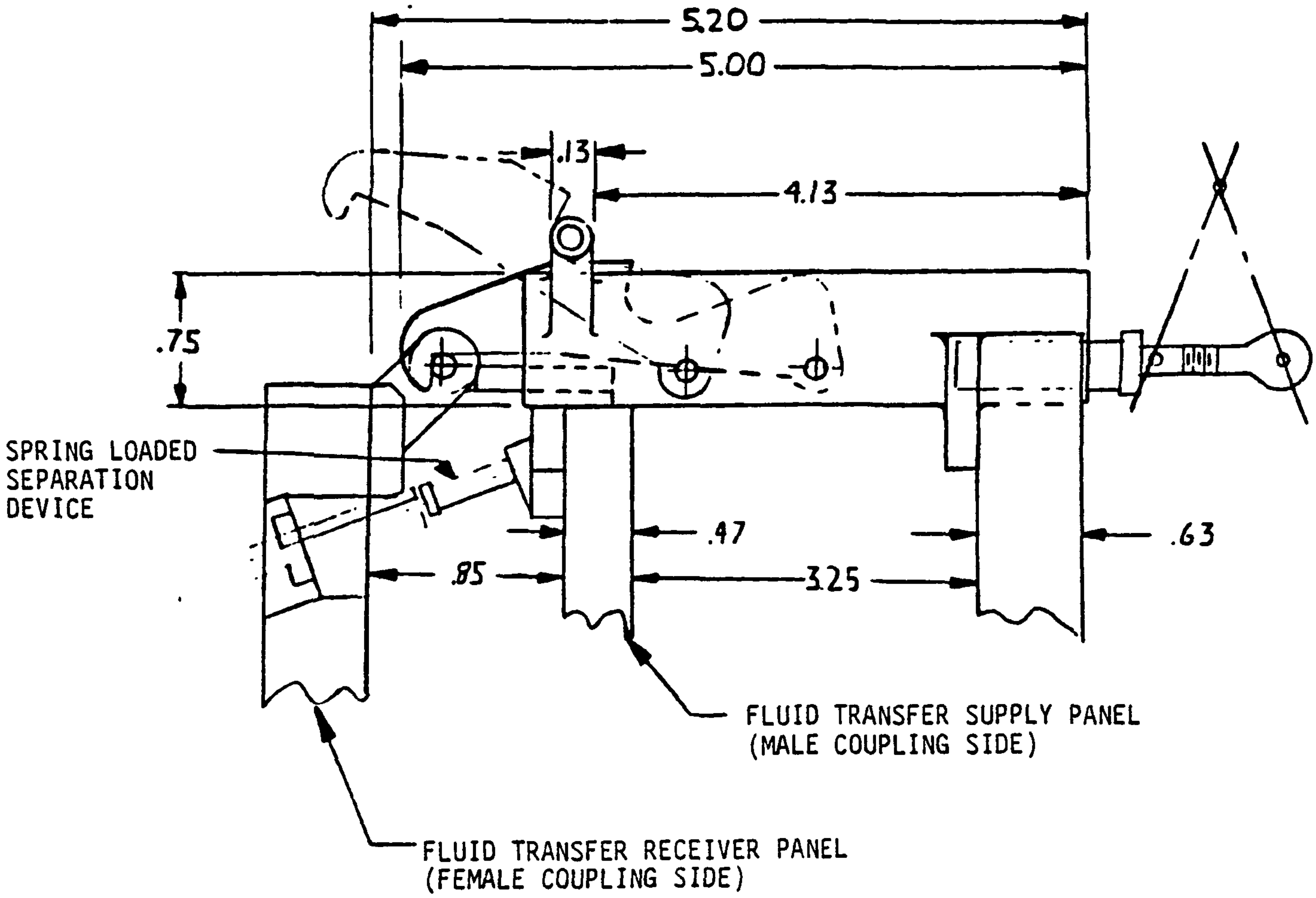

Figure 102. Fluid Interface Panel Latch

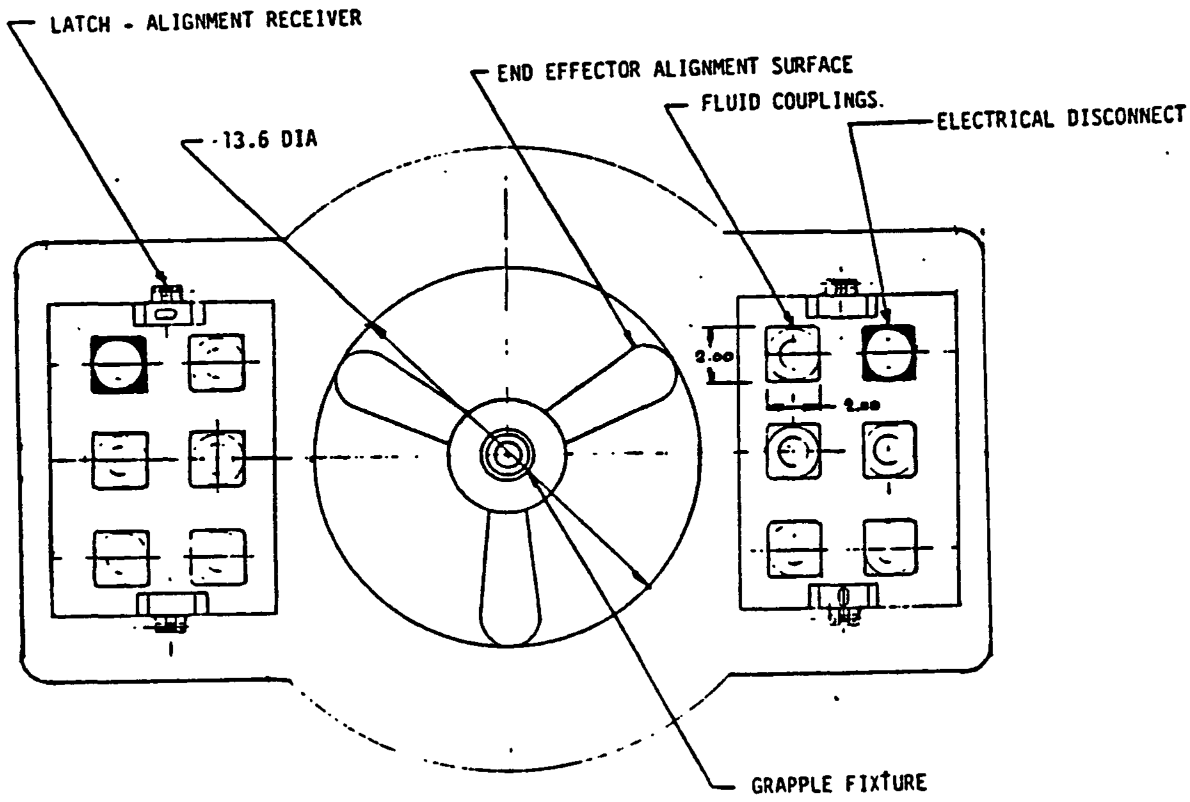

Figure 103. Receiver Vehicle
Fluid Interface Panel

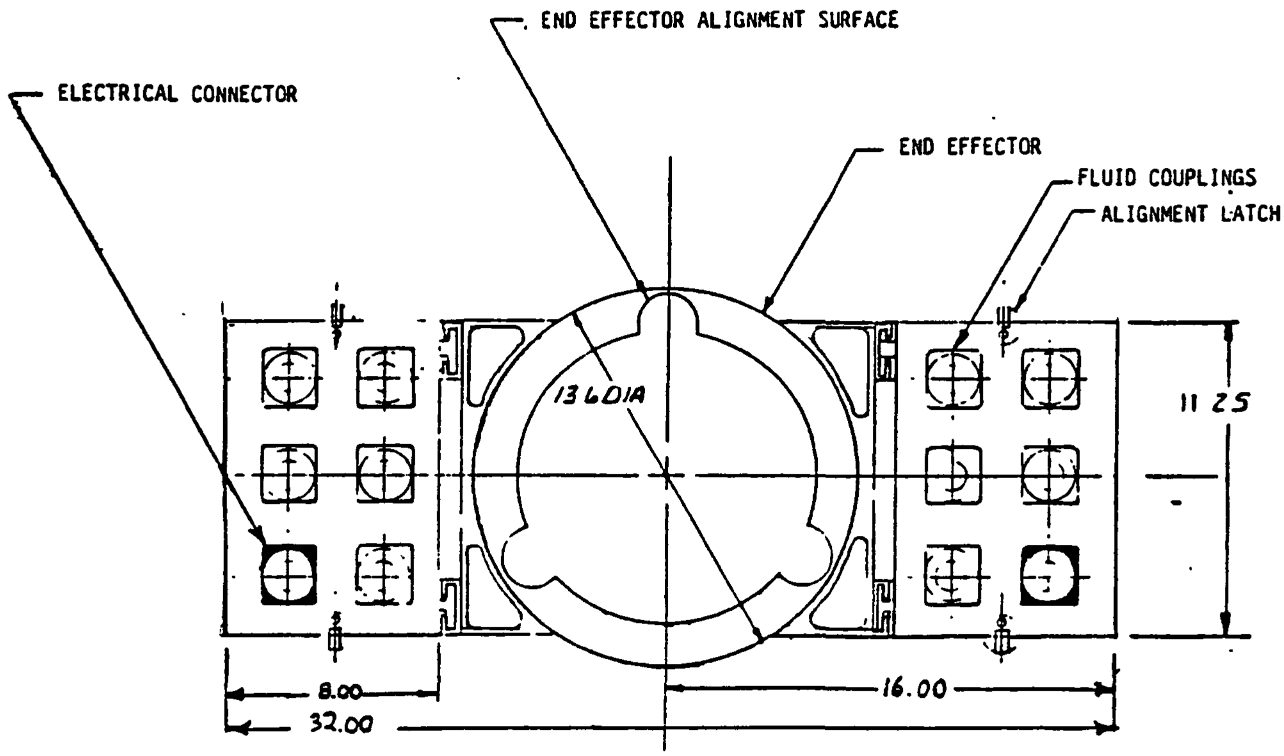

Figure 104. Resupply Vehicle
Fluid Interface Panel

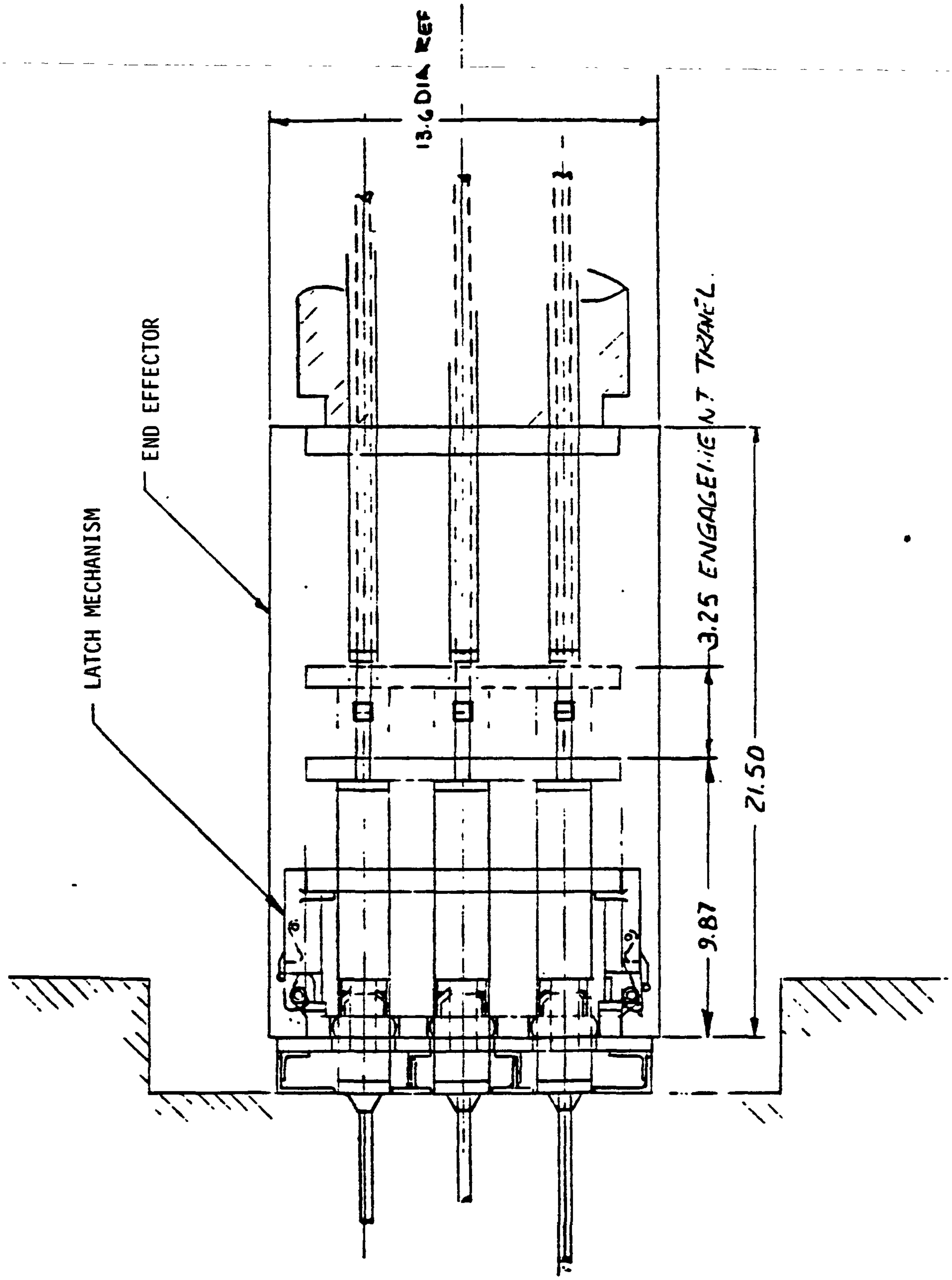

Figure 105. Side View - Fluid Interface Panel

3.2.16 ERM System Weights

Systems and component weights for the ERM are presented in this section. The dry weight of the resupply module was determined for two cases, a high and a low. (See Table 40).

These high and low weight estimates represent the bonds of a 90% confidence range. This indicates that there is less than a 10% probability of the ERM dry mass exceeding 9168 lbs or being less than 8259 lbs.

3.2.17 Receiver Spacecraft Impact Assessment

Four different types of satellite propulsion systems were analyzed as part of the spacecraft impact assessment: hydrazine blowdown, bi-propellant blowdown, pressure regulated (bi-propellant), and pump-fed (bi-propellant).

The modifications required for the on-orbit refueling of spacecaft propulsion systems were minimal. A list of these modifications are as follows:

1. All satellites were assumed to be launched and deployed by the Space Transportation Systems. Therefore, all satellite propulsion systems were modified to meet STS payload safety and contamination requirements including: a) propellant lines leading to an overboard outlet should be controlled by a minimum of three independent mechanical inhibits, and b) pressurant lines leading to an overboard outlet should be controlled by a minimum of two independent mechanical inhibits.

2. Multiple tank systems/subsystems (hydrazine systems, MMH subsystems, and NTO subsystems), require tank isolation valves for propellant loading/ullage transfer control.

3. To insure the integrity of the payload's environment and the local environment of the space station, a disconnect/line purge circuit should be incorporated into the propellant transfer systems of all space vehicles requiring on-orbit refueling operations. High pressure gas (which would be supplied by a resupply module, or any other resupply system), would flow through the purge circuit and clear all refueling disconnects and adjoining lines of residual propellant, prior to disengaging the receiver and resupply vehicles.

The component and weight impact assessment of these modifications, on the four previously mentioned satellite propulsion systems are summarized in Tables 41 through 44. Schematics of these modified propulsion systems are illustrated in figures 106 through 109.

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Table 40. ERM System/Component Weights

ORB/STS	COMPONENT	QTY	CLASS	STS COST BASE	WEIGHT-LB	
1.3.1.2	BODY STRUCTURE		-	-	(2536)	(2784)
	• FRAME & SUPTS	1	NEW	MID FUS STRUCT	1644	1808
	• FITTINGS & BRKTS	172	"	"	182	200
	• COMPONENT PANELS	20	"	"	560	616
	• INTERFACE PANELS	2	"	"	150	165
1.3.1.2	TPS		-	-	(250)	(275)
	• BLANKETS & COVERS	-	NEW	CREW MOD INSUL	250	275
1.3.7.8	SEPARATION MECHANISMS	-	-	-	(201)	(241)
	• PYRO'S	12	OLD	ORB/ET SEP	36	43
	• RMS END EFFECTORS DOCKING	1 pr	OLD	RMS	135	162
	• UMB.ACT'S	1	NEW	AFT UMB DR MECH	30	36
1.3.2.3	FLUID SYS.	-	10Xchg	-	(3721)	(3842)
	• TANKS & PMD	6	MOD	OMS TANKS	2118	2152
	LINES & • COMPONENT	140	NEW/ STS	OMS SYSTEM	386	425
	• UMB. INTERFACE	28	NEW	"	80	96
	• HELIUM TANKS	12	STS	MPS TANKS	977	977
	• PROP PUMP & COMPRSR	4 ea	NEW	NEW	160	192
1.3.3	ELEC.POWER	-	-	-	(800)	(960)
	• BATTERIES (44KW/Hr)	6	NEW	NEW	800	960
1.3.4.6	ELEC.POWER DIST & CNTL	-	-	-	(360)	(504)
	• REG.&DIST	-	MOD	EDPC	200	280
	• SEQ.&CNTLR	-	NEW	EDPC	160	224
1.3.4.3	AVIONICS	-	-	-	(291)	(407)
	• VIDEO	-	MOD	CNTLS & DISPLAY	66	92
	• INSTRUMENTATION	-	MOD	INSTRUMENTATION	100	140
	• DATA PROCESSING	-	NEW	DPS	125	175
1.3.5	ENVIRON. CNTL SYSTEM	-	-	-	(100)	(120)
	• COLD PLATE & HEATERS	-	MOD	EC/LSS	100	120
	WEIGHT-LB	-	-	-	8259	9168

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Table 41. Hydrazine Blowdown Satellite Weight Impacts (Ullage Recompression)

	Impact on Satellite Prop. Systems	Wt./Unit (lbs.)	Source of Information	# of Units Required	Total Wt. (lbs.)
1.	Hydrazine Blowdown Propulsion System				
	a. Ullage Recomp. Modifications				
	i. Interface Panel				
	1. Panel	18.8	Engineering estimate	1 (1/vehicle)	18.8
	2. Disconnects	4.0	LEM design # 4	1	4.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	1	25.0
	2. Valves	2.3	Shuttle component	2	4.6
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle Component	4 (1/tank)	9.2
	System Weight Increase				85.6
	b. Ullage Recomp. Modifications, Disconnect/Line Purge Option				
	i. Interface Panel				
	1. Panel	25.0	Engineering estimate	1 (1/vehicle)	25.0
	2. Disconnects	4.0	LEM design # 4	2	8.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	1	25.0
	2. Valves	2.3	Shuttle component	3	6.9
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle component	4 (1/tank)	9.2
	System Weight Increase				98.1

Figure 106. Impact on Hydrazine Blowdown Propulsion Systems For "Ullage Recompression" On-Orbit Resupply

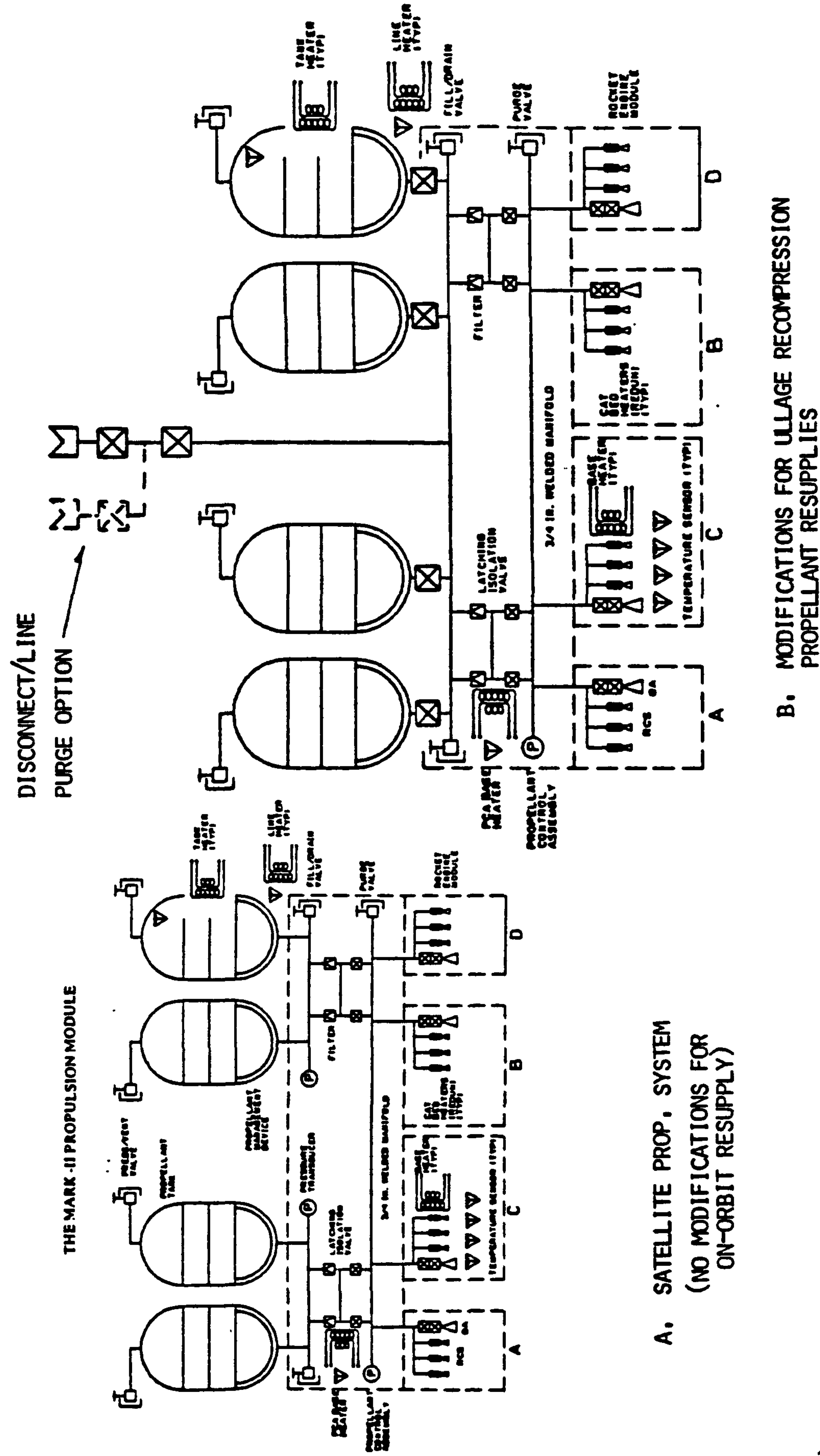


Table 42. Weight Impact On Bipropellant Blowdown Propulsion Systems for Ullage Recompression Resupply

	Impact on Satellite Prop. Systems	Wt./Unit (lbs.)	Source of Information	# of Units Required		Total Wt. (lbs.)
2.	Bi-Propellant Blowdown Propulsion System					
	a. Ullage Recomp. Modifications					
	i. Interface Panel					
	1. Panel	25.0	Engineering estimate	1	(1/vehicle)	25.0
	2. Disconnects	4.0	LEM design + 4	2	(1ox/1fu)	8.0
	3. Electrical Connections	1.0	Engineering estimate	2	(pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1	(1/vehicle)	22.0
	ii. Valve Panels					
	1. Panels	25.0	Eng. est. based on Shuttle	1		25.0
	2. Valves	2.3	Shuttle component	4	(2ox/2fu)	9.2
	iii. Other Components					
	1. Valves (Tank Isolation)	2.3	Shuttle Component	0		0.0
	System Weight Increase					91.2
	b. Ullage Recomp. Modifications, Disconnect/Line Purge Option					
	i. Interface Panel					
	1. Panel	37.5	Engineering estimate	1	(1/vehicle)	37.5
	2. Disconnects	4.0	LEM design + 4	4	(2ox/2fu)	16.0
	3. Electrical Connections	1.0	Engineering estimate	2	(pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1	(1/vehicle)	22.0
	ii. Valve Panels					
	1. Panels	25.0	Eng. est. based on Shuttle	1		25.0
	2. Valves	2.3	Shuttle component	6	(3ox/3fu)	13.8
	iii. Other Components					
	1. Valves (Tank Isolation)	2.3	Shuttle component	0		0.0
	System Weight Increase					116.3

Figure 107. Impact on Bi-Propellant Blowdown Propulsion
For Ullage Recompression On-Orbit Resupply

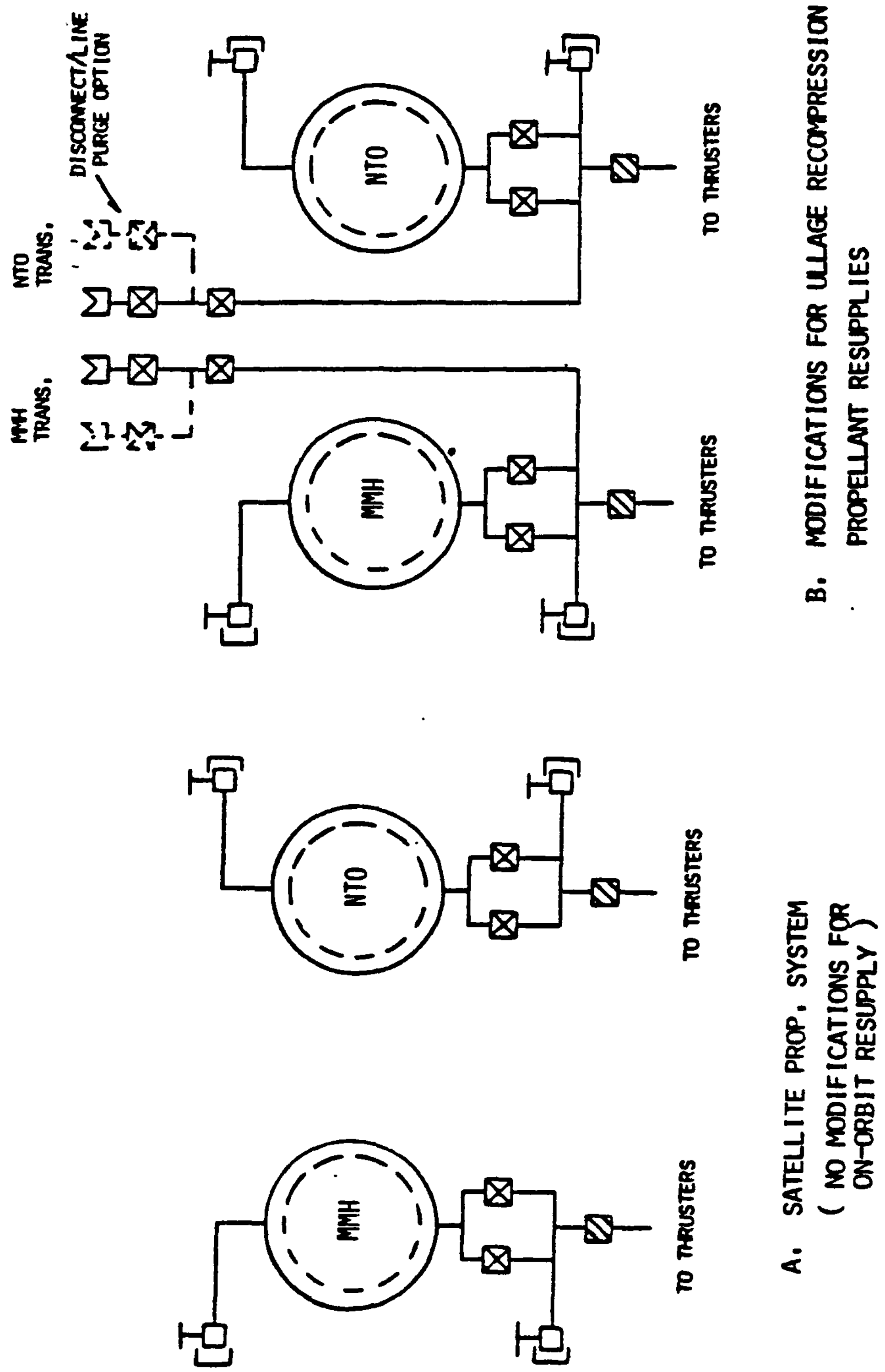
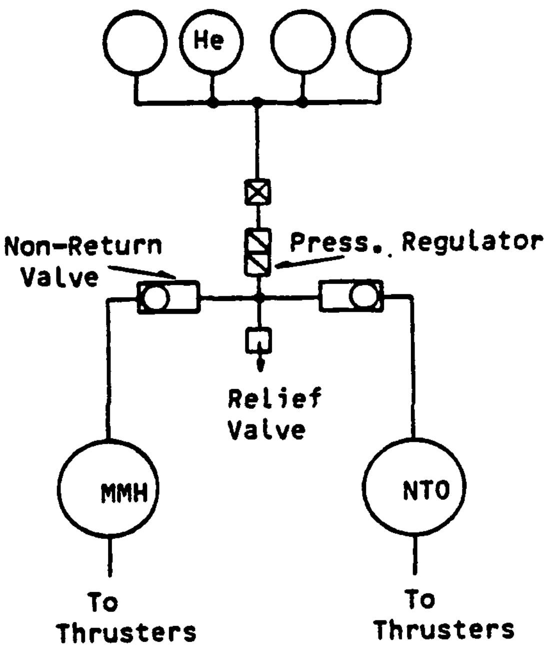
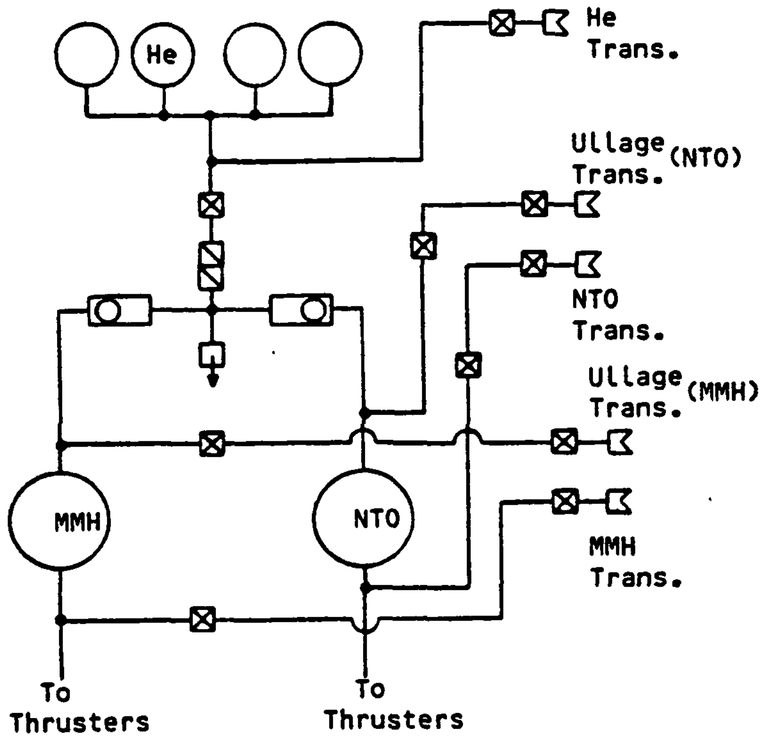


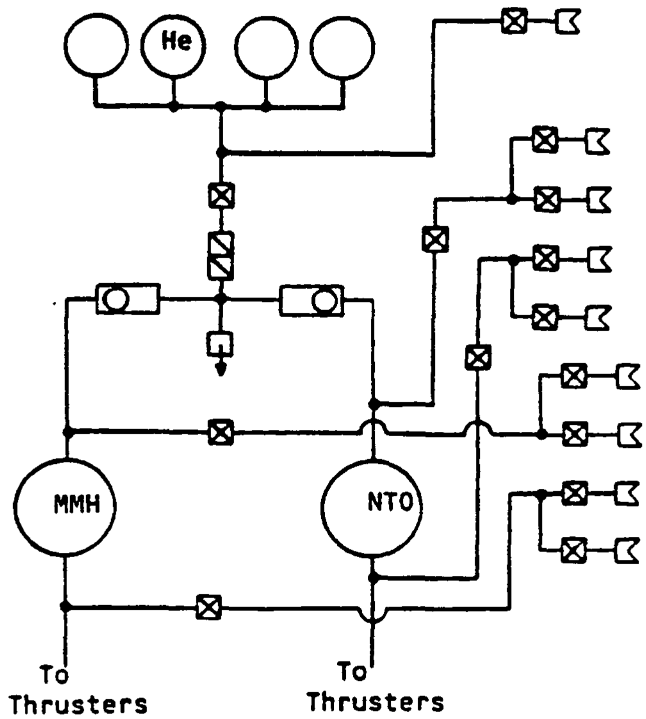
Figure 108. Impact on Pressure Regulated Bipropellant Propulsion Systems for Ullage Exchange Resupply



A. SATELLITE PROP. SYSTEM
(NO MODIFICATIONS FOR ON-ORBIT RESUPPLY)



B. MODIFICATIONS FOR ULLAGE EXCHANGE PROPELLANT RESUPPLIES



C. MODIFICATIONS FOR ULLAGE EXCHANGE PROPELLANT RESUPPLIES (DISCONNECT/LINE PURGE OPTION)

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Table 43. Weight Impact on Pressure Regulated Propulsion Systems for Ullage Exchange Resupply

	Impact on Satellite Prop. Systems	Wt./Unit (lbs.)	Source of Information	# of Units Required	Total Wt. (lbs.)
3.	Bi-Propellant, Pressure Regulated Propulsion System				
	a. Ullage Exchange Modifications				
	i. Interface Panel				
	1. Panel	44.0	Engineering estimate	1 (1/vehicle)	44.0
	2. Disconnects	4.0	LEM design # 4	5 (2ox/2fu/1He)	20.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	2	50.0
	2. Valves	2.3	Shuttle component	9 (4ox,1He), (4fu)	20.7
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle Component	0	0.0
	System Weight Increase				158.7
	b. Ullage Exchange Modifications, Disconnect/Line Purge Option				
	i. Interface Panel				
	1. Panel	69.0	Engineering estimate	1 (1/vehicle)	69.0
	2. Disconnects	4.0	LEM design # 4	9 (4ox/4fu/1He)	36.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	2	50.0
	2. Valves	2.3	Shuttle component	13 (6ox,1He), (6fu)	29.9
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle component	0	0.0
	System Weight Increase				208.9

Table 44. Weight Impact on Pump-Fed Storable Bipropellant Propulsion Systems for Ullage Exchange Resupply

	Impact on Satellite Prop. Systems	Wt./Unit (lbs.)	Source of Information	# of Units Required	Total Wt. (lbs.)
4.	Bi-Propellant, Pump-Fed Propulsion System				
	a. Ullage Exchange Modifications				
	i. Interface Panel				
	1. Panel	50.0	Engineering estimate	1 (1/vehicle)	50.0
	2. Disconnects	4.0	LEM design # 4	6 (2ox/2fu/2He)	24.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	2 (1ox/1fu)	50.0
	2. Valves	2.3	Shuttle component	10 (4ox/1He), (4fu/1He)	23.0
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle Component	4 (1/tank)	9.2
	System Weight Increase				180.2
	b. Ullage Exchange Modifications, Disconnect/Line Purge Option				
	i. Interface Panel				
	1. Panel	75.0	Engineering estimate	1 (1/vehicle)	75.0
	2. Disconnects	4.0	LEM design # 4	10 (4ox/4fu/2He)	40.0
	3. Electrical Connections	1.0	Engineering estimate	2 (pwr,data)	2.0
	4. Grapple Fixture	22.0	Shuttle I/F control doc.	1 (1/vehicle)	22.0
	ii. Valve Panels				
	1. Panels	25.0	Eng. est. based on Shuttle	2 (1ox/1fu)	50.0
	2. Valves	2.3	Shuttle component	14 (6ox/1He), (6fu/1He)	32.2
	iii. Other Components				
	1. Valves (Tank Isolation)	2.3	Shuttle component	4 (1/tank)	9.2
	System Weight Increase				230.4

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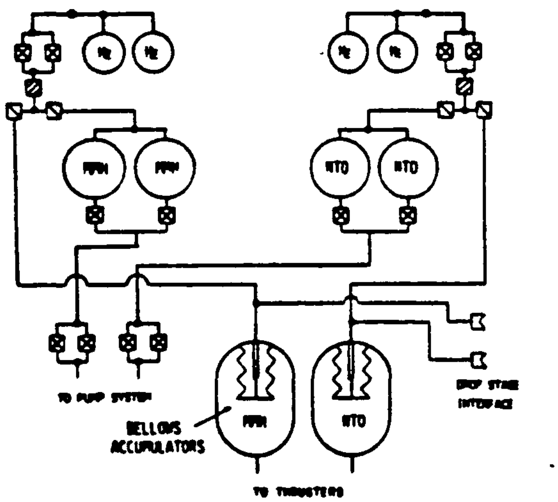
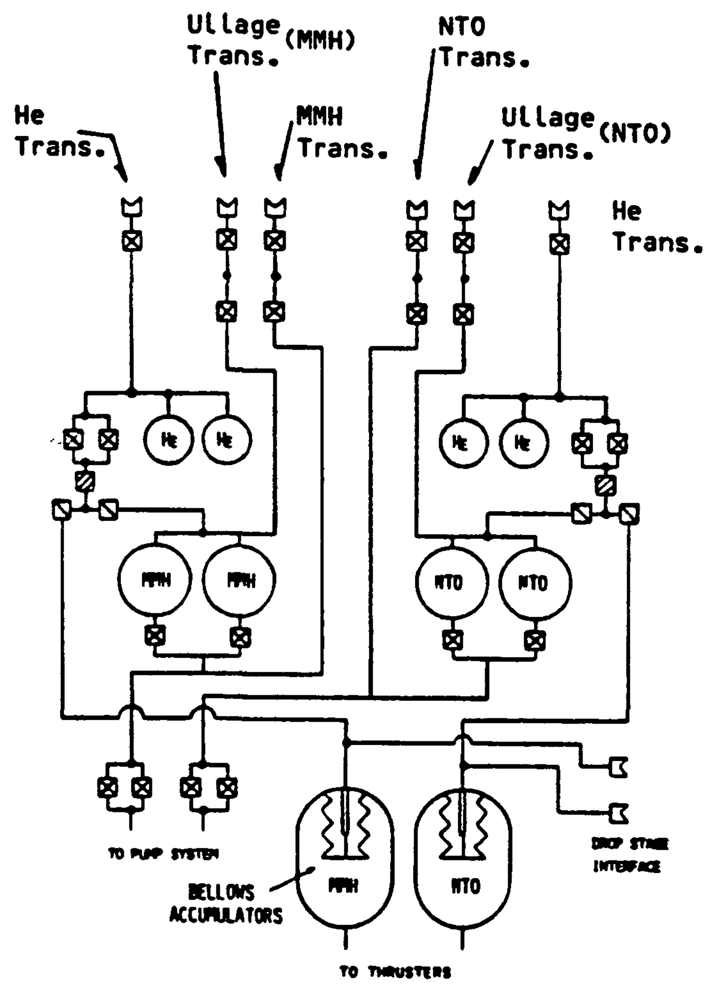
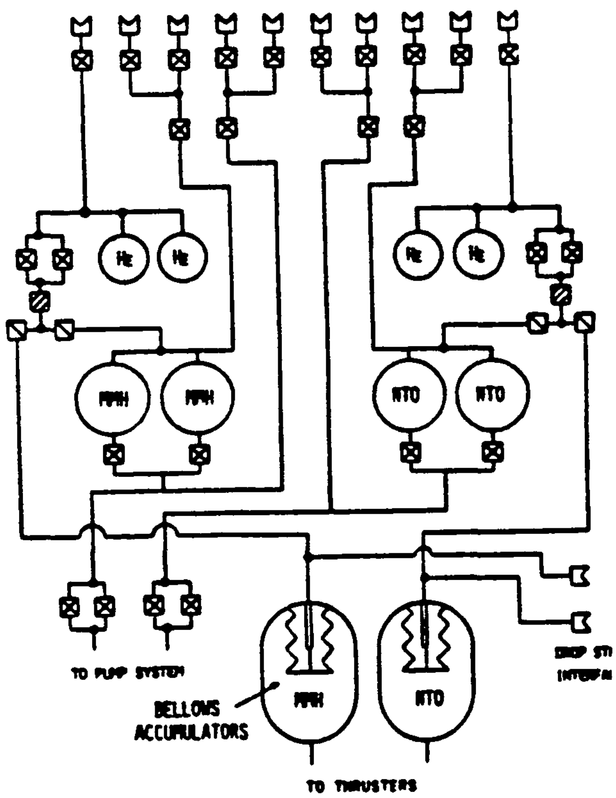
Figure 109. Impact on Bipropellant Pump-fed Propulsion Systems for Ullage-Exchange Resupply



A. SATELLITE PROPULSION SYSTEM
(NO MODIFICATIONS FOR ON-ORBIT RESUPPLY)



B. MODIFICATIONS FOR ULLAGE EXCHANGE PROPELLANT RESUPPLIES



C. MODIFICATIONS FOR ULLAGE EXCHANGE PROPELLANT RESUPPLIES
(DISCONNECT/LINE PURGE OPTION)

3.2.18 Advanced Development Requirements

The essential functions which are required to demonstrate the operational interfaces of the resupply module include the final closure and remote fluid disconnect engagement and actuation, proper ullage positioning of the receiver and supplier's propellant tank PMD for the ullage exchange process, total system operation and checkout utilizing microprocessor control and safe system termination including disconnect disengagement, fluid transfer plate unlatching and safe vehicle separation. Each function requires elements to be included in the flight demonstration program as follows:

- Docking end-effector/grapple fixture snare and final closure to remote fluid disconnect engagement and actuation-demonstrates ability to remotely provide close tolerance alignment for latching and remote fluid disconnect engagement and actuation

- Ullate positioning of the receiver and supply tank's propellan management device is required to exhibit proper ullage control during propellant transfer by ullage transfer process.

- Microprocessor control of system status and health monitoring, fluid transfer including the pumping system, valving, etc., and finally line purging and system safing.

- system safe termination including remote disconnect disengagement, fluid transfer plate unlatching, and finally safe vehicle separation.

Several advanced developments are required to provide the base needed for the types of components to be used for remote resupply. These risk areas include seal develoment for fluid quick-disconnects, propellant pumps, and gaseous compressors, as well as fluid acquisition/ullage positioning in order to provide a contamination free transfer process.

- The components utilizing these advanced developments would include the following:

DEVELOPMENT RISK COMPONENTS

Propellant Pumps	Pump shaft seals and lubrication	Moderate
Gas Compressor	Compressive element seals, staging Requirements and heat rejection and lubrications	Moderate
Quick Disconnects	Sliding redundant seals	Low
Propellant Management	Desin of low-g ullage positioning device	Low, low

The development risk of each technology element is inversely proportional to is technology maturity level (as defined by standard NASA technology development descriptions). The gas compressors, quick disconnects, and fluid pumps have been identified for a special technology development effort that must preceed the ERM program. Contamination control is not expected to be a special problem for the fluid resupply operations anticipated due to the selection of the ullage exchange process for the bi-propellant transfer. Table 49 summarizes the technology status for the resupply module program.

Table 45. Status of Technology for the Resupply Module Program

Technology Maturity Levels		Technology Requirements																	
		Fluid Containment/Transfer									Design				Remote Operations				
No.	Description	Fluid Acquisition	Fluid Transfer	Fluid Pumps	Gas Compressor	Quick Disconnects	Flow/Quantity Measurements	Leakage Monitoring	Contamination Control	Temperature Control	Structure	Plumbing	Components	Electrical	Rendezvous OPS	Docking	Remote Control	Avionics	Power Interfaces
8	Operations																		
7.2	Engineering Model Tested in Space																		
7.1	Engineering Model Qualified																		
6.3	Prototype Developed to Qualify																		
6.2	Prototype Tested in Test Bed Unmanned																		
6.1	Prototype Tested at Contractors																		
5.4	Preprototype Tested at NASA																		
5.3	Preprototype Tested at Contractors																		
5.2	Major Function Tested at NASA																		
5.1	Major Components Tested at Contractors																		
4.3	Critical Hardware Tested																		
4.2	Critical Function Tested Over Time																		
4.1	Critical Function Demonstrated																		
3.2	Conceptual Design Tested Experimentally																		
3.1	Conceptual Design Tested Analytically																		
2	Conceptual Design Formulated																		
1	Basic Principles Observed and Reported																		
Development Risk Assessment		2	2	3	4	3	1	2	3	1	1				1				

1= None, 2= Very Low, 3= Low, 4= Moderate

Three major components can be identified as requiring special developmenal testing to provide concept feasibility. These are the propellant pump, the helium compressor, and fluid disconnect.

The major test/demonstration items required to verify satisfactory component operation and performance are as follows:

- o Evaluate component design concept feasibility with a workhorse-type test unit.

- o Verify operation and performance of a flight-weight prototype test unit to flight application requirements.

The scope of the Test Program should encompass the following:

- o Breadboard-type tests using referee fluids and a development (workhorse) component test unit shall provide the basic data base for component functional design evaluation.

- o Development tests using referee fluids, propellants and prototype flight-weight test components shall provide the basis for verification of the component design concept feasibility and fluid transfer usage. In addition, component life shall also be demonstrated.

The Development Test Schedule for the components requiring special developmental testing to provide concept feasibility has four major sections. The first two include the generation of specifications and requirements and the design phase for the developmental hardware and test setup. The third section includes the breadboard testing of workhorse units for pumps and compressors and runs from 2nd quarter of FY87 through 2nd quarter of FY88. The fourth section - Development/Verification Testing, provides the verification of operation and performance of flight-weight prototype test units for the pumps and compressors and runs from 3rd quarter FY87 through 4th quarter FY88. Figure 110 presents the overall schedule for development test.

At the conclusion of the development/verification testing flight-weight design components will have been shown to be feasible for operation when subjected to flight environments, ground servicing and fluid transfer usage.

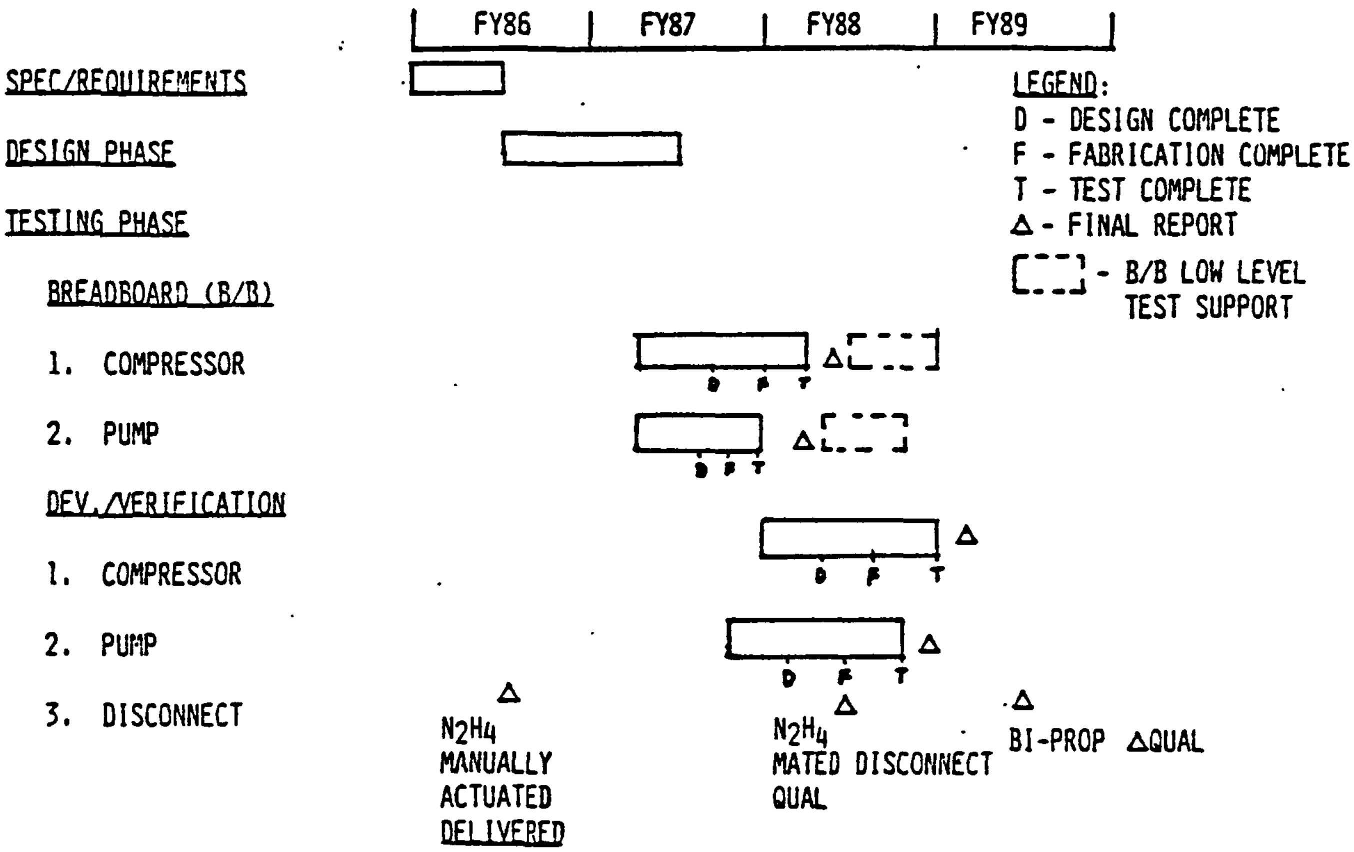


Figure 110. Development Test Schedule for Concept Feasibility

3.2.19 Demonstration Flight Program

The flight demonstration concept will include sufficient fidelity to test the essential functions of remote on-orbit resupply and demonstrate to the user community the viability and attractiveness of such a concept. In order to show proof-of-concept, specific elements will be required on each side of the resupply interface.

Simplicity and economy are emphasized in the selected flight demonstration program (Figure 111). The principle test objective is to validate bi-propellant fluid transfer in a zero-gravity environment. Since the ERM engagement of a spacecraft is accomplished by the OMV with a remote manipulating arm, the remote manipulator system (RMS) on the Orbiter can approximate the OMV engagement capability. The systems on both the receiver and supplier test articles can be attached to either a MPESS (NASA owned) or a SPAS structure (structure only without any subsystems). The Orbiter RMS will lift the receiver test article out of the cargo bay and engage the supplier test article. Repeated fuel transfer tests will then be performed.

Since the supplier test article has the greater need for power and electrical links to the Orbiter, we recommend it remain seated in the cargo bay. This avoids the need for complicated power and electrical links running through the RMS. The supplier test article would be autonomous with its own power source.

The system on both the receiver and supplier test articles (Figure 112) can be attached to either a MPESS (NASA-owned) or a SPAS structure (shown). Grapple fixtures are provided on the receiver simulated vehicle for use in docking with the supplier vehicle and by the RMS to transport the receiver vehicle from a berthed position to the docked position with the supplier vehicle. Fluid transfer system hardware will incorporate to the greatest extend possible final design pumps, compressors, ullage control tankage and disconnects.

Figure 113 presents the resupply module flight demonstration supplies schematic.

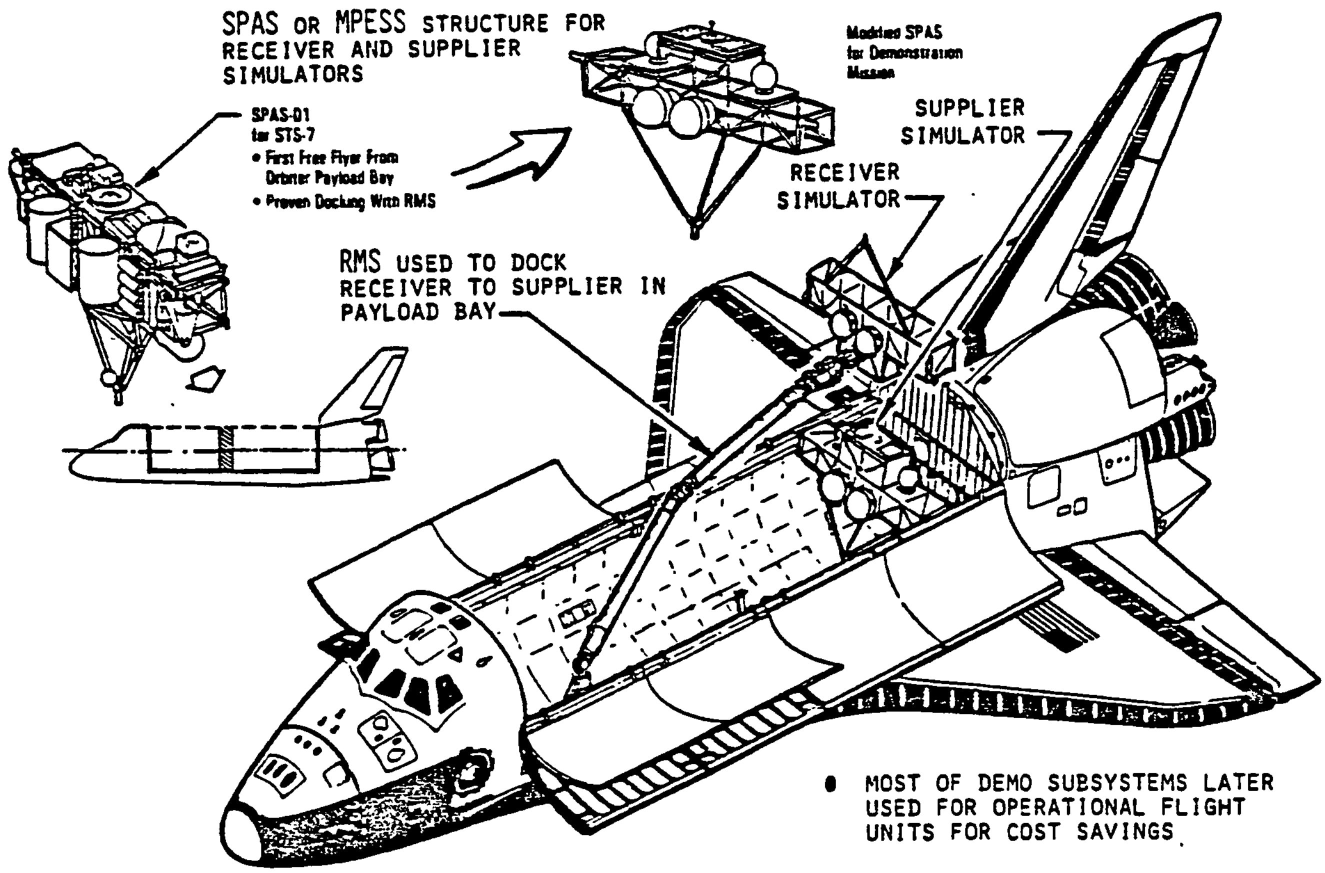


Figure 111.
Bi-Propellant/Helium Transfer Flight Demonstration Concept

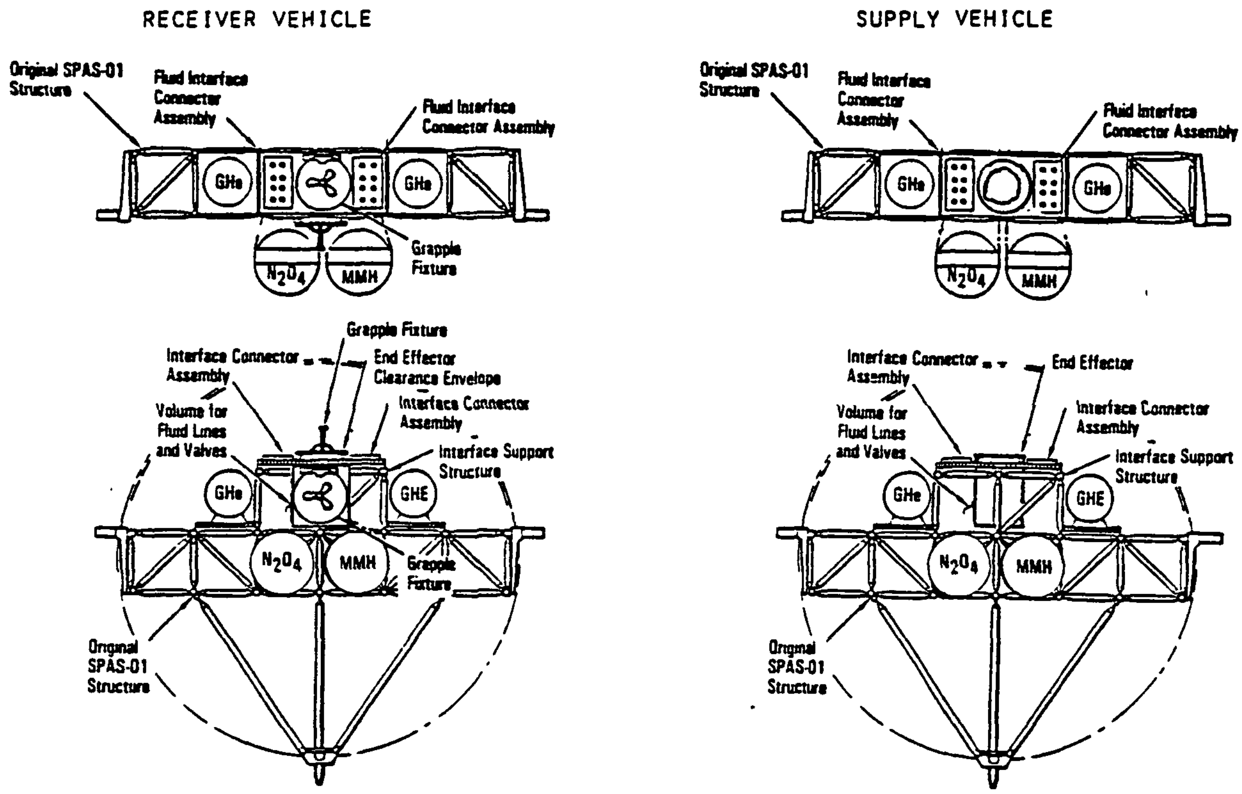


Figure 112. Remote Resupply Flight Demonstration Configuration

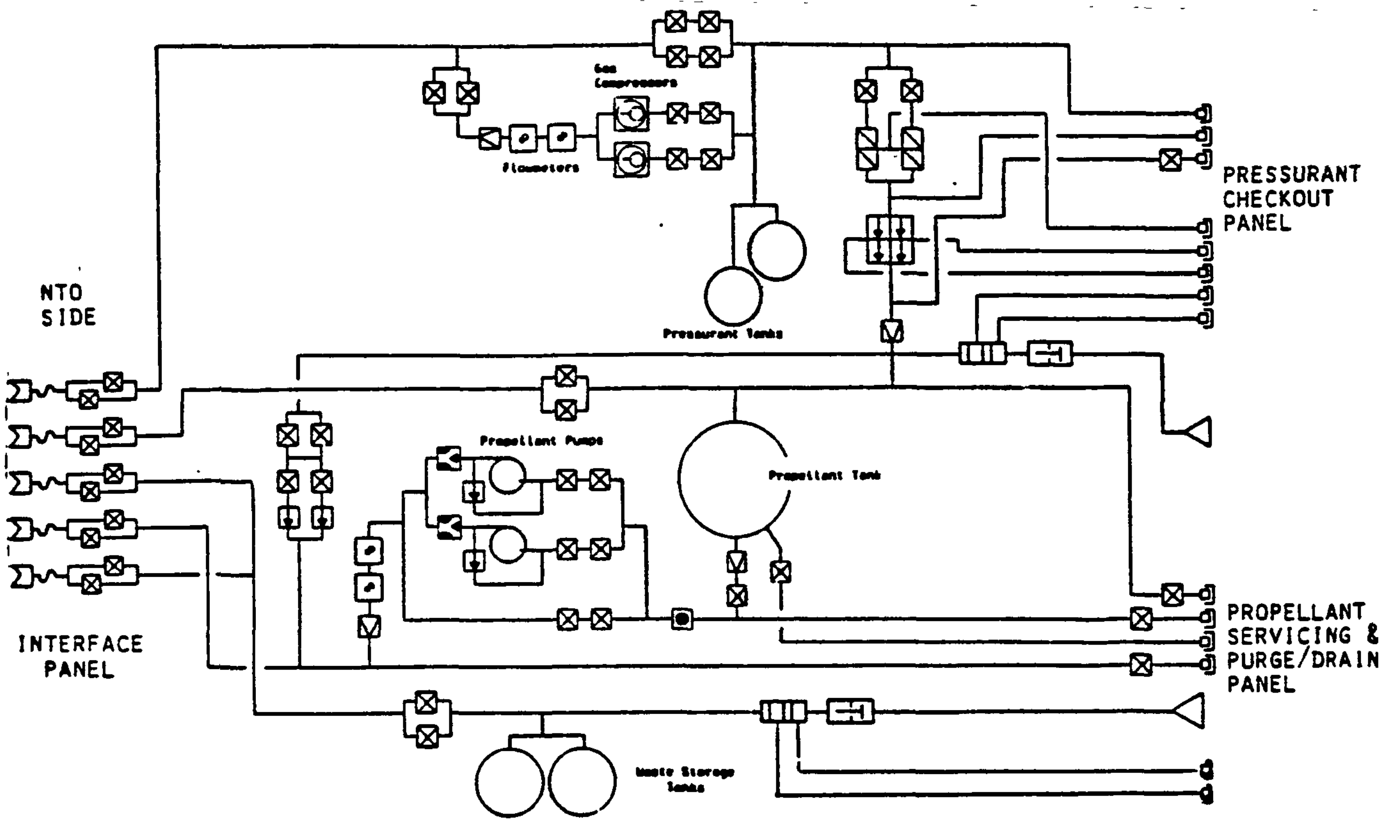


Figure 113.
Resupply Module Flight Demonstration Program Supplier Schematic

3.3 Programmatic and Developmental Planning

The programmatic and developmental planning task included the definition of cost and schedule requirements for the resupply module operational program, technology validation and flight demonstration programs.

3.3.1 Groundrules and Assumptions

Task 3.1, Technology Development, Flight Demo, and ERM Program

The Baseline program consists of two test articles (Developmental and Qualification) and two flight units (one from salvaging systems from the test articles. The earliest possible start for the phase C/D of the ERM program is the fourth quarter FY 1987, and the earliest possible first operational flight is the last quarter of FY 1991.

Task 3.2, Estimated Program Cost

All costs are in constant 1984 dollars. STS launch costs are not included for either the flight demonstration cost or the first ERM operation flight. Costs for GSE and payload support, sustained mission operations/training, maintenance and refurbishment, and additional production units are not included in the baseline ERM program.

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3.3.2 Advanced Development and Demonstration Flight Plans

Two options were considered for the flight demonstration program: An early flight demonstration by the end of the FY 1988 and a late flight demonstration that is part of the ERM phase C/D program that flies a year later at the end of FY 1989.

The technology development program would precede both the flight demonstration program and the ERM phase C/D. All the technology requirements and their maturity status are shown in Figure 114. Standard NASA definitions were used to assess each technology's maturity level. By definition, those technologies with low maturities are the ones that need to be emphasized in the advanced development program. The critical developments are, (1) the gas compressor, (2) fluid pumps, (3) quick disconnects, contamination control, and leakage monitoring.

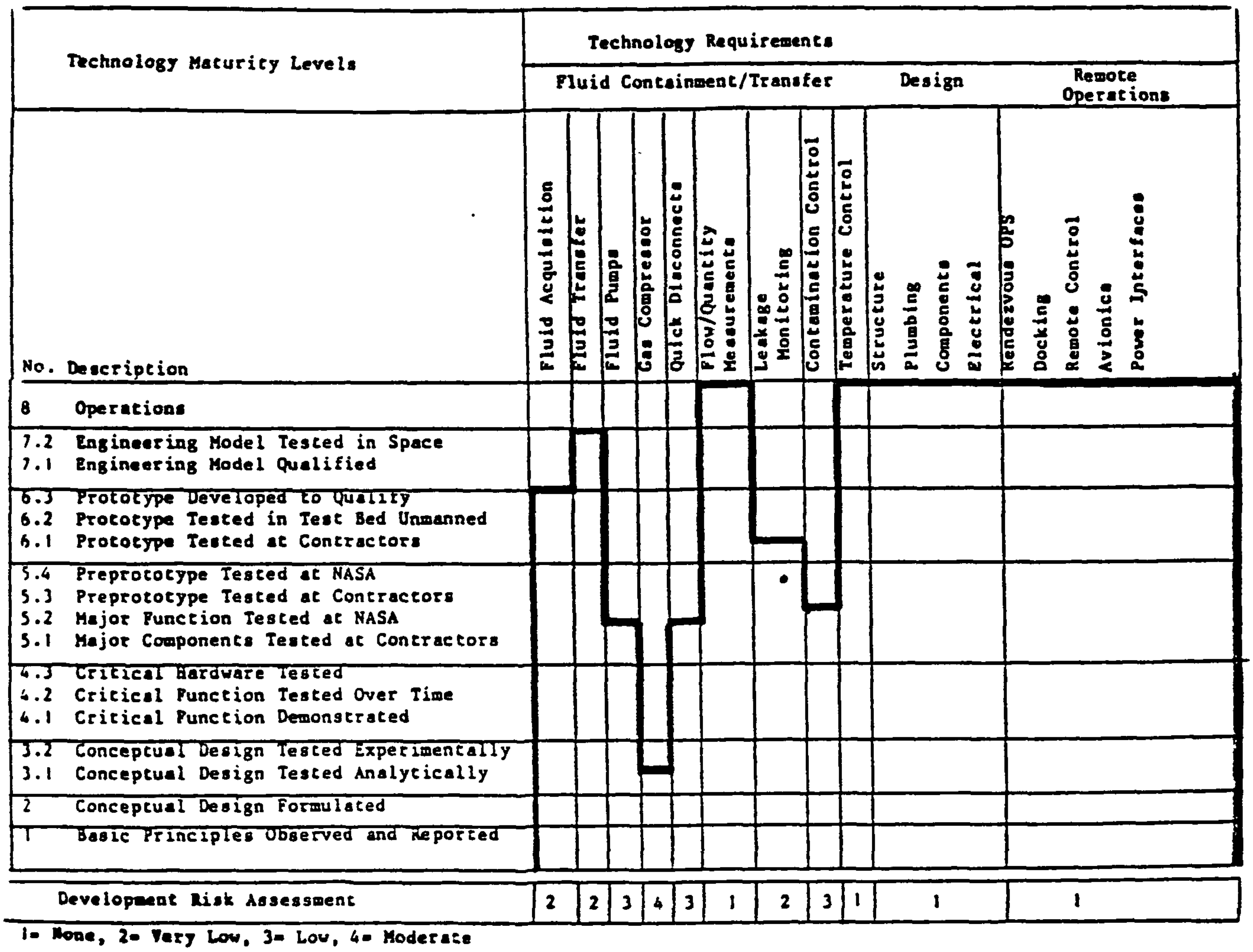

Figure 114. Status of Technology
Required For ERM Program

The schedule for the early flight demonstration option is shown in Figure 115. Breadboard testing precedes the verification testing. The verification testing could culminate with the flight demonstration.

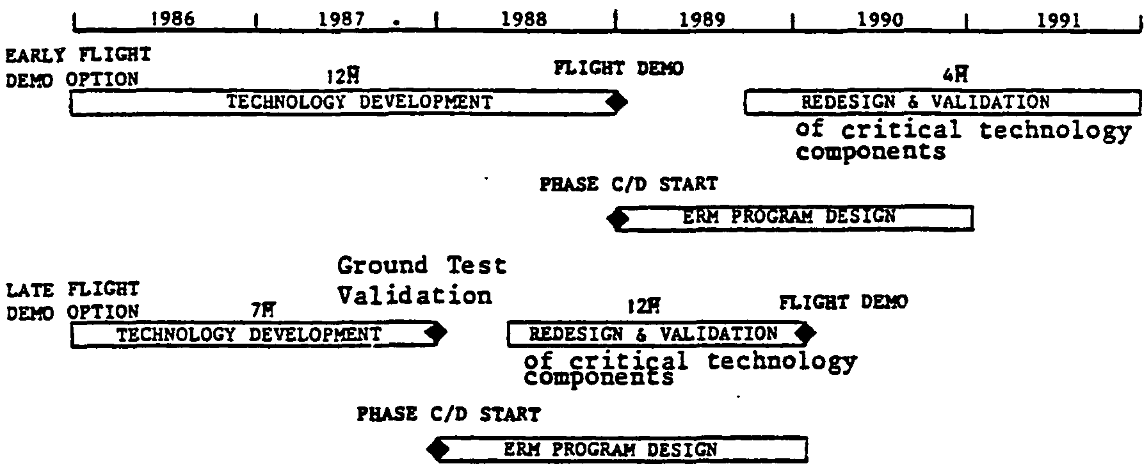

Figure 115
Early Vs. Late Demo Option Schedules/Costs

The schedule for the late flight demonstration option is shown in Figure 116. The technology development phase culminates with the successful verification of key predefined ground tests with breadboard test apparatus, probably by the end of FY 1987.

This schedule is shown in Figure 116. Given a successful verification at the breadboard level, the ERM phase C/D could start. After the specification and requirements for the ERM program are defined, the flight demonstration program would begin. Changes to previous specification and requirements defined in the advanced development program will probably result in a redesign/modification to the critical components in parallel with the design of the flight demonstration.

Each of these two options for the flight demonstration program have their advantages and disadvantages. The cost of the two options are nearly the same as shown in Figure 113. The early estimated cost for the flight demonstration program is $12M with $4M required to modify and validate any specification or requirement changes once the ERM phase C/D program starts. The estimated cost for the late flight demonstration is $7M for the technology development and then $9M for the flight demonstration. Both options—technology development and flight demonstration combined—cost nearly $16M.

The other considerations in closing between the two options are, (1) risk, and whether an early ERM phase C/D program start is more desirable than an early customer acceptance/commitment to the ERM program. The advantages and disadvantages of the two options are displayed in Figure 117. Many issues are still unanswered before making a selection between these two options or the creation of other options:

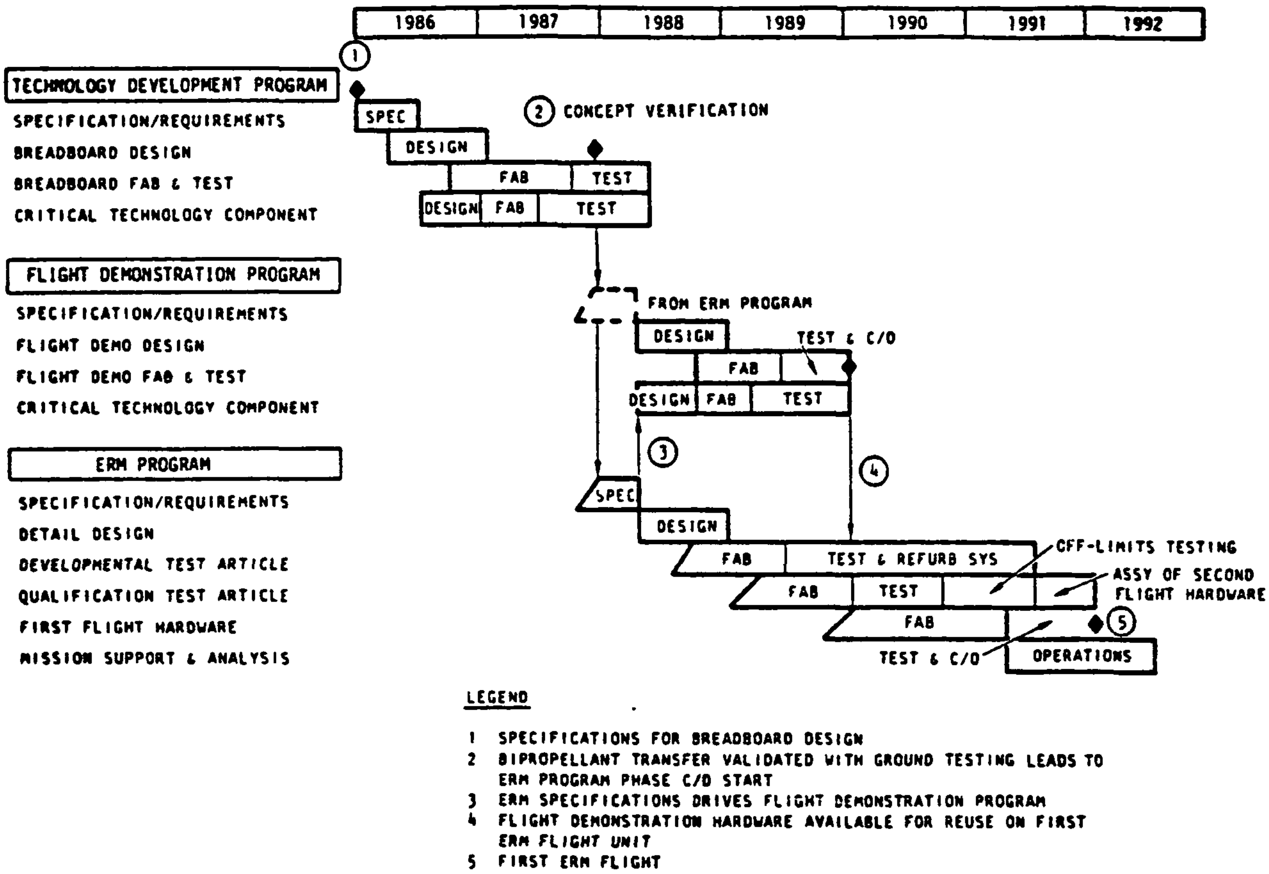

Figure 116. Late Demonstration Flight Schedule

a) Is the flight demonstration needed to gain customer acceptance/commitment?
b) Is the flight demonstration needed to reduce the technical risk of the program? or both

If the user community could be persuaded to support and commit to the ECM program without the flight demonstration, then we recommend the late flight demonstration option and the schedule shown in Figure 116. A technology development program would be initiated prior to the start of an operational expendables resupply program. The Technology Development Program would use ground testing to develop results sufficient to initiate the operational program.

Preferably, the specifications and requirements for the Technology Development Program would be oriented to the ERM Program so that the critical technology components (pump, compressor, and quick disconnects) developed would be usable without extensive redesign. More likely, however, considerable learning will take place during the Technology Development Program and the Phase B of the ERM Program. Thus, the critical technology components will probably require some redesign and, consequently, recertified in the Flight Demonstration

Figure 117. We Recommend a Late Flight Demo & Early ERM Program Start

	EARLY FLIGHT DEMO (END OF FY 1988)	LATE FLIGHT DEMO (END OF FY 1989)
ADVANTAGES	- CRITICAL TECHNOLOGIES TESTED IN SPACE - LOWER RISK - EARLY CUSTOMER ACCEPTANCE/ COMMITMENT TO ERM PROGRAM.	- EARLY ERM PROGRAM START (3RD QUARTER FY 1987) - ERM SYSTEMS TESTED ON FLIGHT DEMO - LOWER RISK
DISADVANTAGES	- ONE YEAR DELAY IN ERM PROGRAM START - DOES NOT TEST SYSTEM DESIGNED FOR THE ERM PROGRAM - HIGHER RISK	- ERM PHASE C/D START BEFORE CRITICAL TECHNOLOGIES ARE TESTED IN SPACE - HIGHER RISK - LATER CUSTOMER ACCEPTANCE/ COMMITMENT TO ERM PROGRAM

Program. Of critical importance, is the fact that all the systems and hardware in the Flight Demonstration Program reflect the specifications and requirements of the ERM Program. Then, systems tested in zero-gravity would be those of the operational expendables resupply program. When flight demonstration is completed at the end of FY 1989, the systems and hardware can be salvaged for reuse on the ERM first flight unit, which is scheduled for launch two years later.

Successful completion of predefined key ground tests, that raise the development of the critical technologies to an acceptable level, lead to the start of the ERM Program. Six months later, after the ERM specifications and requirements are defined, the Flight Demonstration Program would begin, designed to use ERM systems and hardware.

3.3.3 Work Breakdown Structure

The Space Platform Expendable Resupply WBS, as defined herein, was selected mainly to support the program planning, scheduling, and costing performed in the programmatics Task 3. The top level WBS is shown in Figure 118. The selected work breakdown structure for the ERM program consists of the following elements:

1.1 Program management and system engineering is the core of the program and provides the overall program perspective necessary for continuity and decision-making.

1.2 Engineering design

1.3 Developmental, qualification, flight demonstration, and functional testing.

1.4 Developmental, qualification, flight demonstration, and functional testing.

1.5 Mission planning, flight support, and check-out necessary to support the first launch and post-flight analysis.

1.6 Payload support for the resupply module in the orbiter payload bay is GFE (this hardware is available from other programs).

1.7 & 1.8 Operational support beyond the baseline program has not been costed.

The WBS for the Expendables Resupply Module Systems are Orbiter/STS designations, as shown in Figure 119.

The weights for end ERM systems range from a low to a high estimate, reflecting an 80 percent confidence that the actual design weight will be within this range. The total weight of the ERM will range between a low of 8260 and a high of 9170 pounds. The system weights by WBS are given in Figure 120.

Figure 118. Work Breakdown Structure For ERM Program

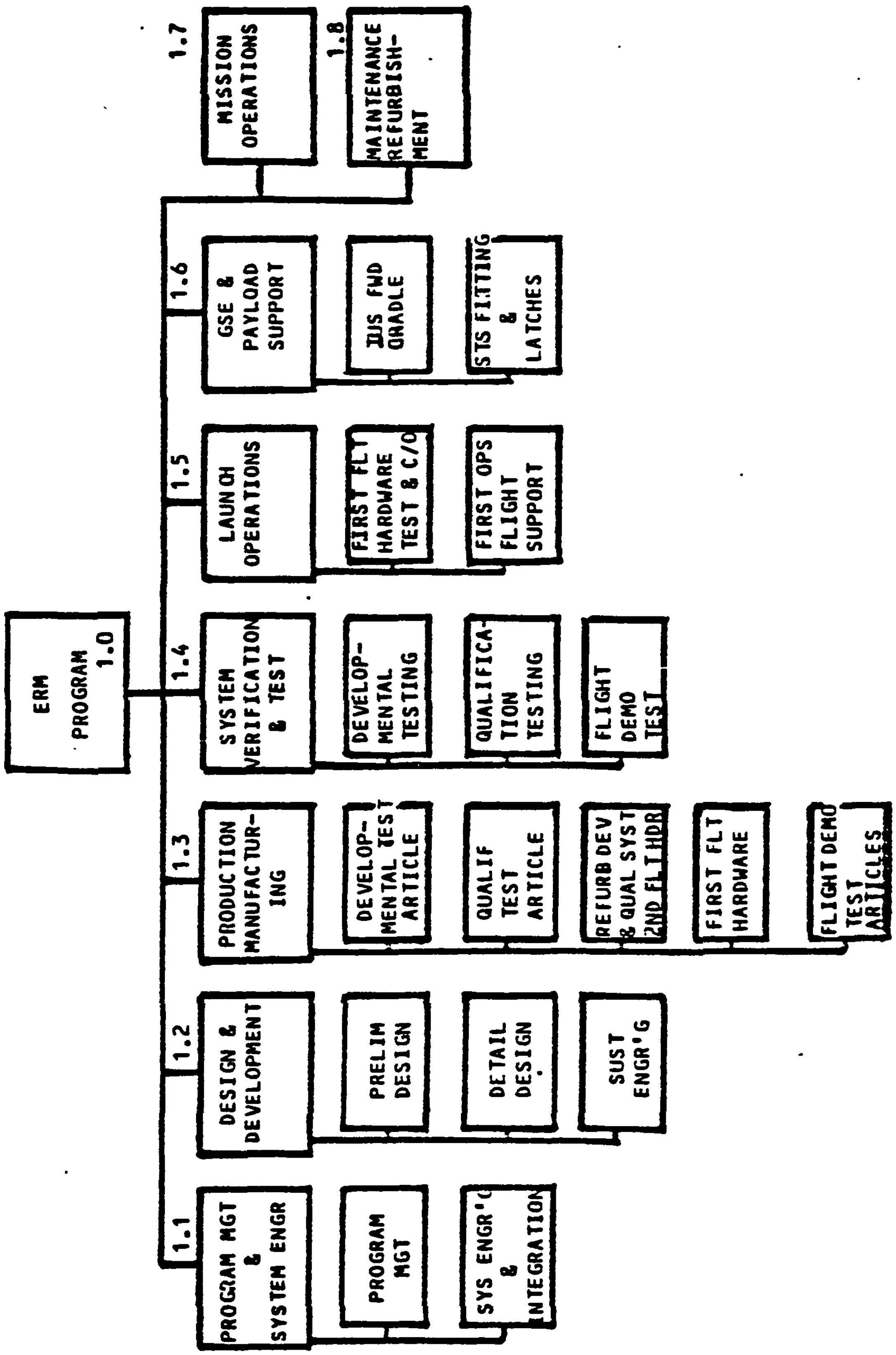

Figure 119. ERM Subsystems Work Breakdown Structure

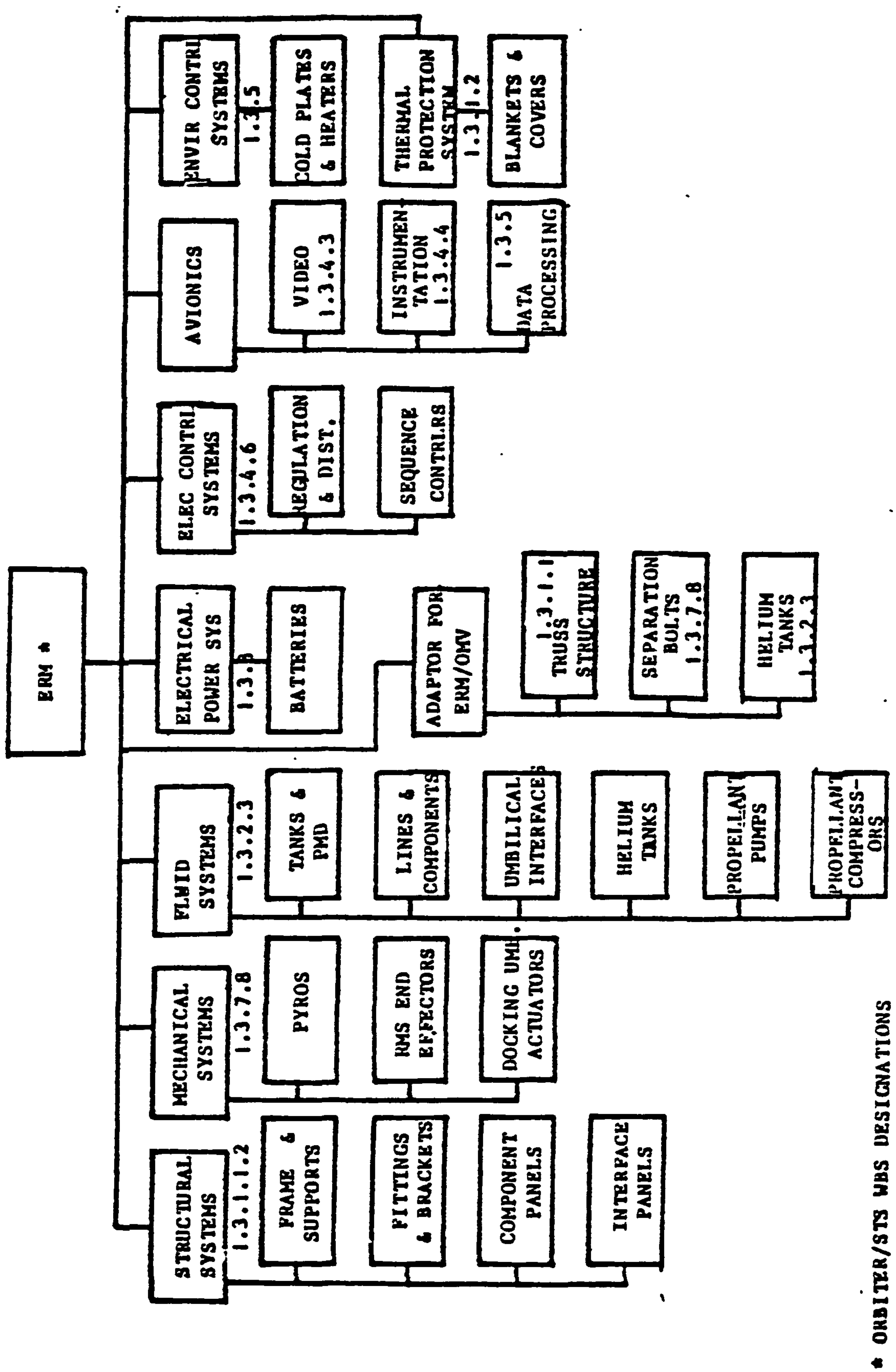

Figure 120. ERM System Weights

ORB/STS	COMPONENT	QTY	CLASS	STS COST BASE	WEIGHT-LB	
1.31.12	BODY STRUCTURE		-	-	(2536)	(2784)
	• FRAME & SUPTS	1	NEW	MID FUS STRUCT	1644	1808
	• FITTINGS & BRKTS	172	"	"	182	200
	• COMPONENT PANELS	20	"	"	560	616
	• INTERFACE PANELS	2	"	"	150	165
1.3.1.2	TPS		-	-	(250)	(275)
	• BLANKETS & COVERS	-	NEW	CREW MOD INSUL	250	275
1.3.7.8	SEPARATION MECHANISMS	-	-	-	(201)	(241)
	• PYRO'S	12	OLD	ORB/ET SEP	36	43
	• RMS END EFFECTORS DOCKING	1 pr	OLD	RMS	135	162
	• UMB.ACT'S	1	NEW	AFT UMB DR MECH	30	36
1.3.2.3	FLUID SYS.	-	10%chg	-	(3721)	(3842)
	• TANKS & PMD	6	MOD	OMS TANKS	2118	2152
	LINES & • COMPONENT	140	NEW/ STS	OMS SYSTEM	386	425
	• UMB. INTERFACE	28	NEW	"	80	96
	• HELIUM TANKS	12	STS	MPS TANKS	977	977
	• PROP PUMP & COMPRSR	4 ea	NEW	NEW	160	192

ORB/STS	COMPONENT	QTY	CLASS	STS COST BASE	WEIGHT-LB	
1.3.3	ELEC.POWER	-	-	-	(800)	(960)
	• BATTERIES (44KW/Hr)	6	NEW	NEW	800	960
1.3.4.6	ELEC.POWER DIST & CNTL	-	-	-	(360)	(504)
	• REG.&DIST	-	MOD	EDPC	200	280
	• SEQ.&CNTLR	-	NEW	EDPC	160	224
1.3.4.3	AVIONICS	-	-	-	(291)	(407)
	• VIDEO	-	MOD	CNTLS & DISPLAY	66	92
	• INSTRUMENTATION	-	MOD	INSTRUMENTATION	100	140
	• DATA PROCESSING	-	NEW	DPS	125	175
1.3.5	ENVIRON. CNTL SYSTEM	-	-	-	(100)	(120)
	• COLD PLATE & HEATERS	-	MOD	EC/LSS	100	120
	WEIGHT-LB	-	-	-	8259	9168

Potential Demand for ERM Units

In Scenarios 1 and 2, the resupply module provides the OMV with propellant for transportation to GEO. This mode was found to be more economical than using an expendable OTV to transport the OMV to GEO. The propellant transfer (and concurrent module exchange servicing, when required), is accomplished directly from the OMV after separation from the resupply module (RM). These missions are expected until the late 1990's. When they do occur, there will be a maximum of three, placing three OMS into orbits where there are clusters of satellites (over North America, Europe, and Japan). In Scenarios 3 through 10, the resupply module operates in LEO and is reusable.

- Some RM's will be parked at the propellant depot (Scenario 4) and participate in propellant scavenging operations from the STS (Scenario 10).

- Some RM's will perform resupply and/or top-off operations:

  o S/C with integral propulsion systems going to GEO (Scenario 3)

  o Reusable perigee vehicles (Scenario 5)

  o Reusable OMV/RM perigee vehicles (Scenario 8)

  o Space-based OMV (Scenario 9)

  o Top-off of RM for Scenarios 1 and 2 (Scenarios 6 and 7)

At a minimum, two ERM's are required to provide a threshold level of service. One ERM will be located at the depot and participate in propellant scavenging from the orbiter. One ERM will perform the propellant transfer and top-off operations. These ERM's are interchangeable and probably rotate functions. When one ERM is filled with propellant from the scavenging operations, it will switch roles with a near empty ERM to perform resupply and top-off missions.

The potential resupply scenarios for the ERM program and the number of ERM units required initially in the early 1990's, and later in the late 1990's, is shown in Figure 121.

3.3.4 Baseline Program

The ERM baseline program includes two test articles, the developmental and qualification test articles, so that testing can be performed in parallel. This gives the program schedule slack and reduces schedule risk. The ERM baseline program also includes two flight units. One of these flight units is assembled from the salvageable system on the two test articles. The ERM phase C/D program could start as early as the third quarter of FY 1987 and have its first operation flight by the end of FY 1991. After two periods of docking, scavenging and fluid transfer demonstrations, expendables resupply operations could commence in FY 1992.

Figure 121. Potential Resupply Scenarios And Estimated Number of Expendables Resupply Modules by Year 2000

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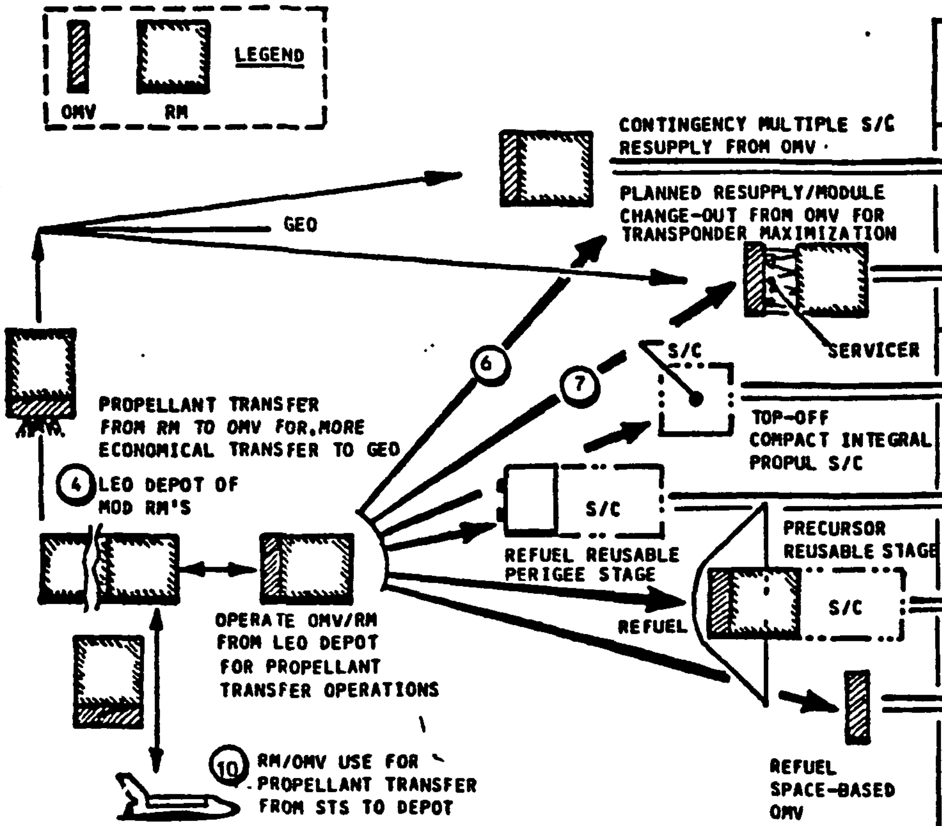
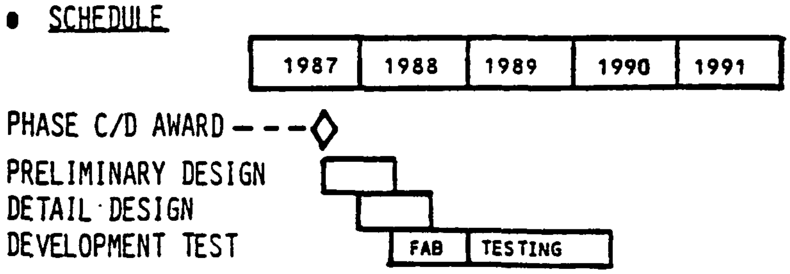


POTENTIAL MISSION SCENARIO	ERM TYPE	ESTIMATED NO. OF RM'S REQD.	
		MIN	MAX
1, 2	EXPEND	EARLY 1990's 0	EARLY 1990's 3
3, 5, 8, 9	REUSE	2	4
	TOTALS	2	7

3.3.5 ERM Developmental Test Program

The WBS systems required to perform developmental testing as shown in Figure 122 are the mechanical, fluid, electrical and avionics. The body structure will resemble the final RM configuration because it will be fabricated from the machine proofings preceding the machining of parts for the flight qualified hardware.

- DEVELOPMENTAL TEST ARTICLE
  - BODY STRUCTURE (1.3.1.1) FROM LOW COST MATERIAL RUN THROUGH MANUFACTURING MACHINES TO VERIFY NUMERICAL CONTROL SETTING
  - MECAHNICAL (1.3.7.8) - EXCEPT PYROS
  - FLUID SYSTEMS (1.3.2.3)
  - ELECTRICAL POWER DISTRIBUTION/CONTROL (1.3.4.6)
  - AVIONICS (1.3.4.3)
- SCHEDULE



- DEVELOPMENTAL TEST OBJECTIVES
  - FLUID TRANSFER TESTS
    - DOCKING AND RELEASE
    - PRESSURANT TRANSFER - BASIC FUNCTION, MULTIPLE REPEATS, CONNECTION RELIABILITY LEAKAGE MONITORING
  - RESUPPLY MODULE/ORBITER COMPATIBILITY TESTS
    - INSTALLATION/INTERFERENCE

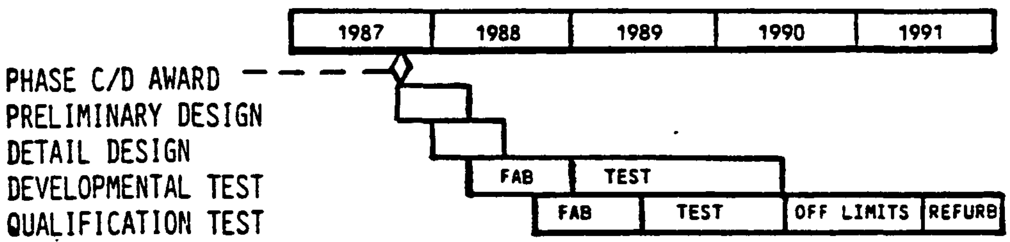
Figure 122. ERM Developmental Test Program

Pyro, environmental control, thermal protection and batteries are not needed for the developmental test program. Parallel testing permits added flexibility to the developmental test program. This test article has a year-and-one-half slack before its systems are needed for assembly into the second flight unit. Therefore, developmental testing could continue for longer than the anticipated one-year period if necessary without disrupting the program.

3.3.6 Qualification Test Program

The systems required on the qualification test article are shown in Figure 123. This test program is likely to leave most sensitive systems unusable for flight hardware. Thus, inexpensive mass simulators will be substituted whenever possible.

- QUALIFCATION TEST ARTICLE
  - BODY STRUCTURE (1.3.1.1) - FIRST PRODUCTION UNIT
  - COLD PLATES/HEATERS (1.3.5) & MASS SIMULATORS
  - THERMAL PROTECTION BLANKET (1.3.1.2)
  - FLUID SYSTEMS (1.3.2.3)
  - ELECTRICAL POWER DISTRIBUTION & CONTROL (1.3.4.6) & MASS SIMULATORS
  - MASS SIMULATORS FOR MECHANICAL SYSTEMS, AVIONICS, BATTERIES
- SCHEDULE



- QUALIFICATION TEST OBJECTIVES
  - ACOUSTIC VIBRATION TESTS
  - THERMAL VACUUM TESTS
  - RANDOM VIBRATION TESTS
  - FLUID TRANSFER OVER SPECIFIC OFF-LIMIT RANGES

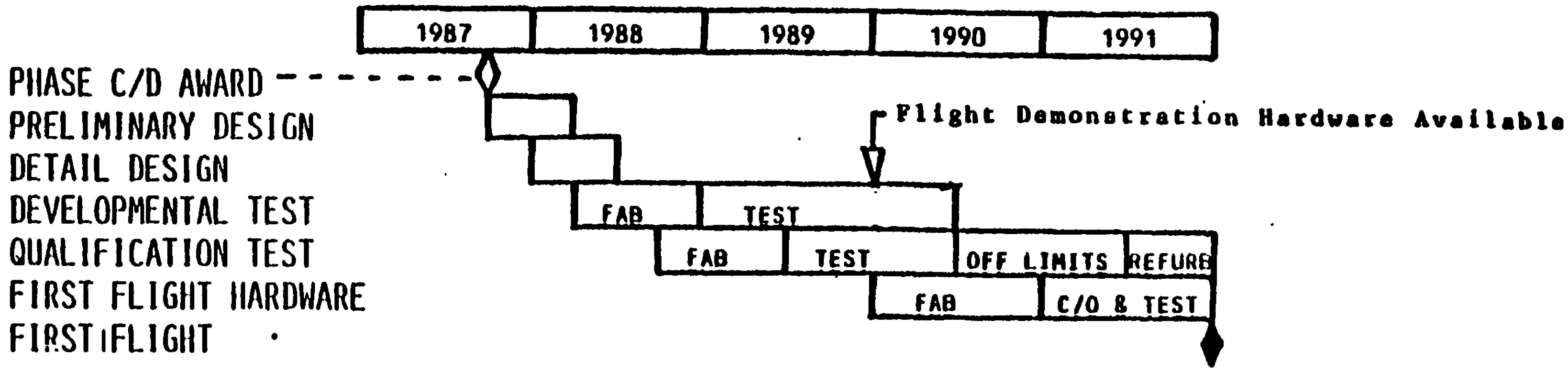
Figure 123. ERM Qualification Test Program

The parallel testing approach to this program allows sufficient time for off-limits testing to validate the ERM for operations outside its design range.

FIRST ERM FLIGHT HARDWARE

The first flight unit can be produced six months early as shown in Figure 124 and available if necessary to supplement the test validation program before it is needed for flight check-out. The first operational flight will occur in the last quarter of FY 1991.

- **FLIGHT CHECK-OUT AND SYSTEMS/FUNCTIONAL TEST ARTICLE**
  - ALL SUBSYSTEMS EXCEPT ADAPTER FOR ERM/OMV
- **SCHEDULE**



- **FLIGHT CHECK-OUT AND SYSTEMS/FUNCTIONAL TEST OBJECTIVES**
  - VALIDATE SYSTEMS/FUNCTION OPERATIONS
  - TEST MALFUNCTION PROCEDURES

Figure 124. First ERM Flight Hardware

SECOND ERM FLIGHT HARDWARE

The second ERM flight unit is assembled from the systems salvaged from the development and qualification test articles as shown in Figure 125. This will avoid $12 to 18 million in additional costs. Thus, the additional cost to assemble the second flight unit is only $4 to 6 million.

($12-18M) SUBSYSTEMS SALVAGED FROM DEV. & QUAL TEST ARTICLES	+($4-6M) OTHER SYSTEMS AND ASSEMBLY & C/O	= SECOND ERM FLIGHT HARDWARE

O DEVELOPMENTAL TEST ARTICLE:
- BODY STRUCTURE (1.3.1.1) - PROOFING/EXPENDABLE -- ⊗
- MECHANICAL SYSTEMS (1.3.7.8) - EXCEPT PYRO
- FLUID SYSTEM (1.3.2.3)
- ELECTRICAL POWER DISTRIBUTION/CONTROL (1.3.4.6)
- AVIONICS (1.3.4.2)

O QUALIFICATION TEST ARTICLE:
- BODY STRUCTURE (1.3.1.1) - FIRST PRODUCTION UNIT
- COLD PLATE/HEATERS (1.3.5)-- ⊗
- THERMAL PROTECTION BLANKET (1.3.1.2)
- FLUID SYSTEMS (1.3.2.3)-- ⊗
- ELECTRICAL POWER DISTRIBUTION & CONTROL (1.3.4.6)-PARTIAL

O OTHER SUBSYSTEMS:
- BATTERIES (1.3.3)
- MECHANICAL - PYROS (1.3.7.8)
- COLD PLATES/HEATERS (1.3.5)

→ INTEGRATE REUSABLE SUBSYSTEMS INTO A SECOND ERM FLIGHT HARDWARE

FIGURE 125. Second ERM Flight Hardware

3.3.7 Build and Flow

The manufacturing plan for fabricating the ERM structure is shown in Figure 126. The planning, calculating the numerical control settings for the machines, and proofing (running low cost material through the machines to verify the adequacy of the numerical control settings) are only performed once. The first production unit is the qualification test article which will be later assembled into the second flight unit. The second production unit will take 42 weeks to complete and will be used as the first flight unit.

3.3.8 Advanced Development and Flight Demonstration Cost

Ideally, the same personnel involved in the Advanced Development Program will roll over to support the Flight Demonstration Program. They will also indirectly support the ERM Program by helping to shape its specifications and requirements.

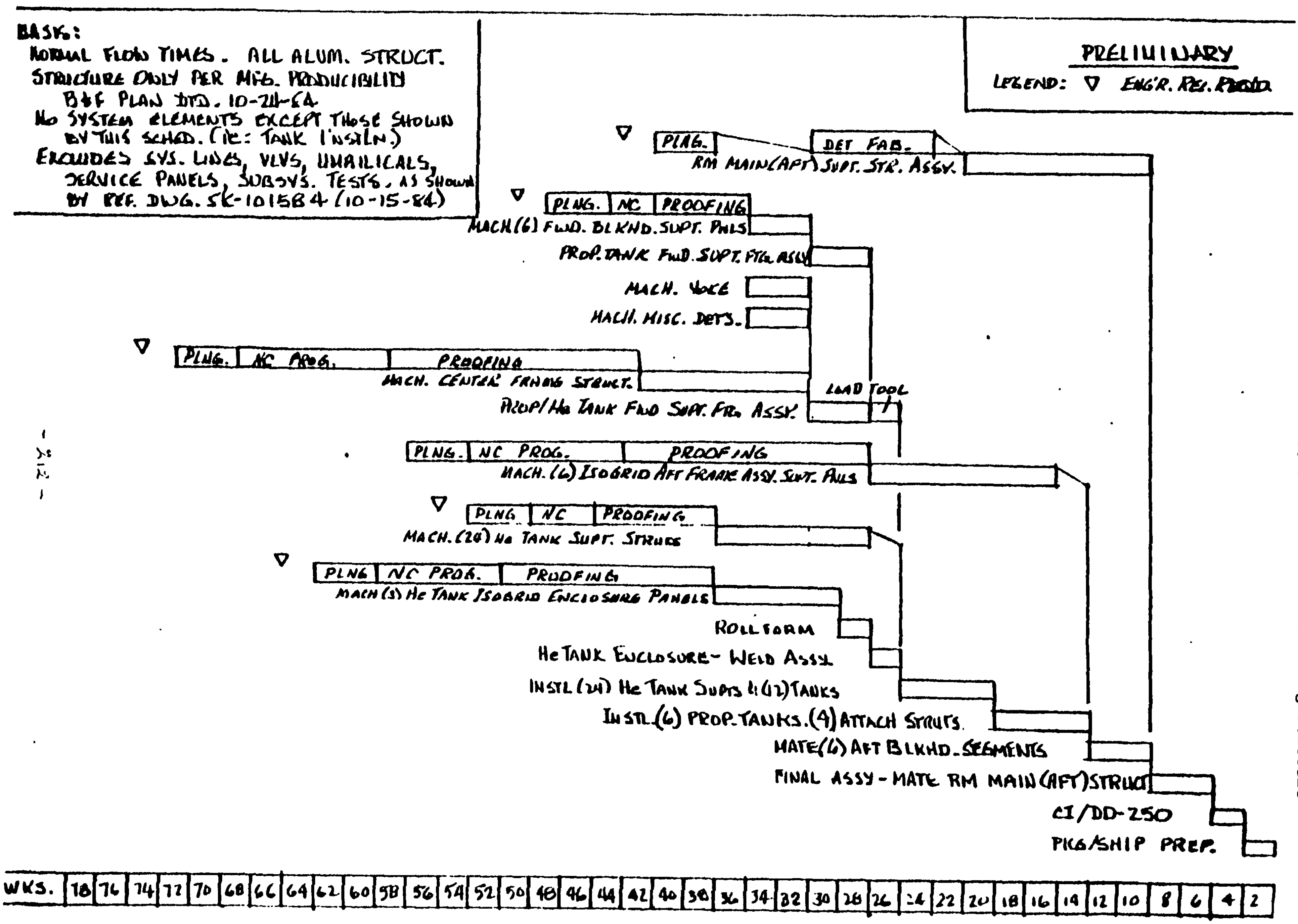


Figure 126. Expendables Resupply Module Manufacturing Schedule

There is time delay between the Advanced Development Program and the Flight Demonstration Program while the ERM Specifications and Requirements are defined as shown in Figure 127. If the user/customer community can be persuaded to support/commit to the ERM program without an early flight demonstration program, then we recommend the late flight demonstration option. This schedule is preferred over a continuous Advanced Development Program that culminates with a flight demonstration for two reasons:

1. If the ERM Program is deferred until a flight demonstration, the program cannot start until the end of FY 1988 or 12 to 15 months later.

2. The systems and concepts verified on the flight demonstration may be different than those eventually specified for the ERM Program.

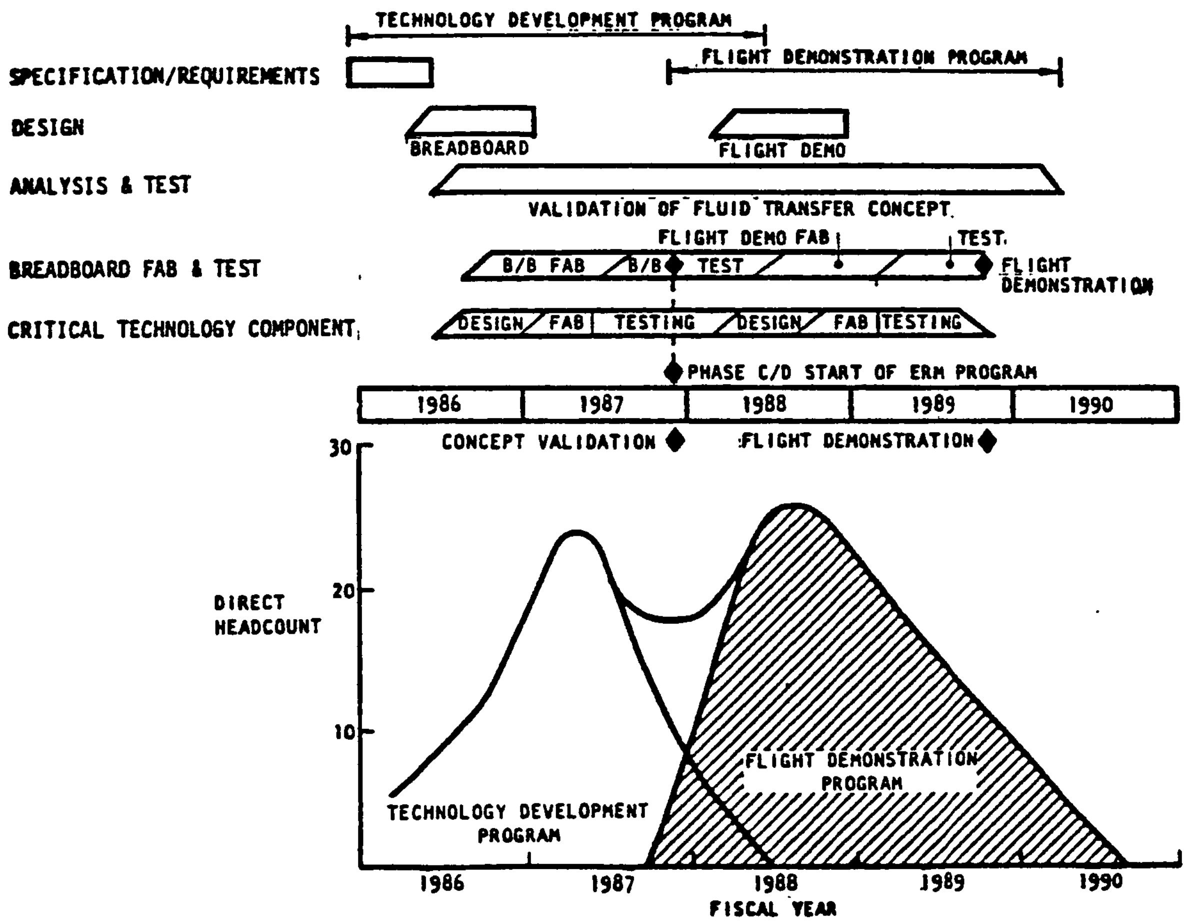


Figure 127. Advanced Development Program Supports the Flight Demonstration Program

A rough best estimate for the total cost of the Advanced Development Program and Flight Demonstration Program is $7M and $9M, respectively. Details of the $16M, or cost estimate are presented in Figure 128 from manpower requirements, labor rates, and the percentage of the program subcontracted. The labor rate includes direct and fringe benefits, overhead, and G&A. None of the cost estimates include subcontractor fees. The labor rate is uncertain because of the level of indirect support allocated to overhead.

TECHNOLOGY DEVELOPMENT AND FLIGHT DEMONSTRATION

	LOW	BEST ESTIMATE	HIGH
DIRECT HOURS	133,400 HR -10%	148,200 HR +20%	177,800 HR
LABOR RATE	$50/HR	$55/HR	$60/HR
% SUBCONTRACTED	30%	40%	50%
	9.5M	13.6M	21.3M
SPAS AND TANKS	1.9M	2.5M	3.4M
	11.4M	16.1M	24.7M

TECHNOLOGY DEVELOPMENT PROGRAM

	LOW	BEST ESTIMATE	HIGH
DIRECT HOURS	51,900 HR -10%	57,700 HR +20%	69,200 HR
LABOR RATE	$50/HR	$55/HR	$60/HR
% CONTRACTED	30%	40%	50%
	3.7M	5.3M	8.3M
TANKS	1.2M	1.6M	2.4M
TOTAL	4.9M	6.9M	10.7M

FLIGHT DEMONSTRATION PROGRAM

	LOW	BEST ESTIMATE	HIGH
DIRECT HOURS	81,500 HR 10%	90,500 HR +20%	108,600HR
LABOR RATE	$50/HR	$55/HR	$60/HR
% CONTRACTED	30%	40%	50%
	5.8M	8.3M	13.0M
SPAs	0.7M	0.9M	1.0M
TOTAL	6.5M	9.2M	14.0M

Figure 128. Cost Estimate for Technology Development and Flight Demonstration Programs

If the specifications and requirements for the critical technology components do not change in the ERM Program, then these components need not be redesigned. The cost of the Flight Demonstration Program will be $4M less if it can use the components from the Technology Development Program and "borrow" systems/parts from the first ERM flight unit. Thus, if the ERM specifications and requirements do not change, the total cost for the technology development and flight demonstration is $12M.

3.3.9 Baseline Program Cost

The ERM baseline program cost was estimated by two separate and independent methods: Parametric and detailed engineering. The purpose was to confirm our cost estimate for using two methods.

Parametric Estimate for ERM Program

The parametric cost estimating relationships for each subsystem (except software) depends on weight, a factor for degree of design modification (a new design has a factor of 1.0), and design complexity relative to a baseline program for which costs are known (a system as complex as the Orbiter for instance would rate a factor of 1.0).

System DDT&E — Cost Estimate $= A * Nd * D_c * (\text{weight})^B$

System Production — Cost Estimate $= C * Pc * (\text{weight})^D$

The high parametric cost for each system reflects the worst combination of weights and complexity factors and the low parametric cost reflects the best. These costs are presented in Figure 129.

ORBITER WBS	COMPONENT	SOURCE OF COST ESTIMATING RELATIONSHIP	WEIGHT RANGE (LB)	DDT&E COST ($\bar{M}$ OF 84$)	NEW DESIGN FACTOR	DESIGN COMPLEXITY	THEORETICAL FIRST AMT ($\bar{M}$ OF 84$)	PRODUCTION COMPLEXITY
1.3.1.1	BODY STRUCTURE	ORBITER & FACTORS	2536-2789	7.7-24.2	1.0	.1-.3	5.0-7.7	.4-.6
1.3.1.2	THERM PROTECT BLKT	SPACE STATION	250-275	.1-.3	.2-.3	.1-.3	.2-.3	.5-.7
1.3.7.8	SEPARATE/MECHANISM	ORBITER & FACTORS	201-241	.3-2.8	.2-.3	.1-.3	1.3-2.2	.5-.7
1.3.2.3	FLUID SYSTEMS	ORBITER & FACTORS	3721-3842	6.8-21.2	.4-.6	.25-.5	4.1-5.9	.5-.7
1.3.3	ADV. BATTERIES	N_1H_2 QUOTE	800-960	3.8-4.5	1.0	PURCHASE	1.9-3.2	.5-.7
1.3.4.6	ELEC PWR DIST/CONTR	SPACE DIV CER	360-504	.5-1.7	1.0	.1-.3	1.1-2.0	.5-.7
1.3.4.3-5	AVIONICS	'83 PROCURE	291-407	1.2-1.5	-	PURCHASE	1.2-1.5	PURCHASE
1.3.5	COLD PLATE/HEATERS	SPACE STATION	100-120	.8-1.6	1.0	.1-.2	.8-1.6	.1-.2
			SUBTOTAL	21.2-57.9			15.6-24.4	
1.3.1.1	TRUSS STRUCT	ORBITER & FACTORS	148-163	1.9-6.1	1.0	. -.3	1.4-2.2	.4-.6
1.3.7.8	SEPARATION BOLTS	ORBITER & FACTORS	36-43	.1-.1		PURCHASE	.1-.1	PURCHASE
1.3.2.3	HELIUM TANKS	'83 PROCURE	160	.3-.3		PURCHASE	.3-.3	PURCHASE
			SUBTOTAL	2.3-6.5			1.8-2.6	
	SOFTWARE			.1-.2				
			TOTAL	23.6-64.6			17.4-27 $\bar{M}$	

Figure 129. Parametric Cost Estimate for ERM Program

To account for special features in the ERM baseline program as given in Figure 130, several cost elements must be added to the low estimate: $15.6M for the first flight hardware and $5M to assemble the systems on the developmental and qualification test articles into a second flight unit.

Since the range of factor values in the cost estimating relationships are defined over a range with an 80% confidence internal, the high $89M and low $44M parametric cost estimates also approximate an 80% confidence internal range.

• MILLIONS OF 84 DOLLARS

ESTIMATE	LOW	HIGH
DDT&E	$23.6	$64.6
FIRST FLIGHT HARDWARE	$15.6	$24.4
REFURBISH DEV & DUAL TEST ARTICLES INTO SECOND FLIGHT HARDWARE	$ 5	INCLUDED
TOTAL COST RANGE	$44	$89.0
PROBABILITY THAT THE ACTUAL COST WILL LIE WITHIN THIS RANGE IS 80%	← 80% →	
PROBABILITY THAT THE ACTUAL COST WILL BE LESS THAN THE LOW ESTIMATE	10%	
PROBABILITY THAT THE ACTUAL COST WILL EXCEED THE HIGH ESTIMATE		10%

Figure 130. Parametric Cost Estimate - Basline Program

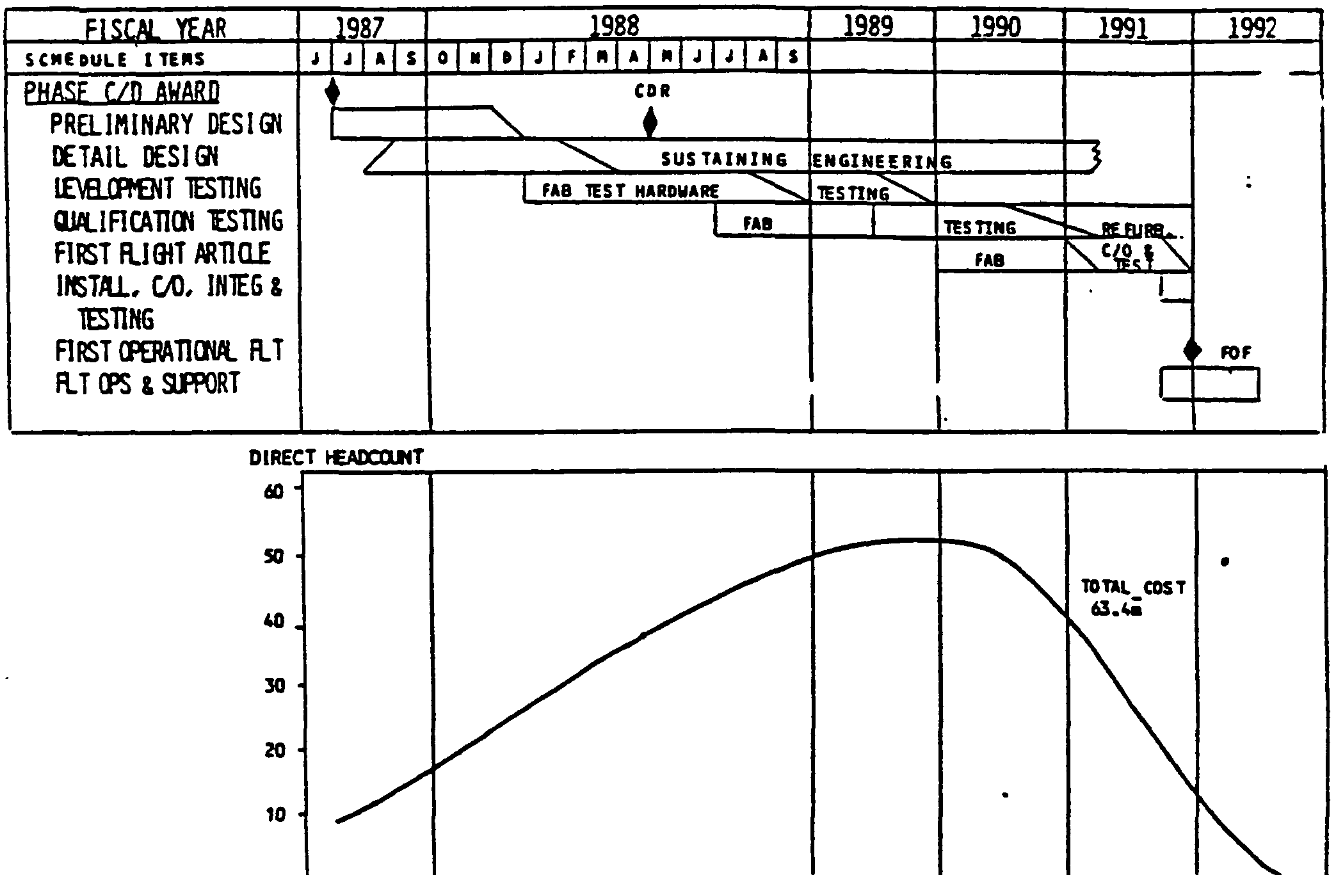
<u>Detailed Engineering Cost Estimate for the ERM Baseline Program</u>

This cost estimating method is based on a program funding curve that has a definitive start, a slope rising to a peak and falling thereafter, and concluding with a definitive finish as illustrated in Figure 131. The ERM program funding profile was estimated from direct personnel expected to support this program. With a direct dollar labor rate and an estimate for the portion of the program that is subcontracted, the total cost was estimated at $64M (Figure 132).

The total direct hours, the direct labor rate, and the percentage of work subcontracted is uncertain and their estimates vary over a range of likely values. The set of worst and best conditions determines the high-and-low-cost estimate, respectively. The combination of high estimates for labor hours, labor rate, and portion of the effort subcontracted results in a high cost estimate of $100M for the ERM program. The combination of low estimates for labor hours, labor rate, and portion of the effort subcontracted results in a low-cost estimate of $45M for the ERM program.

<u>Expected Cost - Best Estimate</u>

The cost estimate range for both the parametric and detailed engineering methods were calcuated so that there was only a 10% chance that the actual cost would be outside the ange, either above the high or below the low. Thus, there is an 80% confidence/probability that the actual cost will be between



	1987	1988	1989	1990	1991	1992
HEADCOUNT REQUIREMENTS	- 12 15 18	21 24 27 30 33 36 38 41 43 46 48 49	51 52 52 52	51 50 47 42	35 28 20 14	8 4 -
ANNUAL DIRECT LABOR HRS	7800	75,600	107,600	98,800	50,400	6,200
TOTAL COST (M$) 55 DIRECT 70% SUBCONTRACTED	$1.4	13.9	19.7	18.1	9.2	1.1

Figure 131.
Detailed Engineering Cost Estimate for ERM Program

the low and high cost estimate. The probability distribution between the high and low cost estimate is spread by a Beta Distribution. A 65/35 Beta Distribution was selected to reflect the fact that the cost distribution is skewed upward. The expected cost is the estimate at the 50% confidence/probability point by definition. Both the parametric ($62M expected cost) and the detailed engineering ($66M expected cost) methods led to similar cost projections as shown in Figure 133. The midpoint of these two expected cost projects - $64M - is our best estimate for the cost of the baseline ERM program. Thus, there is a 50% chance that the program could be less than $64M, but there is also a 50% chance that the baseline ERM program could be greater than $64M. Furthermore, there is only a 10% chance that the baseline ERM program cold be completed for $45M, and there is only a 10% chance that the baseline ERM program could cost more than $95M.

• MILLIONS OF 84 DOLLARS	RANGE COST ESTIMATE		
	LOW ESTIMATE	BEST ESTIMATE	HIGH ESTIMATE
TOTAL DIRECT HOURS	311,760 ← -10%	346,400	+20% → 415,680
% EFFORT SUBCONTRACTED (MATERIAL & PURCHASED PARTS)	65%	70%	75%
DIRECT LABOR RATE (FRINGE BENEFIT & OVERHEAD & G&A)	\$50/HR	\$55/HR	\$60/HR
TOTAL ERM PROGRAM COST	$\frac{311,760 \times \$50}{.35} = 45M$	$\frac{346,400 \times \$55}{.30} = 64M$	$\frac{415,680 \times \$60}{.25} = 100M$

Figure 132.
Detailed Engineering Cost Estimate Range

The \$64M best estimate for the ERM baseline program was allocated to its main elements so that other program variations can be costed. Some other program variations are offered in Figure 134. For instance, if the first flight hardware is removed from the baseline program, the expected cost would drop by nearly \$18M, down to \$46M. The first operational flight could still occur that last quarter of FY 1991, if the off-limits testing of the qualification test article was eliminated. The flight article would be assembled from the system salvaged from the developmental qualification, and flight demonstration test articles. This flight article would be available for functional testing and check-out. On the other hand, the production of flight units 3 and 4 could be included in the baseline program for an additional \$35M and a total program cost of \$99M.

The time-phased funding profile for the expected program cost of \$64M was calculated from the detailed engineering cost estimates. Peak funding occurs in FY 1989 at \$20M as shown in Figure 135.

ERM Production Cost

A 91% learning rate is possible for follow-on production units. The \$35M cost estimate for production units 3 and 4 assumes they are integrated with the baseline program so that material part/systems procurement, manufacturing set-up, and learning are not lost. Again, costs are estimated over a range from high to low. These costs were estimated parametrically in Figure 136.

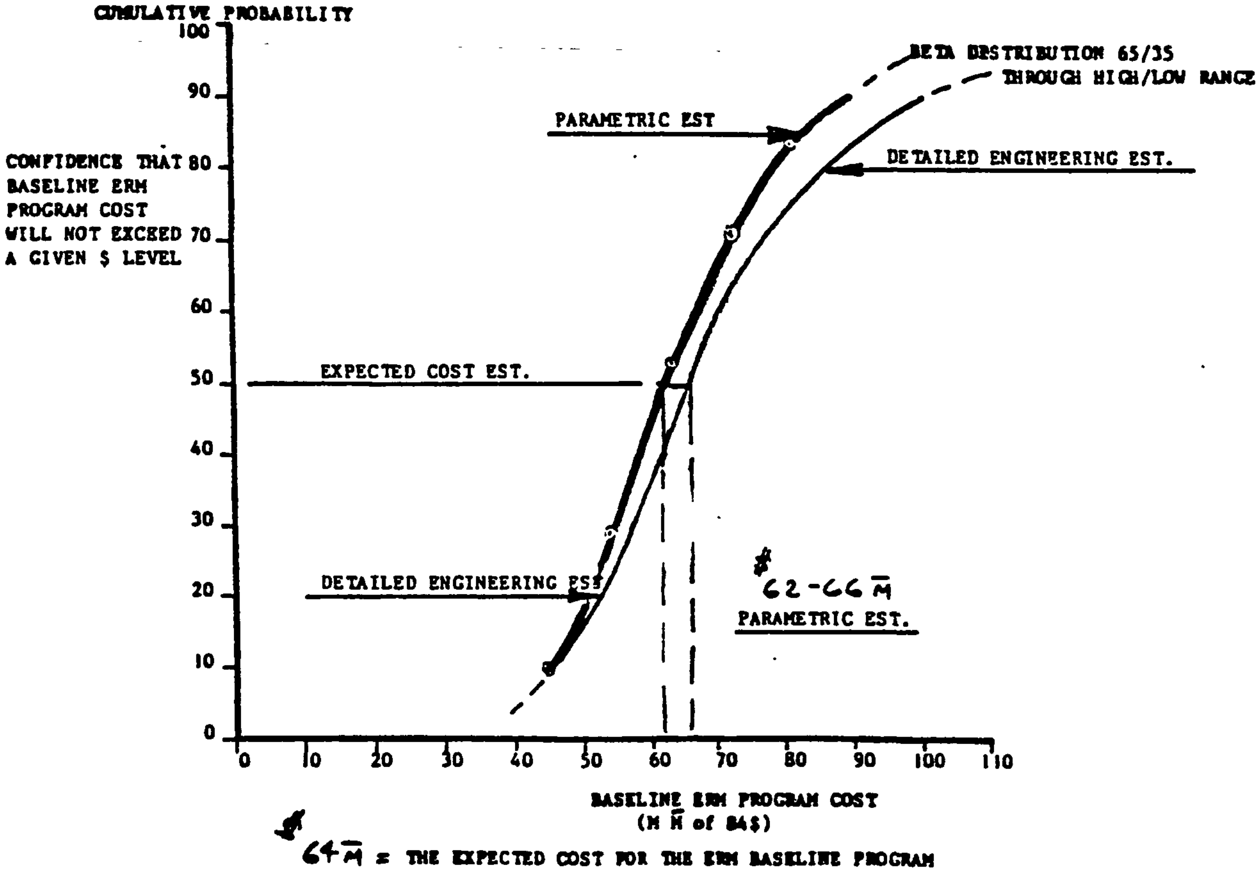


Figure 133. Confidence/Probability of Reaching Baseline ERM Program Cost Target

3.3.10 Cost Estimate for Satellite Mod Kit to Enable Resupply

This cost estimate was computed both by parametric and by detailed engineering techniques as shown in Figure 137. The parametric cost estimates were calculated by adding the scar weight of the mod kit to a 2,000 pound satellite and computing the increase in cost from the cost estimating formulas.

There is reasonably good agreement between the parametric and detailed engineering cost estimate range for DDT&E. However, the production cost range estimated by the parametric method is low because the additional parts in the mod kit include expensive quick disconnects and valves that are more expensive than the average plumbing parts in the cost estimating formula. Thus, the detailed engineering cost estimate more nearly represents the true cost of the mod kit.

The expected DDT&E costs estimated by parametric and detailed engineering methods, is between $1M to $1.25M as shown in Figure 138.

ERM BASELINE PROGRAM	EXPECTED COST
• DDT&E	41M
• REFURBISH SYSTEMS ON DEV & QUAL TEST ARTICLES INTO A SECOND FLIGHT HARDWARE	5M
• FIRST FLIGHT HARDWARE, C/O FUNCTIONS & SHUTTLE INTEGRATION	18M
• GSE & PAYLOAD SUPPORT	-
TOTAL EXPECTED COST	64M

ERM EXTENDED PROGRAM	EXPECTED COST
• PRODUCTION OF FLIGHT UNITS 3 & 4	35M
• MISSION OPERATIONS & TRAINING	NOT COSTED
• MAINTENANCE & REFURBISHMENT - SPARES	NOT COSTED
• PROGRAM MANAGEMENT & SUSTAINING ENGINEERING (LESS THAN 1M)	NOMINAL

Figure 134. Cost Breakdown for the ERM Baseline Program

The production cost range, estimated parametrically, was assumed to reflect the lower half of the parametrically confidence interval (below 50%). This resulted in a new upper cost estimate of nearly $1,000,000 instead of $770,000 originally calculated. Still, the spread between the expected costs estimated by parametric formulas and detailed engineering is wide, $770,000 to $1,240,000, reflecting more uncertainty in this cost estimate. We believe the true cost is towds the upper end, however.

Summary

If the user/customer can be influenced to support/commit to the ERM program without an early flight demonstration, we recommend that the flight demonstration be incorporated within the ERM program and fly a year later, if at all.

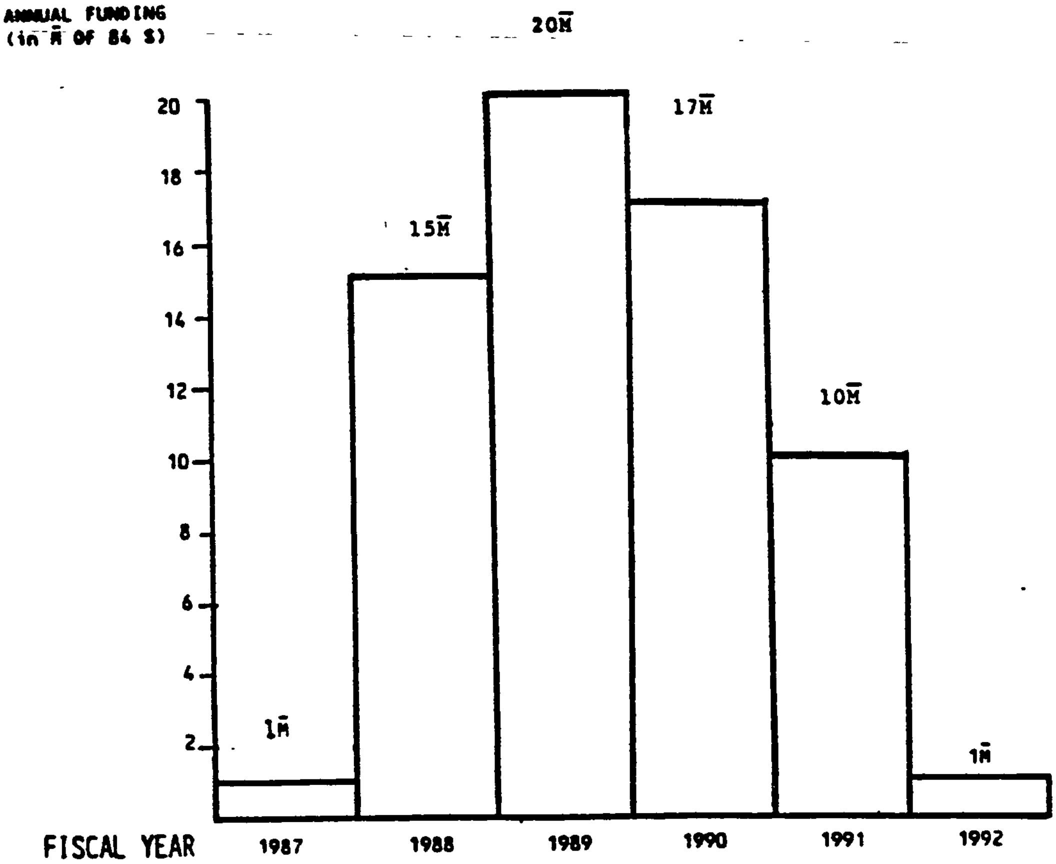


Figure 135. Time Phased Cost Estimate for Baseline ERM Program

The expected cost of the flight demonstration program can be kept to $5M if: 1) performed in the orbiter payload bay, 2) the MPESS and SPAS structures are used, 3) all major systems are salvaged for reuse on the first flight unit, and 4) the critical technology components don't need to be redesigned.

The ERM baseline program consists of two test articles (development and qualification) and two flight units.

The second flight unit is assembled from major systems salvaged from the development and qualification test articles, thus cutting program costs by $12 to 18M.

The expected program cost is $64M with a peak funding of $20M in 1989 if the phase C/D starts in the fourth quarter of FY 1987 and the first operational flight occurs in the last quarter of FY 1991.

The average expected cost to produce additional ERM's is $18M per unit.

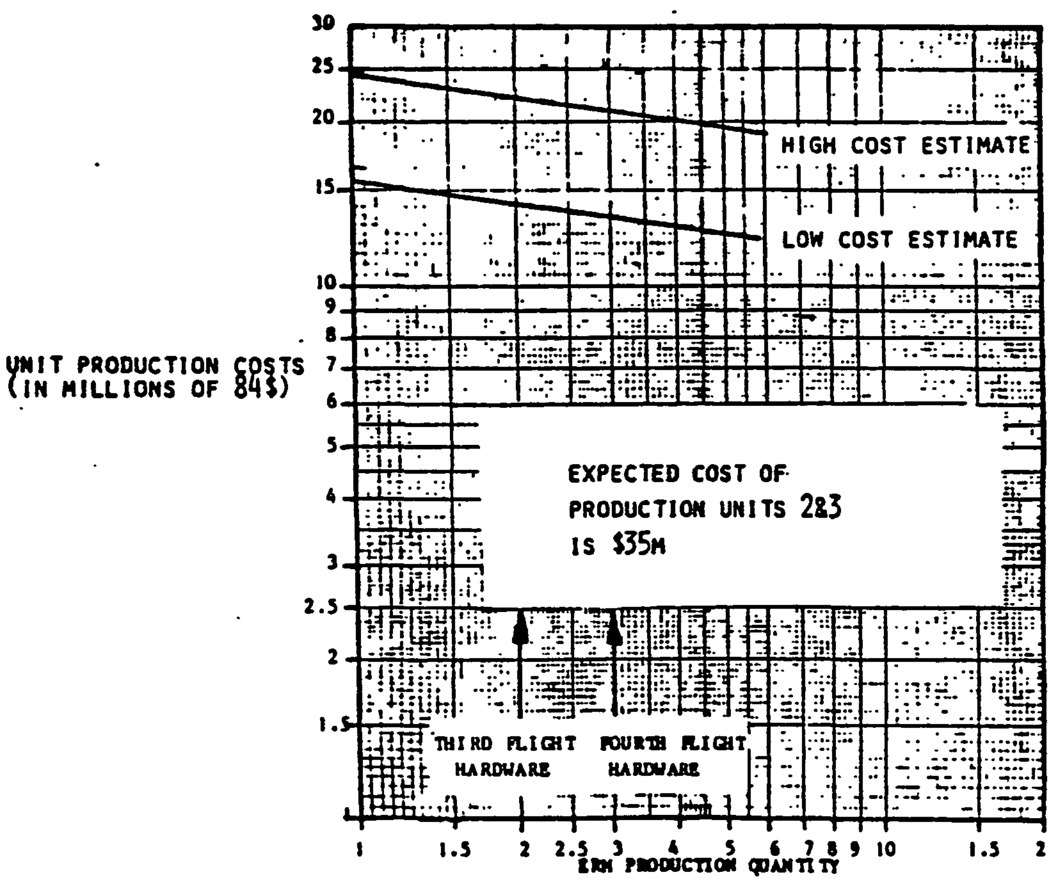


Figure 136. ERM Production Costs

	PARAMETRIC		DETAILED ENGINEERING	
	LOW	HIGH	LOW	HIGH
DDT&E	DESIGN COMPLEXITY=.3 $3.3\bar{M}$ x .3 = $990,000	DESIGN COMPLEXITY=.5 $3.3\bar{M}$ x .5 = $1,650,000	$50/HR x 10,400 HRS .75 = $690,000	$60/HR x 15,600 HRS .60 = $1,560,000
	COST ESTIMATING RELATIONSHIP EXTRA PLUMBING COST ESTIMATE FROM - USAF UNMANNED S/C GRAPPLE TRUSS, DOCKING FIXTURE & VALVE PANEL - NASA-PRC			
PRODUCTION COSTS FOR TFU	PRODUCTION COMPLEXITY .7 $.77\bar{M}$ x .7 = $540,000	PRODUCTION COMPLEXITY 1.0 $.77\bar{M}$ x 1.0 = $770,000	QUICK DISCONNECT 5x75,000=$375,000 VALVES 9x40,000=$360,000 ELECT. CONNECTIONS 2x40,000=$80,000 GRAPPLE FIXTURE & PANELS $160,000 ASSY $100,000 TOTAL $1,075,000	QUICK DISCONNECT 5x100,000=$500,000 VALVES 9x50,000=$450,000 ELECT. CONNECTIONS 2x50,000=$100,000 GRAPPLE FIXTURE & PANELS $200,000 $150,000 TOTAL $1,400,000

Figure 137. Cost Estimate for Satellite Mod Kit to Enable Resupply

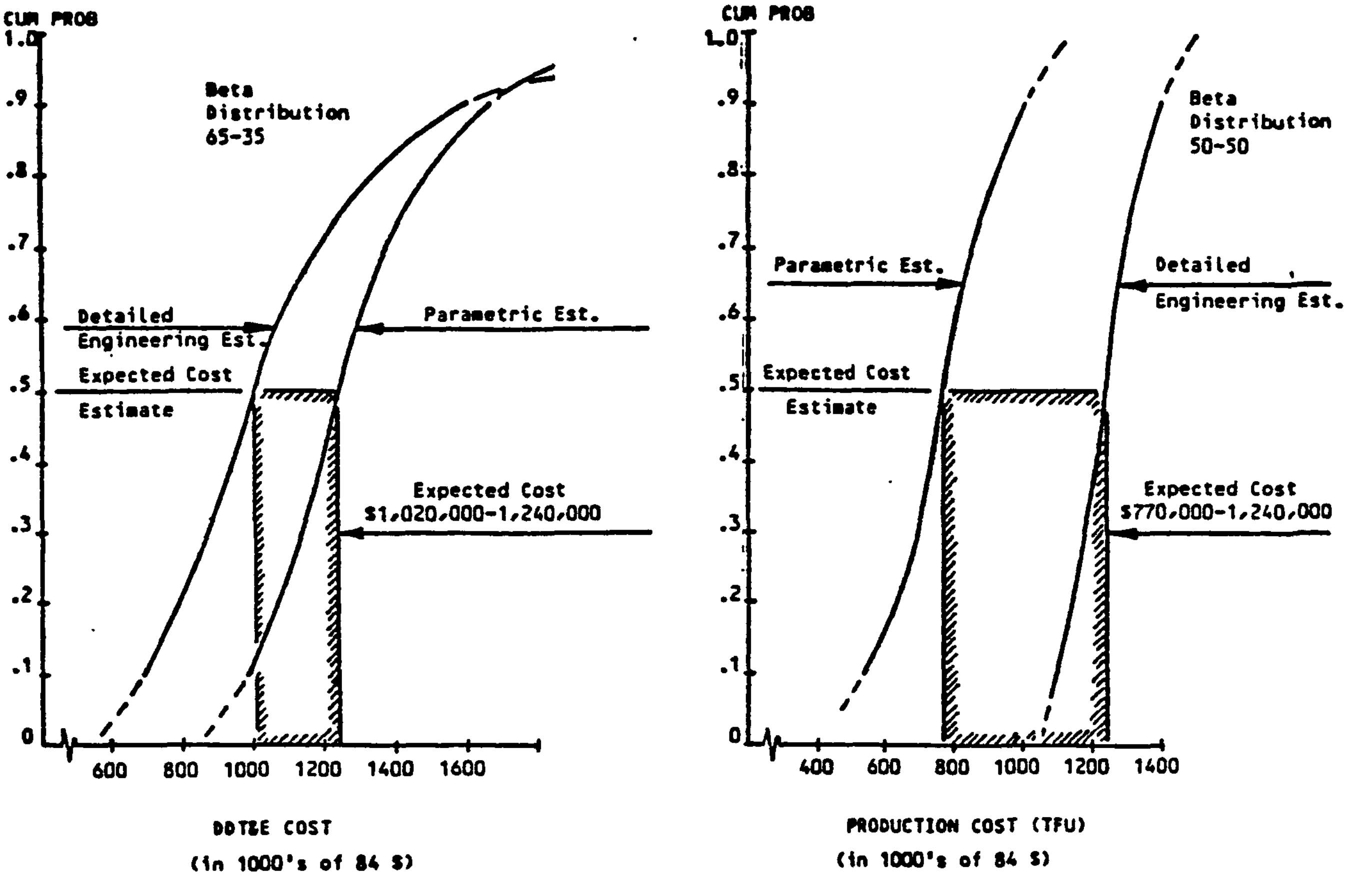


Figure 138. Expected Value Cost Estimate for Satellite Mod Kit to Enable Resupply

Appendix

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Rockwell International

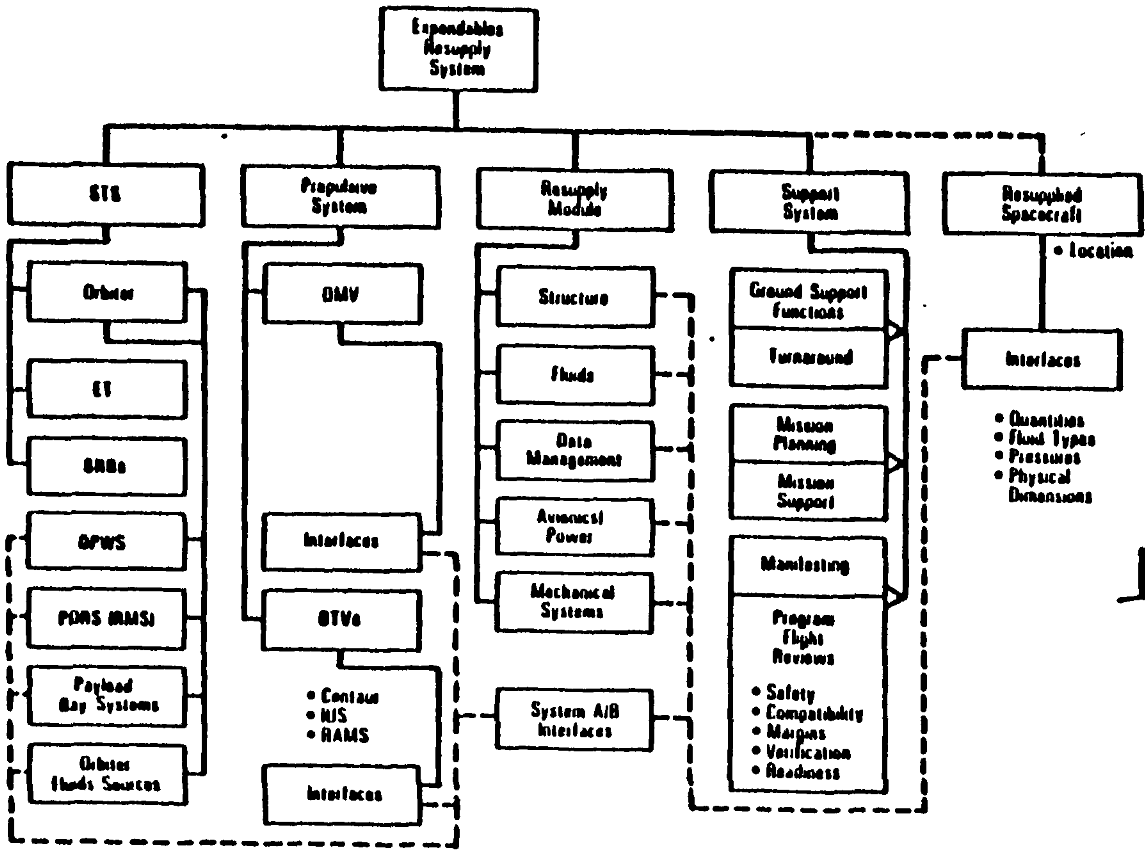
EXPENDABLES RESUPPLY MODULE

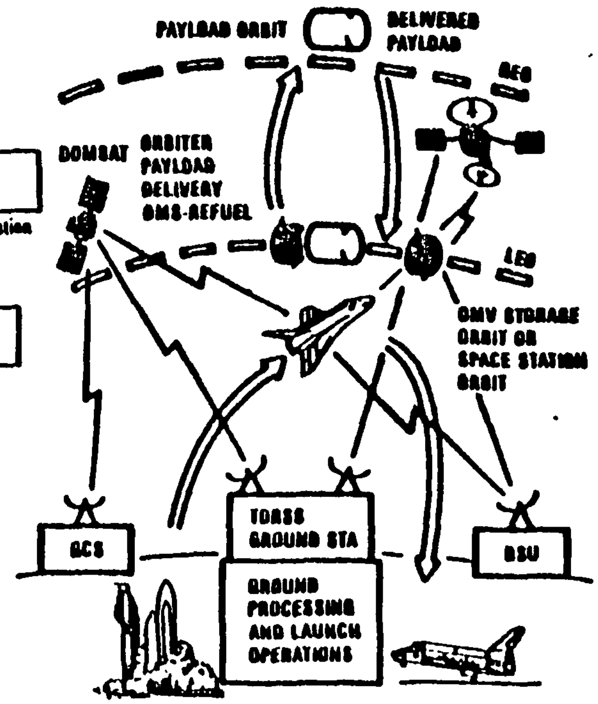
REQUIREMENTS DEFINITION DOCUMENT

ERM SYSTEMS INTERF 'E REQUIREMENTS

INTERFACE-SYSTEM	OPERATIONS	REDUNDANCY/AUTONOMY	REQUIREMENTS	CONSTRAINTS
RECEIVER SATELLITE	O FLUID TRANSFER RECIPIENT	O RECEIVER CAN MONITOR & DOWNLIST DATA O DATA CAN PROVIDE THERMAL CONTROL RECEIVER	O STANDARD RECEIVER UMBILICAL I/F O MONITOR RESUPPLY OPS, STATUS & MEASUREMENTS OF RECEIVER	O PASSIVE GNC O CONTROL ADIABATIC HEATING O LOW-G FOR FLUID TRANSFER
CARRIER VEHICLES	o GN & C (INCLUDES DOCKING) o COMMUNICATIONS	EXTERNAL AUTONOMY EXTERNAL AUTONOMY	o TRANSPORT & DOCK ERM TO RECEIVER o UPLINK & DOWNLINK ERM COMMANDS & DATA	o M-I-L DOCKING CONTROL W/TIME LAG O OMV CONSTRAINED
ORBITER	o LAUNCH/ENTRY VEHICLE PAYLOAD OPS o SYSTEM MONITOR o SUPPLY ERM FLUIDS	O OPTION TO GCS	o PROVIDE LAUNCH & RETURN CAPABILITY o PROVIDE POWER SEP & RMS HANDLING o CONTROL PANEL FLUID TRANSFER & MONITOR	O GROUND BASED OMV o ORBITER P/L SUPPORT O COMM REQUIRED
SPACE STATION	o CARGO OPS o ERM & TANK STORAGE o SYSTEM MONITOR	O OPTION TO GCS	o PROVIDE HANDLING CAPABILITY o TANK FARM TRANSFER & RESERVICING o CONTROL PANEL	O SPACE BASED OMV CAMPATIBILITY W/CARGO BAY O COMM REQUIRED
GSE	o GROUND SERVICING		o FLUID FILL & DRAIN GROUND HANDLING	O COMPATIBLE W/GSE FLUID LOADING & PAYLOAD OPERATIONS
GCS	O SYSTEM MONITORING	O OVERRIDE ERM CONTROL CAPABILITY	O MISSION SUPPORT CONTROL PANEL COMMUNICATION LINK SYSTEM CHECKOUT	O COMM REQUIRED
COMM SATELLITES	O COMMUNICATION LINK VIA OMV COMM SYSTEM	O OMV SYSTEM		O OMV LIMITS

System Definition





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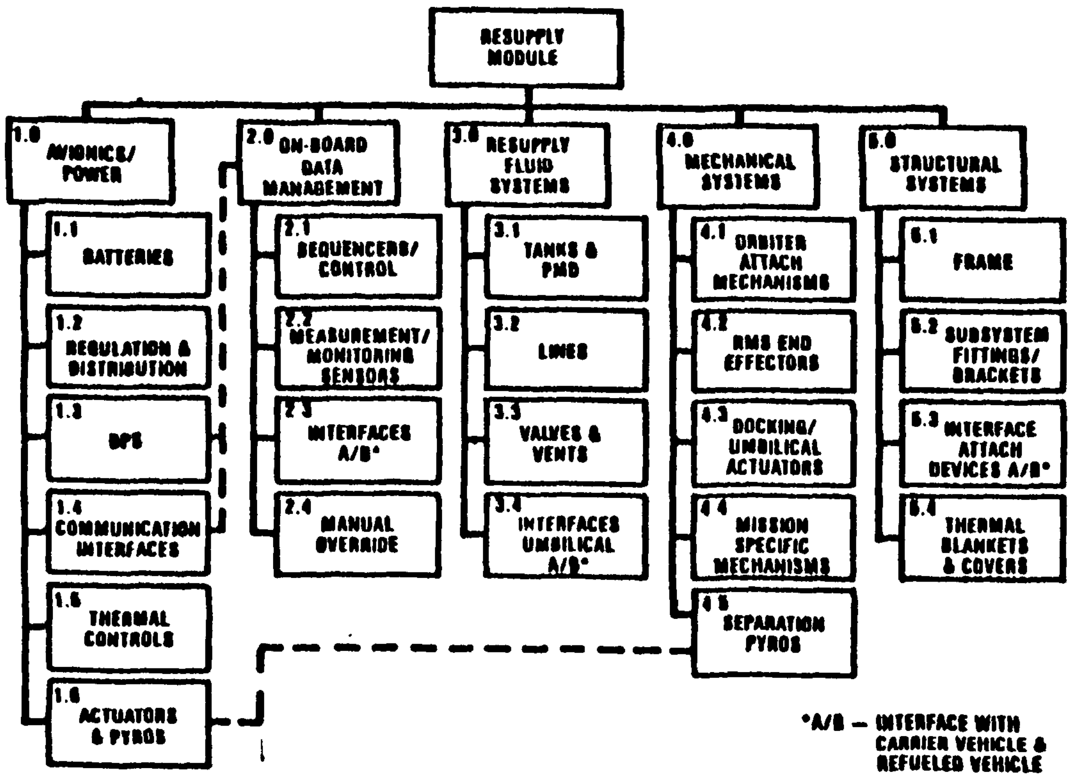
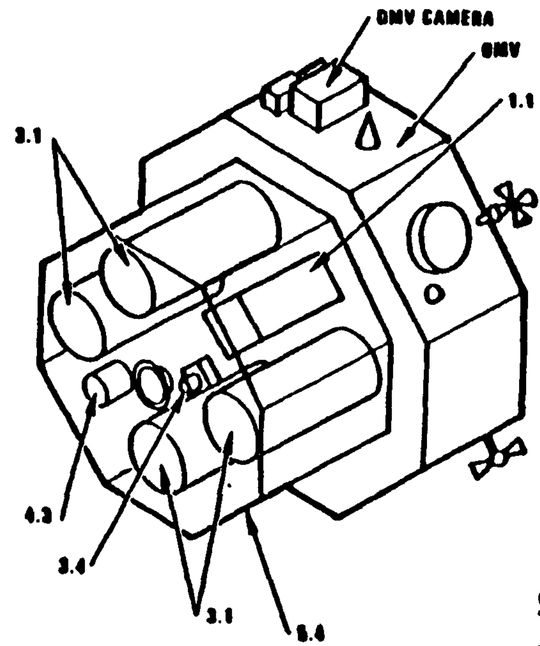
74SSV152535

ERM Subsystems Identification



*A/B — INTERFACE WITH CARRIER VEHICLE & REFUELED VEHICLE



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1.0 AVIONICS & POWER SUBSYSTEMS

1.1 Batteries

1.2 Regulation and Distribution

1.3 DPS

1.4 Communications Interfaces

1.5 Thermal Controls

1.6 Actuators and Pyros

1017e/5

1.0 AVIONICS & POWER SUBSYSTEMS DEFINITION

The avionics and power subsystems shall provide hardware and software necessary to perform and control the following functions and operations:

- o Communication Processin
- o EPDC
- o Fluid Transfer Operations
- o Data Processing
- o Active Thermal Conditioning
- o Umbilical Attach and Detach

Resupply mission functions which are performed by ERM system interface elements are:

Communication - OMV subsystem

GN & C (inc. docking) - OMV subsystem

The avionics/power subsystem requirements are shown in tabular form in Figure 1a.

1.1 Batteries

The power generation system aboard the ERM shall provide the electrical power requirements for all operational phases of the resupply mission. During quiescent phases of the resupply mission power may be provided in the powered down mode.

Electrical power will provide for the following subsystems: data processing, active thermal control, actuators, and fluid storage and transfer.

When in the cargo bay power will be provided from the Orbiter through the cargo interface.

Backup power to the ERM could be provided by the OMV carrier vehicle thru an existing power interface necessary to power the OMV docking camera and lights mounted on the ERM.

1.2 Power Regulation & Distribution

The Power Regulation and Distribution subsystem shall be capable of regulating both internal and external power generation. The autonomous internal power generation shall provide for an operational and quiescent mode. The quiescent power down mode shall provide for system and status monitoring and provide for active thermal control during non-transfer operations. During the operational phase, power will be required to operate valves and actuators, DPS and thermal control during the transfer.

1.3 Data Processing Subsystem

The data processing subsystem shall provide hardware and software capabilities for system performance monitoring on board and ground checkout subsystem sequencing and master timing.

The software shall be capable of supporting in conjunction with GSE software pre-mission system and interface checkout. The DPS shall be able to request and accept digital data, process allorithms and generate necessary output commands for all ERM subsystems.

Data Management Subsystem, Section 2.0, will further elaborate on the on-board data management and interface requirements.

1.4 Communication Interfaces

The Communications Processor shall provide the interface with the Carrier's Communication and Tracking Subsystem via the DPS. It shall process ground commands and format telemetry data for communication uplink and downlink. The required RF signals shall be compatible with the Carrier's telemetry transmission and/or reception links including direct link and TDRS/DOMSAT relay.

The communication processor must also provide interface with the Orbiter's communication and tracking subsystem via a payload interface. It shall be compatible with the Orbiter's telemetry transmission and/or reception links.

Certain operational functions or sequence initiation may require command control or monitoring by an operator. Communication coverage regions and times must be understood prior to critical ERM operations.

1.5 Thermal Control

The thermal control subsystem is conjunction with the passive thermal protection system shall maintain within acceptable limits through all mission phases, the thermal environments, plume environment and/or temperatures of all subsystems and components. This shall be accomplished by utilizing techniques such as thermal insulation, coatings, thermal isolators, heat sinks, heat conduction paths, radiators, and heat sources. Prelaunch and post-abort landing ground thermal conditioning of the vehicle shall be employed when possible to assist the thermal control system in performing its functions.

Numerous subsystem alternatives exist for thermal control of earth storable propellants. The most common are thermostatically controlled heaters, insulation, coating, isolators and heat sinks. Typical temperature limits for all mission phases are 40°F to 120°F.

Cryogenic thermal control is more difficult to resolve. Numerous concepts have been developed for thermal control of cryogenic fluids. Concepts that employ integrated structure and cryogenic insulation with multilayer radiation shield is a generally accepted approach. Super insulation may be used in evacuated or purged systems. However, current studies indicate that the uncompressed evacuated system has the lower coefficient of thermal conductivity.

1.6 Actuators & Pyros

The actuators and pyros shall control docking and umbilical connects and disconnects. Electro/Mechanical actuators control the fluid distribution and isolation system for transfer and safing operations. The pyros can insure emergency separation of failed stack docking and mechanical interfaces.

See Section 4.3 and 4.5 for further requirements on actuators and pyros, respectively.

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
		SUBSYSTEMS	REQUIREMENTS	
1.0 AVIONIC/POWER	1.1 BATTERIES	O EPDC O DPS	~ 40 KWH IN 4 MODULES	O POWER GENERATION WHEN DETACHED FROM ORBITER POWER SUPPLY
	1.2 POWER REGULATION & DISTRIBUTION	O BATTERY O ACTIVE THERMAL CONTROL ELEMENTS O DPS O ACTUATORS, SENSORS, PYROS O RECEIVER INTERFACE	O COMPATIBLE WITH ORBITER P/L POWER SUPPLY 28VDC	O REGULATES & DISTRIBUTES BATTERY POWER TO THE DPS, VALVES, ACTIVATORS THERMAL CONTROL ELEMENT DURING QUIESCENT & OPERATIONAL MODES O SUPPLIES POWER TO RECEIVER FOR TRANSFER OPS
	1.3 DPS	O EPDC O ON ORBIT DATA MGMT - THERMAL - COMM - SEQUENCERS - MEASUREMENTS - RECEIVER SYSTEM	O MODULAR/DISTRIBUTED BUS-ORIENTED 64K RAM/64K ROM/CPU 32 BIT MPU 40 MSEC MINOR CYCLE	O ON ORBIT DATA MGMT OF POWER, MECHANICAL & RESUPPLY SYSTEMS O ISSUES COMMANDS & PROCESSES DATA FROM SENSORS & INITIATES FUNCTIONAL SEQUENCES FOR NOMINAL & CONTINGENCY OPS
	1.4, COMM. INTERFACES	O CARRIER COMM SYSTEM O DPS	O 1 K BAUD (OMV COMPATIBLE)	O PROCESSES GROUND COMMANDS & FORMATS TELEMETRY DATA FOR COMMUNICATION UPLINK & DOWNLINK VIA CARRIER COMM SYSTEM
	1.5 THERMAL CONTROL	O FLUID TRANSFERY SYSTEM O DPS O EPDC	O THERMAL CONTROL & CONDITION ALL STORABLE CRYO & GASEOUS FLUIDS	O PROTECTS STRUCTURE & SUBSYSTEMS FROM SPACE & PLUME ENVIRONMENTS
	1.6 ACTUATORS & PYROS	O DOCKING MECHANISM O INTERFACE UMBILICALS O EPD & C O DPS O FLUID DISTRIBUTION SYSTEM	O PROVIDE HIGH RELIABILITY ACTUATORS O SAFE SEPARATION PYROS	O CONTROL DOCKING & UMBILICAN CONNECTS O EMERGENCY SEPARATION O CONTROLS FLUID DISTRIBUTION & ISOLATION SYSTEM FOR TRANSFER & SAFING OPS

FIG. E 1a

2.0 ON BOARD DATA MANAGEMENT

2.1 Sequencers/Control

2.2 Measurement/Monitoring Sensors

2.3 Interfaces

2.4 Manual Override

2.0 ON BOARD.DATA MANAGEMENT SUBSYSTEM

The data management subsystem provides control of all events and sequencing in the RM itself down to the component actuation level.

System and status monitoring fault detection and automated malfunction procedures provide for assurance of subsystem health, while maximizing mission success probability.

In addition, it controls the data uplinked and downlinked via the OMV related to communication of engineering data, status data and the automatic or man-in-the-loop control data relayed through the system met by way of TDRSS and DOMSAT for extended coverage.

The Data Management Subsystem intefaces with the Receiver to monitor and control critical fluid subsystems.

Figure 2a shows in a tabular format the onboard data management subsystem requirements.

2.1 Sequencers/Control

System and status monitoring will check the general health and safety of the ERm during quiescent and operational phases. Leakage for propellants and pressurants, and system pressures and temperatures will be statused as being within operational limits. During quiescent periods constant confirmation and computer self monitoring should determine system safety, heater operation, and fluids and components are within acceptable limits.

The operational status of the fluid transfer system should be verified upon disconnect engagement by the leakage indicator, pressures at critical junctures, valve position indicators, temperatures of the fluids and components. Upon disconnect engagement leak checks should be performed. Electrical path checks should be verified with the electrical umbilicals. During system operation, flow, pressure drops, pressure and temperature measurements will be used to determine satisfactory system performance and the transferred rate and quantity. Durin transfer using the ullage recompressin method for hydrazine, ullage temperatures must be continuously monitored to control the adiabatic compressure heating of the ullage to within acceptable limits. Sufficient instrumentation should be added to provide override protection and process termination in the event of a malfunction.

The completion and safing monitoring of the transfer process will require sufficient pressure and leak detector measurements to evaluate the purging, depressurizing and demating actions. Prior to system safing sub-system and component valve positions, pressurs and temperatures need to be verified.

The process of fault detection resulting in system shutdown should be automatically controlled by an on-board computer. Computer controlled and monitored operations for transfer operations and fault detection with automatic shutdown will provide the most rapid response capability for remote operations. The operator monitoring the activity can then access the situation and determine to restart the transfer operation.

2.2 Measurement/Monitoring Sensors

The instrumentation subsystem shall sense, acquire, condition, digitize, format and distribute data from the RM subsystems for telemetry, checkout, performance monitoring and GSE. Subsystem measurements include temperatures, pressures, strains, acceleration, currents, voltages, events, flow, position, quantity, rates and electrical power.

Tank quantity gaging and flowmetering are the two principle means of determining the quantity of propellant transferred during refueling. Although tank gaging does not directly measure the transferred amount (but infers it from tank quantity), it has the advantage of directly verifying total tank load, which is important for mission confidence. A flowmeter, by itself, cannot do this unless tank load is accurately known prior to transfer. In the case of a spacecraft returning from its mission, this is not known unless a tank gaging system is provided, or the spacecraft tanks are off-loaded to zero quantity before transfer.

PVT is the only state-of-the-art zero-gaging method which has reached an acceptable state of hardware development and flight experience.

The various resupply fluids and transfer systems would warrant the need for both a tank gaging system and a fluid transfer measurement. Integrating flowmeters are the only method available to measure fluid transferred if a quantity gaging system was inoperative (i.e., during an ullage transfer fluid resupply operation). Since an RM should be able to resupply fluid utilizing any of the fluid transfer methods, it is recommended that both types of propellant measurement systems be incorporated.

2.3 Interfaces

The receiver satellite functions and subsystem status will need to be monitored.

The satellite control center will configure the receiver satellite for resupply operations. Confirmation of the receiver satellite systems status (e.g. attitude control) will be required prior to initiating the transfer operations.

Prior to fluid transfer a fluid quantity status of the receiver is necessary to know resupply quantity requirements. During transfer operations critical temperatures and pressures aboard the Receiver must be monitored for accurate resupply quantity determination and to prevent excessive adiabatic heating during an ullage recompression transfer. See Section 3.0-D for further description of receiver satellite requirements.

For transfer of OMV bi-propellants, a DPS interface will be required to measure critical pressures and temperatures and status the various subsystems.

2.4 Manual Override

The operation of the resupply system is envisioned to include operator controlled or monitored activity at an Operators' Control Panel (OCP). The OCP would have function indicators, position indicators, some critical pressure and temperature readouts for monitoring system health, pressures, temperatures, flows and quantities at the various mission phases. Emergency operation termination should be provided at the OCP which would safe all fluid flow circuits.

SUBSYSTEM REQUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
2.0 ONBOARD DATA MGMT	2.1 SEQUENCERS/CONTROL	o FLUID TRANSFER SYSTEM o THERMAL CONTROL SYSTEM - QUIESCENT MODE - OPERATIONAL MODE o DOCKING/UMBILICAL ACTUATORS o RECIVER SEPARATION o PURGE & VENT SYSTEM	o2 SEQUENCERS	oSOFTWARE RUNNED FUNCTIONAL SEQUENCES FOR SYSTEM CHECKOUT - INTERFACE CHECKOUT - TRANSFER OPERATIONS - QUIESCENT MODE - OPERATIONAL MODE - AUTOMATED MALFUNCTION PROCESS
	2.2 MEASUREMENT/ MONITORING SENSORS	o FLUID TRANSFER SYSTEM o THERMAL CONTROL SYSTEM o DOCKING SYSTEM o DPS	o2-WAY FLOW METERS oACCURATE GAGING SYSTEM oRECEIVER MONITORING OF CRITICAL SYSTEMS	oGAGES, X-DUCERS SENSORS TO MEASURE ALL SUB-SYSTEMS AND OPERATIONS
	2.3 INTERFACES - RECEIVER	o DPS o RECEIVER TANK STATUS - THERMAL - MEASUREMENT - POWER	oSTANDARD I/F CONNECTION	oPROVIDE FOR RECEIVER STATUS & MONITORING
	- CARRIER	o COMMUNICATIONS o FLUID TRANSFER (FROM CARRIER) o BACKUP POWER	oSTANDARD I/F CONNECTION	oPROVIDE FOR COMM FOR UPLINK/DOWNLINK PROCESSING
	2.4 MANUAL OVERRIDE	o ERM CONTROL CENTER o DPS o COMM PROCESSOR	• CAPABILITY DURING CRITICAL MISSION PHASES FOR MONITORING AND CONTROL	oEMERGENCY OVERRIDE CAPABILITY

FIGURE 2a

3.0 RESUPPLY SYSTEM

3.0-A Fluid Delivery & Resupply Operations

3.0-B Tanks & PMD

3.0-C Contamination Control

3.0-D Receiver Spacecraft Design Considerations

3.1 Tanks & PMD

3.2 Lines

3.3 Valves & Vents

3.4 Interfaces Umbilicals

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3.0 RESUPPLY FLUIDS SUBSYSTEM INTRODUCTION

Resupply fluid subsystem requirements are very much dependent upon the nature of the resupply fluid, mission objectives, receiver resupply requirements and the fluid acquisition system.

This section will discuss the operational flow of the fluid transfer storable and cryogenic fluid transfer techniques, pressurant transfer techniques, propellant management devices (PMD), contaminant control, Receiver spacecraft interface requirements.

Figures 3a, 3b and 3c summarize in tabular form the resupply fluid subsystem requirements for OMV bi-propellant, hydrazine and super-fluid Helium transfer, respectively.

To effectively discuss the resupply fluids subsystem requirements, a narrative format was chose to present this section.

3.0-A Fluid Delivery and Resupply Operations

This section will cover the unique requirements for various resupply fluids including, monopropellants, bipropellants pressurants, cryogenics and special fluids.

Resupply scenarios include ERM ground and space basing. Space basing could require fluid or tank changeout at the Orbiter or at a Space Station. Orbiter scavenging of OMV bi-propellants or MPS cryogenics residuals would make the ERM a candidate for those fluids. OMV bi-propellants could be transferred through an RM (without tanks) to satisfy receiver bi-propellant candidates.

3.0-A.1 Storable Fluid Acquisition and Transfer

Methods to be considered for resupply of storable propellants should include: ullage recompression, ullage displacement, and ullage vent/repressurization. These should be considered on the basis of resupply fluid and ullage gas, mechanical and operational complexity, time and safety considerations. Each refueling approach is discussed in the next few paragraphs. Highlights of these methods are summarized and schematics provided in figure 3d.

- Ullage Recompression

In this approach, the pressurant gas either remains in the propellant tank or is returned to gas storage bottles during the propellant resupply. Propellant is transferred against an increasing ullage pressure as the incoming propellant compresses the ullage gas. This creates adiabatic heating which must be regulated to permit heat dissipation. This method is primarily applicable to resupplying systems operating in blowdown. The propellant will most likely be resupplied from a regulated pressure propellant tank system. Therefore, the quantity of propellant transferred can be determined using a tank quantity gaging system. It will be necessary however to know the initial load in the receiver tank so that the quantity to be transferred and the final load is known. Therefore, receiver tanks should also incorporate a tank gaging system.

- Ullage Vent/Vacuum Fill

This type of propellant transfer will be necessary on regulated pressure systems with tanks that do not have the capability to position the ullage bubble. This is an undesirable method of transfer because it will probably necessitate a hazardous waste scavenging system be included in the resupply module/extended mission kit. With this method, the receiver tank is vented to vapor pressure, probably to a scavenging system, before transferring propellant. The required amount of propellant is then resupplied, controlled either by flow metering or a supply tank quantity gaging system, at a pressure that need only be great enough to overcome the frictional losses in the refueling lines. The tank is then pressurized collapsing any propellant vapor bubbles trapped in the propellant acquisition device. The critical issue is ga-liquid phase separation to preclude liquid venting. The transfer control system is more elaborate because of variable temperature/ variable pressure conditions. An alternative would be to fill and vent at contant receiver ullage pressure.

- Ullage Transfer

The ullage transfer resupply method will beused for resupplying pressure regulated propellant tanks that have the capability of locating the ullage bubble. This is a constant pressure transfer during which the ullage gas is vented off the receiver tank as propellant is pumped in. The ullage is then transferred to the supply tank to replace the transferred propellant. A system of this type will require a propellant pump to overcome the transfer lines frictional losses. A PVT propellant gaging system will not work on a system of this type. It will therefore be necessary to use flowmeters to calculate the quantity of propellant transferred. Again, a receiver tank propellant gaging system will be required so that the initial propellant load is known. Ullage displacement offers many advantages including the avoidance of pressurant gas loss, but is more complex and therefore useful mainly for the resupply of large capacity tanks, such as OTV's.

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3.0-A.2 Pressurant Resupply

The options for pressurant transfer include high pressure storage supply bottles, moderate pressure source and high pressure pump, or cryogenic storage and conditioned transfer. High pressure storage supply offers the simplest method; however, this results in high residuals after transfer. High pressure ganged bottle sequential transfer has been evaluated to determine the lightest weight and least residual configuration. Moderate pressure source and a high pressure pump should also be evaluated as far as weight, volume and the design complexity of a high pressure pump. Figure 3f shows schematics of the above pressurant transfer techniques.

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3.0-A.3 Cryogenic Fluid Acquisition and Transfer

Acquisition of cryogenic liquids must depend on either a settling acceleration or a CAD. Positive expulsion devices are not viable alternatives. Also, supercritical feedout is not viable due to the high energy requirements and the necessarily high energy state of the fluid. For on-orbit storage and subsequent propellant transfer, the primary areas of concern are (1) Thermodynamic control of the supply tanks, (2) Liquid/vapor interface control, (3) Childown and vent of warm vapor andor inert gases from the receiver tank, and (4) Propellant transfer and fill operations. In a low-gravity environment, prolonged sub-critical storage of the cryogens in the supply tanks is complicated by the lack of positive control of the location of the liquid and vapor phases. Vapor venting must occur to effectively control tank pressure as heat leaks into the tank. A Thermodynamic Vent System (TVS) for hydrogen will insure that only vapor is vented.

Heat is transferred from the stored fluid into the throttled liquid which is supplied to the TVS from a CAD. An internal heat exchanger attached to the CAD would maintain the fluid in the CAD in a subcooled state. Alternatively, an internal TVS with compact heat exchanger and forced recirculation should be considered. An external heat exchanger should be installed at the tank plumbing penetrations to intercept concentrated heat leakage.

An example of a typical cryogenic transfer approach would be:

1. Propellant acquisition in the supply tank provided by simple capillary sponge or vane type devices (Figure _____) in the supply tank rather than use of settling thrust.
2. LH_2 childown boiloff vapors from the receiver tank reabsorbed in the bulk supply by reverse-flow cycling without a vapor return line.
3. Fully autogeneous pressurization of the supply tanks by means of a capillary or pump-driven bootstrap loop eliminates the requirement for helium pressurization.
4. Propellant-swirl techniques in the receiver tank can be used to position the tank ullage space at the tank vent opening, facilitate zero-g gaging of the tank by acoustic resonance techniques, and sweep vapor bubbles from the wall to the vent port thereby reducing or eliminating the need for a thermodynamic venting system and coils.
5. Subcooling of cryogenic propellants by helium bubbling on the Shuttlelaunch pad or active refrigeration at a space base can greatly reduce or eliminate boiloff losses in propellant transfer and on long tankage hold periods (parking orbit, multiple sortie missions, etc.).

3.0-B Fluid & Pressurant Tanks

The fluid tanks selection for the resupply operations must consider the nature of the resupply fluid, mission objectives, receiver satellite resupply requirements and the fluid acquisition system on the receiver satellite. Supply tank compatibility with receiver satellite tanks is discussed in the fluid transfer operations. It is expected that an ERM modular design concept would provide for varying tank size and shaping. Existing qualified tanks could be tailored to particular resupply mission objectives in this manner reduce design/development costs for the ERM. A study of the various fluid acquisition system was conducted.

The results of the study indicate a preference for positive expulsion devices (diaphragms) with those fluids that are compatible with the operational elastomers (N_2H_4 with teflon, or Neoprene). Although bladders and diaphragms are simple to operate and reliable, they are incompatible with reactive oxidizers (N_2O_4, for example) and cryogenic liquids. Surface tension systems need to be used with the storable oxidizers, and based on a philosophy of commonality of hardware they very probably should be used with the related fuels as well (MMH, UDMH or A-30, for example) on the same flights.

Cryogenic liquids present some special problems in the area of liquid acquisition, principally related to thermal leakage and bubble formation in the devices themselves. This can be avoided by appropriate design minimizing thermal inputs into the PMD's, as well as the use of thermodynamic venting and cooling of the PMD themselves. This approach is recommended for use with superfluid helium as well, together with the rest of the special hardware compliments necessary for this fluid, including vacuum jacket vapor cooled shields, porous plug evaporators and thermoacoustic dampers in the lines, as required.

Figure 3g shows the various acquisition devices studied and highlights their main features.

3.0-C Contamination Control

Acceptance of fluid resupply by the satellite user community critically depends on the minimization of impact by the operations on those satellites, as well as economic considerations. Contamination of sensitive instrumentation, chemically active surfacves or thermal covers would compromise the operational effectiveness of the user satellite and therefore reduce the attractiveness of resupply.

This study has concentrated on the conceptualization of resupply system minimizing the impact on the user by starting with user-defined requirements. A contamination control system approach has been developed which minimizes contamination by 1) using zero-dribble-volume quick disconnects, 2) design of a fluid interface which permits purge of all surfaces contacting any fluid considered to be a contaminant, 3) sleeving the Q-D's with purge jackets, 4) using catalytic vents with N_2H_4 to avoid liquid venting and the resulting contamination, and 5) use of vacuum catch bottles for trapping purge products or 6) use of chemical scrubbers to trap purge products.

The purge catch bottles could be vented to space between engagements, at distances considered to be contamination-safe from any sensitive vehicles, in preparation for the next engagement.

3.0-D Receiving Spacecraft Design Considerations for Resupply

Minimizing the overall cost of resupply on-orbit requires that careful consideration be given to design of the spacecraft and its systems to achieve maximum compatibility with resupply operations. It is also desirable from a cost and operations standpoint that the resupply interfaces of the various spacecraft and platforms be standardized as much as possible.

The most obvious of the special provisions needed on such spacecraft are the fluid disconnect fittings which interface with the resupply module. One basic option in the use of such fittings is the "hard docking" mode in which opposing disconnect halves are placed in close proximity by the procedure of mechanical docking between two interface rings. The disconnect halves are then engaged by special drive mechanisms located on the resupply module.

One of the most important spacecraft resupply design factors is the basic configuration of the propulsion system (RCS or ACS) to be serviced. From the standpoint of operating cost and simplicity, the preferred liquid propulsion system would be a conventional monopropellant type using a single "blowdown" pressurization mode. Besides the fact of requiring one rather than two propellant resupply circuits, pressurant gas venting and resupply is not required since the ullage gas is recompressed in the spacecraft propellant tanks by the act of refilling (usually to a 2/3 or 3/4 full condition). Higher propellant resupply pressures are required, however, to accomplish this recompression.

If a storable propellant tank on the spacecraft uses regulated pressurization for its ullage, it is necessary (prior to resupply) to either vent this gas, or return it to the resupply module for recompression or storage in the propellant resupply tank ullage. Venting usually requires a non-propulsive vent using opposed nozzles. If contamination from storable propellant vapors is a problem, a catalyst device may be required to decompose these into less harmful products, and/or a mast may be needed to locate the vent nozzles at an acceptable distance from the spacecraft.

Whenever gas is compressed in a tank during resupply, heat of compression is generated which may have to be dissipated during refueling and thus prolong the transfer operation. This can be minimized by heat-sinking the affected tanks to the spacecraft structure, or to a special radiator by means of heat pipes. Gas orifices or swirl nozzles in the receiver tank can also be used to promote heat transfer to the tank wall. In the case of a blowdown propellant tank using capillary acquisition devices, liquid jets inside the tank can maintain nearly isothermal ullage conditions during refilling. However, if a diaphragm is used for liquid acquisition, such forced heat transfer in the recompressed ullage is not feasible, and several hours may be required for ullage cooling by gas conduction in the zero-g environment.

In the case of cryofluid resupply, a special bypass vent circuit may be needed in the spacecraft to dump chilldown vapors from the initially warm transfer plumbing directly overboard and avoid the addition of thermal energy to the receiver tank.

Other potential spacecraft design requirements for resupply include pressure and temperature instrumentation for propellant quantity gaging by the PVT method. Safing of the spacecraft prior to docking requires redundant isolation valving in the propellant lines and vent lines, and appropriate control interlocks in systems such as RCS, ACS attitude control, electrical and guidance.

SUBSYSTEM QUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
3.0 Resupply Fluid System (OMV Bi-prop) (ullage vent/ repressurize)	3.1 Tanks & PMD (OMV)	• RM is interface system bet. OMV propellant supply and receiver vehicle	• Temp control - In prop temp limits (50^o - 80^o F) • Pressurization $25 \leq P \leq 350$ PSI • Prop WT ≈ 6000 lbs	• Resupply fluid storage • Ullage bubble containment • Re-pressurant Supply
	3.2 Lines 1 in. OD-CRES	• Thermal Control • Flow meters • Vent/purge	• Pressurized lines (300 PSI) • Flow rate • MMH $\leq$ 10 lbs/sec • NTO $\leq$ 16 lbs/sec • System precludes contact of reactants	• Fluid distribution and delivery • Purging operations • Pre transfer • Post transfer
	3.3 Valves & Vents	• Valve drivers • Burst Discs/Relief valves	• 24 VDC Solenoid valves • 50 Micron filters • Vent/Purge gas capture • Fluid System isolation • Redundant sets • < 40 MSEC Actuation • Liquid/gas separation • ΔP per RDD	• Fluid temperature & pressure control • Create ΔP for transfer (per reqt's def. document) • Ullage venting
	3.4 I/F Umbilical - Receiver	• Docking & Latching Mechanism • Fluid Pressurant Transfer • DPS Status & Commands • Umbilical Actuators	• Angle misalignment < 30^o • Axial Misalignment • Spill-Proof Connects • Quick-Disconnect • Redundancy (Seals & Mechanisms) • Common interface	• Structural retention • Fluid Transfer • Status Monitoring • System Repressurize
	-Carrier	• OMV Compatible Interface • Communication • Fluid/Pressurant Delivery	• Leak detection • QD - Forces •Make - < 500 lb •Break - 0 •Hold - < 250	• Fluid delivery • Uplink & Downlink Command, Monitoring & emergency over-ride
	-Orbiter	• Orbiter P/L Power & DPS	• Meet Orbiter cargo bay reqts for deployable P...	• Power & DPS Supply whe when berthed in Orbiter

FIGURE 3a

ORIGINAL PAGE IS OF POOR QUALITY

SUBSYSTEM REQUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
3.0 Resupply Fluid System N_2H_4 (ullage recompression)	3.1 Tanks & PMD	• Ground support Fill & Drain • P & T x-ducers • Purge • Thermal Control • Module Latches to Standard EAM Structure • Pressurant Supply	• Temp control - In prop temp limits (50^0 - 80^0 F) • Pressurization 25< P < 350 PSI • Prop WT < [illegible] lbs	• Resupply fluid storage • Ullane bubble containment
	3.2 Lines 1 in, OD-CRES	• Thermal Control • Flow meters • Vent/purge • Fill & Drain	• Pressurized lines PSI) • Flow rate • Transfer time: 4 hrs •	• Fluid distribution and delivery • Purging operations • Pre transfer • Post transfer • Fill & Drain
	3.3 Valves & Vents	• Valve drivers • Burst Discs/Relief valves	• 24 VDC Solenoid valves • 50 Micron filters • Vent/Purge gas capture • Fluid System isolation • Redundant sets • < 40 MSEC Actuation • Liquid/gas separation • ΔP per RDD	• Fluid temperature & pressure control • Create ΔP for transfer (per reqt's def. document) • Ullage heat control
	3.4 I/F Umbilical - Receiver -Carrier	• Docking & Latching Mechanism • Fluid Transfer • DPS Status & Commands • Umbilical Actuators • OMV Compatible Interface • Communication • Structural Attach	• Angle misalignment < 30^0 • Axial Misalignments • Spill-Proof Connections • Redundancy (Seals & Mechanisms) • Common interface • Leak detection • QD - Forces •Make - < 500 lb •Break - 0 •Hold - < 250	• Structural retention • Fluid Transfer • Status Monitoring • System Repressurize • Uplink & Downlink

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SUBSYSTEM REQUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
3.0 Resupply Fluid System SFHe	3.1 Tanks & PMD	• Ground Support Fill & Drain • P & T x-ducers • Thermal Control • Module Latches to Standard ERM Structure	• Temp control - In prop temp limits < 1.4°K • Pressurization • Fluid WT 600 lbs • Vacuum jackets/vacuum vapor cooled shields	• Resupply fluid storage

	3.2 Lines 1 in, OD-CRES	• Thermal Control
		• Flow meters
		• Vent/purge
		• Porous Plug
			• Flow rate
			• Transfer time < 72 hrs
			• Thermoacoustic dampers
				• Fluid distribution and delivery
				• Purging operations
				  • Pre transfer
				  • Post transfer
				• Thermal conditioning pre-fill

	3.3 Valves & Vents	• Valve drivers • Burst Discs/Relief valves • Porous plug	• 24 VDC Solenoid valves • Porous Plug • Fluid System isolation • Redundant sets • < 40 MSEC Actuation • Liquid/gas separation • ΔP per HDD	• Fluid temperature & pressure control • Create ΔP pulse for transfer (per reqt's def. document)

	3.4 I/F Umbilical
	- Receiver
	
	-Carrier
		• Docking & Latching Mechanism
		• Fluid Transfer
		• DPS Status & Commands
		• Umbilical Actuators
		
		• OMV Compatible Interface
		• Communication
		• Structural Attach
			• Angle misalignment < 30°
			• Axial Misalignment
			• Spill-Proof Connections
			• Redundancy (Seals & Mechanisms)
			• Common interface
			• Leak detection
			• QD - forces
			  •Make - < 500 lb
			  •Break - 0
			  •Hold - < 250
				• Structural retention
				• Fluid Transfer
				• Status Monitoring
				• System Repressurize
				
				• Uplink & Downlink Command, Monitoring & emergency over-ride

FIGURE 3c

Storable Fluid Transfer Concepts

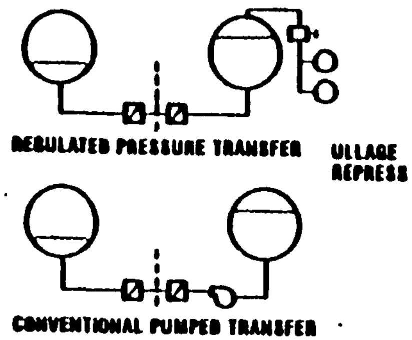
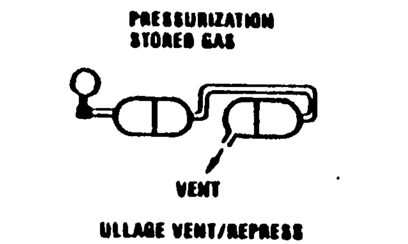
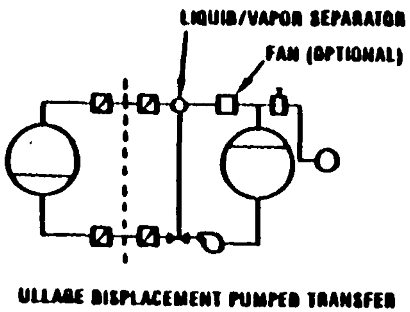
SYSTEM	ADVANTAGES/DISADVANTAGES		FLUID COMPATIBILITY		
			SUPPLY TANK	RECEIVER TANK	PROPELLANT
	• MECHANICAL SIMPLICITY NO REQUIREMENTS FOR PURGING & REPRESSURIZING THE TANK ULLAGE • OPERATIONAL SIMPLICITY. ONLY PROPELLANT IS TRANSFERRED, SIMULTANEOUSLY REFUELING & REPRESSURIZING • PROPELLANT FLOWMETERS ARE NOT REQUIRED PVT DATA MEASURES FUEL LOAD	• ULLAGE COMPRESSION REQUIRES A LARGE HEAT DISSIPATION • ΔP OF SUPPLY MUST BE HIGHER THAN RECEIVER	ANY (SCREEN VANE DIAPHRAGM)	ANY (SCREEN VANE DIAPHRAGM)	ANY (N_2H_4 NTO MMH)
	• LOW PRESSURE PROPELLANT SUPPLY • LOWER HEAT OF COMPRESSION THAN ULLAGE RECOMPRESSION DURING REPRESSURIZATION	• GREATER OPERATIONAL & MECHANICAL COMPLEXITY THAN ULLAGE RECOMPRESSION • PROPELLANT METERING IS REQUIRED	SCREEN VANE DIAPHRAGM	ANY ANY ANY	ANY N_2H_4, NTO, MMH N_2H_4
	• CONSTANT PRESSURE PROPELLANT TRANSFER • LOWEST COMPRESSION HEATING • OPERATIONALLY LESS COMPLEX THAN ULLAGE VENTING • SIMULTANEOUS TANK FILL POSSIBLE	• MECHANICALLY AND OPERATIONALLY MORE COMPLEX THAN ULLAGE RECOMPRESSION • PROPELLANT METERING IS REQUIRED	SCREEN, VANE ANY ANY	SCREEN, VANE SCREEN, VANE DIAPHRAGM	NTO, MMH N_2H_4 N_2H_4

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FIGURE 3d

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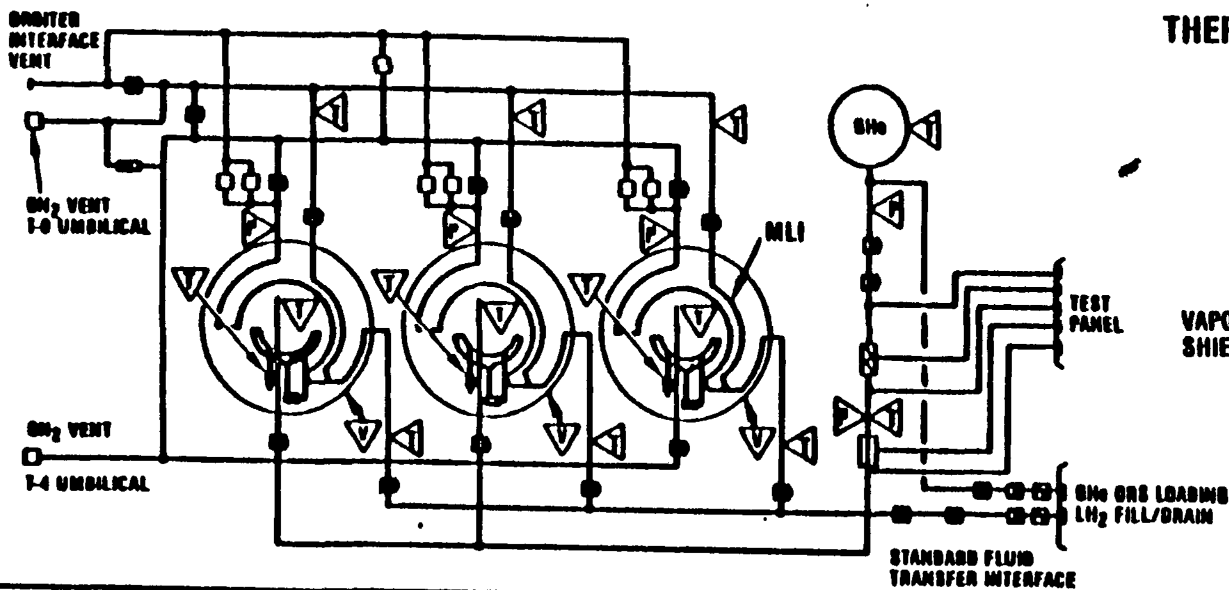
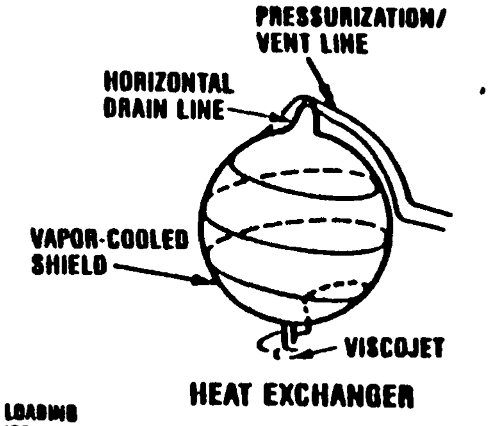
Cryo Transfer Concepts

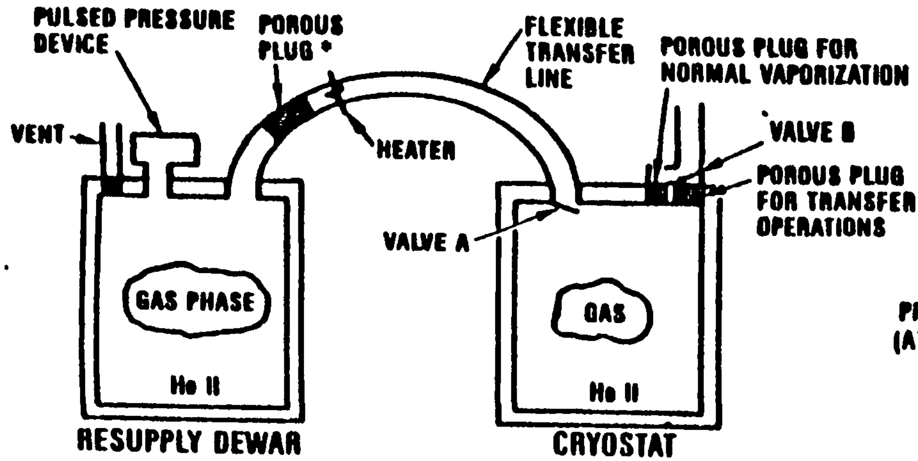


THERMODYNAMIC VENT SYSTEM



HEAT EXCHANGER

SUPERFLUID HELIUM TRANSFER



*POROUS PLUG SIZED FOR DEWAR VOLUME, FLOW RATE, & THERMAL INPUT

OPERATION SEQUENCE
- TRANSFER LINE HOOKUP
- VALVE A & B OPEN
- PPD ACTIVATED
- HEATERS' CONTROL ΔP

NORMAL vs SUPERFLUID HELIUM

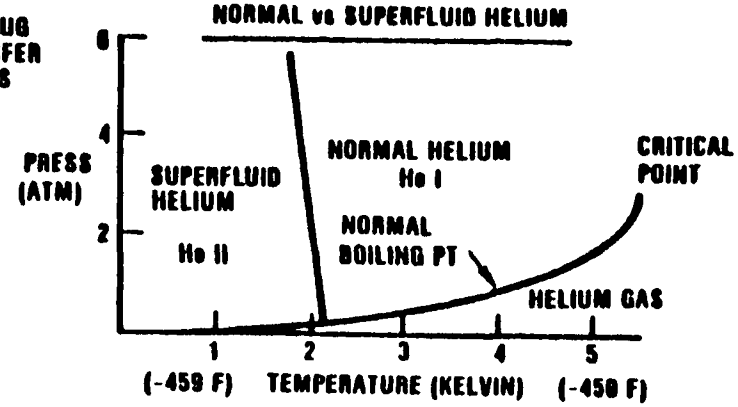


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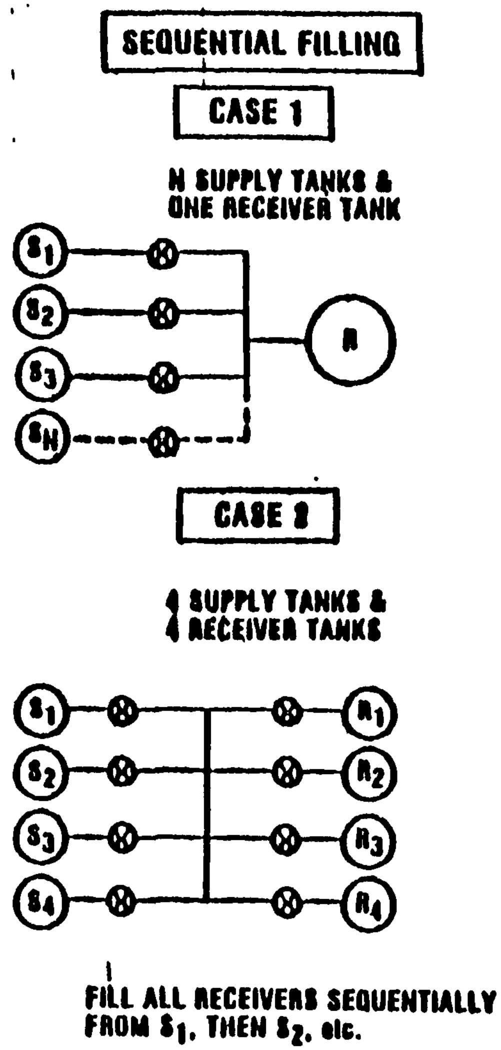
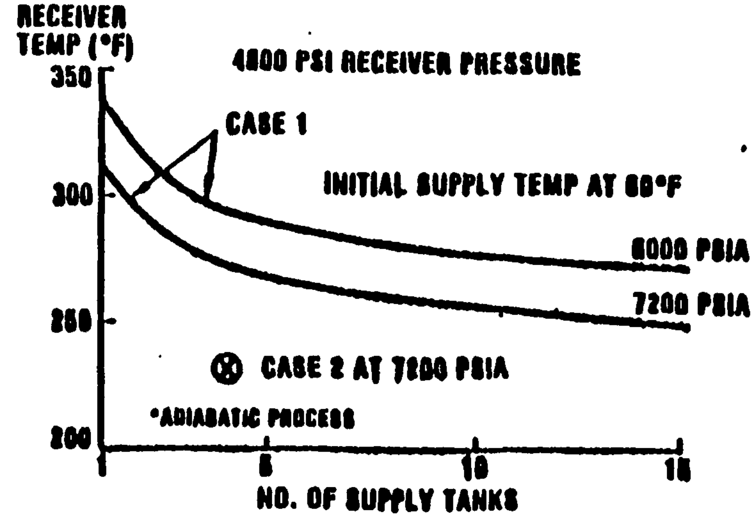
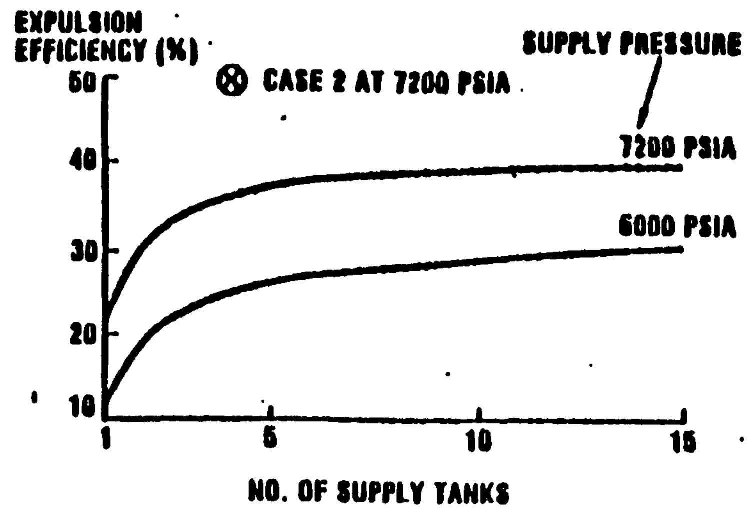
FIGURE 3e

ON-ORBIT PRESSURANT TRANSFER ,
HIGH-PRESSURE GAS SUPPLY TO RECEIVER TANKS







Shuttle Orbiter Division Rockwell International 93SSV139469A

FIGURE 3f

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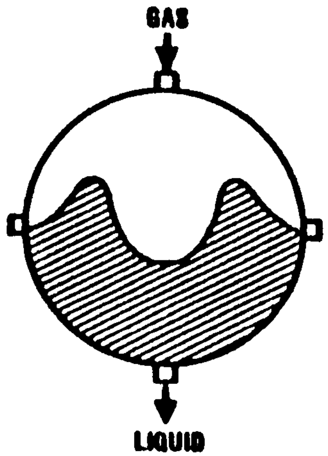
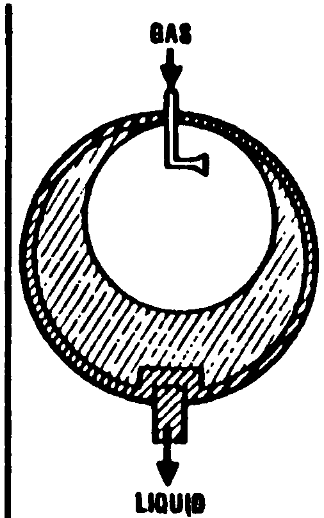
Fluid Acquisition Devices Technology



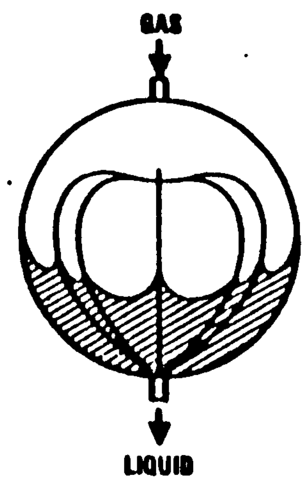
ELASTOMERIC DIAPHRAGM

- USED WITH N_2H_4
- INCOMPATIBLE WITH NTO
- REUSABLE
- EFFECTIVE IN ZERO/MULTI-g
- GAS PERMEABILITY



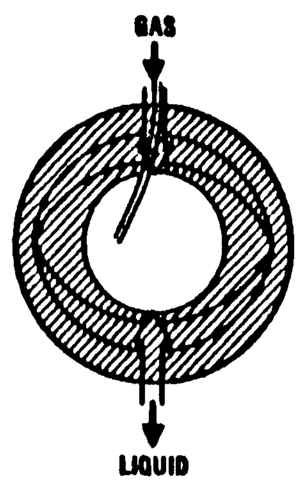
SCREEN GALLERIES

- USABLE WITH ANY FLUID
- REUSABLE
- EFFECTIVE IN 0-LOW-g & AXIAL-g
- REFILL & BUBBLE EXPULSION REQUIREMENT DRIVES RESUPPLY PROCEDURE
- COMPLEX/COSTLY
- HEAT TRANSFER/CONDITIONING REQUIREMENTS FOR CRYOS



INNER SCREEN VANES

- USABLE WITH ANY FLUID
- REUSABLE
- EFFECTIVE IN 0-g/ LOW-g & AXIAL-g
- LIQUID TENDS TO "CORE"
- SIMPLE FABRICATION



OUTER VANES

- USEFUL WITH CRYOS/SFHe
- 0-g ENVIRONMENT
- LIQUID TENDS TO WALL
- HEAT-TRANSFER/CONDITIONING FOR CRYOS

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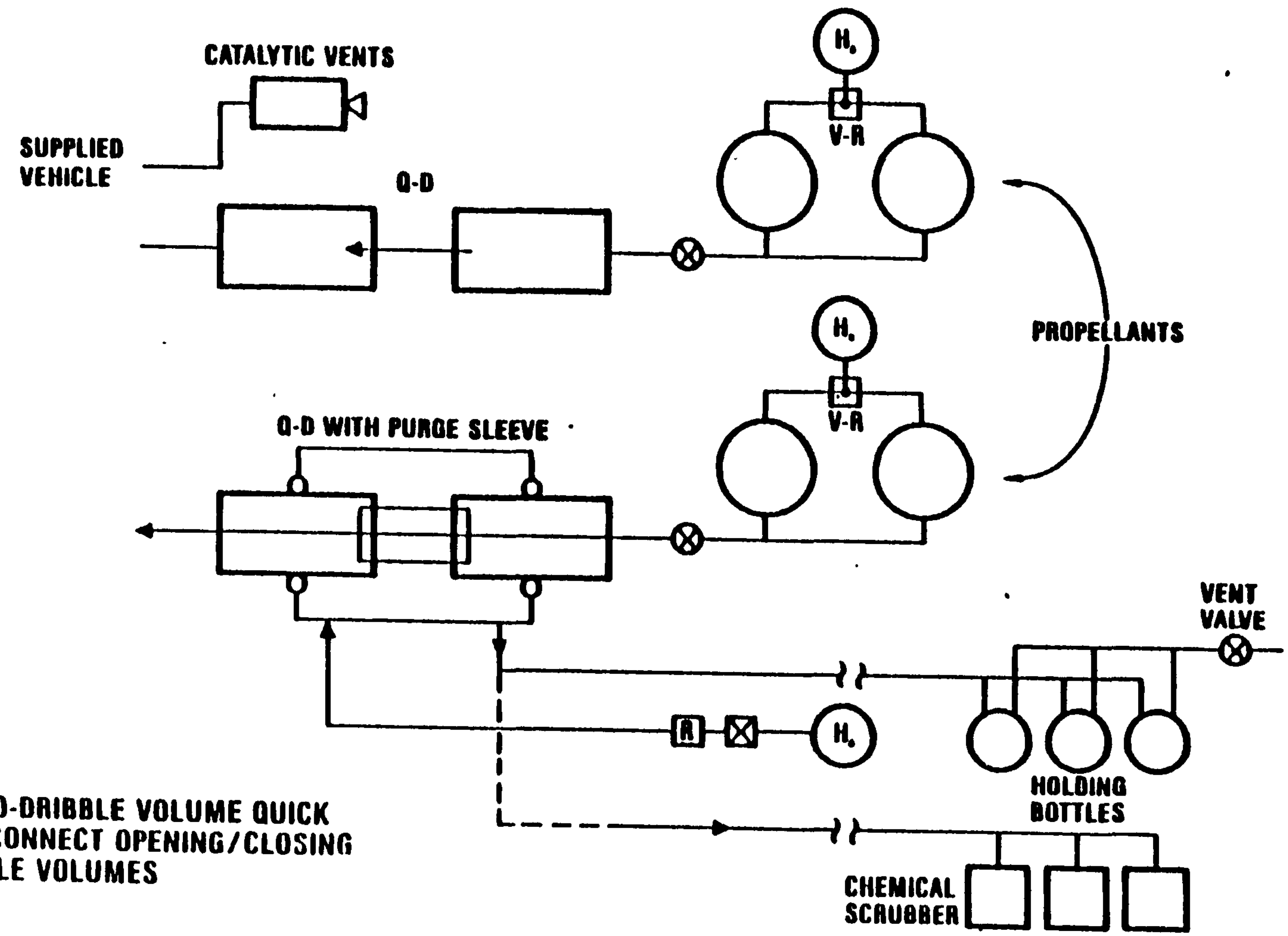
FIGURE 39

Contamination Control Options



ISSUE:

ZERO-DRIBBLE VOLUME QUICK
DISCONNECT OPENING/CLOSING
CYCLE VOLUMES

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FIGURE 38

4.0 MECHANICAL SUBSYSTEM

4.1 Orbiter Attach Mechanisms

4.2 RMS End Effectors/Grapples

4.3 Docking Umbilical Actuators

4.4 Mission Specific Mechanisms

4.5 Separation Pyros

4.0 MECHANICAL SUBSYSTEM INTRODUCTION

The ERM mission scenarios require various structural and mechanical interfaces among the STS and resupply elements. The ERM design should minimize interface hardware and maximize interface compatibility with its elements. The required structural and/or mechanical interfaces follow:

- o ERM/GSE
- o ERM/Orbiter Cargo Bay
- o ERM/Orbiter RMS
- o ERM/Space Station
- o ERM/Carrier Vehicle (OMV)
- o ERM/Receiver

A tabular summary of this section's requirements is provided in Figure 4a.

4.1 Orbiter Attach Mechanisms

The ERM shall provide support trunnions so as to fit in the Orbiter cargo bay as a deployable payload, conforming to ICD 2-19001, "Shuttle Orbiter/Cargo Standard Interfaces". The attached device shall allow the ERM to be replaced in the cargo bay and secured for return to earth.

The same support mechanism could be used to facilitate ground handling and processing operations.

4.2 RMS End Effectors/Grapples

An RMS grapple fixture will allow an Orbiter and/or Space Station RMS to handle the ERM within its cargo bay and facilitate mating and demating of the ERM from its carrier vehicle.

The ERM shall provide another grapple fixture compatible with the OMV carrier forward payload end effector system to attach and detach from the carrier vehicle.

4.3 Docking/Umbilical Actuators

The docking and umbilical actuators will be required to automate the transfer of fluids from the RM to the Receiver. The primary design approach will be the use of remotely controlled and automatic means for docking, fluid and pressurant transfer and separation. The docking system must allow for disconnect misalignment given the navigational accuracy of the approach.

Design goals for remote engagement/disengagement of a self-sealing, automatic open/close disconnect should include the following:

- O Employ "gang-on"/"Gang-off" design concept for mating/demating all disconnects simultaneously for docking/separation
- O Provide semi-balance/balanced design to minimize engagement/disengagement loads
- O Provide swivel design to obtain maximum misalignment to simplify docking maneuvers
- O Incorporate design features to preclude jamming of the ganged disconnects during docking maneuvers
- O Minimize trapped volumes, leakage rates, and retention/separation forces
- O Minimize travel required for engagement/disengagement
- O Incorporate design features to preclude failure of disconnect to disengage under designed retractional forces

A preliminary review of existing hardware indicates that a family of disconnects is available for consideration in a Resupply System. Modifications to existing hardware and demonstrations type testing are considered required to meet the ERM requirements

Two previous disconnect study programs conducted, (MMS in-orbit refueling study analysis and design for Goddard Spacecraft Center, and Space Transportation System Disconnect Program for George C. Marshall Space Flight Center) indicate that remote refueling in-orbit is feasible, and can be defined and based on the use of currently qualified space rated hardware. The only new design innovation on this program for disconnects is docking remotely. However, this requirement is considered within the state-of-the-art for disconnects.

The mating, checkout, operation, and demating of fluid disconnects is one of the more critical resupply operations.

4.4 Mission Specific Mechanisms

The design goal is to keep mission specific mechanism requirements to a minimum. This would be accomplished through standardization of receiver and carrier interface plates. This interface concept is shown in Figure 4b. It is of flexible design to accommodate the full spectrum of fluids which have been determined to be required by candidate receivers. The concept simplicity would result in reliable operation and reduced development cost.

The ERM requirements for retention mechanisms (discussed in sections 4.1 - 4.3) are also driven by the design goal to maximize interface compatibility. Where the ERM is loaded in space (e.g., STS fluid scavenging) or used to transfer OMV bi-propellants from the Carrier to the Receiver, unique requirements arise. The interface plate to the Carrier (or space-based fluid source) would be required to provide fluid and DPS umbilicals. Figure 3a in Section 3 summarizes the fluid transfer system requirements for OMV bi-prop transfer.

4.5 Separation Pyros

Provision must be made to effect separation of the attach mechanisms should the system seize and fail to disengage. Pyrotechnic devices may be necessary on both the ERM carrier interface (should the carrier not provide for payload release) and the Receiver interface.

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SUBSYSTEM REQUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
4.0 MECHANICAL SYSTEMS	4.1 ORBITER ATTACH MECHANISM	• Orbiter Payload Bay - Deployable P/L Actuators - P/L Power - P/L DPS	• Compatible with orbiter cargo bay reqts for deployable payload	• Provide Transport & deployment from Orbiter • Provide cargo bay support to ERM Subsystems
	4.2 RMS END EFFECTORS AND GRAPPLES	• Orbiter RMS • Space Station RMS • Carrier attach	• Compatible with RMS attach effectors • Compatible with carrier attach effector	• RMS Stow/Destow & mate with carrier
	4.3 DOCKING/UMBILICAL	• Receiver I/F Plate • Receiver Grapple Fixture	• Autonomous Engagement latching, Unlatching • High reliability • Failed release	• Dock with receiver • Mate/Demate power, DPS & Fluid Transfer umbilicals
	4.4 MISSION SPECIFIC MECHANISMS	• Unique fluid umbilicals	• Compatible with standard interface on carrier & receiver	• Provide unique mission hardware support
	4.5 SEPARATION PYROS	• Receiver I/F attachment	• Effect emergency & safe separation from receiver	• Enable emergency separation

FIGURE 4a

Spacecraft Interface Concept

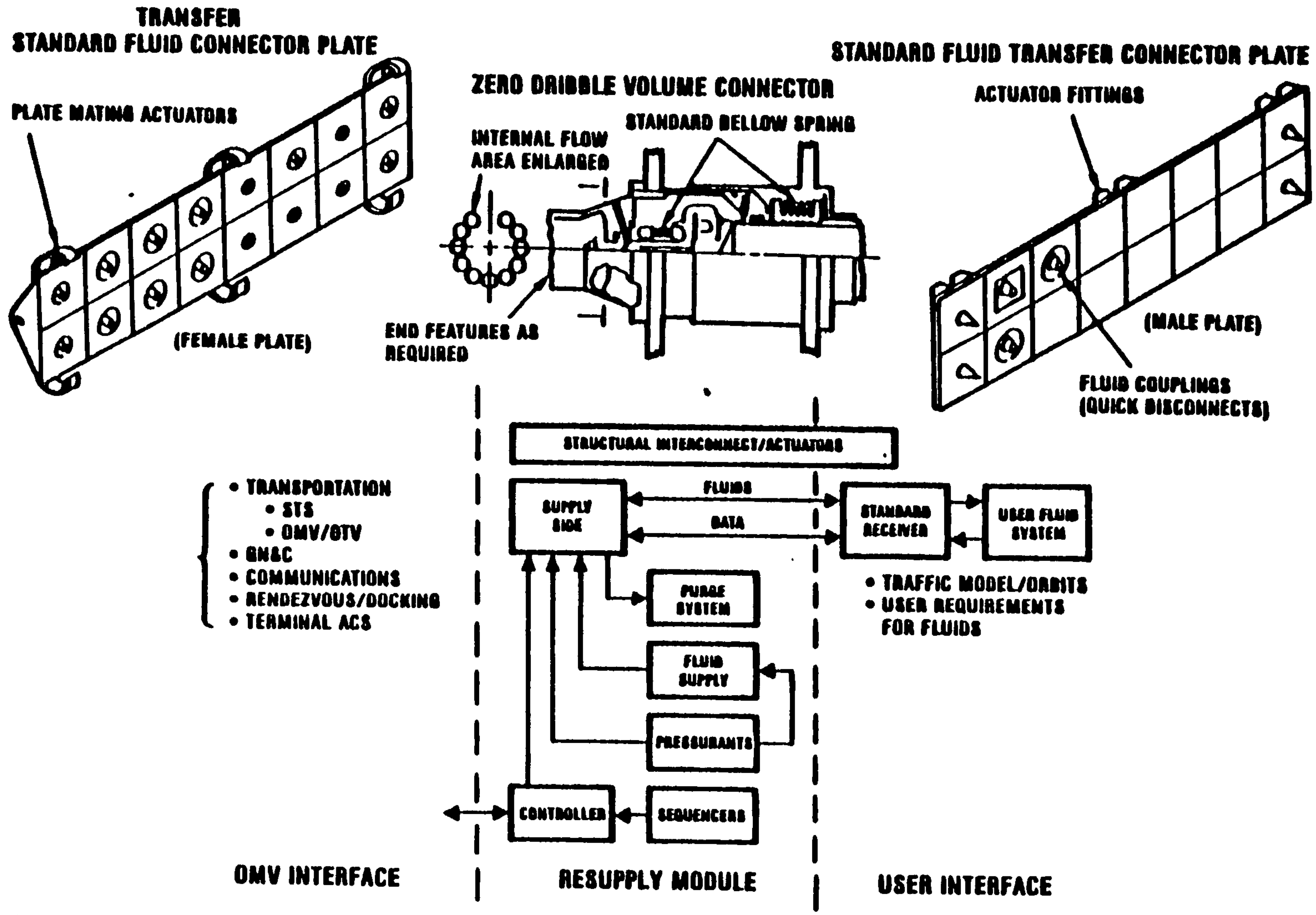

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FIGURE 4b

5.0 STRUCTURES SUBSYSTEM

5.1 Frame

5.2 Subsystem Fittings/Brackets

5.3 Interface Attach Devices

5.4 Thermal Blankets & Covers

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5.0 STRUCTURES SUBSYSTEM INTRODUCTION

The structural configuration of the ERM must accommodate a wide variety of resupply missions, fluids, tank shapes and sizings. The ERM system configurations which support the various mission scenarios include:

- o ERM/GSE
- o ERM/Orbiter Cargo Bay
- o ERM/Space Station Storage
- o ERM/Carrier Vehicles
- o ERM/Receiver and ERM/Carrier
- o ERM/Carrier/Transfer Vehicles

Two of the basic structural requirements shall be adaptability and lightweight design. To achieve these requirements, a number of concepts have been developed and are shown in Figure 5a.

A tabular summary of this section's requirements is provided in Figure 5b.

5.1 Frame

The basic RM framework shall distribute and react external and internal loads resulting from all design on flight and ground loads and their associated operational environments with any ERM configuration defined in the introduction.

The frame must be able to accommodate a wide variety of resupply missions, fluids, tank shapes and sizings. The structure shall be provided with adequate strength and stiffness in its design environment to withstand limit loads without loss of operational capability and to withstand ultimate loads at design temperature without failure. The ERM shall satisfy the structural requirements as specified in Volume XIV, ICD 2-19001, "Shuttle Orbiter/Cargo Standard Interfaces" and in Section 5.0 of this document.

5.2 Subsystem Fittings/Brackets

To achieve the basic structural requirements for adaptability and lightweight design, the subsystem fittings and brackets must provide for mission design flexibility.

The subsystems requiring secondary structural attachments include the avionics and power modules, resupply tanks and their associated plumbing, and the interface hardware.

The RM may be required to provide attachment for the OMV docking camera and lighting equipment which are otherwise excluded during rendezvous approach with the current OMV configuration.

A modular design with a central core could provide for mission specific hardware changeout, common subsystem sharing, eary access and quick reservicing or changeout, and a cost-effective design to meet various mission objectives.

5.3 Interface Attach Devices

To accomplish the remote resupply objectives of the various mission scenarios and to maximize the use of space transportation systems, numerous ERM system configurations have been defined (see Section 5.1). The RM shall provide a structural/mechanical interface for the following configurations:

- o RM/Shuttle Orbiter Cargo Bay
- o RM/Shuttle Orbiter RMS
- o RM/Carrier Vehicle (OMV)
- o RM/Receiver
- o RM/Space Station

The structural interfaces will be required to support static and dynamic loads emanating from launch flight and re-entry environments. Coupled dynamic loads are introduced by the Orbiter propulsion and altitude control system, RMS control maneuvers, the Carrier propulsion and altitude control system, the Receiver (if control system is active), gravity gradient torgues, aerodynamic drag forces, thruster exhaust plumes, and resupply fluid trnasfer and docking operations.

The structural/mechanical interface elements are discussed in Section 4.0.

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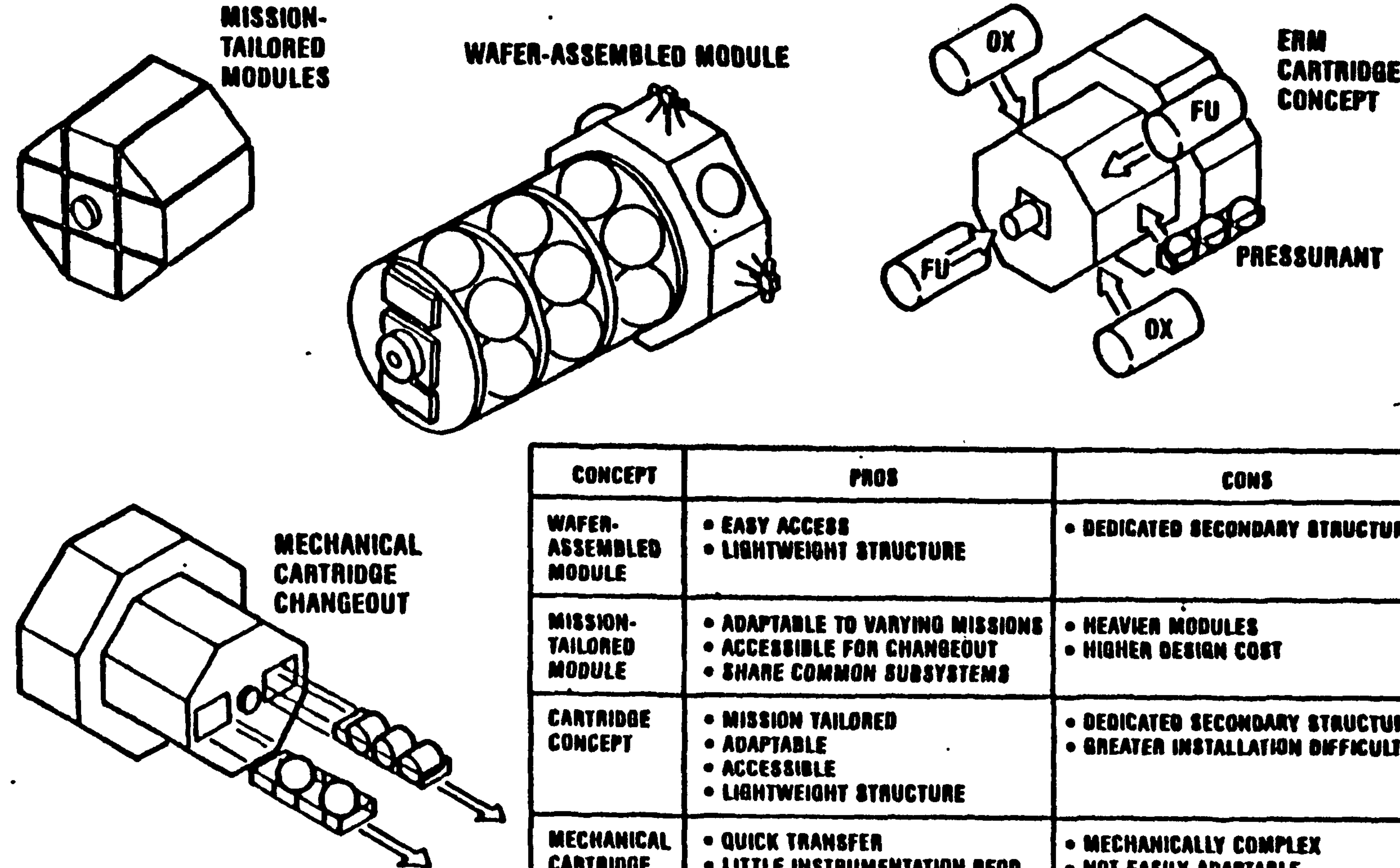
5.4 Thermal Blankets and Covers

Thermal blankets and covers shall provide the RM subsystems with passive thermal control to protect against the natural and induced flight thermal environment.

The thermal environment includes Orbiter ascent, entry and abort thermally induced loads, thruster exhaust plume heating loads, system and subsystem generated heat loads and thermal radiation.

Thermal insulation, coatings and thermal isolators shall be used where possible to reduce active thermal control requirements (see Section 1.5).

ERM — Concept Options



CONCEPT	PROS	CONS
WAFER-ASSEMBLED MODULE	• EASY ACCESS • LIGHTWEIGHT STRUCTURE	• DEDICATED SECONDARY STRUCTURE
MISSION-TAILORED MODULE	• ADAPTABLE TO VARYING MISSIONS • ACCESSIBLE FOR CHANGEOUT • SHARE COMMON SUBSYSTEMS	• HEAVIER MODULES • HIGHER DESIGN COST
CARTRIDGE CONCEPT	• MISSION TAILORED • ADAPTABLE • ACCESSIBLE • LIGHTWEIGHT STRUCTURE	• DEDICATED SECONDARY STRUCTURE • GREATER INSTALLATION DIFFICULTY
MECHANICAL CARTRIDGE CHANGEOUT	• QUICK TRANSFER • LITTLE INSTRUMENTATION REQD • ADAPTABLE TO COMPONENT SERVICING • MINIMUM INTERFACES	• MECHANICALLY COMPLEX • NOT EASILY ADAPTABLE • MAJOR DESIGN IMPACT ON RECEIVER • REQUIRES HIGH RELIABILITY • DIFFICULT TO CHECK OUT RECEIVER

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FIGURE 5a

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SUBSYSTEM REQUIREMENTS

SUBSYSTEM	ELEMENTS	INTERFACES	PERFORMANCE	OPERATIONS
5.0 STRUCTURAL SYSTEMS	5.1 FRAME	o ERM subsystem modules o Carrier, receiver, Orbiter attach devices	o Orbiter P/L bay compatible (14 ft) o Minimum weight (<1000 lbs) o Accessibility for modular changeout & mantenance o Adequate strength & stiffness for all mission phases	o Framework to support various tankage & fluids o Provides light weight framework to conduct resupply operations
	5.2 SUBSYSTEM FITTING/BRACKETS	o ERM frame, tanks, plumbing, harnesses, and all subsystems	o Adaptable to all mission configurations o Allow quick changeout of components	o Provide secondary structure to support standard & mission unique subsystems
	5.3 INTERFACE ATTACH DEVICES - Receiver	o Receiver docking port	o Provide for alignment & rigidity during fluid transfer OPS o Provide standardized interface for all potential receivers	o Facilitate docking operations with receiver
	- Carrier	o Carrier end effector	o Provide for structural attach w/carrier vehicle	o Facilitate use of carrier vehicle as transport vehicle
	- Orbiter	o Orbiter, payload bay o RMS	o Provide for structural attach w/orbiter & STS related interfaces	o Facilitate use of Orbiter & RMS to transport & maneuver ERM in vicinity of orbiter
	5.4 THERMAL BLANKETS & COVERS	o ERM Structure o ERM Subsystems	o Provide for on-orbit thermal control o Reduce active thermal control requirements	o Protects structure & subsystems from thermal & plume environment of space

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