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ADDENDUM

FINAL REPORT

SPACE PLATFORM EXPENDABLES RESUPPLY CONCEPT DEFINITION STUDY

FOR PERIOD JANUARY 1985 - OCTOBER 1985

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Addenda To:
Space Platform Expendables Resupply
Concept Definition Study
Final Report

STS85-0174
Contract NAS 8-35618
For Period January 1985 - October 1985

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ILLUSTRATIONS

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TABLES (Continued)

ACRONYM DICTIONARY

AXAF	Advanced X-Ray Astrophysics Facility
B/B	Brassboard
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
DOD	Department of Defense
DOMSAT	Domestic Satellite
EOS	Earth Observing System
ERM	Expendables Resupply Module
ETR	Eastern Test Range
EUVE	Extreme Ultraviolet Explorer
EVA	Extra Vehicular Activity
FDTA	Flight Demonstration Test Article
GEO	Geosynchronous Orbit
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
JPL	Jet Propulsion Lab
JSC	Johnson Space Center
LaRC	Langley Research Center
LDR	Large Deployable Reflector
LEO	Low Earth Orbit
MIL	Man-in-Loop
MMH	Monomethylhydrazine
MPS	Materials Processing In Space
MSFC	Marshall Space Flight Center
MUTE	Mid-deck Ullage Transfer Experiment
NASA	National Aeronautics and Space Administration
NTO	Nitrogentetroxide (N_2O_4)
N_2	Nitrogen
N_2H_4	Hydrazine
OMV	Orbital Maneuvering Vehicle
OTV	Orbit Transfer Vehicle
PMD	Propellant Management Device
PVT	Pressure, Volume, and Temperature
RCS	Reaction Control Subsystem
RF	Radio Frequency
RM	Resupply Module
RMS	Remote Manipulator Subsystem
ROM	Rough Order of Magnitude
S/C	Spacecraft
SDI	Strategic Defense Initiative
SIRTF	Shuttle Infrared Telescope Facility
SPAS	Shuttle Pallet Satellite
SPER	Space Platform Expendables Resupply
STS	Space Transportation System
TFU	Theoretical First Unit
TDRSS	Tracking and Data Relay Satellite System
USAF SD	United States Air Force Space Division
WTR	Western Test Range

1.0 EXECUTIVE SUMMARY

1.1 INTRODUCTION

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 supplemental study to the original Phase A study and is presented as addenda to that study.

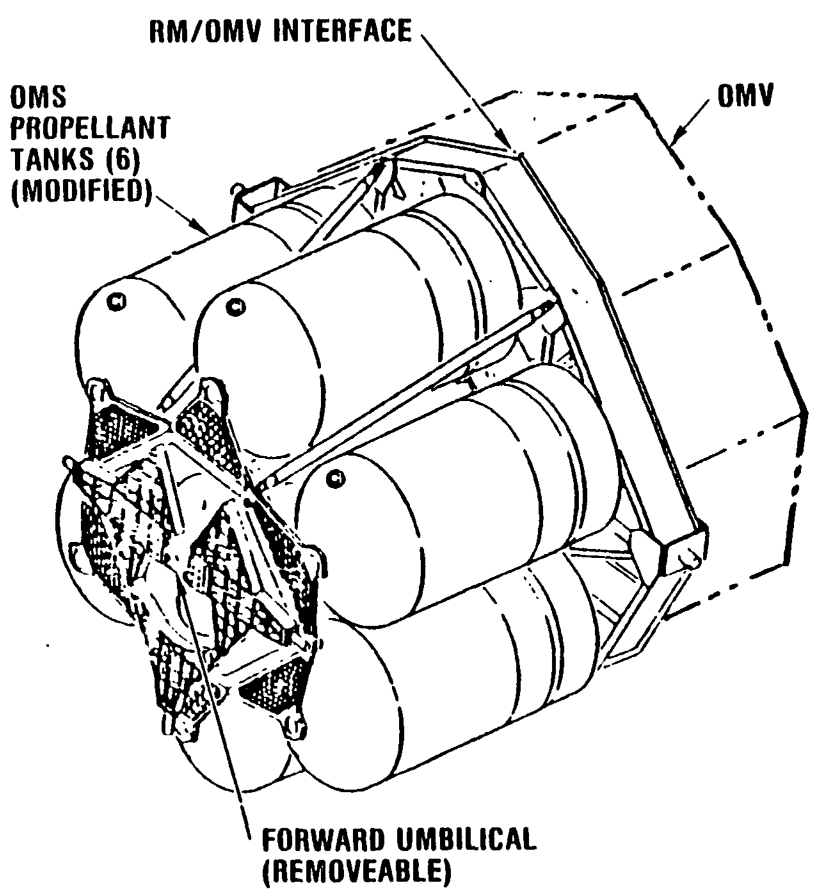
1.2 Background

As background an overview of key results from the original study are presented. The original Phase A study was a $340K effort which ran from March 1984 through January 1985. Long term objectives for this activity consisted of the definition of both an operational Expendables Resupply Module (ERM) and the definition of a flight demonstration for remote resupply. To this end the specific study objectives for the Phase A study consisted of the definition of user needs and requirements, the conceptual design definition of a representative resupply module, a conceptual approach to the flight demonstration of remote resupply, and the definition of a program plan to satisfy the long term objectives.

One key result of this original study activity was the design development of a resupply module concept. This concept and its key features are presented in Figure 1. The key features of this configuration include the capability of the ERM to be utilized as an ERM or an OMV extended mission kit. Six stretched OMS tanks contain 45,500 lbs. of usable propellant for resupply or OMV propulsive use. The dry weight of the ERM for bi-propellant resupply only is 7,960 lbs. For helium and bi-propellant resupply it is 9,140 lbs. The dry weight of the ERM to be utilized as an OMV extended mission kit would be 7,200 lbs. This includes the elimination of helium compressors and associated batteries and simplified propellant management devices to meet OMV propellant transfer requirements.

Also as a result of the original study, a concept for a Bi-propellant/Helium transfer flight demonstration was developed. This concept is presented in Figure 2. The principle test objective for the flight demonstration is to validate storable 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 end effector, the remote manipulator systems (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 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, it was recommended it remain seated in the cargo bay. This minimizes the interface complexity of power and electrical links running through the orbiter. The supplier test article would be autonomous with its own power source.



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 & 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 PMDs & ELIM OF COMPRESSORS/BATTERIES FOR) = 7200 lb

FIGURE 1. RM/OMOV CONFIGURED TO MEET USER NEEDS (DEVELOPED IN ORIGINAL STUDY)

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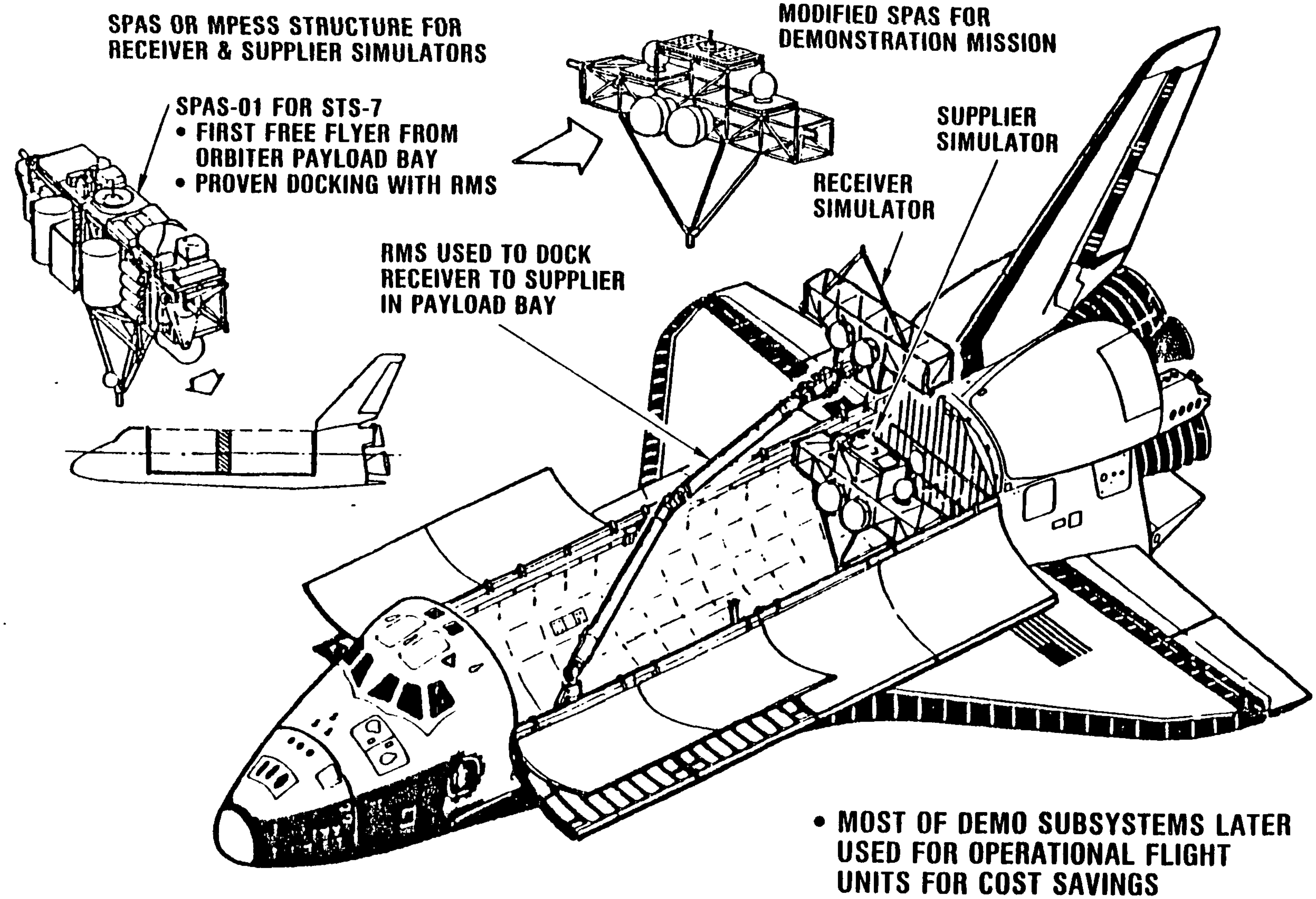

FIGURE 2. BI-PROPELLANT/HELIUM TRANSFER FLIGHT DEMONSTRATION CONCEPT (DEVELOPED IN ORIGINAL STUDY)

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Some additional key conclusions from the original study are as follows:

- o Remote resupply provides substantial benefits
  - o In GEO, mainly with concurrent servicing
    - o But . . . far term horizon
  - o In LEO, in conjunction with storable propellant depot
    - o Supports wide scope of potential operations
- o Single "core" resupply module design is adaptable to both requirement categories
- o Total program cost including adequate technology development & flight demonstration is low/affordable
- o Remote resupply can be operational by early 1990's LEO operations
  - o LEO operations projected first need
- o Evolves to support space station growth

1.3 Supplemental Study Scope and Objectives

The supplemental study was a $98K effort conducted from January 1985 through October 1985. In the supplemental study the requirements task was expanded beyond the original study to include consideration of the evolving space transportation infrastructure. The scope of the design section, in the supplemental study, was limited to the consideration of the transfer process itself and to flight demonstration concepts. The scope of the supplemental study programmatics task was limited to the generation of cost estimates of the resupply options and to provide support in identifying the "Best" demonstration program.

The objectives for this supplemental study were separated into three tasks: requirements, design, and programmatics. The primary objective was to select a preferred flight demonstration program option. This was accomplished by defining user needs, assessing the state of the development of the technology, defining demonstration concepts, and costing those concepts. User needs such as fluid type, quantity, and schedule enable module sizing to be defined. This information was then used as input to the design tasks where the demonstration design requirements and preliminary designs are generated. The programmatics task develops the cost estimates for the resupply concepts and provides decision support for selecting the "Best" demonstration program.

1.4 Summary of Results

Systems Requirements and Scenarios (Task 1)

The system requirements of the original task 1 were revised to include a projection of elements from the evolving space infrastructure. Figure 3 depicts an evaluation of possible key elements of the space infrastructure. These depicted elements are the ones of primary interest to the task of expendables resupply. In accordance with the expanded scope of this contract, a closer look has been taken at the resupply needs and interactions of infrastructure elements.

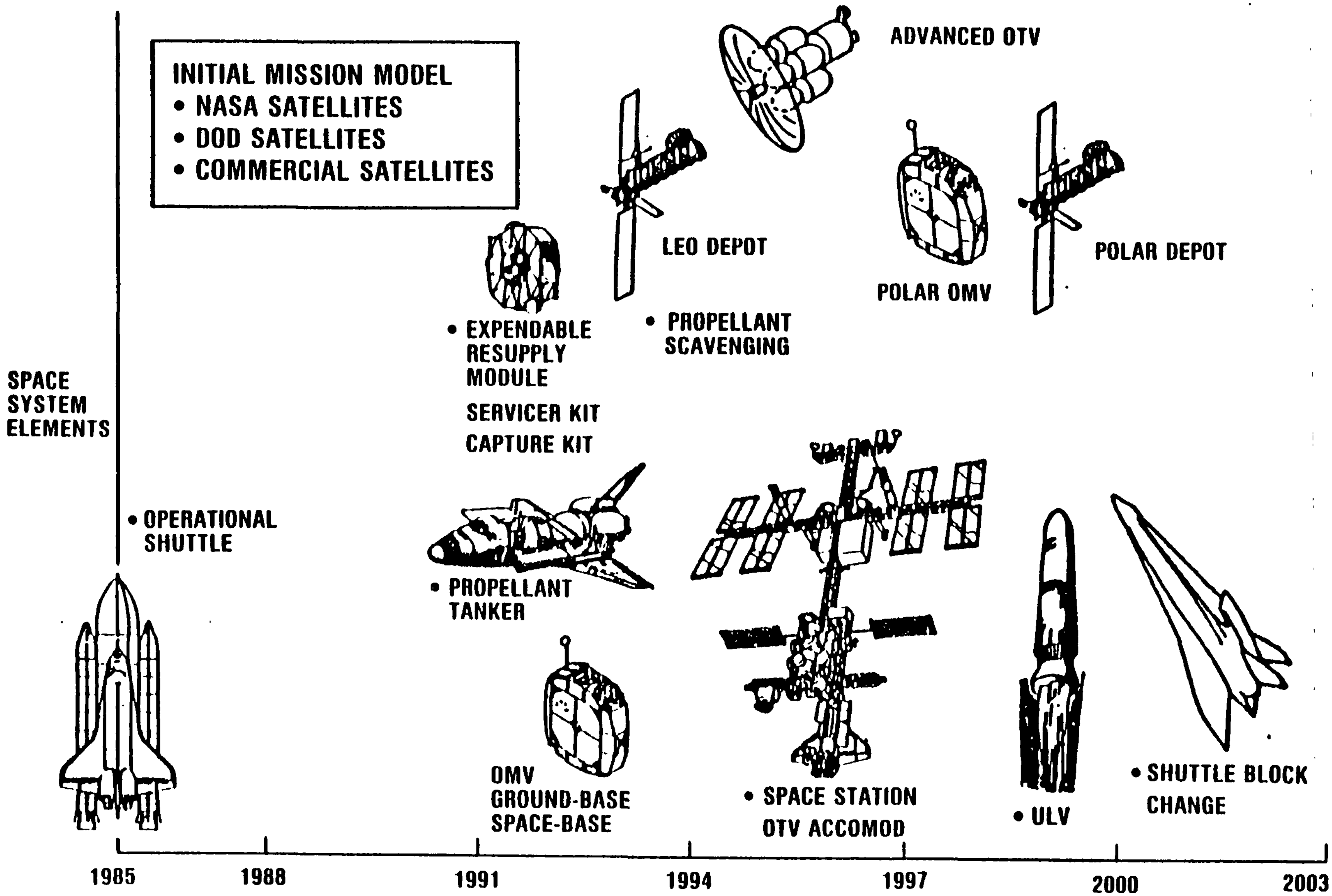


FIGURE 3. REQUIREMENTS DEPEND ON EVOLVING SPACE OPERATIONS

In this analysis the shuttle is being used as the basis for an in-bay propellant tank (under JSC) and later the arrival of the OMV with associated kits, such as the resupply module, servicer kit and capture kit. The OMV and associated kits can be deployed from the ground, and/or space-based at 28.5 degrees or polar orbits. These spacecraft will then be augmented by permanent orbital facilities primarily the space station with its accommodations for OMVs and OTVs. Associated with the station, but separate from it for safety requirements, will be a Leo depot, possibly supplied by propellant scavenging from the STS. Eventually, this depot should be duplicated in polar orbits with its own OMV and associated kits.

Around the turn of the century, new developments such as the shuttle block change and a fully reusable OTV will likely alter the economics of supplying propellant in space. The Shuttle block change could affect the use of propellant scavening, and the OTV will generate requirements for greater amounts of fuels, raising the need for sophisticated storage facilities in space. A secondary space transport and Launch Vehicle will likely arrive in support of SDI. SDI elements are not sufficiently developed yet to predict their impact on orbital fluid requirements and have, therefore, been left out of this projection.

Figure 4 depicts, in graphic form, the fluids to be transfered on-orbit to the elements of the space infrastructure. 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 resupply technique used, no additional pressurants are required for the transfer process. In the latter 1990's, bi-propellants and cryogens are the dominant required fluids. Hydrazine, water and liquid helium constitute a relatively stable base of demand.

In updating the resupply mission requirements extensive contacts were made with the potential user community. In addition to in-house data base and literature searches, three related surveys were conducted Figure 5. They were telephone contacts with various NASA program contacts used in the development of the resupply mission model. The updated data did not contain any major surprises and represented a maturing of several programs which may hold a more positive attitude toward fluid resupply.

In January of 1985, a series of presentations were made to satellite manufacturers TRW, Hughes, and Ford. The presentations were intended to direct feedback on the proposed demonstration program and user benefit analysis. A survey questionaire, prepared under Rockwell IR&D, was sent to over 90 members of the space community in the U.S. and aboard. The contacts included domestic and foreign space agencies, aerospace and communications companies, and space insurers. The survey attempted to find new candidate users of fluid transfer services for either remote or Orbiter-based service. Thirteen useful responses have been received to date, with most supporting the concept of concurrent servicing with resupply. As a result of the requirements re-evaluation and discussions with potential resupply users, six scenarios were developed for the ERM. These scenarios are depicted in Figure 6. In Scenario 1, the expendables resupply module provides the OMV with propellant

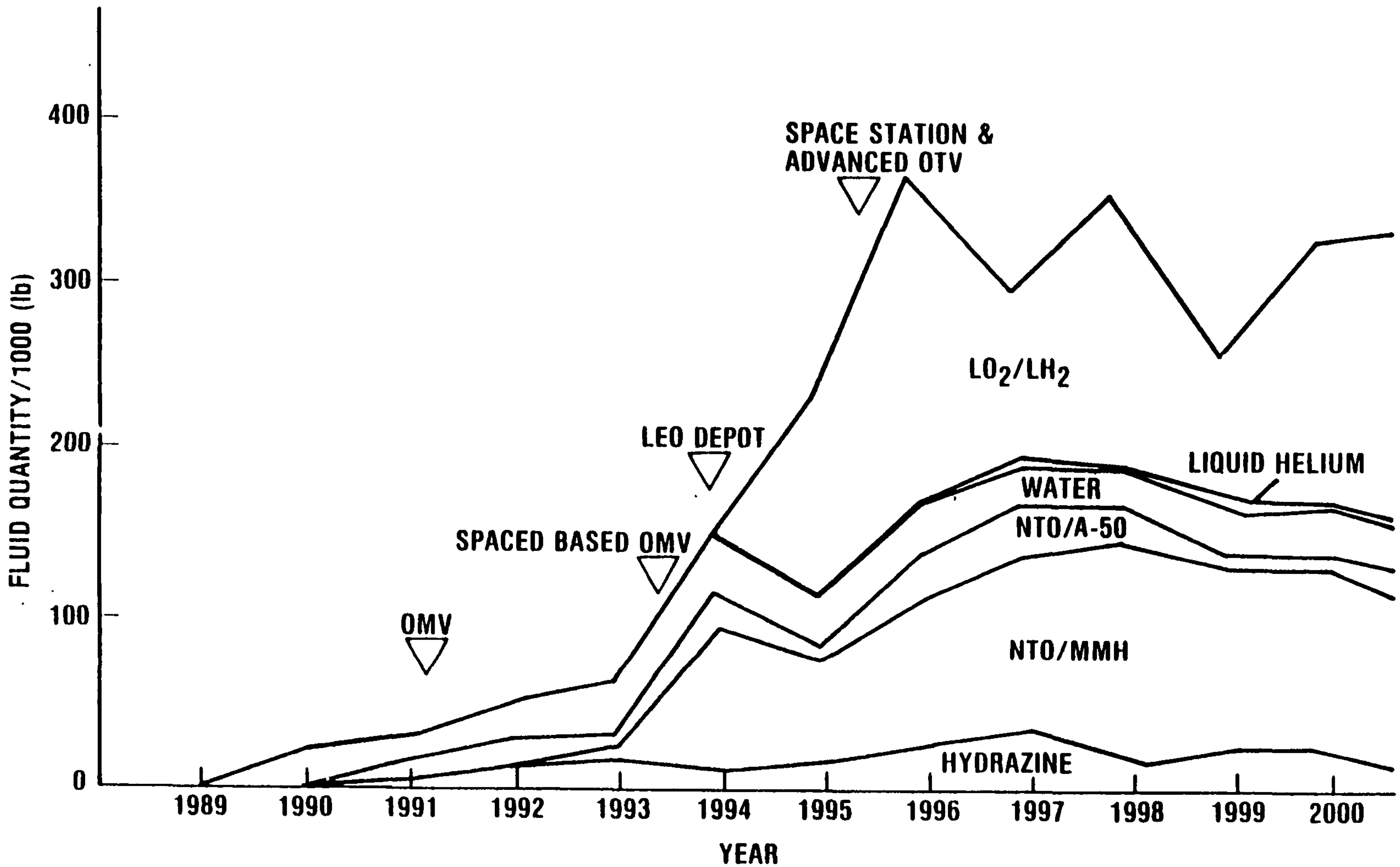


FIGURE 4. FLUID REQUIREMENTS DRIVEN BY SPACE OPERATIONS

TELEPHONE CONTACT
- NASA SCIENTIFIC
- COMMERCIAL MPS

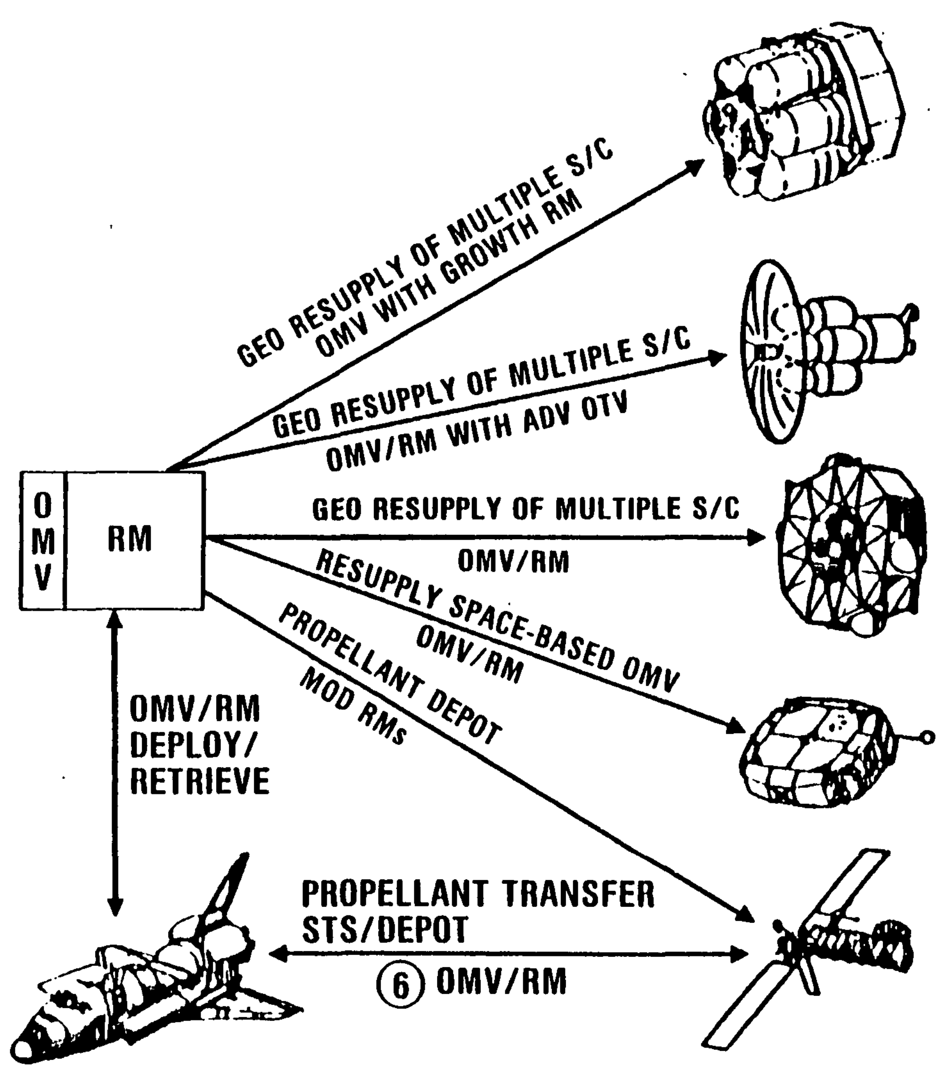
- ABOUT 20 PROGRAM OFFICES CONTACTED
- 12 POSITIVE RESPONSES
- UPDATE OF ORIGINAL STUDY SURVEY
- SPECIFIC RESULTS
  - MORE POSITIVE ATTITUDE

PRESENTATIONS TO SATELLITE SUPPLIERS
- TRW, HUGHES, FORD
- SPECIFIC RESULTS
  - CONTRACTORS INTERESTED IN APPROACH
  - SUPPORTED TECHNICAL FEASIBILITY

SURVEY QUESTIONAIRE
- SENT TO DOMESTIC & FOREIGN SPACE AGENCIES, AEROSPACE & COMMUNICATIONS COMPANIES, & INSURERS
- 90+ QUESTIONAIRES SENT
- 25 RESPONDED TO DATE
- 13 USEFUL RESPONSES
- SPECIFIC RESULTS
  - STRONG SUPPORT OF CONCURRENT SERVICING

SPECIFIC USER NEEDS & COMMITMENT BEING IDENTIFIED

FIGURE 5. THREE TYPES OF SURVEYS CONDUCTED



MISSION SCENARIO NO.	RM BI-PROP (lbs)	RM HELIUM (lbs)	
1	45,500 (GROWTH)	30-90	GEO
2	3,000	TBD	GEO
3	4,000	TBD	LEO POLAR/ 28.5°
4	7,000 (BASELINE)	12	LEO POLAR/ 28.5°
5	20,000 (MULTIPLE RMs)	TBD	LEO POLAR/ 28.5°

MAY 1985

FIGURE 6. USER NEED DETERMINES SPER MODULE SIZING

for transportation to GEO. This mode was found to be more economical than using an expendable OTV to transport the OMV to GEO. Propellant transfer (and concurrent module exchange servicing, when required) is accomplished directly from the OMV after separation from the resupply module in Scenario 2. This uses a reusable cryogenic OTV for Geo transport.

In Scenarios 3 through 6, the ERM operates in LEO and is reusable.

- Some ERM's will be parked at the fuel depot (Scenario 5) and participate in fuel scavenging operations from the STS (Scenario 6).

- Some ERM's will perform supply and/or top-off operations:

The major conclusions from the update of the requirements and scenarios is presented as follows:

- o Fluid transfer should be considered as an integral part of space infrastructure

- o Major customer for fluid transfer are other infrastructure elements: OMV, OTV, Space Station

    - o All potential users support concurrent servicing

- o Selection of MMH/NTO as baseline fluid still justified by user needs

    - o Need for resupply of other fluids
    - o Module sizing determined by operational scenarios

- o Decoupling of tankage from resupply module may be advantageous to development effort

    - o Adequate MMH/NTO available from OMV initially
    - o Focus on key technical/operational risk elements
    - o Lower front-end cost for timely development

The spectrum of resupply missions possible has been summarized into a set of operational scenarios with drivers of resupply module sizing. This results in a confirmation of the selection of MMH/NTO as the baseline fluid while down-sizing the earlier resupply module design (45,500 lbs of bi-propellant). An ERM of some 7 klb of bi-propellants was deemed sufficient, while the larger version serves as a growth version of the design.

A decoupling of tankage from the resupply module was discovered as a potential means for concentrating on the technical issues associated with fluid transfer while preparing flexibility in an evolving market for fluid transfer.

Concept Definition

The Concept Definition task of the supplemental study was divided into two major areas. These areas were: (1) the refinement of the transfer subsystem for the purpose of determining the engineering objectives of the fluid demonstration tests and (2) to develop a fluid demonstration test concept(s) which would satisfy the engineering test requirements.

Several technical areas were studied to refine system requirements for a propellant and pressurant transfer. The technical areas examined are as follows: umbilical purge, pressurant transfer, propellant pump selection, NASA quick disconnect development, and pressure and temperature instrumentation. The results of these studies are presented in Figure 7. Two flight test concepts were developed to provide proof of concept for on-orbit remote expendables resupply. The first is the Mid-Deck Ullage Transfer Experiment (MUTE) which is presented in Figure 8. MUTE's fundamental objective are as follows:

1) To demonstrate the concept of ullage transfer in a micro-gravity environment.

2) To demonstrate the propellant management devices (PMD's) ability to position and control the ullage bubble.

3) To demonstrate the quantity gaging system's accuracy in a micro-gravity environment.

The demonstration of a micro-gravity quantity gaging system is not critical to the success of this flight test. However, MUTE does provide an opportunity to conduct an on-orbit test of the micro-gravity gaging system that is being developed as a separate NASA program.

The second flight test concept is the payload bay Flight Demonstration Test Article (FDTA) which is shown in Figure 9. The Flight Demonstration and Ground Test Article will be used in the orbiter cargo bay to demonstrate a micro-gravity remote fluid transfer. FDTA consists of two platforms capable of being separated or docked using the remote manipulator subsystem (RMS). The upper portion of the test article, the receiver platform, piggy-backs on the supply platform through deployable payload latches. The RMS system is utilized to separate the receiver platform, by use of a grapple fixture, to simulate remote docking and mating of two free-flying spacecraft. A snare type end effector, incorporated at the fluid transfer interface of the supply vehicle, accomplished final mating of the fluid transfer interface panels on each test platform.

Should it become necessary to reduce the cost of the FDTA several options are available. One option is to conduct the tests with only one fluid either MMH or NTO. This enables a reduction in the size thereby lowering system weight and material/component procurement and fabrication costs.

Another approach to lowering flight demonstration cost is through the reduction in test objectives. Requirements analyses have indicated that a compressor may not be need during the early phases of on-orbit resupply operations. Elimination of the compressor from the advanced development

- UMBILICAL PURGE
  - DRIBBLE VOLUME PURGE
    - SIMPLE DESIGN
    - LOW SYSTEM WEIGHT
    - QUICK PURGE TIME
- PRESSURANT TRANSFER
  - RECOMMEND FOUR PRESSURANT BOTTLES
  - USE CASCADE APPROACH
- PROPELLANT TRANSFER PUMPS
  - RECOMMEND GEAR PUMP
  - MAGNETICALLY COUPLED PUMPS REQUIRE MASSIVE ELECTRICAL POWER
- QUICK DISCONNECT
  - AUTOMATED Q.D. DEVELOPMENT IN NASA PLANNING
- INSTRUMENTATION
  - TEMPERATURE
  - PRESSURE
    - NO NEW DEVELOPMENTS REQUIRED

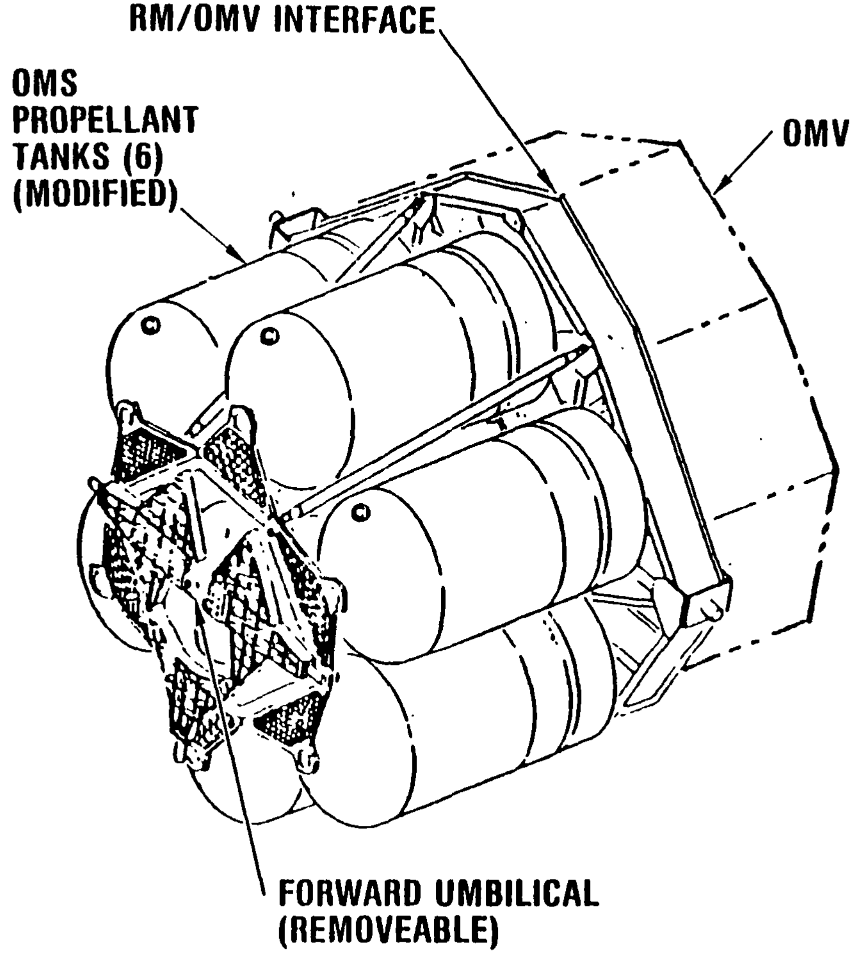
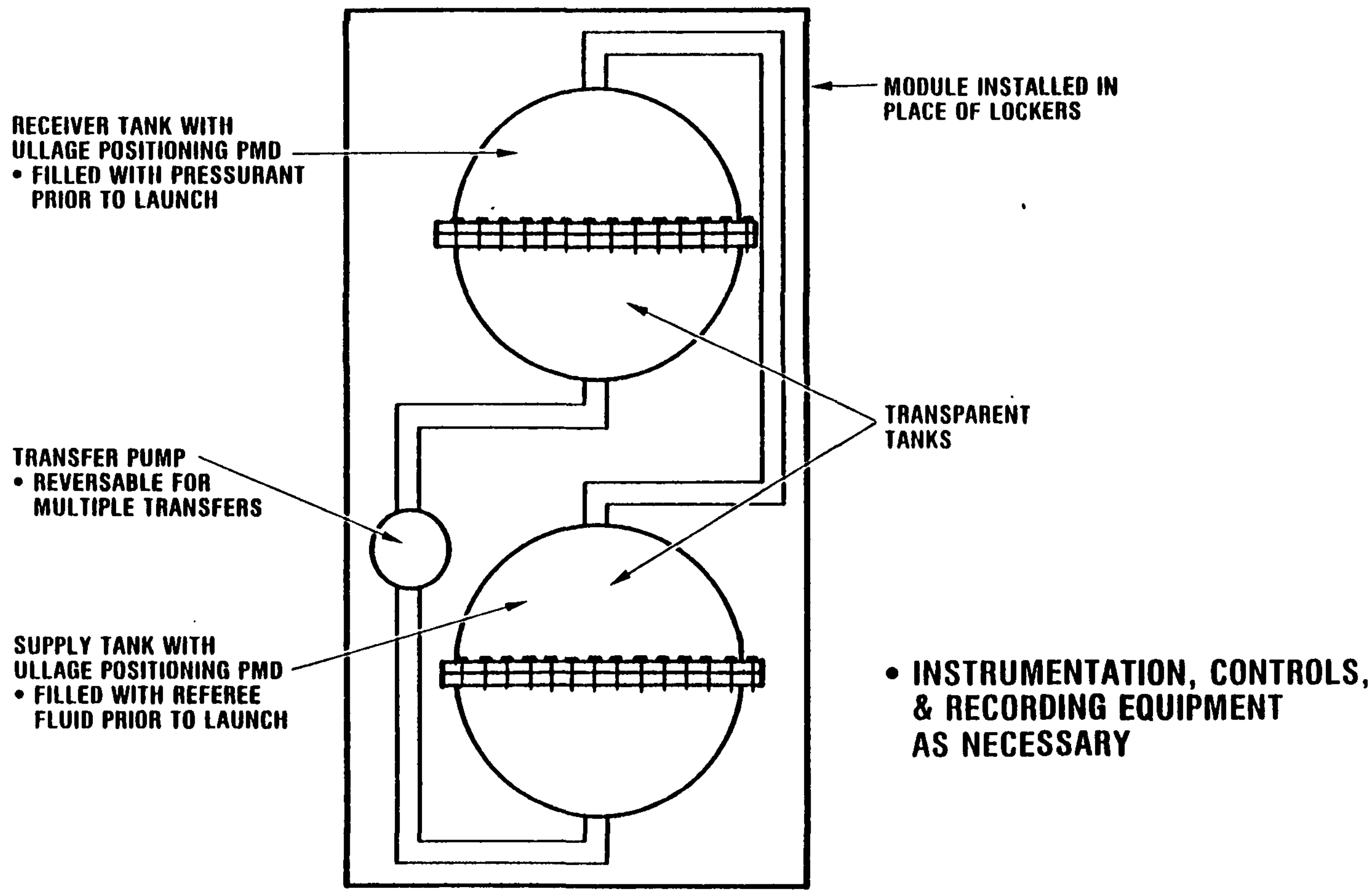
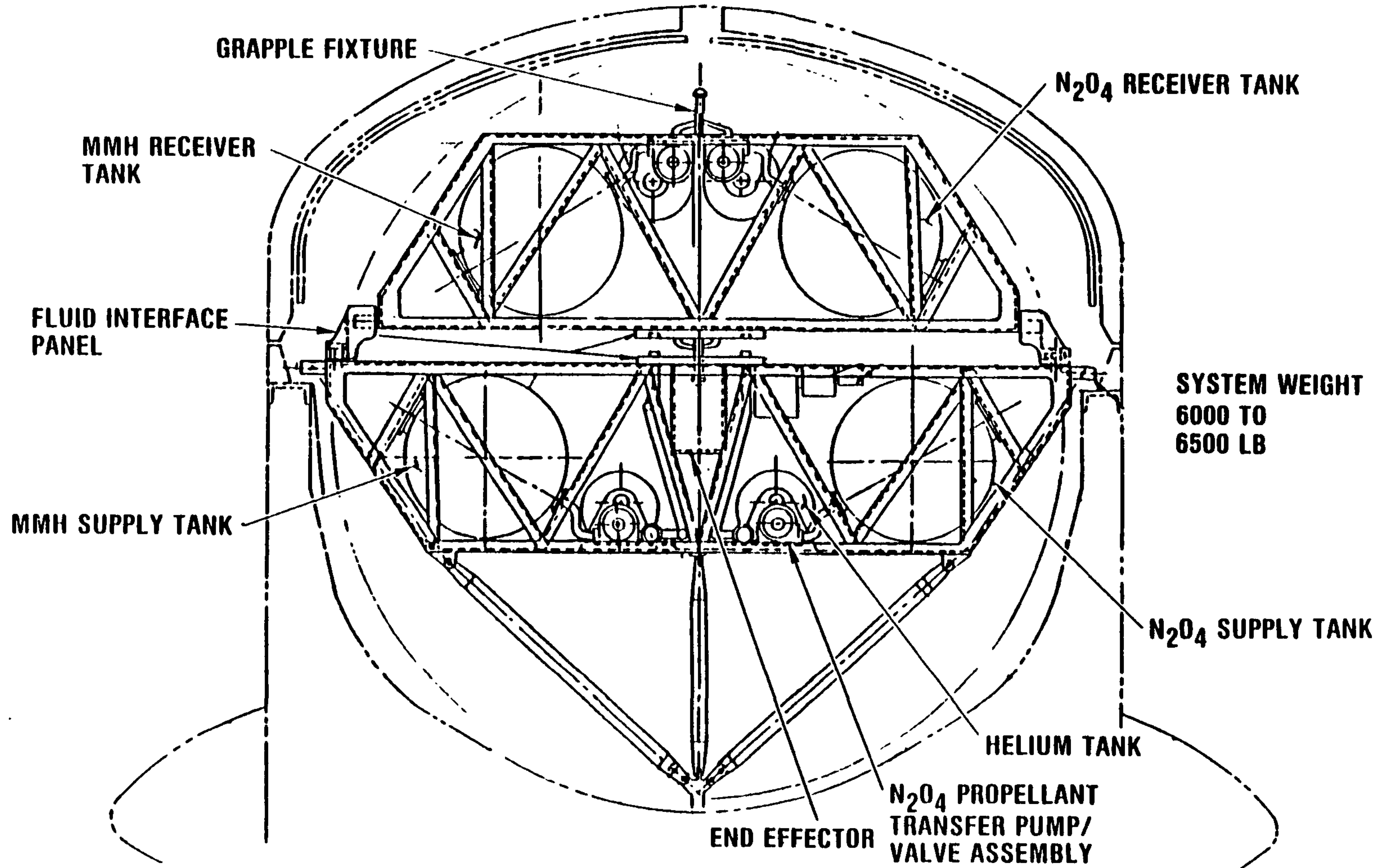


FIGURE 7. TECHNICAL STUDIES DETERMINE DEMONSTRATION DATA REQUIREMENTS



VERIFIES: PROPELLANT MANAGEMENT DEVICE (PMD) & QUANTITY GAUGING

FIGURE 8. MID-DECK ULLAGE TRANSFER EXPERIMENT



VERIFIES: PMD; QUANTITY GAUGING; PUMPS; Q.D. MATING, ALIGNMENT, & ENGAGEMENT; PRESSURANT TRANSFER; THERMAL; & SYSTEM CONTROL

FIGURE 9. PAYLOAD BAY FLIGHT DEMONSTRATION AND GROUND TEST ARTICLE

program and the flight test article could result in considerable program cost reductions. However, as on-orbit resupply operations mature this capacity will become required and therefore, must eventually be developed.

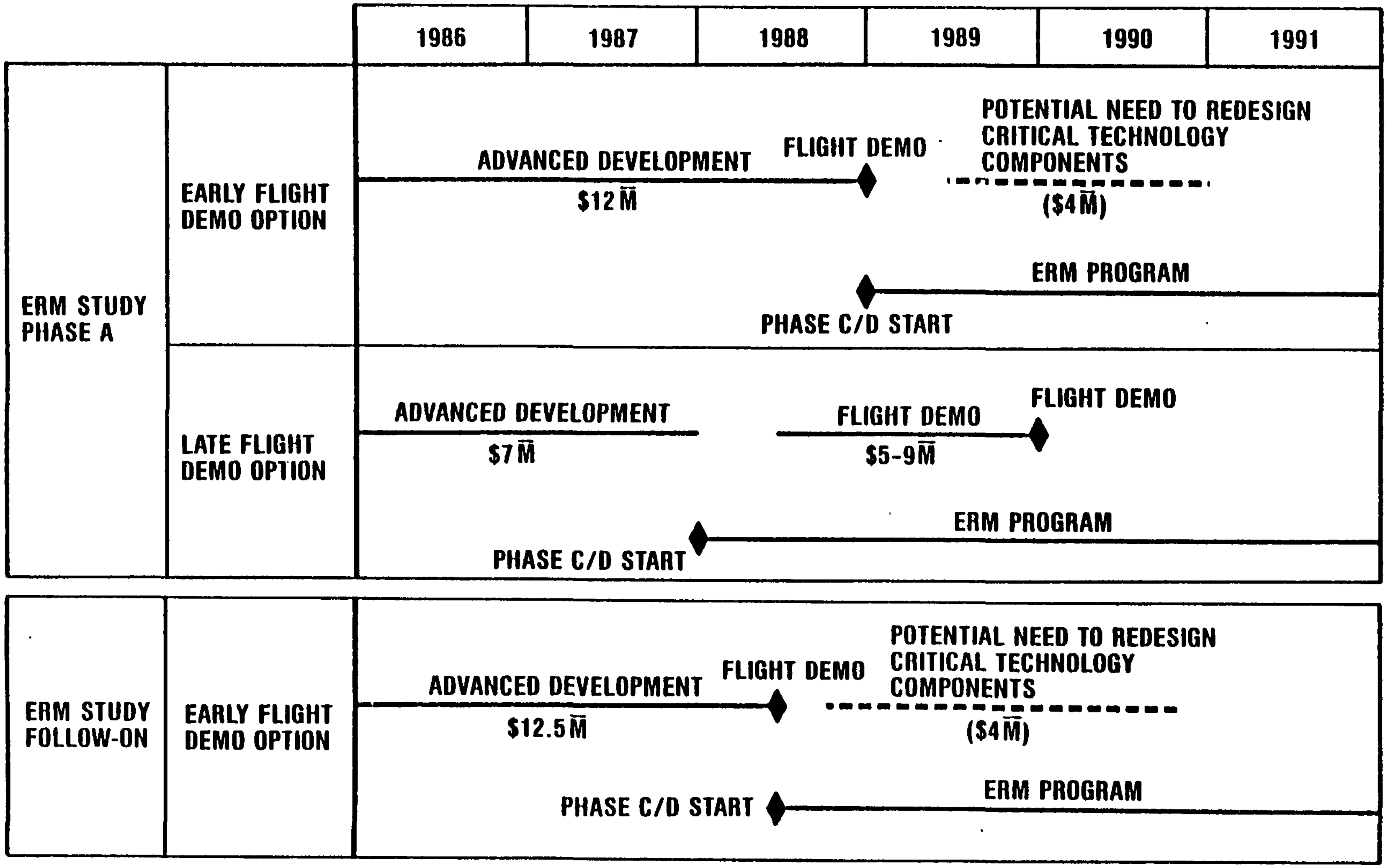
The concept definition task conclusions are summarized as follows:

- o Expendables Resupply Module (ERM) Design
  - o Pressurant purge of disconnects preferred
  - o Cascaded pressurant resupply preferred over compressor approach for small systems
  - o Gear pump most effective in ERM flow & pressure regions
  - o Disconnects & liquid quantity gauging being developed under other programs
- o Mid-deck Ullage Transfer Experiment (MUTE)
  - o Verifies propellant management device (PMD) & quantity gauging
  - o Simple, benign fluids, & max. use of existing equipment
- o Flight Demonstration Test Article (FDTA)
  - o Verifies PMD, pumps, disconnects, pressurant transfer thermal & system control
  - o Operational fluids & max. use of existing equipment
  - o Ground test and flight test

Programmatic and Development Planning

For purposes of comparability, the schedules for advanced development and flight demonstration recommended from the original and supplemental studies have been laid on a common calendar scale (Figure 10). In the initial study we recommended a late demonstration option, because it resulted in a one-year earlier ERM phase C/D start.

As a result, of our supplemental study, we now recommend an earlier flight demonstration option. This is accomplished by using the FDTA as the ground systems test and verification test bed. This resulted in an earlier ERM phase C/D program at approximately the same cost.



ROCKWELL RECOMMENDS THE EARLY FLIGHT DEMONSTRATION OPTION

FIGURE 10. COMPARISON OF ADVANCED DEVELOPMENT/FLIGHT DEMONSTRATION OPTIONS

The ideal schedule leading to the flight demonstration and, hence, the ERM program start is presented in Figure 11. The initial effort is to specify the hardware requirements for the ERM program. With the ERM requirements defined, the design of the flight experiments and the development of the critical technology components can start. Ideally, MUTE can fly in 22 to 24 months after the program starts and the flight demonstration test is scheduled to fly 8 to 12 months afterwards.

The timing in development of several related fluid transfer components could affect this schedule. The micro-gravity gaging system (NASA, JSC) would be an ideal component to include on the Flight Demonstration Experiment. If the ERM and the micro-gravity gaging system development schedules could be made compatible then, could be the Flight Demonstration Experiment expenses could be shared. The function of the micro-gravity gaging system, if not available for the Flight Demonstration Test, could be performed by flow meters.

The Quick Disconnect (NASA, JSC) is essential to demonstrate remote resupply in the Flight Demonstration Experiment. It is currently scheduled to be available in mid-FY 1988, and its schedule may also need to be accelerated slightly or the advanced development program stretched slightly.

The programmatic and development planning conclusions are presented as follows:

- o Early Flight Demonstration - Before ERM phase C/D start
- o Two Experiments - Mid deck ullage experiment & flight demonstration test (orbiter cargo bay experiment)
- o An Integrated Program - Same team for development & test
  - Same development & flight hardware
- o $12.5M Budget - for Advanced Development & Flight Demonstration Programs

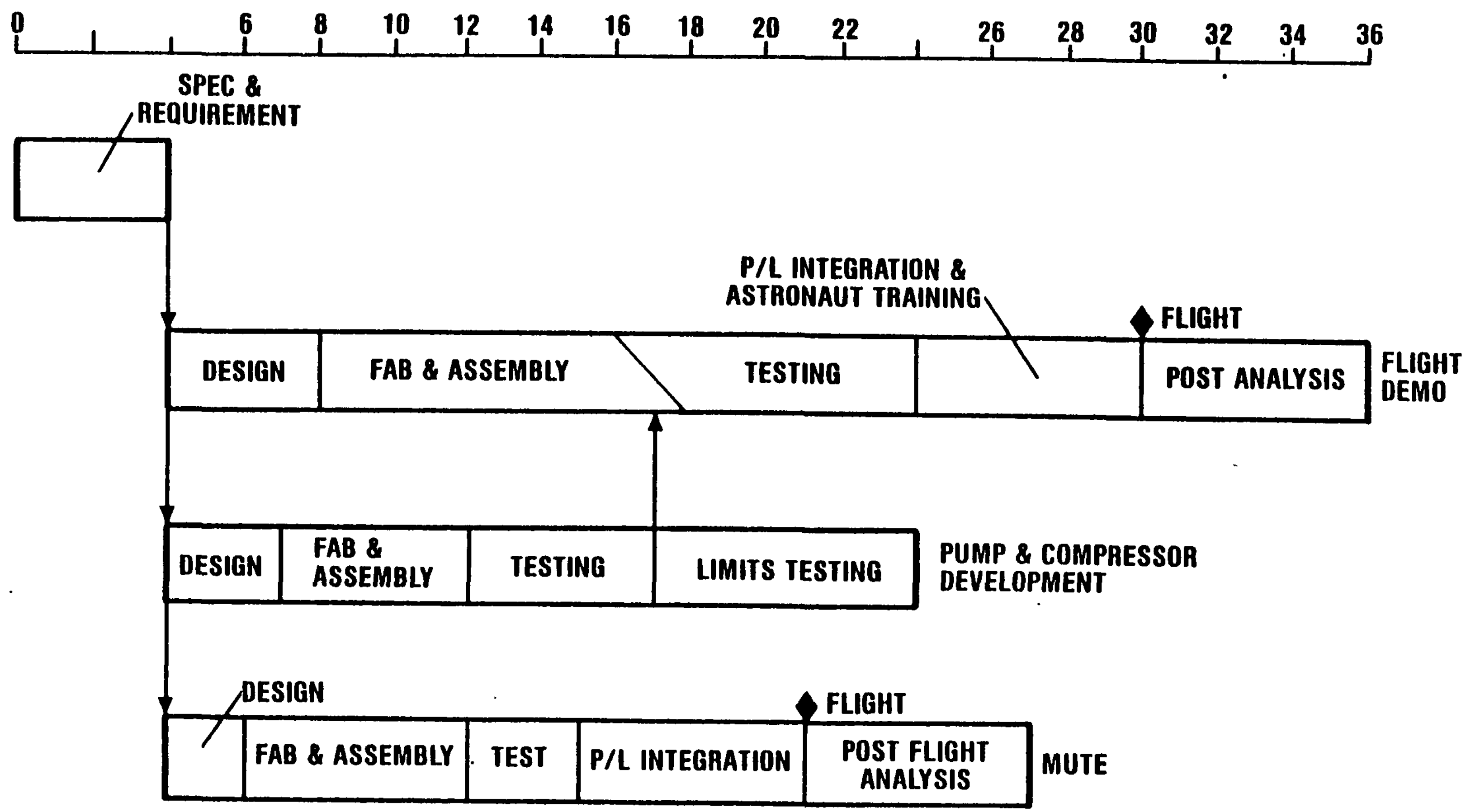


FIGURE 11. ADVANCED DEVELOPMENT AND FLIGHT DEMONSTRATION PROGRAM (MASTER SCHEDULE)

2.0 Study Results

2.1 System Requirements and Scenarios (Task 1)

A review of the task 1 study results is presented in this section of the final report. The topics discussed in this review are the subtasks for task 1 which are summarized as follows:

1) Expand mission model to include space infrastructure elements
2) Define parametric fluid quantities as determined by mission model
3) Survey Potential on-orbit resupply users
4) Reevaluate ERM scenarios from original study
5) Identify ERM sizing options and their driving requirements

Figure 12 depicts an evaluation of key elements of the space infrastructure. These elements depicted are the ones of primary interest to the task of expendables resupply. In accordance with the expanded scope of this supplemental study, a closer look has been taken at the resupply needs and interactions of infrastructure elements.

In this analysis we have the Shuttle being used as the basis for an in-bay propellant tank (under JSC) and later the arrival of the OMV with associated kits, such as the resupply module, servicer kit and capture kit. The OMV and associated kits can be deployed from the ground, and/or space-based at 28.5 degrees or polar orbits. These spacecraft will then be augmented by permanent orbital facilities primarily the space station with its accommodations for OMVs and OTVs. Associated with the station, but separate from it for safety requirements will be a Leo depot, possibly supplied by propellant scavenging from the STS. Eventually, this depot should be duplicated in polar orbits with its own OMV and associated kits.

Around the turn of the century, new developments such as the shuttle shock change and a fully reusable cryogenic OTV will likely alter the economics of supplying propellant in space. Certainly, the Shuttle block change will affect the use of propellant scavenging, and the OTV will generate requirements for greater amounts of cryo fuels, raising the need for sophisticated storage facilities in space. A secondary space transport element, the Unmanned Launch Vehicle, will likely arrive in support of SDI deployments and increased lunar surface activity. The logistics for these initiatives are not sufficiently developed yet to predict their impact on orbital fluid requirements and have, therefore, been left out of this projection.

The expanded mission model (Table 1) reflects a current assessment of the future demand for operational fluid transfer missions. Utilizing resupply modules, OMV's and OTV's, fluid transfer operations are carried out to on-orbit spacecraft and facilities. This model integrates the results of separate OMV, OTV, and Space Station mission models into an identification of fluid users and a schedule of missions. Many of the missions are not purely fluid resupply, but resupply missions conducted in conjunction with other satellite servicing tasks. It is not clear what the division will be between fluid transfer operations occuring at the Orbiter and those occuring in a remote in situ mode. This revision of the orbital fluid supply mission model incorporates the updates and schedule changes since its last publication in

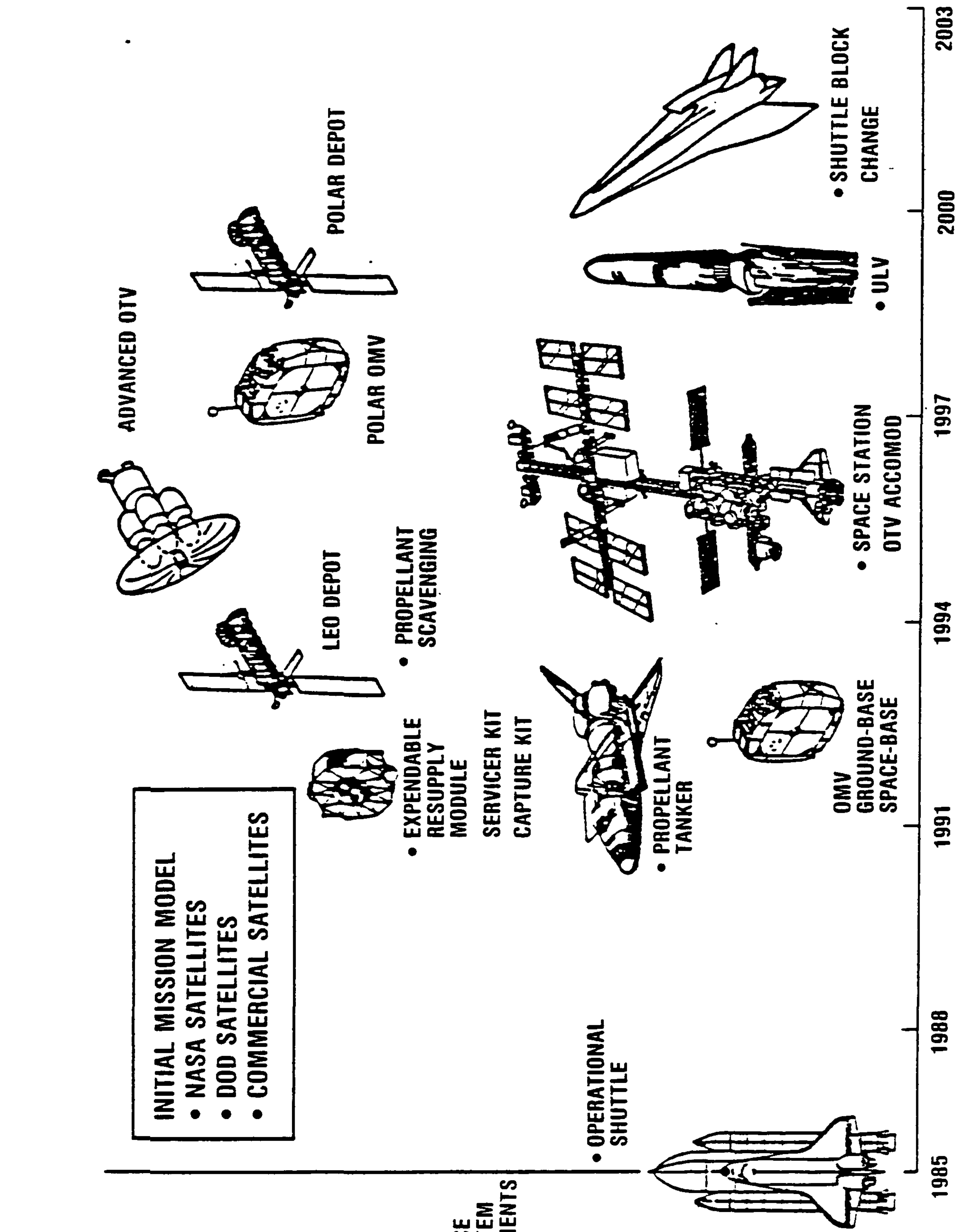


FIGURE 12. REQUIREMENTS DEPEND ON EVOLVING SPACE OPERATIONS

MISSION CATEGORY	89	90	91	92	93	94	95	96	97	98	99	00	01	TOTAL
MAINTENANCE		2	5	9	10	10	10	6	7	3	5	5	5	77
SPACE-BASED OMV					1	4	2	6	1	4	2	2	1	23
SPACE-BASED OTV							4	5	4	5	4	6	5	33
SPACE STATION				4	4	4	4	4	4	4	4	4	4	40
DOD			3	2	2	3	2	5	7	3	4	4	4	39
LEO TOTAL		2	8	15	17	21	22	26	23	19	19	21	19	212
GEO						1	3	5	6	6	8	10	5	44
GRAND TOTAL		2	8	15	17	22	25	31	29	25	27	31	24	256
CUMULATIVE TOTAL		2	10	25	42	64	89	120	149	174	201	232	256	

TABLE 1. MISSION MODEL INTEGRATES ALL FLUID TRANSFER USERS

September of 1984. Changes have been relatively minor; involving shifts in the communications market, new DOD launch schedules, and the addition of some potential foreign missions. The greatest uncertainty involved in this assessment is the impact of Space Station operations.

Based on the expanded mission model Figure 13 shows a projection of all on-orbit fluid transfer engagements for the 1989-2001 time frame. These engagements include the needs of the DoD, large platforms, space station, space-based OTV's and OMV's, as well as other NASA and commercial spacecraft. Engagements allocated to OMV servicing missions means the use of OMV as a carrier vehicle for the resupply module. Materials processing in space (MPS) engagements are based on estimates of factory modules aboard free-flying leasecraft. The number of engagements decline in the latter time frames as MPS operations transition to space station proximity. Further fluid transfers are then accomplished through routine station resupply flights. SDI needs are currently unknown.

Parametric techniques were used to make estimates of the types, amounts and yearly fluid usage rates required by the resupply mission model (Table 2). 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 resupply technique used, no additional pressurants are required for the transfer process. Figure 14 depicts, in graphic for the same information.

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 bi-propellant are the dominant required fluids. In addition, another significant fluid of interest was found to be liquid (primarily superfluid state) helium.

In updating the resupply mission, extensive contacts were made with the potential user community. In addition to in-house data base and literature searches, three related surveys were conducted. They were telephone contacts with various NASA program contacts used in the development of the resupply mission model. The updated data did not contain any major surprises and represented a maturing of several programs which may hold a more positive attitude fluid resupply. In January of 1985, a series of presentations were made to satellite manufactures TRW, Hughes, and Ford. The presentations were intended to direct feedback on the proposed demonstration program and financial analysis. A survey question, prepared under Rockwell IR&D, was send to over 90 members of the space community in the U.S. and aboard. The contacts included domestic and foreign space agencies, aerospace and communications companies, and space insurers. The survey attempted to find new candidate users of fluid transfer services for either remote or Orbiter-based service. Thirteen useful responses have to be received to date, with most supporting the concept of concurrent servicing with resupply. A summary of the results of these surveys is presented in Figure 15.

Table 3 presents a more detailed chart of NASA missions with potential requirements for the use of ERM. The data for this survey resulted from telephone contacts with approximately 20 programs or project offices. Some additional information was obtained from the NASA Space Station mission data. Many of the missions are expected to be resupplied by Space Station, using an OMV. A total of twelve missions were identified on being candidate users.

USER AND CONSUMABLE TYPE	RESUPPLY AMOUNT (LB) 1990	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000	2001
NASA/MPS/Foreign												
Hydrazine	510	4630	1640	5760	2930	4960	1130	5319	2932	5100	1349	5100
Liquid						1675		1278		2382		2382
Water	5000	15000	25000	30000	30000	25000	15000	5000				
Commercial GEO												
Hydrazine								516	932	1291	3770	282
MMH/NTO						500		500	500	500	500	500
DOD LEO												
N204/A50		14000	14000	7000	21000	7000	28000	28000	21000	7000	7000	21000
Hydrazine		70		70		70	8700	17470		17470	17400	70
Liquid He							1200	2400		2400	2400	
DOD GEO												
MMH/NTO					1000	2000	2000	2000	1000	1000	1000	
Hydrazine							3900	2600	1300	3900	1300	3900
Contingency			1500		1500		1500		1500		1500	
OMV												
MMH/NTO				6000	24000	12000	36000	6000	24000	12000	12000	6000
OTV												
LO2/LH2						119400	202900	108050	166150	96150	157400	176350
MMH/NTO					45500	45500	45500	91000	91000	91000	91000	91000
Space Station												
Hydrazine			13040	13040	13040	13040	13040	13040	13040			
LO2/LH2										6800	6800	6800
Water			1800	1800	1800	1800	10000	20000	25000	25000	25000	25000
Total Hydrazine	510	4700	14680	18870	15970	18070	26770	38945	18204	27761	23819	9352
Total MMH/NTO				6000	70500	60000	83500	99500	116500	104500	104500	97500
Total N204/A50		14000	14000	7000	21000	7000	28000	28000	21000	7000	7000	21000
Total LO2/LH2						119400	202900	108050	166150	96150	164200	183150
Total Water	5000	15000	26800	31800	31800	26800	25000	25000	25000	25000	25000	25000
Total Liquid He						1675	1200	3678		4782	2400	2382

TABLE 2. RESUPPLY REQUIREMENTS SUMMARY

	PROJECTED LIFE (YRS)	ALTITUDE (nmi)	INCLINATION (degrees)	LENGTH (ft)	DIAMETER (ft)	MASS (lbs)	NO. OF VEHICLES	INITIAL LAUNCH DATE	CONSUMABLES (TYPE)	QUANTITY (lbs)	COMMENTS
ADVANCED X-RAY ASTROPHYSICS FACILITY (AXAF)	15.0	324	28.5	43	13	18,950	1	1992	LIQUID HELIUM ARGON, ZENON CO_2		RESUPPLY EXPECTED 216 nmi ALTITUDE AFTER 3 YRS
EARTH OBSERVING SYSTEM (EOS)	10.0	381	98.25	59	39 × 13	22,030	3	1992	CRYOGENS	1100	RESUPPLY EXPECTED
EXTREME ULTRA-VIOLET EXPLORER (EUVE)	1.0	300	28.5				1	10-88	NONE		ON PLATFORM SERVICES SATELLITE (PSS) RFP OUT ON PSS
GAMMA RAY OBSERVATORY (GRO)	5.0	270	28.5	25	15	31,000	1	5-88	HYDRAZINE	5000	
GEOSTATIONARY PLATFORMS (GEO)		19,323	0.0	U	U	U		U			VERY PRELIMINARY
GRAVITY PROBE-B (GP-B)	1.0	520	90.0	8	15	4,000	1	EARLY 90'S	LIQUID HELIUM		RESUPPLY NECESSARY ONLY IF PROGRAM IS SUCCESSFUL & LAUNCH IS SUCCESSFUL
LARGE DEPLOYABLE REFLECTOR (LDR)	10.0	500	28.5	131	115	121,170	1	1997	SUPERFLUID HELIUM	552	PERIODIC SERVICING BY OMV
MATERIAL PROCESSING IN SPACE (MPS)	INDE-FINITE										6 MONTH RESUPPLY PLANNED FOR
OCEAN CIRCULATION TOPOGRAPHY EXPERIMENT (TOPEX)	3.0-5.0	720	63.4	U	U	U	1	1992	HYDRAZINE	510	3 YR ESTIMATE OF CONSUMABLE USE. NO RESUPPLY EXPECTED
PROTEUS PLATFORM	10.0	216	28.5	3.3	15	2,203	2	1992	HYDRAZINE	551	2 YR USE OF CONSUMABLES. RESUPPLY REQUIRED
SHUTTLE INFRARED TELESCOPE FACILITY (SIRTF)	5.0	378	28.5	28	13	8,812	1	LATE 1993	SINGLE-CRYOGEN SUPERFLUID HELIUM	1,278	3 YR USE OF CONSUMABLES. RESUPPLY REQUIRED. POSSIBLE EXTENSION TO 10-15 YRS, IF SUCCESSFUL
UPPER ATMOSPHERE RESEARCH SATELLITE (UARS)	1.5	320	57.0°	21	15	11,000	1	OCT 1989	SOLID HYDROGEN	661	POSSIBLE EXTENSION OF 1 YR

TABLE 3. UPDATE OF SPER MODULE MISSION MODEL DATA BASE - NASA

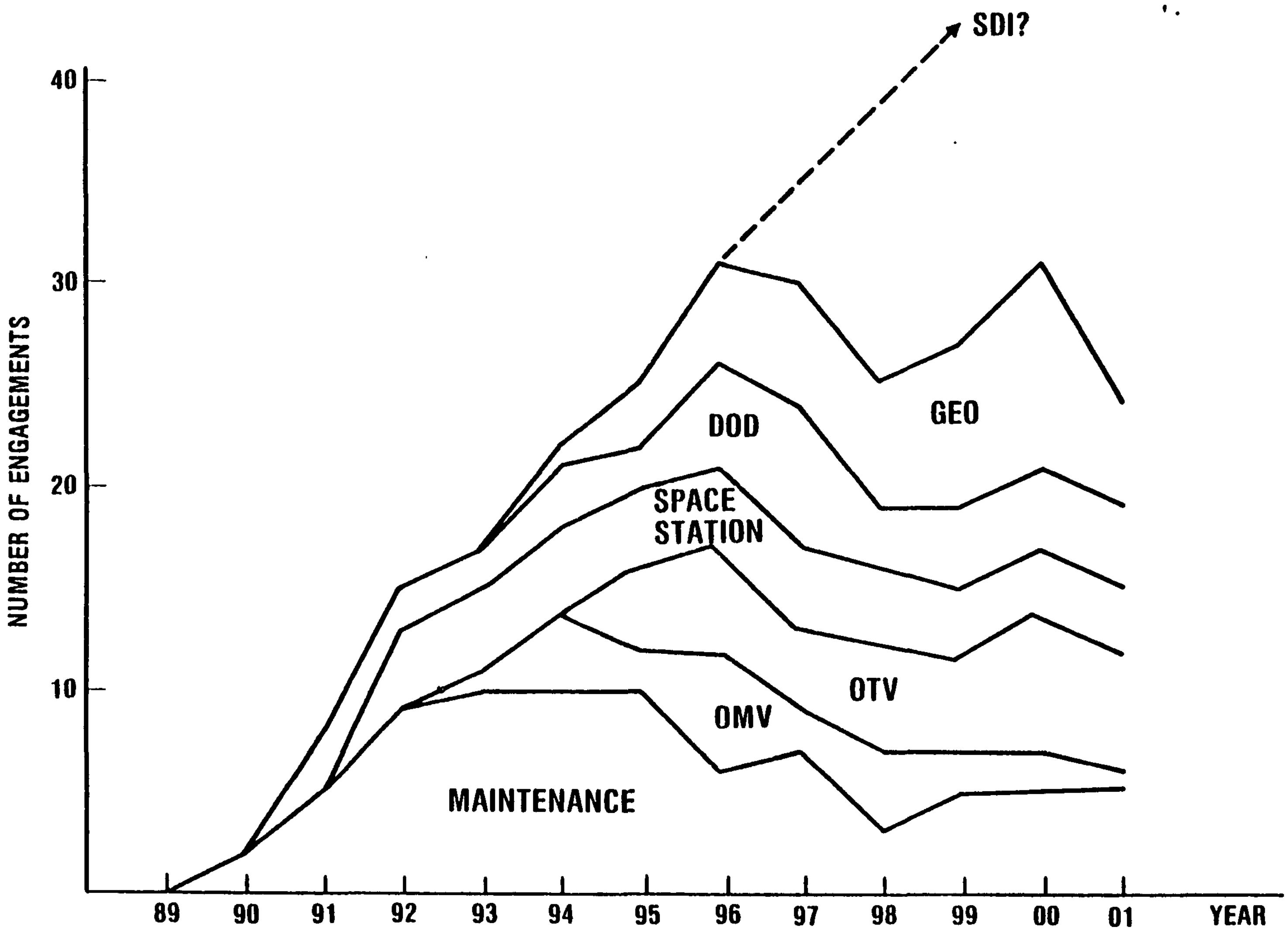


FIGURE 13. PROJECTED ANNUAL FLUID TRANSFER ENGAGEMENTS

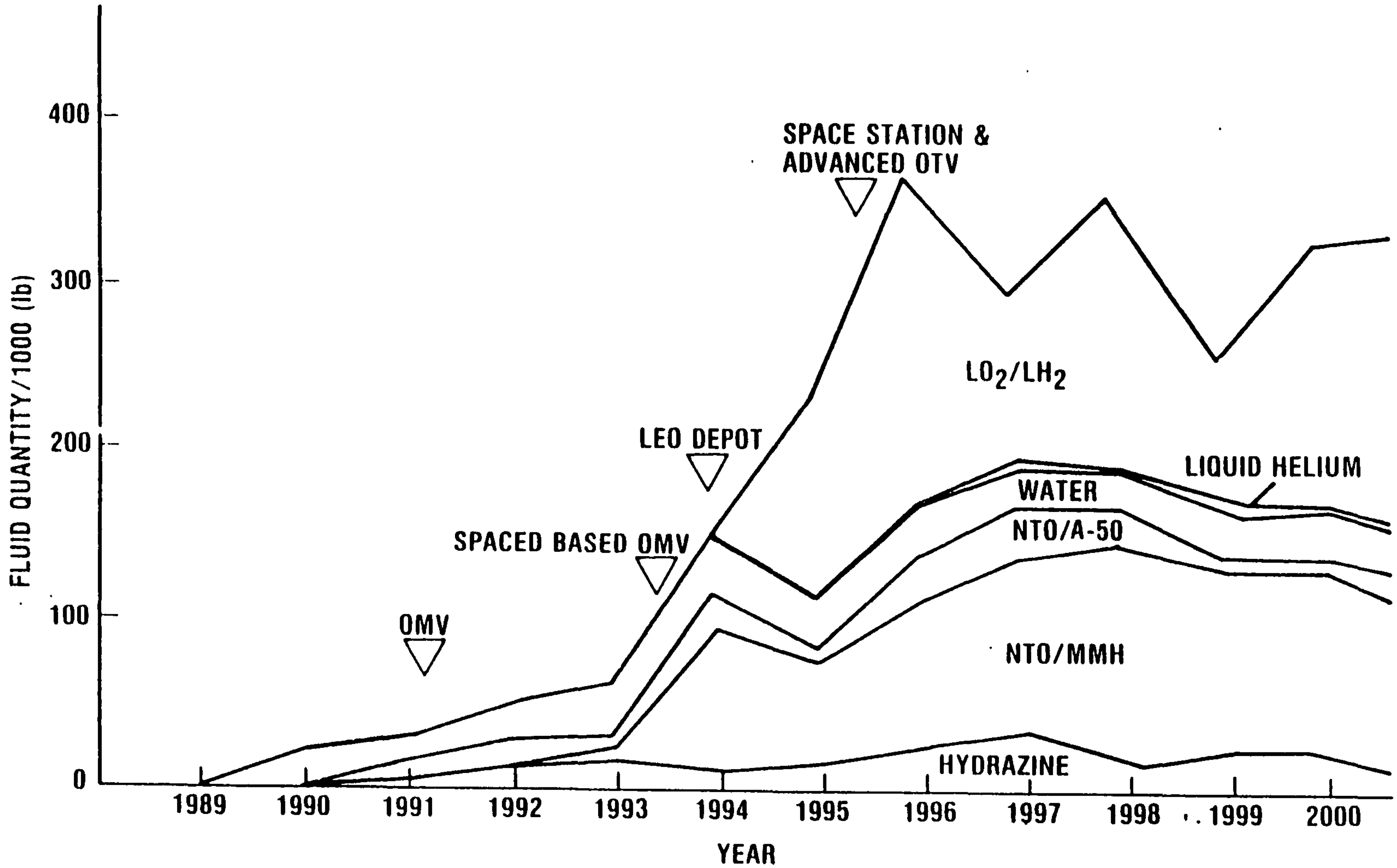


FIGURE 14. FLUID REQUIREMENTS DRIVEN BY SPACE OPERATIONS

TELEPHONE CONTACT

- NASA SCIENTIFIC
- COMMERCIAL MPS

- ABOUT 20 PROGRAM OFFICES CONTACTED
- 12 POSITIVE RESPONSES
- UPDATE OF ORIGINAL STUDY SURVEY
- SPECIFIC RESULTS
  - MORE POSITIVE ATTITUDE

PRESENTATIONS TO SATELLITE SUPPLIERS

- TRW, HUGHES, FORD
- SPECIFIC RESULTS
  - CONTRACTORS INTERESTED IN APPROACH
  - SUPPORTED TECHNICAL FEASIBILITY

SURVEY QUESTIONAIRE

- SENT TO DOMESTIC & FOREIGN SPACE AGENCIES, AEROSPACE & COMMUNICATIONS COMPANIES, & INSURERS
- 90+ QUESTIONAIRES SENT
- 25 RESPONDED TO DATE
- 13 USEFUL RESPONSES
- SPECIFIC RESULTS
  - STRONG SUPPORT OF CONCURRENT SERVICING

SPECIFIC USER NEEDS & COMMITMENT BEING IDENTIFIED

FIGURE 15. THREE TYPES OF SURVEYS CONDUCTED

A series of on-site presentations were made to a series of satellite suppliers. At the request of NASA HQ, ERM contract results were presented, with emphasis on Geo satellite economic analysis, technical inputs to resupplied satellites, and acceptable resupply demonstrations. A series of technical and economic comments were received and documented, some comments were common to more than one supplier. Summary conclusions from these contacts are presented as follows:

- o Need sophisticated resupply/services module and logistics support system
- o Some propellant transfer advanced development necessary
- o Early performance of flight test segments appears to be a sufficient demonstration
- o Near-term resupply of government spacecraft should convince insurers and satellite users of concept viability
- o Directly involve satellite suppliers for further technical/economic studies

Over ninety survey forms, related to potential users of fluid transfer services, were sent out in May. The contacts included domestic and foreign space agencies, aerospace and communications companies, and space insurers. Of the twenty-one responses received to date, twelve were considered to have useful data. Figure 16 summarizes those contacts which submitted useful responses and what the general nature of these comments were. No major surprises were received, although some speculative missions were identified. Strong interest in Orbiter-based resupply was voiced for the Canadian Radarsat program and remote resupply for the TDRS program, depending on potential costs. Tables 4 and 5 list recipients of the fluid transfer survey. The check marks denote who provided responses and the check mark in parenthesis denotes a response still in work. No check mark denotes no further contact of this date.

Based on the resupply mission model and the potential users contacts six ERM scenarios were developed (Figure 17). In Scenario 1, 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. Propellant transfer (and concurrent module exchange servicing, when required) is accomplished directly from the OMV after separation from the resupply module in Scenario 2. This uses a reusable cryogenic OTV for Geo transport.

In Scenarios 3 through 6, the resupply module operates in LEO and is reusable.

- Some ERM's will be parked at the fuel depot (Scenario 5) and participate in fuel scavenging operations from the STS (Scenario 6).
- Some ERM's will perform supply and/or top-off operations.

U.S. AEROSPACE

- BOEING
- (✓)TRW
- FORD AEROSPACE
- ✓ HUGHES SPACE & COMM
- MORTION-THIOKOL
- ✓ MARTIN-MARRIETTA
- ✓ GRUMMAN AEROSPACE
- FAIRCHILD SPACE CO
- LOCKHEED MISSILES & SPACE CO
- ✓ RCA GOVT SYSTEMS DIV/ASTRO-ELECTRONICS
- EAGLE ENGINEERING
- SPACE SERVICES INC
- ✓ ASTROTECH
- GENERAL DYNAMICS
- McDONNELL-DOUGLAS ASTRONAUTICS
- OAO CORP
- ORBITAL SYSTEMS LTD
- TELEDYNE BROWN ENGINEERING
- ORBITAL SCIENCES CORP
- SPACE AMERICA
- BELL AEROSPACE SYSTEMS DIV
- WESTINGHOUSE ELECTRIC
- VOUGHT CORP

U.S. COMMUNICATIONS

- FEDERAL EXPRESS
- AT&T
- SKYLINK
- SBS
- ✓ SPACE COMM CO
- INTELSAT
- U.S. SATELLITE SYSTEMS
- WESTERN UNION

INSURERS

- MARSH & McLENNAN
- INTEC

CONSULTANTS

- CENTER FOR SPACE POLICY
- CONGRESSIONAL RESEARCH SERVICE
- BOOZ-ALLEN & HAMILTON
- BATTELLE LABORATORIES
- PLANNING RESEARCH CORP
- ✓ DCS CORP
- ✓ SCIENCE APPLICATION INC
- ANALYTIC PRODUCTS
- SPACE TECHNOLOGY CENTER
- JOHN BOSMA (MILITARY SPACE)
- CRAIG COVAULT (AW+ST)
- LEONARD DAVID (SPACE WORLD)
- DAVE DOOLING (HUNTSVILLE TIMES)

TABLE 4. COMMERCIAL FIRMS/CONSULTANTS SURVEYED

DOD

✓	USAF SPACE DIVISION (12 SENT, 8 RESPONSES)
	HQ USAF SYSTEMS COMMAND
	SPACE COMMAND
	KIRKLAND AFB/AFCMD
	OFFICE OF THE SECRETARY USAF
	AEROSPACE CORP
	NAVY ELECTRONICS SYSTEM COMMAND
✓	SPACE & NAVAL WARFARE SYSTEMS COMMAND
(✓)	NAVAL RESEARCH LABS

OTHER USG

	JPL
	ARC
	GSFC
✓	NOAA — DEPT OF COMMERCE

FOREIGN GOVT

✓	NASDA (JAPAN)
✓	MITI (JAPAN)
(✓)	EMBASSY OF FRANCE
✓	CNES-SPOT (FRANCE)
✓	CANADA
	ESA
	ERNO-USA (W. GERMANY)

NOTE: ✓ — RESPONSE RECEIVED
(✓) — RESPONSE IN WORK

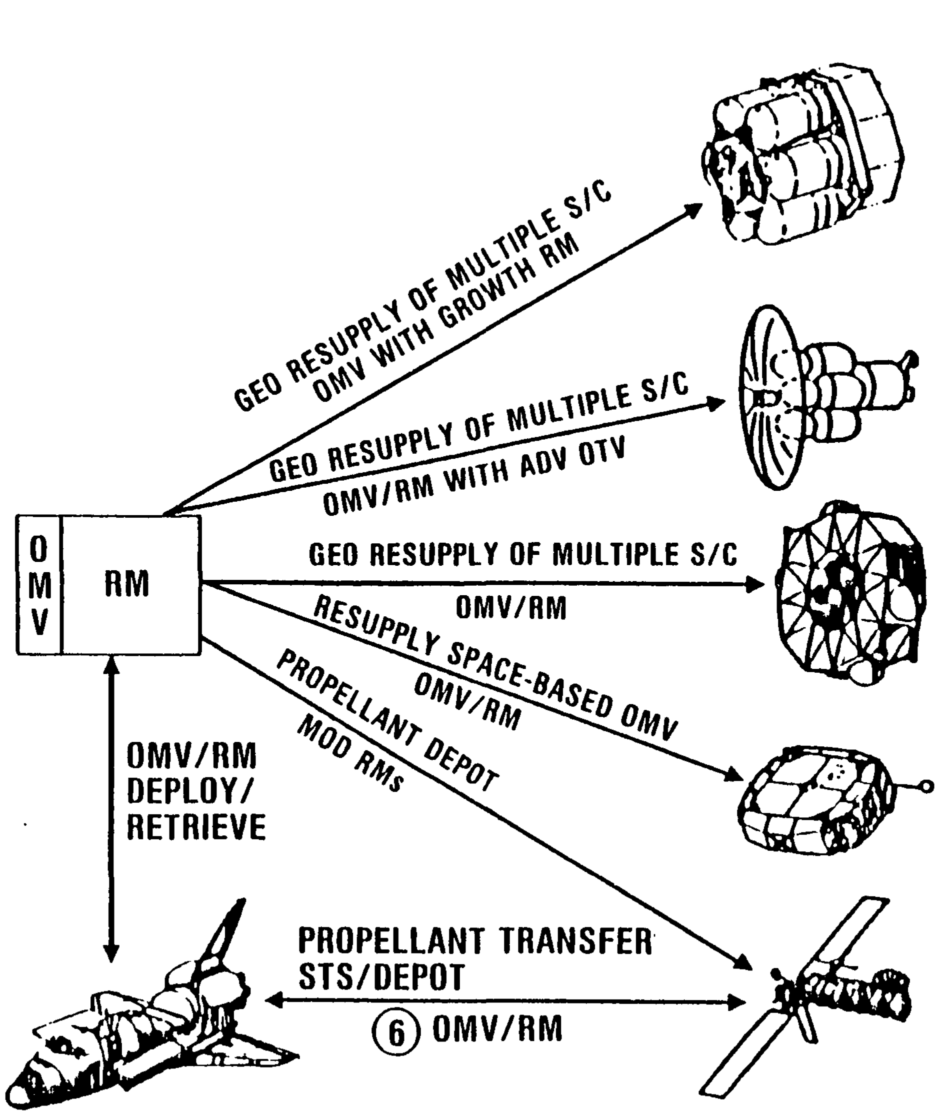
TABLE 5. UNITED STATES GOVERNMENT AND FOREIGN GOVERNMENTS SURVEYED

90+	SURVEY FORMS SENT (MAY '85)
25	RESPONSE TO DATE (JULY '85)
13	USEFUL DATA RESPONSES

USEFUL RESPONSES	COMMENTS
U.S. COMPANIES	
SPACE COMMUNICATION CO	RESUPPLY OF TDRS
HUGHES SPACE & COMMUNICATIONS GROUP	ORBITAL RESUPPLY NOT COMPETITIVE WITH GROUND LOADING
ASTROTECH	INTERESTED IN STUDY RESULTS
DCS CORP	RESUPPLY OF "SPACE CRUISER"
SCIENCE APPLICATIONS INC	RESUPPLY SUPPORT OF PROBE TO MARTIAN MOONS
RCA	MULTIPLE SPACECRAFT; VARIOUS ALTITUDES & INCLINATIONS; FIRST SERVICING 1992
U.S. GOVERNMENT	
NOAA — DEPT OF COMMERCE	SUPPORTIVE OF RESUPPLY NOAA POLAR PROGRAMS
SPACE & NAVAL WARFARE SYSTEMS COMMAND	NO REQUIREMENTS, POTENTIAL USAGE YEAR 2000+
USAF SPACE DIVISION	INTEREST FROM GPS, DMSP, P80-1, SBL
CANADIAN GOVERNMENT	RESUPPLY OF RADARSAT

FIGURE 16. RESUPPLY SURVEY QUESTIONAIRE RESULTS TO DATE (OCTOBER 1985)

3474e



MISSION SCENARIO NO.	RM BI-PROP (lbs)	RM HELIUM (lbs)	
1	45,500 (GROWTH)	30-90	GEO
2	3,000	TBD	GEO
3	4,000	TBD	LEO POLAR/ 28.5°
4	7,000 (BASELINE)	12	LEO POLAR/ 28.5°
5	20,000 (MULTIPLE RMs)	TBD	LEO POLAR/ 28.5°

MAY 1985

FIGURE 17. USER NEED DETERMINES SPER MODULE SIZING

From examining the resupply traffic model, fluid requirements and operational scenarios, a variety of sizes are possible for the resupply module (Table 6). Our perception of the sizing requirement for he Orbiter in-bay tanker has been included for comparison purposes. The range of sizes for all these options are roughly similar (i.e. 2-5 klbs for hydrazine, 2-7 klbs for bi-propellant) as the same general market is being serviced. Operational distinctions, such as the need for concurrent servicing, Orbiter flexibility, and resupply locations, will determine the type of remote or in-situ resupply.

The major conclusions from Task 1 are summarized as follows:

- Fluid transfer an integral part of space infrastructure
- Major customer for fluid transfer are other infrastructure elements: OMV, OTV, Space Station
  - Other potential users support concurrent servicing
- Selection of MMH/NTO as baseline fluid still justified by user needs
  - Need for resupply of other fluids
  - Module sizing determined by operational scenarios
  - Decoupling of tankage from resupply module maybe advantageous to development effort
    - Adequate MMH/NTO available from OMV initially
    - Focus on key technical/operational risk elements
    - Lower front-end cost for timely development

The updating of the mission model provided no surprise but the expansion of contract scope to include space infrastructure assets emphasized major area of fluid transfer requirements. End user satellites typically required concurrent servicing with resupply, while assets such as the OTV, OMV and Space Station required resupply along (while acquiring their payloads on separate mission). The spectrum of resupply missions possible was summarized into a set of operational scenarios with drivers of resupply module sizing. This resulted in a confirmation of the selection of MMH/NTO as the baseline fluid while down-sizing the earlier resupply module design (45,500 lbs of bi-propellant). An ERM of some 7 klb of bi-propellants was deemed sufficient, while the larger version serves as a growth version of this design. The decoupling of tankage from the resupply module was also discovered as a potential means for concentrating on the technical issues associated with fluid transfer while preparing flexibility in an ending market for fluid transfer.

2.2 Concept Definition (Task 2)

A review of the Task 2 Study results is presented in this section of the final report. A summary of the eight subtasks is listed below.

1) The conceptual propellant and pressurant resupply system defined in support of the Space Platforms Expendables Resupply NASA Contract Study (NASA8-35618), has been reevaluated and revised to incorporate

ORBITER IN-BAY TANKER	1990-2010(?) FLUID TYPE: AMOUNT: DRIVER:	HYDRAZINE 4-5 klbs GRO & 1 OTHER S/C	MMH/NTO 6-7 klbs 2-3 S/C	PRESSURANT TBD TECHNICAL
RESUPPLY MODULE/ OMV	LEO OPERATIONS FLUID TYPE: AMOUNT: DRIVER:	1990-2001 HYDRAZINE 4 klbs 2-3 S/C RESUPPLY	MMH/NTO 6-7 klbs 1 OMV FUELING	PRESSURANT TBD TECHNICAL
RESUPPLY MODULE/ OMV	GEO OPERATIONS FLUID TYPE: AMOUNT: DRIVER:	1995-2001 HYDRAZINE 4 klbs DOD/COMMERCIAL S/C 1-2 YEARS DURATION	ADV OTV ASSUMED MMH/NTO 2-3 klbs DOD/COMMERCIAL S/C 1-2 YEARS DURATION	PRESSURANT TBD TECHNICAL
RESUPPLY MODULE/ OMV	GEO OPERATIONS FLUID TYPE: AMOUNT: DRIVER:	1993-2001 HYDRAZINE 4 klbs DOD/COMMERCIAL S/C 1-2 YEARS DURATION	OMV WITH EXTENSION KITS ASSUMED MMH/NTO 45,500 lbs TRANSPORT OF OMV/RM TO GEO	PRESSURANT TBD TECHNICAL

OPERATIONAL RESUPPLY SCENARIOS DEFINE SIZING

TABLE 6. SPACE PLATFORM EXPENDABLES RESUPPLY MODULE SIZING OPTIONS AND DRIVERS

new subsystem design options and operational requirements. In addition to fluid resupply subsystem design changes, which include 3 separate disconnect/line purging designs, the bi-propellant resupply quantity of the Expendables Resupply Module (E.R.M.) has been reduced from the original 45,500 lbs. to 7,000 lbs.

2) The on-orbit transfer of propellants with pumps will make possible propellant scavenging and extend satellite life. Three different types of pumps were considered and compared: centrifugal, gear and peristaltic. This subtask reports the power requirements versus flow rate at different P's for the propellants NTO, MMH, and N_2H_4. Power requirements for the centrifugal pump were determined from reference (3). Power requirements for the gear pump were determined from data supplied by the Sundstrand Corporation.

3) An analysis of two pressurant transfer methods (cascade and compressor) was performed and the optimum method and supply configuration for resuppling a 7000 lb bi-propellant's pressurant system was determined.

4) Subtask 4 requires that the instrumentation requirements for microprocessor control and system status monitoring during resupply/quiescent periods be defined. Reference 10 was used as a baseline. The types of instrumentation considered are pressure transducers, thermocouples, valve position indication feedback switches, flowmeters, and contact sensors.

5) Commonalities with on-going programs were assessed and presented.

6) The objectives and tests to verify these objectives are presented for the Middeck Ullage Transfer Experiment (MUTE).

7) The test objectives and tests to verify these objectives are presented for the Flight Demonstration Test Vehicle (FDTV).

8) The purpose of the ERM ground test programs is to develop the Flight Demonstration Test Vehicle to be used to conduct the fluid transfer during the ERM flight demonstration.

A re-evaluation of the ERM module's design reference mission scenario (Task 1) (Reference 1), has redefined the expendables resupply capability of the vehicle. The ERM., developed under the SPER. Concept Definition Study, is now foreseen as a potential growth version (i.e., OMV extended missions kit, space based propellant depot) of a more modestly sized module design. A ERM of, at best, 7,000 lbs. of bi-propellant will adequately cover resupply needs through the year 2010. The 7,000 lbs. bi-propellant capacity will require a tank volume approximately equal to 6 Shuttle Orbiter RCS propellant tanks.

In addition to the changes in the expendables resupply capacity, minor changes have been incorporated into the conceptual ERM fluid transfer system design. The system design changes include 1) isolating pressurant tanks, 2) regulating compressor inlet pressure, 3) additional pump and compressor plumbing for ERM propellant and pressurant tank refueling, and 4) three separate disconnect/line purge subsystem designs.

High pressure solenoid valves have been positioned at each pressurant tank outlet for a more efficient management of resupply pressurant. Isolating pressurant tanks will ensure a sufficient supply of high pressure helium or nitrogen gas, for multiple engagement resupply missions.

In-house data, concerning the operational characteristics of high pressure gas compressors, reveals that regulating the compressor inlet pressure will be necessary, to obtain the desired design compression ratio and outlet pressure.

A rearrangement of the conceptual fluid system schematic's pump and compressor subsystem plumbing allows the use of these components in the refueling of the ERM's propellant and pressurant tanks. This modification is illustrated in Figures 18, 19 and 20.

Three separate disconnect/line purge subsystems are illustrated in the enclosed figures. Figures 18 and 18a illustrate a pressurant purge design, the same design as illustrated in the SPER Conceptual Design Study. Figure 19 illustrates a vacuum pump purge design, and Figure 20 a dribble volume purging system. The fluid transfer schematics illustrated in Figures 19 and 20 also use a quick disconnect design similar to the NASA/Fairchild, triple inhibit, standardized refueling coupling (NASA 9-17333 umbilical). A detailed description of the operational characteristics of each disconnect/line purge design is documented in reference 2. Table 7 describes the major advantages and disadvantages of each purge subsystem design.

Evaluation of Propellant Resupply Pumps

Propellant transfer can be accomplished by one of the following three methods: ullage recompression, ullage vent and ullage exchange. In the ullage recompression method, the flow rate is determined by the amount of heat that the propellant tank can dissipate. The flow rates for the other two methods are usually limited by the type of propellant acquisition devise (vane, screen, diaphragm) in the receiver tank. Due to the different flow rates required, the power requirements in this report were determined for several P values at flow rates up to 10 GPM.

Although a peristaltic pump can provide enough P for both an ullage vent and ullage exchange transfer it cannot accomplish an ullage recompression transfer. This is due to the fact that a head rise of at least 250 psi is required to complete an ullage recompression propellant transfer. Consequently, the peristaltic pump was not longer considered as a candidate for propellant transfer since it is capable of a maximum head rise of 100 psi.

Figures 21, 22, 23 and 24 present gear pump power requirements versus flow rate at various head rise values for the propellants MMH, NTO, a system of MMH and NTO, and hydrazine. Data supplied by Sundstrand Corporation was extrapolated to complete the family of head curves seen in Figures 21, 22, 23 and 24. These curves are for a 10 GPM optimum flow rate gear pump and represent power required to pump and do not include motor efficiency. According to a representative of Motronics, motor efficiency could vary anywhere from 70 to 90% depending on how the motor is cooled, what materials are used to make the motor, lamination, air flow (if any), heat dissipation, etc. Due to all of the above factors, the motor efficiency will not be taken into account in the power consumption calculated for the 10 GPM optimum gear pump.

Line Pressurant Purge	- Purge receiver and ERM QDs, plus some line leading to QDs. - Quick Purge Time - No Dribble Volume	- High System Weight - More QDs to Mate
Vacuum Pump Purge	- Low System Weight - No Dribble Volume - Purges Receiver and ERM QDs	- Long Purge Times - Problems with Compressive Heating - Limited Life Pumps
Dribble Volume Purge	- Low System Weight - Simple Design - Quick Purge Time	- Receiver and ERM QDs are not purged. - Some propellant will go into the environment 0.2 cc

TABLE 7. DISCONNECT/LINE PURGE OPTIONS, MAJOR ADVANTAGES AND DISADVANTAGES

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OF POOR QUALITY

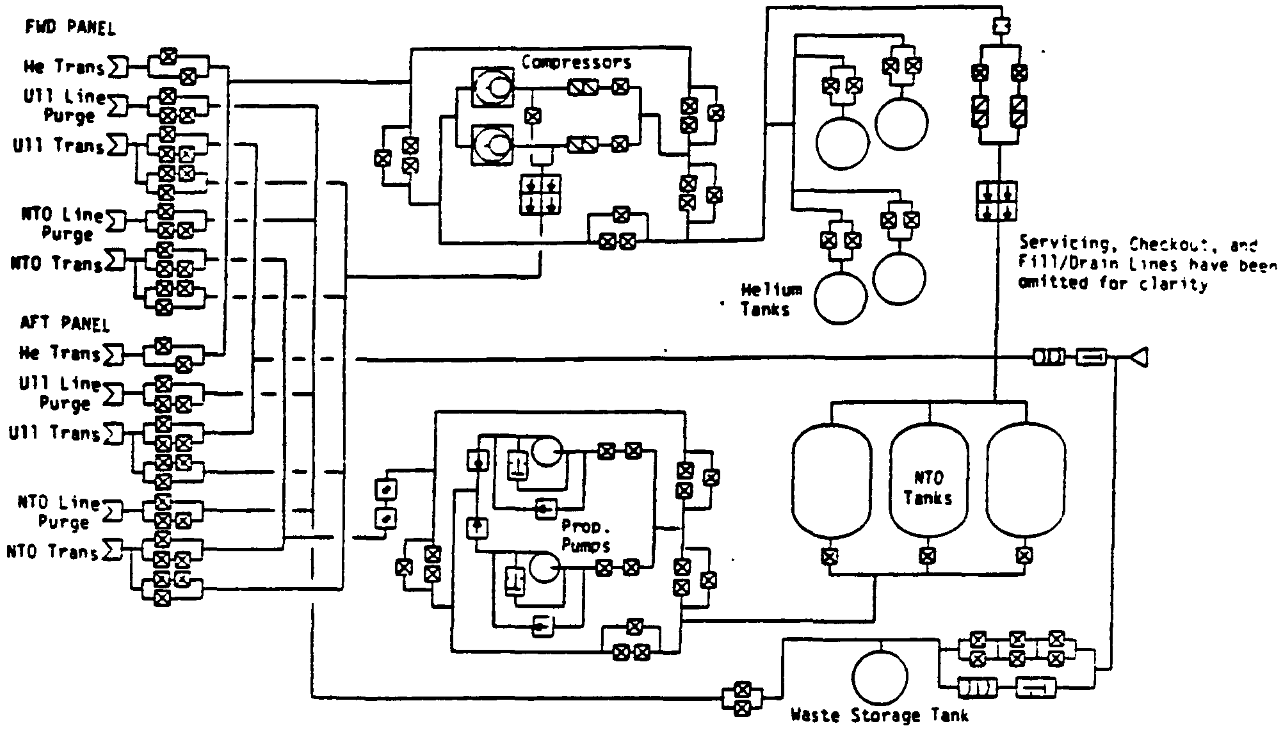


FIGURE 18. FLUID TRANSFER SCHEMATIC, PRESSURANT PURGE DESIGN

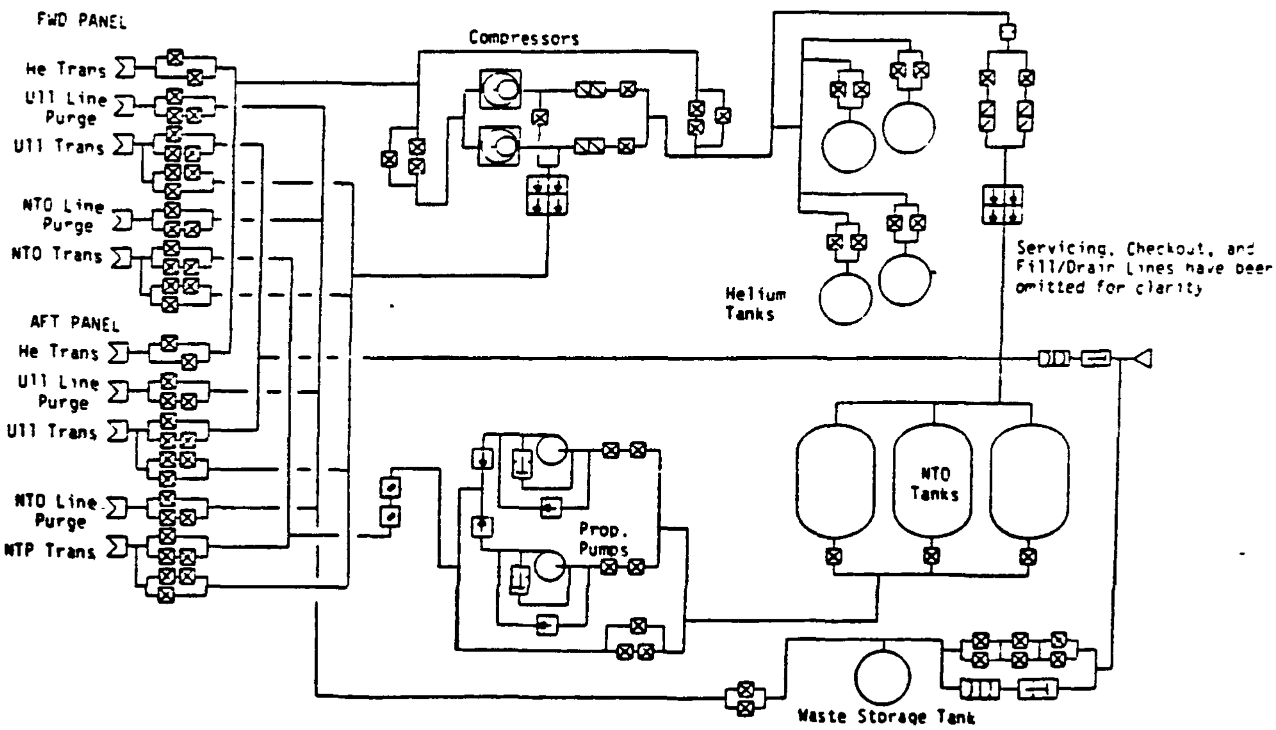


FIGURE 18a. FLUID TRANSFER SCHEMATIC, PRESSURANT PURGE DESIGN, WITHOUT SELF REFUELING PLUMBING

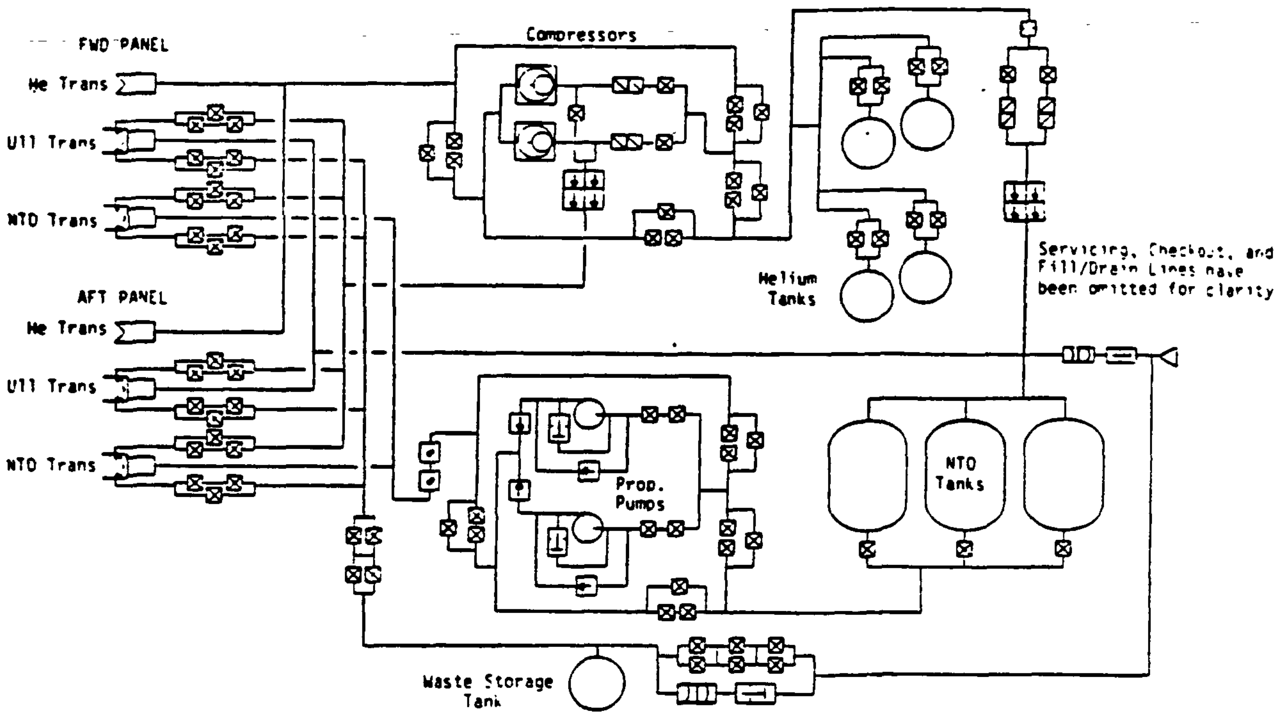


FIGURE 19. FLUID TRANSFER SCHEMATIC, DRIBBLE VOLUME PURGE DESIGN

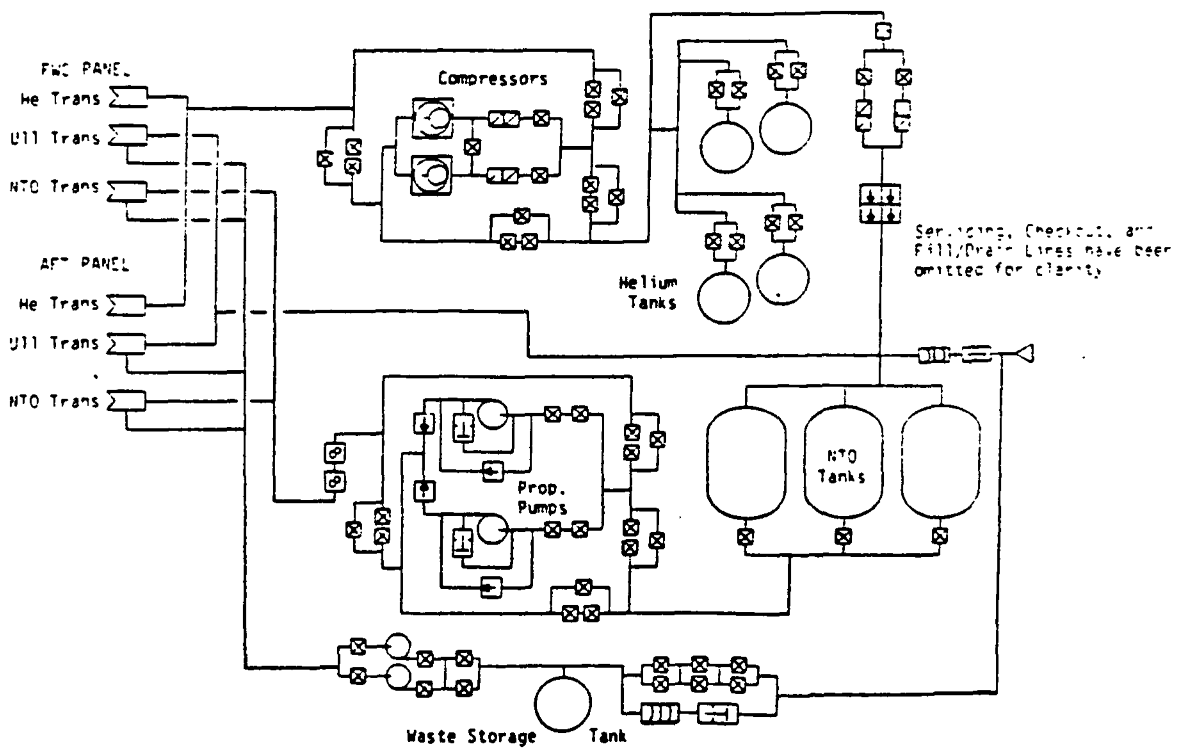


FIGURE 20. FLUID TRANSFER SCHEMATIC, VACUUM PUMP PURGE DESIGN

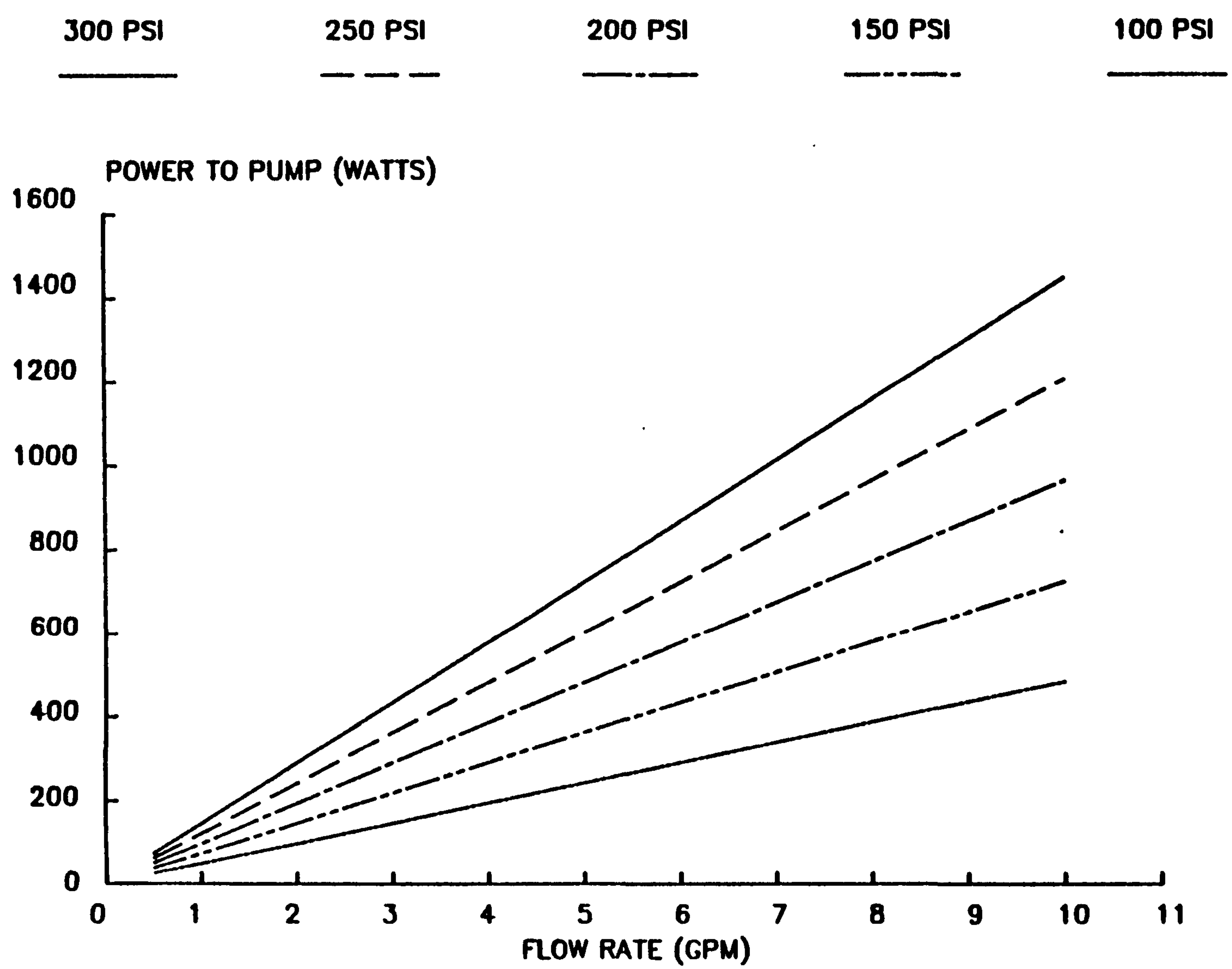


FIGURE 21. REQUIRED GEAR PUMP POWER AT VARIOUS DELTA P FOR MMH

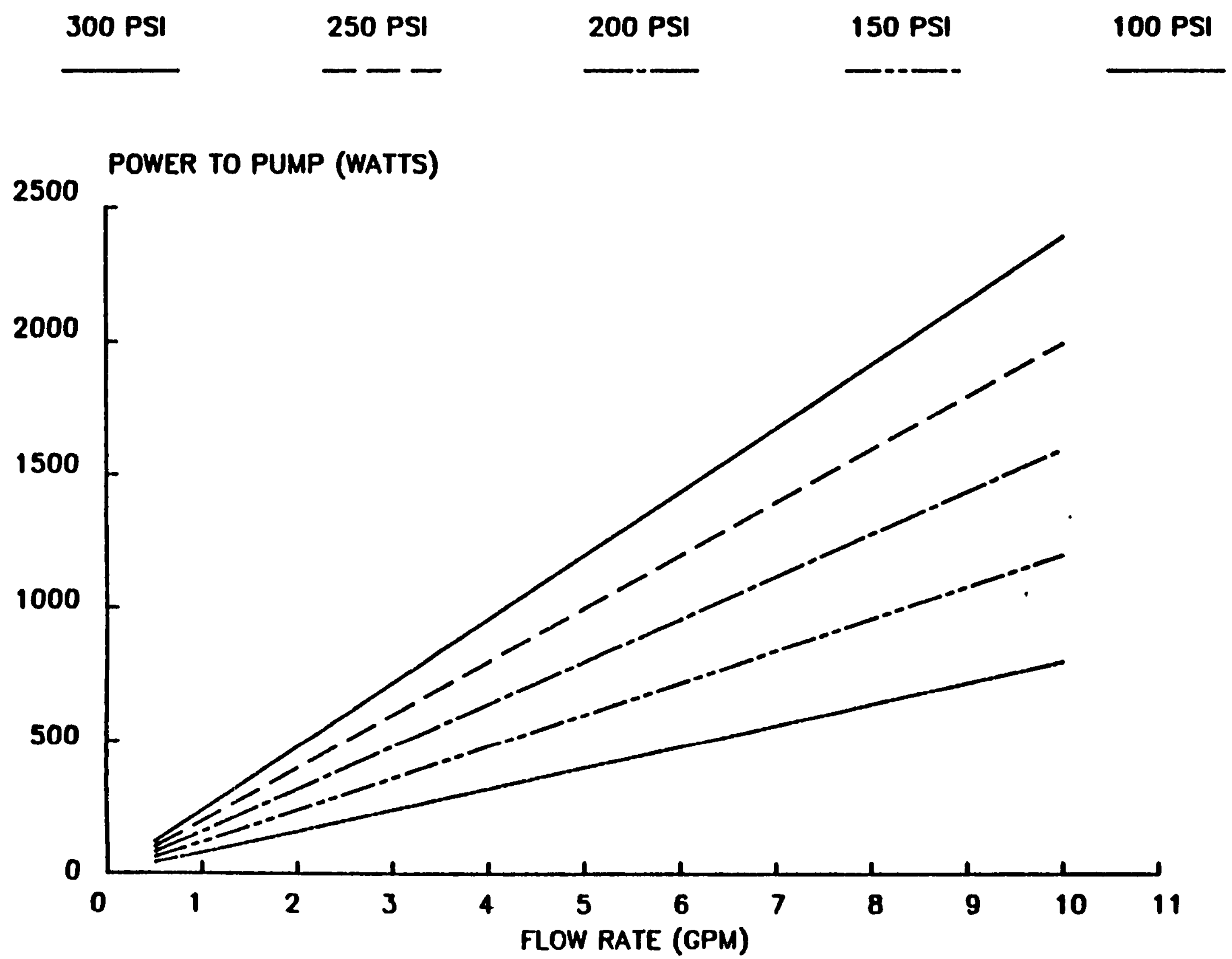


FIGURE 22. REQUIRED GEAR PUMP POWER AT VARIOUS DELTA P FOR NTO

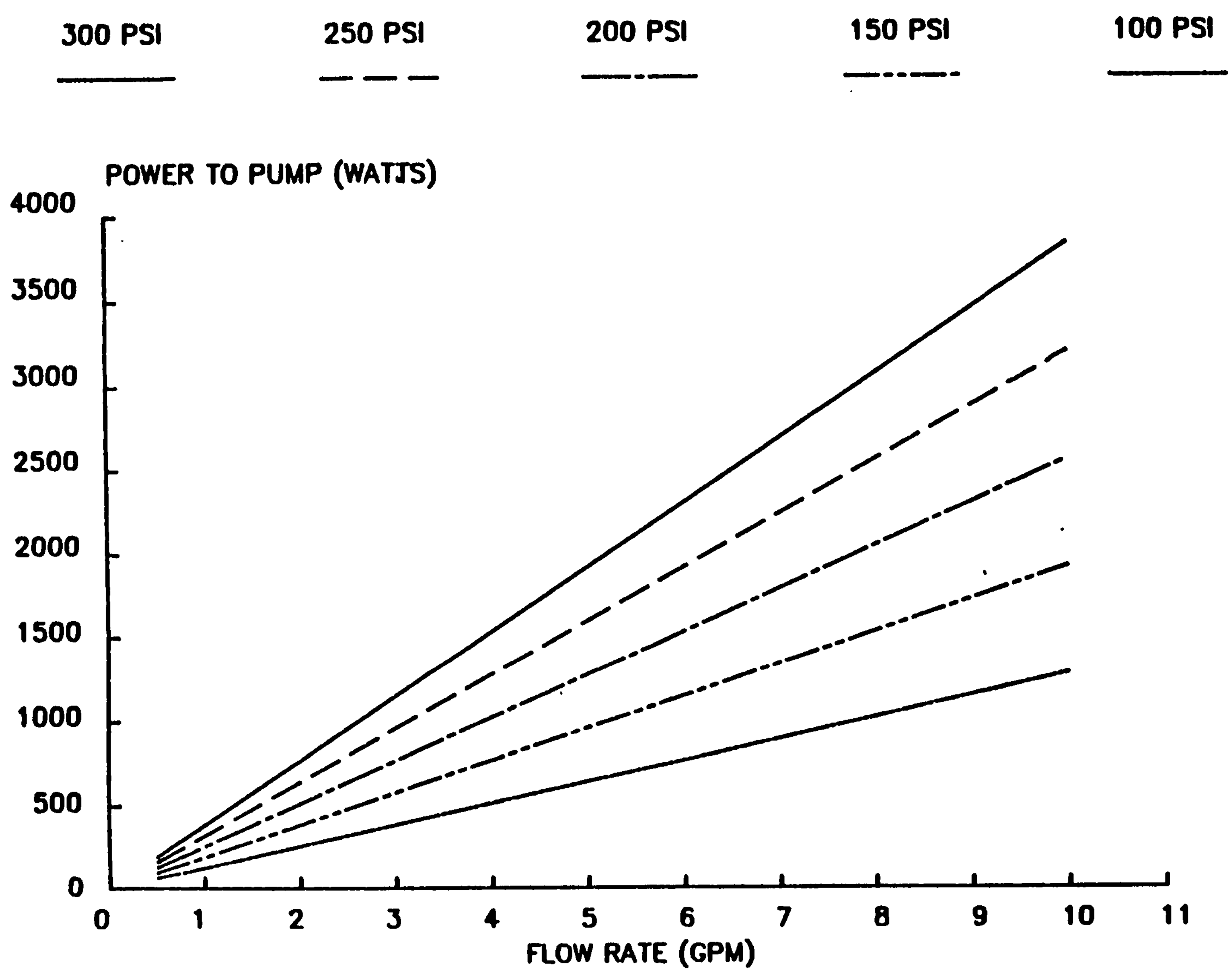


FIGURE 23. REQUIRED GEAR PUMP POWER AT VARIOUS DELTA P FOR NTO AND MMH

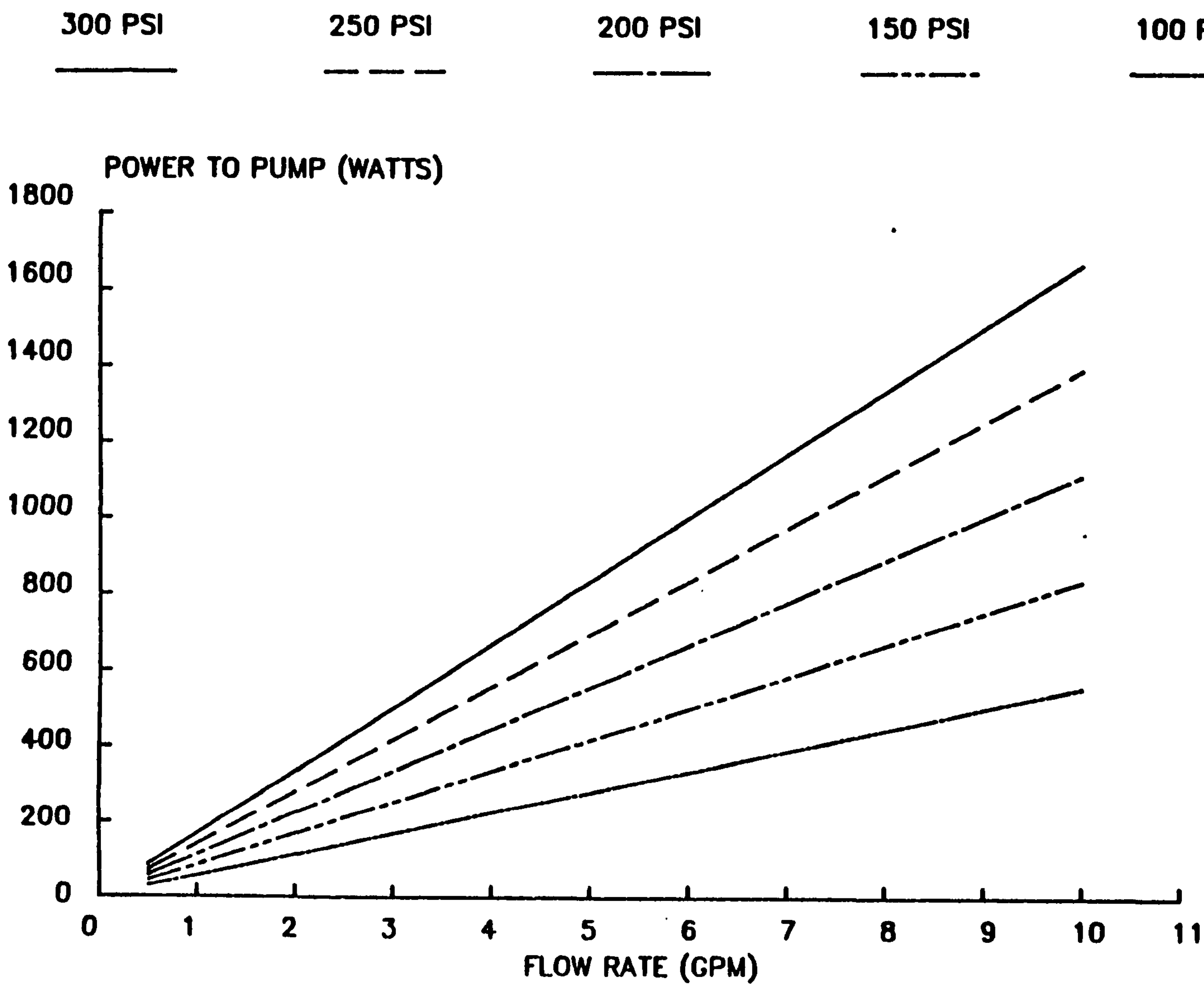


FIGURE 24. REQUIRED GEAR PUMP POWER AT VARIOUS DELTA P FOR HYDRAZINE

The power requirement is accurate at the optimum flow rate of 10 GPM but only an approximation at the other flow rates. The justification for the above approximation is that gear pumps are usually only run at their optimum flow rates, and even though they can be run at off-design speeds, no data was found for this case. A major advantage of the gear pump is its ability to handle a wide range of head values. This 10 GPM optimum flow rate pump weighs approximately five pounds (not including motor weight).

Figures 25, 26, 27 and 28 present pump power requirements of a centrifugal pump versus flow rate at various head rise values for the propellants MMH, NTO, a system of MMH and NTO, and hydrazine. Figures 25, 26, 27 and 28 were duplicated from reference 3.

As can be seen in Figure 23, the power to pump NTO and MMH with a gear pump at a flow rate of 9 GPM and a P of 250 psi is about 2900 watts. The power to pump NTO and MMH with a centrifugal pump under the same conditions is about 2700 watts (Figure 27). The difference in power required by a centrifugal and by a gear pump is not significant; therefore, further study into the power consumption of the motors that will be coupled to these pumps is required.

The major advantage of the gear pump is its ability to provide a wider range of P's than a centrifugal pump can provide. A gear pump can provide P's up to 1500 psid while the centrifugal pump is not capable of much more than delta P's in the 350 psid range at its optimum operating speed. Therefore, centrifugal pumps are inefficient when run at off-design speeds. Variations in operating speed are not critical for the gear pump.

Micropump Corporation believes they can make a magnetically coupled gear pump to pump at 3 GPM with a P of 300 psia. To run the motor and pump combination requires 1730 watts to pump MMH and 2880 watts to pump NTO. This pump (with motor) weighs about twenty pounds, is 7-1/2 inches in diameter and is about twenty inches long. To achieve pressure rises greater than 300 psia, multiple stage pumps will be required.

An analysis of two pressurant transfer methods was performed and the optimum method and supply configuration for resuppling a 7000lb bi-propellant's pressurant system was determined.

The two transfer methods analyzed were a cascade only method and a cascade with compressor method. The cascade only method involves multiple supply tanks at a higher pressure being opened sequentially to a receiver tank until pressure equalization occurs; however, this transfer requires considerably larger quantities of helium than is required for the resupply. Therefore, larger and heavier tanks are required for this method. In contrast, the cascade with compressor method requires only enough helium for the resupply since it performs a cascade transfer first followed by the use of a compressor to complete pressurization. However, this method requires a battery system and two compressors for redundancy which increase system weight.

Supply system volumes and weights were calculated for three supply tank pressures of 6000, 5500, and 5000 psia and 1 to 6 supply tanks. The optimum transfer method and supply tank configuration was determined to be as follows: A cascade only transfer with four 4200 cubic inch supply tanks at 6000 psia for a total system weight of 277 lbs.

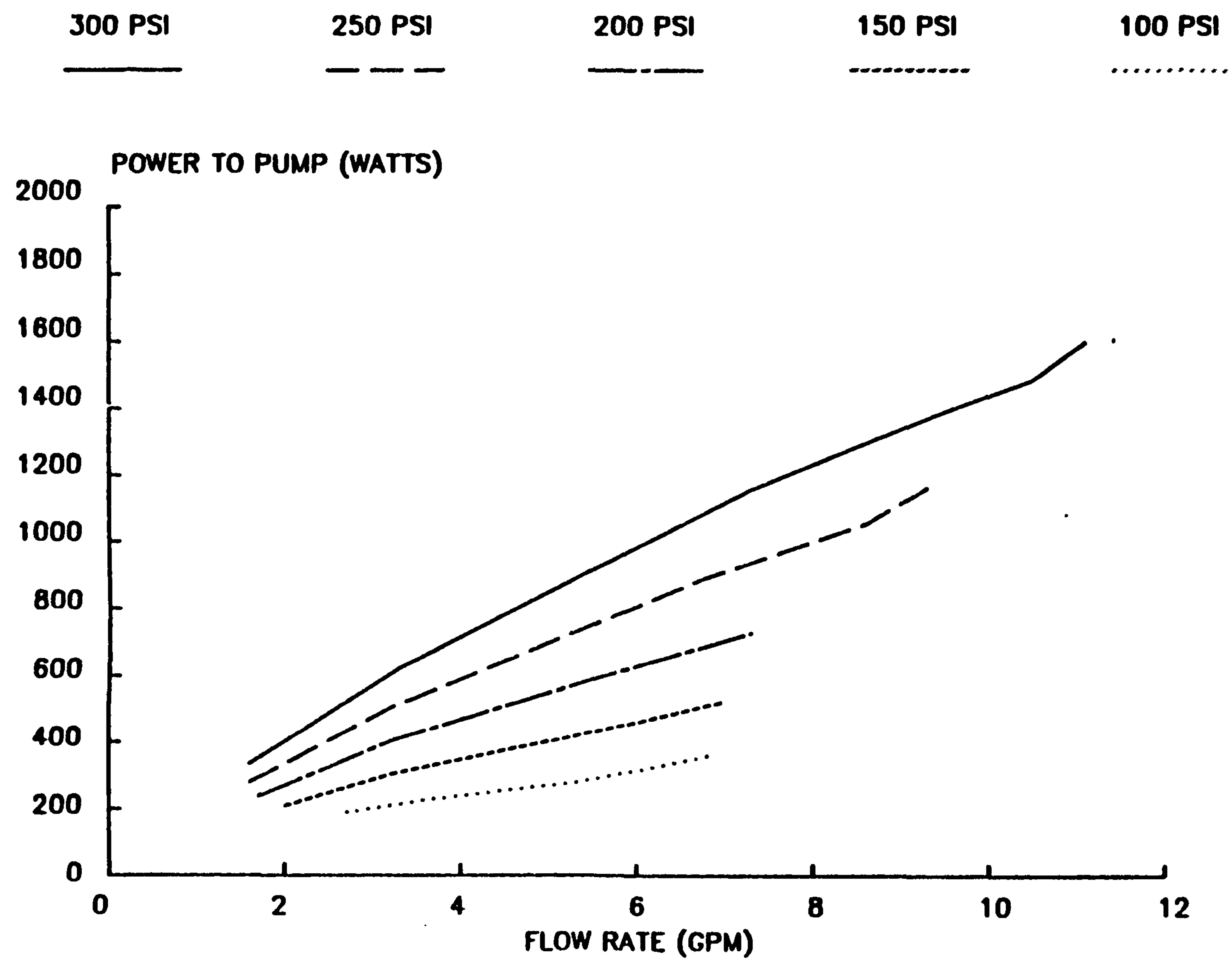


FIGURE 25. POWER TO PUMP AT VARIOUS DELTA P FOR MMH

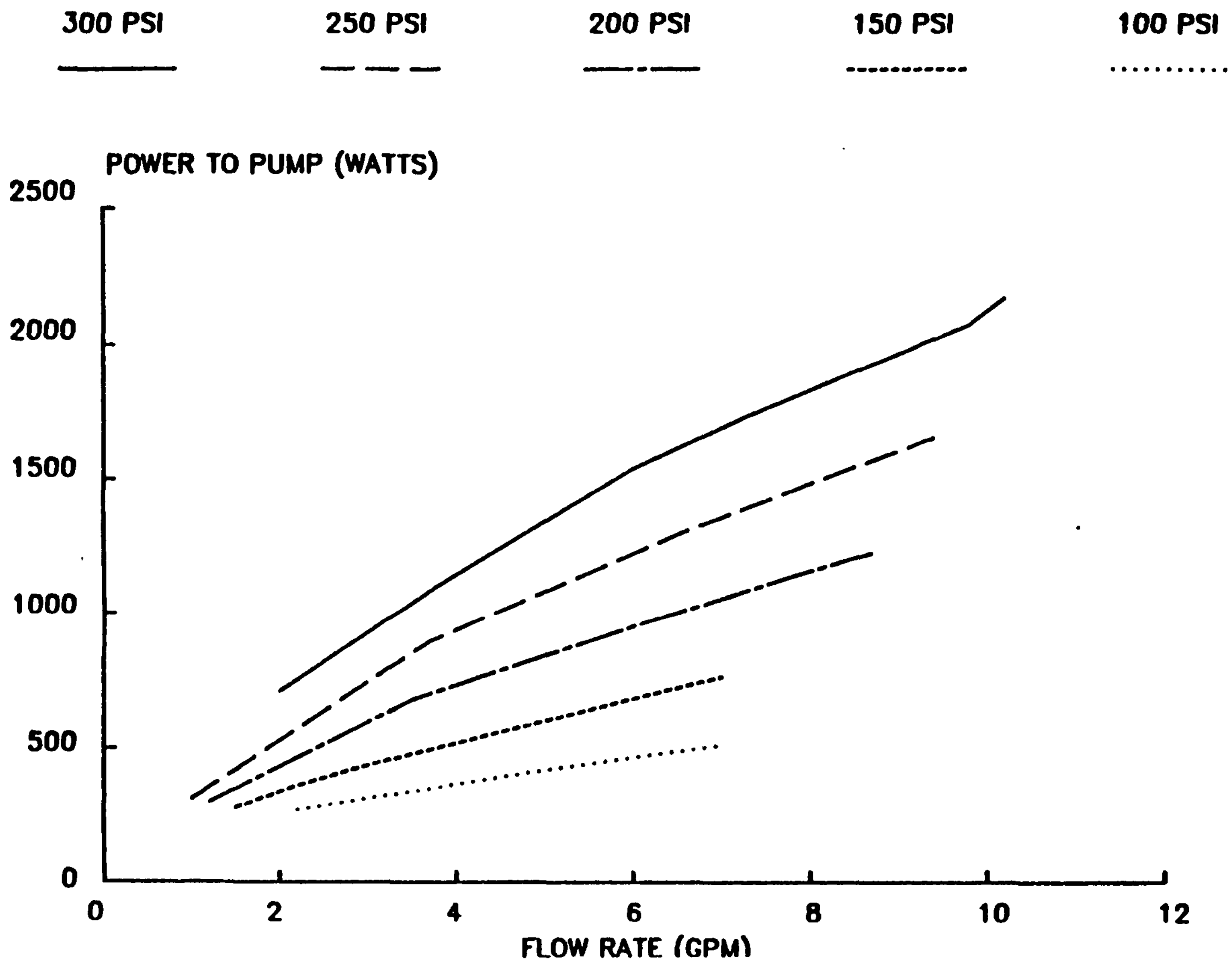


FIGURE 26. POWER TO PUMP

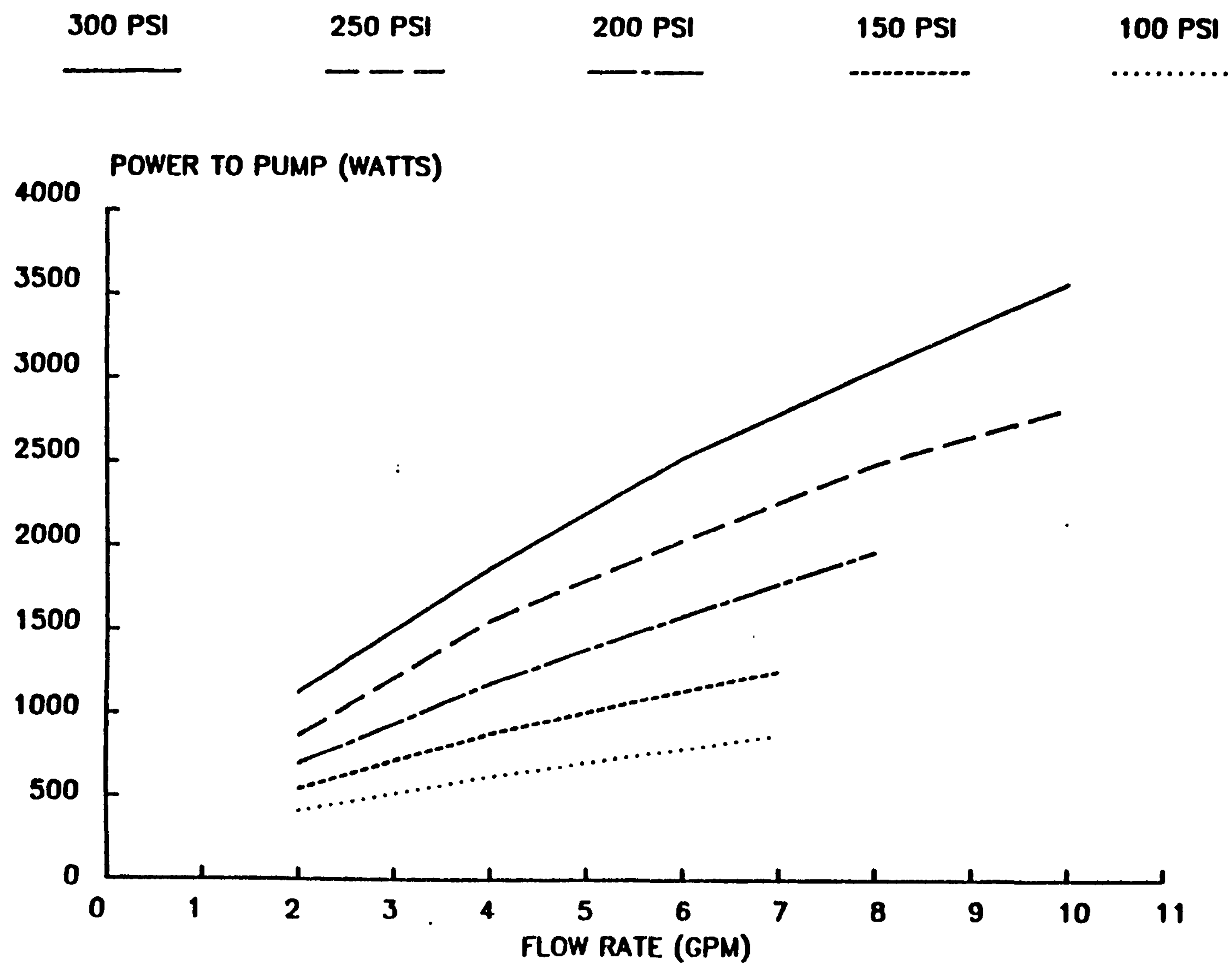


FIGURE 27. POWER TO PUMP AT VARIOUS DELTA P FOR NTO AND MMH

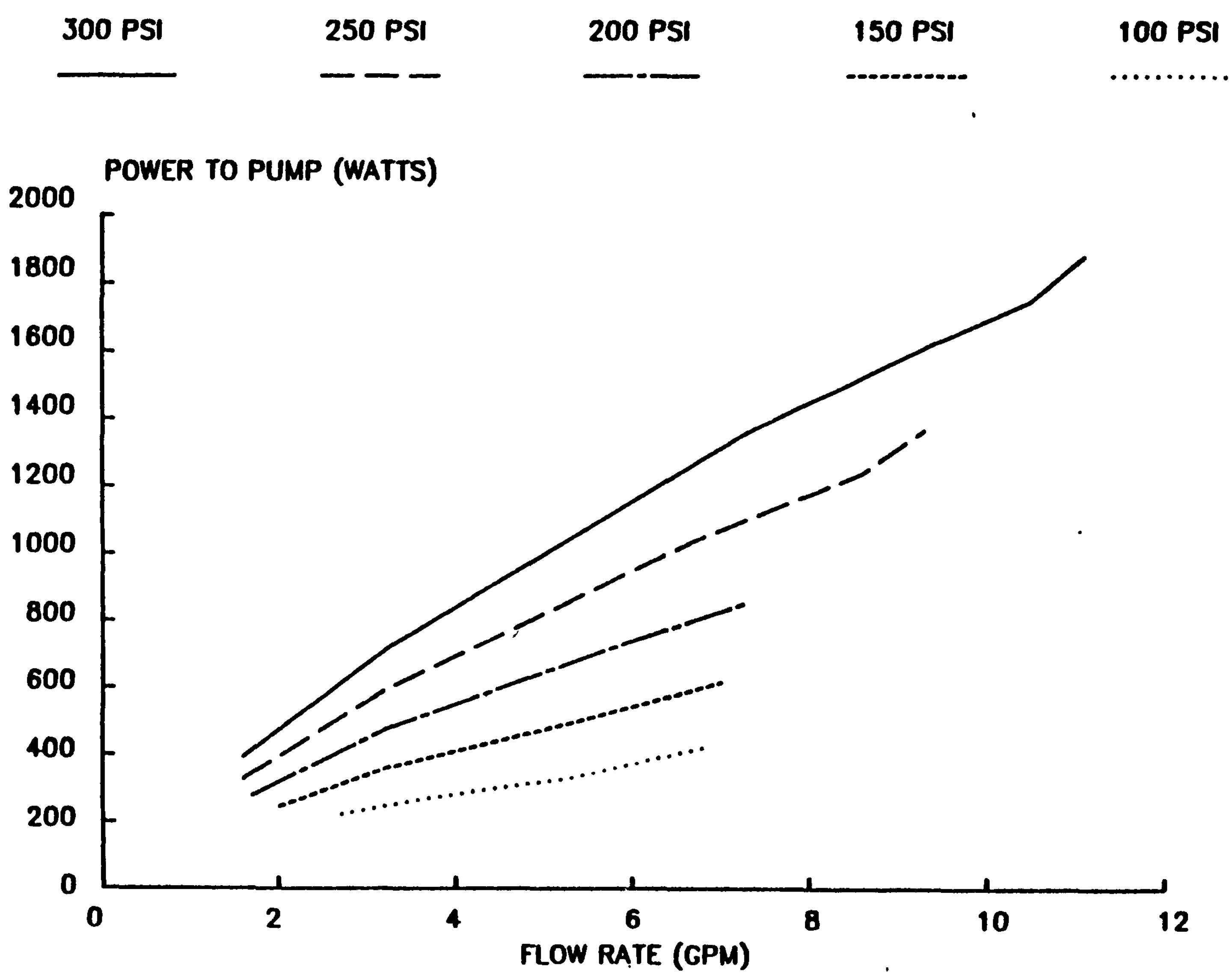


FIGURE 28. POWER TO PUMP AT VARIOUS DELTA P FOR HYDRAZINE

The Concept Definition Study, Ref. 7, identified two possible pressurant transfer approaches: a cascade or blowdown only method and a cascade with compressor method. The cascade only method involves multiple supply tanks at a higher pressure being opened one at a time to a receiver tank until pressure equalization occurs, thus pressurizing the receiver from 500 psia and 70 F to a level of 3600 psia and 70 F. The advantages of this method are its simple operating procedure, sequential valve openings, and its minimal equipment requirements, tanks and isolation valves only. However, due to this pressure equalization technique, a larger volume of helium is required to be carried along than is required for transfer; thus adding more system weight in the form of larger tanks and excess helium. The cascade with compressor method, although, requires only the amount of helium necessary for the resupply since it performs a cascade transfer initially then utilizes a compressor to finish receiver pressurization. However, this method also has its disadvantages; it requires a battery and two compressors, for redundancy, which add considerably to the system weight and complexity.

The supply system was sized for a total receiver volume of 13000 cubic inches since it was determined that this volume of helium at 3600 psia would expel a worst-case amount of 7000 lbs of bi-propellant and still have the required 500 psia residual. A previous task analysis, Ref. 4 had determined that during the helium transfer the supply tank would cool to approximately 40 F and the receiver tank will heat up to 160 F. Since this occurs in relatively short periods of time when compared to the overall transfer time, it was assumed that the transfer was made isothermally at 40F and 160 F. This provides an important worst-case assumption in that by assuming the receiver tank is at 160 F instead of 70 F, less helium was calculated to have been transferred than actually was transferred due to the pressure difference between the two temperatures. therefore, a larger supply system size and weight was calculated to perform the transfer. Figure 29 depicts this pressure difference for the cascade with compressor transfer method.

The actual sizing procedure for the cascade only method used the computer program utilized in the analysis performed for Ref. 6 and, in particular, involved guessing a total supply system volume and calculating the final receiver tank pressure from the ideal gas equation with compressibility then comparing that to the desired level of 3600 psia. System volumes for the three supply tank pressures of 6000, 5500, and 5000 psia and 1 to 6 supply tanks were calculated. Table 8 presents these values.

The sizing of the supply system for the cascade with compressor method was easily accomplished once the three supply pressures were known. Since this transfer method utilizes a compressor to achieve the desired receiver pressure, only the amount of helium required to resupply the receiver is needed in the supply system. Therefore, by using the ideal gas law and knowing the receiver requirements and supply system pressure, the system volume was calculated. Table 9 also presents these values.

The weight of a supply tank was calculated from an equation obtained during a phone conversation with Mr. Newhouse, Ref. 8.

$$(Pb*V)/675{,}000=Wt$$

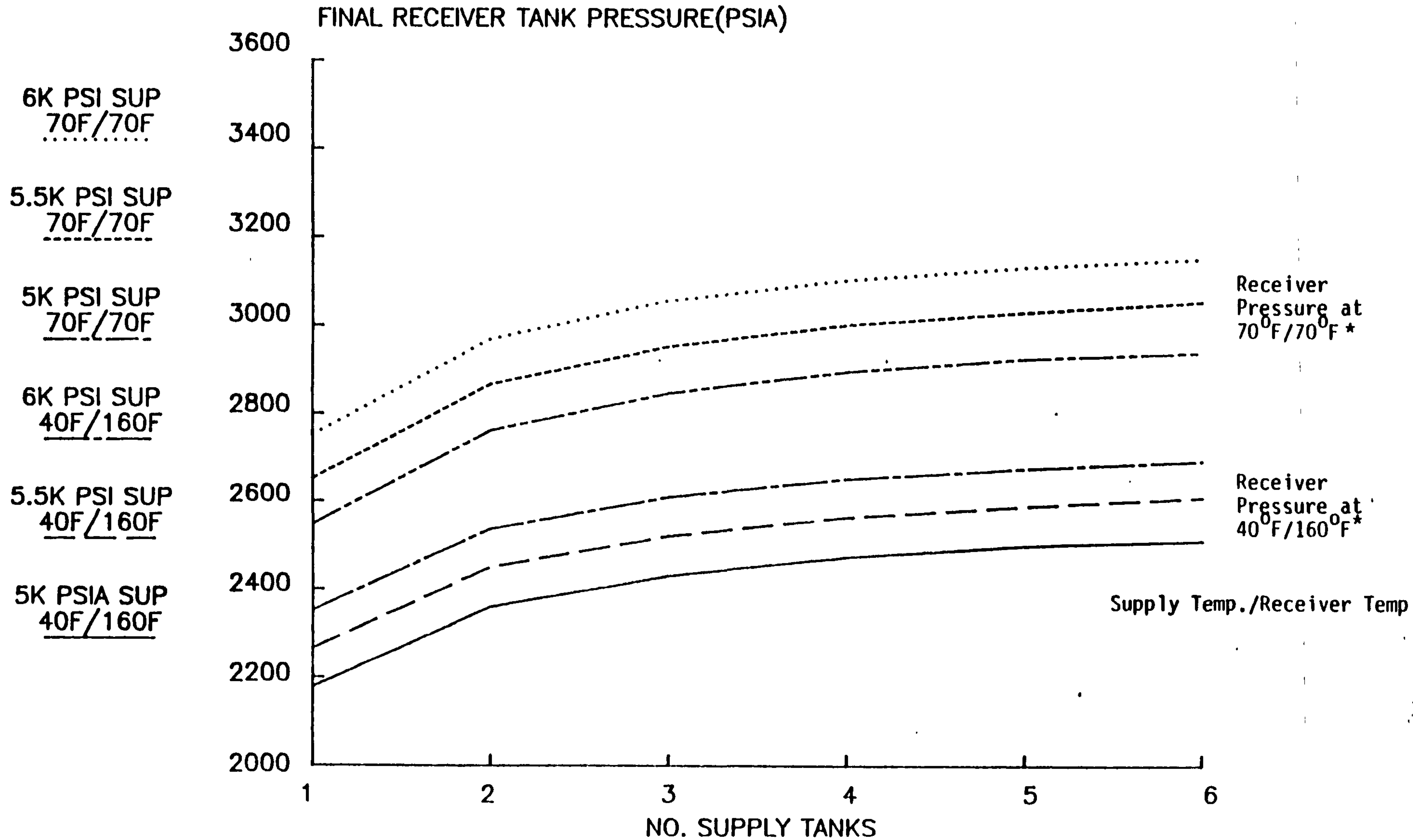
NUMBER OF TANKS	TOTAL SYSTEM VOLUME (CUBIC INCHES)					
	6000 PSIA SUPPLY PRESSURE		5500 PSIA SUPPLY PRESSURE		5000 PSIA SUPPLY PRESSURE	
	CASCADE ONLY	CASCADE W/ COMP	CASCADE ONLY	CASCADE W/ COMP	CASCADE ONLY	CASCADE W/ COMP
1	26800	7800	37500	8500	61000	9400
2	19400	7800	24800	8500	35000	9400
3	18000	7800	23400	8500	31500	9400
4	16800	7800	21200	8500	28000	9400
5	16500	7800	20500	8500	27500	9400
6	16200	7800	19800	8500	27000	9400

TABLE 8. SYSTEM VOLUME COMPARISON

NUMBER OF TANKS	SYSTEM WEIGHT (LBS)					
	6000 PSIA SUPPLY PRESSURE		5500 PSIA SUPPLY PRESSURE		5000 PSIA SUPPLY PRESSURE	
	CASCADE ONLY	CASCADE W/ COMP	CASCADE ONLY	CASCADE W/ COMP	CASCADE ONLY	CASCADE W/ COMP
1	417	380	535	384	790	389
2	308	* 376	360	* 380	460	* 385
3	291	378	342	382	419	387
4	* 277	380	* 318	385	* 379	389
5	277	384	313	388	377	393
6	277	388	311	392	375	397

* - ASTERISK DENOTES OPTIMUM CONFIGURATION

TABLE 9. SYSTEM WEIGHT COMPARISON



* NOTE: COMPRESSOR FINISHES TRANSFER TO 3600 PSIA FROM THIS LEVEL

FIGURE 29. FINAL RECEIVER PRESSURE VS NO. SUPPLY TANKS

where:

Pb = Tank Burst Pressure (psia) = Max Operating Press*1.5 (S.F.)
V = Tank Volume (cubic inches)
Wt = Tank Weight (lbs)

The total system weight for the cascade only method includes the weight of the tanks, helium, and 2 isolation valves per tank. The total weight for the cascade with compressor method includes tanks, helium, 2 compressors (redundancy), a battery system, and 2 isolation valves per tank. The compressors were assumed to weigh 100 lbs each based upon estimations by various companies for a 2 scfm flowrate and 2/3 hp compressor. The tank solenoid valves were assumed to weigh 2.3 lbs each based upon valves currently being used in the aerospace industry. The battery system weight varied according to the amount of helium required to be transferred. This weight included the weight of the Li/TiS battery, the coolant system, the power conditioning system, and the system's structure. To calculate the battery system weight the following equation was used:

$$(WtHe * Pcomp)/Pconv = \text{Battery system wt}$$

where:

WtHe = Mass of Helium required to be transferred (lbm He)
Pcomp = Power required by compressor per pound helium (Wh/lbm He)
Pconv = Efficiency of battery system (Wh/lbm battery)

The power required by the compressor was determined to be 402.6 Watt-hours per pound of helium transferred. The efficiency of the battery system was determined to be 40 Wh per pound of battery once the coolant system, power conditioning system, and system structure was included. Table 10 presents this system weight, the mass of helium required to be transferred, and the amount of power required by the compressor.

Figures 30, 31 and Table 9 present the total system weight vs. the number of supply tanks for the two transfer methods. A line has been drawn through the optimum tank configuration in the figures and an asterisk denotes them in the table. Referring to Table 8 and Table 9, these optimum configurations are: four 4200 cubic inch supply tanks (16800 cubic inches total) at 6000 psia with a system weight of 277 lbs for the cascade only method and two 3900 cubic inch supply tanks (7800 cubic inches total) at 6000 psia with a system weight of 376 lbs for the cascade with compressor method.

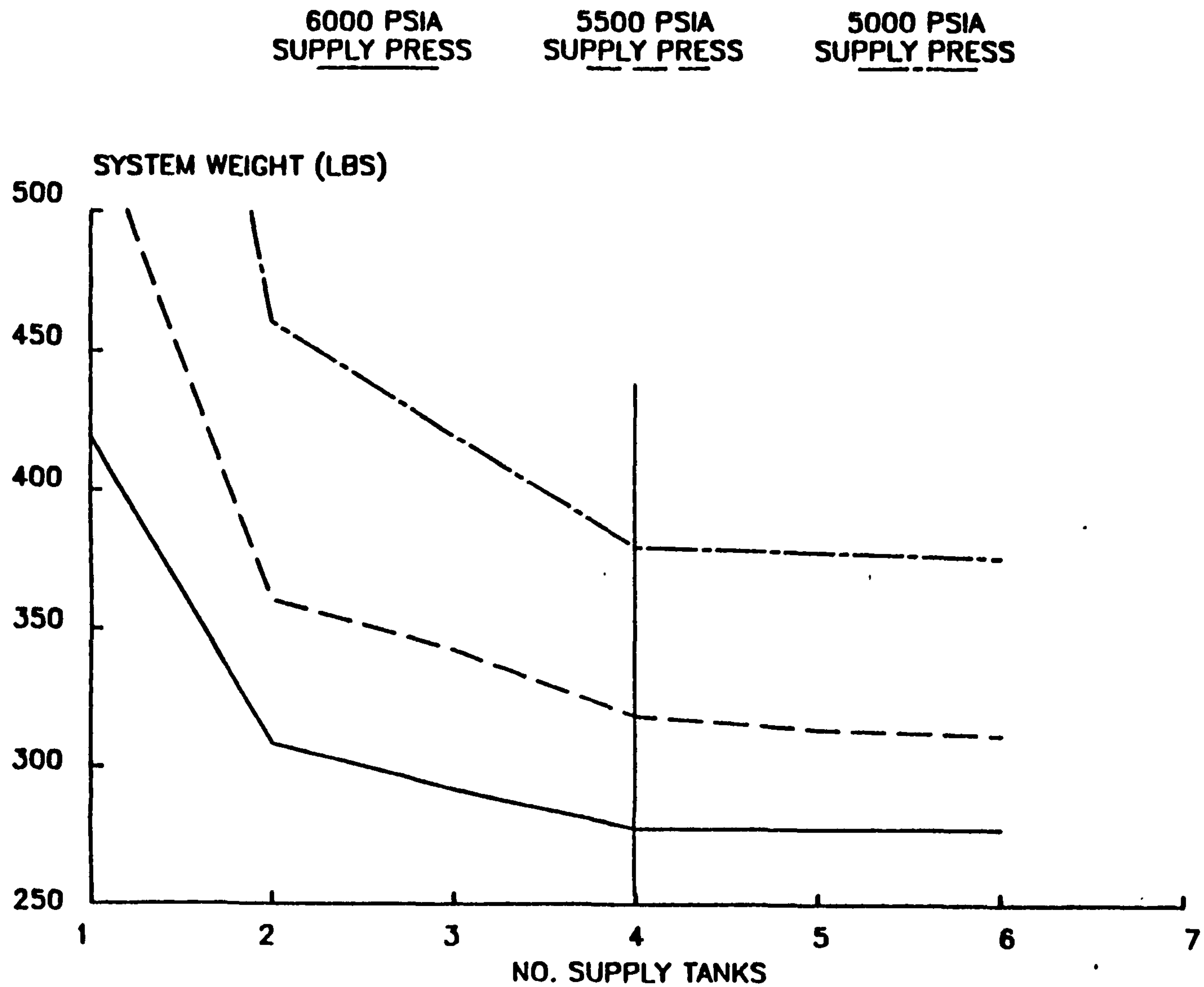
The overall optimum transfer method and tank configuration can now be identified to be a cascade only transfer with four 4200 cubic tanks at 6000 psia since it weighs 99 lbs less than the other method.

Instrumentation Requirements for Microprocessor Control

Pressure Transducers: Pressure Transducers were identified to monitor helium tank inlet pressure, propellant tank ullage pressure, compressors/pumps inlet and outlet pressures, quick disconnect inlet pressure, and waste tank pressure. The minimum pressures to be monitored within the receiver vehicle during resupply are helium and propellant tank pressures. The pressure

Number of Tanks	Supply Pressures (psia) 6000			5500			5000		
	Mass He (lbm)	Power (KW)	Bat. Sys. Wt. (lbs)	Mass He (lbm)	Power (KW)	Bat. Sys. Wt. (lbs)	Mass He (lbm)	Power (KW)	Bat. Sys. Wt. (lbs)
1	5.64	2.27	55.3	6.08	2.45	59.6	6.49	2.61	63.6
2	4.85	1.95	47.5	5.24	2.11	51.4	5.65	2.27	55.4
3	4.51	1.81	44.2	4.91	1.98	48.1	5.32	2.14	52.1
4	4.32	1.74	42.3	4.73	1.90	46.4	5.13	2.07	50.3
5	4.22	1.70	41.4	4.61	1.86	45.2	5.02	2.02	49.2
6	4.14	1.66	40.6	4.52	1.82	44.3	4.97	2.00	48.7

TABLE 10. BATTERY SYSTEM AND POWER REQUIREMENTS FOR CASCADE WITH COMPRESSOR METHOD



NOTE: SYSTEM WEIGHT INCLUDES WEIGHT OF TANKS, HELIUM, AND VALVES

FIGURE 30. SYSTEM WEIGHT COMPARISON CASCADE ONLY RESUPPLY METHOD

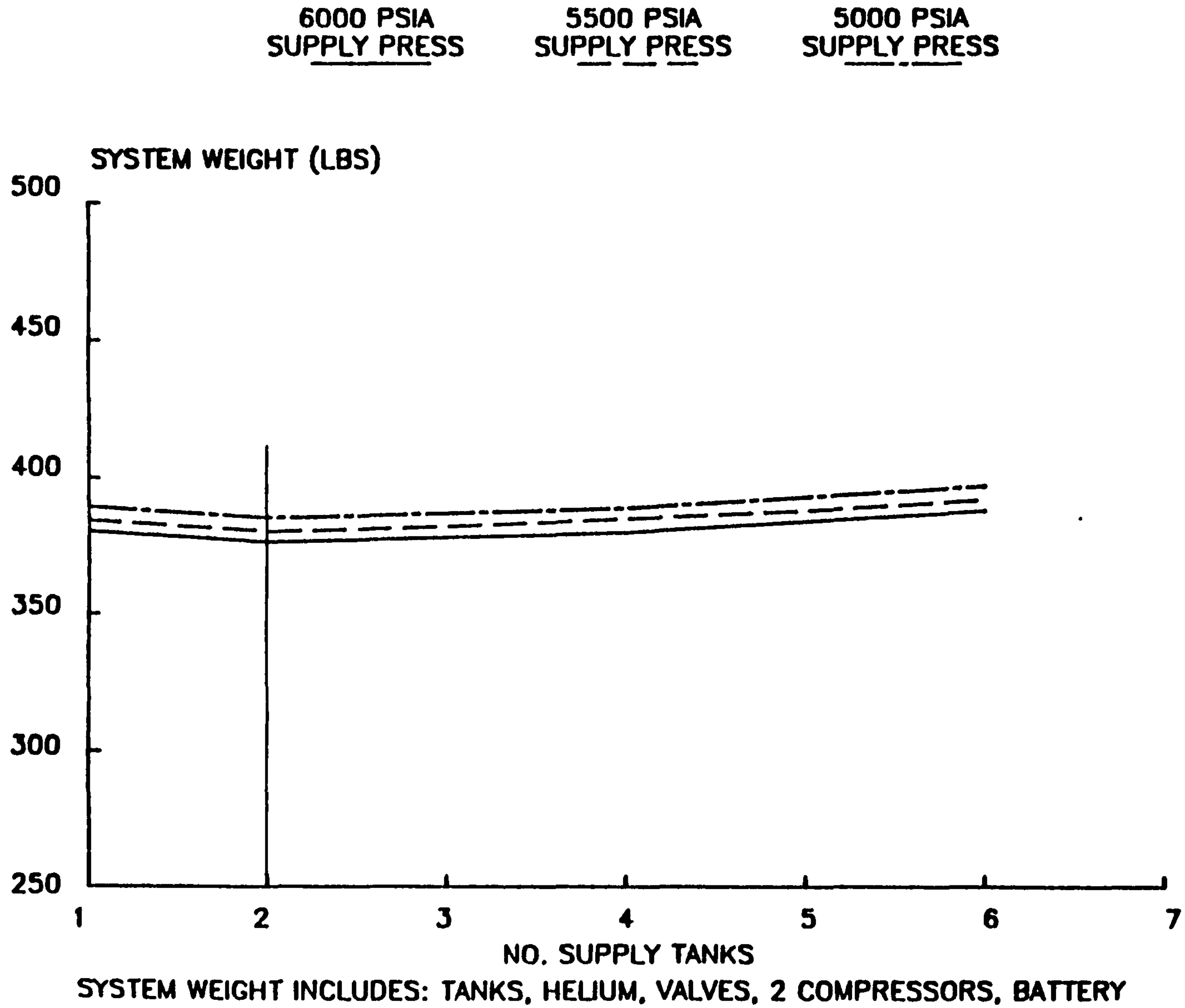


FIGURE 31. SYSTEM WEIGHT COMPARISON CASCADE WITH COMPRESSOR RESUPPLY METHOD

transducer locations are shown in Figure 32. A brief description of each instrument and how it will be used is shown in Table 11. The pressure transducers on the quick disconnect that will be used for leak detection (pressure decay) are not discussed in this letter as they are recommended to be an integral part of the disconnect. The sampling rate for adequate microprocessor control is recommended to be 100 samples per second during dynamic resupply operations and then reduced to one sample per second during quiescent periods.

Thermocouples: Thermocouples were identified to support PVT gaging, helium loading, pump/compressor operation, and vent operations. The minimum temperatures to be monitored within the receiver vehicles during resupply are helium and propellant tank locations. The thermocouple locations are shown in Figure 32 and a brief description of each instrument and how it will be used is shown in Table 12. The thermocouples on the quick disconnects are not discussed in this letter as they are recommended to be an integral part of the disconnect. The sampling rate for adequate microprocessor control is recommended to be one sample per second during resupply and quiescent periods.

Valve Position Indication Feedback Switches - Each valve will be equipped with redundant position indication feedback switches.

Flowmeters - Two flowmeter feedback channels are required to monitor flow in both the forward and reverse flow directions.

Contact Sensors - Contact sensors are necessary to verify proper mating of the quick disconnect panel and can be of the simple spring loaded type.

Commonalities with On-Going Programs and the ERM

The expendable resupply module (ERM) is compatible with two on-going programs and a few flight demonstrations. The two programs of interest are: the orbital maneuvering vehicle (OMV) and the orbital spacecraft consumables resupply (OSCRS). As is seen in Figure 33, OSCRS, OMV, and the ERM are expected to be operational by 1990, 1991-2, and 1993 +, respectively. Therefore, a reduction in development testing of the ERM can be expected by using the developmental results of OSCRS and OMV.

The following development tests are required for the ERM.

A) Rendezvous with active or passive targets.

B) Docking with subsequent release.

C) Mating and alignment for the fluid interface with latching.

D) Propellant transfer - basic function, multiple transfers, connection reliability with leakage monitoring. Major test components - propellant pump, coupling, and tank acquisition device with ullage control.

E) Pressurant transfer - basic function, multiple transfers, connection reliability with leakage monitoring. Major test components - compressor and coupling.

SENSOR I.D.	LOCATION	SYSTEM STATUS MONITORING			
		GENERAL HEALTH	RESUPPLY OPERATIONS	FAULT DETECTION	GAGING
P1 THRU P6 P6A P6B	HELIUM TANK INLET/OUTLET	• MONITOR HELIUM TANKS	• HELIUM LOADING • HELIUM TRANSFER • PROPELLANT TRANSFER • DISCONNECT PURGE • HELIUM SCAVENGING	• HELIUM LEAKAGE O VALVES O REGULATORS O COMPRESSORS • OVERPRESSURE LIMITS	• PVT SENSORS
P7, P8	COMPRESSOR INLET, REGULATOR OUTLET	• MONITOR REGULATOR OUTLET & COMPRESSOR INLET	• HELIUM LOADING • HELIUM SCAVENGING	• HELIUM LEAKAGE • REGULATOR FAILURE	• NOT APPLICABLE
P9	COMPRESSOR OUTLET	• MONITOR COMPRESSOR OUTLET	• HELIUM LOADING • HELIUM SCAVENGING	• COMPRESSOR FAILURE	• NOT APPLICABLE
P10, P11	PROPELLANT ULLAGE	• MONITOR REGULATOR OUTLET & TANK ULLAGE	• PROPELLANT LOADING • PROPELLANT TRANSFER • DISCONNECT PURGE • PROPELLANT SCAVENG-ING	• LEAKAGE • TANK BURST LIMITS	• PVT SENSOR
P12	VENT INLET	• MONITOR VENTS	• VENT ULLAGE • VENT WASTE TANKS	• RELIEF VALVE ACTUATION	• NOT APPLICABLE
P13, P14	PROPELLANT TANK OUTLET	• MONITOR TANK OUTLET AND FLOWMETER INLET	• PROPELLANT LOADING • PROPELLANT TRANSFER • PROPELLANT SCAVENG-ING	• LEAKAGE • FLOWMETER LIMITS	• NOT APPLICABLE
P15, P16	PUMP OUTLET	• MONITOR PUMP PERFORMANCE	• PROPELLANT TRANSFER • PROPELLANT SCAVENG-ING	• COMPRESSOR OVER/REVERSE PRESS.	• NOT APPLICABLE
P17	WASTE TANK INLET	• MONITOR WASTE TANK ULLAGE	• DISCONNECT PURGE • ULLAGE VENT/DIS-POSAL	• TANK BURST LIMITS • LEAKAGE	• NOT APPLICABLE
P18, P19, P20, P21	QUICK DISCONNECT MANIFOLD INLET	• MONITOR MANIFOLD	• QUICK DISCONNECT ENGAGEMENT/DISENGAGEMENT	• LEAKAGE • QUICK DISCON-NECT OVER PRESS	• NOT APPLICABLE

TABLE 11. PRESSURE INSTRUMENTATION

SENSOR I.D.	LOCATION	SYSTEM STATUS MONITORING			
		GENERAL HEALTH	RESUPPLY OPERATIONS	FAULT DETECTION	GAGING
T1, T2, T3, T3A	HELIUM TANKS	• MONITOR HELIUM TANKS	• HELIUM LOADING • HELIUM TRANSFER • HELIUM SCAVENGING • PROPELLANT TRANSFER • DISCONNECT PURGE	• HELIUM LEAKAGE • OVER-TEMPERATURE LIMITS	• PVT SENSORS
T4, T5	COMPRESSOR HOUSING	• MONITOR HOUSING	• HELIUM TRANSFER • HELIUM SCAVENGING	• OVER-TEMPERATURE LIMITS	• NOT APPLICABLE
T6	COMPRESSOR OUTLET	• MONITOR COMPRESSOR PERFORMANCE	• HELIUM TRANSFER • HELIUM SCAVENGING	• OVER-TEMPERATURE LIMITS	• NOT APPLICABLE
T7	VENT INLET	• MONITOR VENT	• VENT ULLAGE • VENT WASTE TANKS	• VENT FREEZING	• NOT APPLICABLE
T8, T9, T10	PROPELLANT TANKS	• MONITOR PROPELLANT TANKS	• PROP. LOADING • PROP. TRANSFER • PROP. SCAVENGING	• TANK LIMITS	• PVT SENSOR • CORRECT FLOWMETER FOR DENSITY
T11, T12	PUMP HOUSING	• MONITOR HOUSING	• PROP. TRANSFER • PROP. SCAVENGING	• OVER-TEMPERATURE LIMITS	• NOT APPLICABLE
T13, T14	PUMP OUTLET	• MONITOR PUMP PERFORMANCE	• PROPELLANT TRANSFER • PROPELLANT SCAVENGING	• PUMP OUT-FLOW LIMITS	• CORRECT FLOWMETER FOR DENSITY
T15	WASTE TANK	• MONITOR WASTE TANKS	• DISCONNECT PURGE • ULLAGE VENT/DISPOSAL • WASTE DISPOSAL	• COMPRESSION LIMITS • OUTFLOW LIMITS	• NOT APPLICABLE
T16 THRU T25	DISCONNECT INLETS	• MONITOR DISCONNECT INLET	• LOADING ○ TRANSFER ○ SCAVENGING ○ PURGING	• DISCONNECT TEMP LIMITS • DISCONNECT LEAKAGE	• NOT APPLICABLE

TABLE 12. TEMPERATURE INSTRUMENTATION

SCHEMATIC

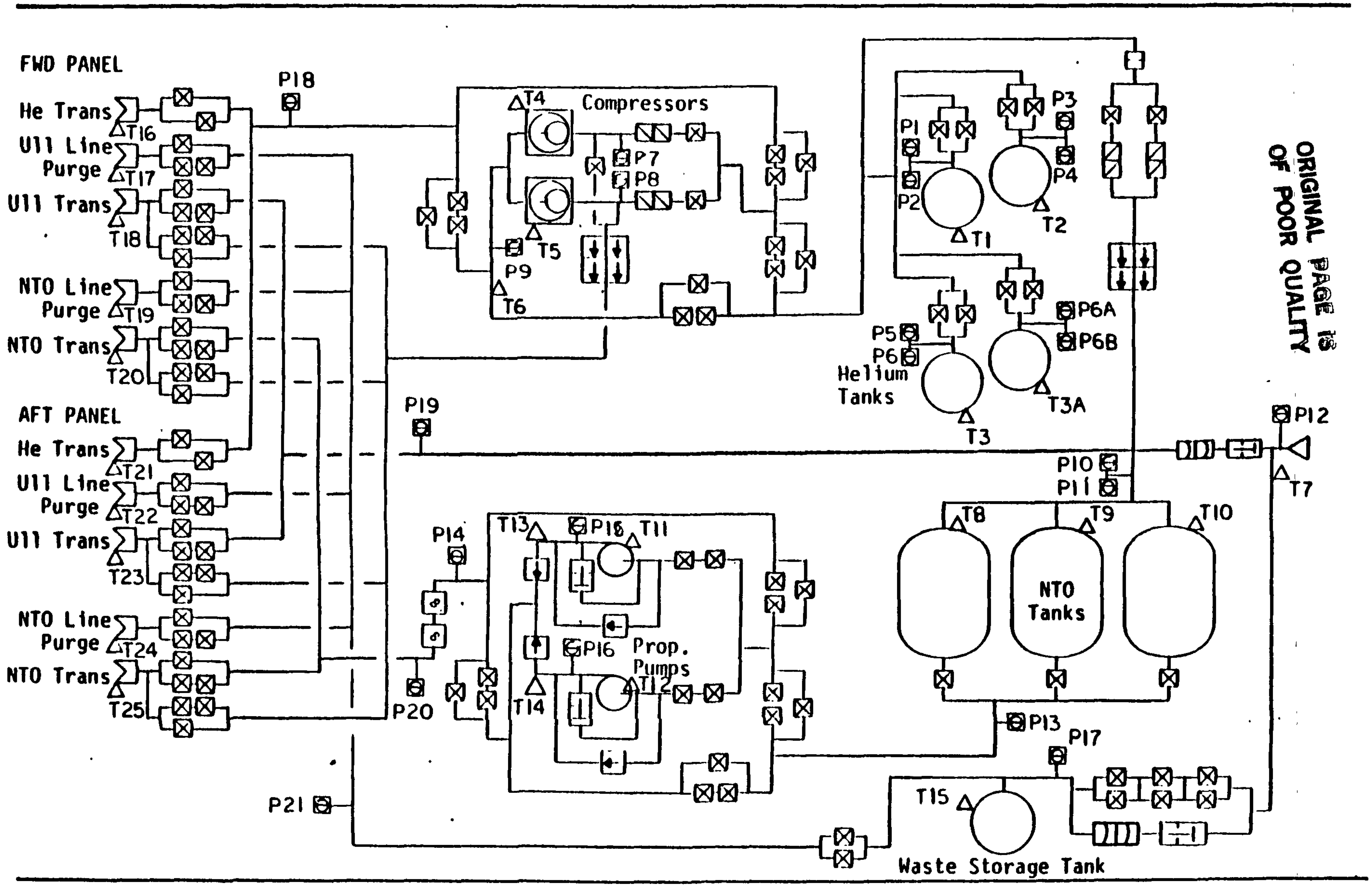


FIGURE 32. TEMPERATURE AND PRESSURE LOCATION

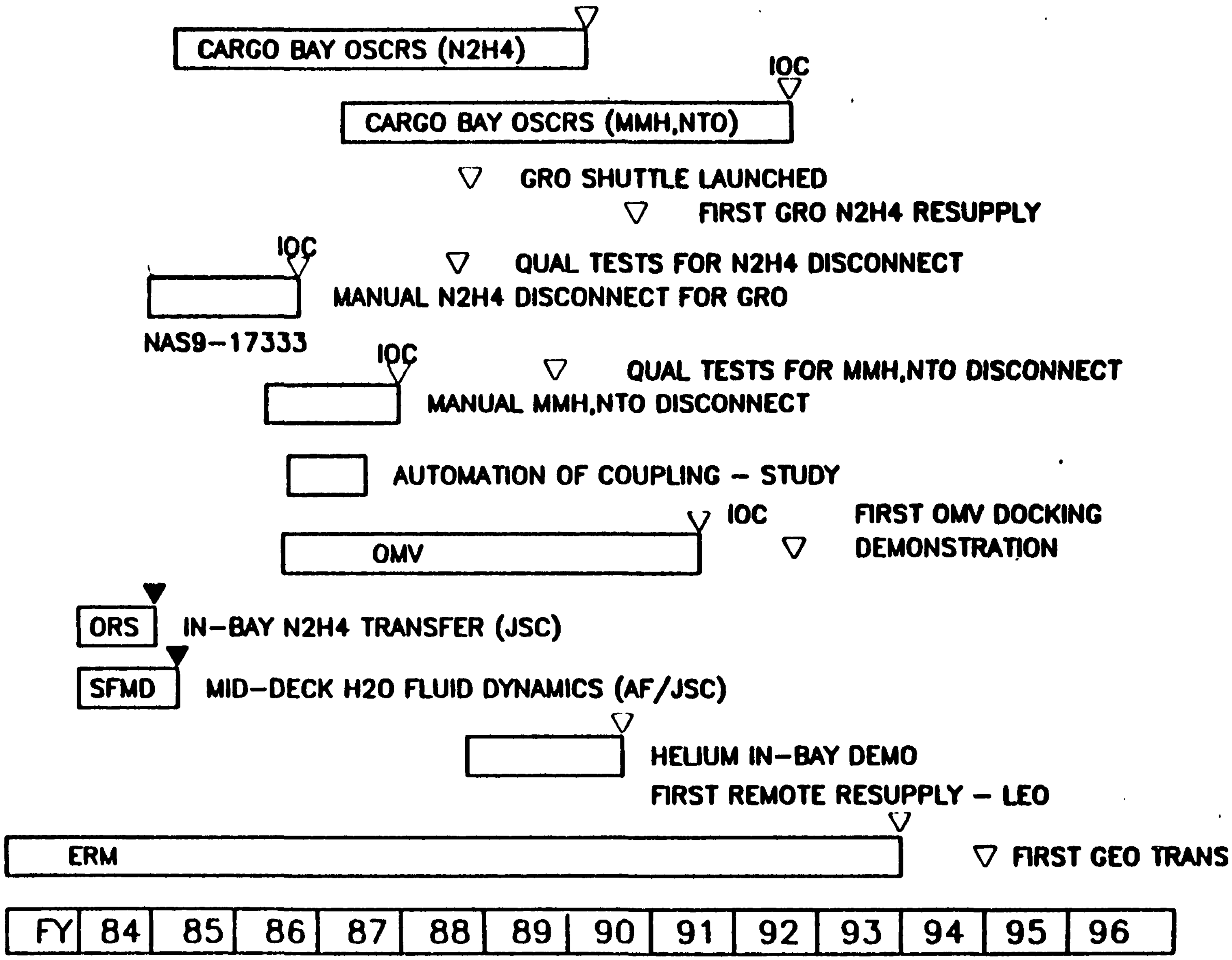


FIGURE 33. REPRESENTATIVE INTERACTION OF REMOTE RESUPPLY DEVELOPMENT

A few compatible areas between the OSCRS and the ERM, are as follows:

1) Some form of docking mechanism will be required for the ERM, OSCRS will demonstrate the use of the MMS berthing system.

2) OSCRS will perform a propellant transfer in which a manual coupling device will be used. This will allow testing/demonstration of the coupling system.

3) Heat transfer determinations during ullage recompression in the receiver diaphragm tank will help to establish optimum flow rates.

A manual coupling is being developed by Fairchild. The coupling is to be developed as a standardized coupling for N_2H_4, MMH, NTO, GN_2, GHe, and $C_2Cl_3F_3$ (trichlorotrifluroethane). OSCRS' usage of the coupling will be to transfer hydrazine to the GRO in 1990, but by using appropriate seals all listed fluids can be transferred. The Fairchild coupling will be available by mid 1986 to transfer N_2H_4 and it could be available by the end of 1987 to transfer other fluids. The coupling is manually engaged and five inhibits must be manually actuated in sequential order to allow a propellant transfer. In Figures 34 & 35 the inhibits are labeled T1-T3 for the tanker side and S1-S3 for the spacecraft side. The inhibits cannct be opened until the two halves of the coupling are engaged or the coupling disengaged before the inhibits are manually closed. The manual engagement and disengagement force is about 36 pounds. A rough leak detection system will be performed by pressure transducers at the indicated ports shown in Figure 34. A weight analysis indicates that the tanker and spacecraft coupling will weight 13.0 and 10.8 pounds, respectively. Fairchild also has funding to start preliminary designs for the automated coupling in early 1987. Fairchild indicated that the present manual design will not be automated as is.

Another possible area of compatibility that OSCRS may have with the ERM is the development of a propellant pump. At this time the use of a propellant pump is only considered a option for hydrazine transfer.

The OMV is a Shuttle-based/launched, remotely controlled, free-flying vehicle capable of performing a wide range of on-orbit services. The initial OMV (launched about mid 1991) will perform payload placement and/or retrieval in LEO when delivered to standard STS orbit by the orbiter. As the space program advances and mission needs materialize the reference design OMV can be upgraded by modifications to perform more sophisticated missions, to be space-based at either LEO or GEO, and to accommodate DOD missions.

There are two areas of compatibility between the ERM and the OMV, they are as follows.

1) Rendezvous with a cooperative or non-cooperative target.

2) Remote docking with subsequent release.

The first OMV docking will probably occur with the Shuttle in mid 1992. All development required for rendezvous and remote docking will be performed on the OMV before the ERM.

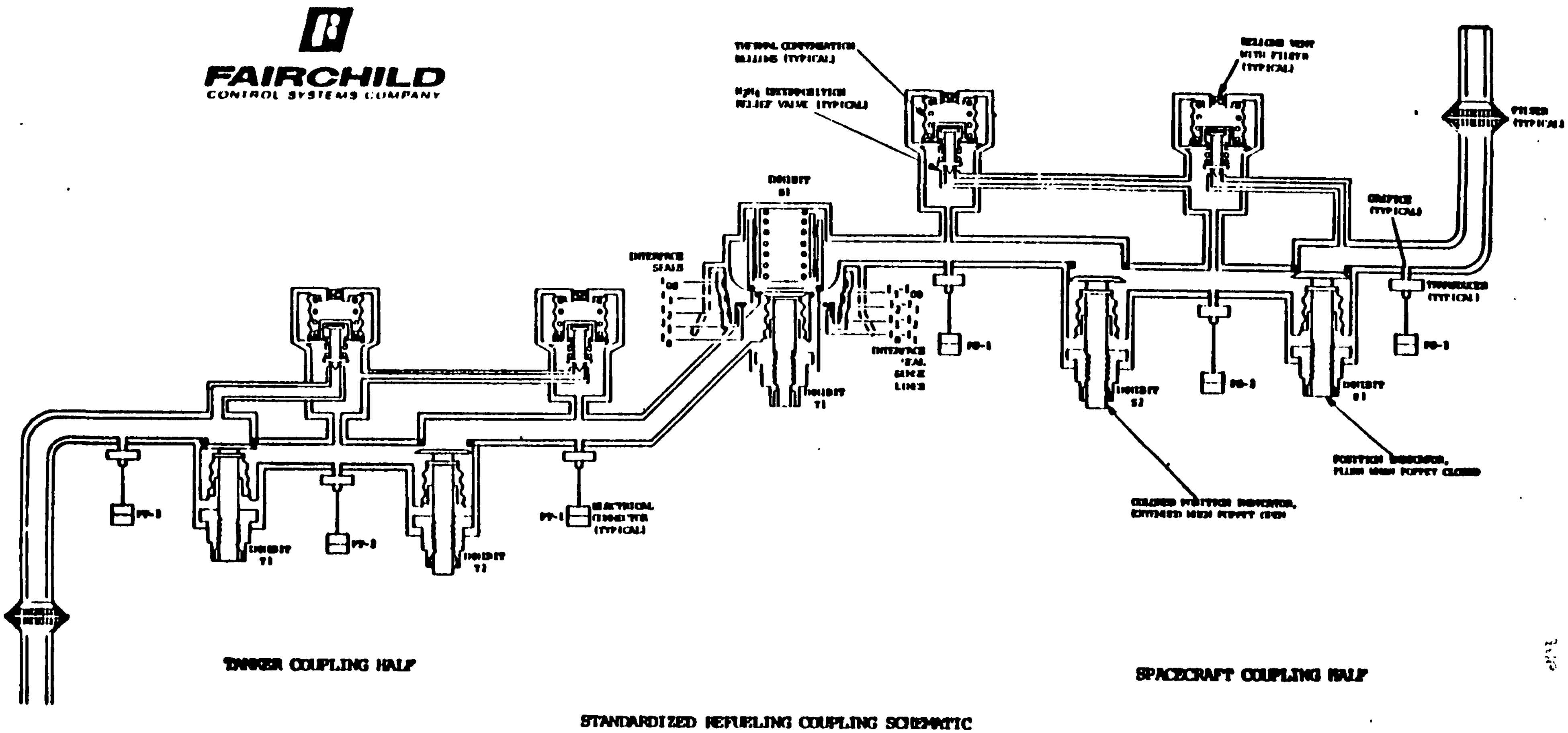


FIGURE 34. STANDARDIZED REFUELING COUPLING SCHEMATIC

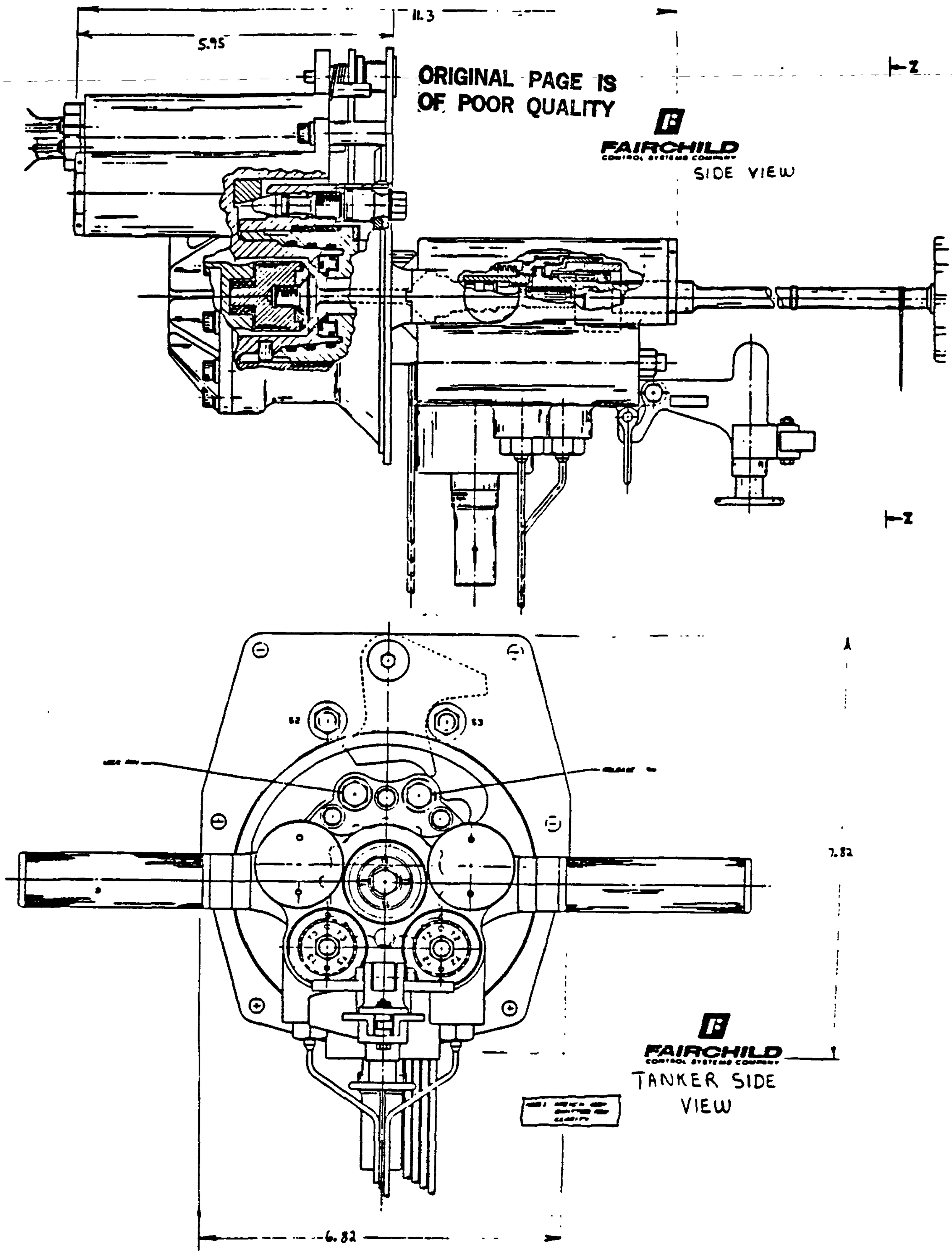


FIGURE 35. FAIRCHILD MANUAL COUPLING

The ERM will be able to use only limited developmental work from OSCRS and the OMV. OSCRS will provide information on the propellant coupling, propellant transfer, leak detection, and docking with the MMS berthing system. OMV will provide information on the rendezvous and remote docking and release with other spacecraft.

<u>On-Orbit Middeck Ullage Transfer Experiment</u>

MUTE is an Orbiter middeck experiment designed to investigate micro-gravity fluid transfer dynamics using a water solution, capillary/screen type tanks, micro-gravity gaging system, and a small reversible fluid pump. The fundamental objectives of MUTE are as follows:

1) To demonstrate the concept of ullage transfer in a micro-gravity environment.

2) To demonstrate the propellant management device's (PMD's) ability to position and control the ullage bubble by observing the ullage transfer process. The PMD will include a ullage/propellant separator and will be placed in each tank.

3) To demonstrate a micro-gravity quantity gaging system's accuracy in a micro-gravity environment.

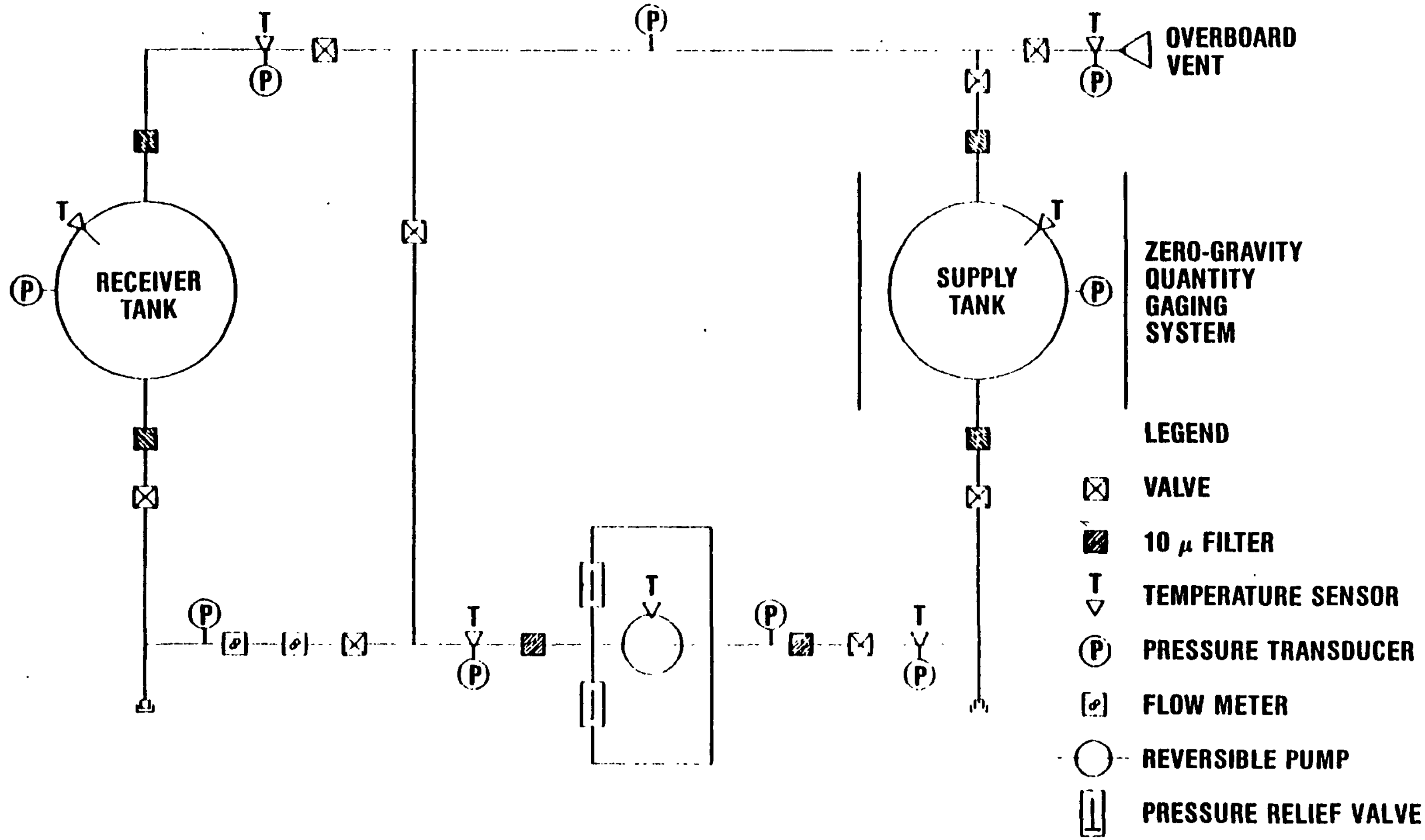
These objectives are to be met by using off-the-shelf equipment and by modeling the structure and tanks after the storable fluid management demonstration (SFMD).

MUTE consists of a supply tank, a receiver tank, a PMD in both the supply and receiver tank, a reversible pump, a micro-gravity gaging system, a flow meter, and associated valves, filters, lines, pressure gauges, and temperature sensors. MUTE has no power or command interfaces with the Orbiter; and is basically self-contained, with the exception of an overboard vent connected to the Orbiter waste management system. The entire experiment will be designed to fit in place of several lockers in the orbiter middeck. Support equipment, from STS, will include cameras and associated lights, power to be supplied by Orbiter outlets, (for the cameras and associated lights) a micro-gravity accelerometer measurement system (MGAMS), and a data recorder.

Control of MUTE on-orbit will be accomplished by the crew through a sequence of manual valve and switch operations. The small fluid pump is the motive force for a series of test fluid transfers into and out of the receiver tank. Data sources will consist of the video cassette recorder, 35 mm still photos, MGAMS acceleration data, data recorder, and astronaut comments/logs. All data inputs will be correlated.

A preliminary schematic to perform the recommended tests is presented in Figure 36. The recommended tests are presented in Table 13, with expected test objectives in Table 14. The following is a few general operating comments:

1) Prior to tests, the crew will connect the MUTE's vent line to the Orbiter waste water overboard vent system.



SIMPLICITY, BENIGN FLUIDS, & MAX. USE OF EXISTING EQUIPMENT — MINIMIZES EXPERIMENT COST, REDUCES PROGRAM RISK

FIGURE 36. MID-DECK ULLAGE TRANSFER EXPERIMENT SCHEMATIC

Test No.	Transfer Fluid from Supply Tank to Receiver Tank	Transfer Fluid from Receiver Tank to Supply Tank
1.	Transfer to dry tank at low flow rate with ullage exchange. Stop filling at each 10% of mass transfer.	Transfer fluid back to supply tank at a low flow rate with ullage exchange.
2.	Transfer to wet tank at low flow rate with ullage recompression until tank is 50% full.	Transfer fluid back to supply tank at a low flow rate with ullage recompression.
3.	Transfer to wet tank at moderate flow rate with ullage exchange.	Transfer fluid back to supply tank at a moderate flow rate with ullage exchange.
4.	Transfer to wet tank at high flow rate with ullage exchange.	Transfer fluid back to supply tank at a high flow rate with ullage exchange.
5.	Transfer to wet tank at low, moderate, and high flow rates with ullage exchange.	Transfer back to supply tank at a moderate flow rate with ullage exchange.
6.	Transfer 75-80% of fluid to wet tank at moderate flow rate with ullage exchange during various Orbiter accelerations. Stop flow and accelerate until screen breakdown. Wait for rewetting of screen to occur before transferring remaining fluid.	Transfer fluid back to supply tank at a moderate flow rate with ullage exchange.

TABLE 13. RECOMMENDED TESTS

Objective	Test No. 1	2	3	4	5	6
1) To evaluate PMD's Ullage Control						
propellant/ullage separation	X		X	X	X	X
wetting pattern of PMD	X		X	X	X	X
wetting time of PMD	X		X	X	X	X
orbiter acceleration effects						X
2) To evaluate Zero-Gravity Gaging System						
accuracy determinations						
steady state flow rate	X	X	X	X		
variable flow rate	X				X	
orbiter acceleration						X
response time						
steady state flow rate	X	X	X	X		
variable flow rate	X				X	
orbiter accelerations						X
comparison to PV gaging system		X				

TABLE 14. MATRIX OF TEST OBJECTIVES

Component	Weight (lbs) each	system
SFMD		250 lbs
removal of valves, gas cylinders, and cylinder		-50 lbs
pressure transducers	2	12
temperature sensor	1	7
pump		5
batteries		25
flow meters	3	6
zero-gravity gaging system		40
+10%		33
Total		363 lbs

TABLE 15. MUTE SYSTEM WEIGHT

2) After each test the test fluid will be returned to the supply tank before going on the next test.

3) All test operations are controlled manually by the crew using the proper valves, switches, and instrumentation.

MUTE's system weight is presented in Table 15. Since MUTE is modeled after SFMD, SFMD's system weight was used as a baseline by removing weights of components not required and then adding in new component weights plus an extra 10%. MUTE's system weight is expected to be about 360 lbs.

Orbiter Cargo Bay Fluid Transfer Flight Demonstration

FDTV will be used in the Orbiter cargo bay to demonstrate a micro-gravity remote fluid transfer. There will be two brass board vehicles capable of being separated or docked using the remote manipulator subsystem (RMS). The upper portion of the test vehicle, the receiver vehicle, piggybacks on the supply vehicle through deployable payload latches. The RMS system is utilized to separate the receiver vehicle, by use of a grapple fixture attached to the receiver vehicle, to simulate remote docking and mating of the two vehicles. A snare type end effector, incorporated at the fluid transfer interface of the supply vehicle, accomplishes final mating of the fluid transfer interface panels on each brass board test vehicle.

The primary test objectives to be performed on the FDTV are as follows:

1) To demonstrate proper docking and alignment of the fluid transfer interface during a remote operation.

2) To demonstrate proper QD mating, leak integrity verification, operation, and purging with safing, before, during, and after a fluid transfer.

3) To evaluate the PMD's ullage control and separation capability during a propellant transfer.

4) To evaluate propellant transfer by the methods of ullage transfer and ullage recompression. Different pumped flow rates, and the ability of the tanker to resupply itself will also be examined during the ullage transfer tests.

5) To evaluate pressurant transfer by two methods: 1) a pressurant tank cascade and 2) by compressor usage.

6) To evaluate the micro-gravity quantity gaging system for both accuracy determinations and variable flow rate effects.

7) To demonstrate thermal and remote system control.

These test objectives are to be met by using the brass board test components and as many off-the-shelf components as possible.

FDTV contains four forward RCS propellant tanks, consisting of two receiver and two supply tanks, for the MMH/NTO propellant transfer. All four tanks will be aligned with the +X-axis of the Orbiter's coordinate system during launch and contain a 60% nominal load (55% of total volume). Only minor modifications to the pressurant inlet area of the RCS propellant tank will be made so that ullage/propellant separation will occur. The propellant transfer will be accomplished by the use of three small propellant pumps. As is seen in Figure 37, the MMH side will provide pumping to the receiver tank and will also resupply the tanker side by using only one pump. The NTO side, Figure 38, will transfer propellant by use of two pumps, one on the receiver vehicle and one on the supply vehicle with simpler valving. There are two different pressurant transfer systems for comparison and demonstration purposes. The MMH side will perform a pressurant cascade from four 4800 psia tanks to fill a RCS helium tank to 3600 psia, while the NTO side will use a compressor to perform the pressurant transfer after a blowdown pressure equalization transfer.

Control of FDTV on-orbit will be accomplished by the crew through a sequence of remote operations. The RMS will be the motive force during mating and demating steps, requiring intensive crew attention; but the resupply process should only require crew monitoring with a minimum interaction between the crew and the FDTV.

Table 16 presents the recommended tests to be performed by FDTV to satisfy the test objectives presented in Table 17. Each of the four tests are broken down into their basic steps required for the completion of each test. The first test will examine the docking, mating, leak integrity verification, purging, safing, and the demating process. The second test will perform the pressurant transfer by the cascade method and compressor method, and perform an ullage recompression propellant transfer. The third test will perform an ullage transfer at a low flow rate. The last test will demonstrate the ullage transfer at a higher flow rate. Propellant transfer will occur back and forth between the receiver and supply tanks during each test.

FDTV's system weight is presented in Table 18. The expected dry weight is 3309 lbs. To this was added a 15% potential growth factor plus the required propellant and pressurant requirements for a total payload weight of 6735 lbs.

The purpose of the ERM-GTP is to develop the Flight Demonstration test vehicle (FDTV) to be used to conduct the fluid transfer during the ERM Flight Demonstration (ERM-FD). The FDTV shall be capable of conducting remote mating and demating operations, performing propellant transfer by means of "ullage transfer" and "ullage recompression" methods, together with the transfer of pressurant by means of "cascade" and compressor methods. In addition the test vehicle shall contain a propellant management device (PMD) capable of separating gas and liquid. (for ullage transfer tests) and a zero-gravity fluid quantity gaging system (gaging system) to determine the amount of propellant in the tanks. To date, the ERM study has defined the need to develop three critical components for this transfer demonstration. Reiterating, these components are the propellant transfer pump, the pressurant compressor and the mating fluid couplings. It should be noted that for this program it shall be assumed that the NASA will develop the couplings for this application (including remote configuration) and that this development effort shall produce qualified units applicable for the ERM-FD test vehicle. The

Test No.	Steps in Each Test
1	1) Unlatch receiver vehicle from the supply vehicle. 2) Using the RMS move receiver vehicle away from the supply vehicle and back again for docking step. 3) Docking and alignment of interface. 4) QD mating. 5) QD leak integrity verification 6) QD purging and safing. 7) Demating of QD. 8) Using the RMS move receiver vehicle away from the supply vehicle and back again for docking step.
2	1) Docking and alignment of interface. 2) QD mating. 3) QD leak integrity verification. 4) Transfer pressurant by cascade method on the MMH side. 5) Transfer pressurant by compressor method on the NTO side. 6) Using supply pump transfer MMH by the ullage recompression method from 60 to 90% of the filled volume. 7) Using supply pump transfer NTO by the ullage recompression method from 60 to 90% of the filled volume.
3	1) Allow pressurant to equilibrate the pressure on the supply and receiver propellant lines and tanks; then prepare for the ullage transfer method. 2) Using the supply pump transfer MMH by ullage transfer from the receiver tank back to the supply tank, from 90 to 30% of the filled volume. 3) Using the receiver pump transfer NTO by ullage transfer from the receiver tank back to the supply tank, from 90 to 30% of the filled volume. 4) QD safing and purging. 5) Demating of QD 6) Disengagement and separation of vehicles.
4	1) Docking and alignment of interface. 2) QD mating. 3) QD leak integrity verification. 4) Using the supply pump resupply the MMH receiver tanks by ullage transfer at a higher flow rate from 30 to 90% of the filled volume. 5) Using the supply pump resupply the NTO receiver tanks by ullage transfer at a higher flow rate from 30 to 90% of the filled volume. 6) QD safing and purging. 7) Demating of QD. 8) Disengagement of and separation of vehicles. 9) Latch receiver vehicle to supply vehicle.

TABLE 16. RECOMMENDED TESTS

Objectives	Test Numbers 1	2	3	4
1) To Demonstrate Docking and Alignment of Interface	X	X		X
2) To Demonstrate QD Integrity Verification, which include:	X	X		X
QD Mating	X	X		X
Leak Integrity Verification	X	X		X
Purging and Safing	X		X	X
3) To Evaluate the PMD's Ullage Control with:		X	X	X
Propellant/Ullage Separation			X	X
Propellant Supply Capability		X	X	X
4) To Evaluate Propellant Transfer by:		X	X	X
Ullage Transfer with:			X	X
Pump Usage			X	X
Different Flowrates			X	X
Tanker Resupply			X	
Ullage Recompression with:		X		
Thermal Control		X		
Pump Usage		X		
5) To Evaluate Pressurant Transfer by the:		X		
Cascade Method		X		
Compressor Method		X		
6) To Evaluate Micro Gravity Gaging System for:		X	X	X
Accuracy Determinations		X	X	X
Flow rates Effects		X	X	X
7) To Demonstrate Thermal & System Control	X	X	X	X
8) To Demonstrate Demating and Separation of Vehicles	X		X	X

TABLE 17. MATRIX OF TEST OBJECTIVES

Subsystems	Weight
1) Fluid Transfer System	
propellant tank	300
pressurant tank	287
waste tank	8
pump	60
compressor	200
disconnects	240
valves, regulators, etc.	325
pressure end temperature sensors	90
2) Interface panels	
end effector	65
grapple fixture	44
face plates	38
latch	20
electrical connects	8
3) Structural System	
frame and support	665
fittings and latches	624
thermal blankets and covers	125
4) *Avionics/Power	
Wire harness	40
regulation & distribution	100
thermal control	30
on-board data management	40
Subtotal	3309
+15% Growth	496
5) Propellant and Pressurant	2930
Total Payload Weight	6735 lbs.

*Allocation - Analysis to be performed at a later date.

TABLE 18. FLIGHT DEMONSTRATION TEST VEHICLE SYSTEM WEIGHT

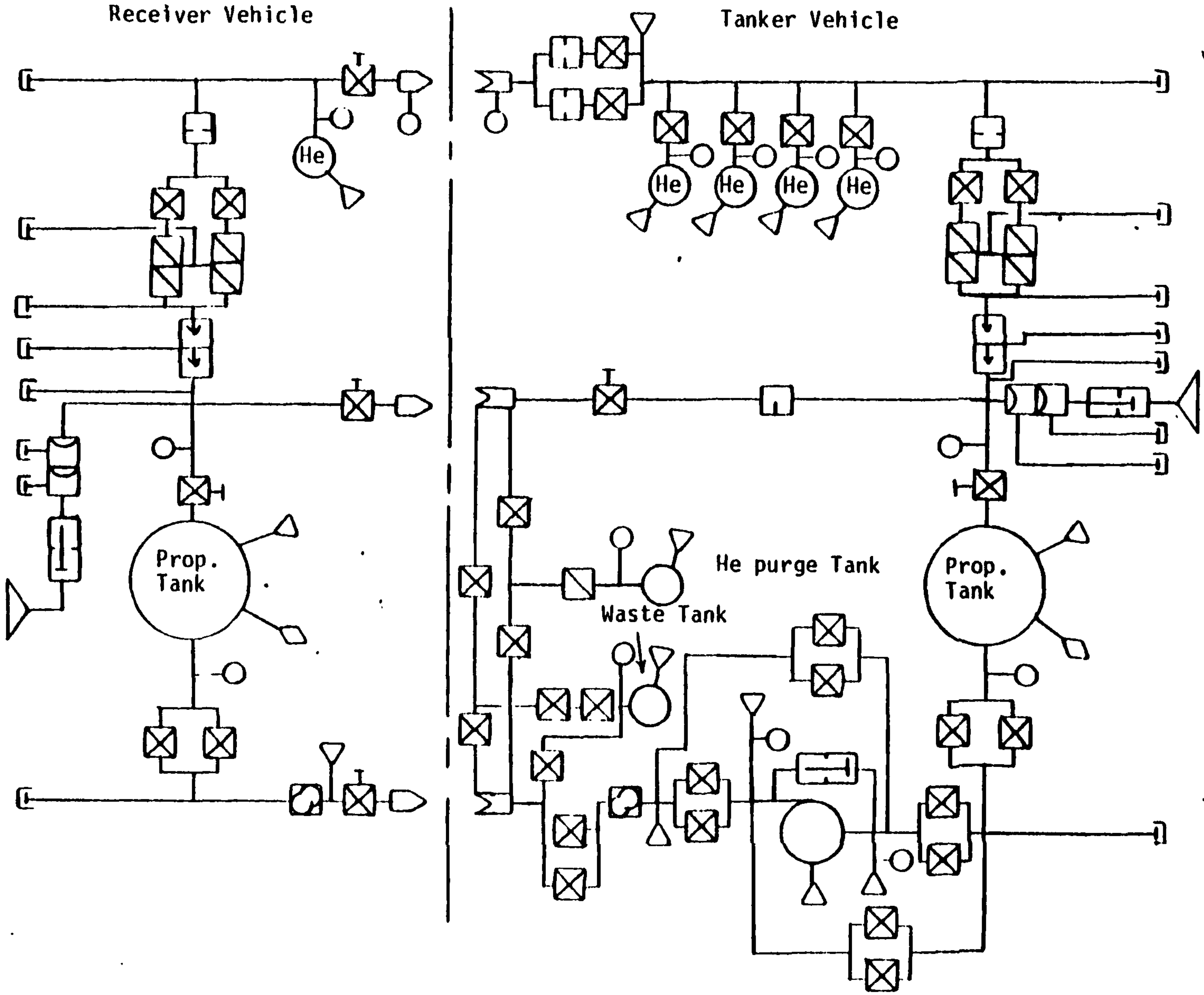


FIGURE 37. FLIGHT DEMONSTRATION TEST VEHICLE SCHEMATIC - MMH HALF

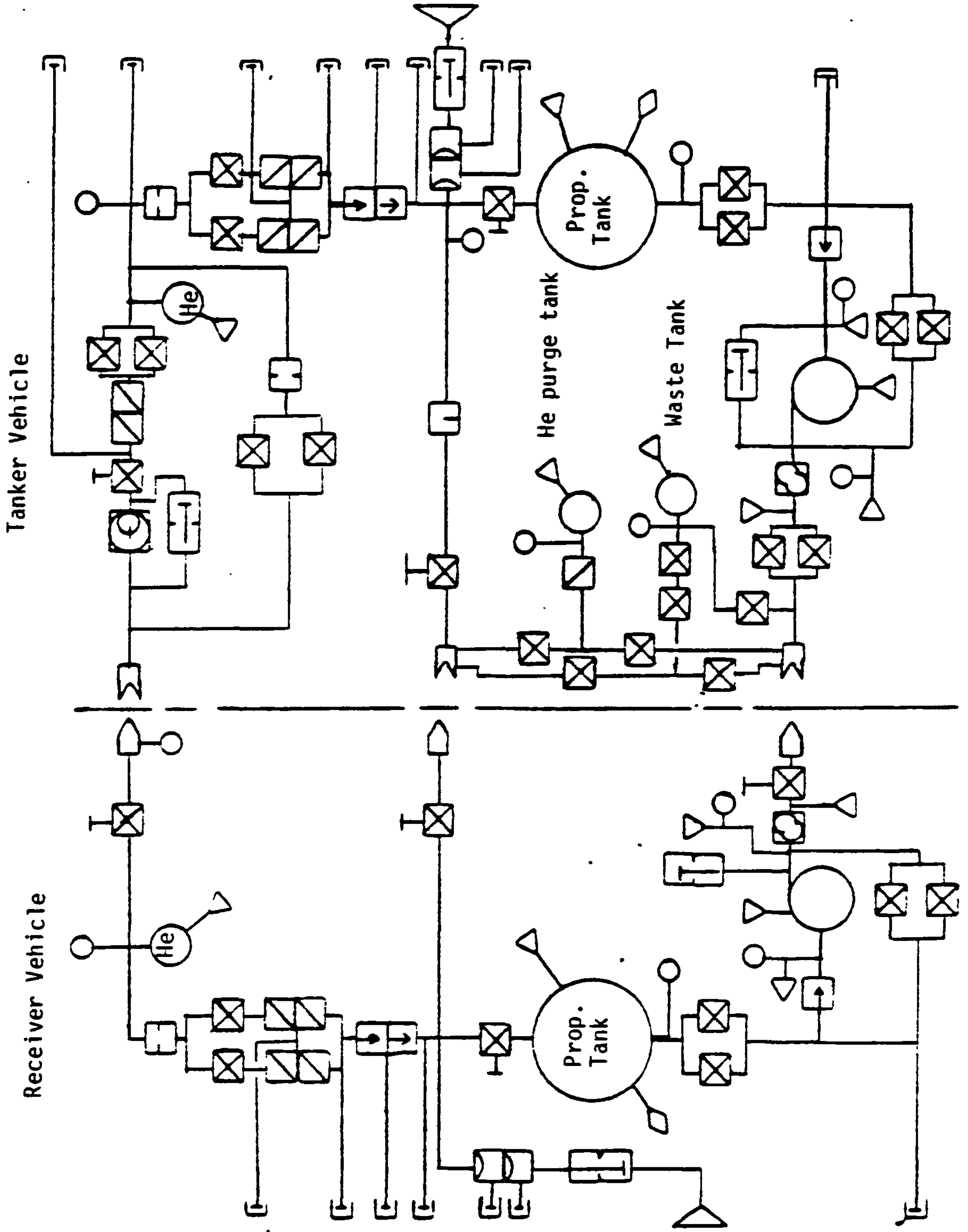
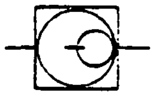


FIGURE 38. FLIGHT DEMONSTRATION TEST VEHICLE SCHEMATIC - NTO HALF

Legend

 check valve

 compressor

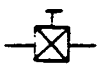
 dual burst disk

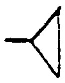
 dual pressure regulator

 flowmeter

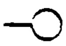
 gaging system

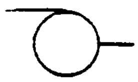
 liquid propellant detector

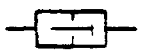
 manual valve

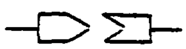
 non propulsive vent

 orifice

 pressure sensor

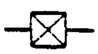
 pump

 relief valve

 remote quick disconnect

 servicing/test port

 temperature sensor

 valve

gaging system, which will be developed under a separate contract, shall be customized to the test vehicle tankage and verified during the ERM-GTP. The development of the liquid/gas separator unit of the PMD shall be tested and/or analytically assessed as a separate unit, with no plans to assess its operation during the system fluid transfer ground testing.

Ground Test Program (GTP)

The ERM-GTP will be divided into four major tasks:

1) Pump and compressor specifications, including supplier design and fabrication of breadboard test units.

2) Component testing of pump and compressor breadboard units. Testing and/or evaluation of the liquid/gas separator together with the gaging system configuration requirements will be performed during this time period.

3) FDTV design and fabrication. During this time frame the test and checkout of the coupling mating assemblies to assure alignment and determine engagement and separation loads will be performed. Procedures for coupling operation will be assessed.

4) FDTV development and verification test program. This period is divided into two test series with allowance for potential modifications between the two test series. During this period, low level support using breadboard development test apparatus may be advantageous and cost effective to isolate unique component problem areas.

The ERM-GTP proposed timeline is presented in Figure 39.

Test Program

The Test Program is divided into two major areas (1) breadboard testing of prototype (pump and compressor) units to assess component features unique to space application and for proof of concept of required advanced state-of-the-art component features and (2) Brassboard testing of the FDTV transfer system using flight components and hardware.

Breadboard Tests

Prior to these tests, a final assessment of component design from various potential suppliers shall be performed and the most promising of these components will be selected for breadboard (B/B) development. The breadboard test results will be used to enable the selection of a unique component design for each fluid transfer application. If, based upon the B/B tests there are more than one viable component design for a unique application both designs will be further evaluated during the FDTV brassboard test program. The breadboard test results will also enable the finalization of component design specification requirements.

The B/B test effort will be concentrated chiefly to the development of the propellant pump, gas compressor (includes assessment of gas and compressor cooling requirements), and the liquid/gas separator device.

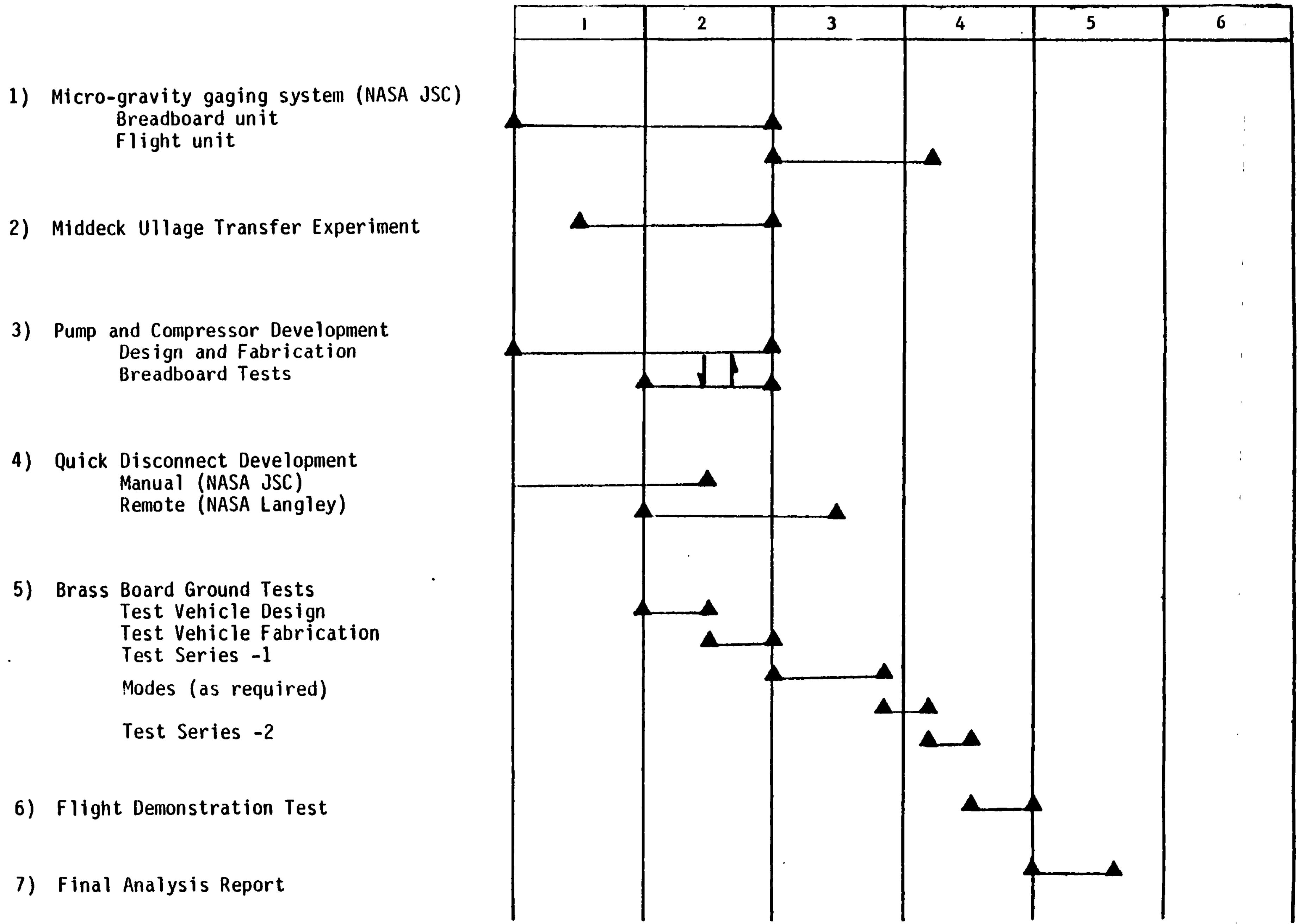


FIGURE 39. EXPENDABLE RESUPPLY MODULE TEST PROGRAM

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Propellant Pump Tests

The breadboard test setup shown in Figure 40 shall be the test-bed for the evaluation of potential pump configurations to be used in the FDTV. Flight configuration flowmeters, if available, shall also be incorporated in order to enhance test results. Three types of pumps shall be evaluated (Gear, multi-stage centrifugal, and piston), however, only the two types with the highest potential will be selected for breadboard development. All breadboard testing will be performed utilizing water and/or freon-113 as the test fluid.

The test program will consist of 12 flow tests using the ullage transfer technique (4 flow tests each at 5, 10, and 15 gpm), and 12 flow tests using the ullage recompression technique (4 flow tests each at 1, 3, and 5 gpm).

The primary test objectives are to assess the pump performance and, if necessary, recommend design changes based upon the test results. Detailed test objectives are listed below:

1) Evaluate pump performance using ullage transfer method and various flowrates and tank pressure conditions.

2) Evaluate pump performance using ullage recompression method and various flowrates and tank pressure conditions.

3) Define any pump deficiencies and provide recommended changes to the flight-type pump designs.

4) Obtain preliminary data on ullage recompression receiver tank ullage temperature effects (secondary objective).

5) Pump power requirements assessment.

6) Determine the pump magnetic coupling drive performance at various pump speeds and pressure differentials (optional only if magnetic coupling is used).

Gas Compressor Tests

The compressor breadboard test setup is shown in Figure 41. A series of helium flow tests at various flowrates, temperatures and pressures will be performed to assess compressor performance and design parameters. The objectives for these tests are as follows:

1) Evaluate compressors performance at various temperatures, pressures, and flowrates. See test matrix in Table 19.

2) Evaluate compressor cooling requirements.

3) Obtain preliminary data on requirements far receiver tank gas cooling.

4) Compressor power requirements assessment.

5) Evaluate the compressor magnetic coupling drive performance at various flow conditions. (Optional only if magnetic coupling is used).

TEST CONDITIONS (Pressure = PSIA, Flowrates = SCFM)	TEST NO. 1,2	3	4	5	6	7	8	9,10
1. COMPRESSOR								
INLET PRESSURE A. 500; B. 700	A	A	A	A	B	B	B	B
MAXIMUM OUTLET PRESSURE A. 3000; B. 5000	A, B	A, B	A, B	A, B	A, B	A, B	A, B	A, B
COMPRESSOR FLOWRATES A. 5; B. 10	A, B	A, B	A, B	A, B	A, B	A, B	A, B	A, B
GAS INITIAL INLET TEMPERATURE A. 70°F, B. 100°F;	A	B	A	B	A	B	A	B
2. AMBIENT CONDITIONS								
PRESSURE A. Ambient; B. Vacuum	A, B	A	A	A	A	A	A	A, B
TEMPERATURE A. 70°F; B. 100°F	A	B	A	B	A	B	A	B
3. HEAT EXCHANGER								
FLUID TEMPERATURE A. 40°F; B. TBD	A, B	A	A, B	A	A, B	A	A, B	A, B
LIQUID FLOWRATE A. TBD	A	A	A	A	A	A	A	A

TABLE 19. COMPRESSOR B/B TEST MATRIX

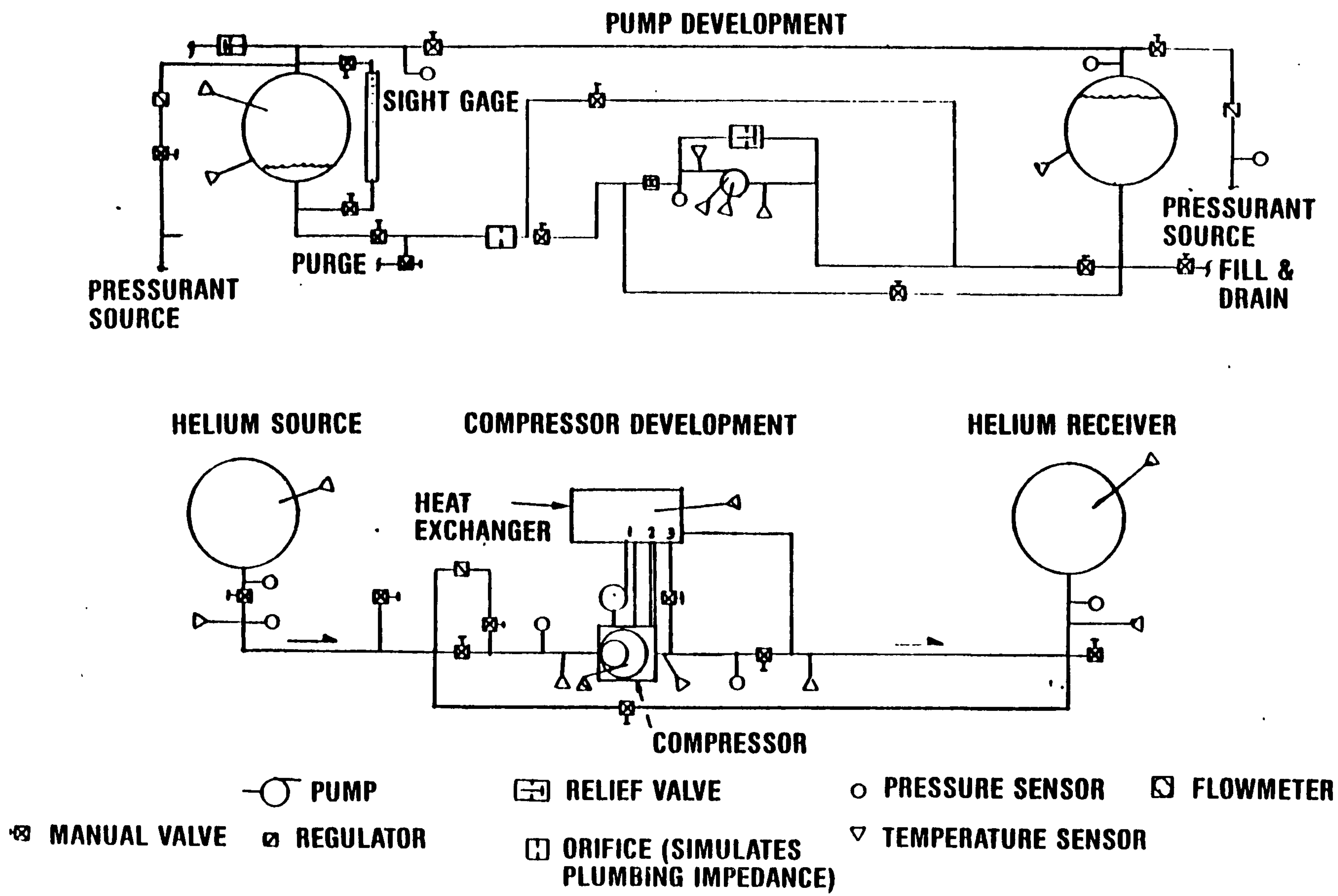


FIGURE 40 AND 41. COMPONENT TEST SCHEMATIC

6. Define any compressor design deficiencies and provide recommended changes to the flight compressor design.

Liquid/Gas Separator Tests

The liquid/gas separator testing will be performed at the component level and will be restricted to the verification that this unit will maintain its structural and mechanical integrity when subjected to launch environments.

Brassboard Ground Test Program

This test program will provide the final phase of the fluid transfer system development and will also provide the verification tool to establish the FDTV fluid transfer capability for the flight demonstration.

Test Vehicle Description

The Test Vehicle Description (T/V) shall be designed and fabricated to be installed in the space shuttle payload bay and shall be in two separate sections; the tanker side, which will be fixed to the orbiter, and the receiver side which will simulate a spacecraft being resupplied. The two sections shall be launched in the demated configuration. Once in orbit the mating of the two sections is performed by the RMS and the grapple and end effector (Figure 42). the T/V shall therefore be designed to perform satisfactorily after being exposed to launch and landing environments. The FDTV fluid schematics are presented in Figures 37 and 38 for the fuel (MMH) and oxidizer (N_2HO_4) systems respectively. the helium transfer systems for use with the compressor and cascade methods are also included. A preliminary FDTV design concept showing the major component layout is presented Figure 42. The test Vehicle (T/V) configuration shall be the same as for the flight demonstration except that the Grapple Fixture and End Effector need not be included for the conduct of the brassboard tests. The Test Vehicle will include seven major hardware and component items critical to the fluid transfer operation. These are; propellant tanks (4), helium pressurant tanks (7), fluid couplings (6), gas compressor (1), zero-gravity quantity gaging system (4) including four flowmeters as backup, propellant pump (3), and miscellaneous sinall tanks (4) to purge and safe the couplings.

A heat exchanger will be required to cool the compressor oil, compressor housing (requires circulating pump), and the compressed gas. The heat exchanger shall be designed to tie into one of the space shuttle payload bay cooling loops using Freon-113 as the fluid media.

This circuit of the compressor heat exchanger will also require a small circulating pump. The ground tests will include a flowmeter in this circuit in order to assess flow requirements.

In addition the FDTV will include a microprocessor for data processing, recording and display and an electrical panel consisting of switches, component status indicators, displays, etc, to enable control of the fluid transfer operations.

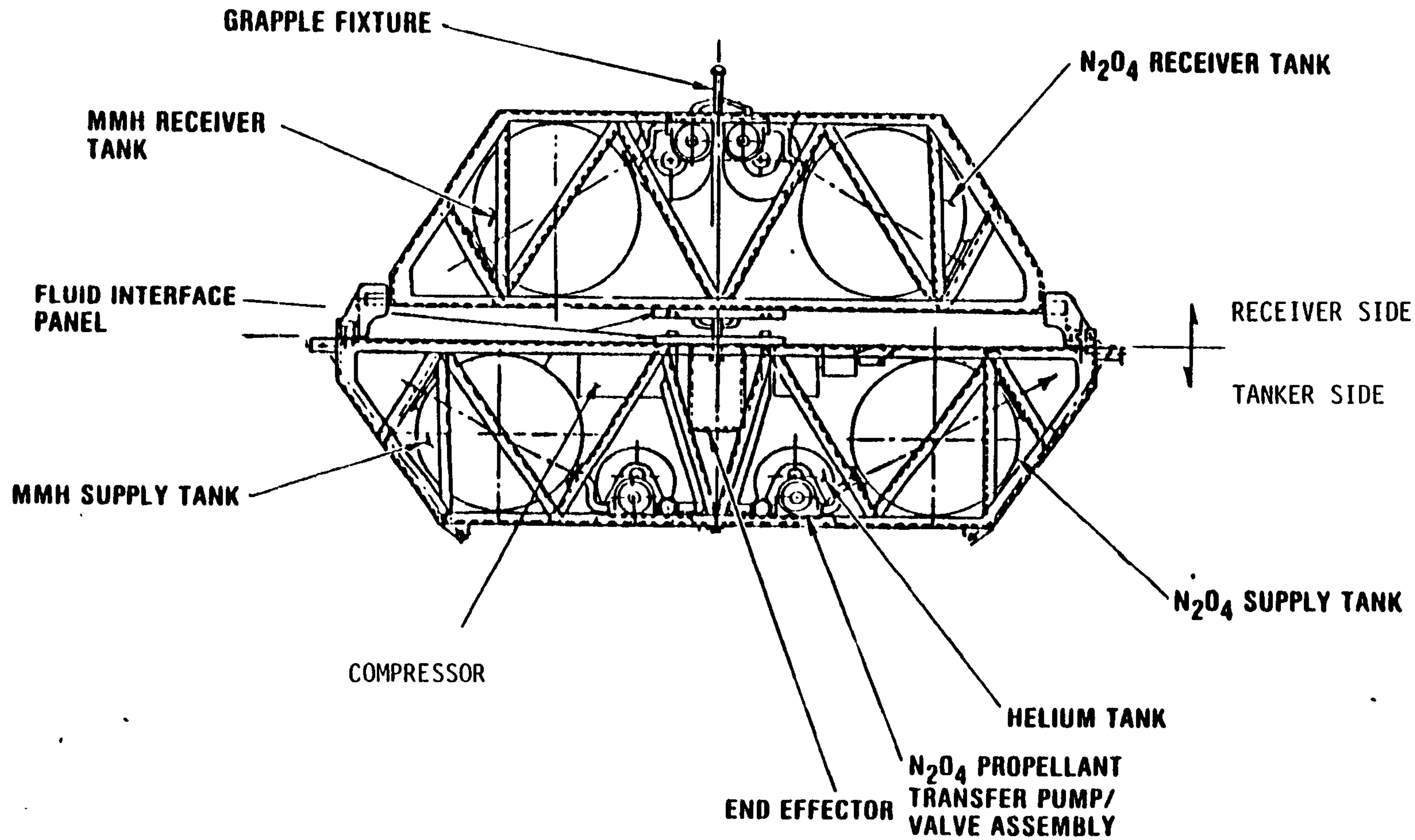


FIGURE 42. FLIGHT DEMONSTRATION TEST VEHICLE (PRELIMINARY DESIGN CONCEPT)

Test Program Description

The test program will consist of a series of tests directed at developing a flight test vehicle which will be utilized in a fluid transfer demonstration aboard the space shuttle payload bay. The results from these tests will develop and assess the fluid transfer system capabilities as well as provide the tool to define the flight demonstration procedures and regimes.

Final development of the gas compressor and propellant transfer pump will be accomplished during this testing phase. In addition, operation, verification and safing of the fluid couplings during mating and demating, together with the verification of the gaging and flowmeter system will be demonstrated.

The test program will include ambient and vacuum testing (zero-gravity tests will not be performed). All thermal sensitive areas such as gas compression during helium gas transfer and during ullage recompression propellant transfer will be diagnosed in a vacuum atmosphere. Except for the zero-gravity environment, the proposed flight demonstration timeline an procedures will be performed as verification and also as a tool for determining system predictions and behavior.

Brassboard Ground Tests

These tests shall be designed to meet the following objectives:

1. Propellant pump verification.

   1-A. Flowrate - Evaluate pump performance at low average, and high flowrates.

   1-B. Pressure conditions - Evaluate pump performance at various inlet, outlet and delta pressures.

   1-C. Temperature conditions - Evaluate pump performance at nominal and high temperature conditions (fluid and ambient).

2. Fluid coupling verification.

   2-A. Mating/demating - Define mating and demating loads alignment requirements, and procedures. Electrical connector mating and demating will also be evaluated and requirements established.

   2-B. Verification of Interfaces - coupling interface verification procedures shall be established and demonstrated for mating, fluid flow preparation, safing and demating operations.

3. Gas compressor verification.

   3-A. Fillrates - Determine compressor performance at low, average and high fillrates.

   3-B. Pressure Conditions - Determines compressor performance at various inlet, outlet, and delta pressure conditions.

3-C. Temperature Conditions - Determine compressor performance at various temperature conditions.

3-D. Compressor/gas heat exchanger - Determine heat exchanger performance (heat exchanger requirement established from breadboard tests).

4. Zero-gravity quantity gaging system verification - This gaging system shall be developed by NASA/JSC under a separate contract.

   4.A Installation - The gaging system shall be installed and calibrated to the FDTV propellant tanks.

   4.B Performance - The gaging system performance shall be verified during flow tests using flowmeters and PVT methods.

   4.C Zero gravity simulation - Gas ingestion and tank orientation tests will assess gaging sensitivity to varying fluid location.

5. Helium Tanks.

   5.A Tank servicing procedures shall be developed and verified.

   5.B The liquid/gas separator design and installation shall be analytically verified for flight application (i.e. launch environment effects and design concept).

6. Helium Tanks.

   6.A Heat exchanger - verify the performance of the helium tank heat exchanger. The need for the heat exchanger shall be determined from the breadboard tests.

7. Helium Transfer

   7.A Verify and demonstrate the optimum transfer configuration using the cascade method.

   7.B Determine transfer characteristics at various temperatures, flowrates and pressures using a gas compressor.

8. Propellant Transfer

   8.A Verify and demonstrate the optimum transfer procedures using the ullage transfer and the ullage recompression transfer techniques for various flowrates, temperature and pressure conditions.

   8.B Assess and establish the capability of the micro gravity quantity gaging system.

9. Flight experiment procedures.

   9.A Establish and define the flight demonstration gas and propellant transfer procedures and estimate the timeline required.

9.B Develop a semi-automatic control of the transfer operations utilizing a micro processor in conjunction with minimal astronaut involvement.

9.C Define orbiter interface and orbiter support requirements such as compressor cooling (heat exchanger) operations.

10. Test Vehicle flight worthiness

10.A Flight environment - FDTV flight worthiness shall be verified analytically with vibration/acoustic testing support performed only if analysis deems it necessary.

10.B Instrumentation & Components - all instrumentation and components shall be verified during testing, and calibration data together with component checkout data verified after completion of the transfer tests. These data shall be part of the FDTV data package.

10.C System leak checks - The system shall be verified to be leak free within the specified requirements prior to and after completion of the test program.

TASK 2 CONCLUSION:

The key results of this task analysis are as follows:

1) New subsystem design options have been incorporated into the conceptual propellant and pressurant resupply system defined in support of the S.P.E.R. Concept Definition Study (NASA8-35618). Design changes include 1) isolating pressurant tanks, 2) regulating compressor inlet pressure, 3) additional pump and compressor plumbing for E.R.M. self refueling, and compressor plumbing for E.R.M. self refueling, and 4) three separate disconnect/line purge designs.

2) An E.R.M. bi-propellant capacity of 7,000 lbs., tank volume approximately equal to 6 RCS propellant tanks, will be adequate for resupply needs through the year 2010.

3) Power requirements are specified for MMH, NTO and N_2H_4 at different flow rates and head rises or both gear and centrifugal pumps.

4) Recommend selection of the gear pump over the centrifugal pump for its versatility over a wide range of P's.

5) The tank volumes required to resupply a 13000 cubic inch receiver tank to 3600 psia using two different pressurant transfer methods and three different supply pressures were calculated and presented in Table 2.

6) The total transfer system weights were calculated from the determined system volumes and are presented Table 3.

7) The overall optimum transfer method and supply tank configuration was determined to be a cascade only transfer with four 4200 cubic inch supply tanks (16800 cubic inches total) at 6000 psia for a system weight of 277 lbs.

8) For a resupply only tanker, with a propellant load of 7000 lbs, the cascade method of pressurant resupply is preferred, but for the maximum system versatility a compressor is required.

9) The ERM will still require the following development tests:

   A) Mating and alignment for the fluid interface with latching.

   B) Propellant Transfer - Major test components - propellant pump, automated coupling, and tank acquisition device with ullage control.

   C) Pressurant Transfer - basic function, multiple transfers, connection reliability with leakage monitoring. Major test components - compressor and automated coupling.

10) The Middeck Ullage Transfer Experiment (MUTE) is designed to test the concept of ullage transfer and the associated PMD, and micro-gravity gaging system. These objectives were examined and a series of tests proposed to verify the ullage transfer process.

11) The Flight Demonstration Test Vehicle (FDTV) is designed to test several aspects of remote on-orbit fluid transfer. The fluids to be transferred will include NTO, MMH, and Helium. The proposed demonstration will test docking, QD mating, leak verification, purging, safing, ullage control and separation during a propellant transfer, cascade and compressor pressurant transfer, and thermal and remote system control.

12) The ERM ground test program will be divided into four major subtasks.

   A) Pump and compressor specifications, including supplier design and fabrication of breadboard test units.

   B) Component testing of pump and compressor breadboard units. Testing and/or evaluation of the liquid/gas separator together with the gaging system configuration requirements will be performed.

   C) FDTV design and fabrication. The test and checkout of the coupling mating assemblies to assure alignment and determine engagement and separation loads will be performed. Procedures for coupling, operation will be assessed.

   D) FDTV development and verification test program. Low level support using breadboard development test apparatus may be advantageous and cost effective to isolate unique component problem areas.

2.3 Programmatics (Task 3)

Summary

Rockwell's recommendations in the follow-on ERM Study for the advanced development and flight demonstration differ significantly from the initial study.

The significant difference between our original and current recommendations are presented in Table 20. In this study, Rockwell defined and weighed four criteria for selecting the best advanced development and flight demonstration program. These criteria and their weights were approved by the NASA Contracting Officer Representative at the mid-term review. Based on these criteria, Rockwell recommends the early flight demonstration program which culminates at the end of the advanced development program. We reviewed 12 different programs and from these selected the one that maximized the NASA's goals, as defined by the NASA approved criteria and weights we applied to the selection process.

Rockwell's recommended program includes two flight experiments as well as the advanced development of the critical components to enable remote resupply servicing. The first flight experiment (Mid-Deck Ullage Transfer Experiment - MUTE) is a low cost validation ($1.8 million) of ullage transfer in a micro-gravity environment. Not only will this experiment remove the uncertainty from the propellant management devices ability to position and control the ullage bubble, but it demonstrates to potential users NASA's resolve to develop remote resupply servicing. Stimulating user support for remote resupply servicing is one of NASA's goals, and hence it was a criteria in selecting our recommended program.

The second flight experiment (Flight Demonstration Test - FDT) is the validation of the advanced developed for remote resupply servicing. Rockwell's low cost approach is to "fly the brass board", the hardware developed and tested during the advanced development phase. The primary flight test objective of FDT is to verify remote propellant transfer in a micro-gravity environment.

Groundrules and Assumptions

- o Customer Goals:
  - * Least cost advanced development and flight demonstration program.
    - Budget target is $12M.
    - Relative importance of goal is 50-60%.
  - * Reduce technical risk and stimulate user support for the ERM program.
    - This is to be accomplished by an advanced development and flight demonstration programs.
    - Relative importance of goal is 40-30%.
  - * Earliest ERM program start date.

SIGNIFICANT DIFFERENCES	INITIAL ERM STUDY	FOLLOWING ERM STUDY
SCHEDULE OF FLIGHT DEMONSTRATION	LATE — AFTER ERM PROGRAM START	EARLY — BEFORE ERM PROGRAM START
NUMBER OF FLIGHT DEMONSTRATIONS	ONE DEMONSTRATION • PROPELLANT TRANSFER EXPERIMENT	TWO DEMONSTRATIONS • MID-DECK ULLAGE TRANSFER EXPERIMENT • FLIGHT DEMONSTRATION TEST
RELATIONSHIP BETWEEN ADVANCED DEVELOPMENT & FLIGHT DEMONSTRATION PROGRAMS	SEPARATE • ADVANCED DEVELOPMENT BEFORE PHASE C/D • FLIGHT DEMONSTRATION AFTER PHASE C/D START	INTEGRATED • BOTH ADVANCED DEVELOPMENT AND FLIGHT DEMONSTRATION COMPLETED BEFORE PHASE C/D START
COST	7$\overline{M}$ TECHNOLOGY DEVELOPMENT 5$\overline{M}$ FLIGHT DEMONSTRATION \$12$\overline{M}$ TOTAL	1.8$\overline{M}$ MUTE 10.7$\overline{M}$ FDTA \$12.5$\overline{M}$ TOTAL

TABLE 20. PROGRAMMATICS OVERVIEW

- Relative importance of goal is 10%.

- o All cost estimates are in constant 1985 dollars.
- o STS launch costs are not directly included in the cost estimate, but launch costs were considered in selecting the top-and-bottom truss experiment over the side-by-side truss experiment in the Orbiter cargo bay.
- o A nominal six month waiting period was assumed for Orbiter payload integration activity and this cost is not included except for direct contractor support.
- o NASA costs, such as astronaut training and test facilities at KSC for pre-flight propellant transfer testing are not included.

Rationale For Recommendation

Rockwell has based its selection for the advanced development and flight demonstration on the NASA's selection criteria and their relative weights. The most important criteria was cost (weight of 50 to 60%). All options were designed to fit within the scope of NASA's target of $12 million. The most costly programs were eliminated in the selecting process. The surviving options were within $1 million of one another. So, cost was eliminated as a criteria in the subsequent selection process. The next two most important criteria were highly correlated, so they were combined into a single composite criteria. Flight demonstration options of greater technical scope and challenge tended to reduce the technical risk of the subsequent ERM program. And because of their greater visibility and realism, technical challenging demonstrations were also more likely to increase user support for the NASA's remote resupply program. The least important criteria was an early ERM phase C/D start (weight of 10%). Thus, with cost eliminated, the dominate criteria was technical risk and user support.

A comparison between our recommendations in the initial ERM study and the follow-on study are shown in Table 21. Rockwell's current recommendation (the early flight demonstration option) greatly contributes to most important criteria reducing technical risk and soliciting user support (weight of 75-80% once the cost criteria is removed). The only disadvantage is a later ERM phase C/D start. However, due to our efforts to find a low cost flight demonstration, ("fly the brass board" discussed later), our program plan is shortened by six months. Thus, our low cost program cuts what would otherwise be a 12 months delay in the ERM phase C/D start to only a 6 months delay (in comparison to the late flight demonstration option originally recommended).

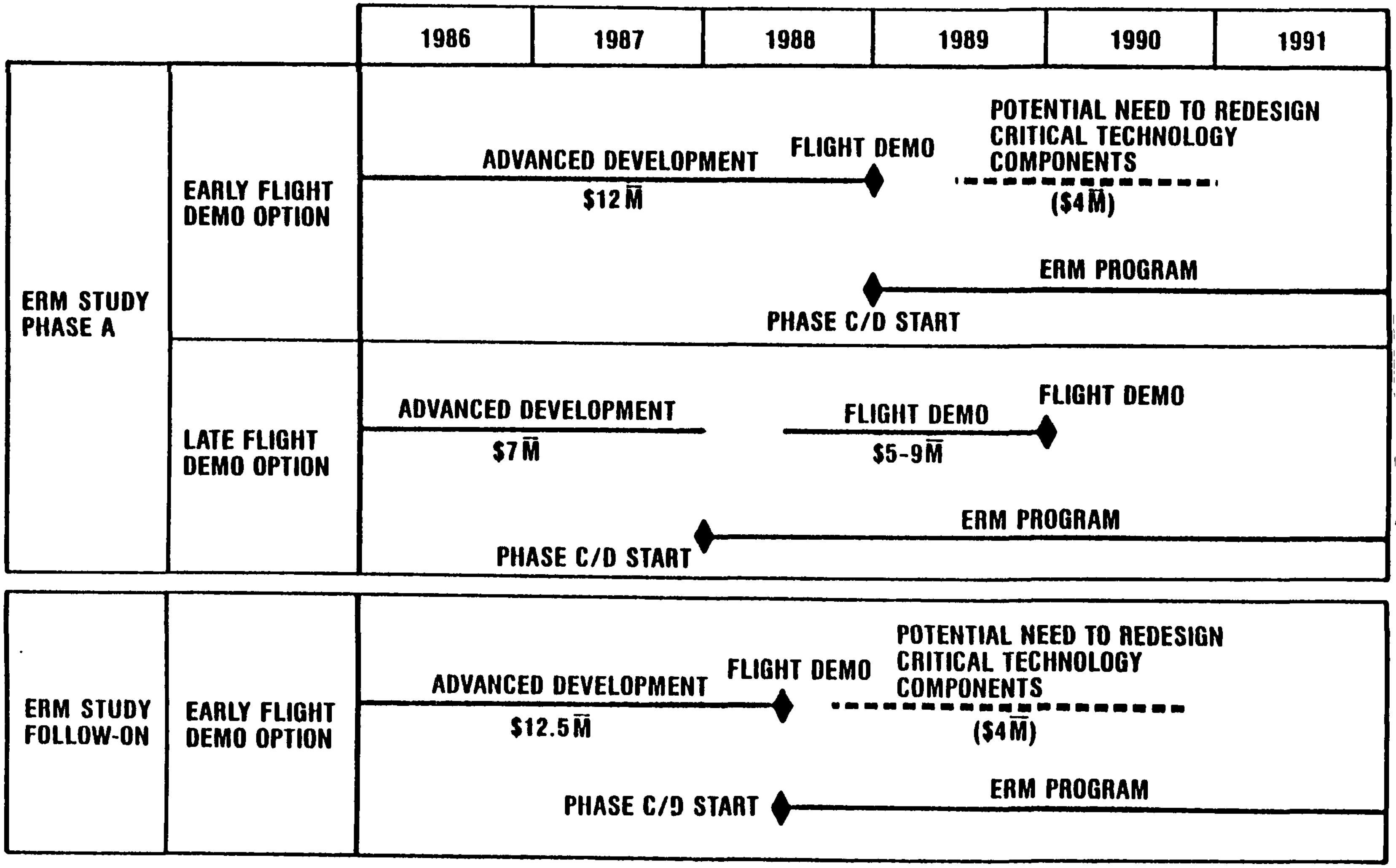
The schedules of the early and late flight demonstration options from the initial ERM study and the early flight demonstration option currently recommended from the follow-on ERM study are displayed in Figure 43.

For purposes of comparison, all program schedules have been laid on a common calendar scale. Schedule differences for the flight demonstration and ERM phase C/D start can be observed by comparing the milestones. The program with the earliest flight demonstration option is the one we currently recommend.

Early Flight Demonstration Option	Late flight Demonstration Option	Criteria	Importance* of Criteria
Technology Uncertainties removed before ERM program starts.	Parallel effort exposes ERM programs to risk of schedule delay and cost increases	Technical Risk	
			75-80%
Accelerated by 18 months	18 month delay	User Support	
6 months	6 month earlier ERM program start	ERM Program Start	25-20%

*Normalized from criteria approved by customer at mid-term briefing (cost 60-50%, Support & Risk 30-40%, Program Start 10%)

TABLE 21. THE EARLY FLIGHT DEMONSTRATION OPTION HAS SUPERIOR ADVANTAGES AND ONLY MINOR DISADVANTAGES



ROCKWELL RECOMMENDS THE EARLY FLIGHT DEMONSTRATION OPTION

FIGURE 43. COMPARISON OF ADVANCED DEVELOPMENT/FLIGHT DEMONSTRATION OPTIONS

It is 18 months earlier than the program we proposed in the initial ERM study. The ERM phase C/D start that we currently recommend is only six months later than the program we recommended originally.

In summary, significantly more benefits have been gained (less technical risk and 18 month earlier flight demonstration) than have been lost (six month later ERM phase C/D start).

Program Master Schedule

The master schedule for the program recommended by Rockwell, leading to the earliest flight demonstrations is presented in figure 44. The initial effort is to specify the hardware requirements for the ERM program. With the requirements for the ERM program defined, the design of the flight experiments and the development of the critical ERM components can start. Ideally, MUTE can fly on schedule in 22 to 24 months after the program starts, and the Flight Demonstration Test can fly in 8 to 12 months afterwards.

The timing in development of several related fluid transfer components could affect this schedule. The micro-gravity gaging system (NASA, JSC) would be an ideal component to include on the Flight Demonstration Test If the timing was coincidental, the micro-gravity gaging system could be flight-tested on the Flight Demonstration Test without the expense of an additional independent flight demonstration of its own. This probably will not happen unless the program is accelerated by six to nine months so that the flight hardware is ready by early to mid FY 1988. The function of the micro-gravity gaging system, if not available for the Flight Demonstration Test, could be performed by fluid gaging.

The Quick Disconnect (NASA, Langley) is essential to demonstrate remote resupply in the Flight Demonstration Experiment. It is currently scheduled to be available in mid-FY 1988, and its schedule may also need to be accelerated slightly.

Mid-Deck Ullage Transfer Experiment

The significant characteristics of the Mid-deck Ullage Transfer Experiment are presented in Table 22. The experiment was selected because it met the NASA's main criteria of low cost, reduction in technical risk, and potential to estimate potential user interest and support in remote resupply servicing.

The time required to design, fabricate and assemble, and test MUTE is twelve months. Ground test time is only scheduled for three to five months. Since this experiment is designed mainly to test the effects of a micro-gravity environment on the ullage transfer, only limited testing can be performed on the ground. The schedule and manpower estimate is presented in Figure 45.

The schedule includes an effort to insure that the test plan is human-engineered since the orbiter crew will be directly responsible for running the experiment. Crew training is planned during the nominal six-month period of payload integration activity at KSC.

Test Objective	Verify ullage transfer with water solution in a micro-gravity environment
Customer Benefit	Reduce technical risk in Propellant Transfer Experiment
	Early flight demonstration to stimulate potential user support low cost ($1.8M)
Test Schedule	21-24 months after start of advanced development program.

TABLE 22. MID-DECK ULLAGE TRANSFER EXPERIMENT SIGNIFICANT CHARACTERISTICS

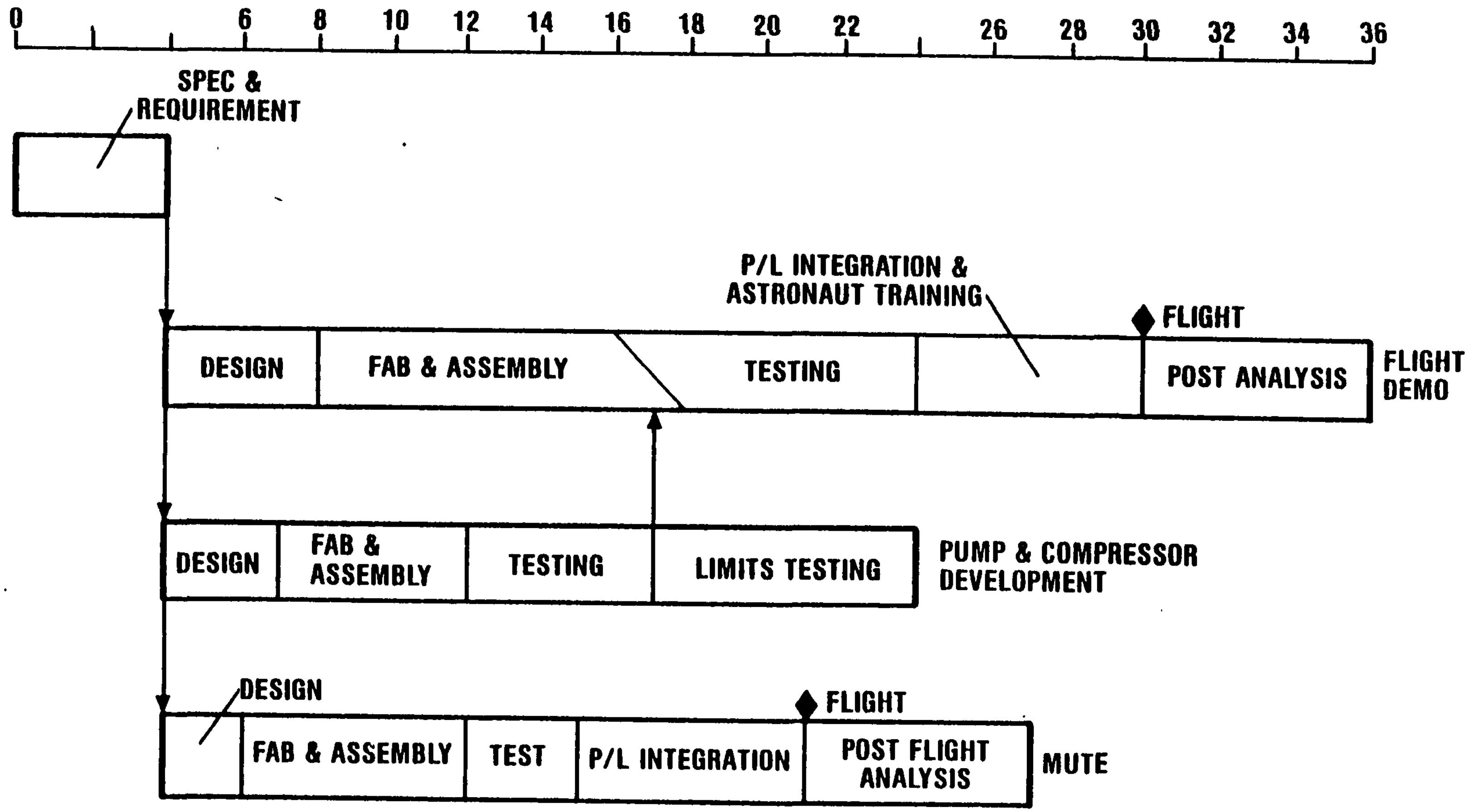


FIGURE 44. ADVANCED DEVELOPMENT AND FLIGHT DEMONSTRATION PROGRAM (MASTER SCHEDULE)

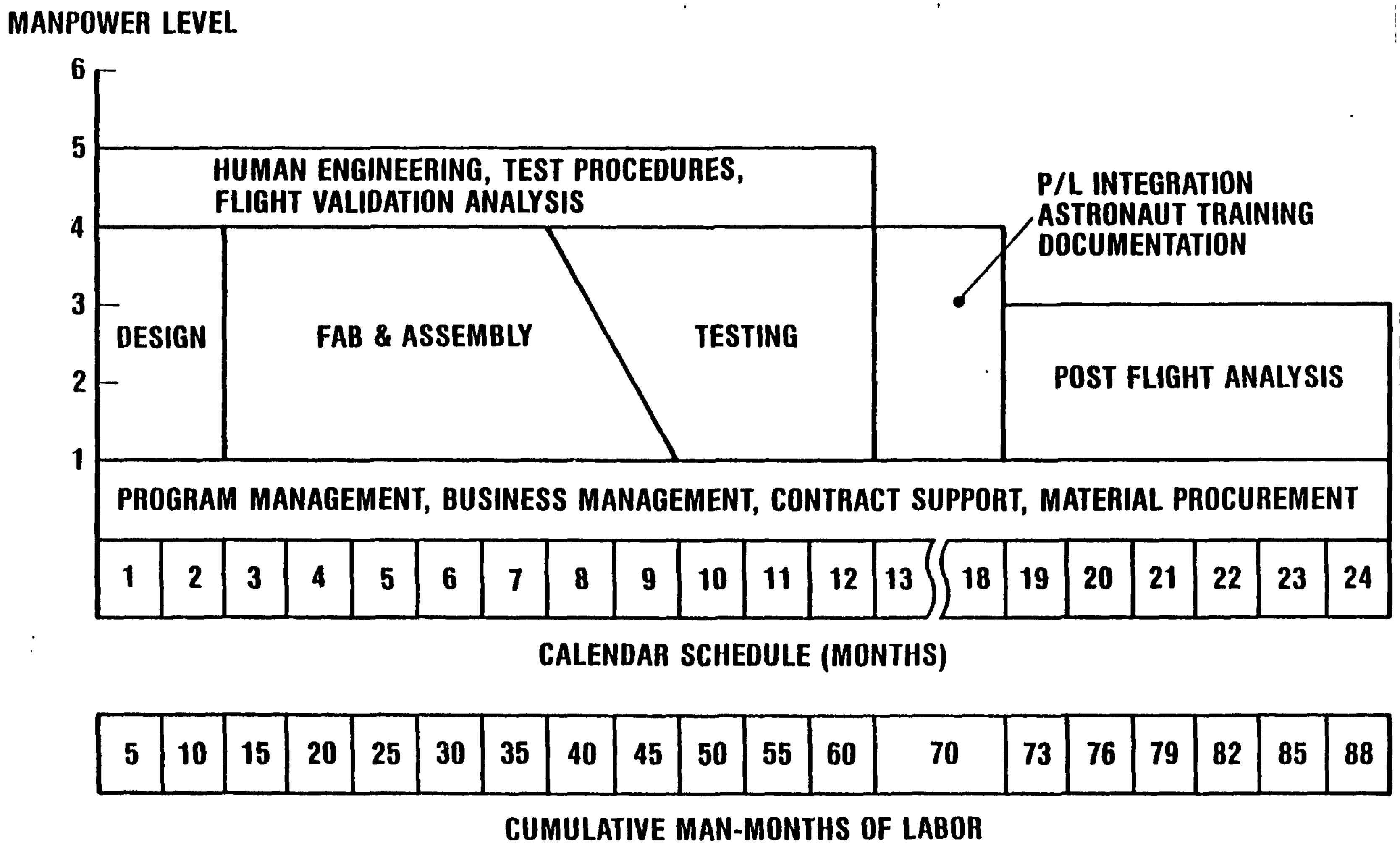


FIGURE 45. MID-DECK ULLAGE TRANSFER EXPERIMENT SCHEDULE AND MANPOWER ESTIMATE

In case of non-critical failures in the ullage transfer that might occur during the flight experiment, there are twelve months available to diagnose and correct them. The modifications would then be retested on the Flight Demonstration Test. Thus, the back-up to MUTE is the Flight Demonstration Test.

This schedule is based on a success-oriented program that assumes no major difficulties are encountered.

The grass-roots (detailed engineering) cost estimate is uncertain because the three key variables - direct labor hours, labor cost rates, and direct material - are uncertain. To provide a cost range, likely values were estimated for each variable as presented in Table 23. The best combination of cost conditions results, by definition, in the low cost estimate. The worst combination of cost conditions results, also by definition, in the high cost estimate. The combination of most likely conditions results in the most likely cost estimate.

These cost estimates only reflect the estimating error given the program plan as reflected in the MUTE schedule and manpower estimate. The program plan, and hence the cost estimate, reflects a success-oriented program with no major difficulties encountered.

The low cost estimate reflects that there is only a 10% chance that the cost of the experiment would be lower, and the high cost estimate reflects a 10% chance that the cost would be higher. The probability curve between these two extreme points was spread by a 35/65 Beta distribution function (ogive curve) and is displayed in Figure 46. The expected cost (50% cummulative probability that the actual cost could be less than or greater than this cost estimate) is $1.87M. The best estimate cost from Table 23 was $1.71 million. The true best estimate is somewhere in the vicinity of these two discrete estimate. If the midpoint is taken, then the best estimate for the MUTE is $1.8M (Figure 46).

In Rockwell's effort to meet NASA's primary goal of a low-cost advanced development and flight demonstration program, MUTE was designed to low cost. Some of the factors we have identified to control costs are listed as follows.

- Simple Experiment with Limited Test Objective
- Non-Rigorous Packaging Constraint - Design to Fit Mid-Deck Locker Space
- Minimum Electronic and Microprocessor Control - Maximize Orbiter Flight Crew "Hands-On" Effort
- Human Engineering Effort to Simplify Mechanics of Experiment
- Maximum Use of GFE (Existing Lockers and Video-Recorder Equipment)
- Minimize Data Recording by Visual Inspection and Video-Recording

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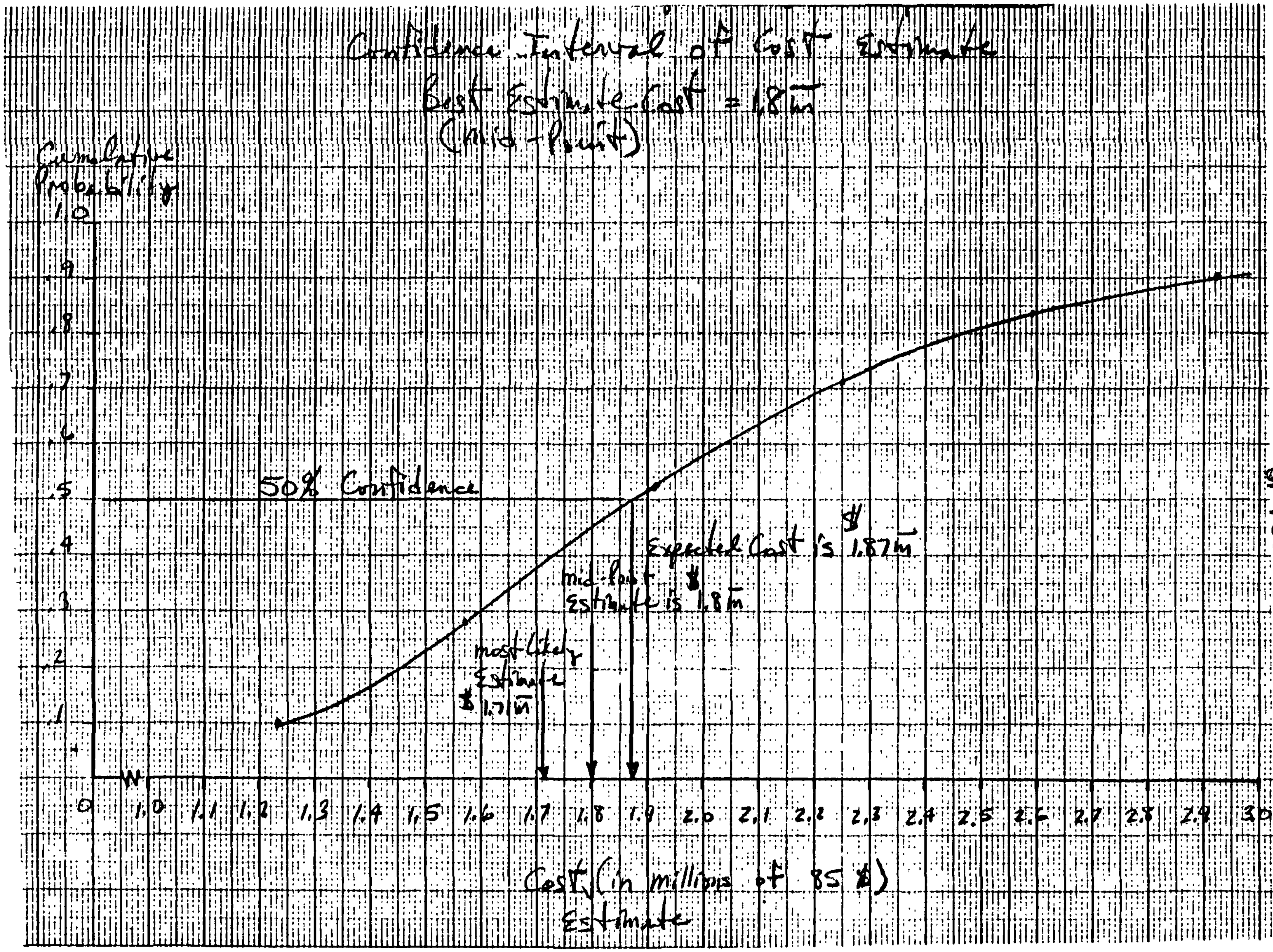

FIGURE 46. MID-DECK ULLAGE TRANSFER EXPERIMENT

TABLE 23. MID-DECK ULLAGE TRANSFER EXPERIMENT COST RANGE ESTIMATE

	Low Estimate	Most Likely Estimate	High Estimate
Labor (Less Program Management) Man Months	64	64 +20%	76
Labor Rate	$8250/mm	$9075/mm	$9900/mm
Total Direct Labor Cost	$528,000	$580,000	$752,400
Material (As % Total Direct Hrs)	40%	50%	60%
Material Cost	$484,000 (88 x 8250mm x 2/3)	$798,600 (88mm x 9075/mm x 1)	$1,574,100 (106mm x 9900/mm x 3/2)
Material Procurement @ 17% Material Cost	$82,280	$135,760	$267,580
Total Material Cost	$566,280	$934,360	$1,841,680
Subtotal	41.094M	$1.52M	$2.594M
Program Support @ 12.5% Subtotal (Incl. Auto Comp. Travel Documentation)	$.137M	$.189M	$.324M
TOTAL	$1.23M	$1.71M	$2.92M

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Besides a simple, low cost experiment to verify that the ullage transfer works as expected in a micro-gravity environment, MUTE also contributes to NASA's objectives to reduce technical risk and to support the marketing effort to gain user interest and support for remote resupply services.

Flight Demonstration Test

The significant characteristics of this experiment are presented in Table 24. The Flight Demonstration Test represents the cummulation of the advanced development. The cost of advanced development was previously estimated at $7 million. Thus, the marginal cost to "fly the brassboard" as Rockwell proposed, is an additional $3.7 million.

This flight demonstration meets the NASA's cost target. In addition, it reduces the technical risk by verifying remote fluid transfer in a micro-gravity environment. Furthermore, the second flight demonstration builds on the momentum of the first flight experiment (MUTE) to gain potential user interest and support for remote resupply services.

The schedules for the brassboard and the subcontracted effort for developing the pump and compressor are integrated. The schedule and manpower estimate is shown in Figure 47. The development effort by NASA for the remote quick disconnects and micro-gravity gaging system are assumed available during the brassboard assembly to support this schedule.

The component subcontractors are scheduled to deliver a prototype unit for the pump and compressor sixteen months after program start. The flight demonstration brassboard will test remote fluid transfer initially with a prototype of the critical ERM components and water solution fluids. Four to six months later, the flight test hardware will be delivered, integrated, and tested on the brassboard.

During the nominal six-month period for payload integration at KSC, we plan to perform fluid transfer testing with the actual hypergolic fluids planned for the flight demonstration.

The Flight Demonstration Test is scheduled for 30 - 36 months after program start, with a six-month post analysis of the results. Given a completely successful flight test, the ERM phase C/D program could start as early as 30 months from the start of advanced development and flight demonstration programs.

The grass-roots (detailed engineering) cost estimate is uncertain because the four key variables - direct labor hours, labor rates, direct material, and the subcontracted cost to develop the pump and compressor - are uncertain. To provide a cost range, likely values were estimated for each variable and presented in Table 25. The best and worst combinations of cost conditions results in the low and high cost estimates, respectively. The most likely combination of cost conditions results in the most likely cost estimate.

Test Objective	Verify remote propellant resupply with receiver and supplier hardware.
Customer Benefit	Reduce technical risk by verifying that all the critical ERM components (quick disconnects, pumps, compressor, and propellant management devices) work as designed in a micro-gravity environment. Early flight demonstration stimulates potential user support. Low cost ($10.7M) due to: o Integration of advanced development and flight demonstration program, and o Less test hardware with our plan to "fly the brassboard".
Test Schedule	30 - 36 months after start of advance development program.

TABLE 24. FLIGHT DEMONSTRATION TEST SIGNIFICANT CHARACTERISTICS

	Low Estimate	Most Likely Estimate	High Estimate
Labor (Less Program Support and Procurement)	224	224	297
Labor Rate	$8250/mm	$9075/mm	$9900/mm
Total Direct Labor Cost	$1.848M	$2.033M	$2.94M
Material (As % Total Direct Hrs Including Program Support and Procurement)	40%	50%	60%
Material Cost	$1.694M (308mm x 8250/mm x 2/3)	$2.795M (308mm x 9075/mm x 1)	$6.504M (438mm x 9900/mm x 3/2)
Material Procurement @ 17% Material Cost	$.288M	$.475M	$1.106M
Total Material Cost	$1.982M	$3.21M	$7.16M
Subtotal Effort			
Labor Man Months	144	144	192
Labor Rate	$8250/mm	$9075/mm	$9900/mm
Total Sub. Labor Cost	1.188M	$1.307M	$1.9M

TABLE 25. FLIGHT DEMONSTRATION TEST COST RANGE ESTIMATE

	Low Estimate	Most Likely Estimate	High Estimate
Material (% of Total Direct)	40%	50%	60%
Labor Cost	$.792M ($1.188M x 2/3)	$1.307M ($1.307M x 1)	$2.85M ($1.9M x 3/2)
Material Procurement @ 17% Labor Cost	$.135M	$.222M	$.485
Subtotal and Subcontractor Fee @ 8%	$2.28M	$3.06M	$5.24M
	Low Estimate	Most Likely Estimate	High Estimate
Subcontract Procurement and Contract Management @ 17% Subcontractor Cost	$.39M	$.52M	$.89M
Program Subtotal Program Support @ 12.5% (+Auto Comp. Travel, Documentation and Other)	$6.5M .81M	$8.88M $1.11M	$16.23M $2.03M
Total Program Cost	$7.3M	$10M	$18.3M

TABLE 25. FLIGHT DEMONSTRATION TEST COST RANGE ESTIMATE (Continued)

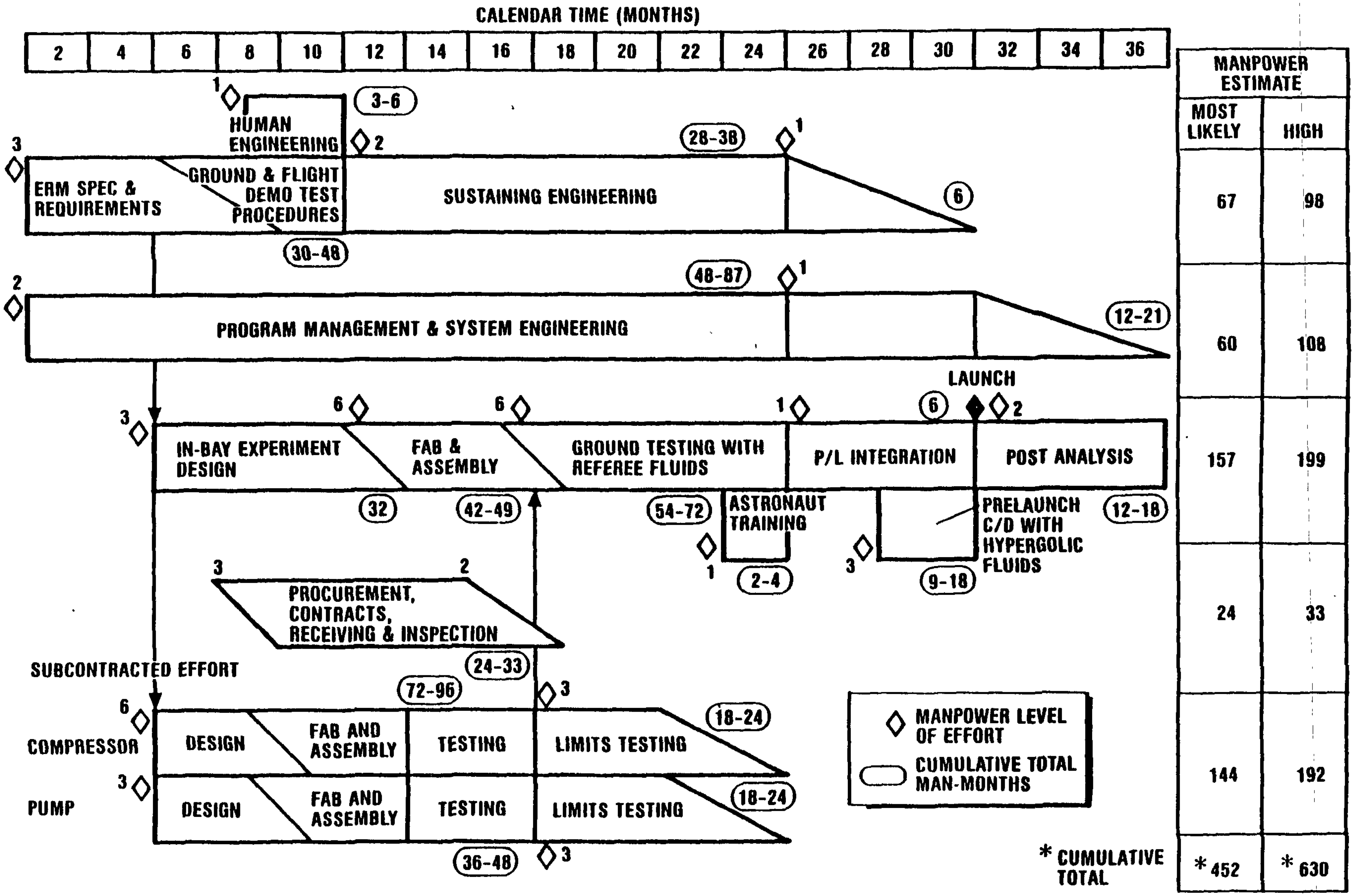

FIGURE 47. FLIGHT DEMONSTRATION TEST SCHEDULE AND MANPOWER ESTIMATE

These cost estimates only reflect the estimating error, given the program plan as reflected in the schedule and manpower estimates for the Flight Demonstration Test. The program plan however, and hence the cost estimate, reflects a success-oriented program with no significant difficulties encountered.

The low cost estimate reflects that there is only a 10% chance that the cost of the experiment could be lower and the high cost estimate reflects that there is only a 10% chance that the cost could be higher. The probability curve between these two extreme points was spread by a 35/65 Beta distribute in function (ogive curve) and is displayed in Figure 48. The expected cost (50% cummulative probability that the actual cost could be less than or greater than this cost estimate) is $11.5 million. The best estimate cost from Table 25 was $10.0 million. The true best estimate is somewhere in the vicinity of these two discrete estimates. If the mid-point is taken, then the best estimate for the Flight Demonstration Test is $10.7 million.

Given cost target of $12M for the Advanced Development and Flight Demonstration Program, Rockwell has designed a program that not only to meet this goal but has cut the cost of the Flight Demonstration Test sufficiently to fund an additional flight test (the MUTE). The significant cost reduction items are summarized as follows:

- Use Flight Demonstration Test Article as Systems Ground Test Article
  - Avoid Duplication
  - Reduces Early Demonstration Schedule by Six Months
- Borrow Orbiter RCS Propellant and Pressurant Tankage
- Less Structure than Side-by-Side Approach
  - Lower Weight
- Use Lower Cost Valves and Regulators than Orbiter
- Potential Use of MPESS for Lower "Half" of Structure

The three most significant cost avoidance items account for most of the cost reduction.

One, we propose to "fly the brassboard". Instead of two separate test articles, one for advanced development and one for the flight demonstration, will be developed we plan to develop the technology for remote fluid transfer, using the structure designed for the Flight Demonstration Test.

Two, we propose to "borrow" the RCS propellant and Helium pressurant tanks from the Orbiter logistical spares inventory instead of purchasing them outright as a separate cost item. This could avoid costs of up to two million dollars.

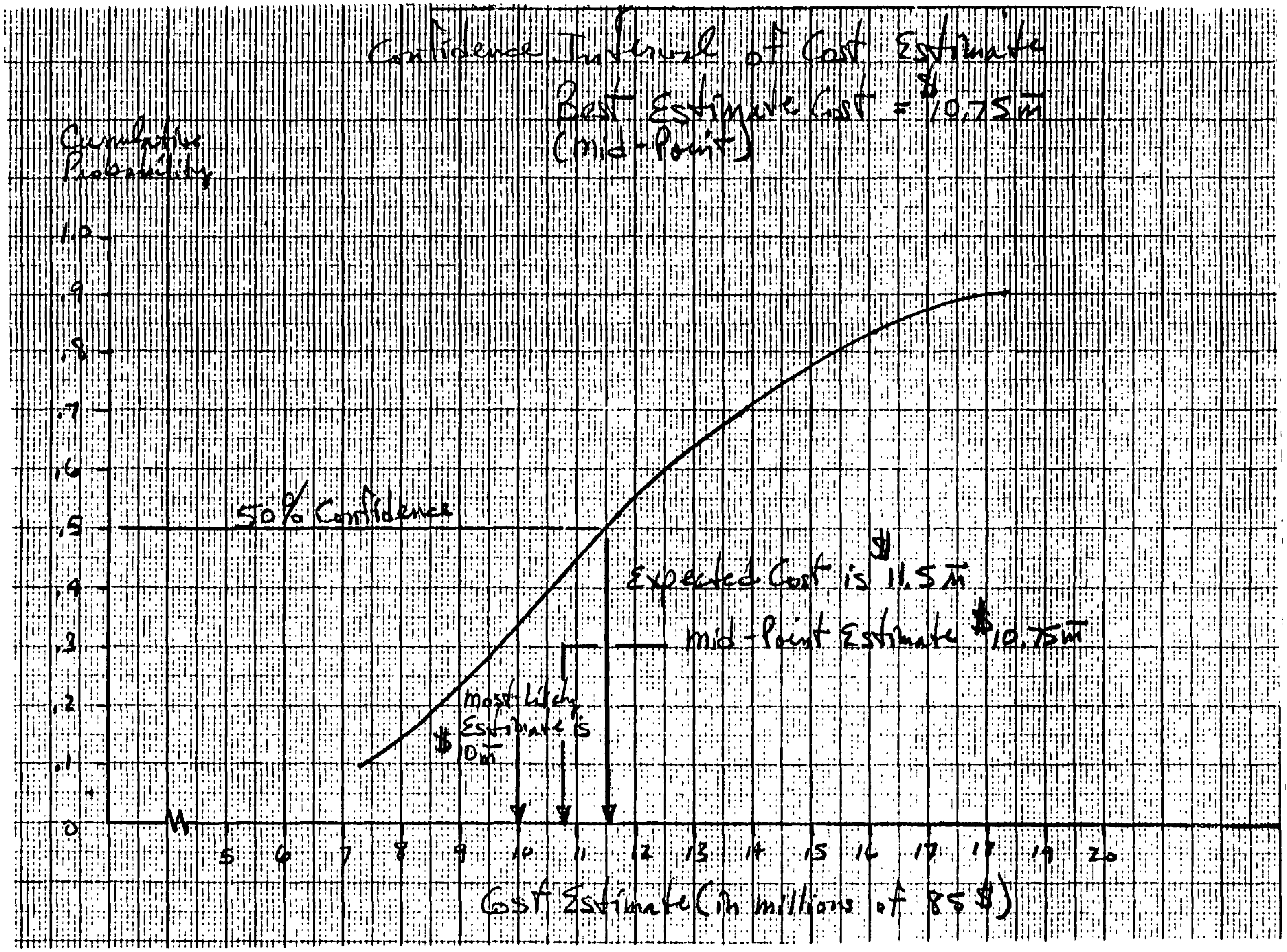

FIGURE 48. IN-BAY FLIGHT TRANSFER EXPERIMENT

Third, we propose a top-and-bottom test structure rather than our original side-by-side test structure. Not only will the weight and, hence, structure cost be less, but the flight transportation expense will decline because the space usage in the Orbiter cargo bay is cut in half. Presently, the width of our Flight Demonstration Test structure is only slightly wider than the RCS tank, or 39 inches.

The Flight Demonstration Test is a simple, low cost experiment to verify remote propellant transfer between a simulated supplier and a receiver spacecraft. This demonstration was selected because its contributes is to NASA's objectives to reduce technical risk and gain user interest and support for remote resupply services.

REFERENCES:

1. Pace, S. N., "Development Scenario and Requirements for Expendables Resupply Module," RI IL AE-MOA-85-178, June 3, 1985.

2. Saxton, H. A. Saxton, "Pump Transfer of Propellant, Task Analysis - Fluid Transfer Study. A Transmittal of Results for Subtask 2.6, IR&D Project 252, RI IL 287-201-85-111, August 22, 1985.

3. H. A. Saxton, "Pump Transfer of Propellant, Task Analysis - Fluid Transfer Study. A Transmittal of REsults for IR&D Project 283, November 21, 1984.

4. Compressor Transfer of Pressurants, Task Analysis - Storable Fluid System Conceptual Design, I.L. 287-201-84-143, November 20, 1984.

5. Development Scenario and Requirements for Expendable Resupply Module, I.L. AE-MOA-85-178, June 3, 1985.

6. Transmittal of the Rationale for Pressurant Resupply Process, I.L. 287-201-84-114, October 16, 1984.

7. Space Platforms Expendables Resupply Concept Definition Study, STS 85-0174, April 1985.

8. Phone conversation on July 9, 1985 with Norm Newhouse, Supervisor-Brunswick Corporation Product Design. (402) 464-8211.

9. "Establishment of Subtasks and Time Line for the Expendable Module Follow-On Study." RI IL #287-201-85-063, F. O. Chandler to J. J. Gernand, April 25, 1985.

10. "System Status Sensing and Measurement Techniques for On-Orbit, Propellant and Pressurant Transfer System - A Transmittal of Results for subtask 2.5 IR&E Project 286; H. A. Saxton to J. J. Gernand, July 27, 1984.

11. "Orbital Maneuvering Vehicle - Preliminary Definition Study", Marshall Space Flight Center, January 1985.

12. "Preliminary Design Review for Standardized Refueling Coupling", Fairchild Control Systems Company, April 4, 1985.

13. Sooter, C. W., "Preliminary Flight Demonstration Options and Their Relative Costs" IL AE-MOA-85-184, dated June 4, 1985.

14. Clement, J. L., "Cargo Systems Manual: SFMD", JSC-19484, July 1, 1984.

15. Sooter, C. W., "Preliminary Flight Demonstration Options and Their Relative Costs" IL AE-MOA-85-184, dated June 4, 1985.

16. Matus, S. J., "Expendables Resupply Brass Board Test System" IL 385-121-85-065, dated June 11, 1985.