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ORBITAL TRANSFER VEHICLE CONCEPT DEFINITION AND SYSTEM ANALYSIS STUDY

VOLUME II
OTV CONCEPT DEFINITION AND EVALUATION
BOOK 2
OTV CONCEPT DEFINITION

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FOREWORD

This final report, Volume II, Book 2 -- OTV Concept Definition, was prepared by Martin Marietta Denver Aerospace for NASA/MSFC in accordance with contract NAS8-36108. The study was conducted under the direction of NASA OTV Study Manager, Mr. Donald R. Saxton, during the period from July 1984 to October 1985. This final report is one of nine documents arranged as follows:

Volume I	Executive Summary	
Volume II	OTV Concept Definition and Evaluation	
	Book 1	Mission and System Requirements
	Book 2	OTV Concept Definition
	Book 3	Subsystem Trade Studies
	Book 4	Operations
Volume III	System and Program Trades	
Volume IV	Space Station Accommodations	
Volume V	Work Breakdown Structure and Dictionary	
Volume VI	Cost Estimates	
Volume VII	Integrated Technology Development Plan	
Volume VIII	Environmental Analyses	
Volume IX	Study Extension Results	

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Table of Contents

LIST OF FIGURES

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF TABLES

LIST OF TABLES (Continued)

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1.0 INTRODUCTION

This portion of the OTV Concept Definition and System Analysis Study, Volume II, Book 2, summarizes the flight vehicle concept selection process and results. It presents an overview of OTV mission and system design requirements and describes the family of OTV recommended, the reasons for this recommendation, and the associated Phase C/D Program.

Figure 1.0-1 depicts the overall process followed in developing the OTV concept definitions during this study. Design driver missions were selected and overall system design requirements were identified in Task 1. The results of this activity are summarized in this overview. The first step in the definition process was to do a parametric assessment of the reasonable propellant and staging options. This activity was supported by analyses conducted under Task 6 system trades. These results were coarse screened in accordance with criteria negotiated between MSFC and contractor personnel. Those concepts judged to have no possibility of being developed into a winner were not studied further.

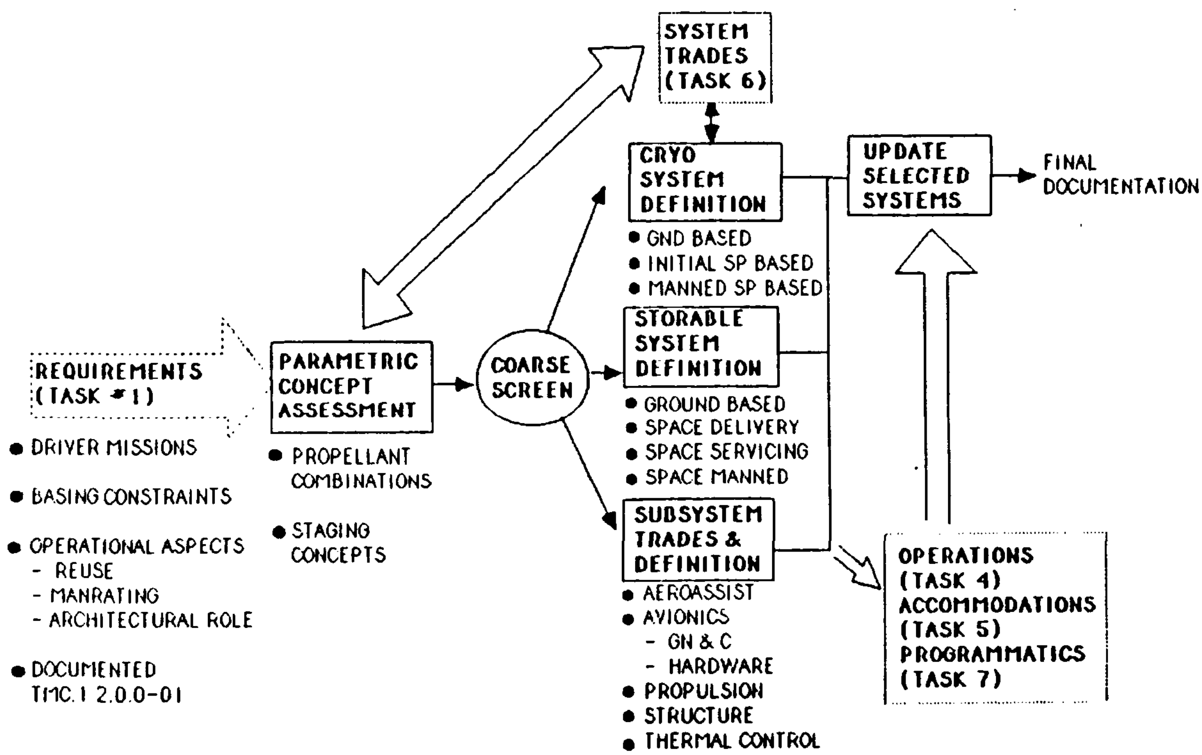

Figure 1.0-1 Vehicle Concept Definition Approach

The bulk of the vehicle concept definition activity was concentrated in the next step. An iterative process of defining the highest potential OTV concepts was conducted. Ground-based and space-based concepts were developed with a view to providing a reasonable evolution from one to the other. We maintained separate system definition activities for both storable and cryogenic options. The reason for this dual approach was that it is impossible to decide between the concepts at the vehicle definition level. At this level, cryos appear to be a clear winner. Storable advantages appear at the operations and space-basing levels, and a selection between them required awaiting the programmatic assessment of Task 4 and 5 evaluations. The system definition activity for storable and cryogenic concepts was supported by system level trades conducted in Task 6 as well as subsystem level trades in the basic vehicle design areas indicated. The subsystem level trades employed system sensitivities to support their specific selections.

Operations and space-based accommodations assessments of the high potential configurations were conducted in Tasks 4 and 5 and the results fed into the programmatics task to support major program decisions. These program decisions coupled with specific design recommendations from the operations and accommodations assessments were then incorporated into the final OTV concept definitions, and final documentation prepared.

The selection process involved a significant change in mission model at the midterm point of the contract. The first portion of the study concentrated on the selection and optimization of high potential vehicle concepts capable of meeting the requirements imposed by the "nominal" Revision 7 OTV Mission Model, and was completed at midterm. After midterm these concepts were evaluated from the launch and flight operations and space-based accommodations viewpoints, and a preferred program capable of meeting the requirements of the 'low' Revision 8 OTV Mission Model was selected. The change in mission models did not have a significant impact on the high potential configuration concepts. It did impact the selection of which of these configurations to include in the preferred program concept. The high potential concepts did not change in spite of a significant reduction in the driving manned mission payload weight because the concept driven by the lesser 20K delivery mission met the new reduced manned performance capability requirement. Changes in traffic levels and initial operational dates did have a significant impact on program selection. A basic MSFC direction was to make decisions that could be justified on the basis of low model traffic levels. While this direction did not change any fundamental decisions, it did make narrow the margin on some of the choices (for example, cryogenics over storables).

Results are presented in the following sequence. Section 2.0 presents a requirements overview. Sections 3.0 through 5.0 present the complete process of selecting the high potential Orbital Transfer Vehicles. The candidate concepts considered are identified, the major system trade results that discriminate between concepts are summarized, and the resulting high potential concepts in both cryogenic and storable categories are selected. These high potential concepts are described in detail in Sections 6.1 and 6.2. The storable concepts were not recommended for development, but they represent a significant data base and could prove desirable in certain mission scenarios. The final sections in this report (Sections 6.3 to 6.5) present the reasons for recommending the selected evolutionary cryogenic OTV program, and a description of the schedule and cost of this program.

2.0 REQUIREMENTS OVERVIEW

This section presents a summary of the mission and system requirements that were most influential in establishing the preferred OTV design concepts. Driving missions are reviewed, and the most significant system requirements are discussed. A more complete treatment of the mission and system requirements is documented in Volume II, Book 1: Mission and System Requirements.

2.1 DRIVING MISSIONS

The missions from the Revision 8 OTV mission model that drive the design of the flight vehicle are summarized in Figure 2.1-1. Drivers are categorized by operational era (pre and post Space Station) and model (low and nominal). The nature of the low model is particularly important, as it is to be used to justify major configuration decisions. As far as the driver missions are concerned, the only differences between the low and high models are deletion of the driving lunar mission and the less difficult planetary missions. The most important aspect of the low model is its lower traffic level, which tends to make it difficult to justify expenditure of development money. It is important to note that the Rev. 8 model is downgraded from the Rev. 7 model by the incorporation of the Mobile Geosynchronous Service Station (MGSS)concept. This concept reduces the geostationary roundtrip requirement from 14000 pounds to 7500 pounds. The following OTV concept development is keyed to find the best way to perform these driving missions.

	MISSION TYPE	LOW MODEL				NOMINAL MODEL			
		UP LB	DN LB	L-DN FT	MISSION DAYS	UP LB	DN LB	L-DN FT	MISSION DAYS
PRE-SPACE STATION	GEO DELIVERY	12000	2000	5	3	12000	2000	5	3
	PLANETARY DELIVERY	5000	@ $C_3 = 50$		6	40000	@ $C_3 = 9$		6
POST-SPACE STATION	GEO DELIVERY	20000	0	-	3	20000	0	-	3
	PLANETARY DELIVERY	3497	@ $C_3 = 98$		6	12000	@ $C_3 = 60$		6
	UNMANNED GEO SERVICING	7000	4500	9	12	7000	4500	9	12
	MANNED GEO SERVICING	7500	7500	10	20	7500	7500	10	20
	MANNED LUNAR SORTIE	-	-	-	-	80000	15000	12	15

* REQUIRES TWO LAUNCHES

Figure 2.1-1 Design Driver Missions

Prior to the time when Space Station is available, the mission model requires only payload delivery missions. The only retrieval requirement is for the OTV itself and, in the case of multiple GEO delivery missions, the multiple payload airborne support equipment. The merit of retrieving this equipment is, of course, subject to economic evaluation. The driving planetary mission in the pre-space-based portion of the low model is not a payload performance driver, but it does impose unique mission design problems. For example, retrieving OTV from planetary inject mission involves a retro maneuver and a return orbit perhaps as long as four days. During this period of time, the Orbiter's orbit precesses out of the OTV orbit plane and complex plane change maneuvers are required. The nominal model, pre Space Station, planetary driver mission does drive payload capability and will require multiple STS missions and onorbit assembly to implement it. This need is, by MSFC direction, not to drive the selection of OTV systems.

The low model in the post Space Station era introduces a number of new driving missions. The 20,000 pound delivery mission is the pacing payload requirement. The unmanned servicing mission is the first to require retrieval of a sizable payload. The manned GEO servicing mission increases this retrieval payload requirement, and introduces the problem of man-rating the vehicle. the nominal model introduces the very difficult, from a payload performance point of view, manned lunar sortie mission. The 80K lb up and 15K lb back requirement drives both propellant required and the retrieval weight designing the retrieval system (probably an aeroassist device). As in the case of the nominal model planetary mission, this lunar mission requirement is not to drive OTV system selection.

Figure 2.1-2 shows the time phasing associated with the major driving missions for the low and nominal mission models. The dry points are associated with the delay in introducing new capabilities associated with the low model. Availability of the space-base is delayed two years, the requirement for manned operation six years. The manned lunar sortie is extended out of the window under consideration in this study. Since major decisions are to be justified by the low mission model, this situation is critical to this study.

2.2 SYSTEM REQUIREMENTS

System requirements have been derived from these driving missions, and are compiled in their entirety in Volume II, Book 1 of this report. The following paragraphs present highlights of these system requirements and the considerations that led to their selection.

ARCHITECTURAL ROLE--The basic design philosophy followed for the OTV concepts developed in this study is illustrated in Figure 2.2-1. OTV provides the muscle to reach high earth orbit. Finesse for advanced operations at high orbit is to be provided by the payloads. In the case of unmanned servicing, a GEO OMV is carried aloft, and it has the six degree of freedom translation capability and servicing systems required to perform the servicing functions. Similarly, manned servicing functions are accomplished using an MGSS. For the interface between the Space Station and the OTV, departure is implemented by provision of a small delta-V to achieve safe separation distance by the hangar system. OTV retrieval from a safe separation distance is to be implemented with OMV maneuvers. In this way, the design of the OTV has been concentrated on providing efficient orbit transfer.

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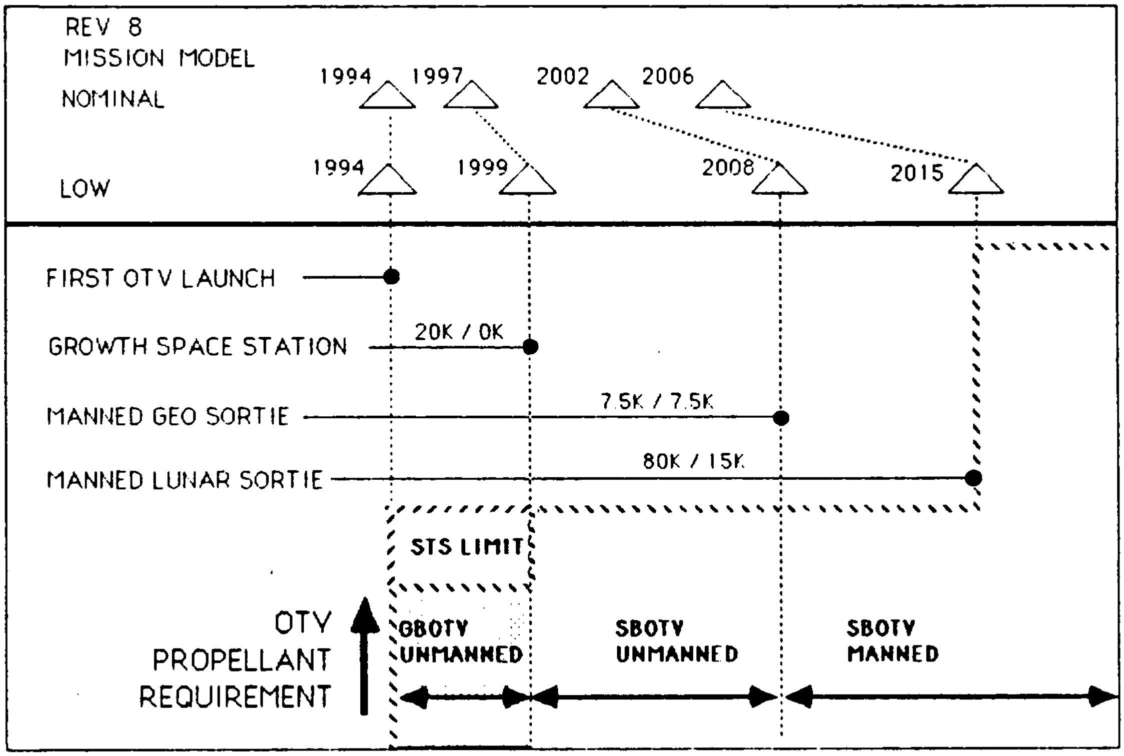

Figure 2.1-2 Time Phasing of Mission Requirements

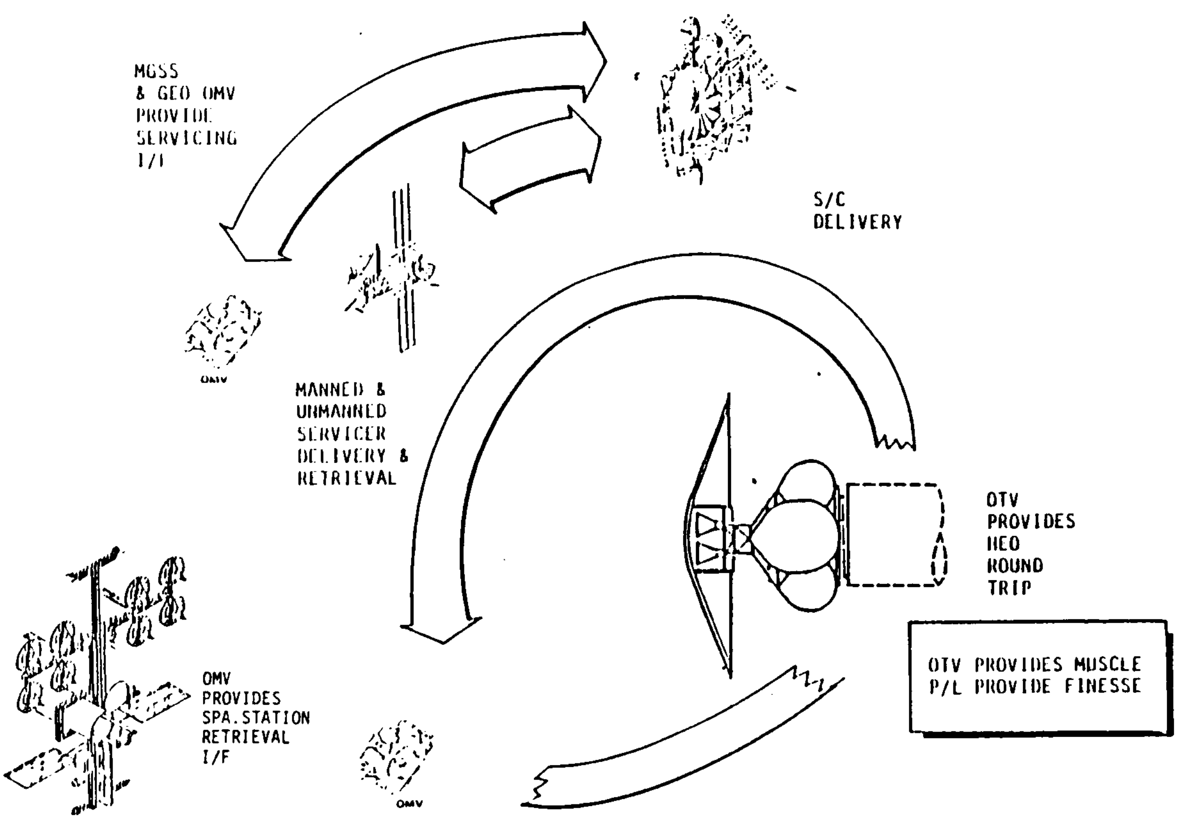

Figure 2.2-1 OTV's Architectural Role is HEO Truck

REUSE--The OTV defined herein eventually become, by study requirement, reusable. Retrieving the OTV for reuse decreases payload capability on a single ground-based launch, or decreases mission payload delivered per pound of propellant required at a space station. Thus the reusable OTV must be justified on the basis of the value of the retrieved stage, the value of retrieving mission hardware, or the value of manned operation in high earth orbit. Trades to date indicate reusability is justifiable, but that the winning margin can be increased by the use of aeroassisted OTV retrieval. This concept is incorporated in all of the high potential configurations selected.

AEROASSIST--The basic requirements imposed on the aeroassist device are that it survive the aeroheating environment imposed, and that it perform its maneuver accurately in the face of anticipated variations in upper atmosphere density, navigation sensor accuracy, et.al. The designs presented in this report reflect selection of ballistic coefficients that are compatible with projected heatshield materials and a lift to drag ratio that provides adequate maneuver control. Our studies have shown that the ballistic coefficient during reentry from GEO must be less than 10 pounds per square foot and that the L/D must be 0.116 or greater. In addition, the heat shield must be large enough to protect the stage and attached retrieved payload from the aerodynamic wake. All of the configurations presented in this report meet these requirements.

MAN RATING--Based on agreements reached at the First Quarterly Review, all manned OTV must at a minimum provide a fail-safe-return capability. This is interpreted to mean that the OTV must be able to return its crew safely to the vicinity of Space Station (or Orbiter, if ground based) after suffering one credible failure. Unmanned vehicles should use a degree of redundancy that is economically justifiable.

GROUND/SPACE TRANSITION--The ground-based OTV defined herein must, by study requirement, evolve to space-based operation. It is not a requirement that the ground-based vehicles be operable in the space-based mode. The changes between configurations should be a cost effective compromise between: Proof of system and subsystem concept in an initial ground-based mode; maximizing ground-based single launch capability; minimizing space-based propellant requirements; and providing efficient OTV packaging for delivery of the ground-based OTV to LEO as an assembled, loaded unit and the space-based OTV in sub-units readily assembled at space station.

UNIQUE GROUND-BASED REQUIREMENTS--This study considered two techniques for carrying a loaded ground-based OTV to low earth orbit (LEO). Effort was concentrated on ascent in an aft cargo carrier (ACC) attached to the external tank. These ACC OTV were packaged to fit the dedicated ACC as defined in Reference 1. If necessary, an additional 36 inches of length is possible within the ACC compartment. Cargo bay ascent of cryogenic OTV was explored in previous Phase A studies. Comparisons were made with these cryogenic cargo bay configurations after previous results were normalized with current subsystem design data, as described in Volume III. Storable cargo bay OTV were not previously studied, and special emphasis was required in this study. In this case, the requirements of Reference 2 were met in defining OTV general arrangement. In the payload bay application, it is important to minimize OTV length so its payload can be as long as possible. Return to earth in the cargo bay is the only retrieval option available.

ACC or cargo bay utilization has a marked effect on the design requirements the OTV must meet, as summarized in Figure 2.2-2. The net weight available for the flight ready OTV and its payload is shown assuming reasonable estimates for required ASE and orbiter fuel required to support necessary rendezvous maneuvers. Note that the ACC operations scenario selected requires an additional rendezvous between the orbiter and OTV during the ascent phase of the mission. In both cases JSC's projected orbiter capability (Reference 3) and a 140 nautical mile, 28.5 degree inclined ascent injection orbit have been assumed. Net weight available for ACC and cargo bay OTV and their payloads is nearly identical. Loaded OTV adaptation to the ACC structure requires less structural ASE than adaptation to the cargo bay longerons and keel and provision of propellant dump capability. This essentially compensates for the ACC structural weight penalty. The total volume available is greater for the ACC configuration. The ascent envelope available in the ACC favors short, large diameter OTV configurations and blunt mechanically deployed aeroassist configurations. The cargo bay envelope demands long configurations that are under 15 feet in diameter, and favors the use of the inflatable ballute aeroassist approach. The ACC configuration poses two unique operational problems for the ground-based OTV -- onorbit rendezvous and mating of a payload carried aloft in the cargo bay with the OTV, and partial disassembly of the OTV for retrieval in the cargo bay. Special ASE is required for ACC/OTV retrieval, and it must be launched in the cargo bay. 900 pounds of OTV support equipment is included in the 2100 pounds shown in Figure 2.2-2. This figure reflects an OTV designed to ease cargo bay mounting, but it is recognized that this figure could grow to the point where net LEO STS capability could favor the cargo bay vehicle. In both cases, stowage of a used aeroassist device for return to the ground and reuse appears difficult to accomplish. Another difference between the ascent locations is that abort dump provisions are required for the cargo bay, and higher structural margins are required for the safety of the orbiter crew. The ACC location does result in loss of the OTV in the event of an STS abort during ascent that requires return the launch site, while the cargo bay OTV is recovered in the same situation. Growth of the ground-based OTV capability to accomplish the advanced missions in the mission model requires multiple launches to achieve the required lift capability for both ascent locations, although volume constraints, when encountered, are eased with the use of the ACC.

DESIGN CONSIDERATIONS	ACC ASSESSMENT	CARGO BAY ASSESSMENT
PERMISSABLE OTV GLOW	o 67279 LB TO 140 NMI o REFLECTS 2100 LB ASE	o 67000 LB TO 140 NMI o REFLECTS 5000 LB ASE
VOLUME CONSTRAINTS	o FAVORS SHORT, MULTITANK CONFIG. o MORE P/L VOLUME	o CYRO REQUIRES TANDEM TANKS o P/L VOLUME CONSTRAINED
RETRIEVAL	o CYRO REQUIRES LH2 TANK REMOVAL o UNIQUE RETRIEVAL ASE	o ASCENT PROVISIONS o SUPPORT RETRIEVAL
SAFETY	o EVACUATE LH2 FOR DISASSEMBLY o STRUCTURE FACTOR 1.25	o PROPELLANT DUMP REQUIRED o STRUCTURE FACTOR 1.4
AEROSHIELD	o EXPENDABLE DESIGN o FLEX SHIELD	o FLEX SHIELD FOR STORABLE o BALLUTE FOR CYRO
OPERATIONS	o COMPLEX OTV/STS ASCENTOPNS o ON ORBIT P/L ATTACHMENT o DISSASSY FOR RETRIEVAL	o OPNS LESS COMPLEX o SHORTER MISSION TIMELINE
GROWTH MISSIONS	o MULTIPLE MISSIONS FOR LIFT CAPACITY	o MULTIPLES FOR LIFT o VOLUME LIMITED
ASCENT ABORT	o OTV LOST	o OTV RECOVERED

Figure 2.2-2 Ground-Basing Imposes Unique Design Constraints

UNIQUE SPACE-BASED REQUIREMENTS-The general layout of the space based OTV and concepts for its operation at a spacebase were evolved together. The limited availability of crewmen for EVA activity at the Space Station made it a derived requirement that a design that would be space maintainable/ serviceable using automation be evolved. While the space-based OTV need not be delivered as an operable unit, it is necessary that deliverable sub-units be efficiently packaged for delivery to Space Station by STS, readily assembled in space, and returnable in the Orbiter cargo bay. It is also a derived requirement that a capability be provided to load and offload OTV propellant in the micro-g environment of the space station. It is a requirement that the space-based OTV propellant tanks be protected against the meteoroid environment defined in Reference 4. In the contract extension activity reported in Volume IX, the added impact of LEO debris was assessed. Cost effectiveness considering system weight and life cycle maintenance as well as achieving single manned mission probability of no penetration in the order of 0.999 provide acceptable shield selection criteria. Figure 2.2-3 summarizes some of the impacts on the design of the OTV that result from the use of the space-based mode. Our data indicates that more efficient OTV design results from space-basing, particularly for the advanced missions that cannot be accomplished in a single shuttle launch. Counter to this, much of the required operations technology is new and is weighed from a programmatic point of view in our trades. Further impacts of a large OTV exist in required space station accommodations -- larger hangars and propellant farms. Our cost analyses reflect accommodations capable of supporting OTV's large enough to support the low Revision 8 mission model.

DESIGN CONSIDERATIONS	FAVORABLE IMPACTS	UNFAVORABLE IMPACTS
GROSS LAUNCH WEIGHT	NOT CONSTRAINED BY SINGLE LAUNCH CAPABILITY	
SIZE	DESIGN OPTIONS ARE RELATIVELY UNCONSTRAINED	UNIT PACKAGING MUST REFLECT ASCENT VEHICLE CONSTRAINTS
EARTH/LEO TRANSPORT	ONLY DELIVERABLE PARTS TO CB/ACC VOLUME LIMITS	ON-ORBIT ASSY & C/O IS NEW PROBLEM
SAFETY	ESCAPES STRESSFUL LAUNCH-LOADED ENVIRONMENT	METEROID SHIELDING REQ'D NEW REFLT INSPECTION ENV'M'T
AEROSHIELD	ELIMINATION OF REFURL REQUIREMENT ENABLES REUSE	
OPERATIONS	AUTOMATED LAUNCH & MAINT CENTRALIZED FACILITY FACILITATES ON-ORBIT ASSY	REMOTE OPNS IS NEW TECH, REQ. BETTER ACCESS, COMPLEX HANDLING FIXTURES, ET. AL.
GROWTH MISSIONS	NOT LIMITED BY BASIC STS CAPABILITY	

Figure 2.2-3 Space-Basing Favors More Efficient OTV Design

COMMUNICATION INTERFACES--The OTV will receive command updates and navigation information and downlink telemetry during its free flight mission. As a consequence, interfaces between OTV and TDRSS and GPS must be implemented.

PERFORMANCE MARGINS--The OTV's developed in this study reflect the following reserves and margins in estimating payload performance capability:

a. Flight Performance Reserve equivalent to 2% mission delta-V requirements

b. 15% on estimated dry weights

c. 10% on estimated ACS propellant requirements

d. 20% on estimated fuel cell reactant requirements.

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3.0 CONCEPT IDENTIFICATION

The major system candidates that were investigated are summarized in the following paragraphs. An overview of the configuration trades that discriminated between these candidates is presented in Section 4.0, immediately following.

PROPELLANT CANDIDATES--The primary propellant selection issue addressed in this study is the selection between storable propellants and cryogenic propellants. The most appropriate propellants within these categories were also addressed at a subordinate level. Figure 3.0-1 shows the range of propellants that are potentially applicable to the OTV. Of the several high energy propellant combinations, only oxygen/hydrogen is a viable candidate. Only candidates using fluorine are competitive or superior in performance, and the operational problems associated with flourine are not considered acceptable. Similarly, the only viable room temperature storable propellant combination is nitrogen tetroxide/monomethyl hydrazine. The other alternatives are not sufficiently superior in performance to overcome the N204/MMH advantage of being already operationally established in the STS program. Another group of propellants was considered -- the space storable options that involve the use of mild cryogenics. Of these options, liquid oxygen in combination with methane, propane or monomethyl hydrazine provides the leading contenders, and the hydrazine option is considered best. Its performance is representative of the class and it has the advantage that both propellants are operationally established in the STS program. These three propellant combinations (LO_2/LH_2, LO_2/MMH, and N_2O_4/MMH) were considered further in this study activity.

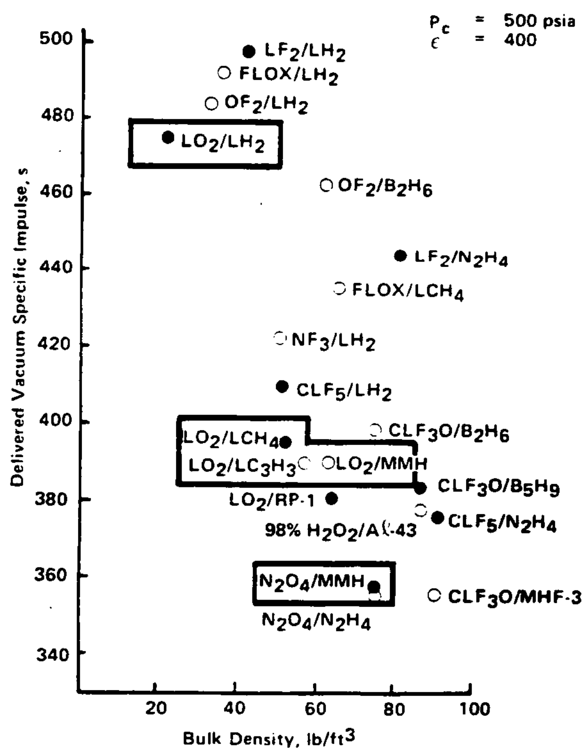

Figure 3.0-1 Candidate Propellant Performance

STAGING OPTIONS—The first step of our concept definition activity, coarse screening, discriminated between the more promising staging concepts, and eliminated those that did not merit the investment of significant study resources. Since different staging options are best for different propellant combinations, separate evaluations were conducted for each propellant. The following four staging options comprise the important options investigated:

- -SINGLE STAGE
- -TWO STAGE
- -PERIGEE STAGE (EXPENDABLE APOGEE STAGE)
- -1 1/2 STAGE (EXPENDABLE DROP TANKS)

Single stage represents the simplest and most operationally desirable reusable approach, providing that its performance is competitive. Two stage is the next most complex reusable solution, one that must be considered for lower performance propellants and very demanding missions. The next alternative requiring consideration is incorporation of a degree of expendability. Two approaches that were considered are expending drop tanks and expending an upper "apogee" stage. We are confident that the best staging solution is encompassed by these candidates. For ground-based vehicles the impact of configuring for both the payload bay and the ACC was considered. While reusable concepts were the primary goal of the study, a comparison with expendable designs was conducted. Resolution of the staging issue is described in section 4.0.

REUSABLE CONFIGURATION OPTIONS -- Three basic reuse issues were considered: Whether aeroassist is superior to expendable and all propulsive reusable approaches; whether zero, low L/D (0.25), or medium L/D (0.25 L/D 0.75) aeroassist devices are superior; and whether the aeroshield structure should be inflatable, foldable or rigid. Since vehicle configuration, control methodology and aeroassist option are closely interrelated, this evaluation become a complex systems issue, Figure 3.0-2 presents a carpet plot of the payload capability of single STS launched GEO delivery OTV missions flown in expendable, aeroassist retrieved and all-propulsive retrieved modes. The greatest payload capability is achieved in the expendable mode. While manned missions demand a retrieval capability, retrieving the OTV for reuse on delivery missions can only be justified by the value of the reflown OTV being greater than the penalty associated with less efficient use of the shuttle flight. Figure 3.0-2 shows the payload capability of the aeroassist approach is superior to the all propulsive approach if the weight of the aeroassist device can be kept sufficiently low. We have, therefore, emphasized the development of a lightweight aeroassist device in our subsystem design efforts.

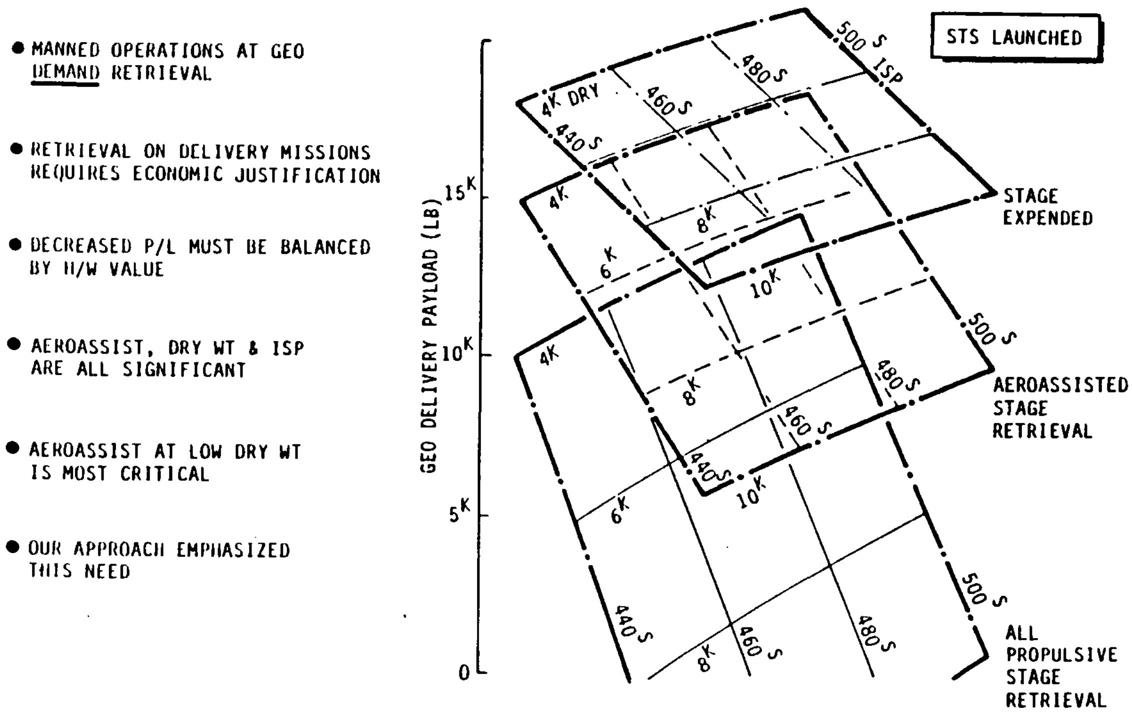

Figure 3.0-2 Efficient Aeroassist in a Key Design Objective

Aeroassist options are highly interrelated with the vehicle general arrangements under consideration, which are in turn interrelated with the ascent to orbit technique (e.g. ACC or cargo bay launch, launch assembled for ground-based operations or modular delivery for space-based operations). The general arrangements deal primarily with how propellant tanks will be arranged, where the engines and aeroassist device will be mounted, and where payloads will be attached. The basic stage arrangement concepts investigated in this study were: (1) Axial with tandem tanks; (2) Axial with cluster tanks; and (3) Transverse with cluster tanks. In case (1) the aerodynamic forces and thrust forces are along the same axis, propellant tanks are mounted in tandem along this axis, and the payload is also mounted along this axis. The only deviation in case (2) is that the propellant tanks are multiple and mounted side by side around the thrust/aerodynamic axis. In case (3), the aerodynamic and thrust axes are transverse to each other, and the tanks are clustered about the aerodynamic axis. Variations within these categories are possible, and some were investigated.

Figure 3.0-3 summarizes the most credible aeroassisted configuration options that were considered in this study. The deployable fabric aerobrake shown in the figure functions well with the four tank configuration using axial thrust. It provides excellent aerodynamic stability and good wake protection for a retrieved payload. The raked ellipse suggested by JSC is a more sophisticated aerodynamic shape that can only be implemented by a somewhat heavier rigid tile system. This configuration avoids the complexity of penetrating the heat shield for the main engine by arranging the thrust axis transversely. Payload is mounted along the aerodynamic axis, inducing significant CG travel complexity. The aeromaneuvering hypersonic sled offers higher L/D with attendant increase in aerosheild weight. Our studies show this approach is not justified by the benefit of aerodynamic turning of GEO retrieval missions. More ambitious mission requirements could justify this approach. The inflatable ballute appears to have considerable merit for configurations launched assembled in the cargo bay. These tandem tank concepts integrate well with the ballute, but tend to require large ballutes to achieve aerodynamic stability, particularly when retrieving long, heavy payloads. We investigated mechanical drag modulation for aeroassist control, as opposed to roll control of low L/D configurations. We found that the weight and complexity of this approach were unacceptable. The aerospike drag modulation concept is illustrated with our ACC configured folding brake. We found the performance advantage of this approach inconclusive and the technical uncertainty beyond our current capability to assess. Further evaluation of these approaches is documented in Volume II, Book 3, Subsystem Trade Studies.

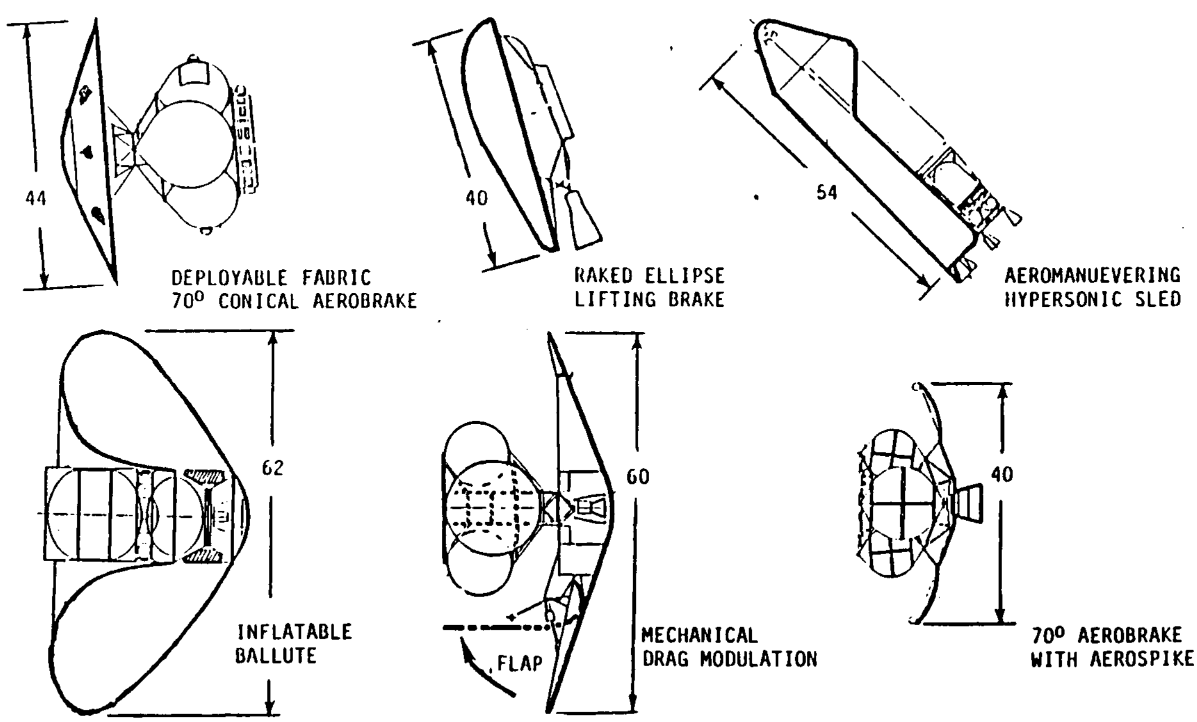

Figure 3.0-3 Configuration Options

MAIN ENGINE CANDIDATES--The main engine issue is a major driver in the development cost of the OTV, and its impact was considered at the system level. The engine options are shown in Figure 3.0-4. The current technology in OTV class engines is the RL10A-3A/B and near term technology is RL10 derivatives. They represent low risk and proven reliability. Advanced engine technology is currently being funded by NASA Lewis with contracts at Aerojet, Pratt & Whitney, and Rocketdyne. Additional work is funded by the Air Force Rocket Propulsion Laboratory (AFPRL) for 500 lb thrust class engines. We visited the three engine contractors to understand the cost and performance issues of derivative and advanced engines. Advanced engine performance and cost could be reduced to obtain a third option: a small 7500 lbf engine that can meet the mission model and evolve into a more advanced engine if future funding and mission constraints dictate.

CRYOGENIC - P&W RL-10 DERIVATIVES
- o CURRENTLY COMMITTED TO STS & CELV CENTAUR
- o PIP UPDATES ON GOING
- o CHOICE OF LOW RISK OPTIONS - TO 470+ SEC
- o HIGH PROVEN RELIABILITY

\- ADVANCED CRYOGENIC ENGINE
- o NASA LERC TECHNOLOGY PROGRAM IN PROGRESS AT ALRC. ROCKETDYNE. P&W
- o USAF - RPL - XLR-134 - 500 LBS

\- IOC CRYOGENIC ENGINE
- o INITIAL GROUND-BASED - 5 HR LIFE ENGINE TO INITIATE OTV PROGRAM
- o CLEAR EVOLUTION TO ADVANCED CYROGENIC ENGINE & MAN-RATING

STORABLE - XLR-132 PUMP FED
- o AFRPL TECHNOLOGY PROGRAMS IN PROGRESS AT ROCKETDYNE AND ALRC
- o HIGH PERFORMANCE 342+ SEC

\- OMS PUMP FED
- o MAN-RATED
- o REUSABLE
- o DERIVATIVE OF FLIGHT PROVEN SYSTEM

Figure 3.0-4 Main Propulsion Candidates

The storable engine technology currently active is the XLR-132 program at AFPRL and pump fed OMS, a derivative to the STS OMS. The XR-132 program is studying 3750 lbf engines, however, we have used parametric data supplied by Rocketdyne in selecting an optimum thrust for storable OTV applications, including the DDT&E for a new engine. These engines represent high performance and the advantages of hypergolic, storable propellants.

The Aerojet pump-fed OMS would use low risk pump technology derived from their XLR-132 work and the flight proven, man-rated and reusable STS OMS engine. No technology work is under way for a new LO_2/MMH engine and this is a negative factor for this this propellant combination.

RELIABILITY AND MAN-RATING--The basic issue is how best to achieve an adequate level of mission success and manned safety. Mission success tends to be an economic issue, while safety merits a higher cost solution. The tools available are parts quality, subsystem internal redundancy, and multiple systems, as far as OTV design is concerned. Programmatically, we have considered and discarded the option of a standby rescue capability. Various combinations of the OTV design approaches have been assessed in developing a basic policy with regard to reliability and failure tolerance.

PROGRAM OPTIONS--The key program issues are: whether or not to start the OTV program with a ground-based phase; whether to configure the ground-based vehicle for the cargo bay or the ACC; and when to introduce a fully man-rated OTV system. Candidate programs illustrating these issues have been devised and evaluated. The results are summarized in Section 6.3 and detailed in the programmatic volume.

4.0 CONFIGURATION TRADES & ANALYSES

This section summarizes the major system trades that led to the selection of the high potential Orbital Transfer Vehicle configurations. They led to selection of the preferred staging concepts for cryogenic and storable propellants, the preferred retrieval options, the preferred main engine selection, the preferred general arrangement and the preferred approach to man-rating and redundancy. In addition, the most significant results of the subsystem trades are summarized. All of these subjects are dealt with more exhaustively in other books and volumes of this final report. See Volume II, Book 4 for operations trades, volume III for system and program trades, and Volume IV for Space Station accommodations.

4.1 STAGING/PROPELLANT TRADE

The first step in our OTV definition Process was to do a parametric assessment of the reasonable propellant and staging options. This activity was supported by analyses conducted under the Task 6 system trades. These results were coarse screened in accordance with criteria negotiated between MSFC and contractor personnel. Those concepts judged to have no possibility of being developed into a winner were not studied further.

Figure 4.1-1 lists the criteria that were used in coarse screening staging concepts. In evaluating ground-based concepts, either cryogenic or storable, it was determined first of all if the concept could capture driver missions in the early years of the mission model. Those missions should also be accomplished within the STS cargo limitation on a single STS launch. Finally, the early ground-based OTV must be designed within the technology level expected to be in place in 1987. Criteria for selecting promising space-based OTV staging configurations includes capture of the driver missions in the complete Mission Model through 2010. Those driver missions were discussed in Section 2.0. For space-based OTV, the required propellant becomes the important factor in measuring relative stage performance because of the cost of delivering the propellant from the ground to the Space Station. Staging simplicity was evaluated recognizing the increased maintenance complexity at the Space Station.

	CRITERIA	BASIS
GROUND-BASED	O CAPTURE DRIVER MISSIONS IN MODEL	ANALYSIS
	O WITHIN STS CARGO LIMITATION	ANALYSIS
	O SINGLE LAUNCH CAPABILITIES	ANALYSIS
	O WITHIN 1987 TECHNOLOGY	ASSESSMENT
SPACE-BASED	O CAPTURE DRIVER MISSIONS IN MODEL	ANALYSIS
	O PROPELLANT REQUIRED	ANALYSIS
	O STAGING SIMPLICITY	ANALYSIS

Figure 4.1-1 Initial Concept Screening Criteria

Separate screening activities were conducted for cryogenic, storable and combination propellants. Perigee stage, 1-stage, 1 1/2 stage and 2 stage options were evaluated.

Figure 4.1-2 presents the results of the ground-based LO_2/LH_2 coarse screening activity. The most important ground-based screening criteria is gross vehicle weight. The gross weight of the stage and payload are shown for several payload weights and staging arrangements. In the case of perigee kick staging, the weight of the required apogee kick stage is included. These values reflect the 460 second Isp from the RL10CAT-IIB engine, appropriate performance margin, and the following stage burnout weight algorithm:

$$W_{BO}=1.3033[2701+.0054688(Prop)+.7838497(Prop)^{2/3}]+.01(Prop)$$

PROPELLANT	MISSION		GROSS WEIGHT			
			PGE KICK*	1 STAGE	1-1/2 STAGE	2 STAGE
LO_2/LH_2 I_{SP} = 460	GEO	8K	37345	48900	46885	55314
		10K	45316	54934	52684	61125
		12K	51352	60991	58489	66953
		14K	58386	67055	64305	72798
		16K	65446	73134	70132	78655
	17050 (PLANETARY)		----	63552	----	----

CONCLUSIONS

- o 2-STAGE SHOULD BE ELIMINATED: POORER PERFORMER THAN ONE STAGE
- o 1-1/2 STAGE SHOULD BE ELIMINATED: INFERIOR TO PERIGEE KICK
- o PERIGEE KICK IS A VALID OPERATIONAL MODE
- o 1-STAGE IS PREFERRED APPROACH (MAY EXPEND CARRIER)

Figure 4.1-2 Ground-Based Cryogenic Screening Results

This algorithm reflects the ground-based design we developed during our 1983 IR&D studies -- a single engine, 4-tank design using a Nextel covered aerobrake that folds forward for stowage in the aft cargo carrier. The crosshatched candidates are clearly unacceptable because they exceed the capability of a single shuttle launch. The two stage designs should be eliminated from further consideration because their gross weight is greater than the simpler staging approaches. All other options are capable of meeting the initial driving mission requirement of delivering a 12.9K lb payload to GEO. Some may not be able to retrieve multimission support equipment. This possibility required further evaluation. 1 1/2 stage, using drop tanks, provides a definite gross weight advantage over single stage, but not as much as the perigee stage approach. Both of these approaches expend mission hardware -- and the perigee stage approach is preferred because of its gross weight advantage. It is clear the ground-based cryogenic OTV should be a single stage. We believe it should be sized to enable maximum performance in the single stage mode, and be used offloaded in the perigee stage mode whenever there is an operational advantage (e.g., S/C is designed with an apogee kick, the mission can be manifested with another spacecraft, etc).

The space-based OTV coarse screening differs from the ground-based case in that the missions anticipated are more ambitious and that the selection criteria changes. Note that the screening shown in Figure 4.1-3 was done for the Rev. 7 mission model, which was more demanding than Rev. 8, but the changes were not in a direction tending to invalidate the conclusions reached here. In the space-based case, the technology availability date does not drive and gross weight no longer needs to be within the STS cargo limitation. Transport to LEO is still the most important part of the LCC equation, but the quantification of single launch capability loses its importance in the Space Station era when propellants will be stored onorbit for when they are needed. The array of selection data was generated using the same weight and specific impulse assumptions as for the ground-based case. The later IOC admits the possibility of an advanced engine development, but this possibility does not effect the relative merit of the various staging arrangements. Our conclusions are as summarized in Figure 4.1-3. Two-stage configurations have no discernable advantage for GEO missions -- but are the preferred approach for the driving lunar mission. While the single stage approach is the simplest for GEO missions, 1-1/2 stage must be considered more carefully because it has a definite propellant advantage. We believe the perigee kick approach should be considered as a valid operational mode. In cases where an expendable apogee kick stage is a viable mission approach, propellant logistics cost can be significantly reduced.

Based on these screening results, we optimized a single cryogenic stage approach to GEO and planetary missions that evolved into a two stage design for the driving manned lunar sortie mission. We investigated the 1 1/2 stage alternative to the single stage, and it proved not to be a winner. We used the perigee kick mode for selected planetary missions. We investigated the use of mixed stages (cryo perigee stage with storable apogee stage) and found no significant benefit.

Figure 4.1-4 is an equivalent evaluation of the ground-based storable staging options.

PROPELLANT WEIGHTS (LB)

PROPELLANT	MISSION	PROPELLANT QUANTITY			
		PGE KICK*	1 STAGE	1-1/2 STAGE	2 STAGE
LO_2/LH_2	GEO DELIVERY 20K	53499	57171	54698	58914
	GEO SERV 7K/4.51K	----	45665	42183	48249
	GEO MANNED 14K/14K	----	80917	77343	82264
I_{SP} = 460	LUN MAN. 80K/15K	----	219059	213990	181263
	PLANETARY (*17065)	----	49123	----	----

* INCLUDES AKM

CONCLUSIONS

- O 2-STAGE CONFIGURATIONS SHOULD BE ELIMINATED FOR GEO MISSIONS: POORER PERFORMANCE THAN 1-STAGE
- O 2-STAGE CONFIGURATIONS ARE PREFERABLE FOR THE MANNED LUNAR SORTIE: LEAST PROPELLANT REQUIREMENT
- O 1-1/2 STAGE MUST BE CONSIDERED FOR GEO MISSIONS SIGNIFICANT PROPELLANT SAVING OVER 1-STAGE
- O PERIGEE KICK IS A VALID OPERATIONAL MODE FOR GEO DELIVERY
- O 1-STAGE IS PREFERRED APPROACH PRIOR TO LUNAR PROGRAM

Figure 4.1-3 Space-Based Cryogenic Screening Results

GROSS WEIGHTS (LB)

PROPELLANT	MISSION		PGE KICK*	1 STAGE	1-1/2 STAGE	2 STAGE
	GEO	8K	43129	74354	70068	74033
N_2O_4/MMH		10K	51644	83364	78537	82328
I_{SP} = 342.3		12K	60184	92412	87038	90662
		14K	68754	101503	95557	99029
		16K	77358	110642	104115	107435

WILL NOT PERFORM REQUIRED MISSIONS

EXCEEDS LIFT CAPACITY OF SINGLE STS FLIGHT

MAXIMUM MISSION IN 1993/1994 -- 12.9K-LB DELIVERY TO GEO

Figure 4.1-4 Ground-Based Storable Screening Results

The parametric data presented in this chart shows the gross weight of the OTV, propellant, and payload for GEO delivery missions from 8 klb to 16 klb. Data is included for the OTV operating as a reusable perigee kick stage, as a reusable single stage, as a reusable stage with expendable drop tanks (1-1/2 stage), and as a completely reusable two stage vehicle. The gross weights in the perigee (PGE) kick stage column include the weight of the expendable apogee kick motor which is required when the OTV operates in this mode. The crosshatching indicates those gross weights that exceed the lift capacity of a single STS flight and a single 1-1/2 stage point that, while marginal for a single STS flight can be discarded because the 8 klb GEO payload will not meet the mission requirements in the ground-based time frame. The conclusion from this study is that for ground-based delivery of payloads to GEO, the perigee mode of operation is the only viable storable OTV staging arrangement.

The parametric data in Figure 4.1-5 indicate the propellant weights required for space-based storable OTVs to perform the driver missions when operating as a perigee stage, single stage, 1-1/2 stage, and two stage vehicle. The gross weight for the perigee kick stage mode includes the weight of the expendable apogee kick motor required to complete the mission. The crosshatching indicates the conclusions that can be drawn from these data. First, it is obvious that the perigee kickstage mode is not suitable for roundtrip missions. The remaining crosshatching indicates those operating modes that require excessive propellant and therefore are not efficient ways to operate. It can be concluded from these data that the perigee kick stage is a very attractive mode of operation for delivery missions from the Space Station. For roundtrip servicing missions to GEO either 1-1/2 stage or two stage operation appears to be the most efficient staging arrangement. The propellant quantities required to accomplish the manned lunar missions with an all storable vehicle are so large as to make the feasibility of the missions questionable.

PROPELLANT	MISSION	PGE KICK*	1 STAGE	1-1/2 STAGE	2 STAGE
	GEO DELIVERY 20K	68819	97506	91898	90683
N_2O_4/MMH	GEO SERV 7K/4.51K	----	81779	74529	77542
I_{SP} = 342.3	GEO MANNED 14K/14K	----	154988	144175	144291
	LUN MAN. 80K/15K	----	404101	388075	299566

* INCLUDES GROSS WEIGHT OF AKM NEEDED TO COMPLETE MISSION (18462 LB)

PROPELLANT WEIGHTS (LB)

EXCESSIVE PROPELLANT REQUIRED

NOT SUITABLE FOR ROUND TRIP MISSIONS

Figure 4.1-5 Space-Based Storable Screening Results

The net conclusion of the storable screening is as follows. All storable delivery missions should use only the perigee kick mode of operation. Both 1 1/2 and 2 stage configurations should be considered for manned and unmanned GEO servicing missions. An all storable concept for implementing the lunar sortie mission appears to require excessive amounts of propellant. As a consequence, a cryo perigee stage in conjunction with a storable upper stage should be considered for this mission in preference to an all storable approach.

The combination propellant LO_2/MMH was investigated in a similar manner. The evaluation resulted in recommendations identical to those reached for the storable options, although the propellant quantities required were somewhat less.

4.2 RETRIEVAL TRADE

Our retrieval trades combined technical and programmatic data to develop a validated position on the economic viability of OTV retrieval and reuse in conjunction with the low Revision 8 Mission Model. We also established the most attractive means of implementing OTV retrieval. Very early in the process, we established the performance advantage of the aeroassisted approach to retrieval over the all propulsive approach. Figure 4.2-1 shows the conditions under which aeroassist provides a net mission propellant savings for two key missions selected from the Rev. 7 mission model. Our technical data shows that an aerobrake weight/recovered weight fraction of 0.19 is achievable for the delivery mission, 0.07 for the 14K lb round trip mission.

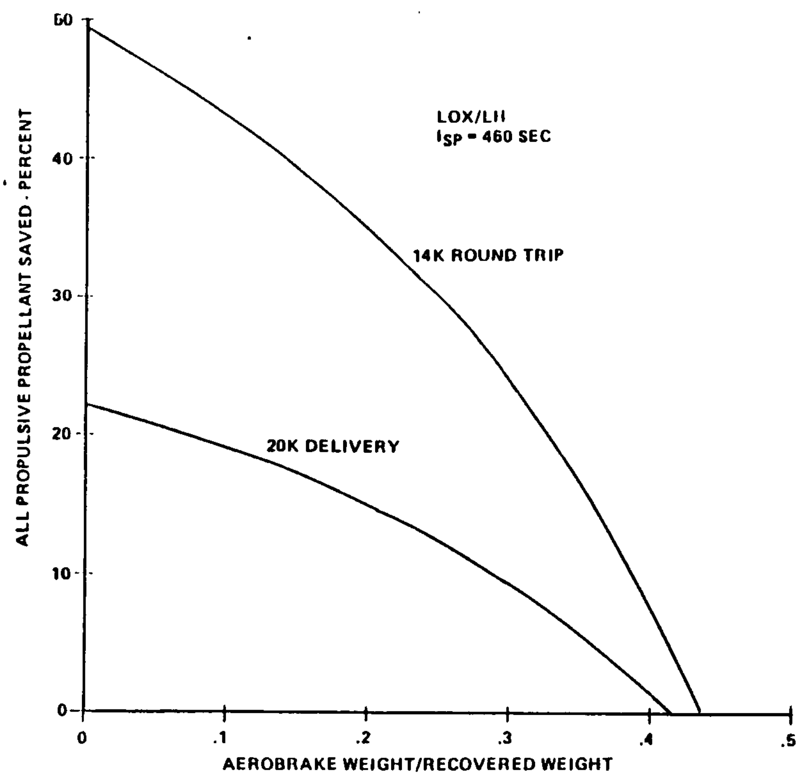

Figure 4.2-1 All-Propulsive vs Aeroassist

This indicates a sizeable propellant savings and, since propellant is a major portion of program cost, should indicate an economic advantage. This says nothing about the merit of retrieval and reuse over expendable stages. During our programmatic analyses we addressed the complete economic issue. Figure 4.2-2 presents the results of this evaluation. This chart shows the time development of benefits after payback for the 145 missions in the Rev. 8 low mission model. The benefit data compares the aeroassisted and all propulsive approaches with the expendable approach. The data reflects discounted dollars. The zero dollar line represents the expendable approach. The expendable reference was constructed assuming a mixed fleet of PAM's, IUS's and Centaurs. A stretched Centaur was conceived to implement the more difficult missions in the model. Retrievals were performed by the Centaur or stretched Centaur, but the Centaur was not reused. The figure shows a net benefit for reuse beginning in 1996, with the aeroassist approach yielding the larger benefit. The low model justifies the use of aeroassist -- the nominal model would increase its benefit margin. More details relative to this evaluation are presented in Volume III. Table 4.2-1 shows several evaluation parameters comparing aeroassist with all propulsive retrieval that were developed and reported in Volume III. Aeroassist is the superior approach in all respects except development cost. The return on investment column in the figure shows that this investment is economically justifiable.

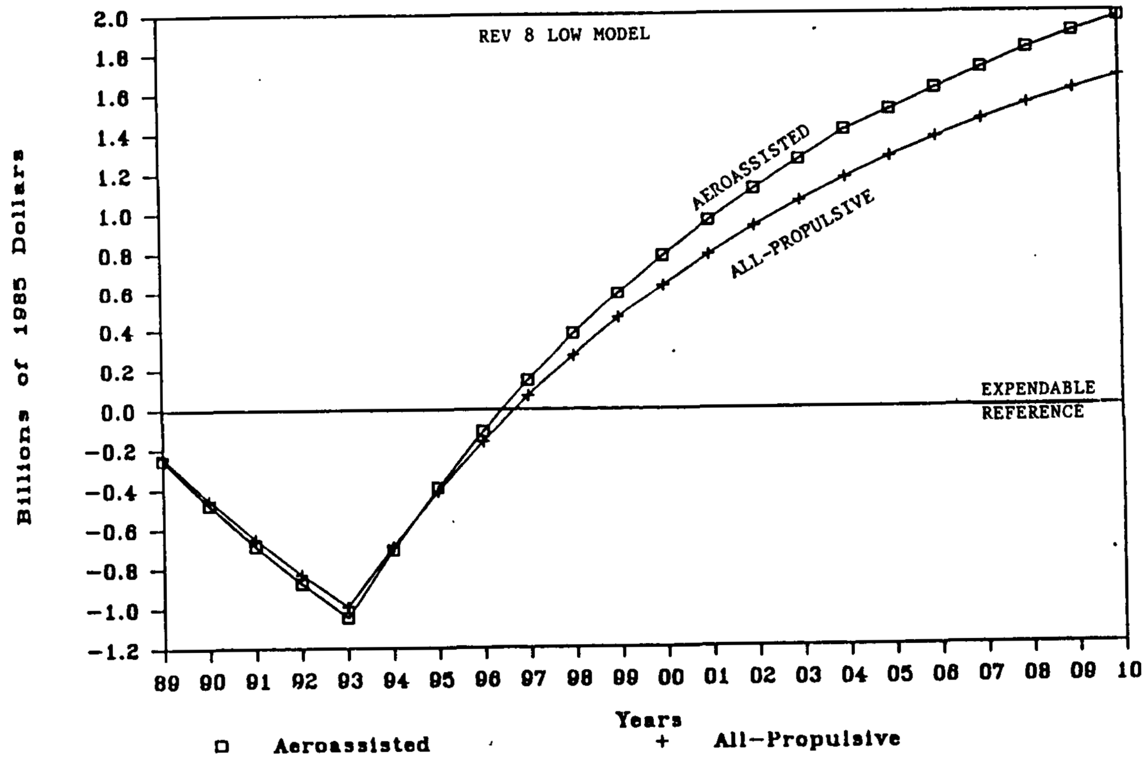

Figure 4.2-2 Aeroassist vs All Propulsive OTV Benefit

Table 4.2-1 All-Propulsive vs Aeroassist Results

REENTRY OPTIONS	ROI (PV)	BENEFITS	DDT&E +PROD (PV)	LCC (CV)	COST/ FLIGHT* (PV)	TOTAL OPS COST (PV)	PAYBACK NO/MISS. (10/YR)	DDT&E/ PROD./OPS TOTAL (PV)
ALL PROPULSIVE	2.5	2684	776	21390	86M*	4035	69	4810
AEROASSIST	3.1	3384	820	17953	77M*	3330	61	4150
RELATIVE SCORE (10 IS BEST)								
ALL PROPULSIVE	8.1	7.9	10.0	8.4	9.0	8.25	8.8	8.6
AEROASSIST	10.0	10.0	9.5	10.0	10.0	10.0	10.0	10.0

* INCLUDES P/L DELIVERY TO LEO

The basic aeroassist concepts investigated were shown in Figure 3.0-3. These included: the deployable, conical, fabric lifting brake; the blunt raked ellipse lifting brake; the aeromaneuvering hypersonic biconic sled; the inflatable ballute with inflation level modulating drag; the pyramidal brake with mechanical drag modulation; and the conical aerobrake with fluid aerospike. The key descriptive parameters of these concepts are summarized in Table 4.2-2. All of them possess a ratio of aeroassist device to vehicle return weight that will yield an advantage over all propulsive retrieval according to the criteria shown in Figure 4.2-1. The first trade we undertook was to decide between the low and mid L/D concepts. Figure 4.2-3 shows specific configurations developed to compare the impact of L/D on both storable and cryogenic propelled vehicles and Table 4.2-3 shows the resulting trade Parameters. In the storable case, the rigid/flexible aerobrake is a clear winner over the hypersonic biconic sled. The 1.0 L/D of the sled configuration is used to aerodynamically implement a portion of the 28.5 degree turn required to return from GEO to the east launched LEO. A 1410 pound fuel savings results. The dry weight of the aerobraked configuration, including both aeroassist and propulsive differences, was 6 KLB less than the sled configuration. The net initial weight advantage goes to the aerobrake by 4662 pounds, in spite of the lower velocity budget associated with the sled concept. A similar trade was performed for cyrogenic configurations incorporating a slant nosed cylinder and low L/D lifting brake concepts. The low L/D concepts were winners for GEO missions. We concentrated further efforts on low L/D concepts.

Table 4.2-2 Aeroassist Characteristics - Config. vs Weight

CONFIGURATION	L/D	W/C_DA	W_A	λ
DEPLOYABLE CONICAL FABRIC LIFTING BRAKE	0.12	10	1500	.07
BLUNT RAKED ELLIPSE LIFTING BRAKE	0.27	15	1800	.08
AEROMANEUVERING HYPERSONIC BIONIC SLED	1.0	70	6800	.27
INFLATABLE BALLUTE	0.0	6	3700	.15
MECHANICAL DRAG MODULATION	0.0	8	5640	.22
70% AEROBRAKE WTH FLUID AERO-SPIKE	0.0	4**	1520	.22**

NOTE: W_A = WEIGHT OF AEROASSIST DEVICE
λ = RATIO OF AEROASSIST DEVICE TO VEHICLE RETURN WEIGHT (14K) PAYLOAD
** = DATA APPLIES TO DELIVERY MISSION

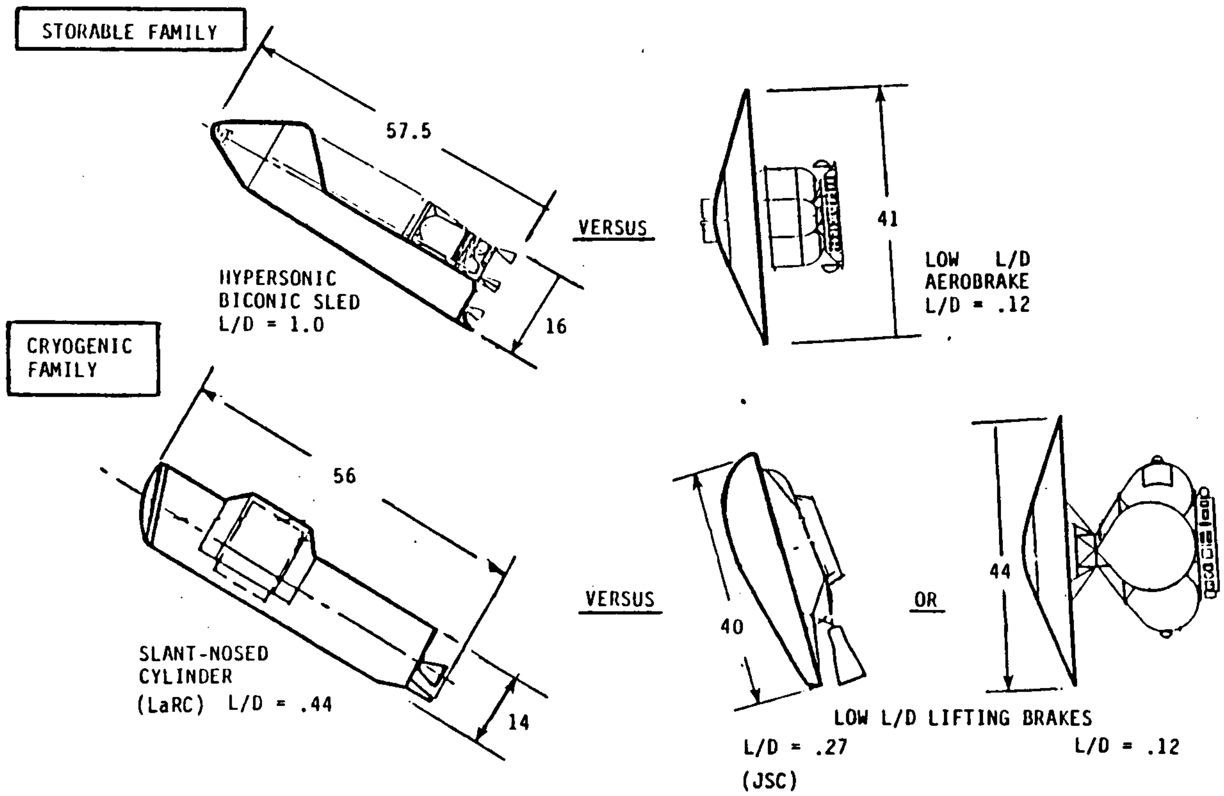

Figure 4.2-3 Low vs Mid L/D Performance Trade

Table 4.2-3 Low vs Mid L/D Trade Results

	STORABLE TRADE		CRYOGENIC TRADE		
CONFIGURATION	HYPERSONIC BICONIC SLED	RIGID/ FLEXIBLE AEROBRAKE	SLANT NOSED CYLINDER	RAKED ELLIPTICAL LIFT BRAKE	RIGID/ FLEXIBLE AEROBRAKE
L/D	1.00	0.12	0.44	0.27	0.12
W/C_DA	70.0	10.8	65.0	15.1	9.9
W_{TPS}	3357	1343	3023	1855	1490
W_{DRY}	12,585	6553	11,574	9757	7640
FUEL SAVINGS	1410			-234	-415
BENEFIT		+4662		+1583	+3519

- PROPELLANT SAVINGS FROM INCREASING L/D DOES NOT OFFSET VEHICLE WEIGHT INCREASE IN TPS.
- THE NET PERFORMANCE BENEFIT IS WITH LOW L/D AND NO INCLINATION STEERING.

The next configuration trade performed was to decide between mechanical drag control, aerospike drag modulation and lift control for low L/D concepts. Our trajectory simulation was used to compare these three basic approaches to aerobraking: Control corridor parametrics were generated for varying levels of aerospike thrust, drag modulation ratio, and L/D. All trajectories are for a ground-based OTV configuration returning from a geosynchronous mission orbit. All the parametrics were normalized to show impact of the various approaches on the aerodynamic control corridor. For the case of aerospike control, it may be seen from Figure 4.2-4 that the control authority is limited to an approximately 6 mile wide corridor (with correspondingly high propellant usage). The geometric constraints of mechanical drag modulation appear to limit its area variation to less than 3:1. From the chart one can see that this corresponds to a control corridor of 3 nm or less. This represents a somewhat marginal control situation, based on our aeroentry error analysis work. The offset C.G. approach (lift control) appears to offer the largest amount of control for the smallest vehicle impact. For example, L/D values of .25 are easily achievable with the 70 degree Viking aeroshell and result in control corridor widths on the order of 12 nm. This is more than adequate to cover trajectory dispersions. Our conclusion is that lift control is the most promising method of controlling the OTV through the aeropass.

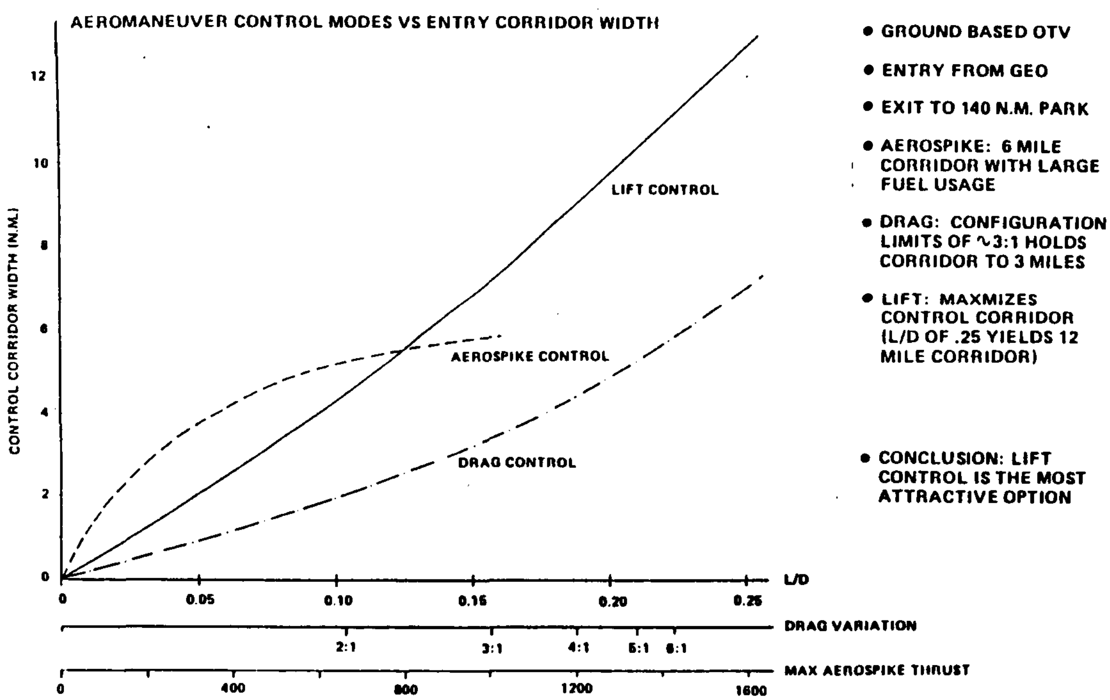

Figure 4.2-4 Aeromaneuver Control Modes

Our next trade study compared alternative means of implementing the low L/D concept with lift control. The first step in this process is establishing precisely how much L/D is required for adequate control. The most efficient configuration will be designed to operate at the lowest L/D capable of meeting control requirements.

Figure 4.2-5 presents an overview of the aeroentry process using low L/D and lift control. The control corridor forms a tunnel within the atmosphere which defines where the vehicle can successfully fly. Note that the bottom of the control corridor is defined by an operational boundary rather than a dynamic one. This is because flying at the bottom of the dynamic corridor causes very depressed perigees in the postaero orbit which requires a large amount of fuel to correct. Just prior to entry the OTV performs a final midcourse correction (entry minus 1 hour), stellar and GPS updates, and a preentry guidance update. After accomplishing these tasks, the OTV establishes an entry attitude which it holds until entry begins at a sensed acceleration of .03 g's.

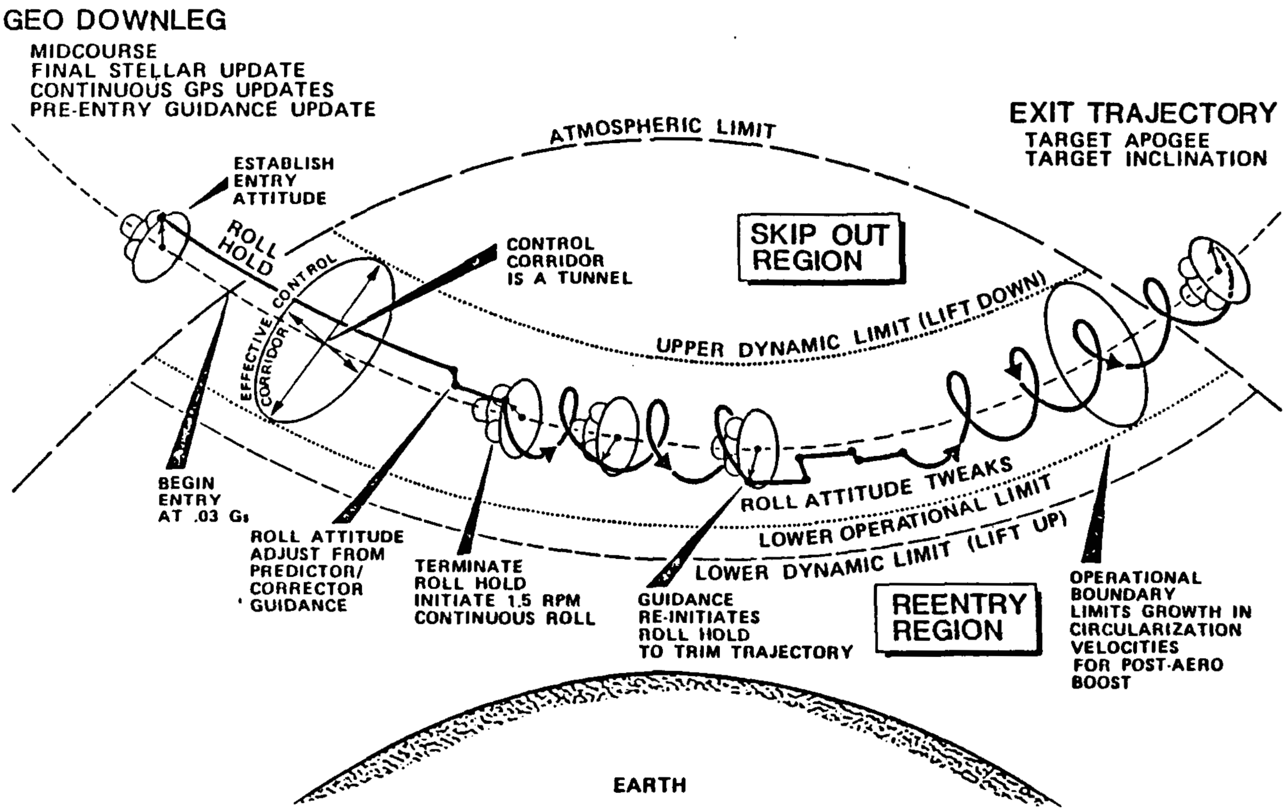

Figure 4.2-5 Aeroentry Overview

As the entry proceeds, guidance updates (every 10 seconds) refine the desired pointing of the vehicle lift vector. Upon achieving velocity targets, the vehicle initiates a continuous roll to null the fixed lift vector. In a typical trajectory, subsequent roll holds are required to tweak the trajectory. This process continues until the vehicle exits the atmosphere, at which time the apogee and inclination targets for the post-aero orbit have been achieved.

A series of error sources were considered and their impacts normalized to an equivalent variation in vacuum perigee. The RSS total of these effects was then used to size the aerocontrol corridor and the L/D of the vehicle. the sources were grouped into two categories: 1) targeting errors which cause OTV to miss its desired atmospheric aiming point and 2) aerodynamic variations which cause the vehicle to fly a different atmospheric trajectory than expected.

1) Tarteting errors - The last opportunity to correct the OTV's downleg trajectory occurs one hour before entry with a midcourse correction burn. All errors prior to this point are nulled out and only those factors that disturb the burn and subsequent trajectory are considered.

   a) Guidance Errors - Experience indicates an error of about 200 ft for this parameter.
   b) Pointing Errors - Midcourse burn attitude errors due to IMU misalignment (after stellar update) and cg trim errors amount to about 0.1 deg. which equates to 130 ft variation in vacuum perigee.
   c) Cutoff Errors - Accelerometer errors and a 10 millisecond shutdown uncertainty.
   d) GPS Error - Estimates of state vector errors for GPS at this stage and 2 fps in velocity. This leads to perigee errors of 845 ft and 9476 ft respectively.
   e) Onboard Clock Error - Very accurate time comes with the use of GPS - not a significant effect.
   f) Nongravitational Effects - Nonbalanced configuration of the RCS jets does not produce pure torques. This is estimated to result in a 320 ft perigee miss. Luni-solar effects will be biased by ground targeting.

2) Aerodynamic Variations - No two aeroentries will be quite the same. The impact of variations in the atmosphere and the vehicle are accounted for here.

   a) Atmospheric Uncertainty - The variation in density has been ground ruled by MSFC at 30%
   b) L/D Uncertainty - An angle-of-attack variation of 1° due to variations in the entry cg
   c) Ballistic Uncertainty - Weight uncertainty - 150 lbs (propellant residual uncertainty), coefficient of drag (C_d) variation = 10% (Shuttle and Viking experience), and brake area variation = 5% (to cover uncertainties in the flex of the support ribs and flexible TPS blanket). The RSS effect of these factors on ballistic coefficient is 12%.

RSS'ing of all the above factors (See Table 4.2-4) yields a net variation in perigee of ±1.27 nm. A control corridor of 3.5 will cover this uncertainty with adequate margin. Further closed loop flight simulations were run with the actual local variations in the upper atmosphere encountered on Shuttle flights. The net impact of these variations is to require an increase in control corridor over that indicated by the RSS analysis just described. The detailed trajectory results are presented in Volume II, Book 3. As a result of these analyses, we increased the control corridor requirement to ± 2.5nmi, or a total width of 5 nmi.

Table 4.2-4 Aeroentry Error Analysis

	EQUIVALENT PERIGEE ERROR		
o TARGETING ERRORS (MIDCOURSE)			
• GUIDANCE ERRORS	= 200 FT		
• POINTING ERROR	= 130 FT	± .1 DEG	
• CUTOFF ERROR	= 490 FT	.33 FPS ACCELEROMETER + 10 MS TIMING ERROR	
• GPS ERROR	= 575 FT	FROM 1020 FT POSITION UNCERTAINTY	
	474 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY	
• NONGRAVITATIONAL	= 320 FT	ACS IMBALANCE	
o AERODYNAMIC VARIATION			
• ATMOSPHERIC UNCERTAINTY	= 5700 FT	± 30% DENSITY	
• L/D UNCERTAINTY	= 4500 FT	± 1° AT = 7.2° ANGLE OF ATTACK	
• BALLISTIC UNCERTAINTY	= 2500 FT	WT = ± 150 LB (RESIDUALS)	± 12% W/C_DA
		C_D = ± 10% (STS FLT DATA)	
		A = ± 5%	
• RSS	= ± 977 FT = ± 0.16 N.M.	FROM TARGETING	
	= ± 7680 FT = ± 1.26 N.M.	FROM AERODYNAMICS	
	= ± 7742 FT = ± 1.27 N.M.	NET VARIATION	

Using the 5 nm control corridor width that results from the aeroentry error analysis it is possible to specify the L/D requirements for the OTV. A series of continuous lift-up and lift-down geosynchronous return trajectories were generated for various L/Ds to define corridor boundaries. The resulting control corridor widths are plotted in Figure 4.2-6. This data shows that an L/D of 0.116 gives the desired 5 nm corridor. This L/D is achieved via an angle-of-attack of 7.2 degrees based on Viking data for this type of aerobrake shape. An analysis of free molecular flow effects shows no significant impact on this angle of attack.

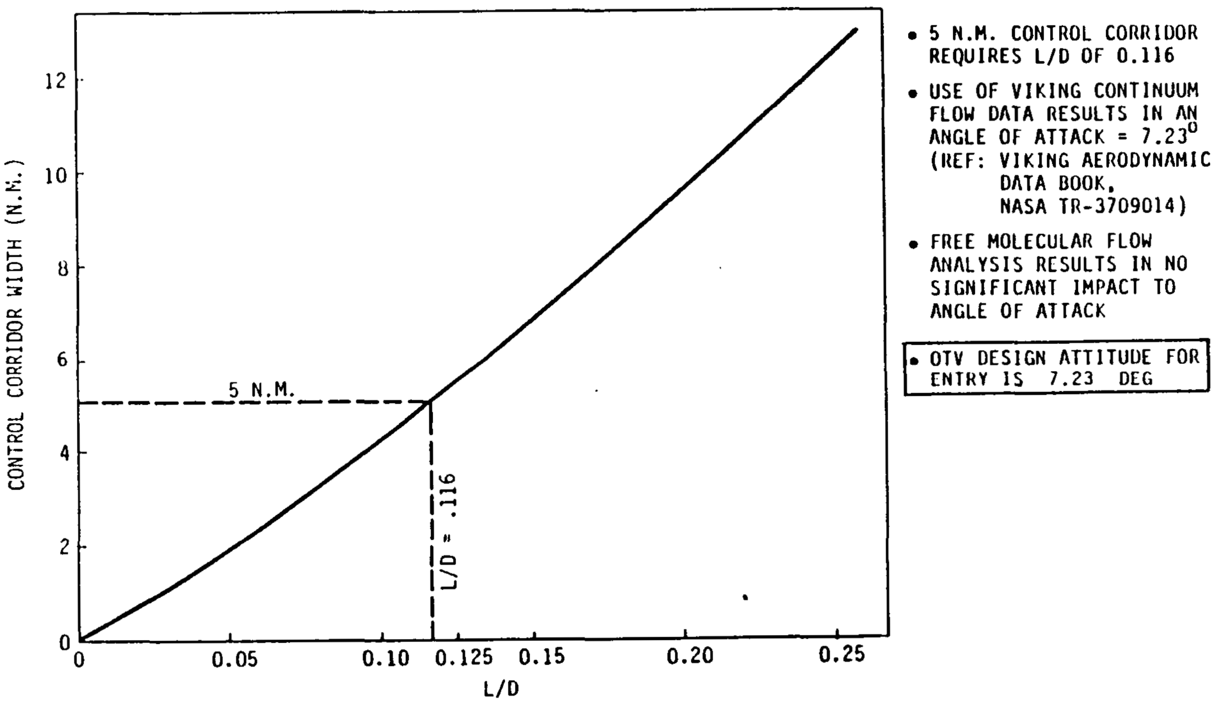

Figure 4.2-6 L/D vs Control Corridor

The primary low L/D concepts are the inflatable ballute, the rigid raked ellptical cone, and the Viking shaped rigid/flexible fabric aerobrake -- as shown in Figure 4.2-7. The ballute and fabric brakes both use flexible thermal protection systems surrounding a rigid spherical segment nose cap with protective doors covering the main engines. The raked ellipse uses rigid thermal protection over the entire exposed area. The propulsive axis is located transversely, eliminating the need for engine doors in the heat shield. Table 4.2-5 provides comparisons of six areas for the three low L/D aerobrake candidates. Design factors for both drag and lift devices; aerobrake/stage characteristics; operational impacts on launch to orbit and Space Station reuse and replacement; payload sizes -- brake dimensions, weights and efficiency ratios; OTV design impacts; and concerns and risks for TPS, control, feasibility, and weight growth are shown. As a final comparison of the ballute concept versus the fixed passive structure, wind tunnel data of these two approaches were compared. References 4 and 5 which were prepared during the Viking development activity provided an additional comparison of an "attached inflatable decelerator" (essentially a ballute) with the rigid shape eventually selected for Viking. The Viking shape was a superior decelerator with a higher drag coefficient, and had a better potential for producing L/D for control purposes. An additional significant difference is static aerodynamic stability. The Viking center of pressure lies 1.01 brake diameters aft of the brake nose, while the ballute is only 0.3 diameters aft. This makes it possible to stabilize a longer stage/payload configuration using a small brake diameter with the Viking shape. considering the comparative data presented, it is our position that the Viking shaped rigid/flexible aerobrake is the superior low L/D aeroassist concept.

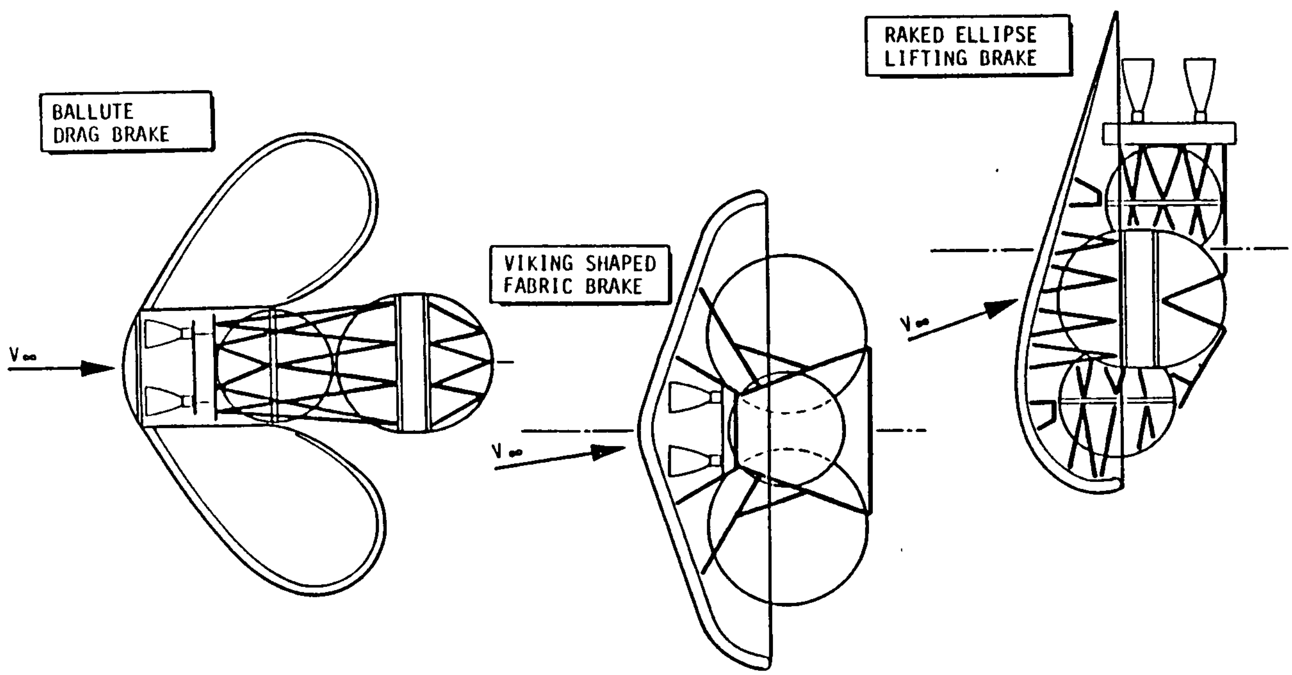

Figure 4.2-7 Low L/D Aero Configuration Concepts

Table 4.2-5 Aerobrake Concept Comparison

FACTOR	INFLATABLE BALLUTE	RAKED ELLIPTICAL CONE	RIGID/FLEXIBLE AEROBRAKE
I.DESIGN SUMMARY • DATA SOURCE • L/D • W/C_DA • CONTROL MODE	 BAC STUDIES ZERO 4.6 / 13.3 AREA VARIATION	 JSC STUDIES 0.3 OR LOWER 8.1 / 15.1 ROLL CONTROL	 MMC STUDIES 0.12 4.0 / 11.6 ROLL CONTROL, OFFSET CG
II.CHARACTERISTICS • GEOMETRY • BRAKE BASE DIA • STAGE DIMENSIONS • AEROSHIELD TPS • LONG. STABILITY (STABLE CG RANGE AFT OF NOSE)	 BLUNT CONICAL SPHERICAL NOSE 50 FT 14D X 34L RIGID/FLEX CP VARIES WITH TDR (25 FT)	 RAKED CONE ELLIPSOIDAL NOSE 40 FT 38D X 14L RIGID 1-RADIUS AFT OF A/B BASE (34 FT)	 BLUNT CONIC SPHERICAL NOSE 44 FT 38D X 25L RIGID/FLEX WIDE CG LATITUDE (43 FT)
III.OPERATIONS • SHUTTLE TRANSPORT TO SPACE STATION • SPACE STATION -REUSE -REPLACEMENT	 SHIP FOLDED FABRIC AS UNIT NOT PRACTICAL, RECHARGE PRESSURANT SIMPLE- INSTALL UNIT	 SECTIONS-ASSY REQUIRED YES - VISUAL CHECK COMPLEX- REPLACE TILES OR ASSY	 SHIP ASSY w FABRIC FOLDED YES - VISUAL CHECK SIMPLE- INSTALL AS A SINGLE ASSY
IV.SIZE 20K P/L DELIVERY • AEROBRAKE DIA-FT • AEROBRAKE MASS • STAGE DRY WT, LB • W_{BRAKE}/W_{RETURN}	LONG. STABILITY 40 1569 8070 194	WAKE HEATING 37 1587 9489 0.167	WAKE HEATING 38 1270 7140 0.178
7.5K MAN GEO SORTIE • AEROBRAKE DIA-FT • AEROBRAKE MASS • STAGE DRY WT, LB • W_{BRAKE}/W_{RETURN}	 50 2452 8950 0.149	 40 1855 9757 0.107	 44 1407 7560 0.093
15K MANNED LUNAR • AEROBRAKE DIA-FT • AEROBRAKE MASS • STAGE DRY WT, LB • W_{BRAKE}/W_{RETURN}	 62 3700 10250 0.146	 40 1923 9825 0.077	 44 1489 7640 0.066
V.OTV DESIGN IMPACT • CONFIGURATION (PARALLEL TANKS INCREASE LENGTH, PLUMBING & RESIDUALS)	 GOOD WITH STORABLE PROP, TANDEM AND TOROIDAL CRYO TANKS	 OVERSIZED FOR MANY MISSIONS, INTEGRATED CONCEPT, OPTIMIZES WITH PARALLEL TANKS	 GOOD FOR ACC USE(NO ASCENT LOADS), NO TANK CONSTRAINT (4 TANKS BEST)
VI.CONCERNS-RISKS TPS	 -SINGLE REUSE -ASSY JOINTS -LOCAL & GLOBAL LOW ΔP FLUTTER -LOBE RADIATION TRAP -PACKAGE VOL	 -ASSY JOINTS -ON-ORBIT ASSY -P/L WAKE HEAT -PROVEN TPS	 -LOCAL ΔP FLUTTER -BASE HEATING -FLEX TPS REUSE -ASSY JOINTS
CONTROL	-PREENTRY SPIN -TURNDOWN RATIO LIMIT -DEFLATION	-SIDE FIRING ENGINES -ASCENT CG OFFSET FOR LONG RETRIEVED P/L	-CG TRIM ERROR -ACS LOCATION
BASIC FEASIBILITY	MODERATE -SHAPE STABILITY -FABRIC FLUTTER	LOW -ONORBIT ASSY & MAINTENANCE	MODERATE -FABRIC FLUTER -MAINTENANCE
WEIGHT GROWTH	MODERATE -8200 LB MAX RETURN WEIGHT FOR 50 FT DIA	LOW -BLOCK CHG TO INCREASE TANK OR BRAKE SIZE -RETURN P/L SHAPE & SIZE VARIABLE	LOW -MODERATE FOR DIA INCREASE -COMPACT STAGE HAS CG MARGIN FOR RETURN GROWTH

4.3 MAIN ENGINE TRADE

Cryo Engine Selection -- The main engine candidates for a cryogenic OTV fit into three classes, as shown in Figure 3.0-4. They can be derivatives of the RL-10 technology, composed of advanced technologies, or established at an initial entry point into the advanced technology. The best selection depends on the use anticipated over the coming decades. We have established our recommendations, as directed by MSFC, based on the Rev. 8 "low" OTV mission model. We performed a comprehensive comparison of the various options that is reported in its entirety in Volume III and supported in Volume II, Book 3. Figure 4.3-1 shows the development of benefits in discounted dollars as mission usage increases. This comparison was made against the RL-10-A-3 as a reference -- this case forms the zero-benefit line on the chart. Five specific engine development possibilities were conceived for comparison with this reference case. The RL10-IIB at 460 seconds specific impulse and 15000 pounds thrust is representative of the RL10 derivatives. The RL10-IIIB at 470 seconds and 7500 pounds in either single or dual installation is programmatically little different. The advanced engine is characterized at 483 seconds specific impulse, 7500 pounds thrust with a 10 hour service life. The IOC engine represents the lower end of the specific impulse range achievable by new higher chamber pressure engine technology, lower mission life qualification, and less sophisticated capabilities such as continuous throttling and condition monitoring. This is an engine that has a clear evolutionary path to the advanced engine. It is characterized by a 475 specific impulse, a 7500 pound thrust, and a 5 hour life. The other two candidates on the chart show the impact of transitioning from the RL-10 derivative or the IOC engines to the advanced engine. Figure 4.3-1 indicates that the highest front end funding produces the most benefits at the end of the low mission model.

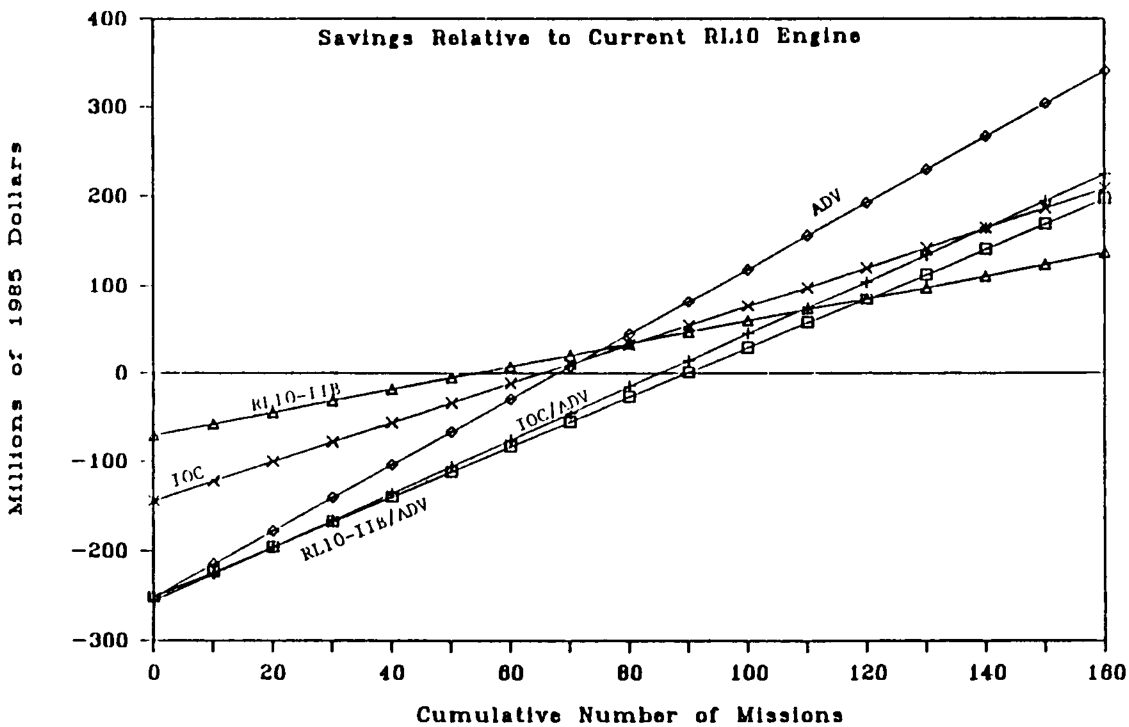

Figure 4.3-1 Engine Payback Comparison

This result must be considered in the light of other considerations such as those shown in Table 4.3-1. These data indicate the RL-10 is 28% better than the IOC engine for return on investment, and 44% lower than the IOC for DDT&E and production and over 40% less expensive in peak funding. The advanced cryogenic engine is 21% better than the RL-10/ADV in benefits, lower in LCC by 11% over the IOC, and lower in engine cost/flight. Payback based on a 54 flight break even for the RL-10-IIB is 11 additional flights for the IOC and 14 additional for the advanced engine. These data indicate the low number of missions biases towards a derivative engine, and the significant advantages of an advanced engine are most beneficial over the long term with increased missions.

Table 4.3-1 Cryo Main Engine Trade Results

MAIN ENGINE OPTIONS		ROI (PV)	BENEFITS (PV)	DDT&E + PROD (PV)	LCC (PV)	PEAK FUNDING	ENGINE COST/FLT CONST $	PAYBACK NO. MISSIONS
RL-10/ADV	1	.79	449	251	2104	40	59	90
IOC/ADV	2	.86	474	255	2083	29	58	82
ADV	3	1.1	535	251	2018	52	55	68
RL-10	4	1.3	159	70	2213	14	66	54
IOC	5	.9	273	143	2172	25	62	65
RELATIVE SCORE (10 IS BEST)								
RL-10/ADV	1	6.1	8.4	2.8	9.6	3.5	9.3	6
IOC/ADV	2	6.6	8.9	2.7	9.7	4.8	9.4	6.6
ADV	3	8.5	10.0	2.8	10.0	2.7	10.0	7.9
RL-10	4	10.0	3.0	10.0	9.1	10.0	8.3	10.0
IOC	5	7.0	4.9	5.6	9.3	5.6	8.9	8.3

The RL-10 derivatives represent existing technology with ongoing product improvement providing an OTV engine at minimum DDT&E cost. Based on the Revision 8 low mission model, the return on investment (ROI) was the highest with a 54 mission payback. The current engine is flight proven without a single mission failure. However, this technology has the highest Life Cycle Cost (LCC), limited growth and the lowest benefits for a future OTV. Alternatively, an advanced engine has the lowest LCC, lowest cost per flight (CPF), and has the greatest benefits over all the competition. The payback period is 68 missions. The front end program cost is high to achieve this technology and could incur schedule risk depending on the level of technology investment and accomplishment prior to ATP.

An OTV engine using low risk technology improvement provides an attractive alternative to the near term technology, low DDT&E approach. It provides the long term benefits of an advanced engine cycle at low risk. Growth capability is retained while keeping LCC, ROI and DDT&E competitive. The payback period is only 11 missions greater than the RL 10 derivative IIB/IIIB and slightly shorter than the advanced technology engine. While the IOC engine has a high cost per flight, it has a 72% improvement in benefits compared to the RL 10-IIB. We recommend this initial operational capability approach because it offers the OTV program a high performance, low front end cost engine with planned growth potential. This provides the opportunity to improve efficiency, performance, cost per flight, and capacity for future OTV delivery, planetary, and manned missions.

Thrust Level vs. Perigee Burns -- Figure 4.3-2 shows the propellant required to perform the 20 klb delivery mission as a function of OTV main engine thrust level and number of perigee burns for Rocketdyne (a) and Pratt and Whitney (b) engine data. Note that optimum thrust level decreases as the number of perigee burns increases. The relative mission cost of multiple burns shown in Figure 4.3-3 was estimated based on: the indicated optimum thrust level; propellant use at $1500/pound; more frequent engine changeout as thrust is decreased; and increased operational cost and higher mission loss cost as mission duration increases with the number of perigee burns. The net effect is less than $1M per flight regardless of the number of perigee burns. Thus multiple burns yields only a small savings that would limit the growth of OTV for planetary and lunar missions and that would increase mission complexity. Any decrease in propellant delivery cost, which is anticipated in the event that ET propellant scanvenging proves feasible, will reduce or eliminate this savings. We elected to size thrust level based on a single perigee burn, selecting a higher thrust that would allow for growth and reduce velocity losses for the planetary and lunar missions. These considerations resulted in recommending a thrust level of 7500 pounds for the OTV main engine.

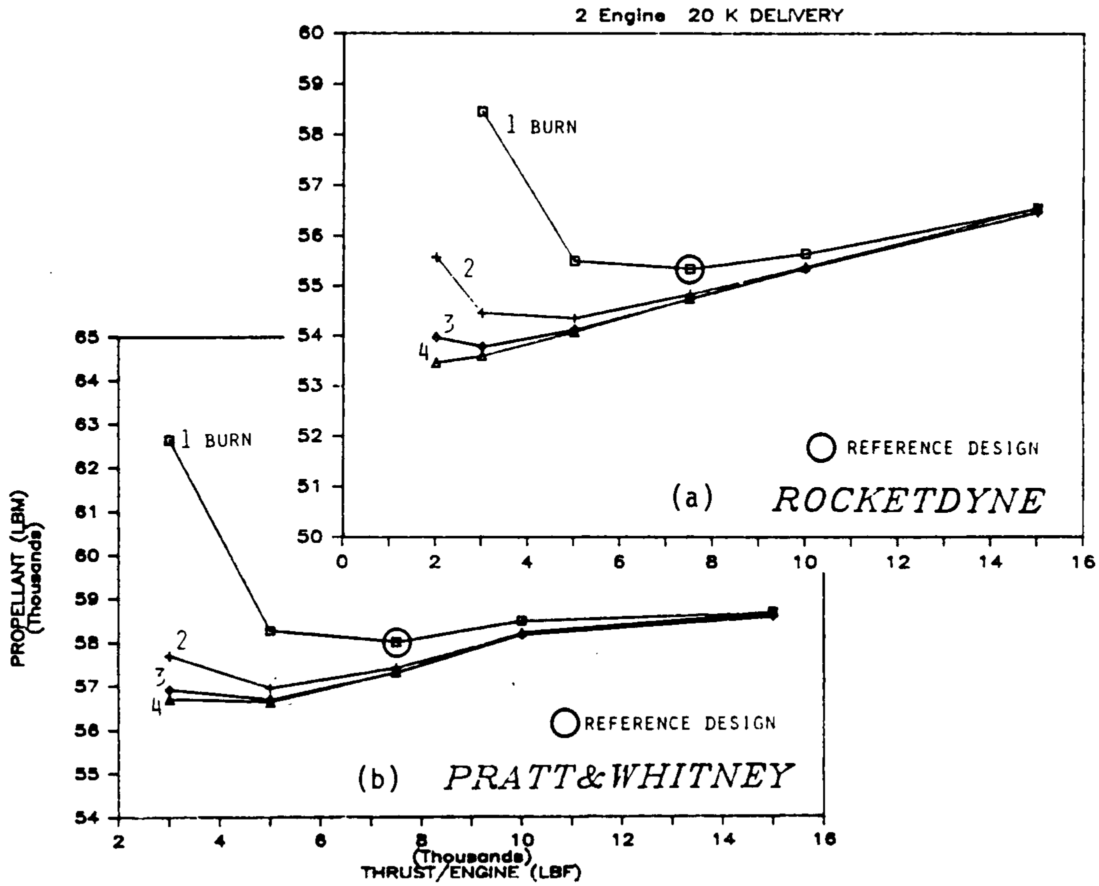

Figure 4.3-2 Cryo OTV Thrust Trade

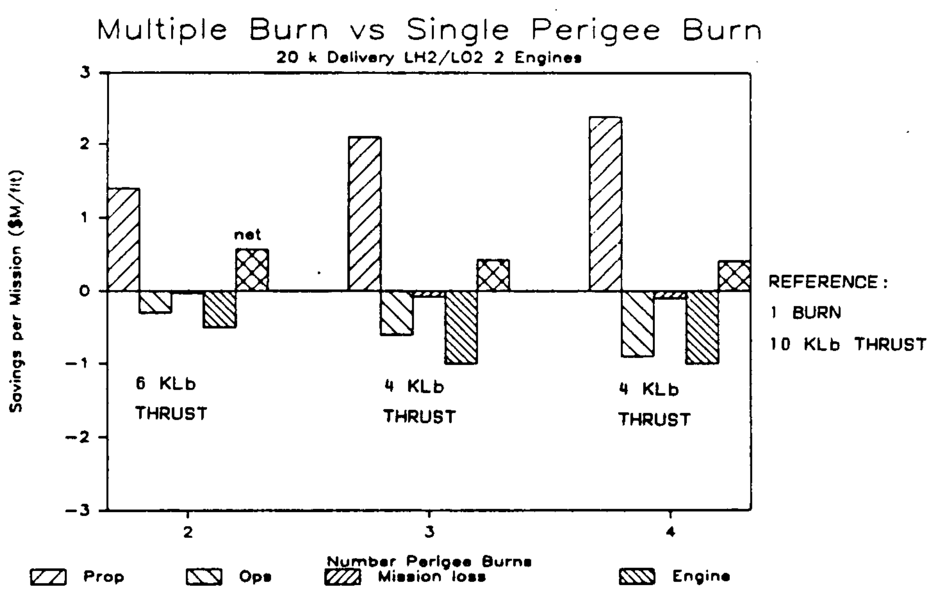

Figure 4.3-3 Multiple Burn Cost Trade

Engine Life - The optimum engine life was determined based on the cost of maintenance and engine life development and testing (assumed at $3M/hr). Engine replacement cost for depot level maintenance was assumed in this analysis with one overhaul over the engine's useful life. The Revision 8 mission model was used and the LCC reflects engine replacements beginning in 1995 at an average cost of $10.93M. The results shown in Figure 4.3-4 indicate an optimum MTBO of 7.5 hours (low) with small savings after 5 hrs. While engine life is sensitive to the number of missions, the effect of the number of units on engine recurring cost was not considered.

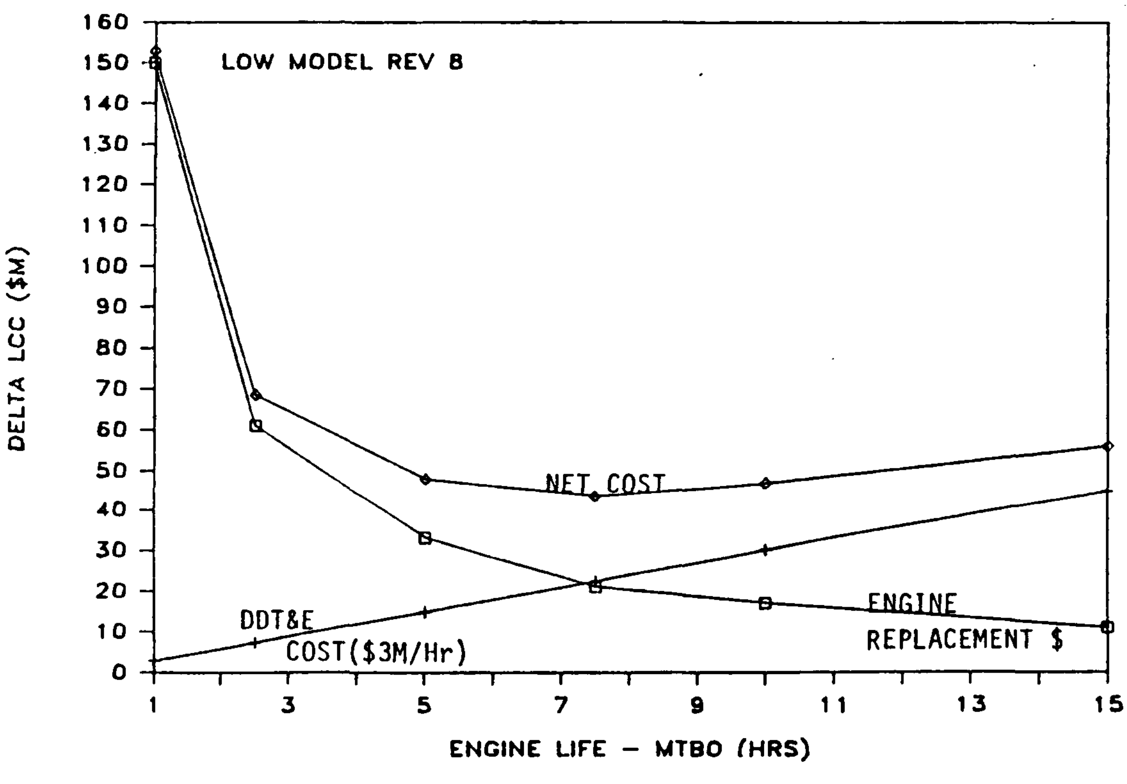

Figure 4.3-4 Optimum Engine Life

Recommended Cryo Engine Requirements - The requirements for our recommended state-of-the-art (SOA) liquid hydrogen/liquid oxygen OTV engine are given in Table 4.3-2. They were derived from our system analysis of the current engine designs. The Isp is based on an economic analysis of a 20,000 lbm payload, 2-engine stage, and a single perigee burn. The dimensions were based on the engine optimization done for both the Pratt & Whitney RL-10 and the Rocketdyne engine. The major driver for engine dimensions is their affect on the aerobrake diameter and consequently its mass. The engine exit diameter effects the spacing between engines and gimbal requirements with attendant impact on stage length, aerobrake diameter, and engine doors. Engine stowed length directly effects both the stage length and aerobrake diameter. Autogenous pressurization was selected for the space-based OTV because of the potential complication of filling an OTV in low-g with a noncondensible gas such as helium present in the tank. This was the driver for the Tank Head Idle (THI) which can eliminate the need for prepressurization prior to engine start. The NPSH values where based on current engine designs and were not optimized. The THI inlet pressure was based on the conditions at which the propellant isstored. Discrete throttling capable of meeting Rev. 8 mission model requirements was selected because continuous throttling will complicate engine development and increase costs. Aerobrake requirements establish the time available to retract nozzles or to eject a failed nozzle and close protective doors. The development cost was based on discussions with the engine manufacturers, affordable engine technology, 5 hr life, and engine changeout as a complete unit. The cost represents our estimate based on data provided by Aerojet and Rocketdyne.

Table 4.3-2 Recommended Engine Requirements

	REQUIREMENT	RATIONALE
PERFORMANCE	$\geq$ 475 SEC @ 6:1 MR	COST TRADE BETWEEN EXISTING AND ADVANCE TECHNOLOGIES, MINIMUM ISP.
THRUST	7500 LBf	PERFORMANCE ANALYSIS- SINGLE PERIGEE BURN AND 2 ENGINES
MASS	280-300 LBm	PERFORMANCE ANALYSIS OF 2 ENGINE VEHICLE
DIMENSIONS DIAMETER LENGTH	 $\leq$ 50" < 60" STOWED < 120" EXTENDED	VEHICLE OPTIMIZATION WITH FIXED AEROBRAKE AND 2 ENGINES GIMBALLED THRU C.G. WITH 20 DEG MAX GIMBAL.
PRESSURIZATION AND CHILLDOWN	GO2/GH2 PRESSURIZATION THI START @ 15 PSIA NPSH 15' H2, 2' O2	NON-CONDENSIBLE PRESSURANT COMPLICATES ON-ORBIT REFILL - ELIMINATES GHE PRESSURIZATION.
THROTTLING	STEP THROTTLING 50% @ $\geq$465 SEC	REV 8 MODEL CONTAINS 6-7 LOW THRUST MISSIONS. CONTINUOUS THROTTLING COMPLICATED ENGINE DEVELOPMENT.
AEROBRAKE IMPACTS	LAST FIRING 1 HR BEFORE AERO-MANEUVER FIRING 10 MIN AFTER EXIT ATMOSPHERE	THI USED FOR MID-COURSE, COULD BE USED FOR RAISING PERIGEE AFTER AEROPASS
DEVELOPMENT COST	$175M, 60 MOS	5 HR LIFE, LOW CHAMBER PRESSURE MINIMIZE TECHNOLOGY RISK, ENTIRE ENGINE IS ORU

Storable Engine Selection - We compared the pump fed OMS-E engine with one that uses the technology being developed by AFRPL (SLR-132) for use on the storable OTV. Figure 4.3-5 shows the propellant mass required to deliver various payload masses to GEO for stages built around these engines. The OMS based OTV uses a 6000 pound thrust, 334 second specific impulse engine reflecting an increase in nozzle expansion ratio over the STS OMS engine. The SLR-132 based OTV used an optimized 7500 pound thrust engine delivering a specific impulse of 344.1 seconds. Both engines are able to meet the ground-based driver mission. The XLR-132 type engine is recommended, however, because it provides superior performance, it can meet the required IOC, and it provides a clear path to the space-based storable OTV where higher performance is extremely important.

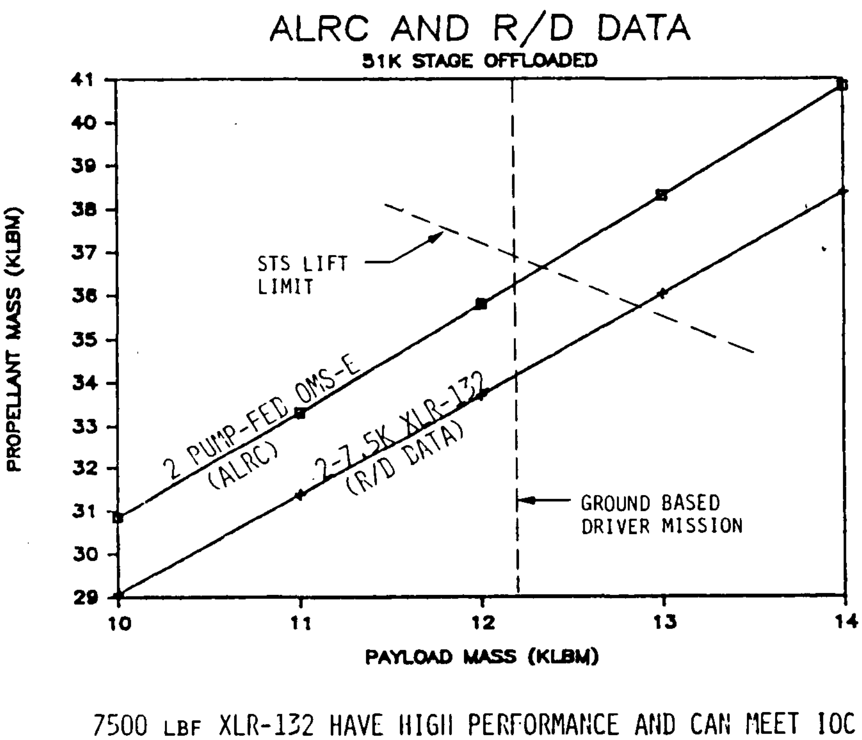

Figure 4.3-5 Ground-Based Storable Engine Selection

4.4 GENERAL ARRANGEMENT TRADES & ANALYSES

A series of trade studies were run to optimize the general arrangement of the OTV configurations. Effort was concentrated on the cryogenic configuration, and the results adapted, where applicable to the storable configurations. The subjects addressed include: 1 vs 1 1/2 stage; 1 vs 2 engines; ground to space commonality; packaging for transport in the cargo bay; and arrangement for space-base assembly and maintenance.

1 VS 1 1/2 STAGE - Parametric evaluation reported in paragraph 4.1 suggested that 1 1/2 stage configurations could have a performance advantage over one stage configurations. As a consequence, a more detailed configuration study reflecting the selected 4-tank, aeroassisted concept was run to establish the merit of conceptual variations using drop tanks. Two drop tank configurations were compared with a reference single stage configuration. This reference held 84,000 pounds of propellant packaged in four near spherical, side-by-side tanks. Figure 4.4-1 shows two potential drop tank arrangements. In the first comparison case, four drop tanks are packaged around a down sized set of four fixed side-by-side tanks. In the second case, the drop tanks are added in tandem with the fixed set of side-by-side tanks. Weights were established and propellant capacity adjusted until the GEO performance capability of the drop tank configurations equaled the capability of the original single stage configuration. The results are indicated in the figure. Both drop tank configurations require greater total propellant usage than the single stage configuration. This result is opposite to the preliminary parametric conclusion reached in Paragraph 4.1, and indicates that, for the selected baseline concept, practical considerations of attachment hardware and aeroassist layout outweigh the theoretical benefit of the droptank approach. Combining this greater propellant requirement with the cost of the expendable drop tanks makes it clear that the single stage configuration is superior. No further effort was expended on drop tank configurations.

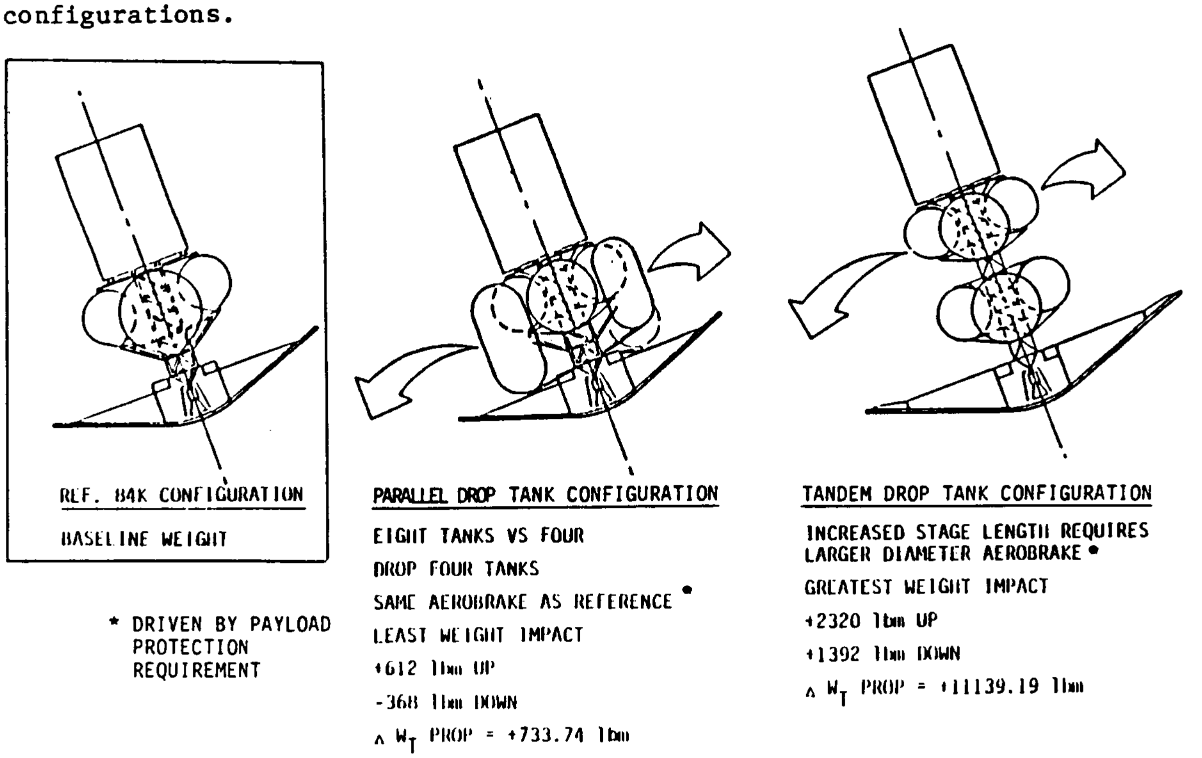

Figure 4.4-1 Cryo Drop Tank Trade

1 VS 2 ENGINES -- It was established that man-rated space-based configurations should have two engines to assure single failure tolerance. This study addressed the feasibility of using two engines on ground-based aft cargo carrier configurations to make evolution to the ultimate space-based configuration more straightforward. This study was conducted using the rather large RL10 derivative configurations. Figure 4.4-2 shows that in order to gimbal through the worst case CG, the two engines must be lowered to the point where the aerobrake extends beyond the maximum allowable ACC envelope. The maximum permissable gimbal angle was established to be 20°. The maximum envelope is based on the special purpose ACC design with a spherical dome extended 7 inches longer than the general purpose ACC design. The engines do not clear the ACC envelope except when gimballed to the full outboard position, and do not leave adequate room for installation of the aerobrake. It was concluded that a single engine configuration is preferred for flight in the ACC, and that two engine commonality through the program is, as a consequent, not practical.

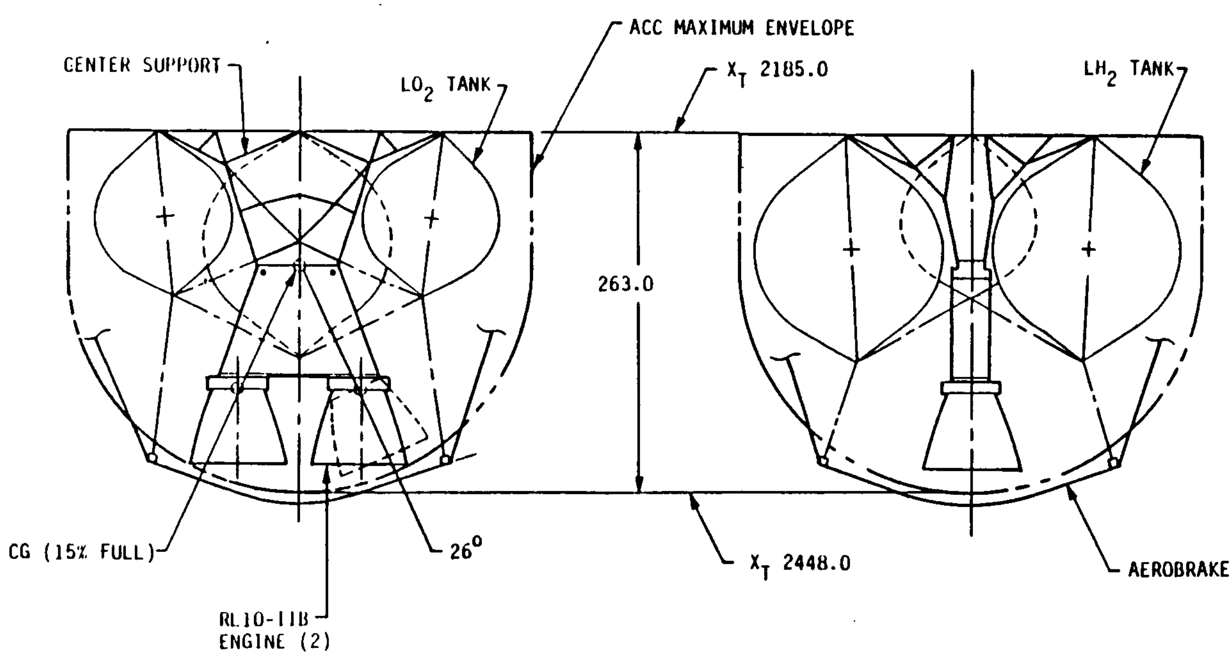

Figure 4.4-2 2-Engine Ground-Based Cryo Packaging

GROUND TO SPACE COMMONALITY — The potential structural commonality between ground and space-based OTV was investigated. The basis for this study was that the ground-based configuration should have one engine and be constrained to fit within the confines of the Aft Cargo Carrier, while the space-based configuration would have two engines. Figure 4.4-3 summarizes a study to determine how much of the ground-based vehicle structure could be used on the space-based vehicle. It was found that only the original center support truss and the structural part of the avionics ring could be counted as truly common. Plumbing attached to the original truss could also be designed to be common. The lower truss and its split plumbing, larger tanks, larger aerobrake and aerobrake supports are all new. Final engine selection and tank size do not affect this result. It was concluded that the space-based structure should be optimized for the space-based application, rather than be compromised to maintain the little commonality that is possible with the ACC constrained single engine configuration.

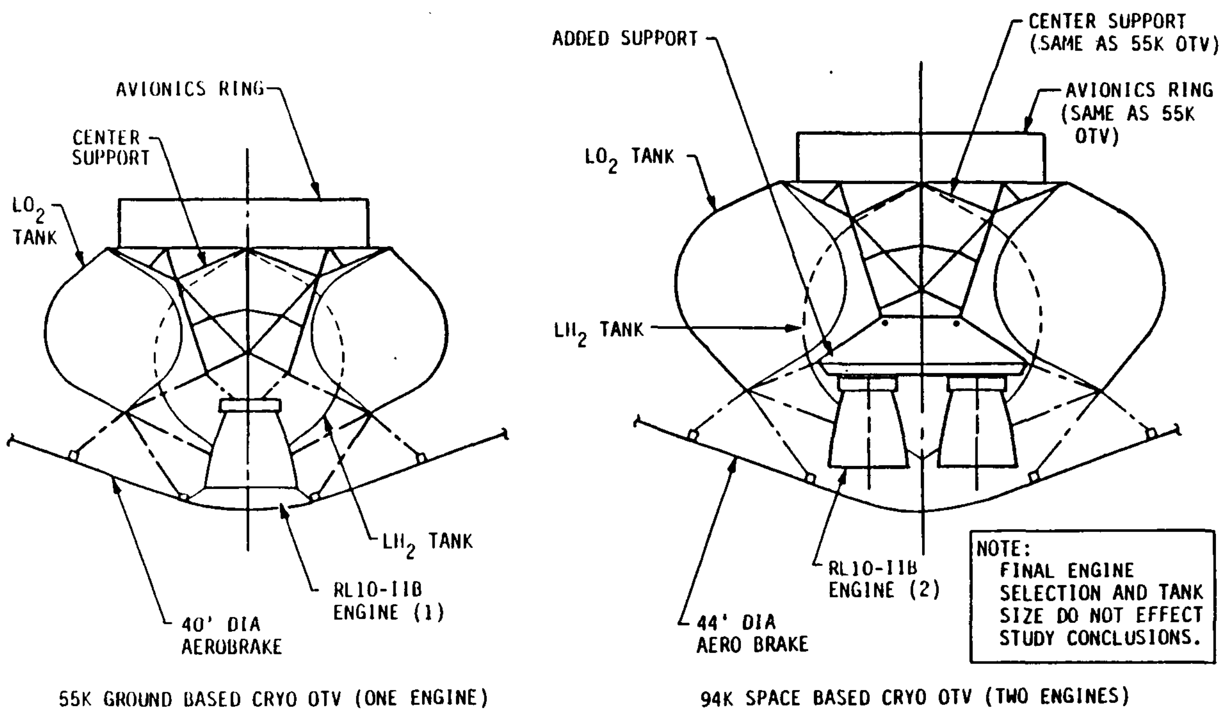

Figure 4.4-3 Ground-Based to Space-Based Cryo Structure Evolution

CARGO BAY PACKAGING — Both the ground-based ACC OTV and the space-based OTV configurations must be accommodated in the cargo bay. In the case of the ground-based ACC OTV, it must be returned to earth in the cargo bay. In the space-based case, it must be delivered to space in the cargo bay and must be returnable to the earth for major maintenance. In both of these cases transport in segments may be acceptable.

In the case of the ground-based ACC OTV, Mr. Larry Edwards of NASA Headquarters, has concieved an efficient approach that was incorporated in the recommended vehicle design. Figure 4.4-4 illustrates the approach. The bulk of the configuration (the primary structure, LOX tanks, avionics, propulsion and attitude control) is configured in a unit that can be stowed longitudinally in the cargo bay. Keel fittings tie the forward OTV structure to the cargo bay, and the forward LOX tank frames fold partway back and are braced (Section A-A) to provide a torsional load path to the cargo bay longerons. Only the hydrogen tanks must be removed from the flight configuration. They are evacuated after flight, removed from the OTV configuration and stowed fore and aft in the cargo bay. These tanks, which weigh approximately 250 pounds each, provide their own torsional strength, and require the support fittings shown in section B-B. The aerobrake is not retrieved for reuse as the material used, while flexible during ascent, is not anticipated to be flexible after use. The structural ASE required to support this retrieval approach is modest and easily stowed in the bay during the ascent portion of the mission.

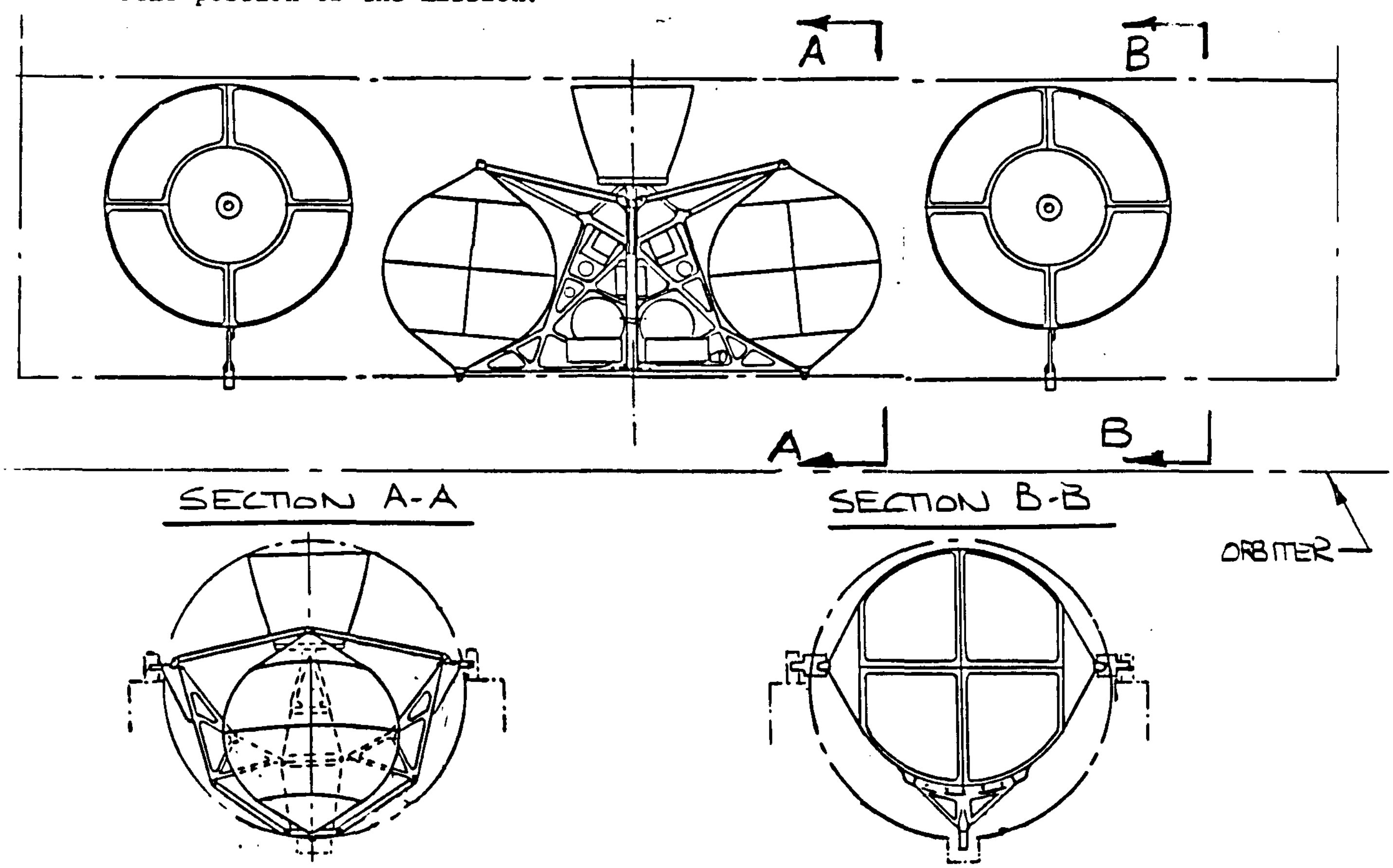

Figure 4.4-4 Ground-Based Cryo ASE

Figure 4.4-5 shows the initial delivery of the disassembled space-based OTV to the space-base. As indicated, all subsystems will fit into the orbiter bay, and delivery will require the equivalent of two shuttle flights volume. since the dry weight of the OTV is on the order of 8000 pounds, it is not advocated that delivery be made in only two flights. Rather, system delivery should be manifested across a larger number of STS flights to achieve full utilization of both Shuttle volume and weight carrying capability.

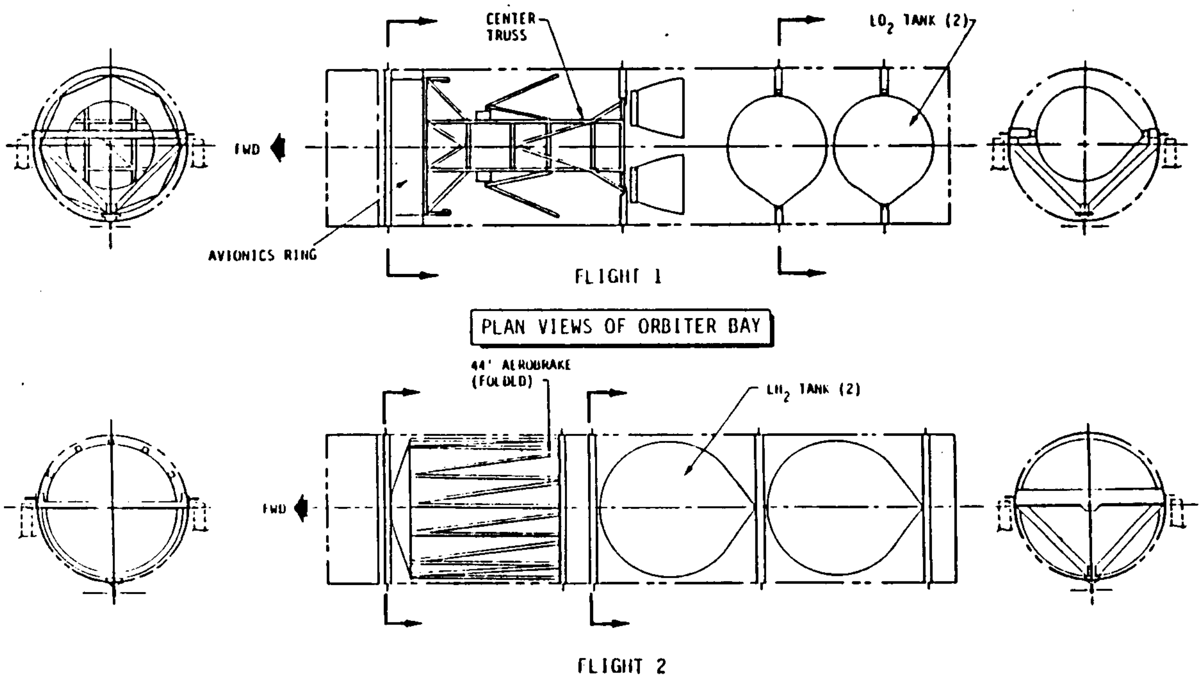

Figure 4.4-5 Cryo Space-Based OTV Delivery

SPACE ASSEMBLY AND MAINTENANCE -- The OTV design has been adjusted to provide for both assembly and maintenance by robotic devices as a primary mode, backed up by remotely operated manipulators and EVA as a contingency mode. Figure 4.4-6 shows the major provisions for grapple fixtures, cradle interfaces and space crane interfaces. These fixtures provide the ability to initially assemble the OTV at the space base, and to perform major parts replacement for maintenance operations. Care has been taken to assure that sufficient space is provided in the vehicle layout to enable these operations with RMS, robotics or space-suited astronauts, as applicable. This requirement has been instrumental in the selection of the open configuration selected for the space-based OTV. Further amplification of this approach is shown in Figure 4.4-7. An octagonal avionics ring was placed at the forward end of the space-based OTVs, providing unobstructed access to all avionics assemblies. The figure shows the locations of these assemblies on the avionics ring. Each replaceable assembly is mounted using MMS type modules that allow removal and replacement with a Module Servicing Tool, which is adaptable to either robotic, RMS or EVA operation. Amplification of assembly and servicing operations is provided in Volume IV.

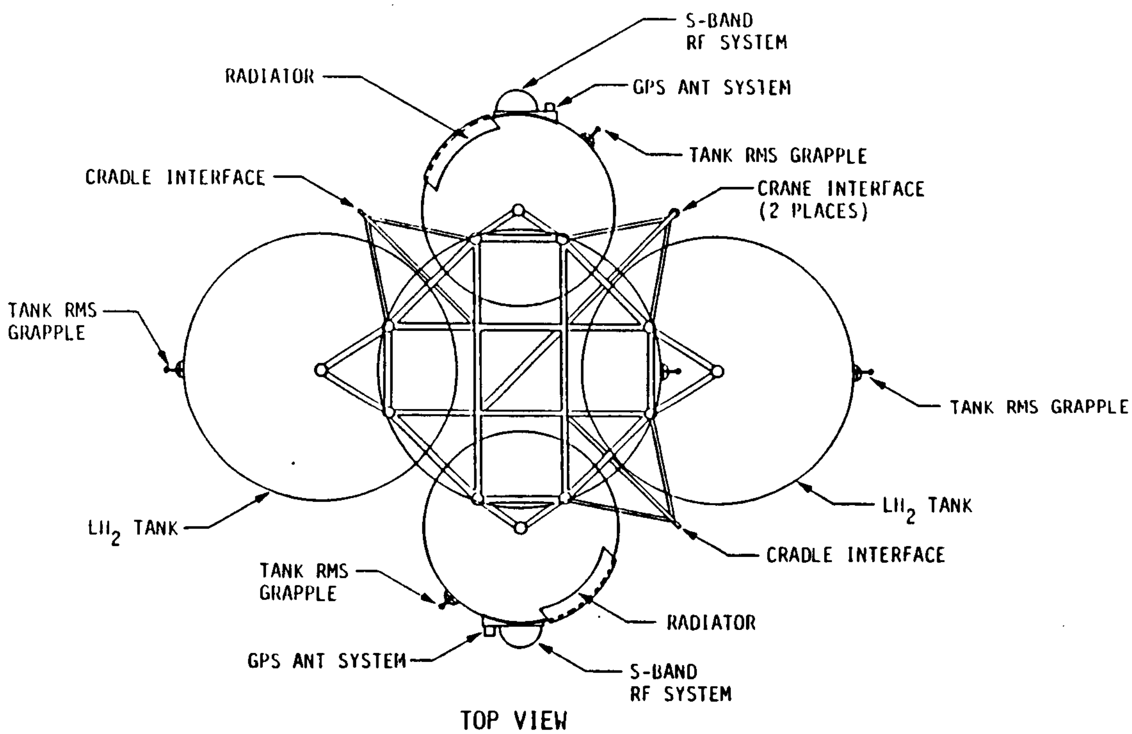

Figure 4.4-6 Cryo Space-Based OTV Subsystem Servicing Locations

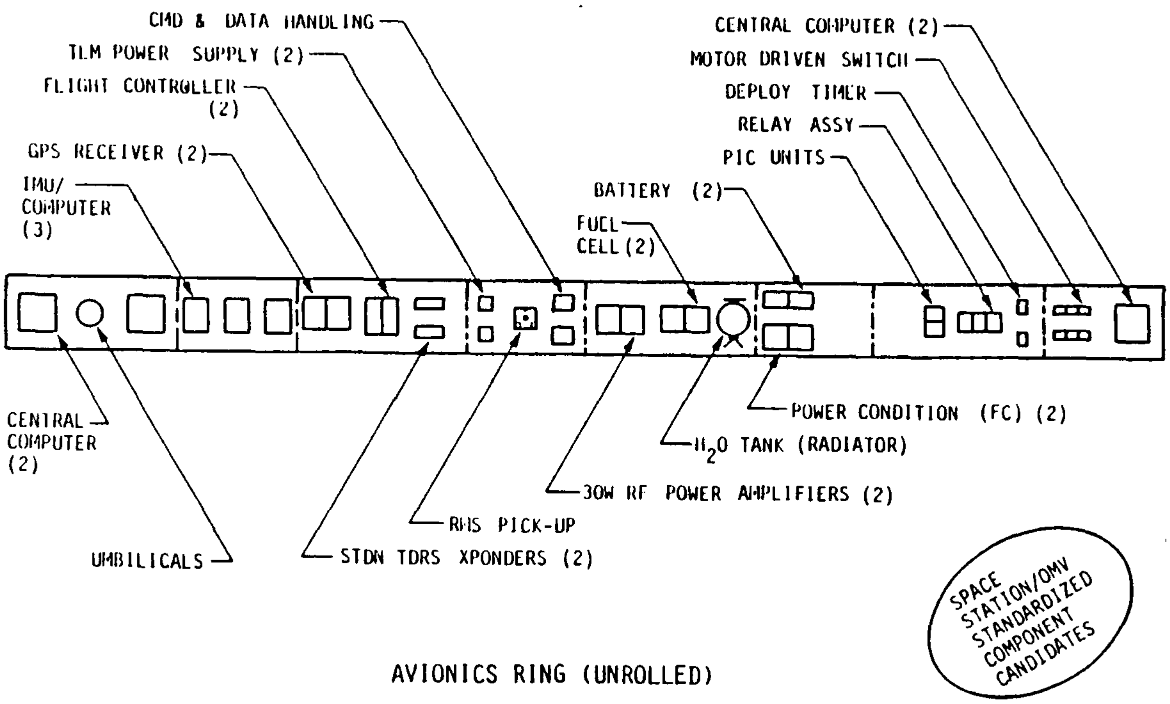

Figure 4.4-7 Space-Based OTV Avionics Servicing Locations

4.5 REDUNDANCY/MAN-RATING

The OTV is to be operated in proximity to the manned Shuttle system from its inception, and is eventually expected to operate in conjunction with the Space Station and to carry men to high orbit. Systems and subsystems must be designed to meet associated safety requirements. In the case of proximity operations, it is necessary to meet the requirements imposed by NASA's safety policy as delineated in NHB 1700.7A. Those systems in use during proximity operations were made dual fault tolerant with respect to credible hardware failures and operator errors. The policy selected for in-flight safety was derived during the course of the study based on the cost of implementing increasingly comprehensive failure policies as illustrated in Figure 4.5-1. Cost increases dramatically with little improvement in system reliability as the most complex policies are implemented. After consideration of these data, NASA direction was to implement a fail safe return policy, where safe return of the crew could be assured in the face of a single failure. This policy must be implemented for manned missions, and may be implemented as much earlier as programmatic considerations indicate to be advantageous.

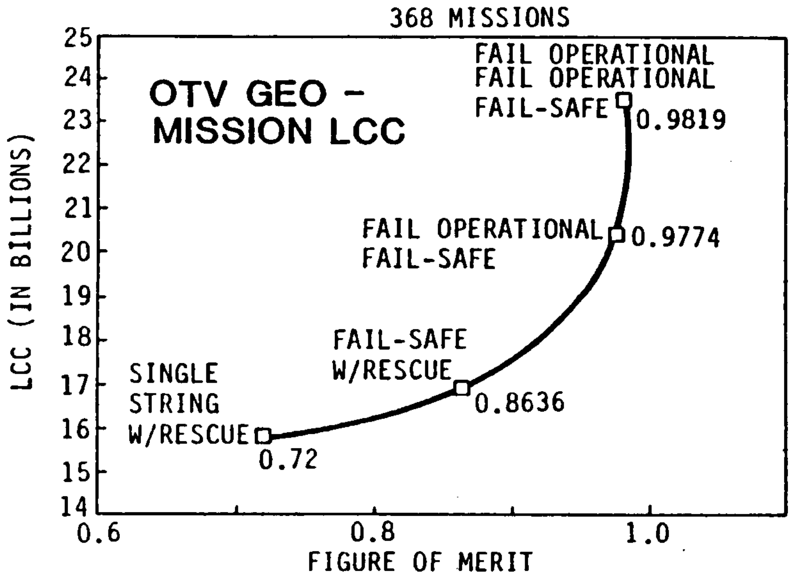

Figure 4.5-1 Redundancy Trade

The fail safe return philosophy is at a point on Figure 4.5-1 that is intermediate between 'fail safe with rescue' and 'fail-op fail-safe'. The system reliability allocation associated with this philosophy was calculated at 0.994 for a 51 hour space mission and 0.946 for a 480 hour space mission. These overall system reliability requirements led to the redundancy levels incorporated in the subsystem designs.

Table 4.5-1 summarizes the results of the various options considered for man-rating the propulsion system. The results show basic trends and ranges, however, they were completed at different levels of maturity in definition of OTV concepts.

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Table 4..5-1 Propulsion System Man-Rating Trade

	OPTION	COST (OPTION COST) / (REF COST)	DRY MASS (LBM) PENALTY	ΔIsp (SEC) (AT OPT 6)	MAINTENANCE SERVICING	MAN-RATING & RELIABILITY	REMARKS
1	1 ENGINE 15000 LBF	REF	REF	REF	SIMPLEST	NOT FAIL-SAFE	o SIMPLE VEHICLE DESIGN AND AEROBRAKE INTERFACE
2	1 ENGINE 15000 LBF BACK-UP TPA	DDT&E 1.05-1.1 UNIT 1.5	240	REF	COMPLICATED BY TPA	FAIL SAFE EXCEPT FOR THRUST CHAMBER AND NOZZLE	o ENGINE DEVELOPMENT TESTING CONCERNS o SIMPLE VEHICLE INTERFACES o ATTRACTIVE APPROACH
3	1 ENGINE 15000 LBF RCS BACK-UP 800 LBF	DDT&E 1.1-1.2 UNIT 1.0	1200 EFFECTIVE (PROP MARGIN)	RCS 440 SEC	COMPLICATED RCS CONDITIONING SYSTEM	FAIL SAFE FOR MANNED MISSIONS	o LOW THRUST GEO DEORBIT o THRUSTER LIFE CONCERNS
4	ONE ENGINE UNMAN 15000 LBF TWO ENGINES MANNED 15000 LBF/EA	DDT&E 1 UNIT 1-2	900	REF	ONORBIT RECONFIGURATION	FAIL-SAFE FOR MANNED MISSIONS	o COMPLICATES DESIGN/DEVELOPMENT o DEVEL. 1 & 2 ENGINES FEED SYSTEM .
5	2 ENGINES 7500 LBF EACH	DDT&E .95 UNIT 1.5	450	-2 TO +2 SEC	2 ENGINES TO MAINTAIN SMALLER VOL. PER ENGINE	FAIL-SAFE ALL MISSIONS	o COMPLEX CONTROLS LARGE GIMBAL ANGLE o LARGE AEROBRAKE DOORS
6	3 ENGINES 5000 LBF EACH	DDT&E .90 UNIT 2.2	750	-2.5 TO +2.7 SEC	3 ENGINES TO MAINTAIN SMALLEST VOL. PER ENGINE	FAIL-SAFE AT LOWER RELIABILITY THAN 5 OR FAIL OP/FAIL SAFE	o MORE COMPLEX CONTROLS o GIMBAL ANGLE SMALLER THAN 2 ENGINES o LARGE AEROBRAKE DOORS

ASSUMPTIONS:
FAIL SAFE - ONE ENGINE OUT MINIMUM REQUIREMENTS FOR MAN-RATING

Option 1 was used as a reference since it represents the minimum unmanned propulsion requirement and high performance.

Option 2 was to back-up the most active component in the engine, the turbopump assembly (TPA). This also included valves and ignitor, but excluded the injector, nozzle, and thrust chamber and might not be considered totally fail safe. It is an approach similar to the Apollo program which had a different requirement.

Option 3 was to use an independent RCS back-up. The common RCS would provide back-up for deorbit from GEO. The propellant margin would be carried as dry mass on the manned mission only, and depended upon the Isp difference, stage/capsule mass, and mission (GEO or Lunar).

Option 4 considered using a second engine for the manned mission while retaining the benefit of the single engine performance for unmanned missions. Sizing the vehicle to accept 1 or 2 engines resulted in a dry mass penalty canceling any benefit.

Option 5 used a two engine vehicle optimized for the GEO delivery missions. This provided fail safe return of the stage for all missions. The vehicle and payload are immediately returned in the case of a failure at perigee. The subsequent burns do not have significant velocity losses for half thrust, therefore, the mission could be completed and the stage returned. For a manned mission, an engine failure would always abort the mission. The engine performance depends upon the area ratio, but in any case there was some performance penalty over a single engine for the two sets of engine data used.

Option 6 investigated adding more than 2 engines. For a single engine out capability, the reliability was lower. For two engine out capability, the stage mass increased without an offsetting increase in performance.

The conclusion was to provide compete engine back-up for the manned GEO and Lunar missions. The back-up TPA is an attractive option because of the design simplification, but the question would be: Is a single thrust chamber, injector, and nozzle fail safe? The RCS back-up requires a large propellant margin and a higher mission loss rate for unmanned missions. The two engine stage was selected because it was fail safe and minimized the performance penalty.

The avionics and power equipment used in the ground and space-based OTV is summarized in the Figure 4.5-2. The component redundancy levels are indicated. The level required for the short duration unmanned ground-based missions is somewhat less than that deemed necessary for the manned space-based vehicles. In the space-based vehicles, we found that the redundancy required by man-rating (a fail safe return philosophy) was somewhat greater than the redundancy suggested by mission 'lost cost' considerations. We elected to incorporate man-rating redundancy in all space-based configurations as indicated in the space-based column of the chart, since our analyses indicated it was not economically desirable to maintain two different avionic configurations in the space-based program. Details on the selection of these subsystems are presented in Vol. II, Book 3.

SUBSYSTEM	COMPONENT	NUMBER OF COMPONENTS GROUND BASED	NUMBER OF COMPONENTS SPACE BASED
GUIDANCE	STAR SCANNER	1	N/A
NAVIGATION	STAR TRACKER	N/A	2
AND CONTROL	IMU	1 DTG	2 RLG
	GPS RECEIVER	1	1
	GPS ANTENNA - LOW ALT	2	2
	GPS ANTENNA - HIGH ALT	1	1
	FLIGHT CONTROLLER	1	2
DATA MANAGEMENT	EXECUTIVE COMPUTER	2	2
	CONDITION MONITOR	N/A	1
TELEMETRY	COMMAND & DATA HANDLING	1	2
AND	TLM POWER SUPPLY	1	2
COMMAND	DEPLOY TIMER	2	N/A
COMMUNICATIONS	STDN/TDRS XPONDER	1	2
AND	20W RF POWER AMP	1	2
TRACKING	S-BAND RF SYSTEM	2	2
ELECTRIC	FUEL CELL (FC)	2	2
POWER	FC RADIATORS	1	2
SYSTEM	EOM REACTANTS TANKS	1 SET	N/A
	FC WATER STORAGE	1	1
	POWER CONTROL & DISTRIBUTION	2	2

Figure 4.5-2 Avionics and Power System Redundancy

4.6 SUBSYSTEM SELECTION SUMMARY

A summary of the subsystem trade studies the results and resulting selections is given in the following pages. The detailed evaluations leading to these results are included in Volume II, Book 3.

AEROASSIST -- The key mission requirement driving aerobrake design for geostationary missions is the weight to be retrieved. The Revision 8 mission model requires return of an empty OTV, return with a 4500 pound manned servicer, or return with a 7500 pound manned capsule. Table 4.6-1 shows aeroheating parameters associated with all the return cases for the ground based OTV using a 38 foot aerobrake. These ground based cases show the trend towards higher temperatures and higher loading as return weight increases. While the 38 foot brake is adequate for the ground based vehicle, we elected to use a 40 foot aerobrake in order to achieve increased design margins. The two space based cases in the table show a minimum sized (38 foot diameter) brake with the early unmanned servicer return mission, and a larger aerobrake with the manned capsule. We elected to use the larger aerobrake throughout the space based program because: We prefer the simplicity of basing only one brake design at the space station; we prefer the increased design margins associated with the larger brake; we like the growth potential for heavier lunar return missions and potentially heavier manned capsules.

Table 4.6-1 - OTV Aerobrake Definition

CONFIGURATION	W/C_DA	BRAKE DIAMETER (FT)	RETURN WEIGHT LB)	q_{MAX}* (BTU/ FT^2SEC)	T* MAX (°F)	DES LOAD (PSF)	
						CTR	OUTBD
GROUND BASED DELIVERY (RETURN EMPTY)	3.20	38	5,700	15.2	2150	22	16
GROUND BASED W/UNMANNED SERVICER	6.13	38	10,970	22.8	2460	35	26
GROUND BASED W/MANNED SERVICER	8.05	38	14,400	25.1	2555	47	35
EARLY SPACE BASED W/SERVICER	6.44	38	11,530	23.4	2485	36	27
GROWTH SPACE BASED W/MANNED CAPSULE	6.00	44	15,670	21.4	2390	34	26

MAN SORTIE - 7,500 lbs CAPSULE, 14 1/2' W x 15' L
* FLEX TPS

PROPULSION -- The major conclusions reached in the propulsion subsystem areas are summarized in Table 4.6-2. The sensitivities shown indicate the factors that drove the decisions. A near term version of the advanced engine was selected to minimize development cost while providing a straight forward evolutionary path to more advanced capability should greater requirements evolve. The man-rating decision, as discussed in section 4.5, was driven by the fail-safe-return man-rating requirement established. Engine thrust, payload mass and number of perigee burns are interrelated as discussed in Section 4.3, and result in the selection reached. The engine throttling requirement is established by the 0.1g payload acceleration mission requirement coupled with the impact of the other system decisions indicated. Engine life reflected a programmatic trade between development cost and frequency of operational replacement. The transition of reaction control system selection is driven by the near term need for economy in the initial ground-based program, and the severe mission requirements imposed by the space-based manned missions.

Table 4.6-2 Propulsion System Conclusions/Recommendations

TRADE/ANALYSIS	CONCLUSION	SENSITIVITY
MAIN ENGINE	IOC ENGINE >475 sec, 5 HRS LIFE, DDT&E $175 M	MISSION MODEL & ENGINE DDT&E vs PERFORMANCE ADVANCE ENGINE PROGRAM SHOULD DEVELOP PROTOTYPE ENGINE
MAN-RATING	2 ENGINES	MANNED SAFETY REQUIREMENTS
ENGINE THRUST	15000 lbf TOTAL, 7500 LBF/EA SINGLE PERIGEE BURN	NUMBER PERIGEE BURNS, PAYLOAD MASS, & NUMBER OF ENGINES
ENGINE THROTTLING	STEP THROTTLING 7.5K ENGINE TO 3.2K	0.1 G LEVEL, PAYLOAD MASS, NUMBER LOW-G MISSIONS, NUMBER OF ENGINES & ENGINE PERFORMANCE
ENGINE LIFE	5 HRS & ORU	REPLACEMENT ON-ORBIT COST & MISSION MODEL SIZE
REACTION CONTROL	HYDRAZINE ON GB COMMON GH2/GO2 ON SB	MISSION MODEL SIZE, GH2/GO2 DDT&E, & DEVELOPMENT OF THRUSTER & COMMON FEED TECHNOLOGY

STRUCTURES -- Configuration and structural trades conducted in this study are described in detail in Volume II, Book 3, paragraph 2.4. Those trades dealing with selection of general arrangement are summarized in paragraph 4.4 of this volume. Other structures trades are summarized in Table 4.6-3.

After a thorough evaluation of available composite and metallic materials, composites were selected for all primary and secondary structural elements, with the exception of propellant tankage. Graphite epoxy was selected for structures below 300 degrees Fahrenheit because of its light weight and ease of fabrication. Graphite polymide was selected for structure above 300 degrees Farenheit, which is required to support the aerobrake thermal shield. This material is able to retain strength at a temperature of 600 degrees Fahrenheit, which in turn establishes thickness requirements on insulation. We selected 2090 aluminum/lithium alloy for cryogenic tanks. This material is expected to display the excellent low temperature and weldability characteristics of the 2219 alloy used for the external tank, while providing significantly lighter weight. We selected 15(V)-3(Cr)-3(Al)-3(Sn) titanium for storable propellant tankage. This is a new alloy that will require further testing but it displays encouraging initial results relative to forming and repairing welding when compared to 6AL4V titanium.

The OTV configurations developed provide adequate protection against the anticipated meteoroid environment. A 0.006 aluminum meteoroid bumper with MLI tank thermal insulation serving as a particle catcher was found adequate for the space based vehicle. The shorter duration ground based missions resulted in a vehicle that was adequately protected with the MLI alone. Subsequent analyses conducted in the extension study indicate that a bumper is required on the ground based vehicle, and a thicker bumper is required on the space based vehicle to provide added protection against the debris environment being defined for space station design. This refinement is not reflected in this volume.

Additional studies were made to establish the appropriate umbilical arrangement and structural interface with the Aft Cargo Carrier. The top level results of these studies are indicated in Table 4.6-3, and more detail is provided in Volume II, Book 3 as indicated.

Table 4.6-3 Structural Design Trade Summary

STRUCTURAL TRADE	KEY ISSUES	RECOMMENDATION	REF
COMPOSITE SELECTION	o OPERATING TEMPERATURE o STRUCTURAL CHARACTERISTICS o FABRICATION CONCERNS	o USE GRAPHTE POLYMIDE FOR AEROBRAKE o USE GRAPHITE EPOXY FOR BASIC STRU.	VOL. II BK. 3 PAR. 2.4.4
METAL SELECTION	o LOW TEMPERATURE STRENGTH TOUGHNESS o PROPELLANT COMPATIBILITY o FABRICATION CONCERNS	o USE 2090 AL/LI FOR CYRO TANKS o USE 15-3-3-3 TITANIUM FOR STORABLE PROPELLANT TANKS	VOL. II BK. 3 PAR. 2.4.5
METEROID PROTECTION	o METEROID ENVIRONMENT o PROTECTION CRITERIA o WEIGHT & VOLUME o FABRICATION	o USE NO PROTECTION ON GND BASED (SUBSEQUENTLY CHANGED FOR DEBRIS) o USE ALUMINUM BUMPER & MLI FOR CATCHER ON SPACE BASED	VOL. II BK. 3 PAR. 2.4.10
UMBILICAL LOCATION	o INSPECTABLITY o USE ET UMBILICAL (ICD80900000025) o ACC CRUCIFORM CROSSBEAM (DWG826AP00231) o STRUCTURAL INTEGRITY AND PHYSICAL FIT	o PENETRATE ACC SKIRT TO PLATES AT CRUCIFORM & LOX TANKS o SEPARATE FLUID & ELECTRICAL UMBILICALS	VOL. II BK. 3 PAR. 2.4.3
ACC BEAM STIFFNESS	o MAXIMIZE PAYLOAD OPTIMUM ACC BEAM (GROUND BASED ACC/OTV)	o 25.5 INCH DEEP PARALLEL CRUCIFORM BEAMS SAVES 18# IN OTV, 110# IN ACC RELATIVE TO ORIGINAL TAPERED BEAMS	VOL. II BK. 3 PAR. 2.4.1
ACC STRUCTURAL ATTACHMENT	o ACC/OTV STRUCTURAL I/F o RESTRAINTS AT 4 ATTACHMENTS o WEIGHT & DEFLECTION	o USE 10 DOF RESTRAINT (TRADED WITH A DEF) o ADD LATERAL RESTRAINT @ LH2 TANK o SAVES 75# & REDUCES DEFL. 2"	VOL. II BK. 3 PAR. 2.4.2

AVIONICS -- Figure 5.6-4 summarizes trade studies performed for all major functional elements in each of the five avionics subsystems: 1) GN&C, 2) DMS, 3) C&T, 4) T&C, and 5) EPS. No unusual results were obtained as a result of these trades. Basic technology advances in disciplines supporting avionics hardware such as microelectronics, opto-electronics, semiconductors, and computer architecture will ensure a continued growth in capability and reliability while maintaining relatively low cost between the present and the OTV Phase B/C/D period. The inertial guidance system for the ground-based OTV could make use of the ring laser gyro technology should schedules push the IOC farther out. The use of redundant, propellant grade reactant fuel cells was selected to reduce logistics and maintenance costs, among other factors, while offering sufficient reliability for the DRMs. Individual electronic subsystems, such as memories, are equipped with built-in battery back-up power. While not mandatory, GPS and TDRS improvements would enhance the OTV program. Addition of an aft (upward) looking antenna on GPS would significantly improve gain margins in obtaining GPS updates at GEO altitude. While TDRS coupled with ground coverage provides adequate OTV command capability, use of a third TDRS and increase of its azimuth steering angle to $\pm 45^{\circ}$ would significantly improve TDRS coverage in the absence of support from ground stations.

Table 4.6-4 Avionics Trade Summary

OTV RF COMMUNICATIONS ALTERNATIVES	o BALL ELECTRICALLY SWITCHED STEERABLE ANTENNA o 20W RF POWER AMP PREFERRED
GN&C STATE VECTOR UPDATE	o GPS IS THE PREFERRED OPTION FOR ALL STATE VECTOR UPDATES
MICROPROCESSOR / MICROCOMPUTERS	o FAIRCHILD 9450 PREFERRED ARCHITECTURE
CENTRALIZED vs. DISTRIBUTED COMPUTER DATA MANAGEMENT	o DISTRIBUTED, NETWORK ARCHITECTURE PREFERRED o DELCO MAGIC V PREFERRED EXECUTIVE COMPUTER
ON-BOARD vs. GROUND CHECKOUT	o ON-BOARD CHECKOUT PREFERRED o STRENGTHENS AUTONOMOUS CHARACTER
BUILT-IN vs. MULTIPLE UNIT AVIONICS BLACK BOX REDUNDANCES	o BUILT-IN, LAYERED FAULT-TOLERANCE APPROACH REDUCES BOX REDUNDANCIES
ELECTRO-OPTICAL NAVIGATION SENSORS	o SOLID STATE STAR TRACKER PREFERRED (GB USES EARLIER STAR SCANNER TECHNOLOGY)
ELECTRICAL POWER GENERATION TECHNOLOGY	o FUEL CELL PREFERRED
GYRO TECHNOLOGY	o RING LASER GYRO PREFERRED OVERALL o DRIRU SUITABLE FOR NEAR TERM USE

5.0 CONCEPT SELECTION

5.1 HIGH POTENTIAL CRYOGENIC SELECTION

An initial concept selection was made at contract midterm to accommodate the 'nominal' Rev. 7 mission model. Subsequent to this selection, MSFC produced a Rev. 8 mission model and directed that development recommendations be justified by the 'low' version of this model. We found that while the 'nominal' Rev. 7 model suggested these OTV development steps (a ground-based vehicle, an initial space-based vehicle, and a growth space-based vehicle), the 'low' Rev. 8 model could be accommodated by the first two of these steps.

Figure 5.1-1 shows the family of cryogenic stages we recommended to capture the Rev. 7 nominal mission model. The ground-based stage is sized at 45,000 pounds propellant capacity to fully utilize STS payload capability when launched in the aft cargo carrier. It would be used to perform single and multiple delivery missions until the initial space-based configuration was to be introduced. We recommended that this stage employ a single 7500 pound thrust advanced technology engine. The configuration is tightly packaged to fit assembled in the aft cargo carrier, and uses a foldable 40 foot diameter fabric covered aerobrake. The aerobrake is designed to support empty stage return at a maximum surface pressure of 23 psf.

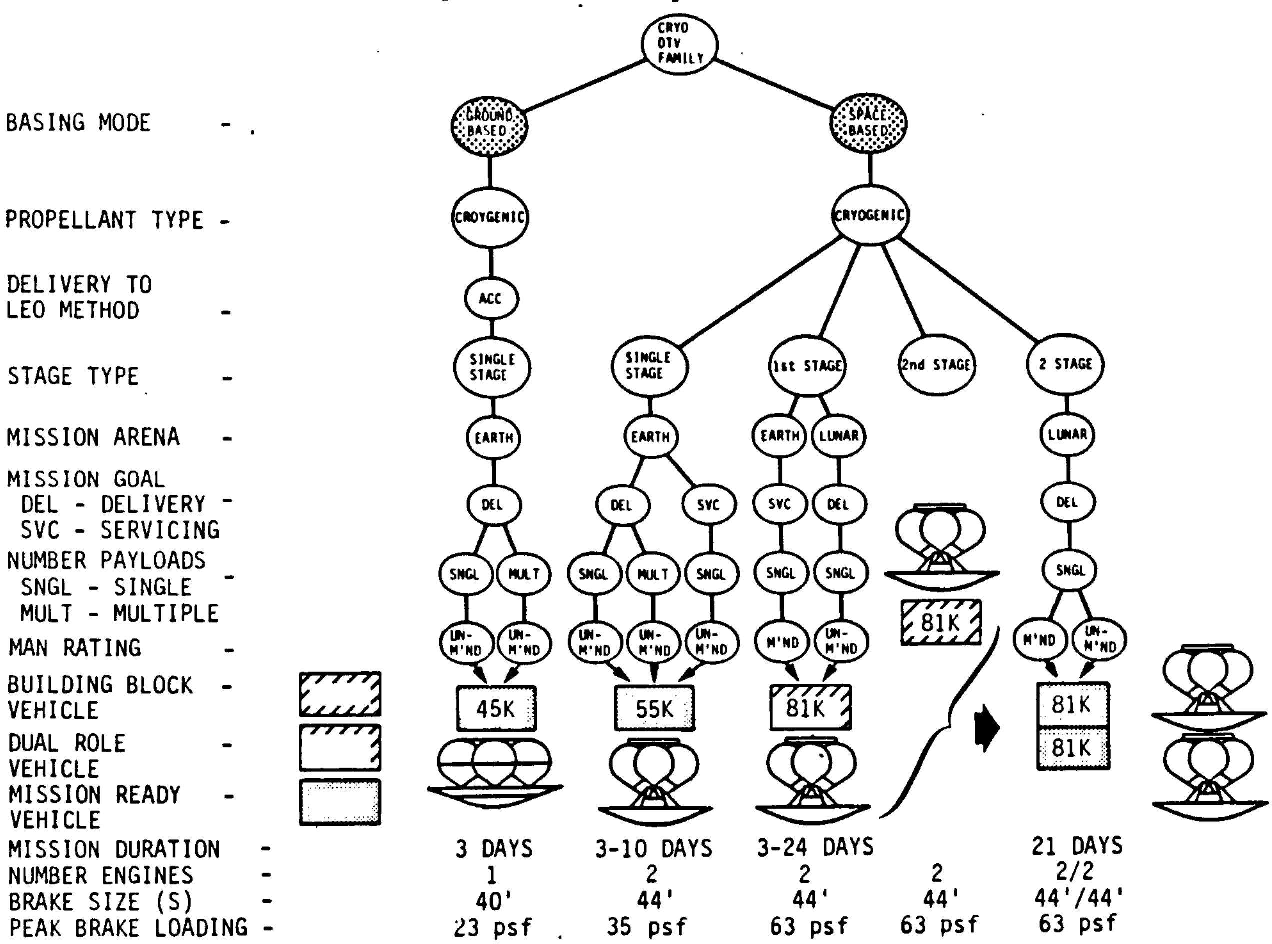

Figure 5.1-1 Cryo Configuration Summary

The initial space-based configuration is derived from the ground-based stage. Its 55,000 pound propellant load was selected to support the driving 20,000 pound delivery mission. It utilizes two engines of the same type, and most of the same avionics components, as the ground-based vehicle. We believe that the general arrangement must be opened up to facilitate maintenance in space. Mission duration is increased to 10 days to support the unmanned servicing mission. A 44 foot aerobrake is required to protect the open configuration, and its 35 psf design pressure supports return with the unmanned servicer.

A growth stage would have been required to support the Rev. 7 manned GEO mission and the larger lunar missions. Both the initial and the growth stages would have been maintained at the Space Station from IOC of the growth stage throughout space-based operations. The growth stage's 81,000 pound propellant load was selected to support the driving Manned Lunar Sortie in a two stage configuration. This is slightly larger than the 75,000 pound load that would have been required to perform the Revision 7 Manned GEO Sortie, but our preliminary programmatic trades indicate the selection of the slightly larger stage would have been cost effective. The mission duration of the growth stage would have been up to 24 days as required by the 14,000 pound Revision 7 Manned GEO Sortie. A 44 foot aerobrake designed for a 63 psf peak pressure enable return with a manned capsule. The two-stage configuration would have been required to support the Manned Lunar Sortie, where 65,000 pounds of payload is delivered in conjunction with a 15,000 pound roundtrip manned sortie.

As previously noted, only the ground-based stage and the initial space-based stage are required to perform the 'low' Revision 8 Mission Model. All three configurations are described in Section 6.1.

5.2 HIGH POTENTIAL STORABLE SELECTION

Figure 5.2-1 is a pictorial presentation of the complete storable OTV family of high potential stages that was selected to perform the 'nominal' Rev. 7 OTV mission model. This selection was not updated to meet the requirements of the "low" Rev. 8 model because it was programmatically demonstrated (including operational and space-basing impacts) that the storable concepts were less desirable than the cryogenic concepts, even with the low use rates involved in the "low" Rev. 8 model. This family of stages will perform the missions identified in the Rev. 7 model with the exception of the heavy lunar missions in the post 2006 timeframe. We proposed to capture those missions using a low technology cryo perigee stage. Two configurations for the early ground-based OTV are defined on the left of the figure; one carried aloft in the ACC and the other configured to fit in the Orbiter payload bay. Both are sized to take advantage of the total lift capability of the STS in the early 1990s and are outfitted to deliver unmanned single or multiple payloads, as identified in the mission model, to GEO operating as a perigee stage. The space-based family is built around three stages with propellant capacities carefully selected to most efficiently perform the broad range of identified missions. The 53,000 lb capacity stage is the workhorse configuration which has application in all GEO missions. Operating as a perigee stage it is the GEO delivery vehicle for single and multiple

6.0 SELECTED CONCEPTS DEFINITION

6.1 HIGH POTENTIAL CONCEPT DEFINITION — CRYO

6.1.1 INITIALLY GROUND BASED CRYO

6.1.1.1 General Arrangement (Ground Based Cryo) - The overall concept of our selected ground-based cryogenic OTV is shown in Figure 6.1.1.1-1, and a more detailed layout in Figure 6.1.1.1-2. The four tank, single advanced technology engine configuration uses the volume and weight efficient principles suggested by Larry Edwards (NASA Headquarters) to fit easily into the Aft Cargo Carrier (ACC). The 40 foot diameter aerobrake folds forward while stowed in the ACC. It is discarded after flight and not stowed in the orbiter bay for retrieval. The aluminum/lithium propellant tanks are designed by engine inlet pressure requirements. Their thinnest gauges are .018 in. for the LO_2 tank and .014 in. for the LH_2 tank. The tanks are insulated with multi-layer insulation. The hydrogen tanks are removed onorbit after mission completion and, with the core system (LO_2 tanks, structure, avionics, propulsion), are stowed in the orbiter bay for retrieval. The propulsion and avionics subsystems are mounted on the central truss, and reflect essentially a single string design. The major exception is redundancy in those systems that require dual fault tolerance while in the vicinity of the Orbiter. The structure is of lightweight graphite epoxy. The propellant load was selected to enable full utilization of projected STS lift capability on GEO delivery missions.

6.1.1.2 Subsystem Summary Description (Ground-Based Cryo)

6.1.1.2.1 Aeroassist (Ground Based-Cryo) - The overall layout of the ground-based cryo OTV aeroassist device is shown in Figures 6.1.1.1-1 and -2. Details of the construction of its surface insulation and the parameters influencing its design are shown in Figure 6.1.1.2.1-1. The basic shape of the aeroassist device is the 70 degree blunted cone proven on the Mars Viking lander. The 40 foot diameter device is designed to retrieve a nearly empty OTV from geostationary transfer orbit. Its center of gravity is offset to cause it to trim out at a 0.12 lift/drag ratio. This has been shown adequate to provide trajectory control when used with a roll modulation control technique. The 40-foot diameter was selected to provide adequate shielding of the OTV from the aerodynamic wake, for trim angles up to an L/D of 0.20. This size, the weight of the OTV at reentry and the physical properties of the aerodynamic surface establish the temperatures and heat fluxes shown in Figure 6.1.1.2.1-1.

The outer portion of the shield folds forward to fit within the Aft Cargo Carrier, and is constructed of the flexible, multilayer material shown in Figure 6.1.1.2.1 backed by graphite polyimide ribs. These ribs can tolerate temperatures up to 600 degrees Fahrenheit, which establishes the thickness of the insulation shown in the figure. The multilayer Flexible Surface Insulation (FSI) outer layer is a woven Nicalon (silicon carbide) fabric which can tolerate high heating rates without becoming brittle. A three dimensional woven structure between the inner and outer surfaces is filled with a ceramic felt insulation. The inner layer is NEXTEL 312, which has superior mechanical properties, impregnated with an RTV gas sealer.

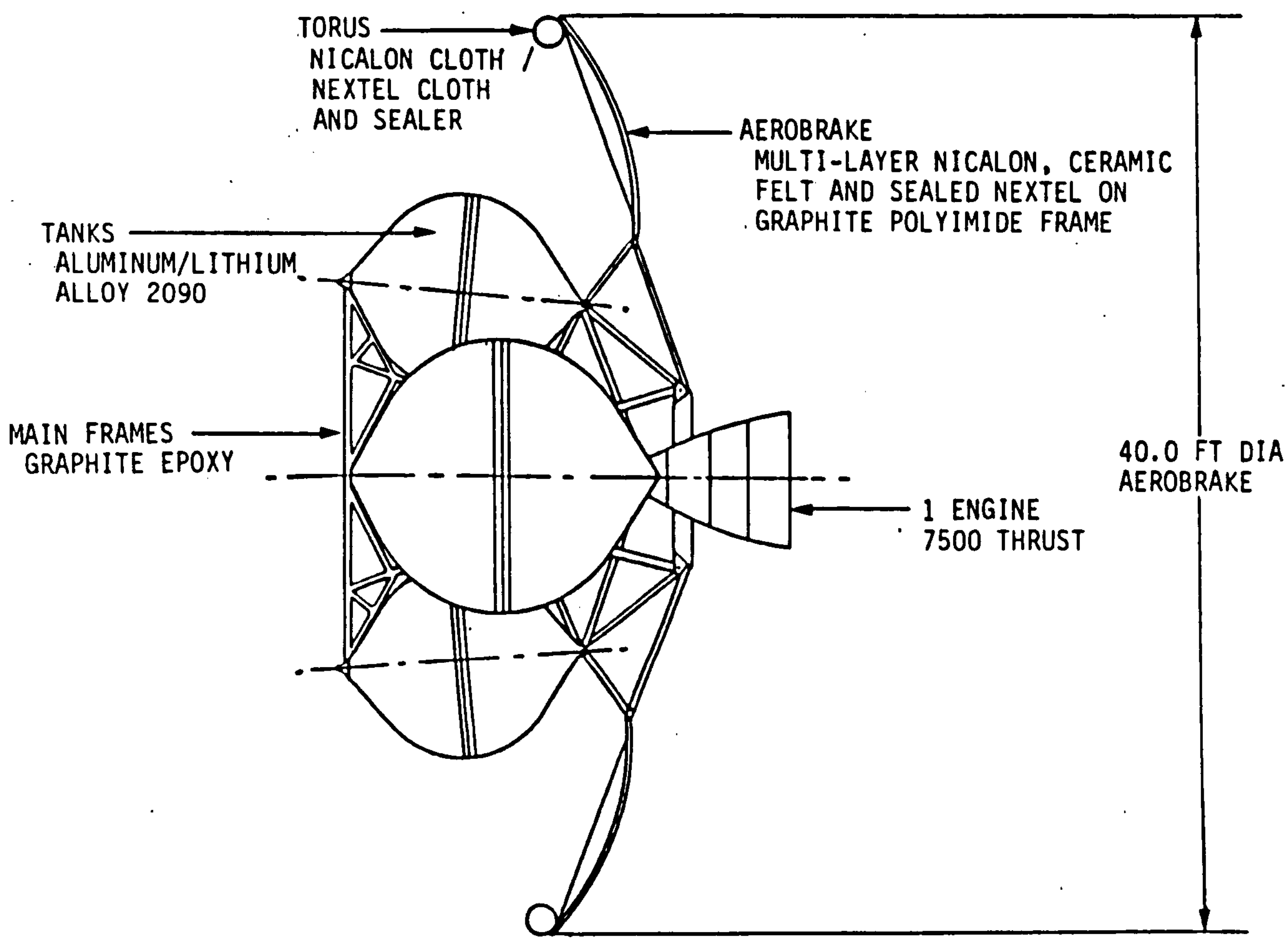

Figure 6.1.1.1-1 Ground Based Cryogenic OTV Concept

payloads. For unmanned roundtrip servicing missions, the 53,000 lb stage is combined with the 25,000 lb stage to form a two stage vehicle. For the demanding requirements of the manned trips to GEO, the 53,000 lb stage is mated to the 90,000 lb stage to form another two-stage configuration. The 53,000 lb stage will be fitted with aerobrakes appropriately sized for the size and weight of the body being returned from GEO. When only the stage is returning, as from delivery missions, only a 25-foot diameter brake is required. When returning from delivery of multiple payloads, the multiple payload carrier returns with the stage and therefore, the required brake size is 32 feet in diameter. Bringing the manned capsule back from the manned servicing mission is the most demanding mission for the 53,000 lb stage and requires a 41-foot diameter brake. The 90,000 lb stage is sized for the first stage application on the manned servicing vehicle; however, it will be the primary vehicle for performing the planetary missions in the mission model. Some of the less demanding planetary missions can be performed by the 53,000 lb and the ground-based stages. The identified application for the 25,000 lb stage in the current mission model is for the second stage of the two-stage unmanned servicing vehicle.

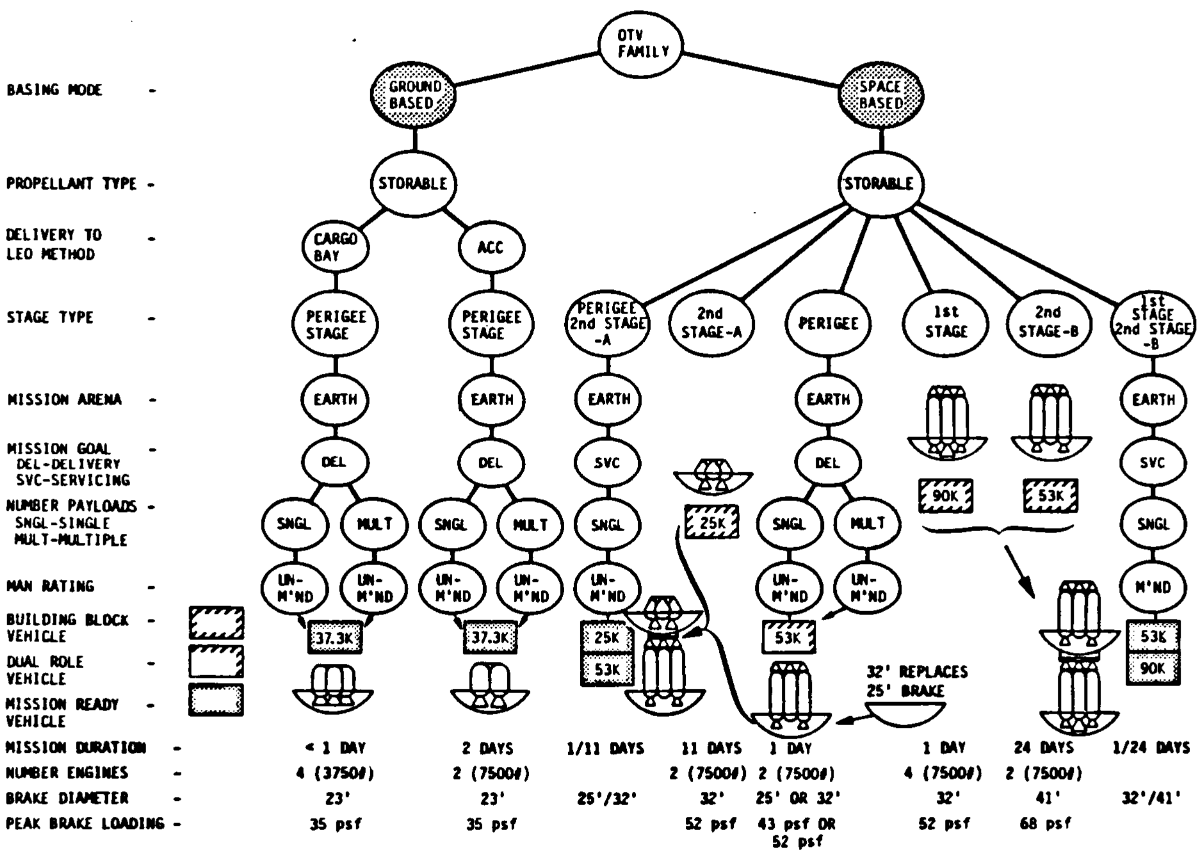

Figure 5.2-1 Storable Configuration Summary

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CONFIGURATION	W/C_DA	BRAKE DIAMETER (FT)	TPS	q_{MAX} (BTU/FT^2 SEC)	q_{MAX} (BTU/FT^2)	T_{MAX} ($^\circ$F)	TPS THICKNESS (IN.)	W_{BRAKE} / W_{RETURN}	DESIGN LOAD (PSF) CENTER	DESIGN LOAD (PSF) OUTBOARD
GROUND BASED	3.7	40	FSI RSI	17.9 21.5	2650 3180	2230 1970	0.34 0.39	0.19	23	17

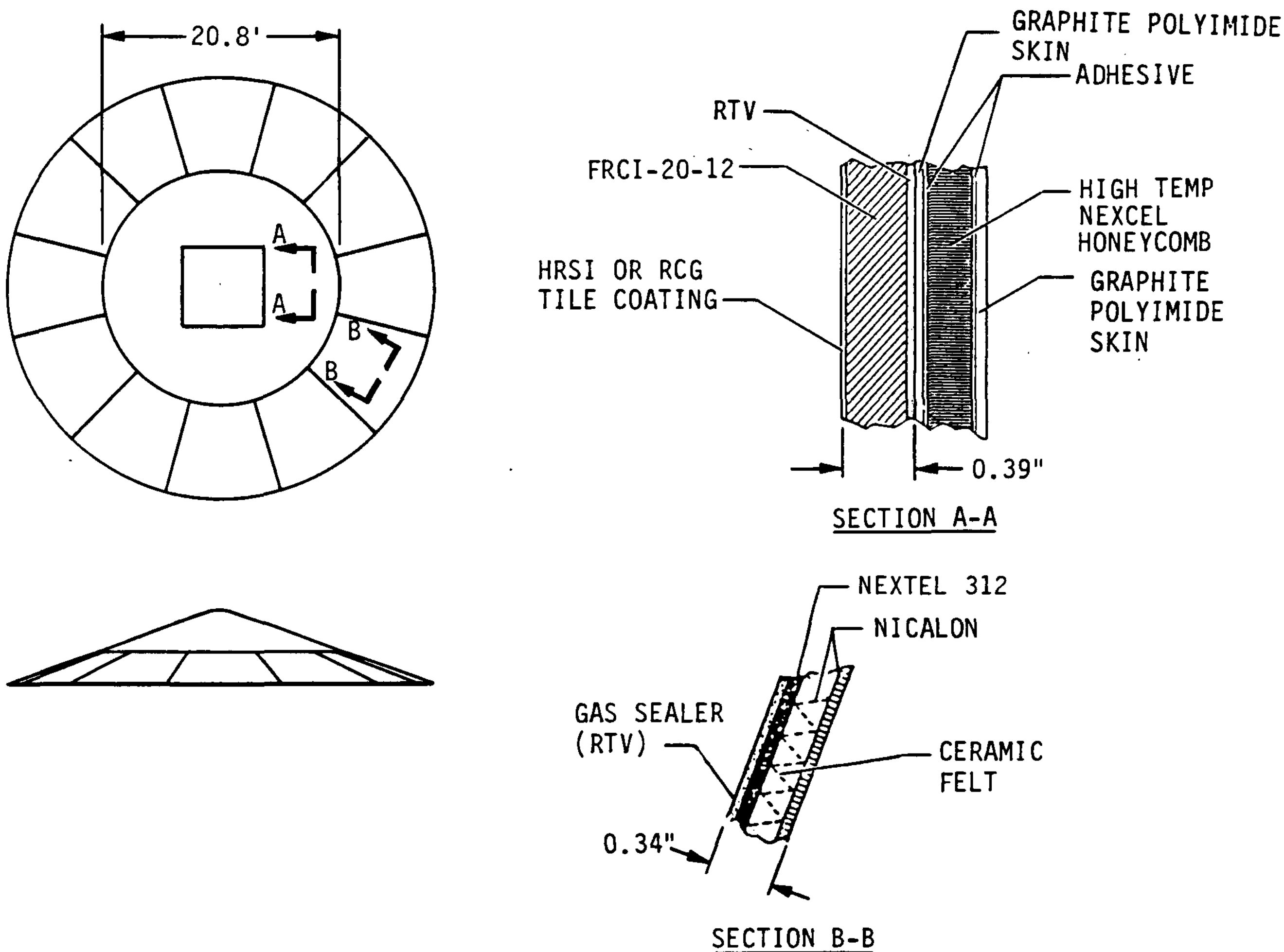

Figure 6.1.1.2.1-1 Ground-Based Cryo Aeroshield

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6-1 to 6-3

The central section of the aeroassist device is rigid and is covered with the same type of ceramic tiles used on the Orbiter. The temperatures associated with the low ballistic coefficient of this configuration are low enough to permit use of the Flexible Surface Insulation (FSI), but the current design uses Orbiter tiles instead. A door is provided in the central heat shield so the two step engine nozzle can be deployed through it. Figure 6.1.1.1-2 shows an FSI door that is rolled back from the opening, but a rigid door has been selected to maintain the blunted 70 degree cone basic geometry.

6.1.1.2.2 Propulsion (Ground Based Cryo) - The propulsion characteristics are shown in Table 6.1.1.2.2-1 for the main propulsion and Table 6.1.1.2.2-2 for the reaction control system. The main engine is a single 7500 lb thrust expander cycle, which can be available in 1993 with an accelerated development program. Current advanced engine research funding is scheduled to demonstrate the required technology by 1990. These research engines would then reduce the risk of a development program which is estimated to take 5-7 years. To provide an evolutionary path from ground-based to space-based, we recommend accelerating this program and the development of a new engine for the ground-based OTV. This could be accomplished with a phased program, i.e., a lower technology engine initially that has the capability to evolve to a space-based design. The engine Isp would be 475 (minimum). The engine can provide propulsive settling of propellants by operating at tank head idle (THI-without rotation of turbo machinery). It can operate with saturated propellants at pumped idle (PHI) thrust level and low NPSH at full thrust level. Capacity for both hydrogen and oxygen autogeneous tank pressurization is provided by the engine. The nominal inflight tank operating pressures are 26 psia and 21 psia for LO_2 and LH_2, respectively. This allows for the negative acceleration head associated with the siphon feed.

The advanced engine has a goal of 300 -500 firings and a time between overhaul of 10 - 20 hours. However, the initial engine would be qualified for only about 5 hrs. The engine development time is 60 months from start of development to the first operational engine. Dual engine installations were also evaluated for the ACC ground-based cryogenic OTV. Layout studies performed at MAF indicated that additional length would be reqired for the ACC in order to accommodate 2 RL10-IIB or RL10-III engines with the capability to gimbal through the center of gravity and provide a fail safe engine capability. This additional length could not be obtained. This study should be revisited for the small advanced engine, including impacts to cargo bay return.

Propellants are stored in a four tank configuration, two liquid oxygen and two liquid hydrogen. The tanks are manifolded together in a parallel flow configuration so that propellants will be depleted simultanously from each of the two tanks. In order to deplete the parallel tanks and both propellants simultaneously, a propellant utilization system is included. This system consists of propellant utilization probes that provide continuous liquid level data during main engine firings and discrete point level sensors that provide data to allow the cancellation of cumulative errors in the continuous mode when the liquid level passes the discrete point sensor. The data from each tank is input to the stage computer. The computer outputs signals to either

Table 6.1.1.2.2-1 Ground-Based Cyrogenic MPS Summary

o	ENGINE	- SINGLE ENGINE, 7.5K THRUST, I_{SP} = 475 SEC, EXPANDER CYCLE
o	PROPELLANT DISTRIBUTION	- DUAL TANK PARALLEL FEED START TRAP
o	PRESSURIZATION	- AUTOGENOUS FROM ENGINE FOR PUMPED IDLE AND FULL THRUST - NOT REQUIRED FOR TANK HEAD IDLE
o	ENGINE FEATURES	- TANK HEAD IDLE (THI) CONDITIONING AND SETTLING, PUMP HEAD IDLE (PHI) FOR LOW THRUST APPLICATIONS 5 HR LIFE
o	VENT	- GROUND/ASCENT - "0"G (TVS)
o	VALVE ACTUATION	- HELIUM, STORED ON STAGE
o	PROPELLANT UTILIZATION	- TANK TO TANK AND MR CONTROL
o	CARGO BAY RETRIEVAL	- SEPARATION OF LH_2 TANKS
o	THERMAL PROTECTION	- H_2 - 1/2" SOFI 1/2" MLI (25 LAYERS DAK) - O_2 - 1/2" MLI (25 LAYERS DAK)
o	STS PROXIMITY OPERATIONS	- TWO FAULT TOLERANT
o	REDUNDANCY	- SINGLE FAILURE TOLERANT EXCEPT FOR ENGINE

Table 6.1.1.2.2-2 Ground-Based RCS Summary

- o PROPELLANT
  - Hydrazine (N_2H_4)
- o ROCKET ENGINE MODULE
  - 30 LB, 7 ENGINES PER MODULE — I_{SP} = 230 SEC
  - 14 THRUSTERS SCARFED INTO AEROBRAKE
  - 3 DOF and +X TRANSLATION
  - FAIL OPERATIONAL
- o PROPELLANT SUPPLY
  - THREE 24" DIAMETER TANKS
  - POSITIVE EXPULSION — 400 LBS OF HYDRAZINE MAXIMUM
  - 400 PSI 2:1 BLOWDOWN
- o SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR STS PROXIMITY OPERATIONS

the tank propellant utilization valves to keep liquid levels in each tank the same or to the engine to shift mixture ratio to assure simultaneous depletion of usable propellants. Refillable traps are included in the outlet of each tank to provide liquid propellants for the chilldown until the remaining propellants are settled over the tank outlet by the engine thrust.

The tank vent system consists of both ground vents for loading and low gravity vent systems for flight operations. The flight system uses a mixer and a thermodynamic vent heat exchanger to minimize the operation impacts for in flight venting and reduce propellant thermal stratification. The mixer and heat exchanger similar to the STS Centaur are mounted inside the tanks. The controls and magnetically coupled drive motors are on the outside so that they can be serviced without entering the tanks.

High pressure helium is stored at 3000 psi in a composite overwrapped vessel for MPS engine purges and valve actuation. The stage propellant system valves are also pneumatically activiated.

The ground-based cryogenic OTV must be disassembled for return to the ground in the orbiter cargo bay because of its size. The design includes the removal of the LH_2 tanks to provide for sufficient clearance in the cargo bay for the remainder of the stage.

The residual propellants (up to 1.5%) will be burned and dumped in a nonoptimum burn after the perigee raising burn that follows the aeropass maneuver. During that burn, the remaining propellant quantity will be determined to an accuracy of 0.25% so that nonoptimum trajectory burn times can be calculated. The first burn of this maneuver will utilize the MPS engine to consume some portion of the residuals. The second burn will start with the MPS engine and finish with an RCS vernier burn during which the remaining propellant, approximately 260 lbs, will be dumped through 2.5" dump valves in the MPS plumbing system.

This complex propulsive dumping maneuver is required to dump the residual propellants without freezing the residual hydrogen. If the hydrogen were dumped nonpropulsively about 70% of the residuals could freeze when the triple point pressure for hydrogen (1.0 psia) was reached. LO_2 is not as prone to freezing because it has a triple point pressure of about 0.022 psia. Before we selected this propulsive mode of operation we looked at several alternatives which are discussed in Volume II, Book 3.

The thermal protection system is 0.5 inches of SOFI and 0.5 double aluminized Kapton multilayer insulation (MLI) on the hydrogen tank and 0.5 inch MLI on the oxygen tank. Both insulation systems are purged on the ground with low dew point nitrogen to eliminate moisture contamination.

The system is two fault tolerant for inadvertent RCS and main engine ignition for proximity operations near the orbiter.

The system is single failure tolerant except for the engine itself as shown in the MPS schematic, Figure 6.1.1.2.2-1. Figure 4.1.2.2-2 shows the legend that identifies the components shown in the propulsion schematics. The pneumatic system is not shown in Figure 6.1.1.2.2-1.

The reaction control system (RCS) uses hydrazine monopropellant pressurized by nitrogen gas operating in a blowdown mode from 400 psi. Fourteen (14) thrusters provide 3 degree of freedom operation and +X translation. The thrusters are 30 lbf each and are clustered with seven (7) thrusters in each module. The thrusters provide an Isp of 230 seconds. The propellant is stored in three 24-inch diameter tanks, each having a usable propellant capacity of about 133 lbs of hydrazine at a 2:1 blowdown. The RCS is two fault tolerant for proximity operations as shown in Figure 6.1.1.2.2-3.

6.1.1.2.3 Structures & Packaging (Ground-Based Cryo) - The configuration, shown in Figure 6.1.1.1-2, consists of two 132 in. diameter spherical LH_2 tanks with cone ends and two 93 in. diameter spherical LO_2 tanks with cone ends and one advanced design engine that generates 7500 lbs of thrust. The engine and the lower support for the four tanks is provided by a central core truss that also provides the interface at four points with the ACC. Upper ends of the tanks are linked together and tie to the upper part of the truss at the LO_2 tanks. The folding aerobrake attaches at the engine end of the core truss. The brake is folded while the vehicle is in the ACC and is deployed by springs after leaving the ACC. Interface with the ACC is on the end opposite the aerobrake and engine. These points also interface with the payload adapter. Umbilical provisions with ACC are also opposite the aerobrake. Avionics are installed on the center core truss. The vehicle has been designed to be partially disassembled so that it can be returned to earth in the cargo bay of the orbiter after jettisoning the aerobrake.

The cryogenic tanks are of fusion welded construction and are made in two halves from 2090 aluminum lithium alloy. Minimum membrane thickness is .014. If problems are uncovered during testing of the 2090 alloy or in developing forming in two halves, the backup alloy would be 2219 aluminum with backup processing to be four gores per head with machined conical caps. If difficulties are encountered in handling .014 thick tanks, membrane thickness would be increased to what is required for handling. The basic air frame truss and tank support struts are graphite epoxy. The aerobrake support structure is designed to operate at 600°F. The center section is made up of a hexcel honeycomb with graphite polyimide skin, covered with shuttle FRCI-20-12 tiles. The outer portion is a flexible surface insulation composed of a Nicalon outer layer and sealed Nextel inner layer separated by Q-felt insulation. It is supported by graphite polyimide ribs that are hinged to permit stowage for ascent in the Aft Cargo Carrier. The structural airborne support equipment (ASE) considerations were shown in Figure 4.4-4. To stow the OTV in the orbiter cargo bay, the two LH_2 tanks are removed and stowed - one forward and one aft of the OTV.

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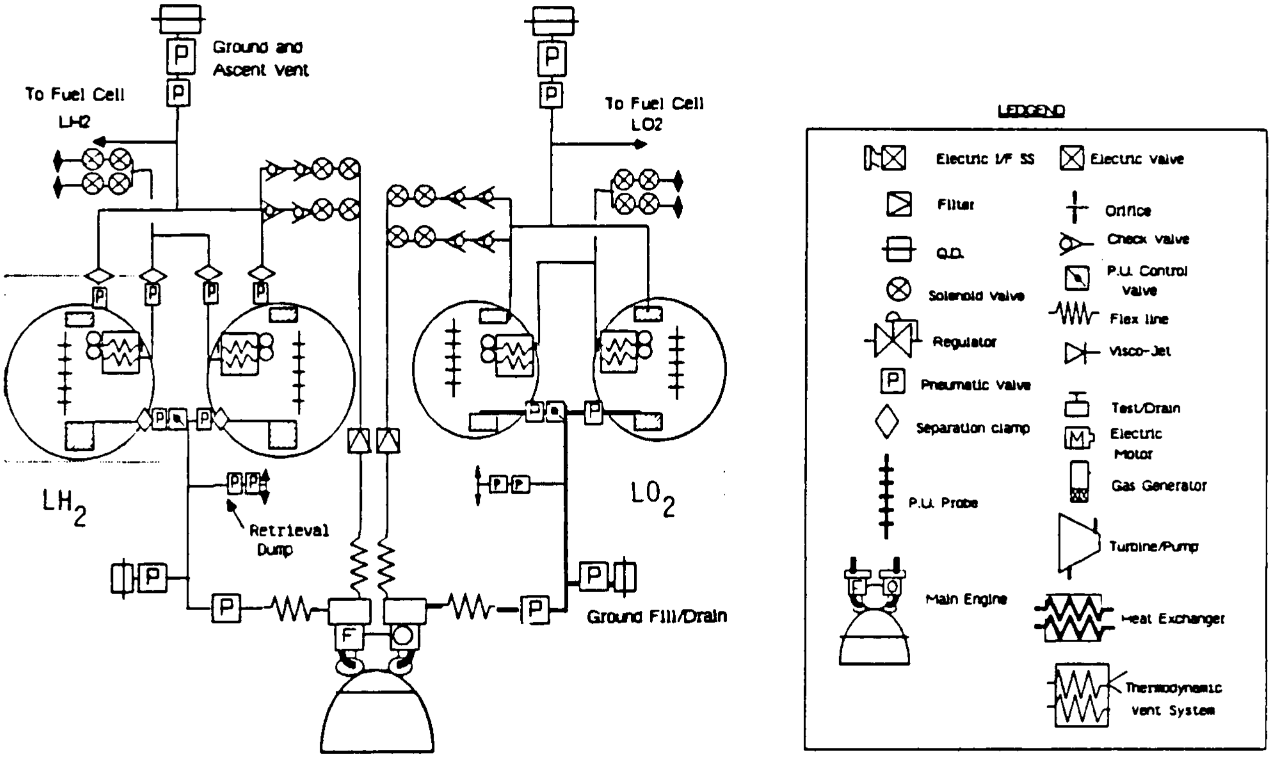

Figure 6.1.1.2.2-1 Ground Based Cryogenic Propulsion Schematic

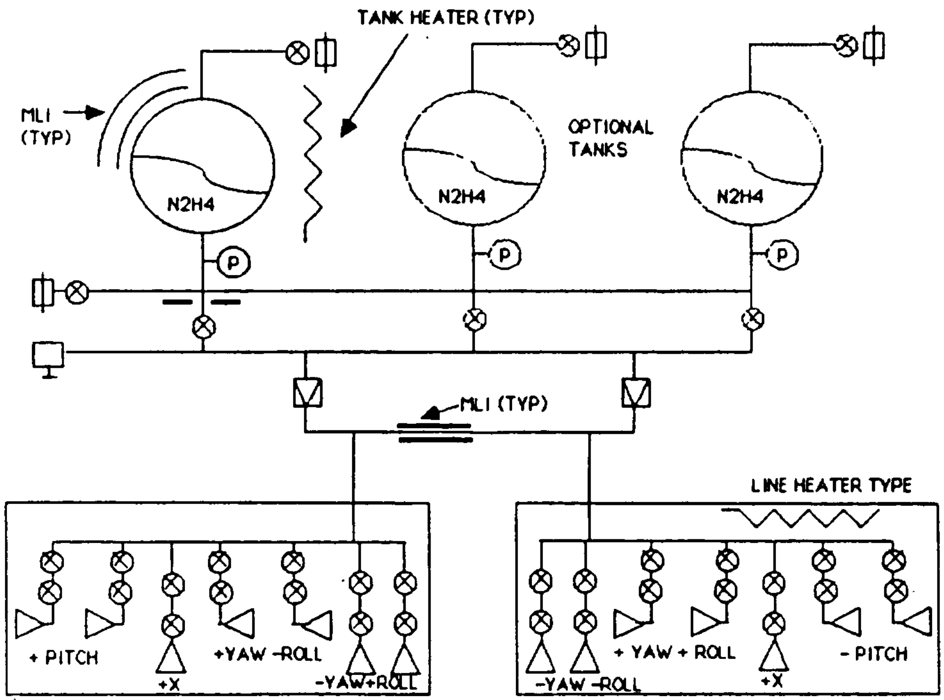

Figure 6.1.1.2.2-2 Ground-Based N_2H_4 RCS Schematics

The LH_2 tanks are supported in the orbiter bay at three points. A fitting is attached to the LH_2 tank that interfaces with the orbiter bay keel fitting as shown in Figure 4.4-4. Two trunnion fittings pick up the LH_2 tank support and interface with the orbiter longeron fittings. The LO_2 tanks remain installed in the OTV airframe. It is designed to interface with the orbiter keel fittings at the lower axle end of the LO_2 tanks. The airframe is attached to the orbiter longerons as shown in Figure 4.4-4, section A-A. The forward hydrogen tank supports are folded back and braced to the central truss by stowage members that comprise a portion of the ASE, stowage members are added to brace the OTV LH_2 truss for in flight loads. All structural ASE will be aluminum and will be stowed in the orbiter payload bay and attached to the OTV by EVA at retrieval rendezvous.

6.1.1.2.4 <u>Avionics (Ground-Based Cyro)</u> - The cyrogenic ground-based, ACC delivered, OTV avionics, Figure 6.1.1.2.4-1, is a modular design that supports technology insertion as well as redundancy enhancement. A significant feature is its distributed computer architecture with a flexible executive operating system that facilitates performance enhancement and permits affordable software development. The design is single fault tolerant through internal component redundancy for mission success and two fault tolerant for critical operations in the vicinity of the Orbiter. An avionics component list and physical description is presented in Table 6.1.1.2.4-1.

6.1.1.2.4.1 <u>Guidance, Navigation and Control (GN&C)</u> - The GN&C hardware consists of the following:

a. Strapdown Inertial Measurement Unit (IMU)
b. Solid State Star Scanner
c. GPS Receiver/Processor and Hi and Low-Altitude Antennas
d. Majority Vote Flight Controller

A detailed description of these elements is presented in Reference 7.

6.1.1.2.4.2 Data Management - The OTV data management subsystem is configured in a distributed architecture that includes two Executive Computers (dual-CPU type) as shown in Figure 6.1.1.2.4-1, each with large shareable mass memories and local memories. Key functional areas under Executive Computer software control are the Executive Operating System, attitude, guidance and navigation management, sequence control, power management, and test and checkout. The Executive and all of the other intelligent avionics subsystems are interconnected via a global network bus. This global network can support a throughput of from 10 to 20 Mbps via fiber optic cable. The network structure permits each subsystem to access the bus using an intelligent, standard protocol interface.

6.1.1.2.4.3 Telemetry and Command (T&C) - The telemetry and command subsystem is designed around a basic SCI Data Acquisition and Control System (DACS) having a single control and I/O interface unit. The central unit consists of an 80C86 CMOS microprocessor-based system with local RAM (32K) and ROM (8K) for conducting telemetry and command processing independent of the executive computer. Command decoding and authentication, time tagging and command override services are provided.

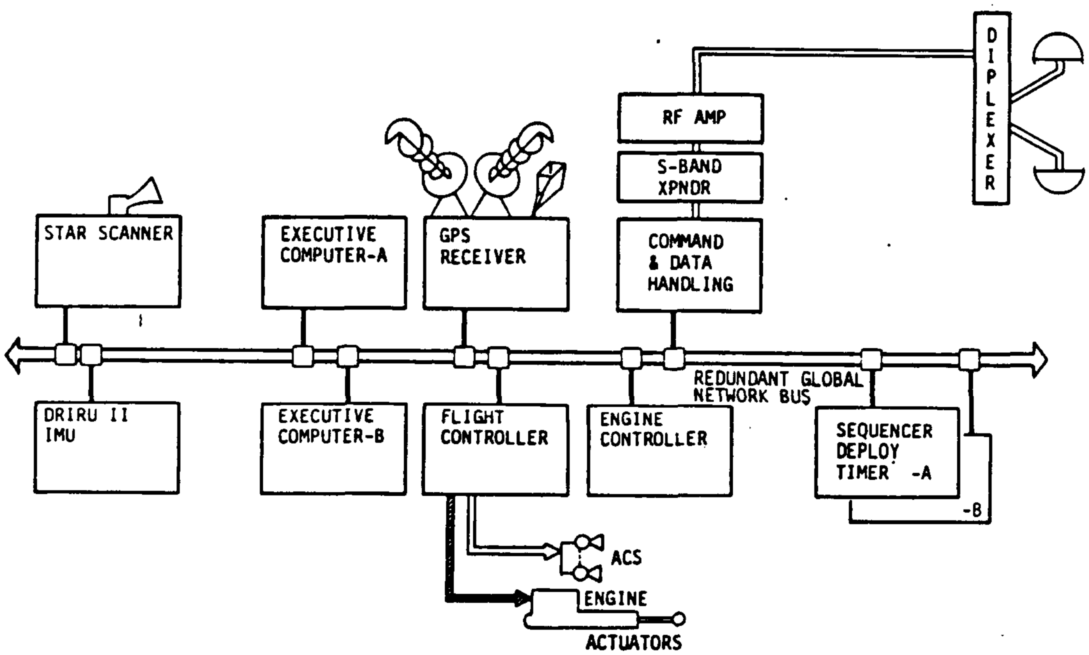

Figure 6.1.1.2.4-1 Block Diagram of the ground-based, ACC delivered, cryogenic configuration

Table 6.1.1.2.4-1 OTV Avionics Equipment List - Ground-Based ACC, Cryogenic Configuration (Sheet 1 of 2)

Subsystem / Equipment	Weight (lbs)	Power (w)	Size (in) H W L	Total Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
GN&C							
Star Scanner	12	7	7x 7x20	1	12	7	5
IMU	37	25	6x12x16	1	37	25	25
GPS Receiver	45	35	8x12x16	1	45	35	25
GPS Antenna-Low Alt	5		6x 6x10	2	10		
GPS Antenna-Hi Alt	5		18x18x26	1	5		
Flight Controller	45	120	8x 8x16	1	45	120	120
Engine Thrust Controller	10	40	6x 8x 9	1	10	40	10
			Subsystem Total		164	227	185
Data Management							
Executive Computer & Mass Memory	10	60	6x 8 x 9	2	20	120	85
			Subsystem Total		20	120	85
Telemetry and Command							
Command & Data Handling	15	35	6x 8x10	1	15	35	22
TLM Power Supply	7	10	4x 7x 7	1	7	10	5
Deploy Timer	6	6	3x 4x 7	2	12	12	4
			Subsystem Total		34	57	31

Table 6.1.1.2.4-1 OTV Avionics Equipment List - Ground-Based ACC, Cryogenic Configuration (Sheet 2 of 2)

Subsystem / Equipment	Weight (lbs)	Power (w)	Size (m) H W L	Total Qty	Total Wt (1b)	Power (w) Max	Power (w) Avg
Communications and Tracking							
STDN/TDRS Xponder	16	55	6x 6x14	1	16	55	55
20w RF Power Amp	6	125	3x 6x10	1	6	125	40
S-Band RF System	90	30		2	180	60	30
Subsystem Total					202	240	125
EPS							
Fuel Cell (FC)	35		11x12x12	2	70		
FC Radiators	35		$35ft^2$x2"	1	35		
FC Plumbing	25				25		
FC Coolant	10				10		
FC H_2O Tank	13				13		
FC EOM Tanks	80				80		
Power Control & & Distribution	27	10	6x 8x12	2	54	20	20
Engine Power		300				300	
Subsystem Total					287	320	20
System Total					707	964	446

6.1.1.2.4.4 Communication and Tracking (C&T) - The C&T subsystem provides both direct and relay communication with the ground. Communication with the Orbiter is either direct or through a ground station. The C&T subsystem operates at S-band and is compatible with STDN/TDRSS and SGLS depending upon the specific mission. Provisions have been incorporated for redundant transponders, RF power amplifiers and COMSEC equipment. Two electronically switched steerable array antennas provide hemispheric coverage. Each antenna includes a redundant microprocessor and redundant switching power divider. The other major components are inherently redundant, i.e., 145 passive elements with associated power drivers. Each antenna also includes an integrated preamplifier to facilitate parallel operation of two receivers (for fault tolerant reception) with minimal RF distribution losses. The direct/relay feature provides maximum flexibility from low earth orbit to GEO in terms of coverage and link margins for the various OTV missions. Relay C&T via TDRSS provides the primary tracking and communications for OTV operations below 10,000Km altitude. Direct C&T is the primary mode for higher OTV altitudes, with TDRSS as a backup where coverage is available. The heart of the C&T subsystem is a dual mode TDRSS/STDN transponder and 20 watt RF amplifier (such as the existing Motorola packages) combined with the Ball Aerospace ESSA. This combination provides the flexibility in spatial coverage and the necessary link margins for the various OTV missions.

6.1.1.2.4.5 Electrical Power Subsystem (EPS) - The OTV Electrical Power Subsystem, Figure 6.1.1.2.4-2, consists of redundant fuel cells, vehicle cabling, power distribution and control, reactants, plumbing, and radiators. Power is distributed through redundant buses to the OTV subsystems. The Power Control and Distribution Assembly (PCDA) contains motor driven switches and relays needed to provide load control and fault protection circuitry. The PCDA also interfaces the command and data systems where commands are received from the OTV data bus, and health and status are passed to the data management subsystem. Each of the OTV fuel cells is sized to deliver 1.2 KW peak which includes 20% design margin. A high current density design for the fuel cells was selected to minimize weight and volume for short and medium duration missions. The fuel cells are also sized to provide coarse bus voltage regulation (28 $\pm$ 4 VDC) during worst case operation at the end of a five year life. This eliminates the requirement for active power conditioning. An active coolant loop and radiator system are used to reject fuel cell waste heat. One 35 square foot radiator is sized to reject the fuel cell waste heat. Reactants are taken from the main propellant system. Redundant fuel cells and plumbing allow the EPS to meet system reliability requirements without battery backup. There is no safety issue associated with this type of a fuel cell application because it is an extension of the STS design. System power up is also simplified because fuel cell initialization consists of warming the catalysts to operating temperature and supplying reactants.

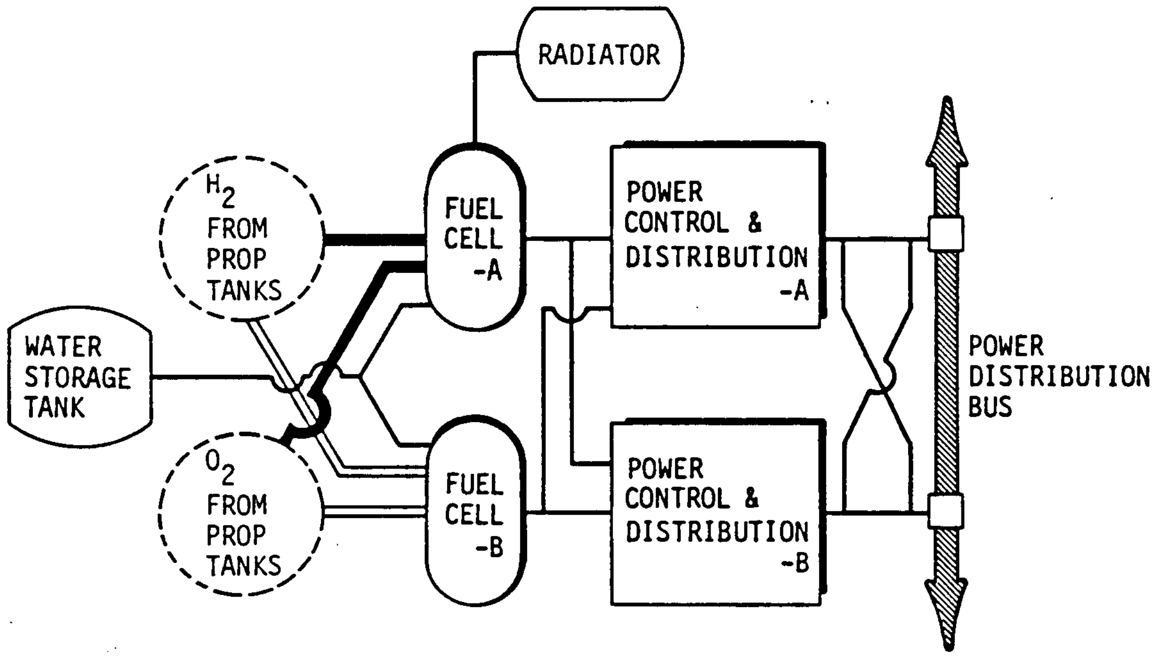

Figure 6.1.1.2.4-2 EPS Configuration for the Ground-Based, ACC Delivered, Cryogenic Configuration

6.1.1.2.5 Thermal Control (Ground-Based Cyro) - This configuration utilizes a fuel cell power system. The fuel cell thermal control system (TCS) is sized for an OTV continuous flight power requirement of 1.2 KW and a nominal 76-hour OTV flight duration. The fuel cell TCS requires 35 sq-ft^2 radiator area to dissipate the fuel cell waste heat effectively. The fuel cell radiator weight is 35 lb (Table 6.1.1.2.4-1). The radiators face outboard (maximum view to space) and are mounted to the oxygen tanks with low conductivity mounts which are blanketed from a sun flux environment. A one layer minimum thermal blanket is fixed to the back of the radiator facing the oxygen tank; the cryo side of this blanket has a low emissivity.

The avionics packages are located on the structural trusses between the cryo tanks. The avionics are passively cooled and mounted on pallets for effective heat sinking energy distribution. Small thermostatically controlled heaters and thermal blankets may be required for certain components to supplement the passive thermal control system. Component surface finishes (i.e., painted or polished) and mounting techniques shall be specified at a later time in the OTV design development.

The payload/OTV interface must be made nearly adiabatic. To accomplish this, 25 to 50 layers of insulation blanket (double aluminized Kapton MLI), is located at the interface.

The cryo tanks require a significant amount of insulation to prevent excessive boiloff. 0.25 inch of SOFI with 0.5 inch of MLI (25 layers of double aluminized Kapton) are used for the H_2 tanks. The layer of SOFI is used to eliminate the need for an MLI purge system. The temperature of the outer side of the SOFI is warm enough not to freeze any component of the dry nitrogen atmosphere provided in the Aft Cargo Carrier. The SOFI is not required for the oxygen tanks. The main propellant feedline insulation is 2 layers of gold foil.

The OTV reaction control system (RCS) requires thermal protection for the RCS tank, feedlines, and propulsion modules. The RCS tank has an MLI blanket (10 to 25 layers) and strip heaters. The feedlines contain hydrazine (freezing point of 35^oF) and requires low power (approximately 25 watt) strip heaters and one or two layers of thermally insulating blankets. The RCS modules will be maintained at sufficienty high temperatures by "thermal pulsing" techniques (i.e., periodic module firings).

The helium tank used in the pneumatic system requires heater tape for adequate thermal control to maintain proper pressurization.

Engine nozzle heating effects are not considered a problem for this configuration.

6.1..1.3 System Weight Summary - Ground-Based Cryo

Total flight vehicle weight for the ground-based cryogenic configuration is presented in Table 6.1.1.3-1. Dry weight, nonpropulsive fluids and usable propellant are summarized. Dry weight is categorized according to the groupings requested by MSFC, and the individual items include a 15% contingency (assuming that all equipment can be considered to be new in this time frame). Table 6.1.1.3-2 shows a detailed dry weight breakdown within each group, including the contingency weight assigned.

Table 6.1.1.3-1 Stage Weight Summary - Ground-Based Cryo 45K Propellant Load

WBS GROUP	WEIGHT (LB)
2. Structures	698
3. Propellant Tanks	603
4. Propulsion Less Engine	728
5. Main Engine	313
6. Reaction Control System	215
7. Guidance, Navigation, Control	180
8. Communications & Data Handling	303
9. Electrical Power	403
10. Thermal Control System	153
11. Aerobrake	1320
Dry Weight Total	4916
12. Fluids	
Reactants, Coolants & Residuals	
Residual - FU (LH_2)	96
Residual - OX (LO_2)	579
FC Coolant	10
Pressurants - He & GN_2	24
Hydrazine - ACS	400
Inert Weight Total	6025
Usable Main Propellants	
Fu-LH_2 (Incl. FPR)	6332
OX-LO_2 (Incl. FPR)	37993
Ignition Weight Total	50350

Mass Fraction

$$\frac{44325 \text{ (Main Prop Incl FPR)}}{50350 \text{ (Ignition Weight)}} = 0.88$$

Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity

WBS GROUP	ELEMENT	WEIGHT (LB)	
2	Structure		
2.1	Air Frame		442
	Truss Work	384	
	Contingency	58	
2.2	Thrust Structure		29
	Engine Truss	25	
	Contingency	4	
2.3	Equipment Mounts		112
	REMS & Hydrazine Tank	12	
	Electrical Equip	39	
	Avionics	46	
	Contingency	15	
2.4	Payload Attachment		46
	Adapter/Attachment	40	
	Contingency	6	
2.5	Micometeroid Shield		0
	N/A	0	
2.6	Handling and Storage Structure		69
	PIDA and Struts	30	
	RMS Grapple Fixtures	30	
	Contingency	9	
	Group 2 Total		698
3	Propellant Tanks		
3.1	Tank Structure		483
	LH2 (2)	242	
	LO2 (2)	178	
	Contingency	63	
3.2	Tank Mounting		120
	LH2	52	
	LO2	52	
	Contingency	16	
	Group 3 Total		603
4	Propulsion Less Engine		
4.1	Press. Pneumatic Sys		131
	Lines, Valve, X-Ducer	114	
	Contingency	17	
4.2	Propellant Feed, Vent & Drain - Fuel		235
	Feed	73	
	Vent & Drain	100	
	Press	31	
	Contingency	32	
4.3	Propellant Feed Vent & Drain - Ox.		205
	Feed	65	
	Vent & Drain	82	
	Press	31	
	Contingency	27	

Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
4.4	Prop Utilization System		129
	Probes	44	
	Computer	68	
	Contingency	17	
4.5	Misc System		28
	Pryo Cable Cutter	24	
	Contingency	4	
	Group 4 Total		728
5	Main Engine		
5.1	Engine (1)		240
5.2	Actuators (2) Elec		32
5.3	Contingency		41
	Group 5 Total		313
6	Reaction Control System		
6.1	REM Assy		43
	REM (6)	37	
	Contingency	6	
6.2	Tank		94
	Hydrazine (2)	82	
	Contingency	12	
6.3	Plumbing & Installation		78
	Line, Valves, X-Ducers	68	
	Contingency	10	
	Group 6 Total		215
7	Guidance Navigation & Control		
7.1	Control & Guidance		166
	Flight Controller & TLM	52	
	IMU Processor	37	
	GPS Receiver	45	
	Thrust Controller	10	
	Contingency	22	
7.2	Navigation		14
	Star Scanner	12	
	Contingency	2	
	Group 7 Total		180
8	Communications & Data Handling		
8.1	Communications		263
	GPS Antenna System	15	
	STDN/TDRS Xponder	16	
	20w RF Power Amp	6	
	S Band RF System	180	
	Deploy Timer	12	
	Contingency	34	

Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
8.2	Data Management		40
	Central Computer Mass Mem	20	
	CMD & Data Handling	15	
	Contingency	5	
8.3	Video		0
	N/A	0	
	Group 8 Total		303
9	Electrical Power		
9.1	Fuel Cell System		109
	Fuel Cell	70	
	Fuel Cell Plumbing	25	
	Contingency	14	
9.2	Radiator System		52
	Radiator	35	
	Plumbing	10	
	Contingency	7	
9.3	Residual H20 System		15
	Tank	8	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Tanks & Plumbing		94
	LH2	9	
	LO2	7	
	LH2 Plumbing	33	
	LO2 Plumbing	31	
	Contingency	14	
9.5	Power Distribution		133
	Wire Harness, Connector,Etc	116	
	Contingency	17	
	Group 9 Total		403
10	Thermal Control		
10.1	Insulation		109
	LH2 Tank	61	
	LO2 Tank	32	
	ACS Tank	2	
	Contingency	14	
10.2	Thermal Control		44
	ACS (Htr Tape)	3	
	FC Sys (Htr Tape)	3	
	Prop Line, F/C System	16	
	Engine Compt	10	
	Electrical Sys	6	
	Contingency	6	
	Group 10 Total		153

Table 6.1.1.3-2 Detailed Dry Weight Breakdown - Ground-Based Cryo 45K Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
11	Aerobrake		
11.1	Heat Shield		857
	TPS Center Dome - (Fixed)	177	
	TPS Quilt (Flex)	568	
	Contingency	112	
11.2	Doors & Mechanism		116
	Doors	101	
	Contingency	15	
11.3	Support Structure		347
	Ribs and Struts	302	
	Contingency	45	
	Group 11 Total		1320
15	Propellants		
15.1	Main		45010
	Usable - LH2 (inc. FPR)	6332	
	Usable - LO2 (incl FPR)	37993	
	Residual - LH2	96	
	Residual - LO2	579	
	Press. Pneum. (He)	10	
15.2	F.C. Coolant & Reactants		10
	Coolant	10	
15.3	ACS		414
	Hydrazine	400	
	Pressurant - GH2	14	
	Group 15 Total		45434

6.1.1.4 Performance on Model Missions: Ground-based Cryo - The following is a summary of the ground rules and assumptions used in the performance analyses contained herein. This decription applies not only to the data presented in section 6.1.1.4 but to the analyses in section 6.2.1.4 as well.

The delta v's used for ground-based geosynchronous delivery missions were as shown in Table 6.1.1.4-1:

Table 6.1.1.4-1 Ground-Based ACC OTV GEO Delivery Delta Vs

BURN	PURPOSE	PLANE CHANGE (DEG)	PROPULSIVE DELTA-V (FPS)
1	Shuttle MECO to 86.4 x 140.0 nmi	0.00	248.7
2	86.4 x 140 nmi to 140 x 140 nmi	0.00	96.0
3	140 x 140 nmi to 140 x 19322.9 nmi	2.19	8073.8
4	140 x 19322.9 to 19322.9 nmi circ	26.31	5855.8
5	19322.9 circ to 45 x 19322.9 nmi	28.50	6059.7
-	Aeropass maneuver to 2.0 x 149 nmi	0.00	0.0
6	2.0 x 140 to 140 nmi circ	0.00	535.0

For ground-based cargo bay OTV missions, the GEO mission delta v's are the same as above except that the first two burns are ommitted.

The above ideal, impulsive delta-v's. Gravity induced velocity losses were added to the initial perigee burns as a function of the burn time involved. Boiloff was accounted for at the rate of 2.8 lbs/hr.

The delta v's used for planetary missions were derived from a hypothetical launch geometry which minimizes the OTV delta-v penalty incurred due to precessing of the Shuttle orbit while the OTV is away. No attempt was made to research actual launch window geometries and there was assumed to be no plane change required to get from the Shuttle orbit to the departure hyperbola at launch time. Since each planetary mission has a unique delta-v budget, we have not listed the planetary delta-v's in tabular form. More information on planetary mission analysis methodology is contained in Reference 8.

For some OTV configurations on some planetary missions it was necessary to add an expendable kick stage (EKS) to the payload. For such cases, the specific orbital energy (or C3) at which the OTV shuts down and the kick stage takes over was chosen so as to minimize the gross weight of the OTV + EKS + payload. In all cases where an EKS was used, they were sized by assuming a mass fraction of 0.95 and an Isp of 310 seconds.

Table 6.1.1.4-2 summarizes the propellant load required to accomplish each of the model missions that are to be performed by the ground-based cryogenic Orbital Transfer Vehicle. Figure 6.1.1.4-1 presents a parametric summary of the performance capability of this vehicle.

Table 6.1.1.4-2 Performance Analysis for Required Missions
Ground-Based Cryogenic ACC, 45K OTV

Isp = 475 Sec

REV. 8

MISSION	P/L UP(lb)	P/L DN(lb)	OTV PROPELLANT(lb)	
GEOSYNCHRONOUS MISSIONS				
13006	12017	0	37485	
18912	12000	2000	40488	
19031	12000	0	37428	
19031 (Reflight)	12000	0	37428	
PLANETARY MISSIONS				EKS MASS (lb)
17075	5000	0	29320	8268
17081	4079	0	14437	4636
17084	4410	0	37287	0

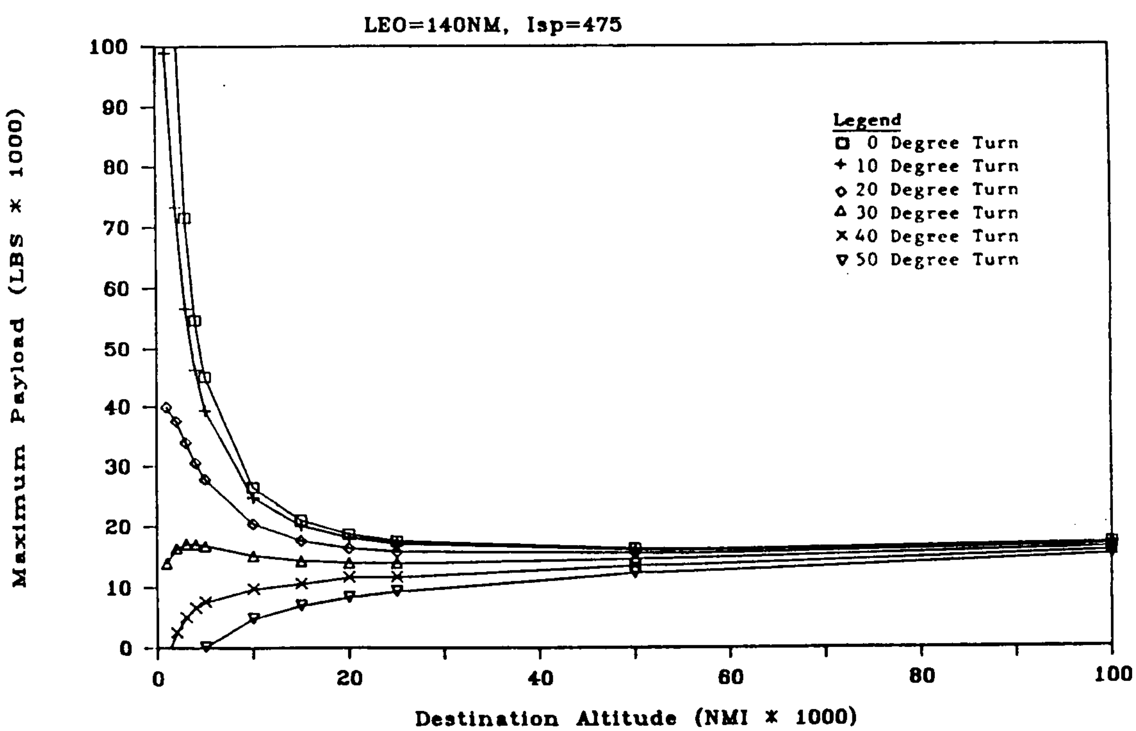

Figure 6.1.1.4-1 Ground Based 45Klb Cryo OTV Performance Capability

6.1.2 SPACE-BASED CRYO FAMILY

6.1.2.1 General Arrangement (Space-Based Cryo) - The space-based cryogenic family of OTV uses two basic stage designs in three configurations. The first configuration provides an initial space-based capability to perform GEO and planetary delivery missions and unmanned GEO servicing missions. This single stage concept is illustrated in Figure 6.1.2.1-1. This vehicle is derived from our ground based concept, but there are several important differences. Propellant capacity has been increased to 55,000 lb. to enable a 20,000 lb GEO payload delivery capability. Minimum tank gauges have been reduced to .010 in. on the LO_2 tank and .012 on the LH_2 tank, reflecting lower tank pressure requirements. Meteoroid shielding has been added to the tanks. The general arrangement has been opened up to permit servicing at the space station, when necessary, by a space suited astronaut. Redundancy, including two main engines, has been added to increase mission reliability. Avionics units have been mounted on an avionics ring at the forward end of the vehicle to simplify space-based maintenance. The aeroshield is designed to withstand a peak pressure of 35 psf, enabling retrieval of the unmanned servicing spacecraft. A more detailed layout of this stage is shown in Figure 6.1.2.1-4.

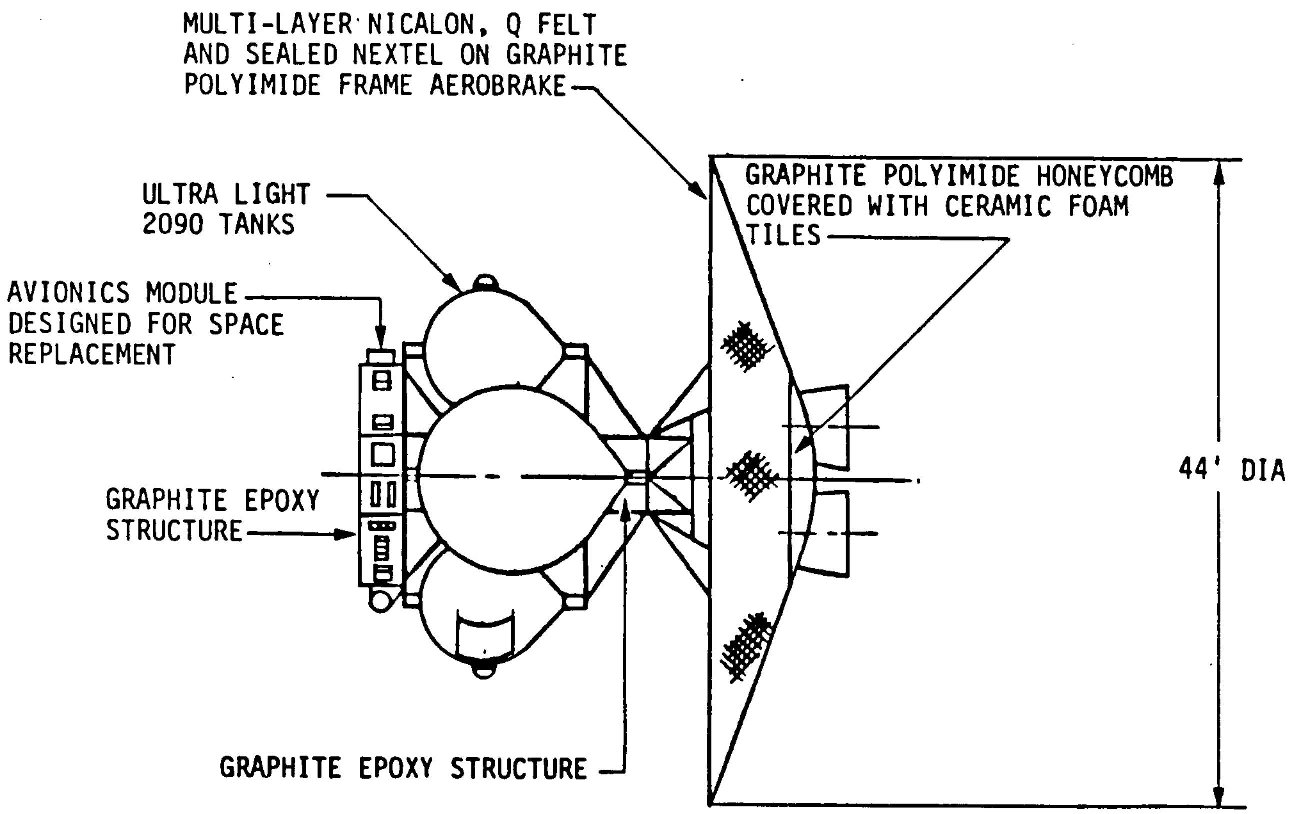

Figure 6.1.2.1-1 Initial Space-Based Cryo OTV

The general arrangement of our selected growth space-based OTV is shown in Figure 6.1.2.1-2. This configuration is not required to support the Rev. 8 low mission model. It is required to support the Manned Lunar Sortie mission in the Rev. 8 nominal model, and is capable of supporting the Manned GEO Sortie mission in the older Rev. 7 mission model. In most respects, this vehicle is identical to the initial space-based OTV. The basic structure is identical. The level of subsystem redundancy is the same. Since the electrical power subsystem and reaction control subsystem are fed from the main propellant tanks, no subsystem changes are required to accommodate different mission durations. Design variation does result from changes in propellant load and heating environment resulting from the delivery and retrieval of heavier spacecraft. Tank size is increased to accommodate an 81,000 lb propellant load. This is large enough to perform the largest lunar missions in a two stage configuration without excessive compromise in meeting the manned GEO sortie mission requirement. Since this vehicle is readily capable of performing the Rev. 7 manned GEO sortie (a 14000 retrieval payload), the aerobrake was sized to be compatible with this capability. In this case, the peak design pressure of the aerobrake is increased to 63 psf. This results in an increase in TPS thickness and aerobrake structural strength. A more detailed layout of this stage is shown in Figure 6.1.2.1-5.

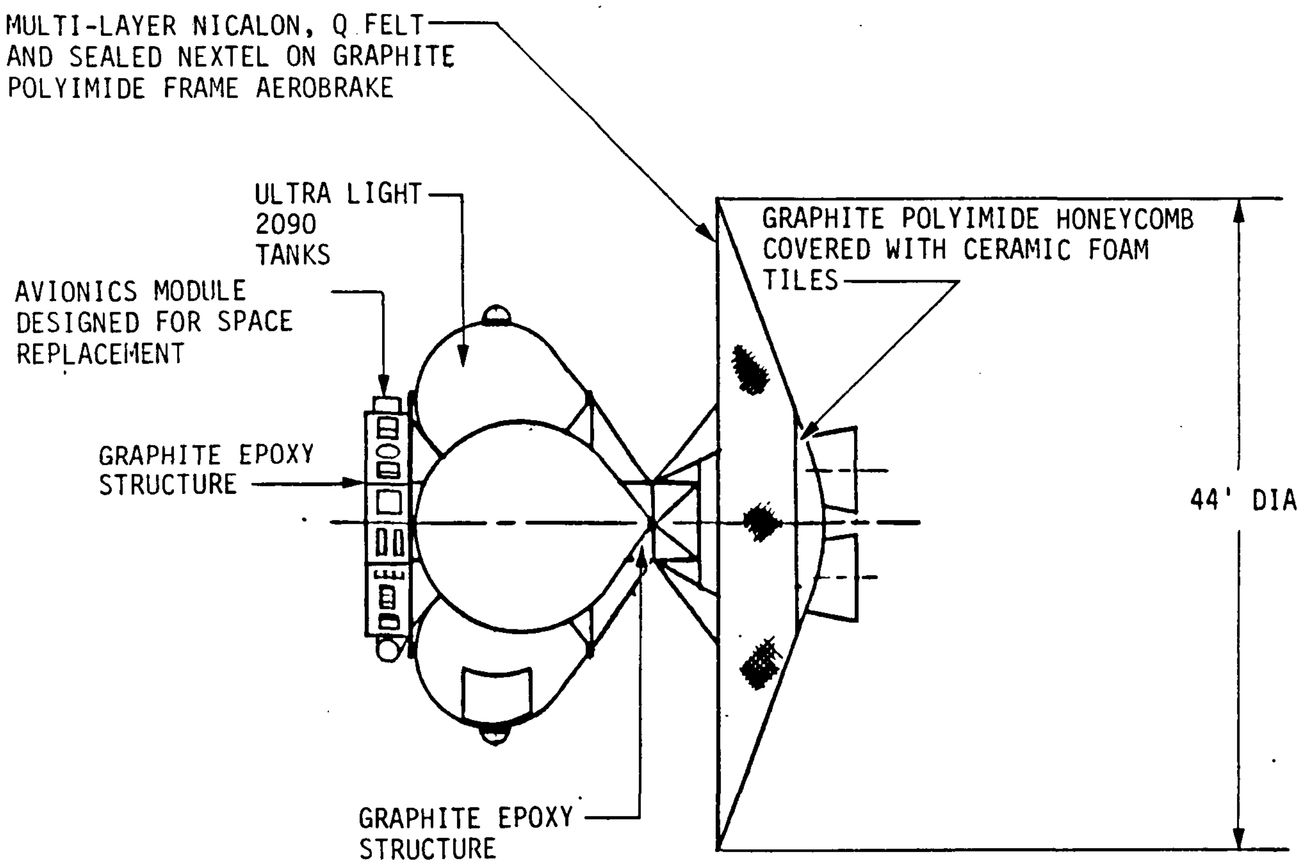

Figure 6.1.2.1-2 Growth Space-Based Cryo OTV

The ultimate spacebased OTV capability requirement, encountered in the nominal Rev 8 mission model, is to perform the Manned Lunar Sortie mission. This requires delivery of 80,000 lb to and return of 15,000 lb from low lunar orbit on a single OTV mission. The 81,000 lb propellant capacity stage was sized to accomplish this mission with the two stage configuration shown in Figure 6.1.2.1-3. No design changes are required to implement this configuration, other than development of an appropriate interstage structure. It is anticipated that the increased severity of the lunar reentry conditions can be accommodated with the same aeroshield design by using a two pass aeromanuever.

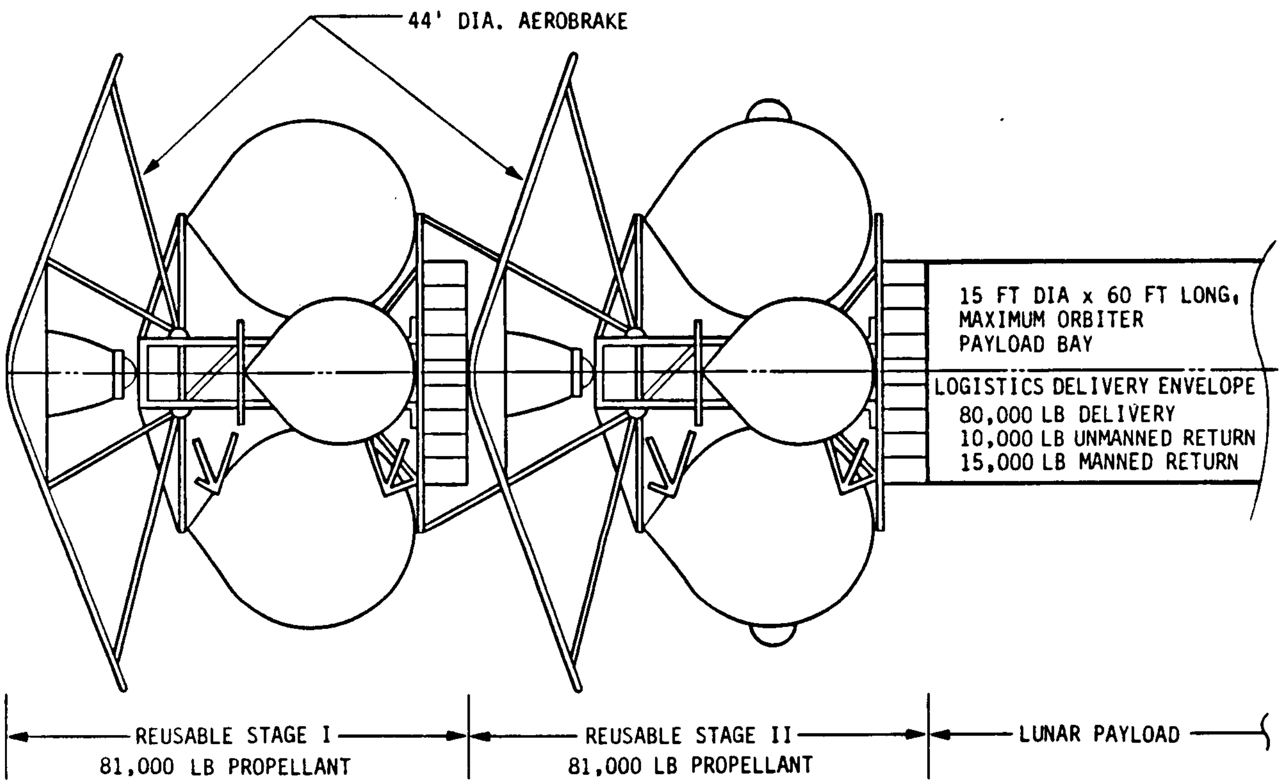

Figure 6.1.2.1-3 Cryogenic Lunar Logistics Vehicle

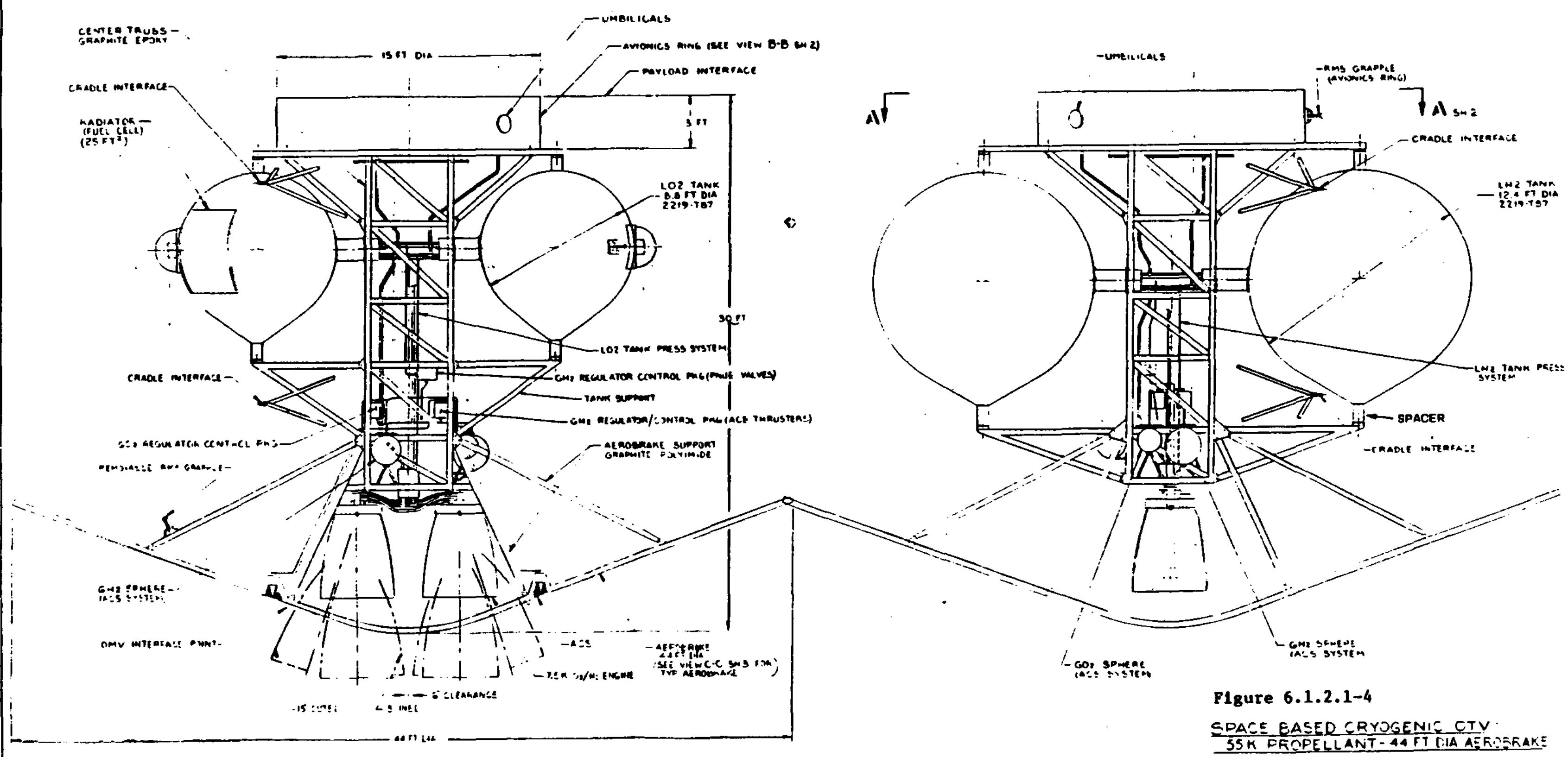

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Figure 6.1.2.1-4

SPACE BASED CRYOGENIC CTV
55K PROPELLANT - 44 FT DIA AEROBRAKE

-021

E | 53100 | SHIR-OIT-C

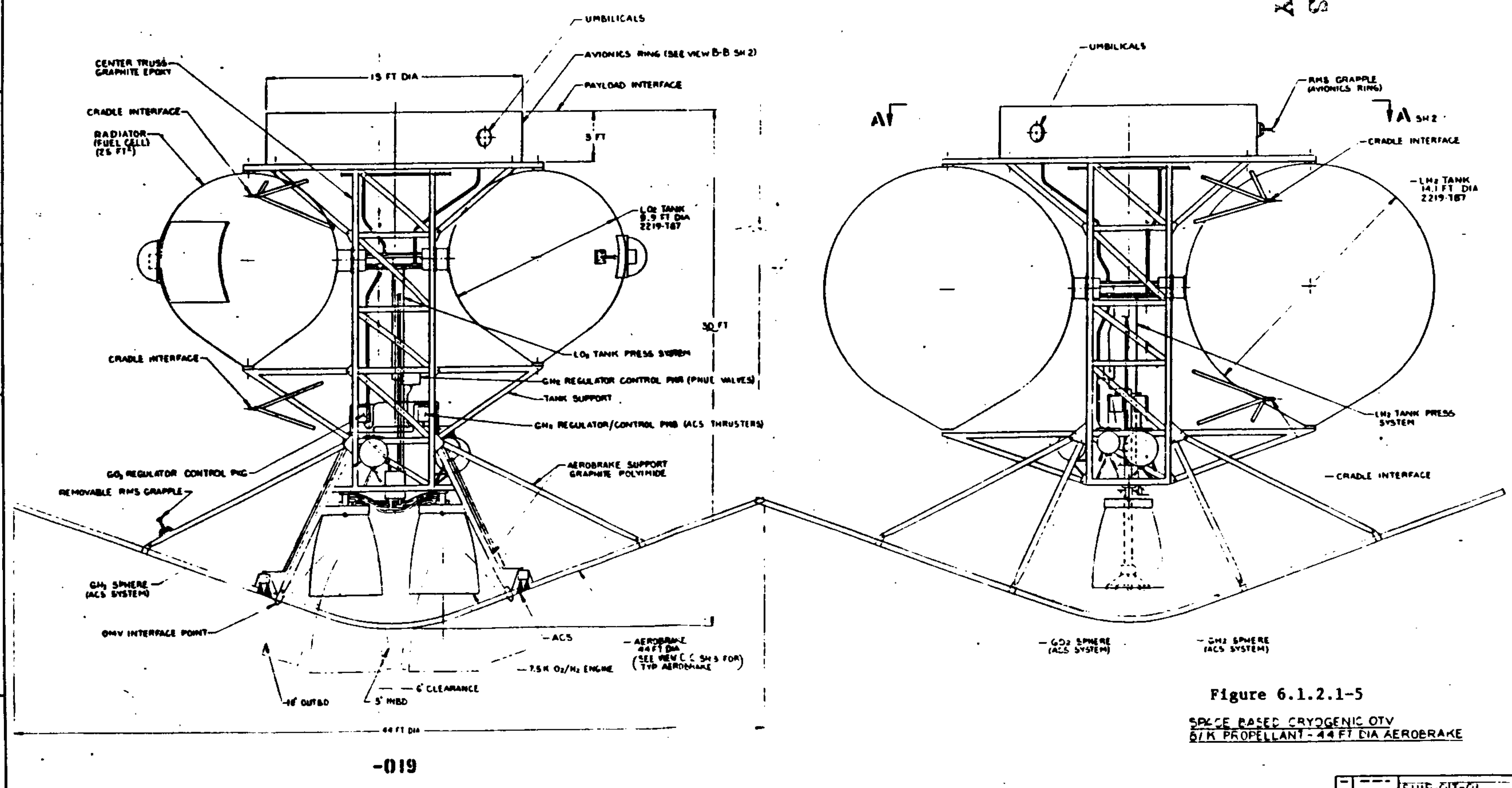
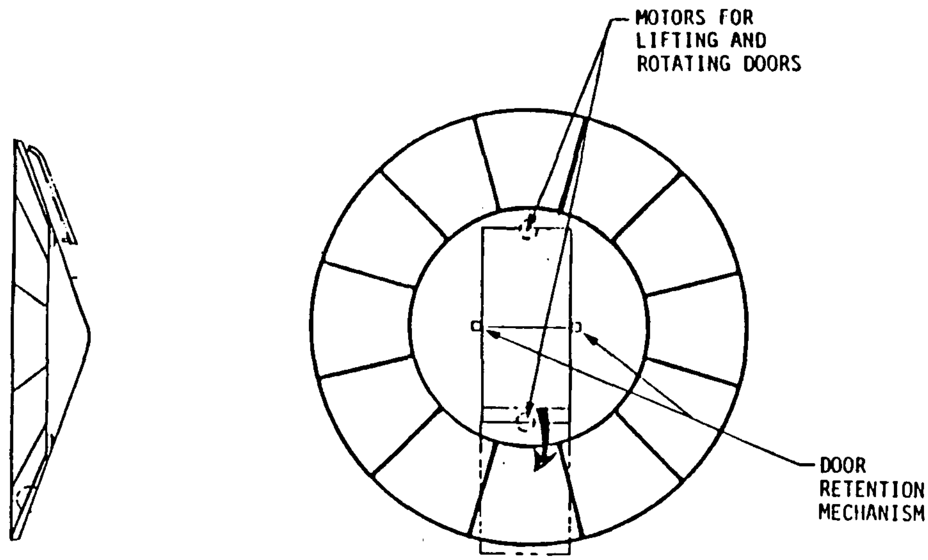


Figure 6.1.2.1-5

SPACE BASED CRYOGENIC OTV
6/K PROPELLANT - 44 FT DIA AEROBRAKE

6.1.2.2 Subsystem Summary Description (Space-Based Cryo)

6.1.2.2.1 Aeroassist (Space-Based Cryo) - The overall layout of the space based cryo configurations were shown in Figures 6.1.2.1-1 through -5. The aeroassist devices used with these configurations are similar in concept to the one used on the ground-based configuration discussed in Paragraph 6.1.1.2.1. The diameter is increased to 44 feet to protect the larger OTV stage and payloads to be retrieved. The total weights to be retrieved are heavier, so design surface pressure and heating is greater, resulting in thicker surface insulation. Figure 6.1.2.2.1-1 summarizes the aeroshield parameters.



CONFIGURATION	$W/C_D A$	BRAKE DIAMETER (FT)	TPS	$\dot{q}_{MAX}$ (BTU/FT^2 SEC)	q_{MAX} (BTU/FT^2)	$\dot{T}_{MAX}$ (°F)	TPS THICKNESS (IN.)	W_{BRAKE}/W_{RETURN}	DESIGN LOAD (PSF) CENTER	DESIGN LOAD (PSF) OUTBOARD
MANNED CAPSULE	6.0	44	FSI RSI	21.4 24.0	3050 3660	2390 2180	0.38 0.43	0.10	34	26
GROWTH * CAPSULE	9.9	44	FSI RSI	25.6 33.3	3680 4420	2600 2520	0.43 0.48	0.06	63	48

* 14,000 lbs, 14½'W x 23' L (A growth capability)

Figure 6.1.2.2.1-1 Space-Based Cryo Aeroshield

C-2

The initial capability aeroshield shown in the figure is mounted on the 55,000 lb propellant capacity stage which is designed to return the unmanned servicing spacecraft from GEO to low orbit. The growth capability aeroshield is placed on the 81,000 lb OTV, and is designed to return the 14,000 lb. manned capsule to LEO. Both of the configurations have sufficiently high ballistic coefficients, and associated reentry heating, that it is necessary to use the rigid surface insulation tiles on the central portion of the aeroshield. Likewise, the use of flexible surface insulation for the engine door is not feasible. It should be noted that the 7500 lb manned capsule of the Rev. 8 mission model can be returned with the initial capability aeroshield. The growth capability aeroshield would not have to be introduced until the manned lunar sortie is encountered.

These configurations employ two main engines to achieve fail-safe-return man rating. As a consequence, the engine doors are larger than those used on the single engined ground-based configuration as well as being constructed using rigid surface insulation. The resulting rigid doors are opened by lifting them from their openings and rotating them as shown in Figure 4.2.2.1-1. The weights of these aeroshields are summarized in paragraph 4.2.3 along with the remainder of vehicle subsystem weights.

6.1.2.2.2 Propulsion (Space-Based Cryo) - The propulsion characteristics of the space-based cryogenic stages are shown in Table 6.1.2.2.2-1 and the reaction control characteristics are shown in Table 6.1.2.2.2-2. The MPS uses two 7500 lb advanced expander cycle engines. Studies show that the development of an engine for OTV is cost effective. Technology development continues with three main engine contractors through funding from NASA/LeRC. Advanced expander cycle concepts all use higher chamber pressure and expansion ratios to obtain performance levels from 475 to 487 seconds, depending on engine contractor performance predictions and the level of technology incorporated in the expander cycle (ranges from hydrogen expander to dual propellant expander). Rocketdyne and Pratt Whitney point designs are in the 7500 and 15,000 lb thrust class, whereas Aerojet is working in the 3,000 lb thrust class. Throttle ratios possible are 10:1 for the expander cycle engines with an ultimate goal of up to 30:1, but with the current Rev. 8 mission model we have selected 50% step throttling as cost effective. All engines have THI and PHI capabilities and autogenous presssurization capability. Optional valve/actuator control is provided by high pressure GH_2 and GO_2. The cycle life varies between 300 starts and 10 hours of life up to 500 starts and 20 hours of life as a design goal. Increasing life beyond the ground-based 5 hrs must be based on mission model and cost.

Lower tank pressures are used on the space-based vehicle because the propellants will be maintained saturated at 1 atm and the engine interface is below the tank outlet. Nominal operating presures are 18 - 19 psia and 17 - 18 psia in the LO_2 and LH_2 tank, respectively.

Table 6.1.2.2.2-1 Space-Based Cryogenic MPS Summary

ENGINE	- TWO ENGINES, 7500 LB THRUST, EXPANDER CYCLE I_{sp} - 475 SEC
PROPELLANT DISTRIBUTION	- DUAL TANK, PARALLEL FEED, TOTAL ACQUISITION DEVICE
PRESSURIZATION	- AUTOGENEOUS FROM MPS ENGINE FOR PUMPED IDLE AND FULL THRUST--NOT REQUIRED FOR TANK HEAD IDLE
VENT	- TVS HEAT EXCHANGERS AND NON-PROPULSIVE VENTS FOR BOTH PROPELLANTS FOR FLIGHT OPS
VALVE ACTUATION	- HIGH PRESSURE GASEOUS HYDROGEN AND OXYGEN
PROPELLANT UTILIZATION	- TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL
THERMAL CONTROL	- H_2 1" MLI (50 LAYERS) - O_2 1" MLI (50 LAYERS)
PROXIMITY OPERATION	- TWO FAULT TOLERANT ISOLATION
REDUNDANCY	- FAIL SAFE
MAINTENANCE	- COMPONENT MODULARITY - ENGINE REPLACED AS UNIT - 5 HR LIFE

Table 6.1.2.2.2-2 Space-Based Cryogenic RCS Summary

- PROPELLANT
  - GO_2/GH_2
- ROCKET ENGINE MODULE
  - NEW DESIGN BASED ON TECHNOLOGY STUDIES, 100 LBF,
    14 THUSTERS (7 THRUSTERS PER MODULE)
    THRUSTER I_{SP} = 400 SEC, I_{SYS} = 378 SEC WITH CONDITIONER LOSSES
  - 3 DOF CONTROL AND +X TRANSLATION
  - FAIL OPERATIONAL
- PROPELLANT SUPPLY
  - COMMON STORAGE WITH MPS TANKS, CONDITIONED BY GAS GEN/HEAT EXCHANGER ASSY TO 1,000 psi
  - REGULATED TO 300 psi FOR THRUSTERS AND PNEUMATICS.
  - 320 LB_M of GO_2/GH_2 AT A MR 4:1
- SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS

Propellants are stored and delivered to the engines using the same general tankage arrangements and components defined previously for the ground-based vehicle. However for the space-based vehicle, system components and tankage will be modularized for ease of replacement on orbit. For instance, the hydrogen tank has a connector on the side that contains the propellant, pressurization, vent, thermodynamic vent, electrical power and control, and instrumentation interfaces. All connections are made simultaneous within one connector to assure ease of tank replacement. The MPS schematic is shown in Figure 6.1.2.2.2-1. A total acquisition system is included to provide for onorbit detanking as a contingency for an aborted mission.

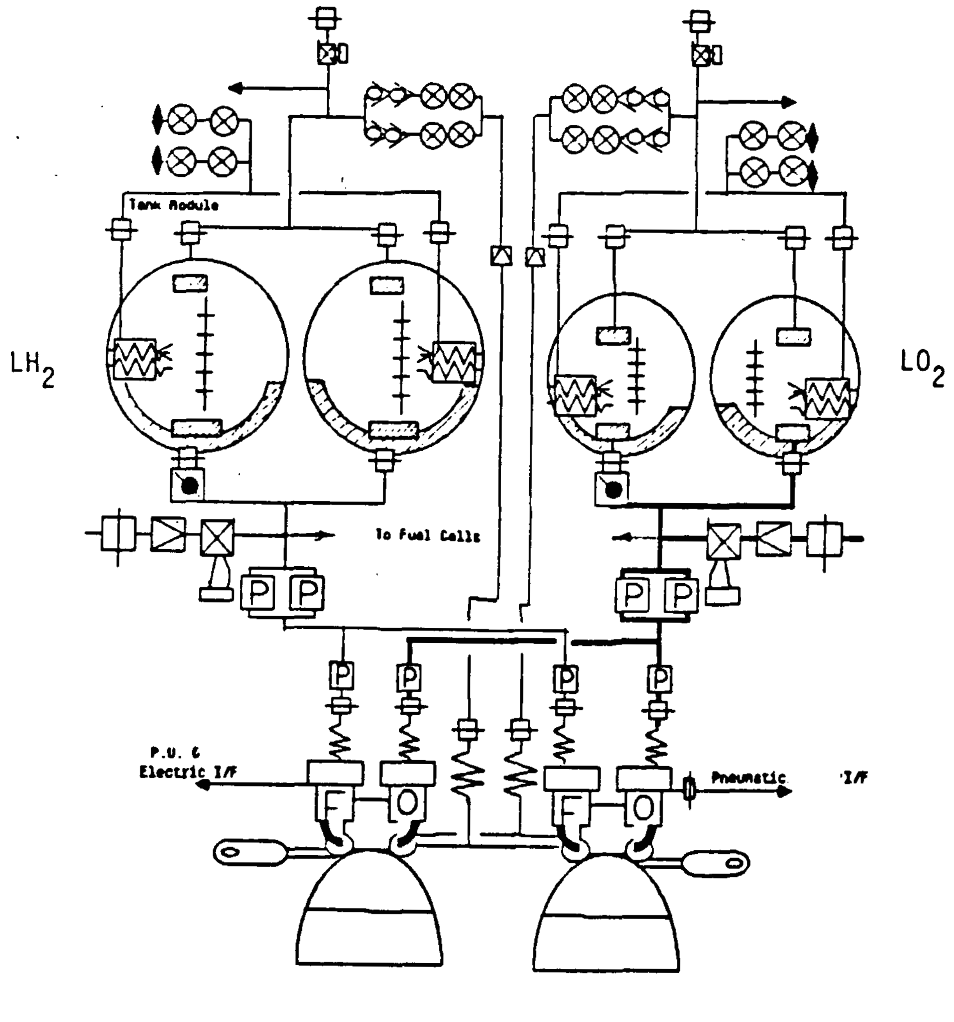


Figure 6.1.2.2.2-1 Space-Based LH_2/LO_2 Schematic

The RCS uses gaseous oxygen and hydrogen thrusters to provide thrust for attitude and translational maneuvers. The propellants are stored in the main propulsion tanks and accumulators are charged from the GGS. The accumulator pressure can vary from 1000 to 300 psi between charges and the pressure is regulated to 300 psi for delivery to the GO_2/GH_2 thrusters. The system will store 12,000 lb-sec of total impulse between recharges. The thruster technology will be new, however, it is based upon LeRC's GO_2/GH_2 thruster development which occurred in the early 1970 time period and demonstrated Isp in the 400 second range. This technology is also now being studied by ALRC and LeRC for the Space Station. To provide gas for recharging the accumulators between main engine burns a gas generator, pump and heat exchanger are provided to condition the propellants. This conditioner is designed to run off optimum performance at a mixture ratio of 1.0. This reduces gas generator temperatures to about 1500°F and allows flexibility in conditioning the system for start and a simplified control system. This lower efficiency operation degrades the system Isp from 400 to 378 seconds. The RCS schematic is shown in Figure 6.1.2.2.2-2.

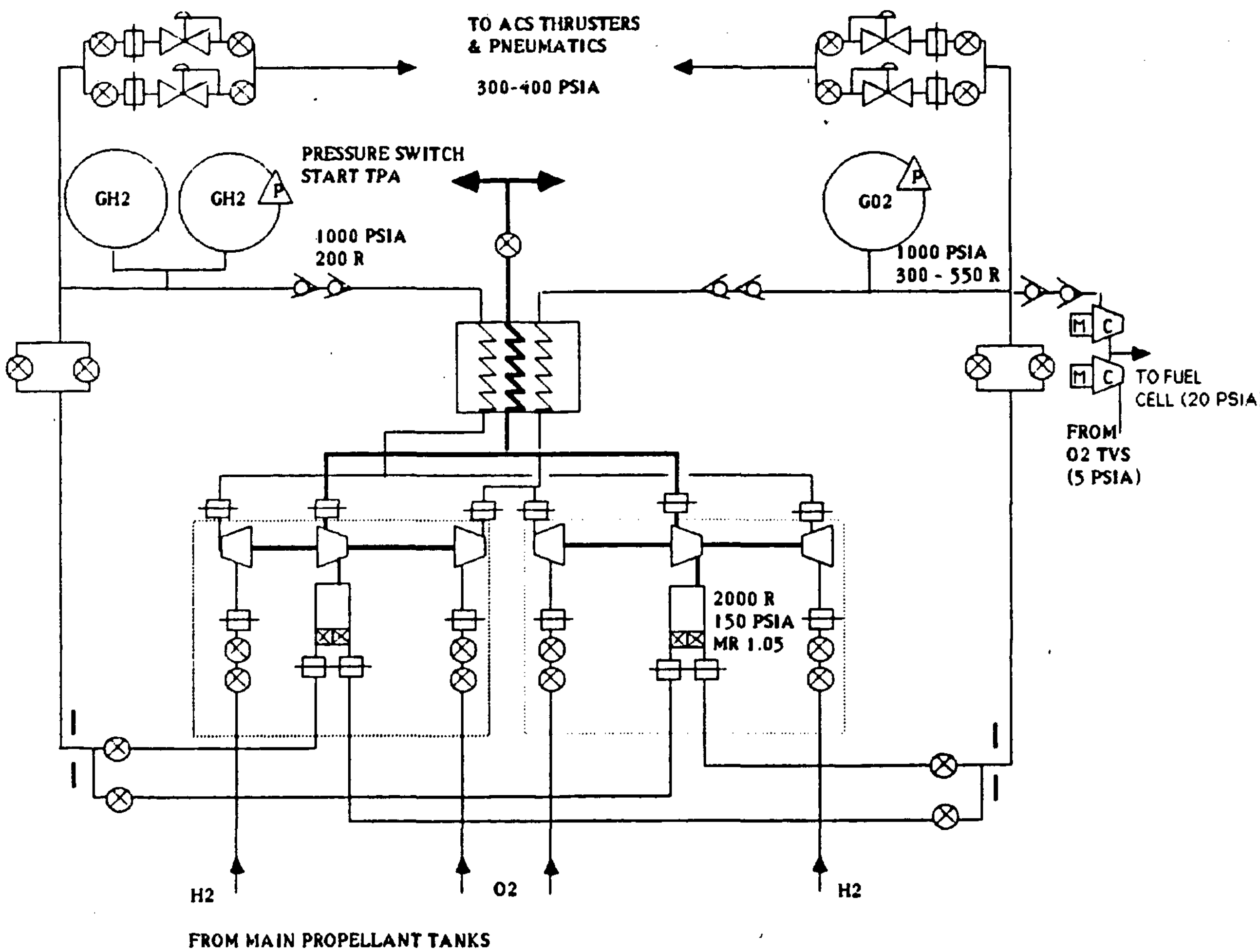
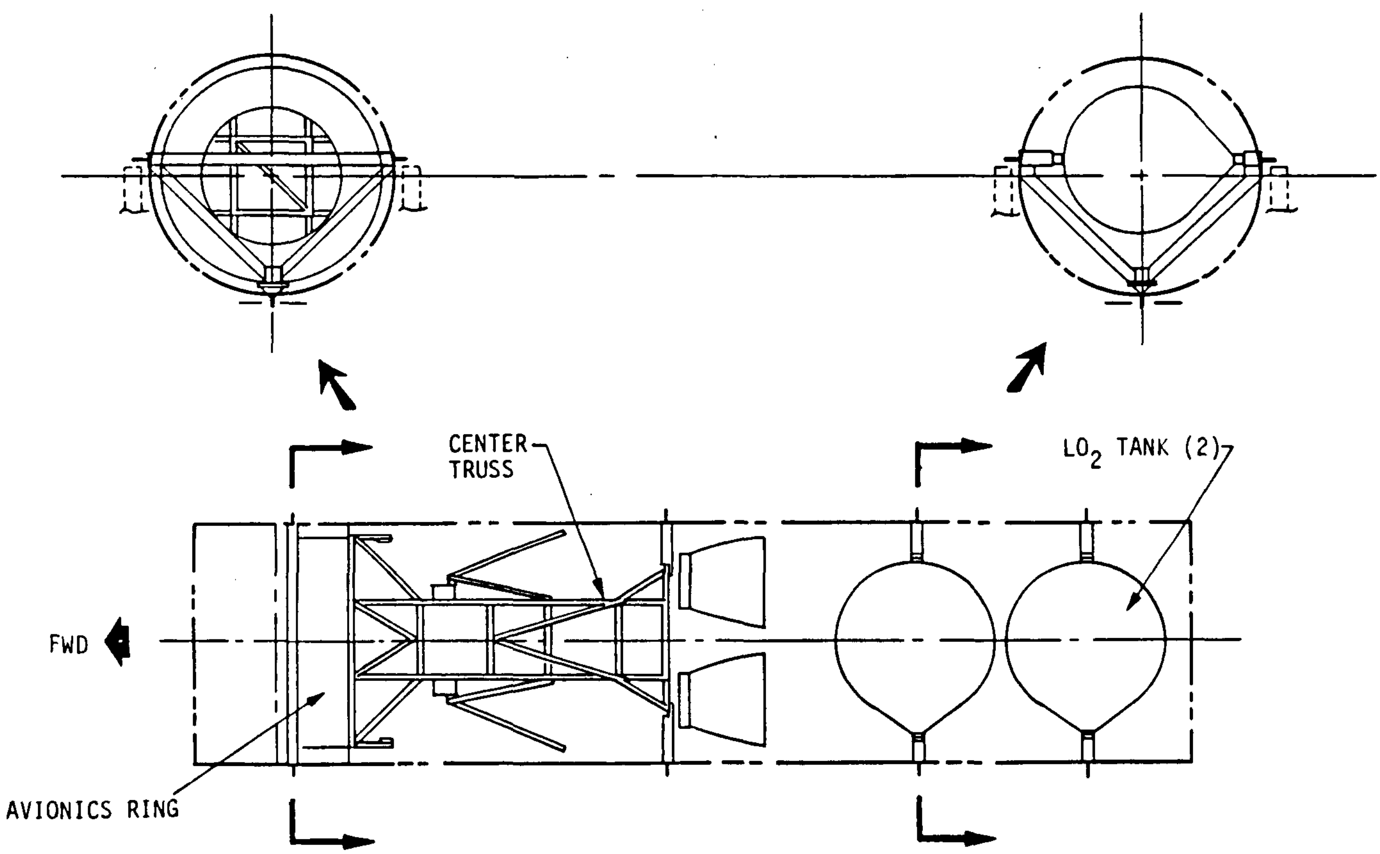


Figure 6.1.2.2.2-2 Space-Based GH_2/GO_2 RCS Schematic

6.1.2.2.3 Structures and Packaging (Space-Based Cryo) - The space-based cryo configurations shown in Figures 6.1.2.1-4 and -5 use two spherical LH_2 tanks, two spherical LO_2 tanks and two 7500 lb thrust engines. A central truss provides the backbone of the stage with engine and aerobrake mounted on one end of the truss and the avionics ring and payload adapter attached on the other. The two space-based cryo stage sizes are implemented by simply changing tanks. The smaller 55,000 lb capacity tanks will use spacers to fill the gaps between them and the structure that was designed to accommodate up to the 81,000 lb tank size.

The OTV is delivered to space in the orbiter cargo bay (Figures 6.1.2.2.3-1 and -2). For orbiter cargo bay delivery, tanks are removed from the central core and aerobrake is removed. The aerobrake is unfolded in space. Grappling fixtures have been provided to allow use of an RMS to handle the tanks and truss.



PLAN VIEW OF ORBITER BAY

Figure 6.1.2.2.3-1 Space-Based Cryo Transportation

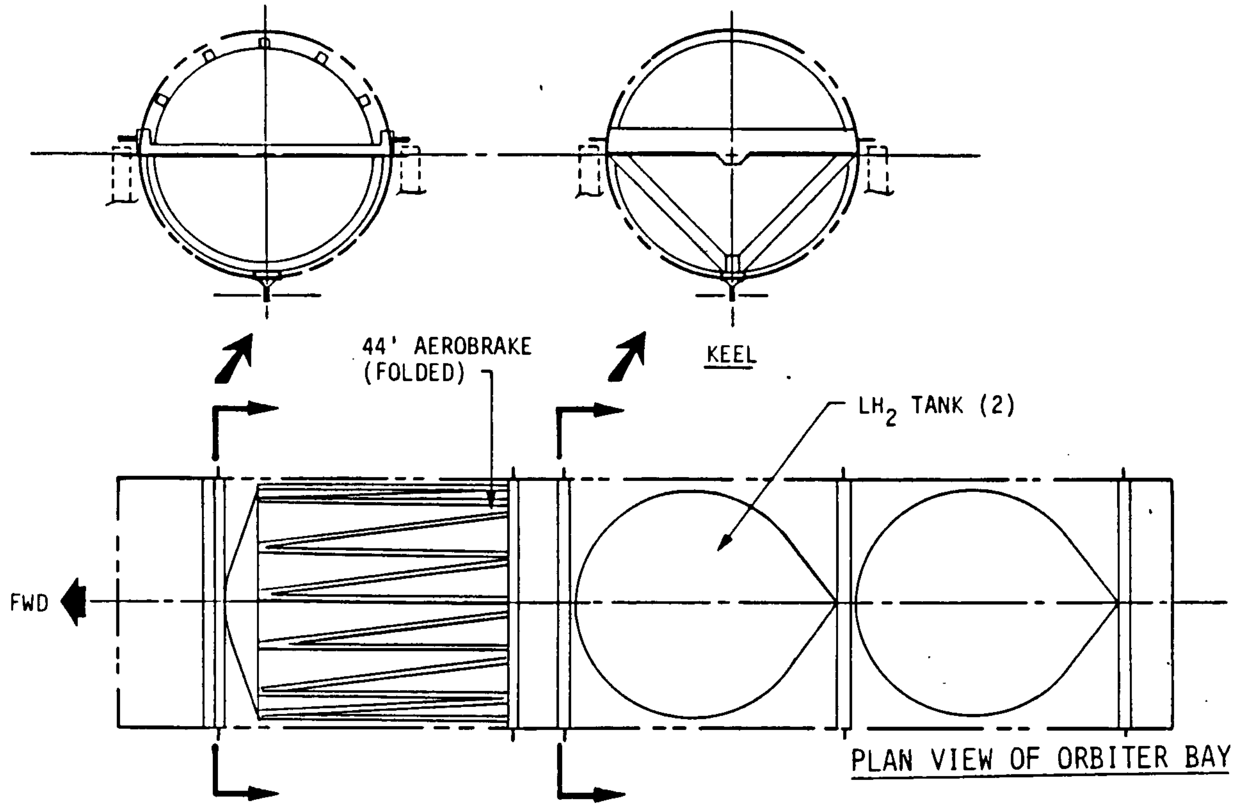


Figure 6.1.2.2.3-2 Space-Based Cryo Transportation

The aerobrake is mounted in the payload bay on an aerobrake deployment assist mechanism (ADAM). The ADAM (Figure 6.1.2.2.3-3) consists of a central shaft with 4 hinged, telescoping arms and slotted guide plates. It is designed to be either returned to earth after deploying the aerobrake or stored in the space hangar with a collapsed aerobrake. The aerobrake is deployed by extending the telescoping arms up and out, which pulls the spring assisted struts through the guide plates and allows the hinged rib to unfold. After being fully extended, the aerobrake is removed from the ADAM, and mounted to the OTV support structure.

The aerobrake structure consists of lower support ring, upper interface ring, hinged ribs and spring assisted, collapsible struts. The center core is composed of graphite polyimide honeycomb with ceramic foam tiles and a quilted outer edge of nicalon, ceramic felt and sealed Nextel.

The release mechanism, (Figure 6.1.2.2.3-4) consisting of a release handle, 12 latch pins and a connecting kevlar rope, is mounted to the OTV support ring and facilitates the attachment and removal of the aerobrake. The release mechanism, when engaged, retracts the 12 latch pins simultaneously via kevlar rope and frees the aerobrake from the OTV. Figure 6.1.2.2.3-5 shows the aerobrake rib deflection, which is considered to be acceptable.

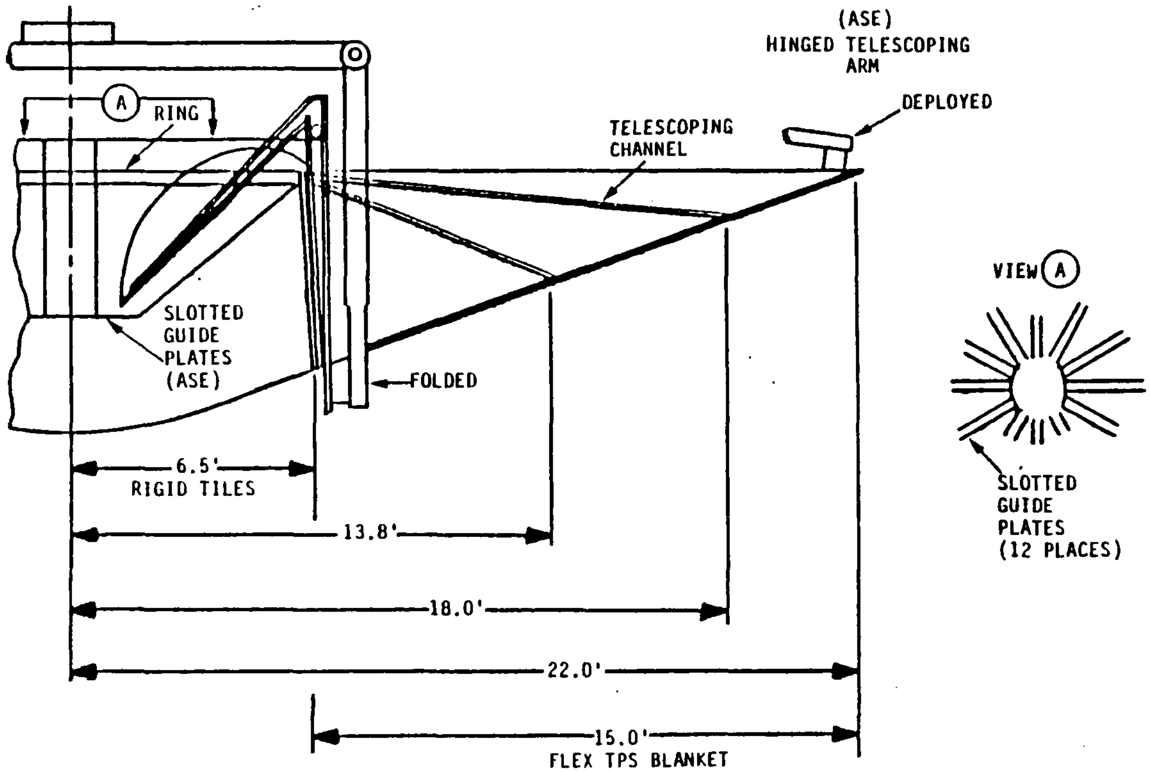


Figure 6..1.2.2.3-3 Space-Based Aerobrake

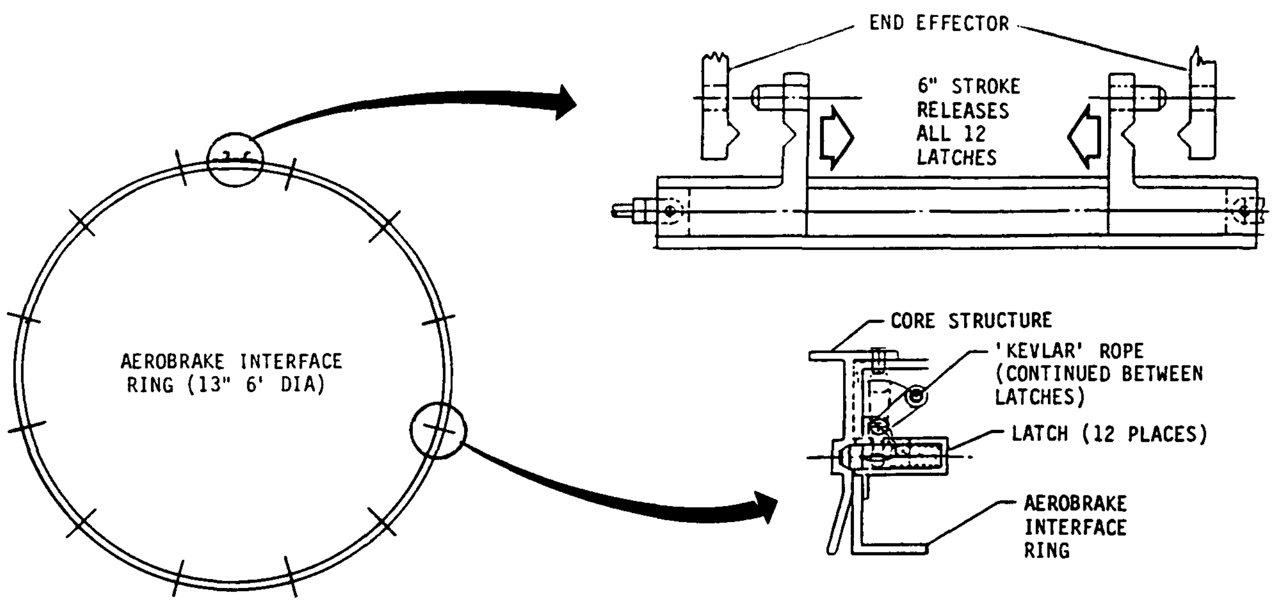


Figure 6.1.2.2.3-4 Aerobrake Release Mechanism

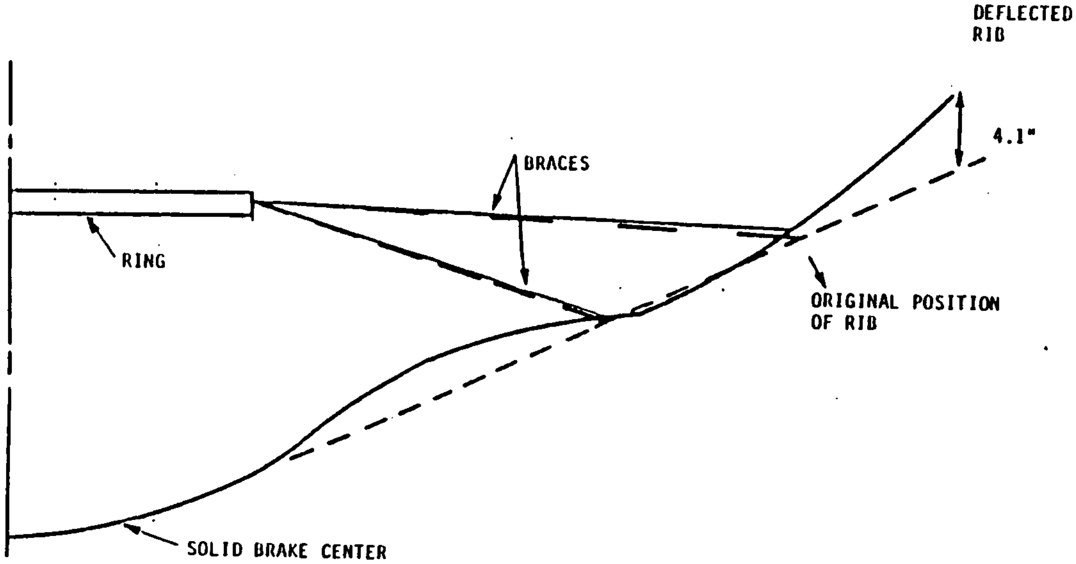


Figure 6.1.2.2.3-5 Aerobrake Rib Deflection

The cryogenic tanks are of fusion welded construction and are made in two halves from 2090 aluminum lithium alloy. LH_2 tank membrane is a minimum of .012 thick. If problems are uncovered during testing of the 2090 alloy or in developing forming in two halves, the back up alloy would be 2219 aluminum with back up processing to be four gores per head with machined concical caps. If difficulties are encountered in handling .012 thick tanks, membrane thickness would be increased as required. The basic airframe truss is graphite epoxy.

The vehicles are equipped with crane and cradle provisions for handling at the Space Station. In addition major components such as aerobrakes and tanks have grapple provisions for component changeout at the Space Station.

6.1.2.2.4 Avionics (Space Based Cryo) - Avionics for the space-based, cryogenic configuration of the OTV is modular in design and similar to the ground based configuration. The space-based configuration has a distributed computer architecture with a flexible executive operating system that facilitates performance enhancement and permits affordable software development. Because of longer mission times, this design has greater fault tolerance features. It retains the two fault tolerant feature for critical operations in he vicinity of the Orbiter. System block diagram is shown in Figure 6.1.2.2.4-1, equipment list in Table 6.1.2.2.4-1.

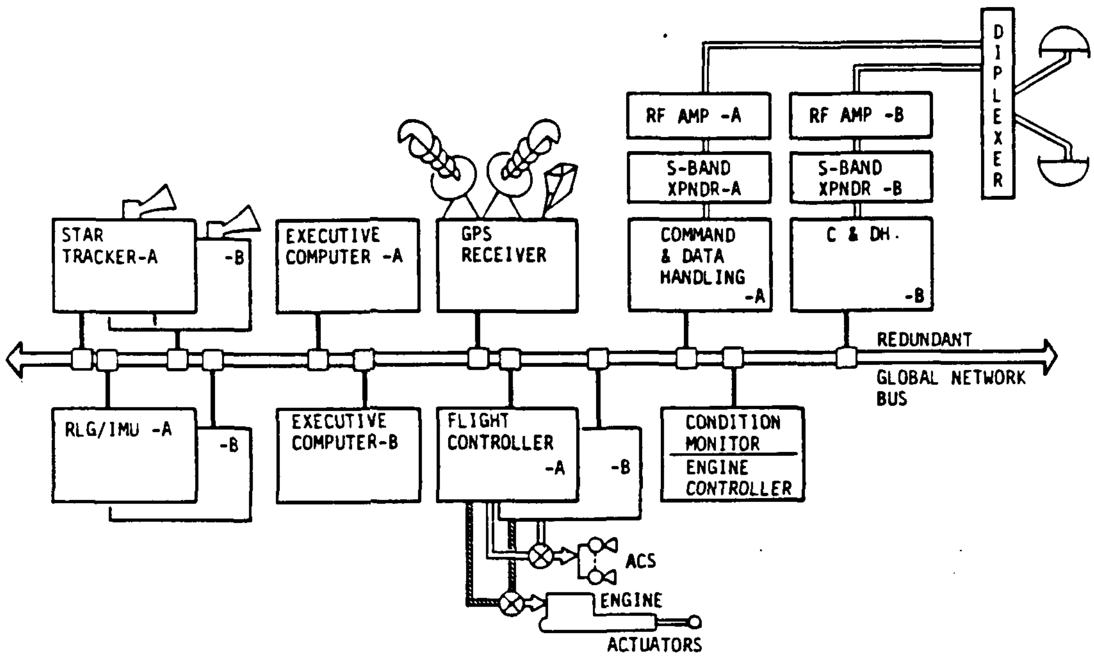


Figure 6.1.2.2.4-1 Block Diagram of the Space-Based Cryogenic Configuration

6.1.2.2.4.1 <u>Guidance, Navigation and Control (GN&C)</u> - The GN&C hardware consists of the following:

a. Dual Redundant Ring Laser Gyro (RLG) Inertial Measurement Unit(s) (IMU)
b. Dual Star Trackers
c. GPS Receiver/Processor & Hi and Low-altitude Antennas
d. Dual Majority Vote Flight Controllers

Two RLG IMUs were selected for the space-based configuration rather than the Teledyne DRIRU II unit because of the longer mission and performance capability of the cyrogenic OTV. Each IMU includes three (3) ring laser gyros (RLGs) and three (3) pendulous mass accelerometers and required computers and power supplies. A star tracker was selected instead of a scanner to take advantage of increased sensitivity of trackers and to minimize required maneuvers. Details of the selected GN&C hardware are presented in Reference 7.

6.1.2.2.4.2 <u>Data Management (DM)</u> - The OTV Data Management subsystem is the same as in Section 6.1.1.2.4.2.

6.1.2.2.4.3 <u>Telemetry and Command (T&C)</u> - The T&C subsystem is the same as in Section 6.1.1.2.4.3.

6.1.2.2.4.4 <u>Communication and Tracking (C&T)</u> - The C&T subsystem is the same as in Section 6.1.1.2.4.4.

Table 6.1.2.2.4-1 OTV Avionics Equipment List - Space-Based Configuration (Sheet 1 of 2)

Subsystem Equipment	Weight (lb)	Power (w)	Size (in) H W L	Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
GN&C							
Star Scanner	11	10	7x 7x20	2	22	20	10
IMU	24	40	8x 8x12	2	48	80	80
GPS Receiver	20	30	8x 8x 9	1	20	30	10
GPS Antenna-Low Alt	5		6x 6x10	2	10		
GPS Antenna-Hi Alt	5		18x18x26	1	5		
Flight Controller	30	90	8x 8x16	2	60	180	120
Engine Thrust Controller	10	60	8x10x 9	1	10	60	60
			Subsystem Total		175	370	214
Data Management							
Executive Computer & Mass Memory	10	60	6x 8 x 9	2	20	120	120
			Subsystem Total		20	120	120
Telemetry and Command							
Command & Data Handling	15	35	6x 8x10	2	30	45	22
TLM Power Supply	7	10	4x 7x 7	2	14	20	5
			Subsystem Total		44	65	27

Table 6.1.2.2.4-1 OTV Avionics Equipment List - Space-Based Configuration (Sheet 2 of 2)

Subsystem / Equipment	Weight (lb)	Power (w)	Size (in) H W L	Total Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
Communications and Tracking							
STDN/TDRS Xponder	16	55	6x 6x14	2	32	65	65
20w RF Power Amp	6	125	3x 6x10	2	12	125	40
S-Band RF System	50	20		2	100	40	20
			Subsystem Total		144	230	125
EPS							
Fuel Cell (FC)	45		11x12x12	2	90		
FC Radiators	25		$25ft^2$x2"	2	50		
FC Plumbing	25				25		
FC Coolant	15				15		
FC Water Storage	15				15		
Power Control & & Distribution	27	10	6x 8x12	2	54	20	20
Engine Power		600				600	
			Subsystem Total		249	620	20
			System Total		632	1405	506

6.1.2.2.4.5 Electrical Power Subsystem (EPS) - The Electrical Power subsystem is essentially the same as in Section 6.1.1.2.4.5 sized for a 1.7 Kw peak output which includes a 20% design margin for each fuel cell. The configuration is shown in Figure 6.1.2.2.4-2. Two 25 sq ft radiators reject fuel cell waste heat. Since the EPS reactant is supplied from the main propellant tanks, the OTV has an inherent ability to support longer duration missions without requiring design changes. End-of-mission fuel cell reactant tankage is not required as propellant tank purge is not required while the fuel cells are operating during space-based operations.

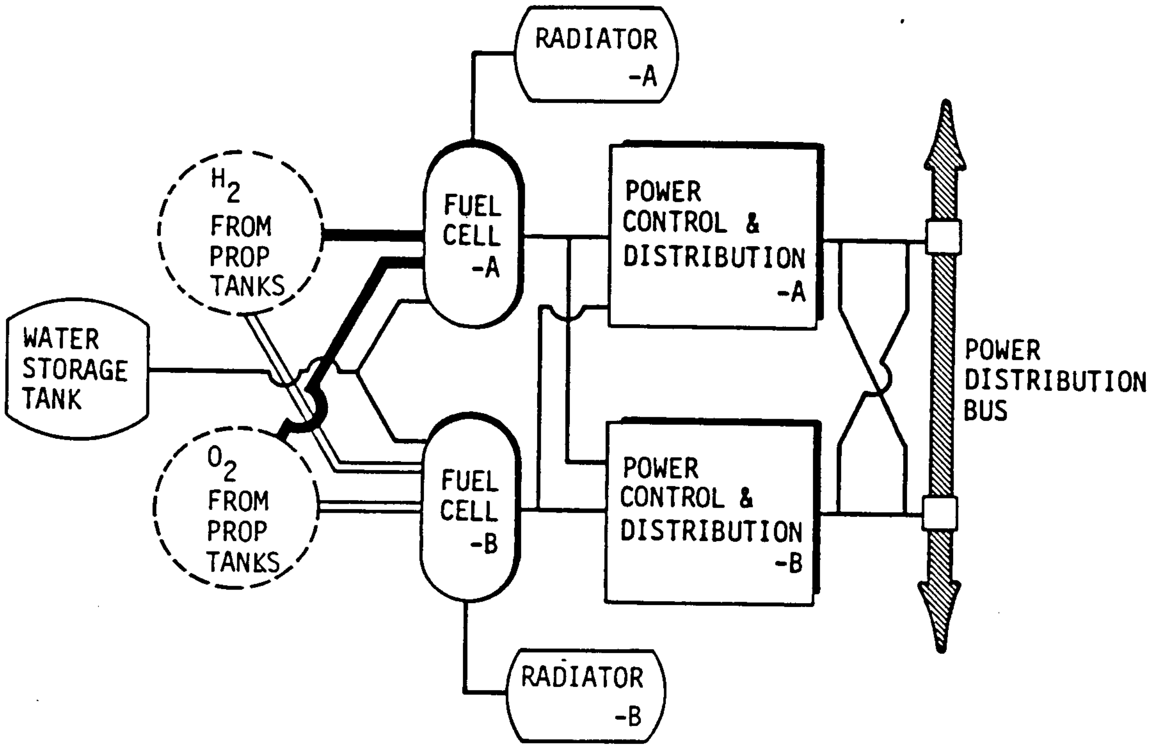


Figure 6.1.2.2.4-2 EPS Configuration for the Space-Based Cryogenic Configuration

6.1.2.2.5 Thermal Control (Space-Based Cyro) - These vehicles have essentially the identical thermal control system (TCS) as the ground-based cryo configuration except for mission duration. The avionics are mounted circumferentially and outboard on the avionics ring located at the payload/OTV interface. The outboard side of the ring is painted with a low alpha over epsilon paint. The avionics are housed in MMS-type boxes. The avionics components are mounted to the skirt in a manner which allows component waste heat to travel freely to the skirt. The location of the avionics on the ring will allow for the component waste heat to be evenly distributed among all the avionics. This reduces supplemental heater power requirements.

The fuel cell TCS is sized for a nominal 25-day OTV flight duration which requires two 25-ft^2 radiators to dissipate fuel cell waste heat. The radiators are located on the avionics ring simplifying the cooling loop system and reducing its weight. The two radiators are mounted on opposite sides of the vehicle to accommodate long duration fixed OTV orientation with respect to the sun vector, thus preventing fuel cell overheating.

All H_2 and O_2 cryo tanks are insulated with 1.0 inch (50 layers) of LI. The main propellant feedline insulation consists of 2 layers of gold foil.

Meteoroid protection is provided for propellant tankage with a stand-off thin wall aluminum bumper outside the multi-layer insulation. The MLI functions as a catcher for meteoroid impact particles as well as an insulation.

6.1.2.3 System Weight Summary - Space-Based Cryo

The flight vehicle weight for the initial space-based cryogenic configuration (55K lb propellant capacity) is presented in Table 6.1.2.3-1. Dry weight, non propulsive fluids and usable propellants are summarized. Dry weight is categorized according to the groupings requested by MSFC, and the individual items include a 15% contingency. Table 6.1.2.3-2 shows a detailed dry weight breakdown within each group, including the contingency weight assigned.

Table 6.1.2.3-3 and 6.1.2.3-4 show the equivalent information for the growth space-based cryogenic configuration (81K lb propellant capacity).

Table 6.1.2.3-1 Stage Weight Summary - Space-Based Cryo 55K Propellant Load

WBS GROUP	WEIGHT (lb)
2. Structure	1775
3. Propellant Tanks	672
4. Propulsion Less Engine	1137
5. Main Engine	626
6. Reaction Control System	304
7. Guidance, Navigation, Control	184
8. Communications & Data Handling	257
9. Electrical Power	357
10. Thermal Control System	167
11. Aerobrake	1885
Dry Weight Total	7364
12. Fluids	
Reactants, Coolants & Residuals	
Residual - FU (LH_2)	118
Residual - OX (LO_2)	707
FC Coolant	15
Inert Weight Total	8204
Usable Main Propellants	
FU-LH_2 (Incl. FPR)	7739
OX-LO_2 (Incl. FPR)	46436
Ignition Weight Total	62379

Mass Fraction

$$\frac{54175 \text{ (Main Prop Incl FPR)}}{62379 \text{ (Ignition Weight)}} = 0.87$$

Table 6.1.2.3-2 Detailed Dry Weight Breakdown - Initial Space-Based Cryo 55Klb Propellant Capacity

WBS GROUP	ELEMENT	WEIGHT (LB)	
2	Structures		
2.1	Air Frame		1015
	Center Truss	485	
	Fwd Truss	244	
	Fittings - Center Truss	56	
	Aerobrake Truss	98	
	Contingency	132	
2.2	Thrust Structure		97
	Engine Truss	84	
	Contingency	13	
2.3	Equipment Mounts		128
	REMS & Accumulators	22	
	Electrical Equip	37	
	Avionics	52	
	Contingency	17	
2.4	Payload Attachment		198
	Adapter & Avionics Ring	172	
	Contingency	26	
2.5	Micometeroid Shield		199
	Bumper	139	
	Stand-off	34	
	Contingency	26	
2.6	Handling and Storage Structure		138
	Crane Interface	30	
	RMS Grapple Fixtures	30	
	Cradle Interface	60	
	Contingency	18	
	Group 1 Total		1775
3	Propellant Tanks		
3.1	Tank Structure		399
	LH2 (2)	236	
	LO2 (2)	111	
	Contingency	52	
3.2	Tank Mounting		273
	LH2 (2)	129	
	LO2 (2)	108	
	Contingency	36	
	Group 3 Total		672
4	Propulsion Less Engine		
4.1	Press. Pneumatic Sys		50
	Lines, Valve, X-Ducer	47	
	Contingency	8	
4.2	Propellant Feed, Vent & Drain - Fuel		265
	Feed	139	
	Vent & Drain	91	
	Contingency	35	

Table 6.1.2.3-2 Detailed Dry Weight Breakdown - Initial Space-Based Cryo 55Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
4.3	Propellant Feed Vent & Drain - Ox.		264
	Feed	138	
	Vent & Drain	91	
	Contingency	34	
4.4	Prop Utilization System		245
	Probes, Sensors, etc.	73	
	Acquisition System	140	
	Contingency	32	
4.5	Misc System		313
	Eng. Removal Q/D	272	
	Contingency	41	
	Group 4 Total		1137
5	Main Engine		
5.1	Engine (2) Advanced 7.5k		480
5.2	Actuators (4) Elec		64
5.3	Contingency		82
	Group 5 Total		626
6	Reaction Control System		
6.1	REM Assy		69
	REM (6)	60	
	Contingency	9	
6.2	Accumulators		71
	GH2	55	
	GO2	7	
	Contingency	9	
6.3	Plumbing & Installation		95
	Valves, Lines, Switches	83	
	Contingency	12	
6.4	Conditioning Units		69
	Turbo Pmp Assy	35	
	Gas Generator	5	
	Heat Exchanger	20	
	Contingency	9	
	Group 6 Total		304
7	Guidance Navigation & Control		
7.1	Control & Guidance		159
	Flight Controller & TLM	60	
	IMU Processor	48	
	GPS Receiver	20	
	Thrust Controller	10	
	Contingency	21	
7.2	Navigation		25
	Star Scanner	22	
	Contingency	3	
	Group 7 Total		184

Table 6.1.2.3-2 Detailed Dry Weight Breakdown - Initial Space-Based Cryo 55Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
8	Communications & Data Handling		
8.1	Communications		199
	GPS Antenna System	15	
	STDN/TDRS Xponder	32	
	20w RF Power Amp	12	
	S Band RF System	100	
	TLM Power Supply	14	
	Contingency	26	
8.2	Data Management		58
	Central Computer Mass Mem	20	
	CMD & Data Handling	30	
	Contingency	8	
8.3	Video		0
	Group 8 Total		257
9	Electrical Power		
9.1	Fuel Cell System		103
	Fuel Cell	70	
	Fuel Cell Plumbing	20	
	Contingency	13	
9.2	Radiator System		58
	Radiator	35	
	Plumbing	15	
	Contingency	7	
9.3	Residual H20 System		17
	Tank Accumulator	10	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Plumbing		29
	Plumbing	25	
	Contingency	4	
9.5	Power Distribution		150
	Wire Harness, Connector,Etc	130	
	Contingency	20	
	Group 9 Total		357
10	Thermal Control		
10.1	Insulation		109
	MPS Tanks	91	
	ACS Tanks	3	
	FC Water Tank	1	
	Contingency	14	
10.2	Thermal Control		58
	Engine Truss & Compartment	16	
	Prop Lines & F/C System	24	
	Electrical & Plumbing	10	
	Contingency	8	
	Group 10 Total		167

Table 6.1.2.3-2 Detailed Dry Weight Breakdown - Initial Space-Based Cryo 55Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
11	Aerobrake		
11.1	Aeroassist Device		1082
	Center Dome - (Fixed)	174	
	Quilt TPS (Flex)	767	
	Contingency	141	
11.2	Doors & Mechanism		111
	Doors	97	
	Contingency	14	
11.3	Support Structure		692
	Ribs and Struts	629	
	Contingency	063	
	Group 11 Total		1885
15	Propellants		
15.1	Main		55000
	Usable - FU LH2 (inc. FPR)	7739	
	Usable - OX LO2 (inc. FPR)	46436	
	Residual - FU LH2	118	
	Residual - OX LO2	707	
15.2	FC Coolant		15
	Coolant	15	
	Group 15 Total		55015

Table 6.1.2.3-3 Stage Weight Summary - Space-Based Cryo 81K Propellant Load

WBS GROUP	WEIGHT (LB)
2. Structures	1821
3. Propellant Tanks	835
4. Propulsion Less Engine	1171
5. Main Engine	626
6. Reaction Control System	304
7. Guidance, Navigation, Control	184
8. Communications & Data Handling	257
9. Electrical Power	357
10. Thermal Control System	210
11. Aerobrake	1885
Dry Weight Total	7650
12. Fluids	
Reactants, Coolants & Residuals	
Residual - FU (LH_2)	174
Residual - OX (LO_2)	1041
FC Coolant	15
Inert Weight Total	8880
Usable Main Propellants	
FU-LH_2 (Incl. FPR)	11397
OX-LO_2 (Incl. FPR)	68388
Ignition Weight Total	88665

Mass Fraction

$$\frac{79785 \text{ (Main Prop Incl FPR)}}{88665 \text{ (Ignition Weight)}} = 0.90$$

Table 6.1.2.3-4 Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity

WBS GROUP	ELEMENT	WEIGHT (LB)	
2.1	Structures		
	Air Frame		1015
	Center Truss	485	
	Fwd Truss	244	
	Fittings - Center Truss	56	
	Aerobrake Truss	98	
	Contingency	132	
2.2	Thrust Structure		97
	Engine Truss	84	
	Contingency	13	
2.3	Equipment Mounts		128
	REMS & Accumulators	22	
	Electrical Equip	37	
	Avionics	52	
	Contingency	17	
2.4	Payload Attachment		198
	Adapter & Avionics Ring	172	
	Contingency	26	
2.5	Micometeroid Shield		245
	Bumper	177	
	Stand-off	36	
	Contingency	32	
2.6	Handling and Storage Structure		138
	Crane Interface	30	
	RMS Grapple Fixtures	30	
	Cradle Interface	60	
	Contingency	18	
	Group 2 Total		1821
3.1	Propellant Tanks		
	Tank Structure		562
	LH2 (2)	330	
	LO2 (2)	159	
	Contingency	73	
3.2	Tank Mounting		273
	LH2 (2)	129	
	LO2 (2)	108	
	Contingency	36	
	Group 3 Total		835
4	Propulsion Less Engine		
4.1	Press. Pneumatic Sys		50
	Lines, Valve, X-Ducer	42	
	Contingency	8	
4.2	Propellant Feed, Vent & Drain - Fuel		265
	Feed	139	
	Vent & Drain	91	
	Contingency	35	

Table 6.1.2.3-4 Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
4.3	Propellant Feed Vent & Drain - Ox.		264
	Feed	138	
	Vent & Drain	91	
	Contingency	34	
4.4	Prop Utilization System		279
	Probes, Sensors, etc.	83	
	Acquisition System	160	
	Contingency	36	
4.5	Misc System		313
	Eng. Removal Q/D	272	
	Contingency	41	
	Group 4 Total		1171
5	Main Engine		
5.1	Engine (2) Advanced 7.5K		480
5.2	Actuators (4) Elec		64
5.3	Contingency		82
	Group 5 Total		626
6	Reaction Control System		
6.1	REM Assy		69
	REM (6)	60	
	Contingency	9	
6.2	Accumulators		71
	GH2	55	
	GO2	7	
	Contingency	9	
6.3	Plumbing & Installation		95
	Valves, Lines, Switches	83	
	Contingency	12	
6.4	Conditioning Units		69
	Turbo Pmp Assy	35	
	Gas Generator	5	
	Heat Exchanger	20	
	Contingency	9	
	Group 6 Total		304
7	Guidance Navigation & Control		
7.1	Control & Guidance		159
	Flight Controller & TLM	60	
	IMU Processor	48	
	GPS Receiver	20	
	Thrust Controller	10	
	Contingency	21	
7.2	Navigation		25
	Star Scanner	22	
	Contingency	3	
	Group 7 Total		184

Table 6.1.2.3-4 Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
8	Communications & Data Handling		
8.1	Communications		199
	GPS Antenna System	15	
	STDN/TDRS Xponder	32	
	20w RF Power Amp	12	
	S Band RF System	100	
	TLM Power Supply	14	
	Contingency	26	
8.2	Data Management		58
	Central Computer Mass Mem	20	
	CMD & Data Handling	30	
	Contingency	8	
8.3	Video		0
	N/A	0	
	Group 8 Total		257
9	Electrical Power		
9.1	Fuel Cell System		103
	Fuel Cell	70	
	Fuel Cell Plumbing	20	
	Contingency	13	
9.2	Radiator System		58
	Radiator	35	
	Plumbing	15	
	Contingency	8	
9.3	Residual H20 System		17
	Tank Accumulator	10	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Plumbing		29
	Plumbing	25	
	Contingency	4	
9.5	Power Distribution		150
	Wire Harness, Connector,Etc	130	
	Contingency	20	
	Group 9 Total		357
10	Thermal Control		
10.1	Insulation		152
	MPS Tanks	125	
	ACS Tanks	5	
	FC Water Tank	2	
	Contingency	20	
10.2	Thermal Control		58
	Engine Truss & Compartment	16	
	Prop Lines & F/C System	24	
	Electrical & Plumbing	10	
	Contingency	8	
	Group 10 Total		210

Table 6.1.2.3-4 Detailed Dry Weight Breakdown - Growth Space-Based Cryo 81Klb Propellant Capacity (Continued)

WBS GROUP	ELEMENT	WEIGHT (LB)	
11	Aerobrake		
11.1	Aeroassist Device		1082
	Center Dome - (Fixed)	174	
	Quilt TPS (Flex)	767	
	Contingency	141	
11.2	Doors & Mechanism		111
	Doors	97	
	Contingency	14	
11.3	Support Structure		692
	Ribs and Struts	629	
	Contingency	63	
	Group 11 Total		1885
15	Propellants		
15.1	Main		81000
	Usable - FU LH2 (inc. FPR)	11397	
	Usable - OX LO2 (inc. FPR)	68388	
	Residual - FU LH2	174	
	Residual - OX LO2	1041	
15.2	FC Coolant		15
	Coolant	15	
	Group 15 Total		81015

6.1.2.4 Performance on Model Missions: Space Based Cryo

The following is a summary of the groundrules and assumptions used in the performance analyses contained herein. This description applies not only to the data presented in section 6.1.2.4 but to the analyses in section 6.2.2.4 as well.

For space based OTV GEO missions, the delta v's used are shown in Table 6.1.2.4-1.

Table 6.1.2.4-1 Space-Based OTV GEO Delivery Delta V's

BURN	PURPOSE (orbit dimensions in nmi)	PLANE CHANGE (DEG)	PROPULSIVE DELTA-V (FPS)
1	270 circ. to 270 x 19322.9	2.26	7856.4
2	270 x 19322.9 to 19322.9 circ.	26.24	5855.8
3	19322.9 circ. to 45 x 19322.9	28.50	6049.7
-	Aeropass maneuver to 32.9 x 270	0.00	0.0
4	32.9 x 270 to 270	0.00	535.0

The above are ideal, impulsive delta-v's. Gravity induced velocity losses were added to the initial perigee burns as a function of the burn time involved. Boiloff was accounted for at the rate of 2.8 lbs/hr.

Lunar mission OTV delta-v's are shown in Table 6.1.2.4-2

Table 6.1.2.4-2 SPACE BASED OTV LUNAR DELIVERY DELTA V's

BURN	PURPOSE (orbits dimensions i nmi)	PROPULSIVE DELTA-V (FPS)
1	270 circ. to trans-lunar inject	11350.0
2	outbound midcourse correction	150.0
3	inject into 70 nmi circ. lunar orbit	2870.0
4	70 nmi circ. lunar to trans-earth	2870.0
5	return leg midcourse correction	150.0
-	aeropass maneuver	------
5	circularize with space station	535.0

The delta v's used for plnetary missions were derived from a hypothetical launch geometry which minimizes the OTV delta-v penalty incurred due to precession of the Space Station orbit while the OTV is away. No attempt was made to research actual launch window geometries and there was assumed to be no plane change required to get from the Space Station orbit to the departure hyperbola at launch time. Since each planetary mission has a unique delta-v budget, we have not listed the planetary delta-v's in tabular form. More information on planetary mission analysis methodology is contained in Reference 8.

For some OTV configurations on some planetary missions it was necessary to add an expendable kick stage (EKS) to the payload. For such cases, the specific orbital energy (or C3) at which the OTV shuts down and the kick stage takes over was chosen to minimize the gross weight of the OTV + EKS + payload. In all cases where an EKS was used, they were sized by assuming a mass fraction of 0.95 and an Isp of 310 seconds.

Table 6.1.2.4-3 presents the propellant loads required to perform each of the low model missions that can be captured by the initial space based cryogenic OTV. Table 6.1.2.4-4 presents the equivalent data for the two stage version of the growth space based OTV. These data were used in the programmatic trade studies discussed in Volume III. Figure 6.1.2.4-1 presents a summary of the performance capability of the initial space based OTV recommended for development as a result of programmatic evaluations of the Rev 8 Low Mission Model.

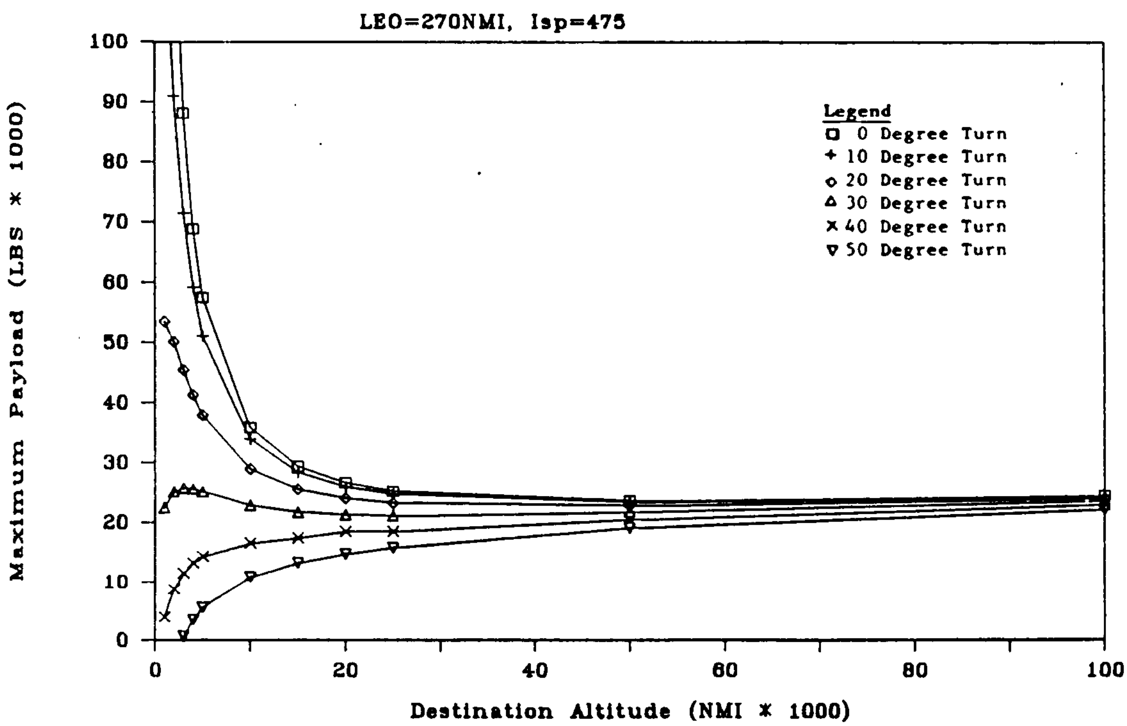


Figure 6.1.2.4-1 Space-Based 55Klb OTV Performance Capability

Table 6.1.2.4-3 Performance analysis for Required Missions
Cryogenic, Growth Space-Based, 55K OTV

Isp = 475 sec

Rev 8 Mission	P/L Up (lb)	P/L Dn (lb)	OTV Propellant (lb)	
Geosynchronus Missions				
13006	12017	0	40596	
13700	20000	0	53099	
18073	12000	0	40571	
18040	20000	0	53099	
18722	20000	0	53099	
18912	12000	2000	43749	
10100	20000	0	53099	
13002	7000	4510	36017	
15700	7500	7500	45636	
15008	13159	0	42366	
15009	13310	0	42366	
15701	12000	2000	43749	
19031	12000	0	40571	
19035	20000	0	53099	
Lunar Missions				
17200	5000	0	29743	
17202	20000	0	53493	
Planetary Missions				EKS Mass
17074	3497	0	50179	15188
17075	5000	0	30014	10458
17078	2205	0	55762	0
17084	4410	0	32228	0

Table 6.1.2.4-4 Performance Analysis for Growth Lunar Missions
Cryogenic, Two Stage, Space-Based, 81K OTV

Isp = 475 sec

Mission	P/L Up(lb)	P/L Dn(lb)	OTV Propellant (lb) Stage 1	Stage 2	Total
Lunar Missions					
17203	80000	15000	81915	81915	163830
17204	80000	0	77325	77325	154650
17205	80000	10000	80884	80884	161768

6.2.1 GROUND BASED STORABLE

6.2.1.1 General Arrangement (Ground Based Storable) - Two ground based storable configurations have been defined. The first is a perigee kick stage packaged to be carried aloft in the Aft Cargo Carrier (Figure 6.2.1.1-1). The second is an identical capability stage packaged to be carried in the Orbiter cargo bay (Figure 6.2.1.1-2).

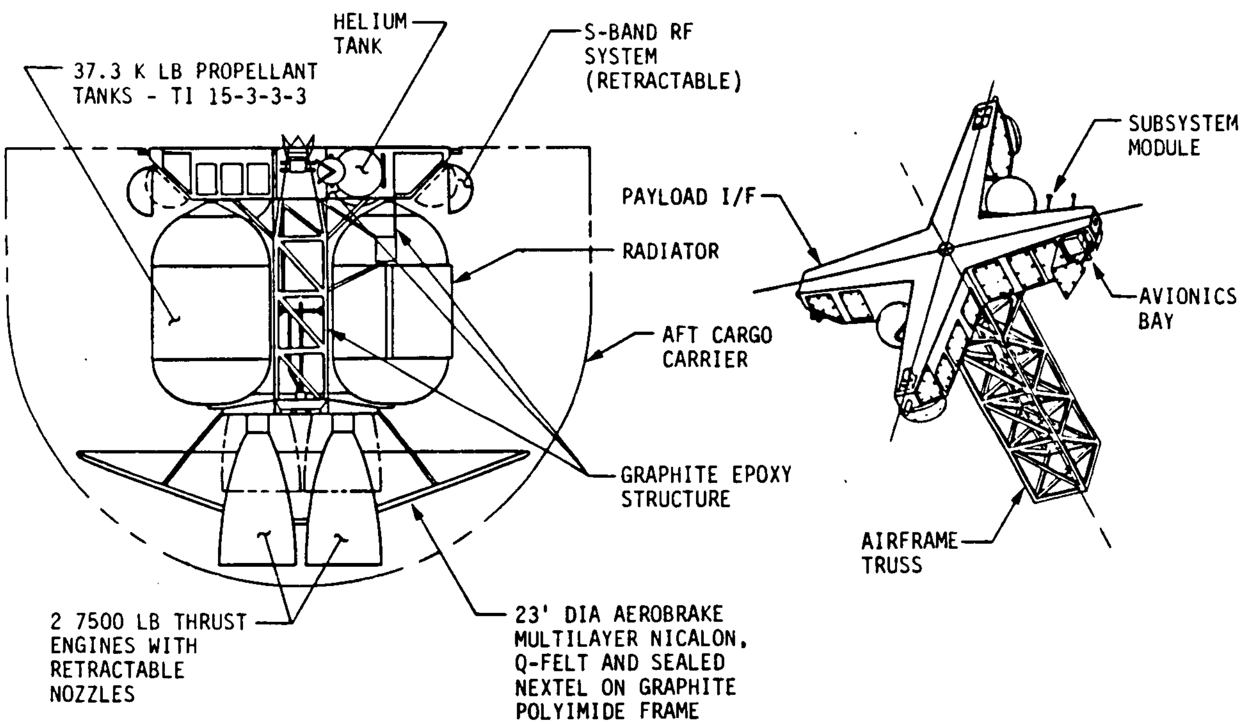


Figure 6.2.1.1-1 Ground-Based Storable - ACC

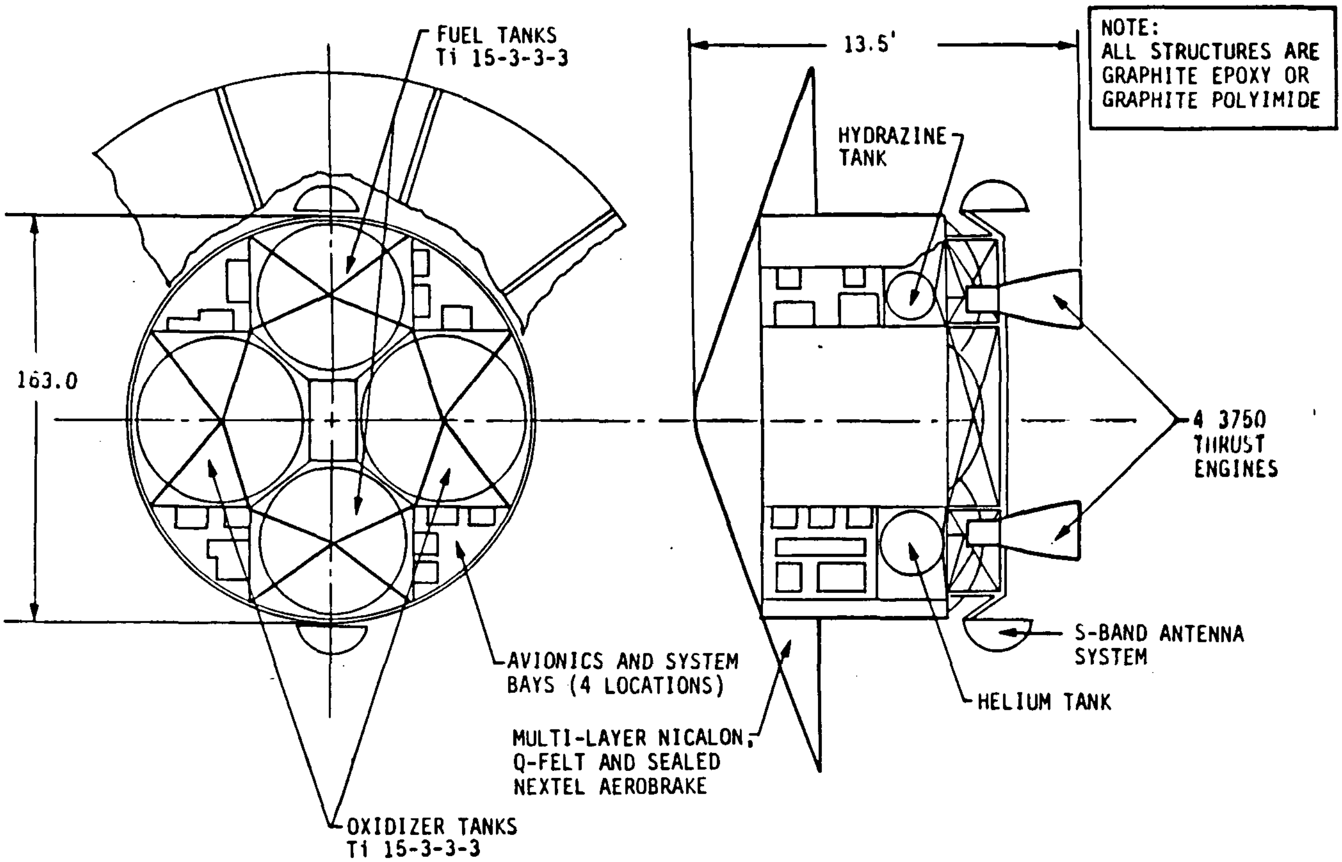


Figure 6.2.1.1-2 Ground-Based Storable - Cargo Bay

ACC -- The general arrangement of the ACC configured storable OTV was selected to maximize commonality with the space based configuration described in paragraph 6.2.2. The propellant tanks all use the same diameter tooling, and the concept of extending the engine through the heat shield, enabling payload retrieval, is retained. The stage is built around a subsystem module and an airframe truss. The subsystem module provides mounting space for the avionics components and support for the main propellant tanks. The airframe truss supports the tanks laterally and provides the attachment for the main engines and the aerobrake. Both the subsystem module and the airframe truss will be constructed of graphite epoxy composite materials to minimize weight. The main tanks are sized to contain 37,300 pounds of storable propellants and will be constructed of 15-3-3-3 titanium. The tank size selected is adequate to perform the mission model in 1993 and 1994 within the projected 72,000 lb lift capability of the Space Transportation System (STS) in that timeframe. The main propulsion system engines are two XLR-132 engines scaled up to 7500 pounds thrust. Extendable exit cones will be provided to an exit ratio of 600 to 1. The nozzles will be extended through the aerobrake while firing and retracted inside the brake contour during the aerobraking maneuver. The aerobrake is 23 feet in diameter and will be constructed of a multilayer fabric material of Nicalon, Q-felt, and Nextel sealed with RTV. The fabric will be supported on a graphite polyimide frame or honeycomb. The center section will contain the door through which the engine nozzles will extend and retract. The physical dimensions and weight of the storable OTV require the aerobrake to be no more than 23 feet in diameter. As shown in Figure 6.2.1.1-1, this allows the fully deployed aerobrake to fit within the dimensions of the Aft Cargo Carrier. No deployment mechanisms will be required. At the end of the mission the outer torus of the aerobrake will be jettisoned and the remainder of the OTV will be installed in the Orbiter bay for return to earth. No further disassembly of the OTV is required to fit within the envelope of the P/L bay. More details of this concept are shown in Figure 6.2.1.1-3

CARGO BAY -- We have designed a minimum length OTV (Figure 6.2.1.1-2) that will fit within the envelope of the STS P/L Bay and still leave adequate space for the longest payloads identified in the Mission Model. Commonality with subsequent space based designs has been sacrificed to obtain short stage length. Fuel and oxidizer tanks are different diameters to achieve equal, short tank length, and the aerobrake has been mounted on the forward end of the stage, enabling use of four short engines tucked into the corners between tanks. The overall length has been held to approximately 13.5 feet which leaves 46.5 feet for payload and ASE. The four main propulsion propellant tanks, sized for 37,300 pounds of storable propellant, are of 15-3-3-3 titanium alloy and supported in a truss and skin structure of graphite epoxy. The subsystem equipment is fitted into the quadrants between the tanks. The main propulsion system will use the XLR-132 engines with 3750 pounds thrust. Fixed nozzles with an expansion ratio of 400 to 1 were selected to minimize length. A 23 foot diameter deployable aerobrake was selected using the same multilayer fabric material selected for the ACC OTV aerobrake design over graphite polyimide support structure. The center support structure is honeycomb and the outer portion is rib construction. Since the aerobrake will be mounted on the forward end of the stage, the payload interface, instead of the engines, will penetrate the heat shield. Thus this perigee stage will have no capability to retrieve payloads in the aerobraked operational mode. No payload retrieval is required in this mode. At the end of the mission, the outer portion of the brake will be discarded and the remainder of the OTV will be installed in the Orbiter P/L bay for return to earth.

6.2.1.2 Subsystem Summary Description (Ground-Based Storable)

6.2.1.2.1 Aeroassist (Ground-Based Storable) - The overall layouts of the ground-based storable configurations were shown in Figures 6.2.1.1-1 through -3. The aeroshield devices used with these configurations are similar in concept to those used on the cryogenic configurations discussed in Section 6.1 The diameter has been reduced to 23 feet as the more compact storable stages are more easily protected from the aerodynamic wake and afterbody re-circulation. This smaller diameter results in a higher ballistic coefficient than that experienced on the ground based cryogenic vehicle. The resulting surface insulation temperature is higher, and its thickness is correspondingly greater. The resulting aeroassist parameters are summarized in Table 6.2.1.2.1-1. Note that the stage configured to be carried aloft in the Orbiter cargo bay has its engines on the opposite end from the aeroshield, and no door is needed to permit their use. Note also that the ACC version is small enough to fit within the ACC envelope without being folded. We have maintained the concept of RSI on the central portion of the brake and FSI on the outboard portion.

Table 6.2.1.2.1-1 Ground-Based Storable Aeroshield

W/C_DA	Brake Dia (ft)	TPS	q_{max} (Btu/Ft^2 sec)	Q_{max} (Btu/Ft^2)	t_{max} (°F)	TPS Thick (In.)	W_{Br} / W_{Ret}	Design Load psf Ctr	Design Load psf Otbd
6.2	23	FSI	15.7	3700	2570	0.43	0.14	35	28
		RSI	18.8	4440	2360	0.43			

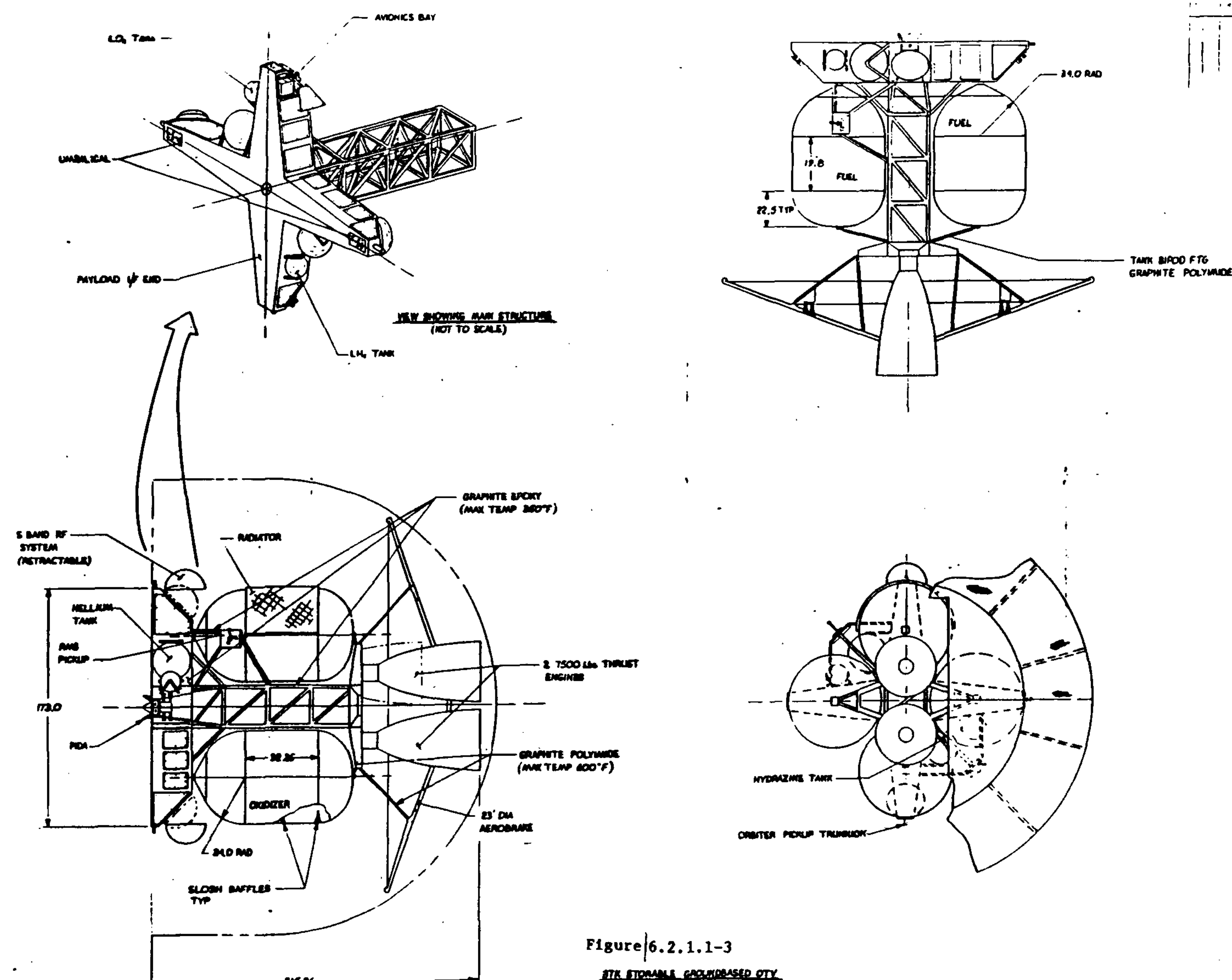


Figure 6.2.1.1-3

STR STORABLE GROUNDBASED OTV
ALL STRUCTURE COMPOSITE
TANK STRUCTURE 15-3-3 TITANIUM
(REF DRAWING GSOR-015)

GSOR-027-01

6.2.1.2.2 Propulsion (Ground-Based Storable) - The propulsion characteristics are shown in Table 6.2.1.2.2-1 and the RCS characteristics are shown in Table 6.2.1.2.2-2. Two main engines provide 7500 lbs of thrust each and are based on technology (XLR-132) currently under development at AFRPL. The engine is a gas generator cycle that uses oxidizer cooling and provides a specific impulse of 345.7 seconds at a chamber pressure of 1500 psia and expansion ratio of 600:1. The engine weighs 253 lbs with a two position nozzle, and can be gimballed up to $\pm$ 16° to track the center of gravity for engine out operations.

The XLR-132 technology is currently being developed by AFRPL. The expendable engine can be available in 1989 and the reusable engine in 1992 under the current planning and funding schedules. As currently designed, the 3750 lbf engine has a life of 1020 seconds and a capability for 10 starts. Based on discussions with AFRPL, the life is expected to be extended to 5 hours with a possible performance reduction. Reusable XLR-132 engine studies are planned to begin in mid 1985.

The propellants, MMH and N_2O_4 are stored in a four tank configuration which can be returned to the ground in the shuttle payload bay without disassembly. The tanks are manifolded together in parallel flow configuration so that each propellant will be depleted simultaneously. A propellant utilization system is included to assure that the usable propellants in each tank can be depleted with a minimum residual propellant weight. This system consists of propellant utilization probes that provide both a continouous output of liquid level and discrete points to update propellant usage data during engine firings to cancel cumulative errors periodically as the propellant is consumed. The outputs of the computer can be used to vary the consumption from individual tanks to maintain liquid levels within acceptable limits, or to vary the engine mixture ratio to achieve simultaneous depletion of usable propellants. Propellant start traps are also provided.

The pressurization gas is helium stored in high pressure vessels and regulated by electronic pressure regulators. The MMH tank pressure is lower than the N_2O_4 tank because of the lower engine interface pressure; 17 psia for MMH vs 37 psia for N_2O_4. This required an additional regulator at the MMH tank. These are shown in Figure 6.2.1.2.2-1. The nominal flight operating pressure for MMH is 20 psia and for N_2O_2 is 50 psia. This assumes 3 psi delta-P for frictional losses and allows for a 10 psi rise in the N_2O_4 tank during coast because of its higher vapor pressure. The storage system also provides helium for engine valve actuation, main engine turbine spin-up, and purge of oxidizer and fuel pump seals.

Table 6.2.1.2.2-1 Ground-Based Storable MPS Summary

o ENGINE	-	TWO ENGINES, 7500 LBS THRUST EACH, GAS GENERATOR CYCLE I_{SP} = 345.7 SEC AREA RATIO 600:1
o PROPELLANT DISTRIBUTION	-	DUAL TANK, PARALLEL FEED, START TRAP
o PRESSURIZATION	-	HELIUM STORED GAS, REGULATOR CONTROLLED TO PROPELLANT TANKS
o VENT	-	GROUND VENT ONLY--NONE DURING ASCENT AND FLIGHT. REDUCE PRESSURE PRIOR TO RETURN IN CARGO BAY
o VALVE ACTUATION	-	ELECTRICAL AND HELIUM, COMMON WITH STAGE PRESSURIZATION SUPPLY
o PROPELLANT UTILIZATION	-	TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL
o CARGO BAY RETRIEVAL	-	RETURN STAGE IN CARGO BAY AND DETANK RESIDUALS ON THE GROUND
o THERMAL PROTECTION	-	HEATER BLANKETS AND MULTILAYER INSULATION
o PROXIMITY OPERATIONS	-	TWO FAULT TOLERANT
o REDUNDANCY	-	FAIL SAFE

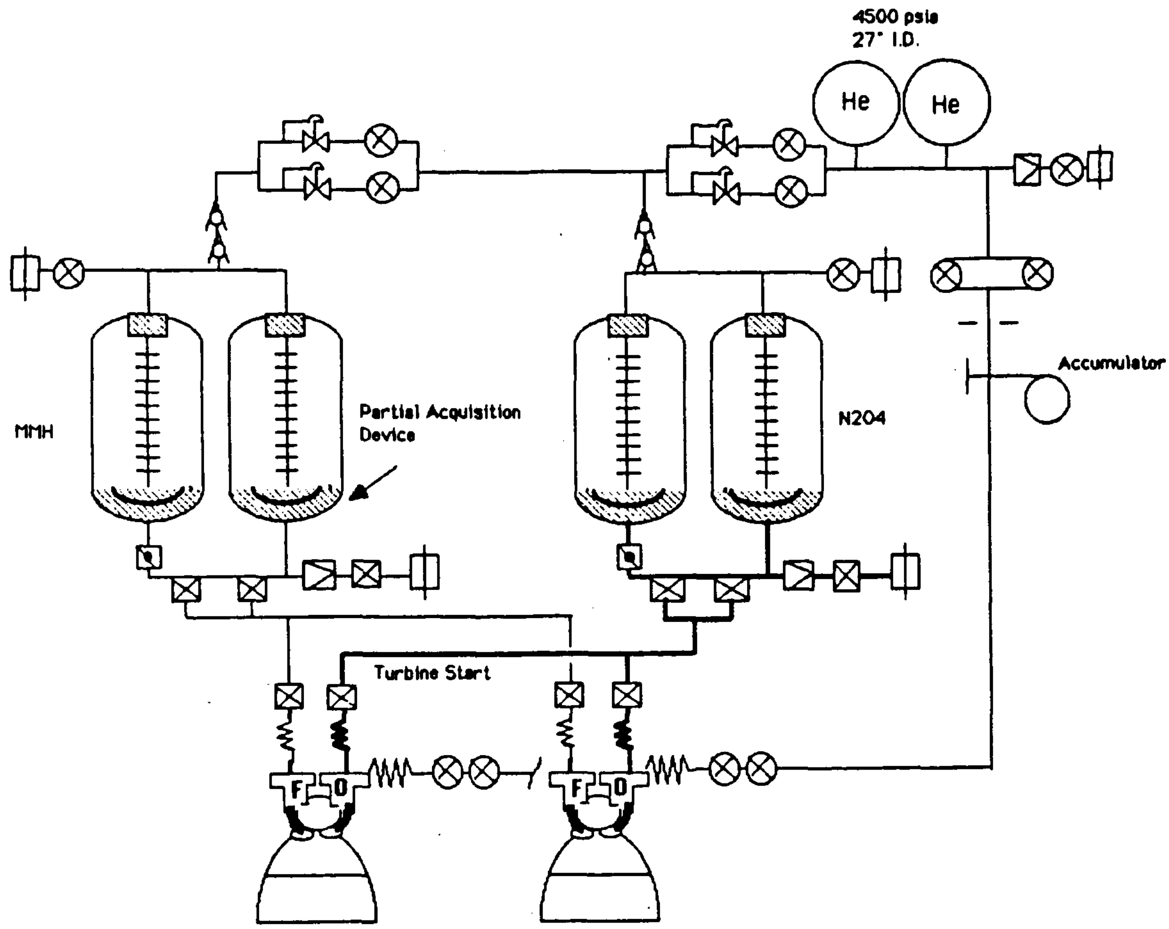


Figure 6.2.1.2.2-1 Ground-Based N_2O_4/MMH Schematic

Table 6.2.1.2.2-2 Ground-Based Storable RCS Summary

- o PROPELLANT
  - HYDRAZINE (N_2H_4) I_{SP} = 230 SEC
- o ROCKET ENGINE MODULE
  - 30 LB, 7 ENGINES PER MODULE
  - 14 THRUSTERS
  - 3 DOF and -X TRANSLATION
  - FAIL OPERATIONAL
- o PROPELLANT SUPPLY
  - 24" DIAMETER TANK
  - POSITIVE EXPULSION
  - 400 PSI 2.3:1 BLOWDOWN

150 LBS OF HYDRAZINE MAXIMUM

- o SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS

The vent system has a vent and relief valve on each tank which provide over pressure relief capability and the capability to depressurize the ullage to acceptable values prior to stowage of the OTV in the cargo bay.

The stage will be returned to the ground in the shuttle cargo bay. Propellants will not be dumped to space; unless future analysis shows the combined stage/propellant weight exceeds 32,000 lbs, which is the shuttle gross payload landing capability; the center of gravity is not within the shuttle center of gravity limits; or if the propellant slosh magnitude is not acceptable to the orbiter's autopilot system during reentry. At present the ground-based stage does not exceed these requirements. Residual propellants will be detanked on the ground.

The thermal protection system includes heaters, and multilayer insulation to maintain propellant temperature within acceptable limits.

For proximity operations near the shuttle, the system is dual fault tolerant and for flight operations, the system is fail safe as shown in Figure 6.2.1.2.2-1.

The reaction control is the same as that use for the ground based cryo (Figure 6.1.1.2.2-3). It uses hydrazine monopropellant pressurized by nitrogen gas operating in a blowdown mode from 400 psi. Fourteen thrusters provide three degrees of freedom and +X translation. The thrusters provide an ISP of 230 seconds. The propellant supply is one 24-inch diameter titanium tank, having a usable propellant capacity of 150 lbs of hydrazine at a 2.3:1 blowdown ratio. The RCS is two fault tolerant for proximity operations.

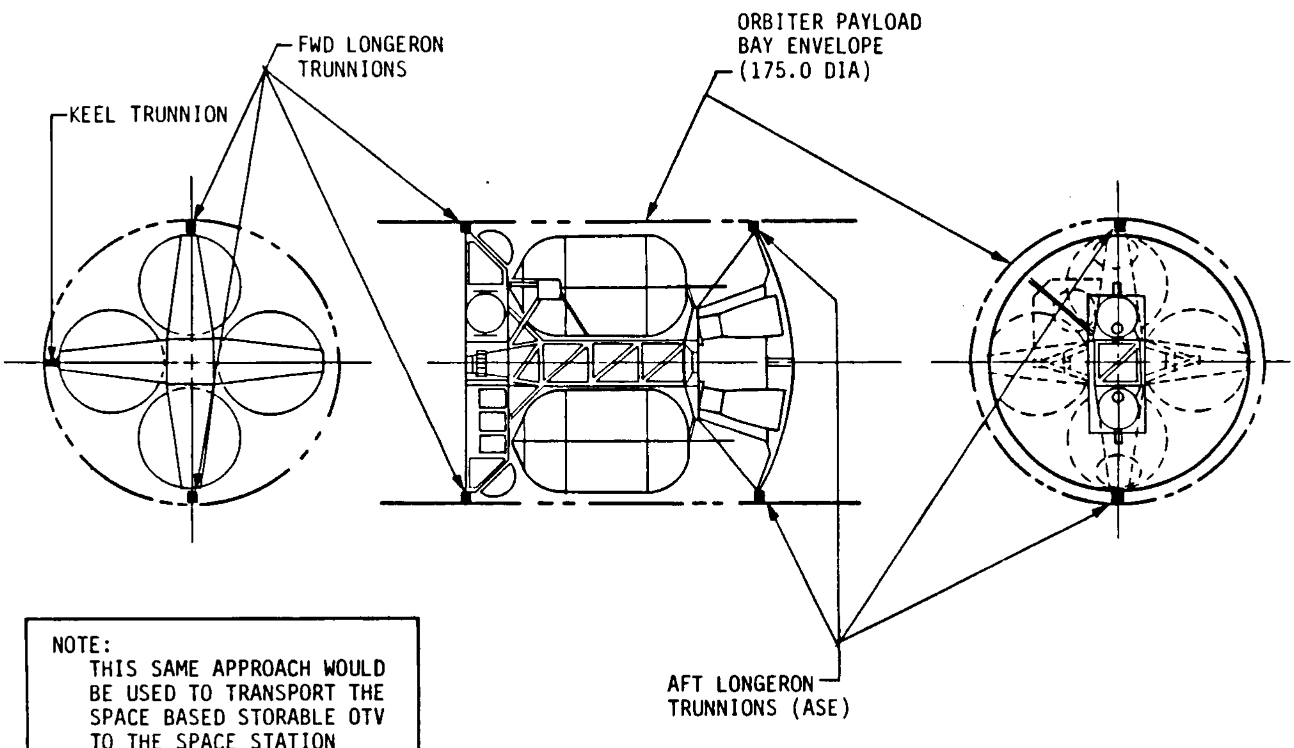
6.2.1.2.3 Structures and Mechanisms (Ground-Based Storable) - The ACC configuration shown in Figures 6.2.1.1-1 and -3 consists of two 68 in. diameter cylindrical MMH tanks with .75 ellipse lower heads and two 68 in. diameter cylindrical N_2O_4 oxidizer tanks with .75 ellipse lower heads. Two 7500 lb thrust engines are mounted on a center core truss arrangement. Tanks are supported off conical shaped forward heads by a cross beam arrangement. The 23 ft diameter aerobrake is mounted on the engine support by graphite polyimide struts.

The vehicle is delivered to space in the ACC. With only a 23 ft brake required, a fixed Viking shaped aerobrake can be used. Interface with the ACC and payload adapter is at the crossbeam end of the vehicle. Umbilical provisions with the ACC are also on that end. Avionics are installed on the outside surfaces of the forward crossbeam to provide ready access for replacement.

The vehicle will be returned to earth in the cargo bay of the orbiter after jettisoning the fabric aerobrake. A grapple fixture is provided on the OTV to interface with the orbiter RMS.

The storable tanks are of fusion welded construction of heat treated 15-3-3-3 titanium. If problems are uncovered during the alloy test program, the back up alloy would be 6AL-4V titanium. The conical heads are made in five pieces with a machined cone shaped cap for tank pick up and four formed and chem milled gores. The elliptical tank head is of similar construction but with a formed center piece. Minimum tank membrane gage is 0.017 inches. The tank barrel is made in two pieces. Main structural crossbeam and engine thrust beam are fabricated from graphite epoxy. Lower tank support beams are of graphite polyimide construction. Aerobrake has a center core of graphite polyimide honeycomb covered with ceramic foam tiles. The remainder of the aerobrake is Nextel, nicalon, ceramic felt, and RTV construction. Generally, similar construction is used on the cargo bay configuration shown in Figure 6.2.1.1-2.

Airborne support equipment (ASE) considerations for the ACC OTV are shown in Figure 6.2.1.2.3-1. The ground-based storable ACC OTV will be supported in the orbiter bay for retrieval at five locations - three forward, two aft. The upper forward mounts will consist of two trunnion and scuff plate fittings mounted to the OTV crossbeam. The trunnion and scuff plate fittings will be of aluminum alloy and located to mate with the orbiter sill longeron bridge fitting. The lower forward mount will consist of one aluminum alloy base plate and trunnion fitting attached also to the OTV crossbeam and mating with an orbiter bay keel fitting. The upper aft mounts will consist of two trunnions and scuff plates attached to the OTV aerobrake structure and mating at the outboard ends with orbiter bay sill longeron bridge fittings. The trunnion and scuff plate fittings will be of aluminum alloy and located to mate with the orbiter sill longeron fittings. The two aft fittings are to be carried up in orbiter payload bay and attached to OTV at rendezvous.



NOTE:
THIS SAME APPROACH WOULD BE USED TO TRANSPORT THE SPACE BASED STORABLE OTV TO THE SPACE STATION

Figure 6.2.1.2.3-1 Ground-Based Storable OTV ASE

The payload bay configured storable OTV concept will be mounted for ascent and descent using the same ASE. The required design has not been finalized, but a cradle will be required. The cradle will pick up longeron and keel fittings at the rear of the cargo bay and cantilever the perigee stage, apogee stage and payload stack. For very long payloads, it is likely that longeron fittings on the payload would be beneficial.

6.2.1.2.4 Avionics (Ground-Based Storable) - The ground-based storable avionics configuration (Figure 6.2.1.2.4-1) is essentially identical to the ground-based cryo configuration (Section 6.1.1.2.4). Fuel cell reactant tankage must be added since it cannot be main tank fed as it was in the cryo configuration. Table 6.2.1.2.4-1 reflects the subsystem and system unit and total values for power and weight for this configuration. A more detailed description of this system is provided in Reference 7.

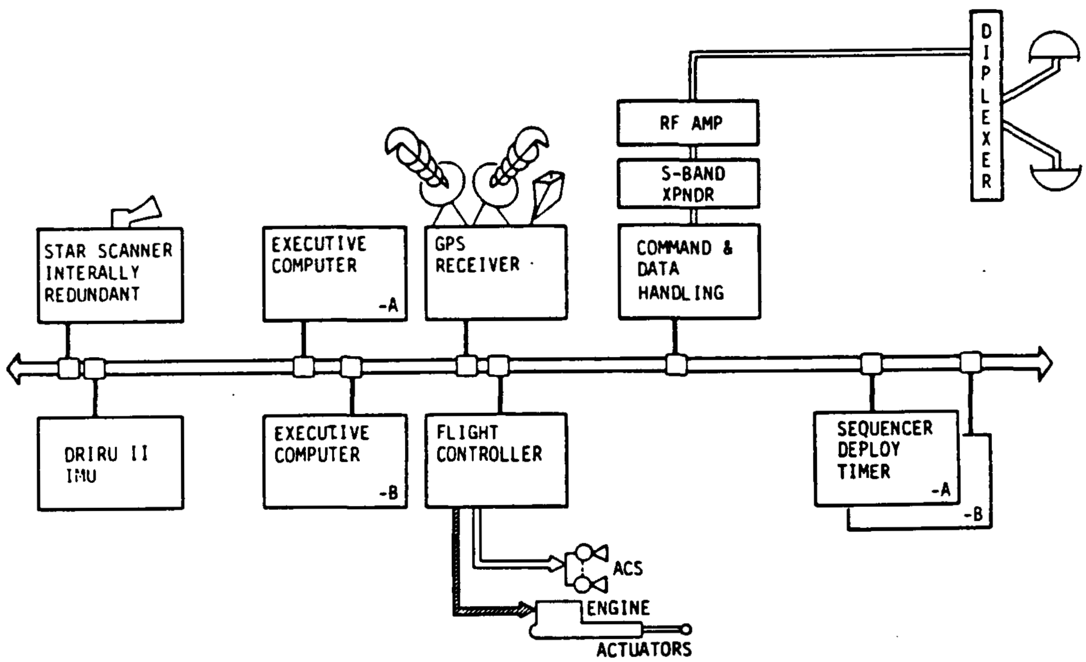


Figure 6.2.1.2.4-1 Block Diagram of the Ground-Based Storable Configuration

6.2.1.2.4.1 Guidance, Navigation and Control - The Guidance, Navigation and Control subsystem is essentially the same as in Section 6.1.1.2.4.1. The engine controller is deleted as it is not required for the storable engine.

6.2.1.2.4.2 Data Management - The Data Management subsystem is the same as in Section 6.1.1.2.4.2.

Table 6.2.1.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 1 of 2)

Subsystem Equipment	Weight (lb)	Power (w)	Size (in) H W L	Total Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
GN&C							
Star Scanner	12	7	7x 7x20	1	12	7	5
IMU	37	25	6x12x16	1	37	25	25
GPS Receiver	45	35	8x12x16	1	45	35	15
GPS Antenna-Low Alt	5		6x 6x10	2	10		
GPS Antenna-Hi Alt	5		18x18x26	1	5		
Flight Controller	45	120	8x 8x16	1	45	120	120
			Subsystem Total		154	187	165
Data Management							
Executive Computer & Mass Memory	10	60	6x 8 x 9	2	20	120	85
			Subsystem Total		20	120	85
Telemetry and Command							
Command & Data Handling	15	35	6x 8x10	1	15	35	22
TLM Power Supply	7	10	4x 7x 7	1	7	10	5
Deploy Timer	6	6	3x 4x 7	2	12	12	4
			Subsystem Total		34	57	31

Table 6.2.1.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 2 of 2)

Subsystem Equipment	Weight (lb)	Power (w)	Size (in) H W L	Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
Communications and Tracking							
STDN/TDRS Xponder	16	55	6x 6x14	1	16	55	55
20w RF Power Amp	6	125	3x 6x10	1	6	125	40
S-Band RF System	90	30		2	180	60	30
Subsystem Total					202	240	125
EPS							
Fuel Cell (FC)	35		11x12x12	2	70		
FC Radiators	35		$35ft^2$x2"	1	35		
FC Reactants	35				35		
FC Reactant Tank LH_2 & Plumb	42				42		
FC Reactant Tank LO_2 & Plumb	38				38		
FC Coolant	10				10		
FC H_2O Tank	13				13		
Power Control & & Distribution	27	10	6x 8x12	2	54	20	20
Heaters		50				50	50
Engine Power		200				200	
Subsystem Total					297	270	70
System Total					707	874	476

6.2.1.2.4.3 Telemetry and Command - The Telemetry and Command subsystem is the same as in Section 6.1.1.2.4.3.

6.2.1.2.4.4 Communications and Tracking - The Communications and Tracking subsystem is the same as in Section 6.1.1.2.4.4.

6.2.1.2.4.5 Electrical Power - The fuel cell reactant tanks are added to this configuration as reactants cannot be taken from the propellant system. The EPS design (Figure 6.2.1.2.4-2) is essentially the same as in Section 6.1.1.2.4.5. Fuel cells are sized for 1.1 kw including a 20% design margin.

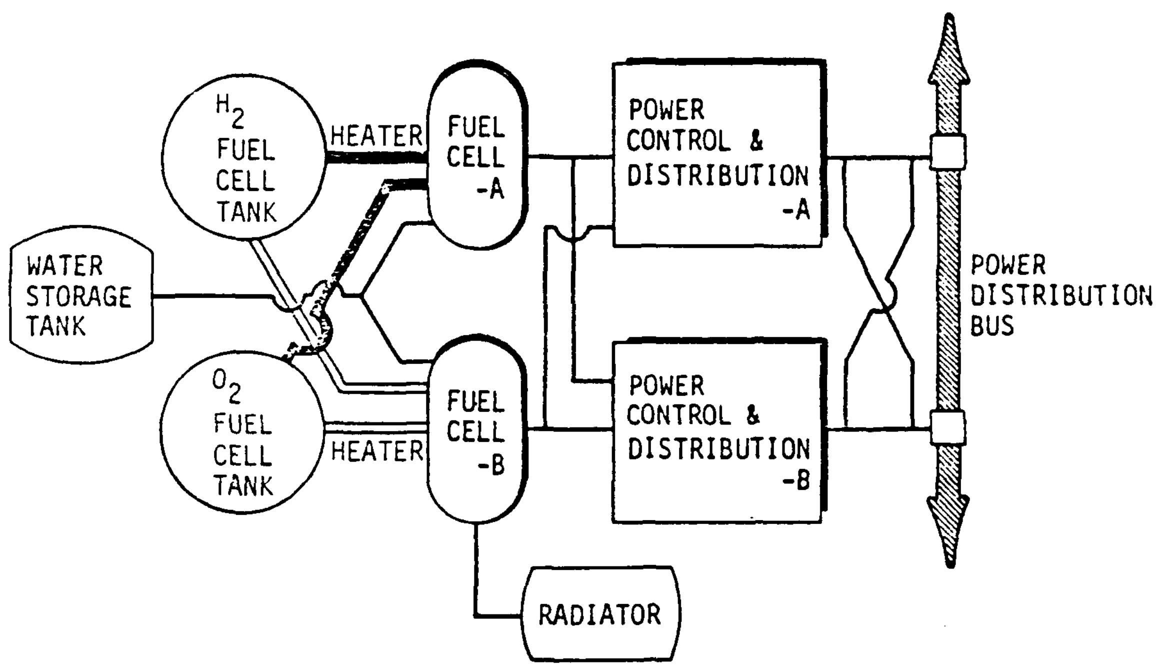


Figure 6.2.1.2.4-2 EPS Configuration for Ground-Based Storable Configuration

6.1.1.2.5 Thermal Control (Ground-Based Storable) - This configuraion utilizes a fuel cell power sytem. The fuel cell thermal control system is sized for an OTV continuous flight power requirement of 1.1. KW and a 2 day flight duration. The fuel cell thermal control system requires 35 sq ft of radiator area to dissipate the fuel cell waste heat effectively. The radiator faces outboard and is mounted on one of the oxidizer tanks.

The avionics are to be passively cooled with the components mounted to high mass heat sink structures which face outboard and have a low alpha/epsilon paint. Some components may require thermostatically controlled heater strips. The avionics bay structure protects the avionics and thermal blankets on all or most of the avionics components since they have no direct view to space. Component surface finishes and mounting techniques shall be defined as the OTV design develops.

The payload/OTV interface is made nearly adiabatic. Approximately 25 to 50 layers of insulation blanket (double aluminized Kapton MLI) is located at the interface.

The OTV reaction control system (RCS) requires thermal protection for the RCS tank, feedlines, and propulsion modules. The RCS tank has an MLI blanket (10 layers) and strip heaters. The feedlines contain hydrazine (freezing point of 35°F) and require low power strip heaters (approximately 25 watts) and one or two layers of thermally insulating blankets. The RCS modules and feedlines will be maintained at sufficiently high temperatures by "thermal pulsing" techniques (i.e., periodic module firing).

The N_2O_4 and MMH tanks shall be insulated with two layers of Kapton thermal blankets. Thermostatically controlled strip heaters are used to maintain sufficiently high propellant tank temperatures. The short flight duration should enable the propellant system capacitance to maintain acceptable propellant temperatures with a minimal supplemental heater power requirement.

The helium pressurant tank requires MLI, since the tank is of composite construction. Thermostatically controlled strip heaters are necessary to ensure adequate helium temperature/pressures.

The heating effects of the engine nozzles are of no concern.

6.2.1.3 System Weight Summary - Ground-Based Storable - Total Flight vehicle weight for the ground-based storable configuration is presented in Table 6.2.1.3-1 for the ACC concept, in Table 6.2.1.3-2 for the cargo bay concept. Dry weight is developed in detail. Propulsive and nonpropulsive fluid loadings are summarized. The breakdown has not been rearranged into MSFC's suggested format as this approach was deleted from contention at an earlier stage in the study.

Table 6.2.2.3-1 System Weight Summary (Ground-Based Storable)

Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - ACC Configured

Description	Weight (lb)		
Structure			826
Basic Airframe Truss		120	
OX Tank		196	
Tank (2)	158		
Fwd & Aft Attach	38		
Fu Tank		96	
Tank (2)	60		
Fwd & Aft Attach	36		
Engine Truss & Attach		101	
Engine Truss	67		
Hardpoints - Engine Attach	30		
Attach - Engine Actuators	4		
Subsystem Module		208	
PIDA Grapple Fixture		20	
RMS Grapple Fixture & Struts		30	
Orbiter Pickup Trunnion (3)		15	
P/L or ACC Attach		40	
Aerobrake Assy - 23 ft			538
Environmental Control			117
Meteoroid Protection		N/A	
Thermal Control/Protection		117	
Ox Tanks - MLI & Heater Tape	41		
Fu Tanks - MLI & Heater Tape	35		
Engine Truss - MLI	1		
Hydrazine Tank - MLI & Heater Tape	11		
He Tanks (2) - MLI	1		
Engine Compartment - MLI	5		
Propellant Lines, Components, MLI, Heater Tape, Coatings, etc.	23		

Table 6.2.1.3-1 Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - ACC Configured
(Continued)

Description		Weight (lb)	
Main Propulsion System			939
Engine (2): XLR-132 7500# Thrust Ea		506	
Propellant Disbribution System		194	
Pressurization/Pneumatic System		170	
ACS Common Feed		N/A	
PU/Acquisition System		N/A	
Instrumentation		5	
Actuators (4) - Electrical		64	
Orientation Control			116
ACS Subsystem - Hydrazine		116	
Rocket Engine Modules (2)	32		
Mounting - REMS	3		
Hydrazine Tank (1)	41		
Mounting - Tank	6		
Propel. Distribution & Pressurization	34		
Electrical Power			352
Fuel Cell System		70	
Reactant Tank (LH2) & Plumbing		42	
Reactant Tank (LO2) & Plumbing		38	
Radiator System		35	
Residual H_2O System		13	
Wire Harness, Connectors, etc.		116	
Mounting Provisions		38	

Table 6.2.1.3-1 Weight Statement
Space-Based Storable
37.3K Propellant Load
Pge Stage - ACC Configured
(Continued)

Description	Weight (lb)	
Avionics		456
Avionics (Equipment List)	410	
Mounting Provisions	46	
Dry Weight		3344
Contingency (15%)		502
Total Dry Weight		3846
Main Propellant (MR 2:1 Ox Wt to Fu Wt)		37300
N2O4	24866	
MMH	12434	
Pressurant (He) - MPS		13
ACS Propellant & Pressurant		157
N2H4	150	
N2	7	
FC Reactant & Coolant		45
Reactant	35	
Coolant	10	
Total Loaded Weight		41361

$$\lambda = \frac{37300 \text{ (Main Propellants)}}{41361 \text{ (Ignition Weight)}} = 0.902$$

Table 6.2.1.3-2 Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - P/L Bay Configured

Description	Weight (lb)		
Structure			1340
Basic Airframe Truss		410	
Ox Tank		250	
Tank (2)	180		
Fwd & Aft Attach	70		
Fu Tank		208	
Tank (2)	142		
Fwd & Aft Attach	66		
Engine Truss & Attach		375	
Engine Truss	243		
Hardpoints & Struts	124		
Attach - Engine Actuators	8		
PIDA Grapple Fixture		20	
RMS Grapple Fixture & Struts		30	
Orbiter Pickup Trunnion (3)		15	
P/L Attach		40	
Aerobrake Assy - 23 ft			542
Environmental Control			117
Meteoroid Protection		N/A	
Thermal Control/Protection		117	
Ox Tanks - MLI & Heater Tape	41		
Fu Tanks - MLI & Heater Tape	35		
Engine Truss - MLI	1		
Hydrazine Tank - MLI & Heater Tape	11		
He Tanks (2) - MLI	1		
Engine Compartment - MLI	5		
Propellant Lines, Components, MLI, Heater Tape, Coatings, etc.	23		

Table 6.2.1.3-2 Weight Statement
Ground-Based Storable
37.3K Propellant Load
Pge Stage - P/L Bay Configured
(Continued)

Description		Weight (lb)	
Main Propulsion System			975
Engine (4): XLR-132 3750# Thrust Ea		456	
Propellant Disbribution System		216	
Pressurization/Pneumatic System		170	
ACS Common Feed		N/A	
PU/Acquisition System		N/A	
Instrumentation		5	
Actuators (8) - Electrical		128	
Orientation Control			116
ACS Subsystem - Hydrazine		116	
Rocket Engine Modules (2)	32		
Mounting - REMS	3		
Hydrazine Tank (1)	41		
Mounting - Tank	6		
Propel. Distribution & Pressurization	34		
Electrical Power			352
Fuel Cell System		70	
Reactant Tank (LH2) & Plumbing		42	
Reactant Tank (LO2) & Plumbing		38	
Radiator System		35	
Residual H20 System		13	
Wire Harness, Connectors, etc.		116	
Mounting Provisions		39	

Table 6.2.1.3-2 Weight Statement
Space-Based Storable
37.3K Propellant Load
Pge Stage - P/L Bay Configured
(Continued)

Description	Weight (lb)	
Avionics		456
Avionics (Equipment List)	410	
Mounting Provisions	46	
Dry Weight		3906
Contingency (15%)		586
Total Dry Weight		4492
Main Propellants (MR 2:1 Ox Wt to Fu Wt)		37300
N2O4	24866	
MMH	12434	
Pressurant (He) - MPS		13
ACS Propellant & Pressurant		157
N2H4	150	
N2	7	
FC Reactant & Coolant		22
Reactant	12	
Coolant	10	
Total Loaded Weight		41984

$$\lambda = \frac{37300 \text{ (Main Propellants)}}{41984 \text{ (Ignition Weight)}} = 0.888$$

6.2.1.4 Performance on Model Missions: Ground-Based Storables - The ground-based storable stage is used as a perigee stage, requiring the use of an additional expendable kick stage to perform some part of the mission, such as circularizing the payload at geosynchronous altitude. In such cases, the EKS was assumed to have a mass fraction of 0.95, an Isp of 310 seconds, and was sized to be just large enough to perform the mission at hand, i.e. "custom fit" to each particular mission. For this reason the performance curve graphs for the GB ACC storable perigee stage is qualitatively different than those for the cryogenic stages which performed their missions "solo". Table 6.2.1.4-1 summarizes the propellant load required and the gross weight of this stage on each of the Rev. 8 model missions on which it will be used. Gross weight includes the weight of the required kick stage. Figure 6.2.1.4-1 summarizes the performance of this stage to high circular orbits assuming full utilization of the Shuttle launch capability (72,000 lb to 140 nmi when launched east from KSC), and the use of expendable apogee kick stages.

Table 6.2.1.4-1
Performance Analysis for Required Missions Storable, Ground-Based ACC, 37.3K Perigee Stage
(Rev. 7 Missions)

Isp = 345.7 sec .

MISSION	P/L UP (lb)	P/L DN (lb)	OTV PROP.(lb)	GROSS WT. OTV + EKS + P/L (lb)
19036	6000	0	18914	36780
19031	8025*	0	23560	45751
19031	9317*	0	26546	51498
18724	10000	0	28133	54543
18064	10163	0	28512	55271
13006	12017	0	32851	63571

*Early Year DoD Equivalent Payloads Projected by MMA

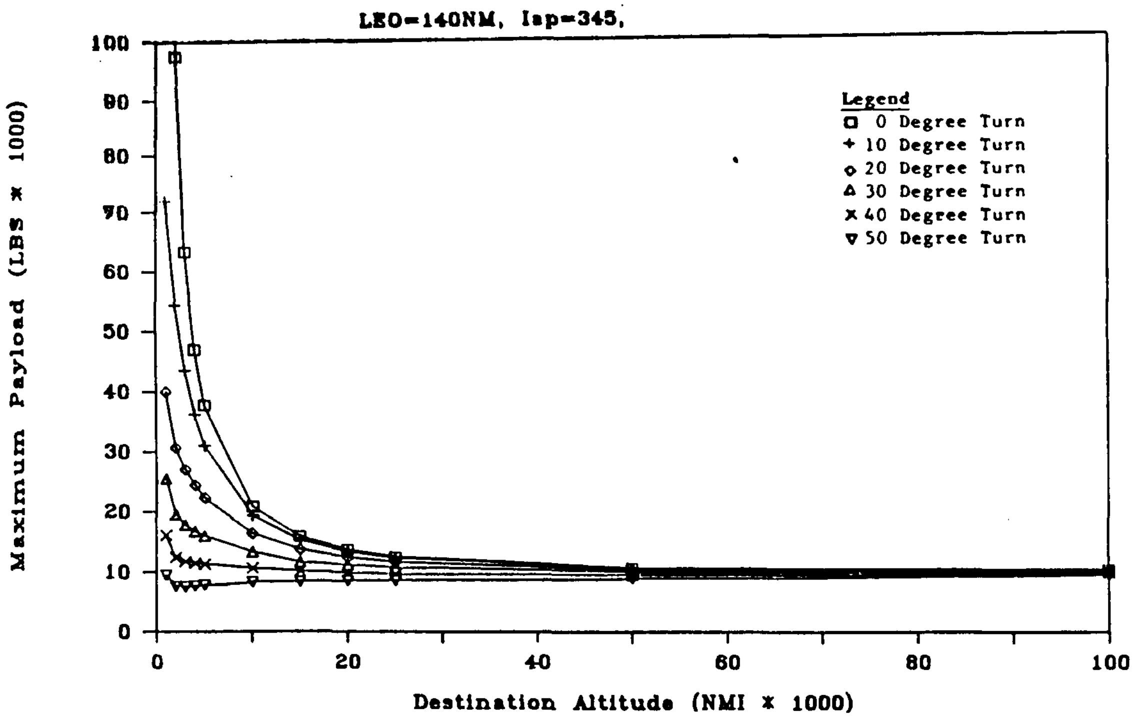


Figure 6.2.1.4-1 Ground-Based 37.3Klb Storable Performance Capability

6.2.2 Space-Based Storable Family

6.2.2.1 General Arrangement (Space-Based Storable) - The space-based storable family of vehicles uses one perigee stage configuration and two two-stage configurations to meet HEO and planetary mission requirements (Figures 6.2.2.1-1 to -3). These configurations use 3 stage sizes, further details of which are shown in Figures 6.2.2.1-4 to -6. Capture of the large lunar missions requires use of a low technology cryogenic perigee stage under the 53,000 lb propellant capacity storable stage.

Either evolving from an initially ground based OTV or starting as a space based OTV, our study shows the basic configuration of the storable space-based GEO delivery OTV, shown in Figure 6.2.2.1-1, should be essentially the same. The subsystem module/airframe truss forms an efficient structure to support the main propulsion system, propellant tanks, and avionics and electric power system equipment. Repackaging of the avionics and propulsion components is necessary to accommodate the space based maintenance activities involving a limited number of technicians for a minimum time. Accessibility to the equipment both by astronauts in EVA gear and by robotics has been a consideration in configuring the space-based vehicles. The basic ground-based vehicle and the space-based vehicles have much in common. The avionics components are essentially the same except for added redundancy. The main engines and feed system are the same 7500 lb thrust XLR-132 engines and feed system selected for the ACC ground-based OTV. The main propellant tanks are sized for the more ambitious missions identified for the later years when the space station will be operational. The tank diameter is the same as selected for the ground-based ACC OTV to assure common tooling between the potential ground based program and the space based program, and to enable delivery of the assembled OTV to the Space Station in the Orbiter payload bay. The OTV configuration selected for delivery of payloads to GEO is shown in Figure 6.2.2.1-1. The vehicle will operate as a perigee stage and an expendable AKM will be provided to insert the payloads into GEO-synchronous orbit. The aerobrake is constructed of the same materials as the ground based ACC brake. The size has been increased commensurate with the return weight of the vehicle. Although shown with a 25 ft diameter brake, this vehicle will be capable of being fitted with a 32 ft diameter brake when required to return with additional equipment (such as the multiple payload carrier) after delivering multiple payloads.

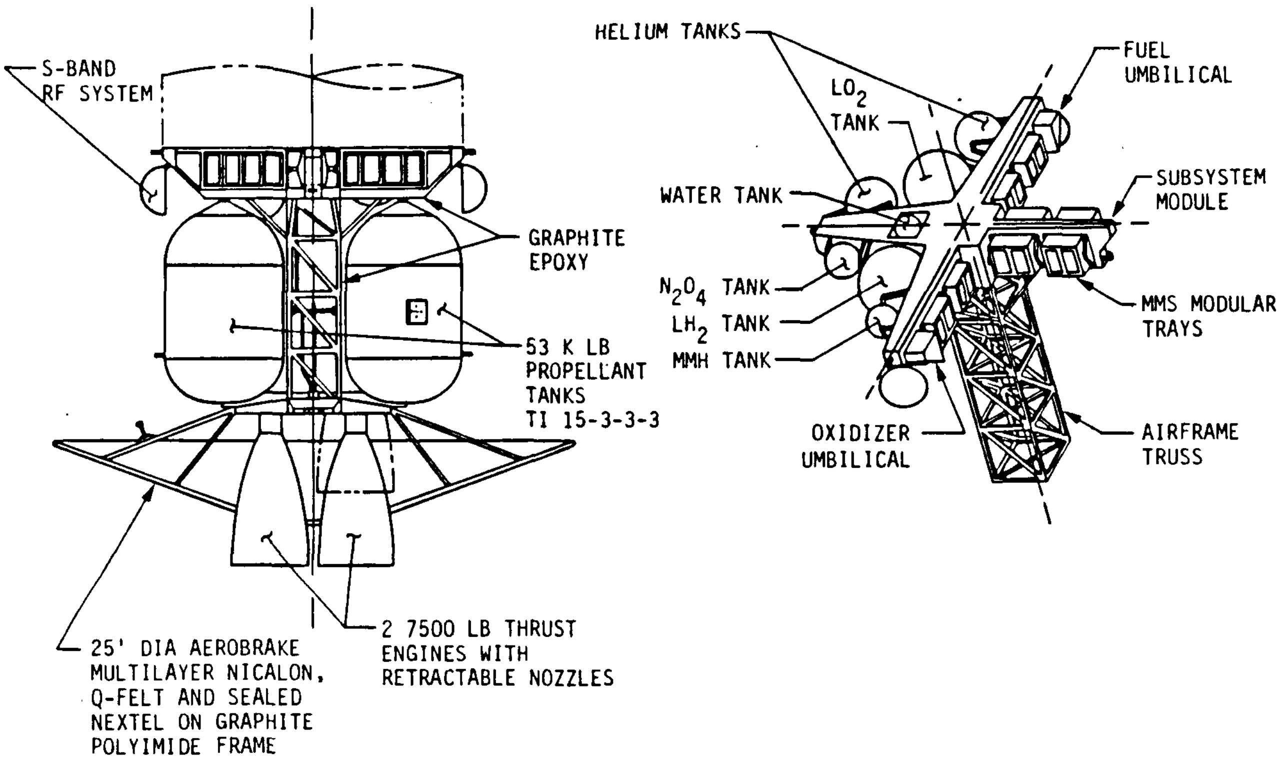


Figure 6.2.1.1-1 Space-Based GEO Delivery Vehicle

The unmanned servicing missions to GEO will be performed with a two stage vehicle (Figure 6.2.2.1-2) made up of the GEO delivery vehicle, described earlier, as the first stage and a smaller propellant capacity stage as the second stage. The first stage will perform the perigee burn, separate from the second stage and payload, coast out to GEO altitude, return for the aerobrake maneuver and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the payload into GEO orbit. The stage will stay in the vicinity of the servicer through the duration of the mission and then perform the deorbit burn to bring the servicer back to the Space Station. The smaller second stage is configured similar to the first stage but with smaller tanks and shorter airframe truss structure. The subsystem module, engines and feed system, avionics equipment, and electric power system are the same for both stages. Tanks for the EPS fuel cell reactants will be larger for the second stage because of the longer duration of the stage two mission. The diameter of the tanks for the second stage are the same as the first stage tanks in order that the tooling for the domes can be common. By welding the domes of the fuel tanks together with no barrel section and adding a short barrel section in the oxidizer tanks, the propellant capacity for the second stage is approximately 25,400 pounds. The aerobrake for the second stage is sized at 32 ft in diameter to bring the unmanned servicer back through the aeromanuever. Construction of the brake is the same for both stages.

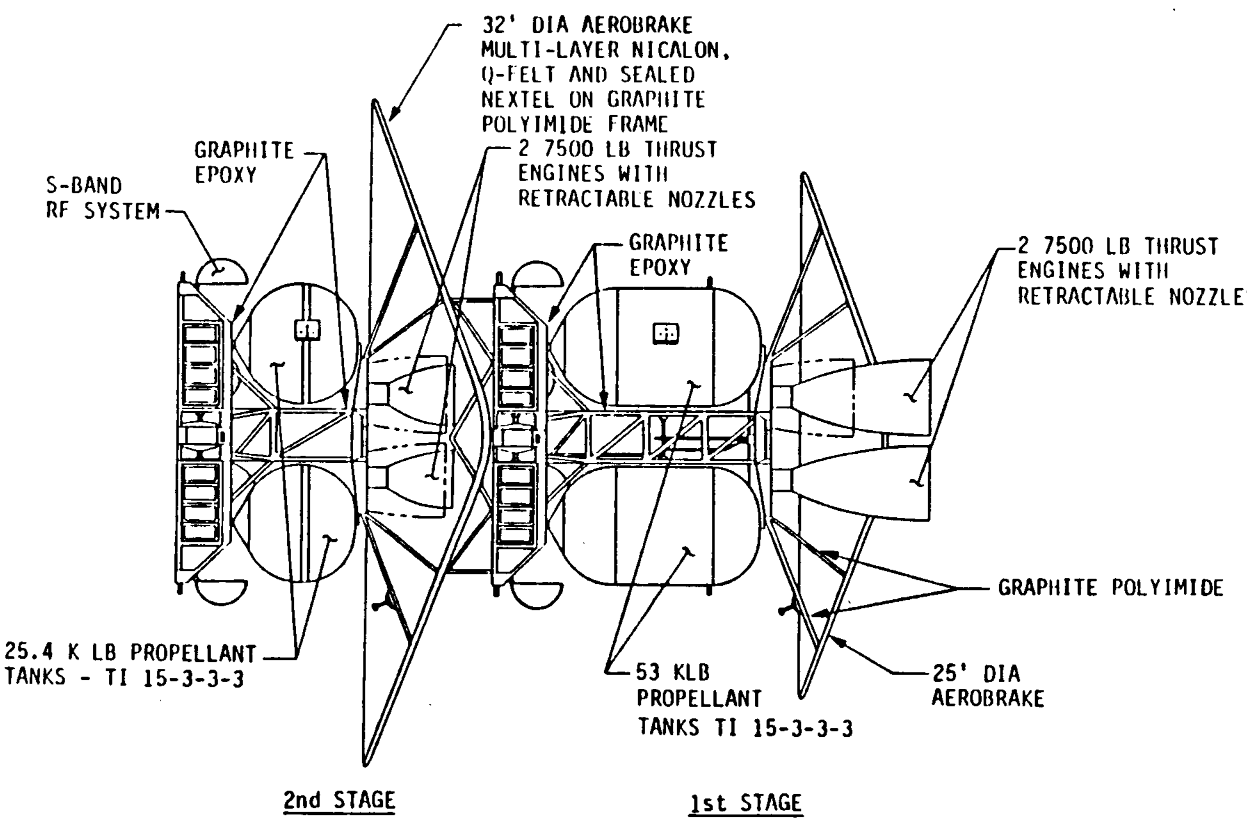


Figure 6.2.2.1-2 Space-Based Unmanned Servicing Vehicle

The manned servicing missions to GEO wil be performed with a two stage vehicle (Figure 6.2.2.1-3) made up of the GEO delivery vehicle, slightly reoutfitted, as the second stage and a larger propellant capacity stage as the first stage. The first stage will perform the perigee burn, separate from the second stage and manned capsule, coast out to GEO altitude, return for the aerobrake manuever and return to the Space Station. The second stage will continue the mission with the apogee burn to insert the manned payload into GEO orbit. The stage will stay in the vicinity of the manned capsule through the duration of the servicing mission and then perform the deorbit maneuvers to bring the servicer back to the Space Station. The larger first stage is configured for maximum commonality with the second stage. As for the small stage for the unmanned servicing vehicle, the major difference is in the tank length and the airframe truss. The tanks are lengthened, but retain the same diameter for tooling commonality, to provide capacity for 90,000 lb of propellant. Because of the mass of the complete vehicle/payload stack, two additional engines have been added. The four engine arrangement uses the same 7500 pound thrust XLR-132 engine but will require a different feed system. The 53,000 pound propellant capacity second stage is the same basic stage as the GEO delivery vehicle and the first stage of the unmanned servicing vehicle except with a larger diameter aerobrake and larger capacity fuel cell reactant tanks. The fuel cell reactant tanks are sized for support of the 24 day manned mission. The aerobrake is increased to 41 ft in diameter because of the weight of the returning stage and manned capsule.

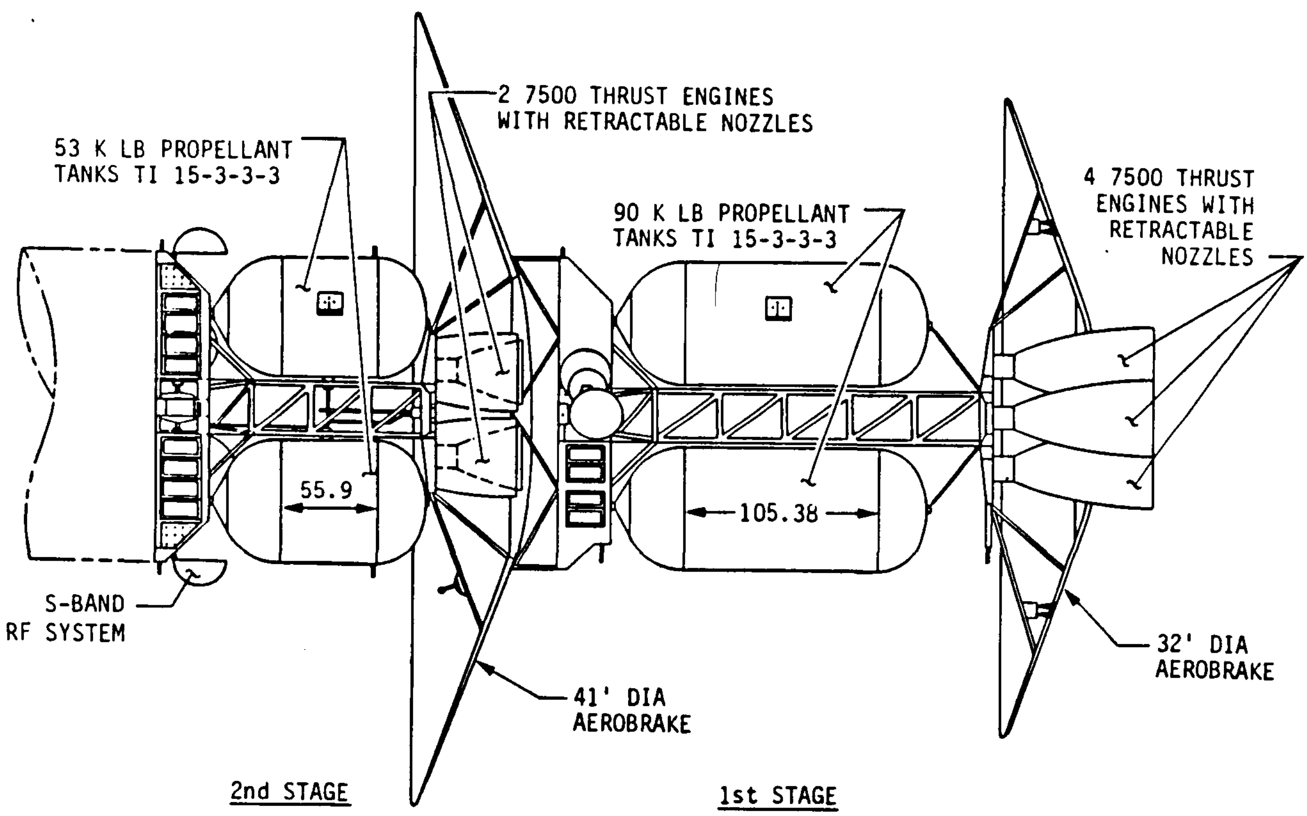


Figure 6.2.2.1-3 Space-Based Manned Servicing Vehicle

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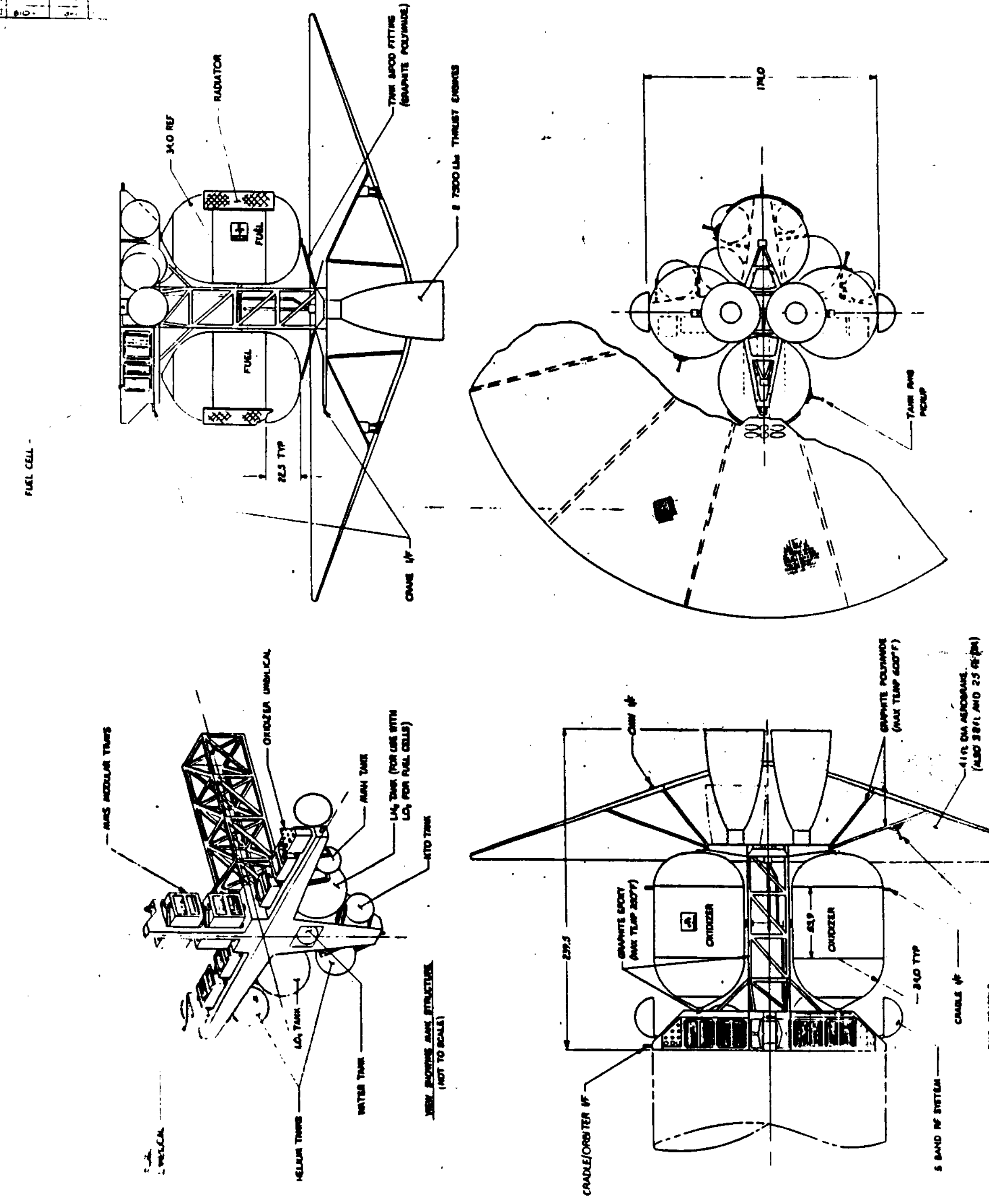

Figure 6.2.2.1-4.

SSF SPACEBASED STORABLE OTV
ALL STRUCTURE COMPOSITE
TANK STRUCTURE 15-3-3-3 TITANIUM
(REF DRAWING GSOR-015)

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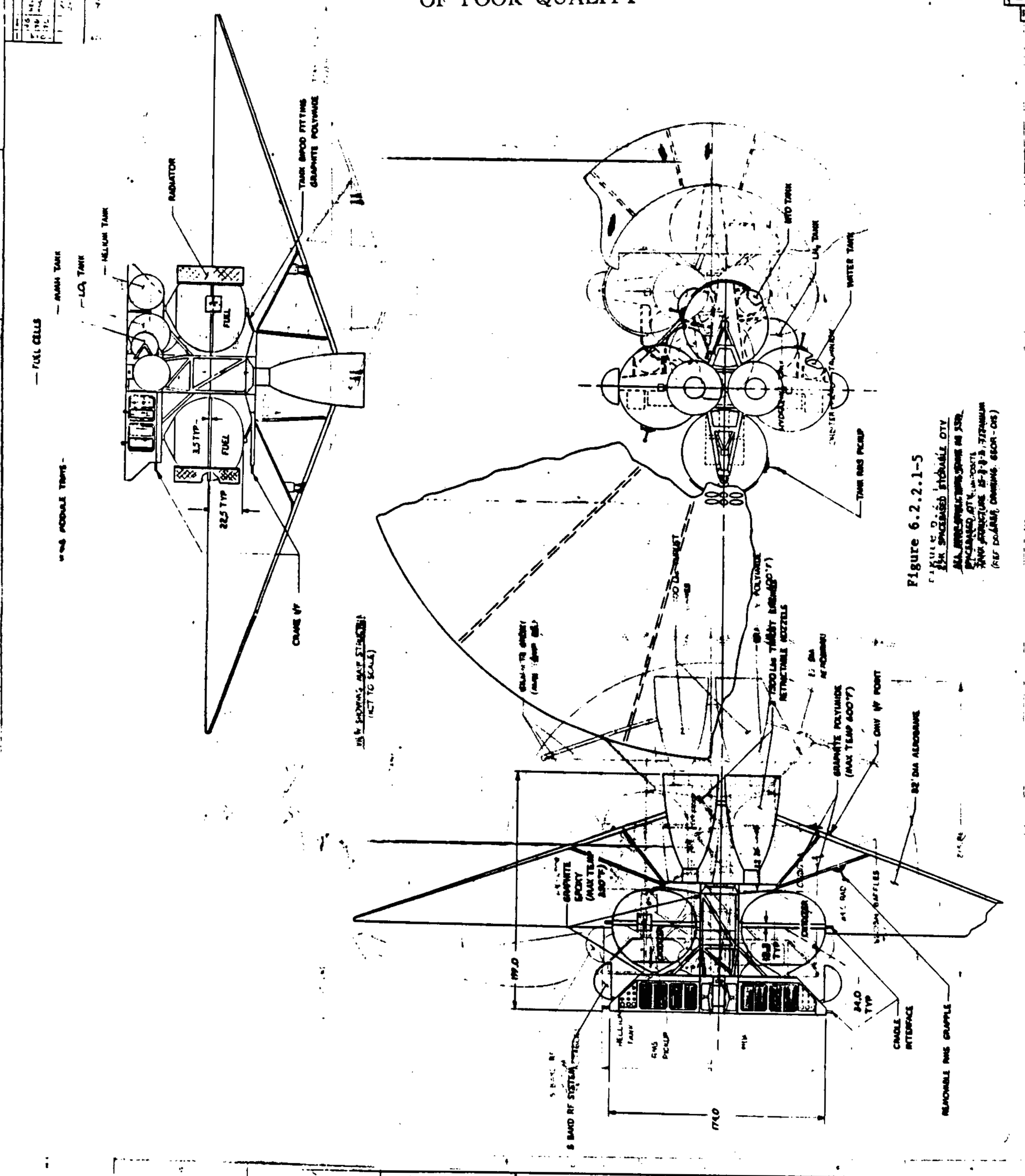

Figure 6.2.2.1-5

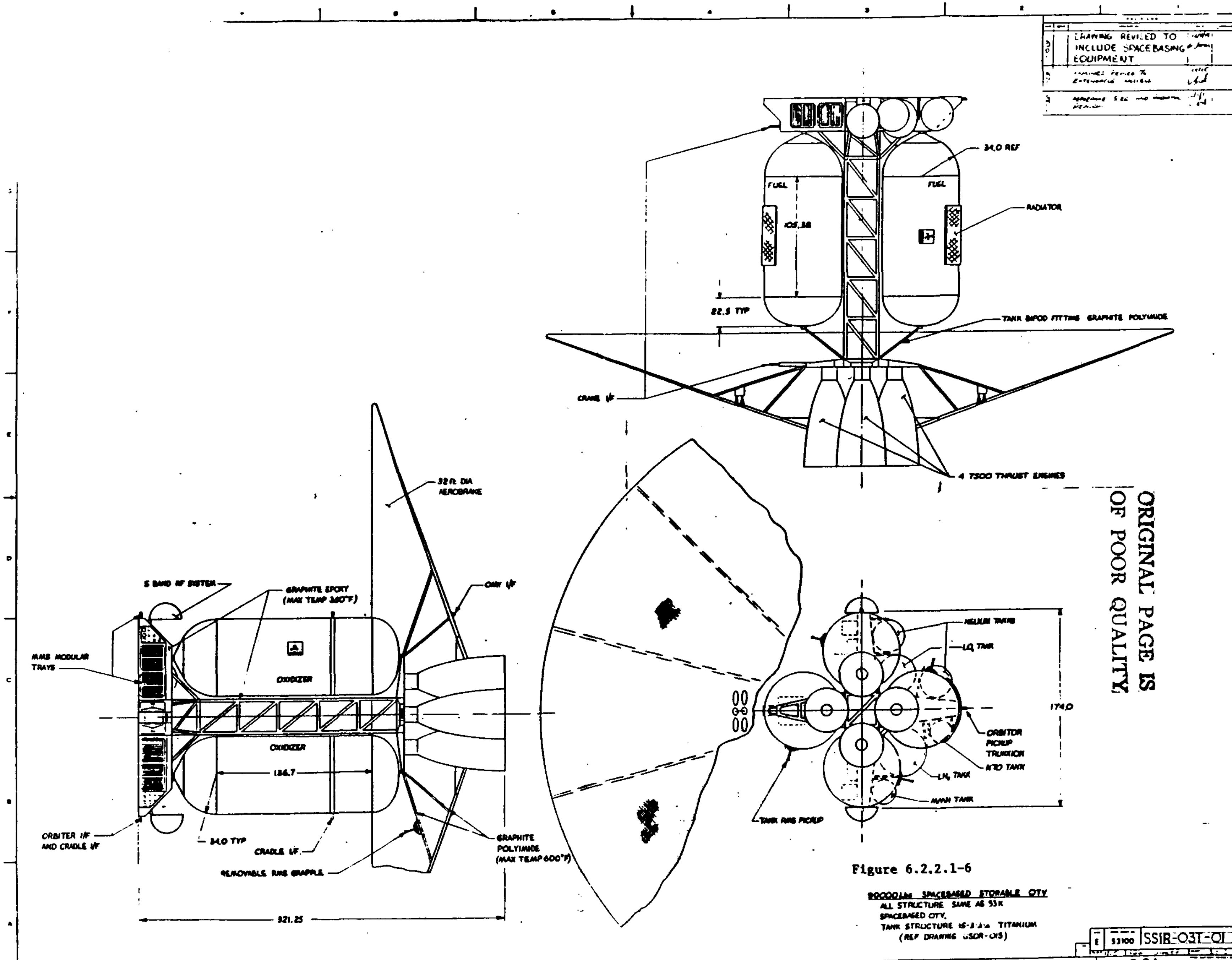

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Figure 6.2.2.1-6

90000Lbs SPACEBASED STORABLE OTV
ALL STRUCTURE SAME AS 53K SPACEBASED OTV.
TANK STRUCTURE 15-3-3-3 TITANIUM
(REF DRAWING USOR-013)

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6.2.2.2 Subsystem Summary Description (Space Based Storable)

6.2.2.2.1 Aeroassist (Space-Based Storable) - The overall layouts of the space based storable configurations are shown in Figures 6.2.1.1-1 through -6. The aeroshield devices used with these configurations are similar in concept to those used on the cryogenic configurations discussed in Section 6.1. Three different aeroshield diameters have been established to protect the several vehicles to be returned. Table 6.2.2.2.1-1 shows the return situations involved. Empty stages of two different sizes will be returned. Three payloads (the multiple payload carrier, the unmanned servicer, and the manned spacecraft) and the appropriate stage will be returned. The table shows the aeroshield diameter required to perform each of these returns. In order to simplify the array of aeroshields to be designed and supported in space, we have used the 32 foot aeroshield to perform the functions indicated for the 28 and 30 foot aeroshields as well. The ballistic coefficients, brake loadings temperatures and surface insulation thicknesses associated with these cases is summarized in Table 6.2.2.2.1-1. The aeroshield weights presented in Section 6.2.2.3 reflect these data.

Table 6.2.2.2.1-1 Space-Based Storable Aeroshield

Aeroshield Parameter	W/C_DA	Brake Diameter (Ft)	TPS	$\dot{q}_{max}$ (Btu/Ft^2) Sec)	Q_{max} (Btu/Ft^2)	t_{max} (^oF)	TPS Thickness (In.)	W_{Brake} / W_{Return}	Design Load (psf)	
									Center	Outboard
Return 53 k-lb Stg	7.46	25	FSI RSI	16.7 20.0	3950 4740	2670 2530	0.45 0.52	0.10	43	32
Return P/L Carrier	7.62	30	FSI RSI	15.8 19.0	3700 4440	2590 2430	0.43 0.48	0.12	43	32
Return 90 k-lb Stg	7.95	28	FSI RSI	15.8 19.0	3750 4500	2590 2430	0.44 0.50	0.09	46	35
Return Servicer	8.62	32	FSI RSI	15.9 19.1	3750 4500	2600 2440	0.44 0.50	0.10	52	39
Manned Return	11.00	41	FSI RSI	16.1 19.3	3800 4560	2610 2460	0.45 0.52	0.07	68	53

6.2.2.2.2 Propulsion (Space-Based Storable) - The propulsion characterics are shown in Table 6.2.2.2.2-1 and the reaction control characteristics are shown in Table 6.2.2.2.2-2. The main propulsion system for the space-based storable is similar to the ground-based storable with the following additions.

Two to four main engines will be required to fly the three stages that have been defined. The engines will be capable of readily being maintained at the Space Station. The engines will also have provisions for the OTV computer to monitor the health of the engine during flight. The 90K perigee stage will use four 7500 lb thrust engines, the 25K second stage will use two 7500 lb, and the 53K second stage and perigee stage will use two 7500 lb thrust engines.

The propellant distribution system components will be modularized for space maintenance and health monitoring provisions will be added. In addition a total tank propellant acquistion system to allow for filling and draining the tanks at the Space Station will be required. The schematic is shown in Figure 6.2.2.2.2-1.

Table 6.2.2.2.2-1 Space-Based Storable MPS Summary

o	ENGINE	-	TWO TO FOUR ENGINES, 7500 LB THRUST EACH, GAS GENERATOR CYCLE, I_{SP} = 345.7 SEC, AREA RATIO 600:1
o	PROPELLANT DISTRIBUTION	-	DUAL TANK PARALLEL FEED, FULL TANK ACQUISITION FOR SERVICING AT SPACE STATION
o	PRESSURIZATION	-	HELIUM STORED GAS, REGULATOR CONTROLLED
o	VENT	-	SPACE STATION VENT ONLY
o	VALVE ACTUATION	-	ELECTRICAL AND HELIUM, COMMON WITH STAGE PRESSURIZATION SUPPLY
o	PROPELLANT UTILIZATION	-	TANK TO TANK AND ENGINE MIXTURE RATIO CONTROL
o	THERMAL PROTECTION	-	HEATER BLANKETS AND MLI
o	MAINTENANCE	-	COMPONENT AND ENGINE MODULARITY

Table 6.2.2.2.2-2 Space-Based Storable RCS Summary

- PROPELLANT
  - N_2O_4/MMH
- ROCKET ENGINE MODULE
  - NEW DESIGN BASED ON SHUTTLE TECHNOLOGY I_{SP} = 280 SEC THRUST 100 lb_f, 14 THRUSTERS (7 THRUSTERS PER MODULE)
  - 3 DOF FOR PERIGEE STAGE, +X TRANSLATION
  - 3 DOF FOR APOGEE STAGE, +X TRANSLATION
  - FAIL OPERATIONAL
- PROPELLANT SUPPLY
  - COMMON STORAGE WITH MPS TANKS
  - MPS ENGINE PUMP FEED TO ACCUMULATORS
  - FEED TO ACS ENGINES AT 400 PSI 3:1 BLOWDOWN
  - 20" ID TANKS WITH SCREEN ACQUISITION DEVICE
  - 430 LB N_2O_4/MMH AT A MR OF 1.65:1
- SAFETY
  - 2 FAULT TOLERANT ISOLATION FOR PROXIMITY OPERATIONS

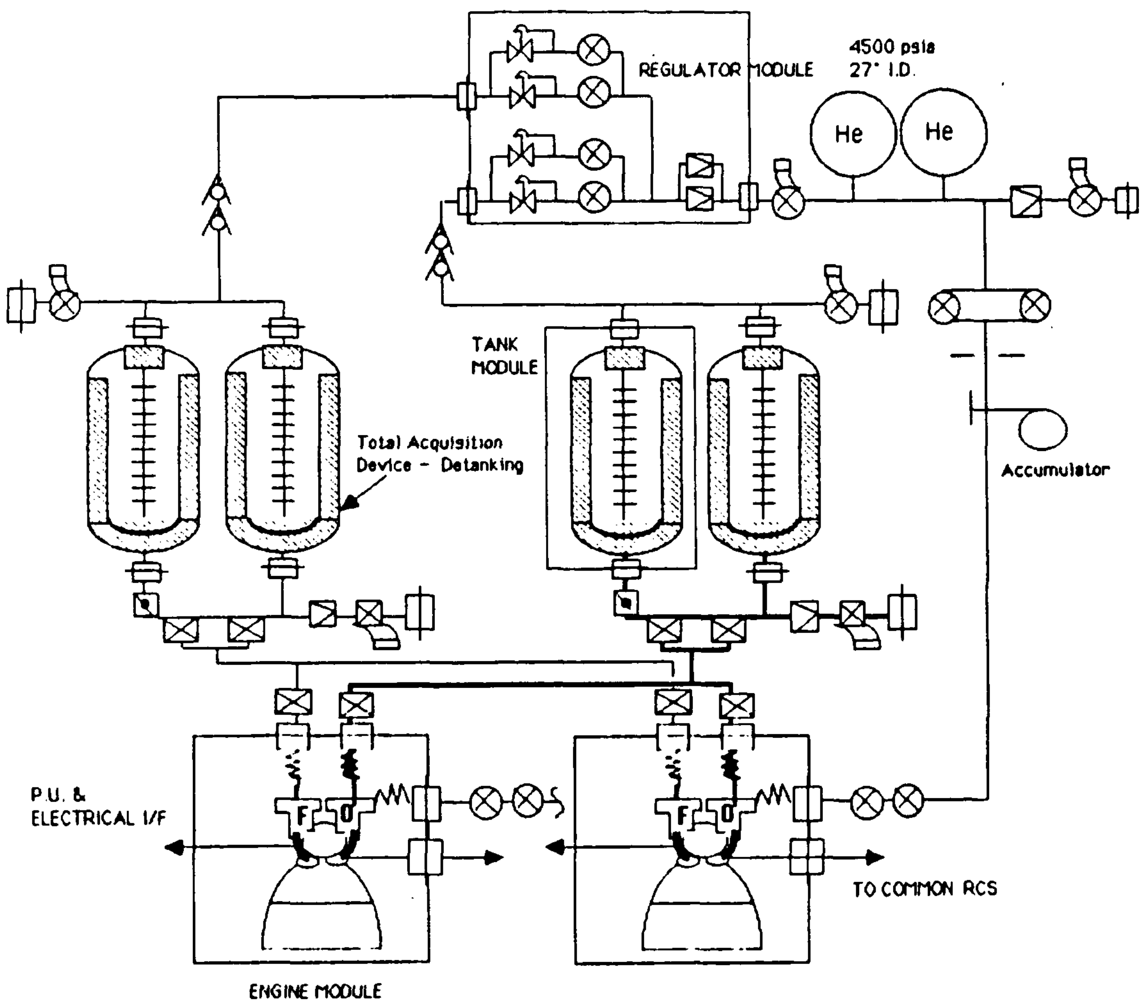

Figure 6.2.2.2.2-1 Space Based N_2O_4/MMH Schematic

The MMH tank operating pressure will remain the same as the ACC OTV. However, the goal of the AFRPL XLR-132 design is to reduce the N_2O_4 NPSH by about 15 feet. This corresponds to about a 10 psi N_2O_4 tank pressure reduction.

For the second stage application, propellants are stored in the main tanks and are transferred to the RCS accumulators from the high pressure sections of the main engine turbo pumps during engine burns. The 20 inch diameter accumulators have screen liquid acquisition devices to prevent gas ingestion during expulsion to the thruster system at about 400 psia. The system will provide 60,000 lb-sec of total impulse between recharges from the MPS engine. The thruster design will be based on shuttle RCS technology level, therefore system sizing has been based on a Isp of 280 seconds. For perigee operation the tanks should not require resupply. Inital RCS propellant loading will be done during MPS fill before the accumulators are pressurized to 400 psia.

The RCS system is isolated in a dual fault tolerant manner for proximity operations as shown in the schematic, Figure 6.2.2.2.2-2.

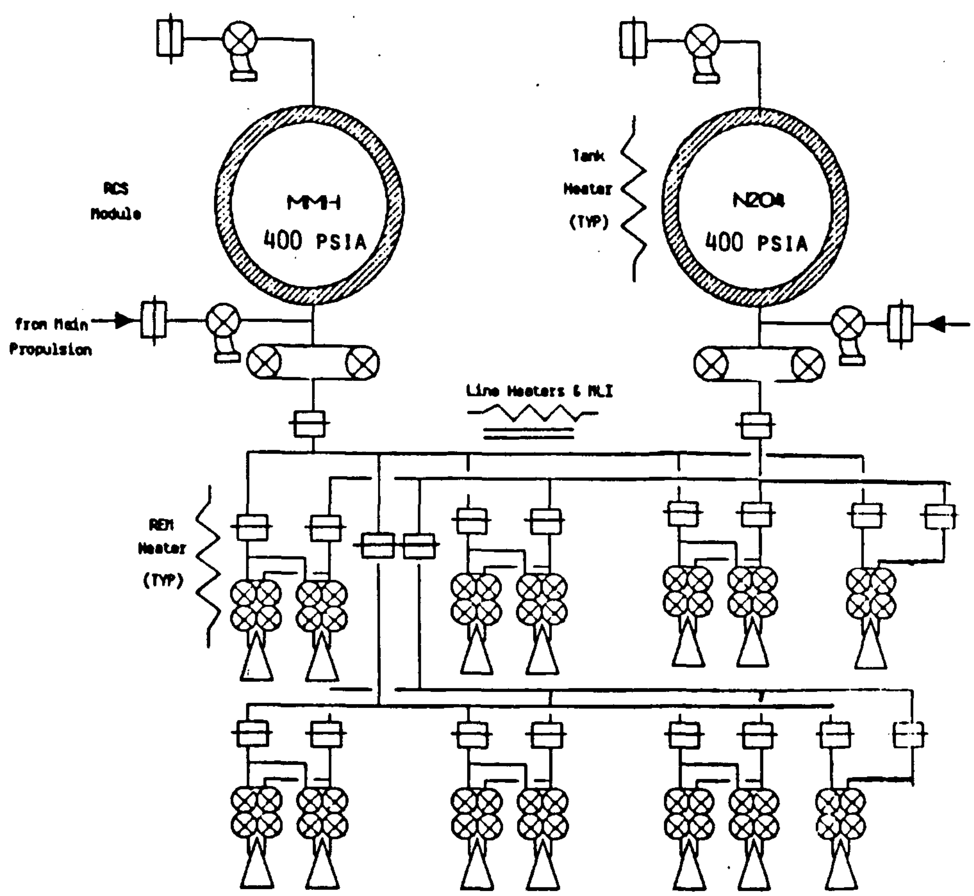

Figure 6.2.2.2.2-2 Space-Based N_2O_4/MMH RCS Schematic

6.2.2.2.3 Structures and Mechanisms (Space-Based Storable) - The space based storable OTV requires three different sized stages (Figures 6.2.1.1-4 to -6) to satisfy the baseline mission requirements. The fleet will consist of a 53K OTV, a 25.4K OTV, and a 90K OTV. All configurations use two 68 in diameter cylindrical MMH tanks with 0.75 ellipse lower heads and two 68 in diameter cylindrical N_2O_4 oxidier tanks with 0.75 ellipse lower heads. Engines are mounted on a center core truss arrangement. Tanks are supported off conical shaped forward heads by a crossbeam arrangement. The aerobrake is mounted on the engine support by graphite polyimide struts.

The vehicles are configured to be delivered to space fully assembled (except for aerobrake) in the space shuttle cargo bay. The aerobrakes will also be delivered in the shuttle cargo bay and deployed and attached to the OTV in space.

The avionics are installed on the outer surfaces of the crossbeam to provide ready access for removal/replacement. The storable tanks are of fusion welded construction of heat treated 15-3-3-3 titanium. If problems are encountered during the alloy test program, the back up alloy would be 6AL 4 V titanium. The conical heads are made in five pieces with a machined cone shaped cap for tank pickup and four formed and chem milled gores. The elliptical tank head is made of similar construction but with a formed center piece. Tank barrel is made in two pieces. Minimum tank membrane gage is 0.006. If difficulties are encountered in handling 0.006 thick tanks, membrane thickness will be increased to what is required.

Main structural crossbeam and engine thrust structure are fabricated from graphite epoxy. Lower tank support beams are of graphite polyimide construction. Aerobrakes consist of a center core structure of graphite polyimide honeycomb and ceramic foam tiles. The remainder of the aerobrake is a nicalon, ceramic felt and sealed Nextel layup.

The major difference between the stages are total length, aerobrake diameter, and number of engines. A table of aerobrake diameter and number of engines per vehicle is shown below.

Stage	No. Engines	A/B Diameter
25.4K OTV	2	32 ft
53K OTV	2	25,32, & 41 ft
90K OTV	4	32 ft

The vehicles are equipped with crane and cradle provisions for handling at the Space Station. In addition, major components such as aerobrakes and tanks have grapple provisions for component changeout at the Space Station.

6.2.2.2.4 Avionics (Space-Based Storable) - The space-based storable avionics configuration (Figure 6.2.2.2.4-1) is essentially identical to the space-based cryo configuration (Section 6.1.2.2.4). Fuel cell reactant tankage must be added since it cannot be main tank fed as it was in the cryo configuration. Table 6.2.2.2.4-1 reflects the subsystem and system unit and total values for power and weight for this configuration.

6.2.2.2.4.1 Guidance, Navigation and Control - The Guidance, Navigation and Control subsystem is essentially the same as in Section 6.1.1.2.4.1. The engine controller is deleted as it is not required for the storable engine.

6.2.2.2.4.2 Data Management - The Data Management subsystem is the same as in Section 6.1.2.2.4.2.

6.2.2.2.4.3 Telemetry and Command - The Telemetry and Command subsystem is the same as in Section 6.1.2.2.4.3.

6.2.2.2.4.4 Communications and Tracking - The Communications and Tracking subsystem is the same as in Section 6.1.2.2.4.4.

6.2.2.2.4.5 Electrical Power - The fuel cell reactant tanks are added to this configuration as reactants cannot be taken from the propellant system. The EPS design (Figure 6.2.2.2.4-2) is essemtially the same as in section 6.1.2.2.4.5. Fuel cells are sized for 1.1KW, which includes a 20% design margin.

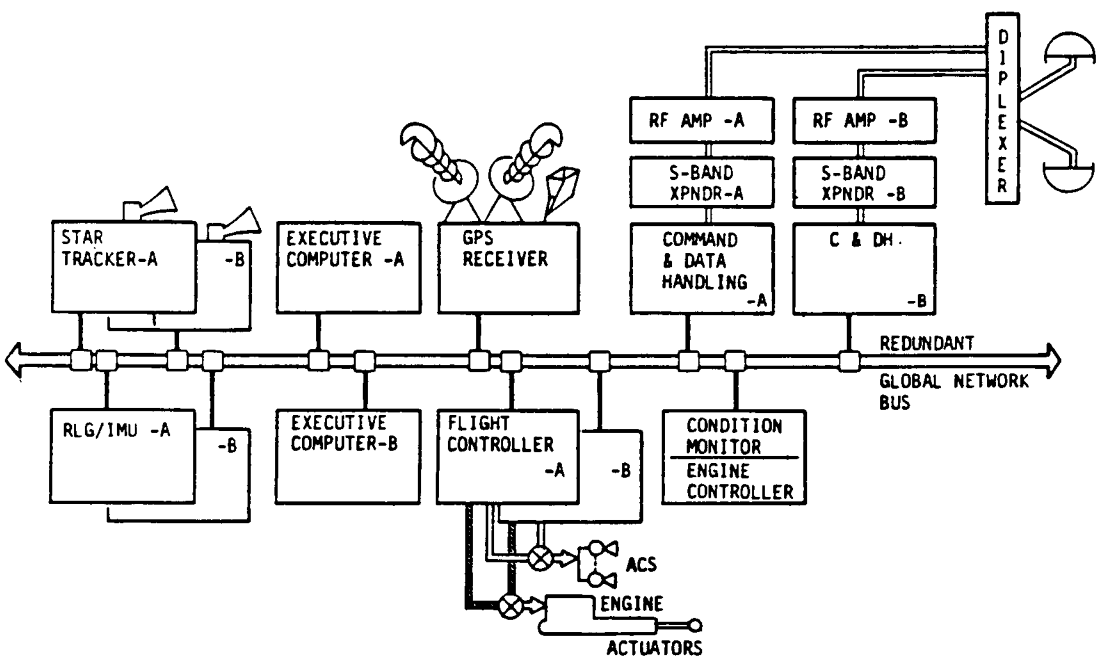

Figure 6.2.2.2.4-1 Block Diagram of the Space-Based Storable Configuration

Table 6.2.2.2.4-1 OTV Avionics Equipment List - Storable Configuration (Sheet 1 of 2)

Subsystem Equipment	Weight (lb)	Power (w)	Size (in) H W L	Qty	Total Wt (lb)	Power (w) Max	Power (w) Avg
GN&C							
Star Tracker	11	10	7x 7x20	2	22	20	10
IMU	24	40	8x 8x12	2	48	80	80
GPS REceiver	20	30	8x 8x 9	1	20	30	10
GPS Antenna-Low Alt	5		6x 6x10	2	10		
GPS Antenna-Hi Alt	5		18x18x26	1	5		
Flight Controller	30	90	8x 8x16	2	60	180	180
Subsystem Total					165	310	280
Data Management							
Central Computer & Mass Memory	10	60	6x 8x 9		20	120	120
Subsystem Total					297	270	70
Telemetry and Command							
Command & Data Handling	10	35	6x 8x10	2	30	45	22
TLM Power Supply	7	10	4x 7x 7	1	7	10	5
Subsystem Total					44	65	27

Table 6.2.2.2.4-1 OTV Avionics Equipment List - Space-Based Storable Configuration (Sheet 2 of 2)

Subsystem	Equipment	Weight (lb)	Power (w)	Size (in) H W L	Qty	Total Wt(lb)	Power (w) Max	Power (w) Avg
Communications and Tracking								
	STDN/TDRS Xponder	16	55	6x 6x14	2	32	65	65
	20w RF Power Amp	6	125	3x 6x10	2	12	125	40
	S-Band RF System	50	20		2	100	40	20
				Subsystem Total				144
230	125							
EPS								
	Fuel Cell (FC)	33		11x12x12	2	66		
	FC Radiators	25		$25ft^2x2"$	2	50		
	FC Reactants	35				35		
	FC Reactant Tank LH_2 & Plumb	42				370		
	FC Reactant Tank LO_2 & Plumb	38				38		
	FC Coolant	10				10		
	FC H_2O Tank	13				13		
	Power Control & & Distribution	27	10	6x 8x12	2	54	20	20
	Heaters		50				50	50
	Engine Power		100				100	
				Subsystem Total			643	170
70								
				System Total		1016	895	622

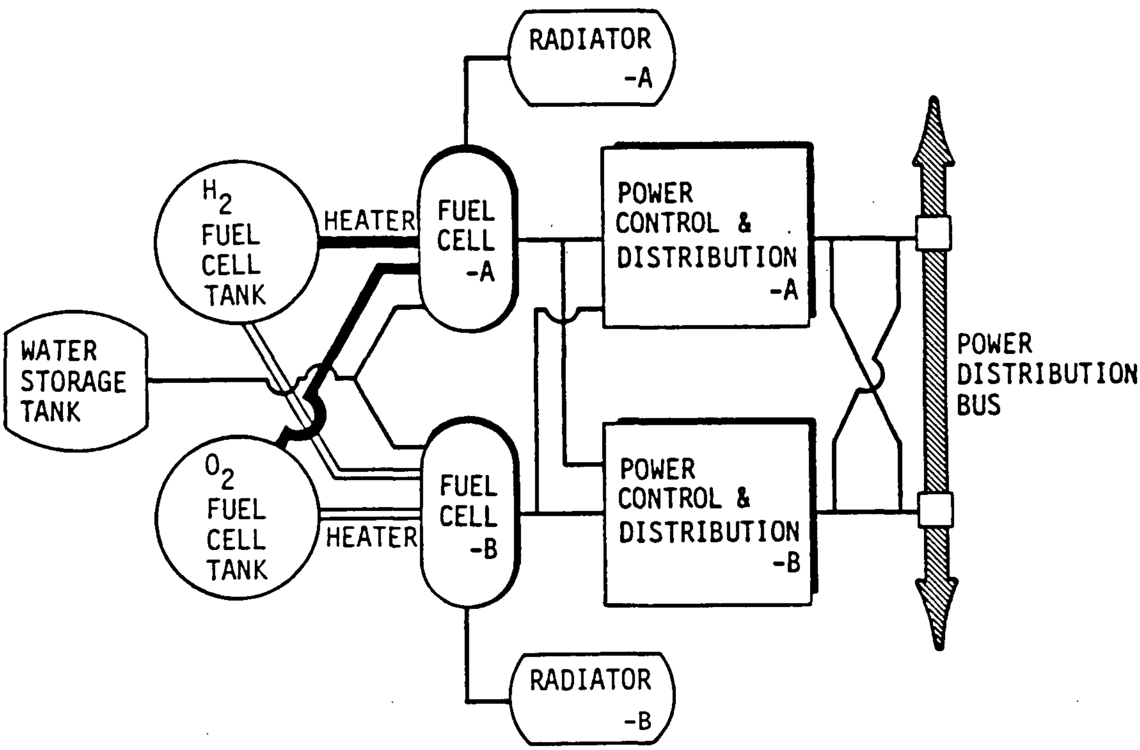

Figure 6.2.2.2.4-2 EPS configuration for the space-based, storable configuration.

6.2.2.2.5 Thermal Control (Space Based Storable) - All space-based storable configurations have fuel cell power systems. Serparte H_2 and O_2 tanks must carry sufficient fuel to support a 2.10 KW continuous power requirements for the flight. Two 25-ft^2 radiators are required and should be mounted on opposite sides of the vehicle.

The avionics are housed in modularized boxes mounted to the avionics bay structure. This isolates many of the avionics components and makes more difficult to evenly distribute waste heat among the various avionics components. The design can, however, be passibely thermally controlled. The high power avionics is mounted to base plates to allow a high heat conductance to the base plate. The base plate, in turn, has a strong radiation tie to space.

The payload/OTV interface is made nearly adiabatic by approximately 25 to 50 layers of insulation blanket (double aluminized Kapton MLI) located at the interface.

The N_2O_4 and MMH tanks are insulated with two layers of Kapton thermal blankets. These tanks are equipped with heaters (thermostatically controlled) to maintain acceptable propellant temperatures for long flight durations.

The impacts the meteoroid shield has on the OTV thermal control system will be evaluated as the OTV design develops.

The RCS tanks, feedlines, and modules require heaters since thermal pulsing would consume too much fuel. The RCS tank requires a 25 layer MLI blanket. The RCS feedlines will be inslulated with one or two layers of thermally insulating blankets.

The heating effects of the engine nozzles are currently of no concern.

The composite helium pressurant tanks will require MLI. Thermostatically controlled strip heaters are necessary to ensure adequate helium temperatures/pressures.

6.2.2.3 System Weight Summary - Space-Based Storable

Total flight vehicle weight for the space-based storable configuration is presented in Tables 6.2.2.3-1 through -4. Table 6.2.2.3-1 presents data relative to the 53K lb perigee stage, while Table 6.2.2.3-2 presents data relative to the 90K lb perigee stage. Tables 6.2.2.3-3 and -4 present data relative to the 25.4K lb and 53K lb apogee stages, respectively. Dry weight is developed in detail. Propulsive and nonpropulsive fluid loadings are summarized. The breakdown has not been rearranged into MSFC's suggested format as this approach was deleted from contention at an earlier stage in the study.

Table 6.2.2.3-1 Weight Statement
Space-Based Storable
53K Propellant Load
Perigee Stage

Description	Weight (lb)		
Structure			915
Basic Airframe		153	
Ox Tank		179	
Tank (2) - Ti	141		
Aft Struts & Fwd Fittings	38		
Fu Tank		104	
Tank (2) - Ti	68		
Aft Struts & Fwd Fittings	36		
Engine Truss & Attach		120	
Engine Truss	86		
Hardpoints - Engine Attach	30		
Attach - Engine Actuators	4		
Subsystem Module		208	
RMS Grapple Fixtures & Struts		56	
OMV I/F Fittings (4)		20	
Cradle/Orbiter I/F Trunnions (5)		25	
Crane I/F Fittings (2)		10	
P/L or ACC Attach (4)		40	
Aerobrake Assy			590
Support (ASE) - S.B. Maintenance			272
Environmental Control			253
Meteoroid Protection		157	
MPS Tanks	138		
ACS Tanks	5		
He Tanks	9		
FC Reactant Tanks	3		
FC Water Tank	2		
Thermal Control/Protection		96	
Thermal Control - Heater Tape @ MPS, ACS Tanks	59		
Thermal Protection -MLI, Tape & Coatings @ Engine Truss, Compartment, Prop;, Lines, etc.	37		

Table 6.2.2.3-1 Weight Statement
Space-Based Storable
53K Propellant Load
Perigee Stage
(Continued)

Description		Weight (lb)	
Main Propulsion System			1198
Engine (2): XLR-132 7500# Thrust Ea		506	
Propellant Disbribution System		113	
Pressurization System		302	
ACS Common Feed		8	
PU/Acquisition System		190	
Instrumentation		15	
Actuators (4) - Electrical		64	
Orientation Control			177
ACS Subsystem: Bi-Prop (MMH & N204)		177	
Rocket Engine Modules (2)	51		
Mounting - REMS	5		
ACS Accumulator Tanks (2)	28		
Mounting - Tanks	3		
Propel. Distribution & Pressurization	82		
Mounting Provisions	8		
Electrical Power			381
Fuel Cell System		66	
Reactant Tank (LH2) & Plumbing		42	
Reactant Tank (LO2) & Plumbing		38	
Radiator System		50	
Residual H2O System		13	
Wire Harness, Connectors, etc.		130	
Mounting Provisions		42	

Table 6.2.2.3-1 Weight Statement
Space-Based Storable
53K Propellant Load
Perigee Stage
(Continued)

Description	Weight (lb)	
Avionics		425
Avionics (Equipment List)	373	
Mounting Provisions	52	
Dry Weight		4211
Contingency (15%)		632
Total Dry Weight		4843
Main Propellants (MR 2:1 Ox Wt to Fu Wt)		53000
N204	35333	
MMH	17667	
Pressurant (He) - MPS		26
ACS Propellant & Pressurant Scavenged from MPS		N/A
FC Reactant & Coolant		22
Reactant	12	
Coolant	10	
Total Loaded Weight		57891

Table 6.2.2.3-2 Weight Statement
Space-Based Storable
90K Propellant Load
Perigee Stage

Description	Weight (lb)		
Structure			1218
Basic Airframe		224	
Ox Tank		307	
Tank (2) - Ti	269		
Aft Struts & Fwd Fittings	38		
Fu Tank		159	
Tank (2) - Ti	123		
Aft Struts & Fwd Fittings	36		
Engine Truss & Attach		169	
Engine Truss	101		
Hardpoints - Engine Attach	60		
Attach - Engine Actuators	8		
Subsystem Module		208	
RMS Grapple Fixtures & Struts		56	
OMV I/F Fittings (4)		20	
Cradle/Orbiter I/F Trunnions (5)		25	
Crane I/F Fittings (2)		10	
P/L or ACC Attach (4)		40	
Aerobrake Assy			887
Support (ASE) - S.B. Maintenance			480
Environmental Control			378
Meteoroid Protection		226	
MPS Tanks	203		
ACS Tanks	5		
He Tanks	13		
FC Reactant Tanks	3		
FC Water Tank	2		
Thermal Control/Protection		152	
Thermal Control - Heater Tape @ MPS, ACS Tanks	99		
Thermal Protection -MLI, Tape & Coatings @ Engine Truss, Compartment, Prop., Lines, etc.	53		

Table 6.2.2.3-2 Weight Statement
Space-Based Storable
90K Propellant Load
Perigee Stage
(Continued)

Description		Weight (lb)	
Main Propulsion System			1941
Engine (4): XLR-132 7500# Thrust Ea		1012	
Propellant Disbribution System		151	
Pressurization System		412	
ACS Common Feed		16	
PU/Acquisition System		196	
Instrumentation		26	
Actuators (8) - Electrical		128	
Orientation Control			197
ACS Subsystem : Bi Prop (MMH & N204)		197	
Rocket Engine Modules (2)	51		
Mounting - REMS	5		
ACS Accumulator Tanks (2)	28		
Mounting - Tanks	3		
Propel. Distribution & Pressurization	100		
Mounting Provisions	10		
Electrical Power			381
Fuel Cell System		66	
Reactant Tank (LH2) & Plumbing		42	
Reactant Tank (LO2) & Plumbing		38	
Radiator System		50	
Residual H20 System		13	
Wire Harness, Connectors, etc.		130	
Mounting Provisions		42	

Table 6.2.2.3-2 Weight Statement
Space-Based Storable
90K Propellant Load
Perigee Stage
(Continued)

Description	Weight (lb)	
Avionics		425
Avionics (Equipment List)	373	
Mounting Provisions	52	
Dry Weight		5907
Contingency (15%)		886
Total Dry Weight		6793
Main Propellants (MR 2:1 Ox Wt to Fu Wt)		90000
N204	60000	
MMH	30000	
Pressurant (He) - MPS		39
ACS Propellant & Pressurant Scavenged from MPS		N/A
FC Reactant & Coolant		22
Reactant	12	
Coolant	10	
Total Loaded Weight		96854

Table 6.2.2.3-3 Weight Statement
Space-Based Storable
25.4K Propellant Load
Apogee Stage

Description	Weight (lb)		
Structure			762
Basic Airframe		113	
Ox Tank		106	
Tank (2) - Ti	68		
Aft Struts & Fwd Fittings	38		
Fu Tank		74	
Tank (2) - Ti	38		
Aft Struts & Fwd Fittings	36		
Engine Truss & Attach		110	
Engine Truss	76		
Hardpoints - Engine Attach	30		
Attach - Engine Actuators	4		
Subsystem Module		208	
RMS Grapple Fixtures & Struts		56	
OMV I/F Fittings (4)		20	
Cradle/Orbiter I/F Trunnions (5)		25	
Crane I/F Fittings (2)		10	
P/L or ACC Attach (4)		40	
Aerobrake Assy			887
Support (ASE) - S.B. Maintenance			272
Environmental Control			204
Meteoroid Protection		108	
MPS Tanks	89		
ACS Tanks	5		
He Tanks	5		
FC Reactant Tanks	7		
FC Water Tank	2		
Thermal Control/Protection		96	
Thermal Control - Heater Tape @ MPS, ACS Tanks	59		
Thermal Protection -MLI, Tape & Coatings @ Engine Truss, Compartment, Prop Lines, etc.	37		

Table 6.2.2.3-3 Weight Statement
Space-Based Storable
25.4K Propellant Load
Apogee Stage
(Continued)

Description		Weight (lb)	
Main Propulsion System			1098
Engine (2): XLR-132 7500# Thrust Ea		506	
Propellant Disbribution System		113	
Pressurization System		202	
ACS Common Feed		8	
PU/Acquisition System		190	
Instrumentation		15	
Actuators (4) - Electrical		64	
Orientation Control			177
ACS Subsystem : Bi-Prop (MMH & N204)		177	
Rocket Engine Modules (2)	51		
Mounting - REMS	5		
ACS Accumulator Tanks (2)	28		
Mounting - Tanks	3		
Propel. Distribution & Pressurization	82		
Mounting Provisions	8		
Electrical Power			401
Fuel Cell System		66	
Reactant Tank (LH2) & Plumbing		54	
Reactant Tank (LO2) & Plumbing		44	
Radiator System		50	
Residual H20 System		13	
Wire Harness, Connectors, etc.		130	
Mounting Provisions		44	

Table 6.2.2.3-3 Weight Statement
Space-Based Storable
25.4K Propellant Load
Apogee Stage
(Continued)

Description	Weight (lb)	
Avionics		425
Avionics (Equipment List)	373	
Mounting Provisions	52	
Dry Weight		4226
Contingency (15%)		634
Total Dry Weight		4860
Main Propellants (MR 2:1 Ox Wt to Fu Wt)		25400
N204	16933	
MMH	8467	
Pressurant (He) - MPS		13
ACS Propellant & Pressurant Scavenged from MPS		N/A
FC Reactant & Coolant		180
Reactant	170	
Coolant	10	
Total Loaded Weight		30463

Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage

Description	Weight (lb)		
Structure			915
Basic Airframe		153	
Ox Tank		179	
Tank (2) - Ti	141		
Aft Struts & Fwd Fittings	38		
Fu Tank		104	
Tank (2) - Ti	68		
Aft Struts & Fwd Fittings	36		
Engine Truss & Attach		120	
Engine Truss	86		
Hardpoints - Engine Attach	30		
Attach - Engine Actuators	4		
Subsystem Module		208	
RMS Grapple Fixtures & Struts		56	
OMV I/F Fittings (4)		20	
Cradle/Orbiter I/F Trunnions (5)		25	
Crane I/F Fittings (2)		10	
P/L or ACC Attach (4)		40	
Aerobrake Assy			1343
Support (ASE) - S.B. Maintenance			272
Environmental Control			260
Meteoroid Protection		164	
MPS Tanks	138		
ACS Tanks	5		
He Tanks	9		
FC Reactant Tanks	10		
FC Water Tank	2		
Thermal Control/Protection		96	
Thermal Control - Heater Tape @ MPS, ACS Tanks	59		
Thermal Protection -MLI, Tape & Coatings @ Engine Truss, Compartment, Prop;, Lines, etc.	37		

Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage
(Continued)

Description		Weight (lb)	
Main Propulsion System			1198
Engine (2): XLR-132 7500# Thrust Ea		506	
Propellant Disbribution System		113	
Pressurization System		302	
ACS Common Feed		8	
PU/Acquisition System		190	
Instrumentation		15	
Actuators (4) - Electrical		64	
Orientation Control			177
ACS Subsystem - Hydrazine		177	
Rocket Engine Modules (2)	51		
Mounting - REMS	5		
ACS Accumulator Tanks (2)	28		
Mounting - Tank	3		
Propel. Distribution & Pressurization	82		
Mounting Provisions	8		
Electrical Power			419
Fuel Cell System		66	
Reactant Tank (LH2) & Plumbing		66	
Reactant Tank (LO2) & Plumbing		49	
Radiator System		50	
Residual H20 System		13	
Wire Harness, Connectors, & etc.		130	
Mounting Provisions		45	

Table 6.2.2.3-4 Weight Statement
Space-Based Storable
53K Propellant Load
Apogee Stage
(Continued)

Description	Weight (lb)	
Avionics		425
Avionics (Equipment List)	373	
Mounting Provisions	52	
Dry Weight		5009
Contingency (15%)		751
Total Dry Weight		5760
Main Propellants (MR 2:1 Ox Wt to Fu Wt)		53000
N204	35333	
MMH	17667	
Pressurant (He) - MPS		26
ACS Propellant & Pressurant Scavenged from MPS		N/A
FC Reactant & Coolant		380
H2	42	
02	328	
Coolant	10	
Total Loaded Weight		59166

6.2.2.4 Performance on Model Missions: Space-Based Storables

The following is a summary of the assumptions used in performance analyses which were unique to storable OTV configurations. Several of the storable configurations were envisioned as perigee stages, requiring the use of an additional expendable kick stage to perform some part of the mission, such as circularizing the payload at geosynchronous altitude. In such cases, the EKS was assumed to have a mass fraction of 0.95, an Isp of 310 seconds, and was sized to be just large enough to perform the mission at hand, i.e. "custom fit" to each particular mission. For this reason the performance curves for the SB 53K, and SB 90K storable perigee stages are qualitatively different than those for the cryogenic stages which performed their missions "solo". Table 6.2.2.4-1 summarizes the propellant load required and gross weight of the 53K lb storable perigee stage on each of the Rev. 7 model missions on which it will be used in conjunction with an expendable apogee stage. Table 6.2.2.4-2 shows propellant and gross weight of the two stage configuration comprised of the 53K lb perigee stage and the 25K lb reusable apogee stage. Table 6.2.2.4-3 presents data analagous to Table 6.2.2.4-1 for the 90K lb perigee stage. Table 6.2.2.4-4 presents data equivalent to Table 6.2.2.4-2 for the 90K lb perigee stage/53K lb apogee stage combination. Figures 6.2.2.4-1 and-2 summarize the performance of the perigee stages to high circular orbits assuming the use of expendable apogee kick stages.

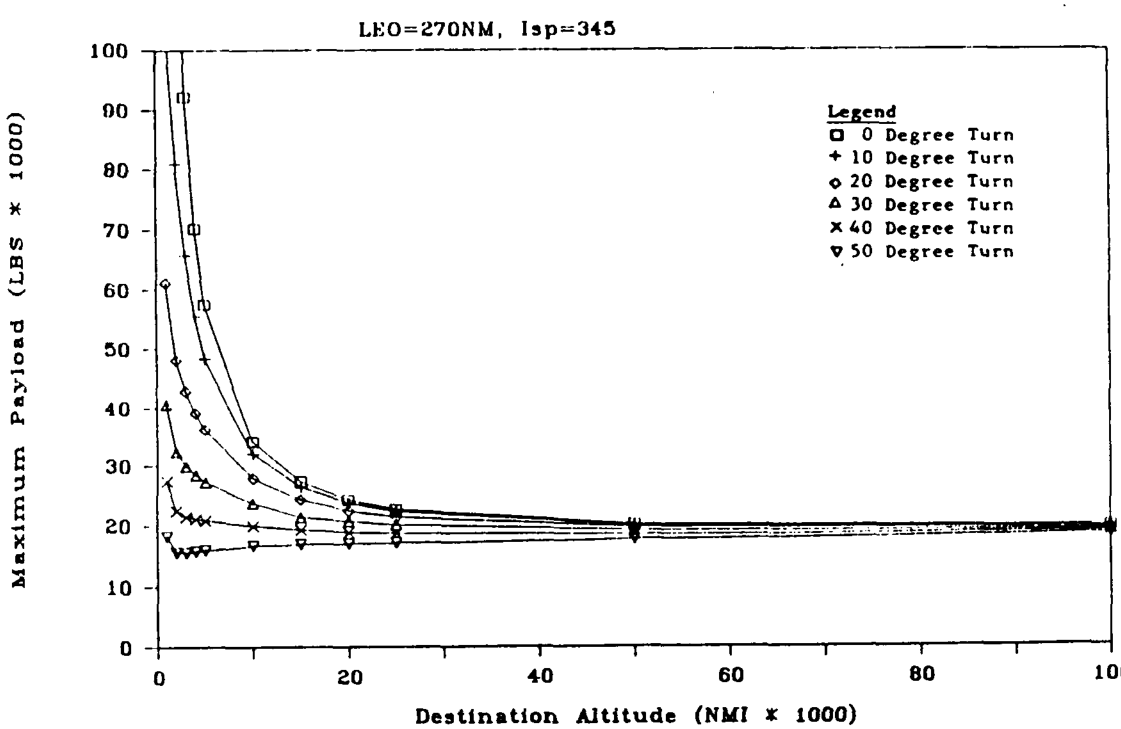

Figure 6.2.2.4-1 Space-Based 53Klb Storable OTV Performance

Table 6.2.2.4-1 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage, Used with Expendable Apogee Stage
(Rev. 7 Missions)

Isp - 345.7 Sec

MISSION	P/L UP	P/L DN	OTV PROPELLANT	OTV PROPELLANT GROSS WT. OTV + EKS + PAYLOAD
Geosynchronous Missions				
19036	6800	0	20305	38869
15009	13310	0	34228	65328
15008	13159	0	33903	64709
18724	10000	0	27164	51828
18064	10163	0	27508	52490
13003	20000	0	49103	93128
18902	11099	2000	23604	44534
DoD Equivalent Geosynchronous Delivery Missions				
19031	19083*	0	6756	89268
19031	13660*	0	34983	66764
19031	13989*	0	35695	68116
19031	13683*	0	35033	66859
19031	13100*	0	33775	64468
19031	12675*	0	32862	62727
19031	10489*	0	28199	53814
19031	18000*	0	44510	84733
19031	15380*	0	38722	73848
19031	17417*	0	43212	82300
19031	17843*	0	44160	84077
19031	18383*	0	45367	86334
19031	18150*	0	44845	85359
19031	18367*	0	45331	86267
19031	19083*	0	46939	89268
Lunar Mission				
17201	5000	0	41314	54074
Planetary Mission				
17075**	9120	0	40115	55086

*DoD Equivalent Payload
Growth Projected by MMA

**Planetary program analyzed here was extracted from a preliminary Rev 8 planetary model supplied by MSFC on 25 Jan 1985 and not redone due to selection of cryo OTV

Table 6.2.2.4-2 Performance Analysis for Required Missions
Storable, Space-Based, 25K Apogee Stage, Used with 53K Perigee Stage
(Rev. 7 Missions)

Isp = 345.7 Sec

			OTV PROPELLANT			
MISSION	P/L UP	P/L DN	OTV PROPELLANT			GROSS WT. OTV + EKS + PAYLOAD
Geosynchronous Missions						
15700	6500	5000	44109	23608	67717	118157
15701	1000	2678	35178	18145	53323	93393
13002	7000	4510	42316	22528	64844	133224

Table 6.2.2.4-3 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage, Used with Expendable Apogee Stage

Isp = 345.7 Sec

MISSION	P/L UP	P/L DN	OTV PROPELLANT	GROSS WT. OTV + EKS + PAYLOAD
Lunar Missions				
17202	20000	0	83361	112567
Planetary Missions				
17070**	15344	0	60004	83364
17071**	15000	0	59280	82289
17072**	13183	0	55469	76623
17073**	13645	0	56436	78061
17074**	19035	0	67826	94955
17076**	14333	0	57878	80206
17077**	22623	0	75532	106327

**Planetary program analyzed here was extracted from a preliminary Rev 8 planetary model supplied by MSFC on 25 Jan 1985 and not redone due to selection of cryo OTV

Table 6.2.2.4-4 Performance Analysis for Required Missions
Storable, Space-Based, 53K Perigee Stage
(Rev. 7 Missions)

MISSION	P/L UP	P/L DN	OTV PROPELLANT			GROSS WT. OTV + EKS + PAYLOAD
			OTV PROPELLANT			
Geosynchronous Missions						
15003	16500	9000	62879	34671	97550	155395
15006	14000	14000	82354	46790	129144	225401
Lunar Missions						
17203	80000	15000	157289	39066	196355	419300
17204	80000	0	147561	30822	178383	392110
17205	80000	10000	154500	36317	190817	395221

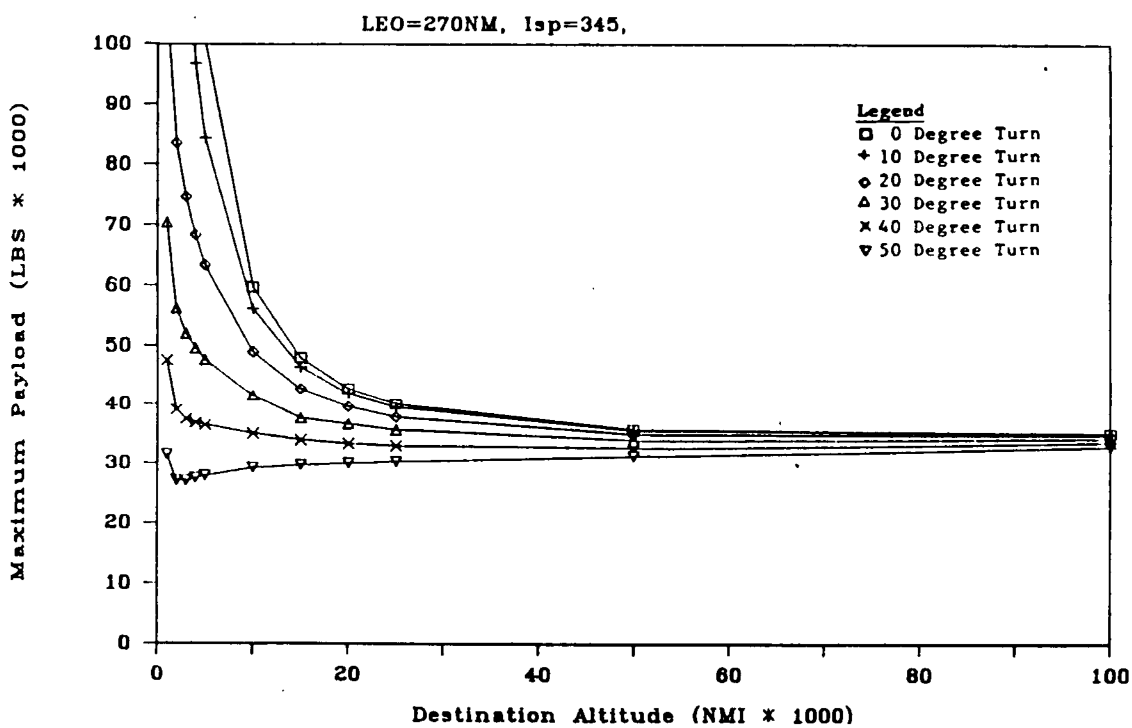

Figure 6.2.2.4-2 SB 90Klb Storable OTV Performance

6.3 MULTIPLE PAYLOAD CARRIER

Figure 6.3-1 shows an OTV multiple payload carrier mounted within the orbiter payload bay. The figure shows two PAM-D class and a IUS/INTELSAT class satellite payloads mounted on a centrally located ASE box frame assembly. The PAM-D satellites are each 7 feet in diameter, 9.25 feet long, and weight 2030 pounds when intended for a cryogenic propellant OTV concept. For storable propellant OTV concepts, this weight increases to approximately 3904 pounds, accounting for the expendable apogee kick motors required. The corresponding weights of the IUS/INTELSAT class payload are 6600 pounds for the cryo OTV, 12,788 pounds (including apogee kick motor) for the storable OTV. The ASE box frame is approximately 8 feet long between satellites and is sized for the heavier loads at 3904 and 12,788 pounds. An additional ASE frame is added to support the opposite end of the heavy payload. Total overall length shown would be approximately 48 feet.

The payloads are each mounted on a conical payload attach fitting with an integral spin table and release mechanism. The spin tables are attached at the payload/OTV subframe which is shaded in the figure. Figure 6.3-2 and 6.3-3 show the configuration of the three payload multiple payload carrier with its payloads when installed on a ground-based cryogenic OTV. Figure 6.3-2 shows a side view, while Figure 6.3-3 is a section plan to showing the attachment of the payloads to the OTV. During the transfer from the orbiter payload bay to the OTV, the payload subframe is disconnected from the control ASE box frame and the forward ASE frame is disconnected from the heavy payload. The payloads are then transferred, while remaining attached to the payload subframe (shaded area), to the OTV attachment grid. The attachment grid has multiple attachment points which permits the payloads to be positioned with the required center of gravity.

Figure 6.3-4 shows a geomod generated view of the multiple payload carrier in the three spacecraft configurations. Figure 6.3-5 shows an exploded geomod generated view of a four PAM-D class configuration as required for installation in the orbiter payload bay. Two payloads are mounted on each of the payload subframes, which are in turn, attached to the central ASE box frame. The central ASE box frame remains in the orbiter payload bay after the payloads have been transferred to the OTV in LEO.

Table 6.3-1 gives the weight estimates for the multiple payload carrier in the four and three spacecraft configurations. These weight estimates are based on an aluminum central ASE frame (the structure that remains in the orbiter). A weight reduction can be realized by optimizing the design with the use composite materials. The subframe structure that flys with the OTV (where weight is more critical) uses composite materials in its design.

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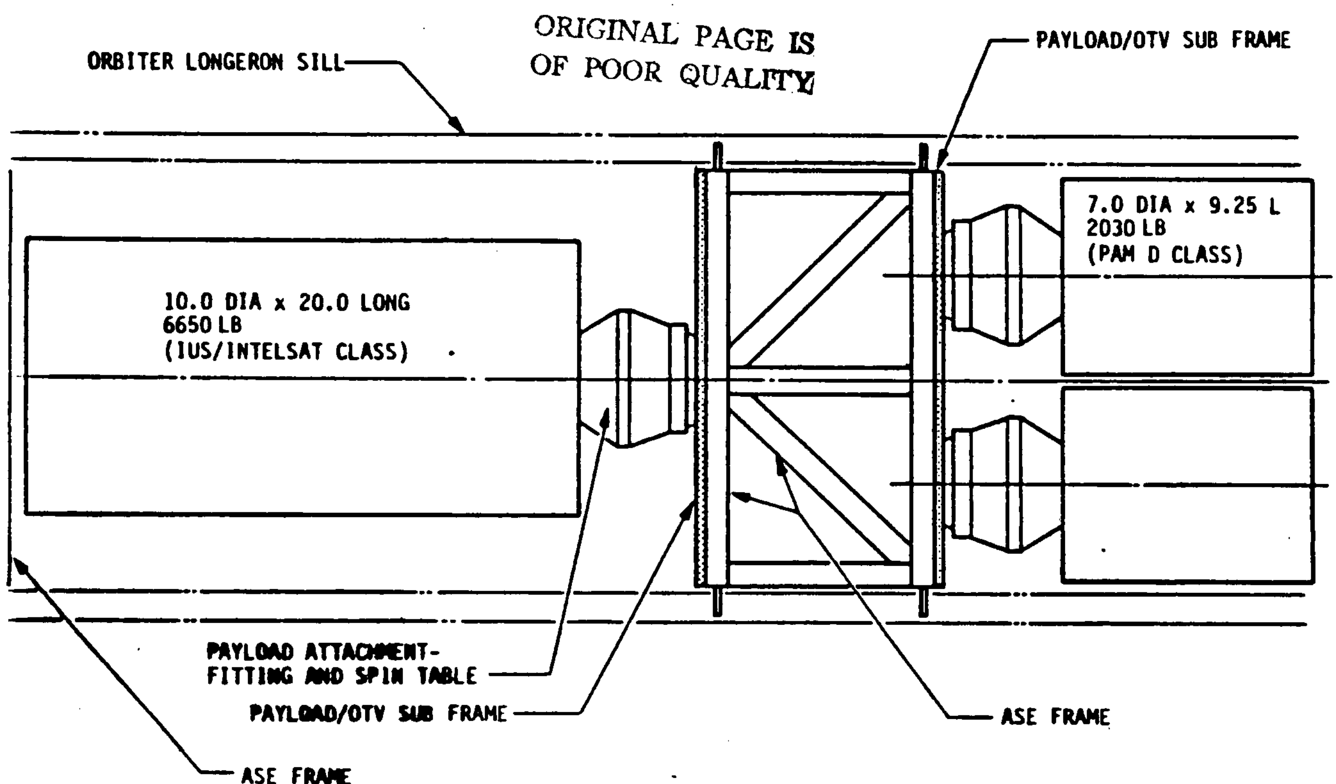

Figure 6.3-1 Multiple Payload Carrier and STS ASE

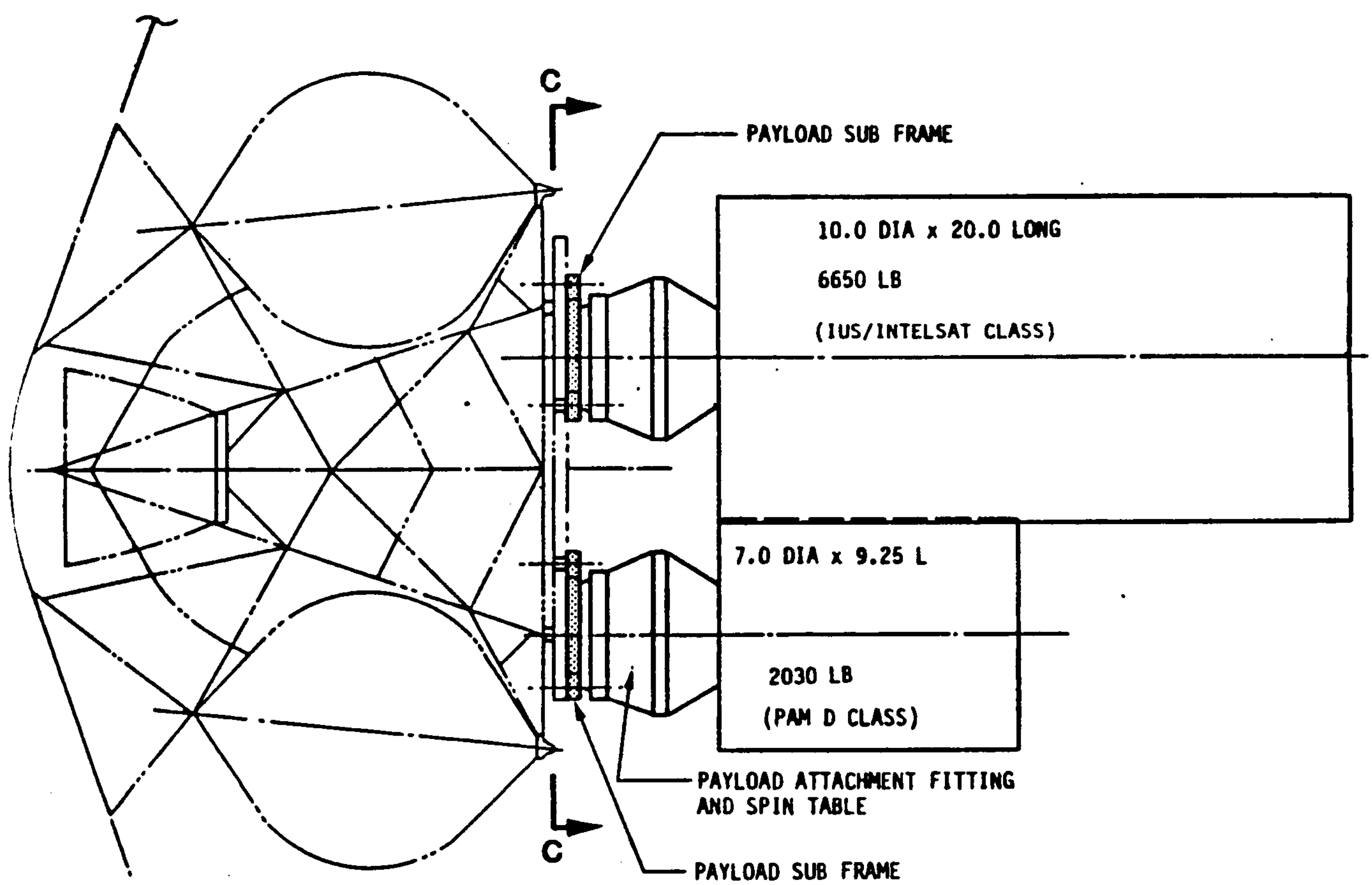

Figure 6.3-2 Multiple Payload Carrier and OTV (Side View)

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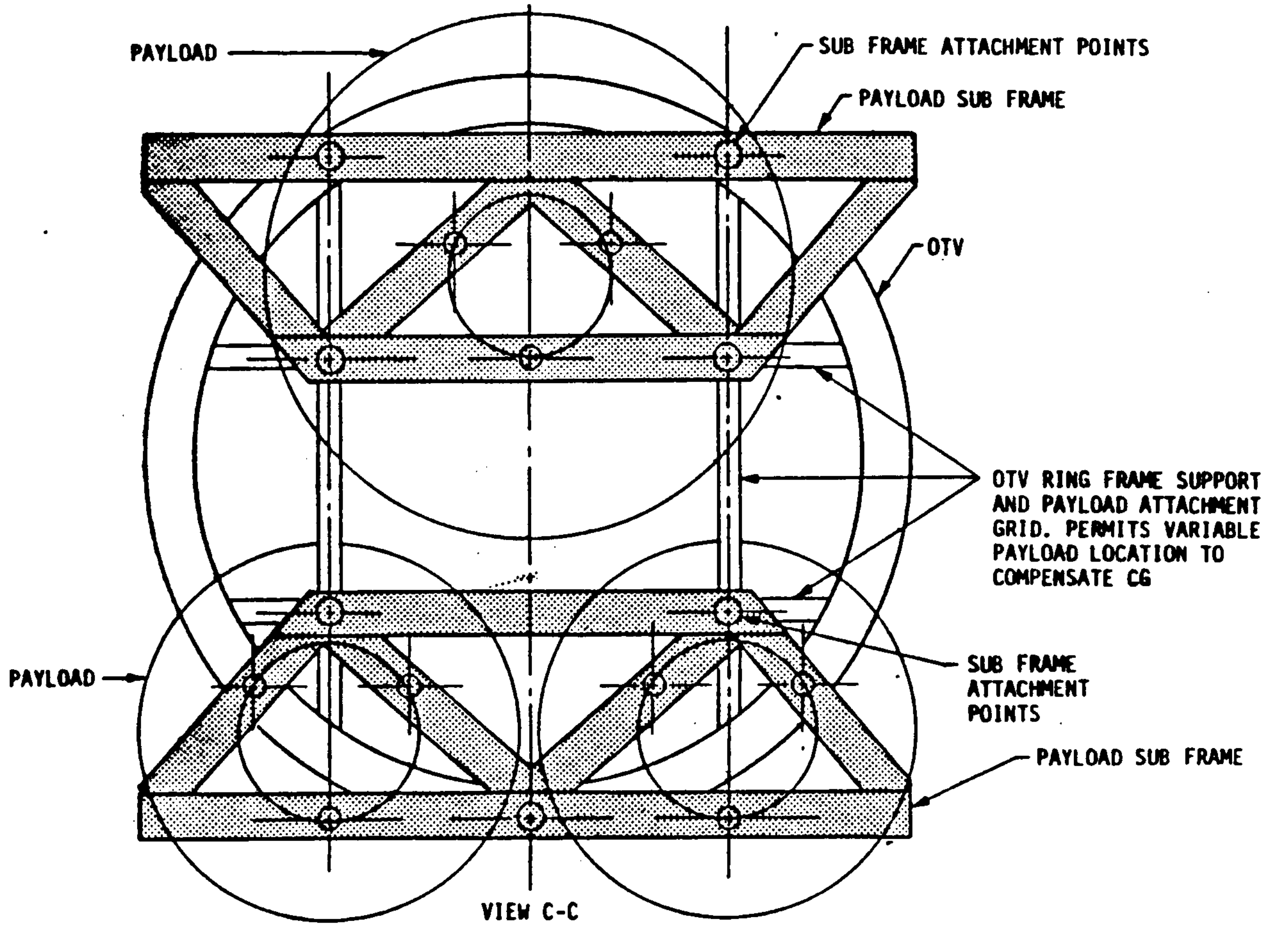


Figure 6.3-3 Multiple Payload Carrier (End View)

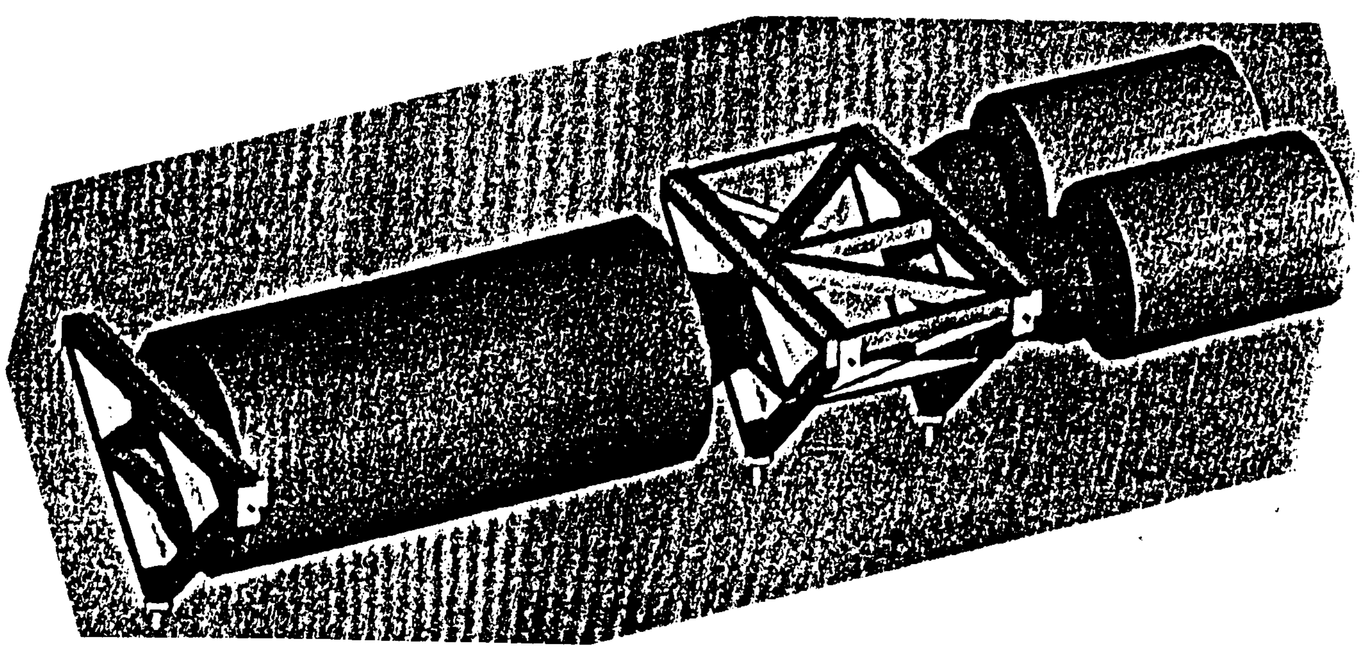


Figure 6.3-4 Multiple Payload Carrier with 3 Spacecraft

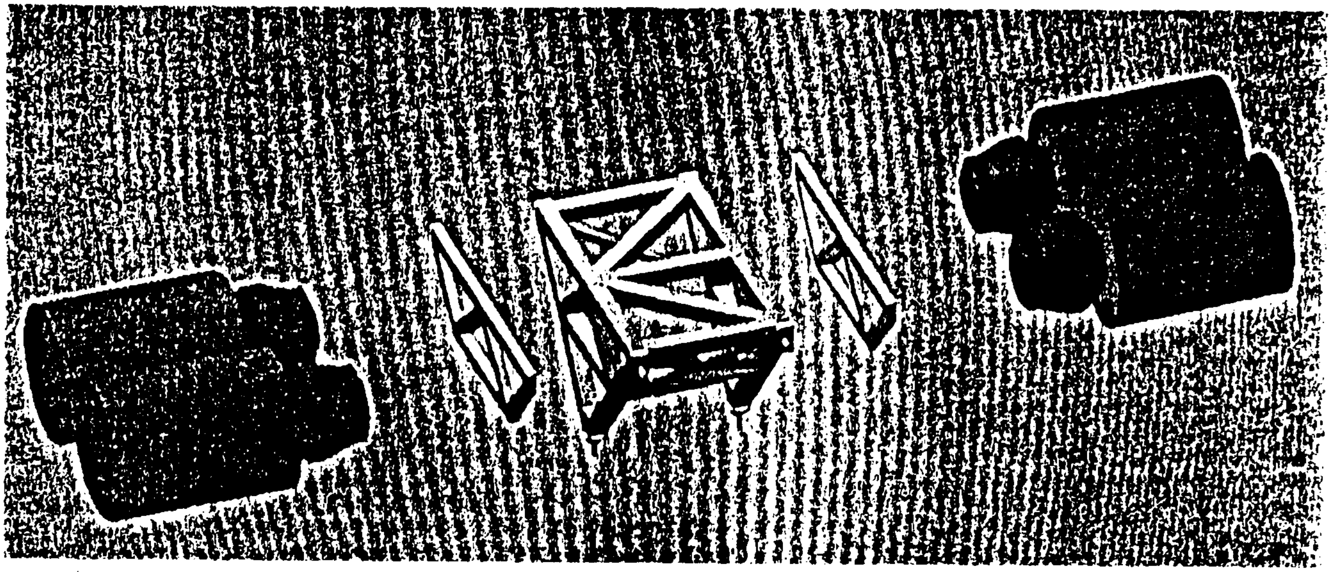


Figure 6.3-5 Multiple Payload Carrier with 4 Spacecraft

Table 6.3-1 OTV - Multiple Payload Carrier (Aluminum)

DESCRIPTION	WEIGHT (LBS) (4) PAYLOADS		(3) PAYLOADS	
AIRBORNE SUPPORT EQUIPMENT (ASE)		2305		3190
FWD TRUSS	--		633*	
AFT TRUSS	1872*		1872*	
KEEL BEAM TRUNNION	44		66	
LONGERON TRUNNION	68		102	
ATTACH - P/L FWD	--		50	
AVIONICS	20		20	
CONTINGENCY (15%)	301		417	
PAYLOAD CARRIER - AIRBORNE TO GEO		485		460
PAYLOAD/OTV SUBFRAME	152		152	
OTV ADAPTER BEAMS	68		68	
HARD POINTS	180		160	
AVIONICS	21		17	
CONTINGENCY (15%)	64		63	
PAYLOAD DEPLOYMENT (SPIN TABLES)		1824		1368
PAYLOADS				
(4) PAM-D/PAM-DII		8120		
(2) PAM-D PLUS (1) IUS/TOS/INTELSAT				10710
TOTAL CARRIER		12734		15728

* COMPOSITE MATERIALS WOULD SAVE APPROXIMATELY 468 LBS AND 634 LBS FOR THE (4) PAYLOAD AND (3) PAYLOAD CARRIER.

6.4 EVOLUTIONARY STRATEGY

This section presents a summary of the logic that went into our selection of the optimum OTV evolutionary strategy. The details backing up this logic are reported in Volumes III and VI. This summary encompasses the programmatic justification for OTV propellant selection, candidate evolutionary paths for the selected cryogenic approach, and program comparison data leading to our specific program recommendation.

The problem was approached in the broadest sense, showing both the benefit of undertaking the program as well as establishing the most desirable approach. A reusable space-based OTV functioning in conjunction with the Space Station was shown to be an important asset in the 1983 "Space Station Needs, Attributes, and Architectural Options Study". An important output of this study is a validation of this conclusion from the vehicle designers perspective. It is particularly important to prove that a significant advantage exists over the current expendable approach to high orbit access. The environment is competitive and the reason OTV is being considered for development is the attractiveness of reducing the cost of payload delivery, as well as providing a new roundtrip capability. If its advantage in the delivery mode is significant, it can justify earlier development of a ground-based capability and can make the STS more attractive to users, thus increasing self sufficiency. Of course any delivery cost advantage must be evaluated in the light of how rapidly it can pay back its development investment.

Any development recommendations are to be justified by the 'low' Revisions 8 OTV Missions Model, by MSFC direction. This mission model is only a projection of the OTV marketplace and should not be viewed as a fixed or absolute opportunity. In this light, the potential growth and flexibility of each option is important. An example of the desired flexibility is to be able to accommodate heavy payloads earlier than specified with little cost impact.

Risks attendant with OTV options and acquisition strategies are important because they reflect the possibility of increased cost. Key factors to be assessed are the risks that cannot be mitigated or controlled by the OTV design, such as STS delivery capacity.

The specific program evaluation factors that are important are as follows:

1) Return on investment
2) Cost per flight vs competition
3) Development cost
4) Payback
5) Risk
6) Growth/flexibility

We have evaluated each of these factors in our assessment of candidate evolutionary strategies.

6.4.1 Cryo/Storable Resolution

We carried the design activity for both cryogenic and storable OTV through the midterm review. A final decision between them was not made until a full operational and space-based accommodations assessment could be included in a full cost analysis. This trade reflects OTV programs that begin with ground based operation and transition to space based operation.

Table 6.4.1-1 summarizes the ground rules and assumptions that were used in conducting this cost analysis. The evaluation was originally conducted for the Rev. 7 'nominal' mission model, but was adjusted to reflect the 'low' Rev. 8 mission model. Figure 6.4.1-1 shows the cumulative discounted comparison of storable and cryogenic systems relative to an all expendable approach using the current/growth expendable upper stage stable (PAM, IUS and Centaur). The comparison shows the difference in program cost (in present value dollars) between the reference expendable program and the reusable program. This shows a payback for a storable investment slightly sooner, but the net advantage over the low model goes to the cryogenic approach.

Table 6.4.1-1 Storable vs Cryo OTV Ground Rules and Assumptions

1) All costs were calculated in 1985 dollars and exclude fees. Present Value (PV) comparisons reflected a 10% discount rate.
2) All cost estimates reflect midterm data (wt. mission model, etc.) generated for the cryogenic and storable stage families
3) DDT&E
   a) Maximum sharing of engineering & tooling efforts between stages was assumed where applicable.
   b) Ground test hardware includes STA, GVTA, MPTA and func. test article
   c) Dedicated flight tests required for the ground-based OTV: no space-based configuration flight test assumed
   d) Flight test articles refurbed to operations spares
   e) Space Station Equipment limited to tank farm impacts
4) Production
   a) Each unique stage assumes an initial production run of 2 units (1 operation, 1 spare (GVTA and FTA are refurbished for GB vehicles)
   b) 92% Wright learning curve assumed: Learning shared across stages
   c) Transportation charges for space-based production hardware included in production (68.5M/STS flt) (1.5 flts/full SB stage)
5) Operations
   a) Payload delivery costs assumed the same, transportation costs not included: No reflights included
   b) Propellant usage based on 421 missions extracted from the midterm, nominal Rev. 7 mission model (32GB, 389SB), adjusted for Rev. 8 low model
   c) ETR launch only: STS CPF = $68.5M: ACC CPF = $2.3M
   d) Mission ops @ 35 man-yrs/yr
6) All cost estimates reflect midterm data (Wt. mission model, etc.) generated for the cryogenic and storable stage families

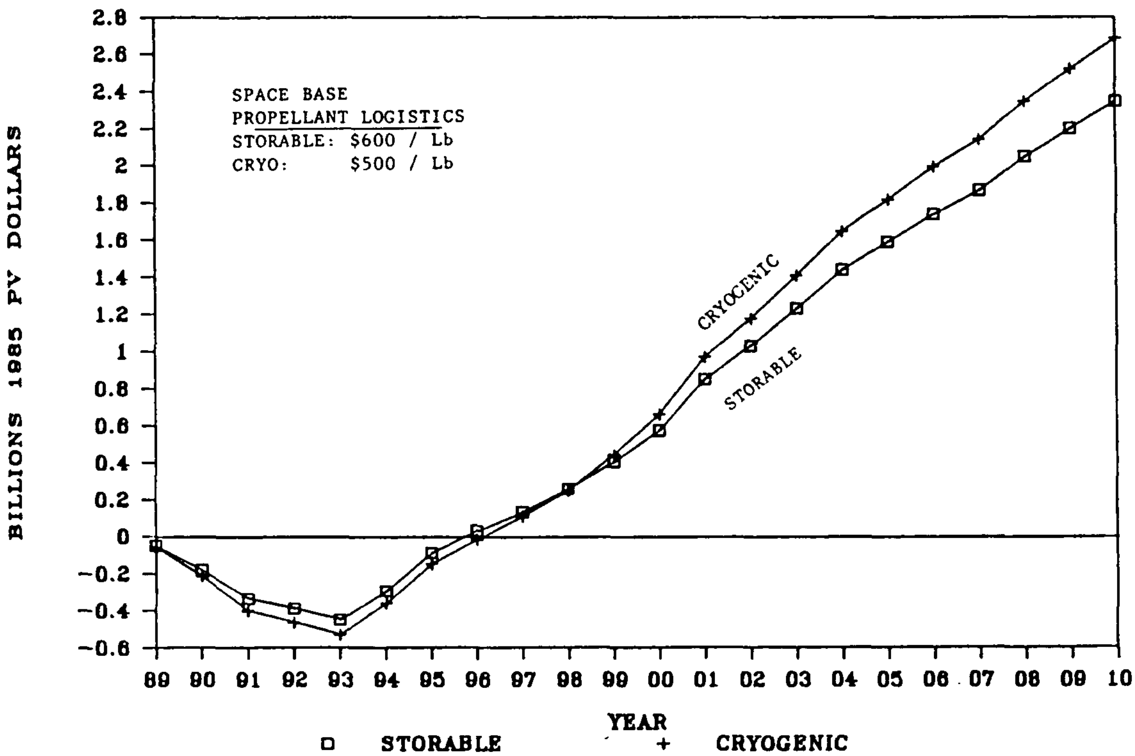


Figure 6.4.1-1 Cumulative Storable/Cryo Comparison

Table 6.4.1-2 summarizes the comparison of investment and return on investment in addition to benefit. The ROI shown is calculated as (operational savings/DDT&E) -1. The storable approach requires less initial investment, but the cryo approach produces more benefit over the expendable fleet and a better return on investment for the low model. This advantage will increase for any more ambitious mission model. As a consequence, even though the cryo advantage is not large for the low model, we feel that cryogenics are clearly the correct selection. Our evolutionary analyses, therefore, were conducted for cryogenic families of Orbital Transfer Vehicles.

Table 6.4.1-2 Cryo/Storable Trade Results (Present Value)

Factors	Storable	Cryo
ROI (Ratio)	3.5	3.7
Benefit (M$)	3010	3415
Investment (M$)	661.5	726.8
Scores		
ROI	9.5	10
Benefits	8.8	10
Investment	10.0	9.1

6.4.2 Alternative Cryogenic Evolutionary Strategies

We evaluated five candidate evolutionary approaches to acquiring a space-based orbital transfer capability, as summarized in Figure 6.4.2-1. A totally ground-based program to achieve the same mission capability was also evaluated. These candidate programs provided the basis for investment/benefits comparisons, and led to our recommendation of the preferred OTV acquisition program. Three of the ultimately space-based programs shown in Figure 6.4.2-1 start with a ground-based OTV while two exist only in the space-based mode. Two of the initially ground-based programs start with an ACC configured OTV, while the third starts with a cargo bay configured OTV. The last parameter explored is whether the space-based OTV should also start unmanned and evolve to a man-rated system. We followed the philosophy that the program selected should be justifiable on the basis of the low Rev. 8 OTV mission model. The manned lunar sortie mission defined in the nominal mission model requires vehicle capability beyond the low model missions. The preferred means for achieving this capability is shown in the last column in Figure 6.4.2-1. These candidate programs were analyzed and are reported on in detail in Volume III of this report, and are summarized in section 6.4.3.

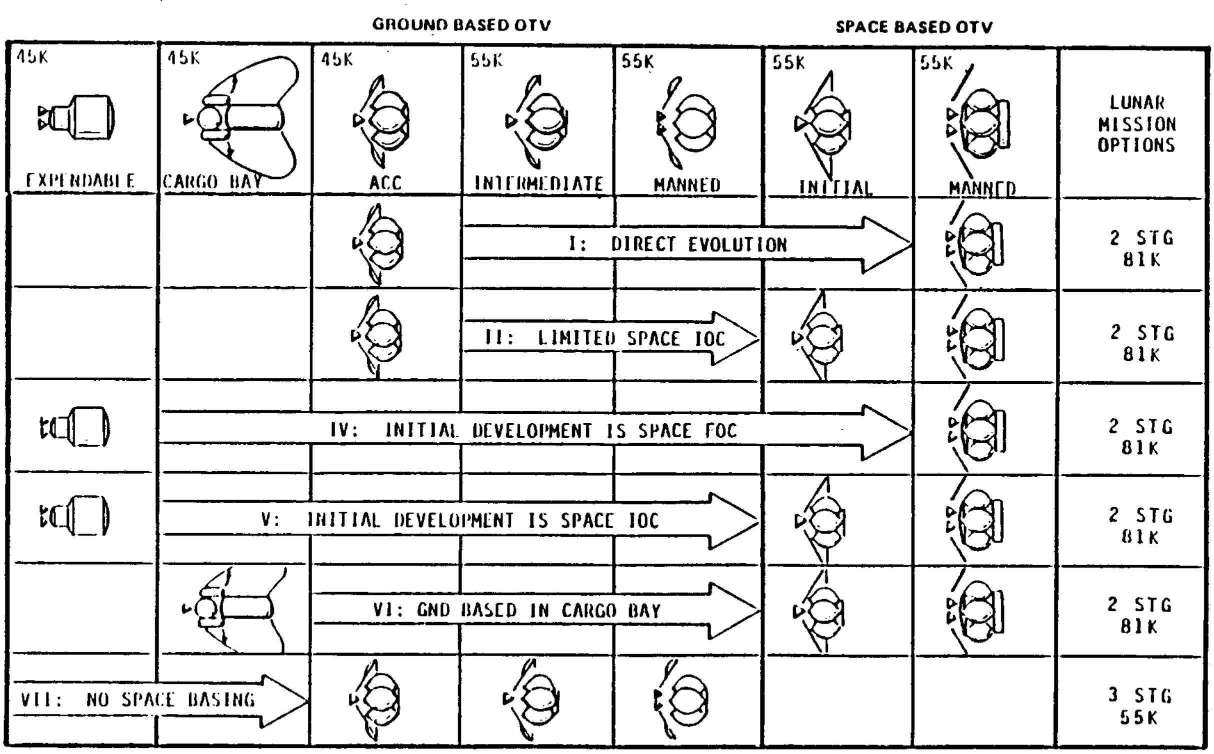
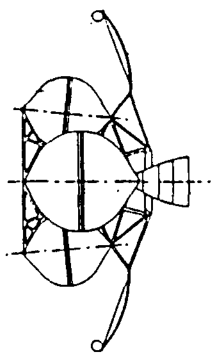


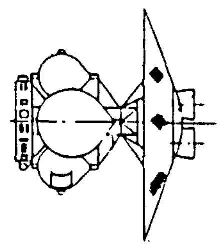
Figure 6.4.2-1 Alternative OTV Growth Paths

Figures 6.4.2-2 through 6.4.2-7 show a high level summary of the vehicles used in each of these program options. Figure 6.4.2-2 shows the two vehicles that comprise Option 1. The initial ground-based vehicle is the one described in detail in section 6.1.1, while the only other configuration is the initial space-based vehicle described in detail in section 6.1.2. Figure 6.4.2-3 shows the vehicles that comprise Option 2. This option adds an intermediate step between the vehicles used in Option 1. An initial non-man-rated vehicle using one main engine and with structure derived directly from the ground-based OTV is introduced. The potential advantage of this approach is lower development cost to become space-based. The full cost of achieving man-rating can be delayed. Option 4, as described in Figure 6.4.2-4, replaces the reusable ground-based ACC OTV with the expendable Centaur. Option 5, Figure 6.4.2-5, is the equivalent of option 2 with the expendable Centaur replacing the ground-based reusable ACC OTV. Option 6 is identical to Option 2 with the cargo bay OTV illustrated in Figure 6.4.2-6 replacing the ACC OTV. The final Option 7, Figure 6.4.2-7, is a totally ground-based approach. The initial step is the ground-based ACC OTV. The final Option 7, Figure 6.4.2-7, is a totally ground-based approach used in Options 1 and 2. It is followed with a non-man-rated version with propellant capacity increased to enable performance of the 20K delivery mission with the use of a second Shuttle flight. The final vehicle in this option is a man-rated version used only for the manned GEO missions.



AVIONICS:	INTEGRAL
STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	40 FT
REDUNDANCY:	NON-MAN RATED
PROP CAP:	45,000 Lb
LOADED WT:	50,363 Lb
ENGINE:	475 Isp/7500 Lb (1)

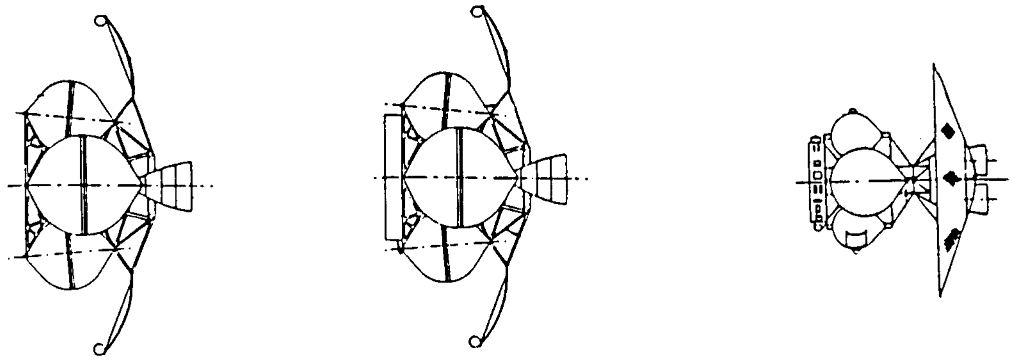
GROUND BASED
ACC DELIVERY



AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	44 FT
REDUNDANCY:	MAN RATED
PROP CAP:	55,000 Lb
LOADED WT:	62,169 Lb
ENGINES:	475 Isp/7500 Lb (2)

SPACE BASED
CB DELIVERY

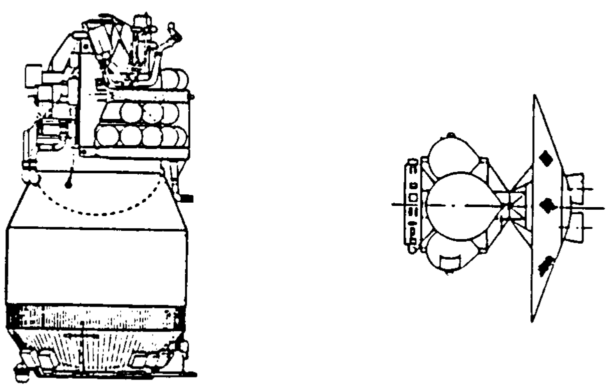
Figure 6.4.2-2 Option 1 - GB to Man-Rated SB



AVIONICS:	INTEGRAL	AVIONICS:	RING	AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY	STRUCTURE:	GRAPHITE EPOXY	STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	40 FT	AEROBRAKE:	40 FT	AEROBRAKE:	44 FT
REDUNDANCY:	NON-MAN RATED	REDUNDANCY:	NON-MAN RATED	REDUNDANCY:	MAN RATED
PROP CAP:	45,000 Lb	PROP CAP:	52,500 Lb	PROP CAP:	55,000 Lb
LOADED WT:	50,363 Lb	LOADED WT:	58,282 Lb	LOADED WT:	62,169 Lb
ENGINE:	475 Isp/7500 Lb (1)	ENGINE:	475 Isp/7500 Lb (1)	ENGINES:	475 Isp/7500 Lb (2)

GROUND BASED ACC DELIVERY	SPACE BASED ACC DELIVERY	SPACE BASED CB DELIVERY

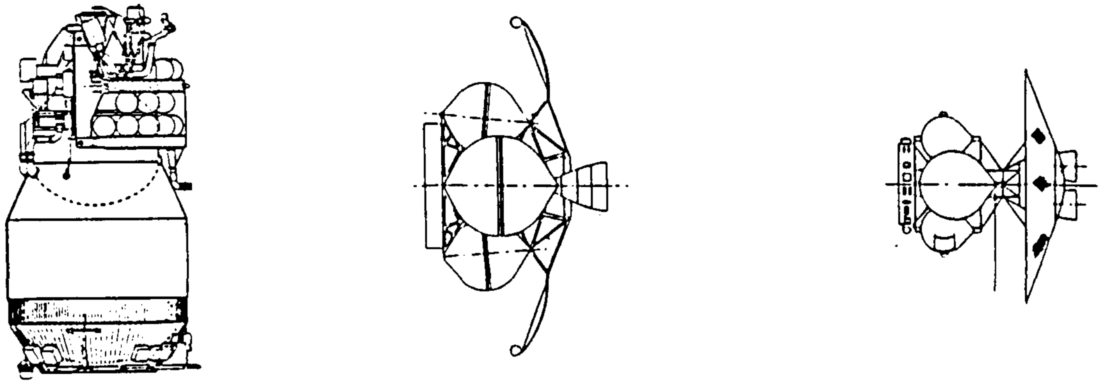
Figure 6.4.2-3 Option 2 - GB to SB Followed by Man-Rating



AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	44 FT
REDUNDANCY:	MAN RATED
PROP CAP:	55,000 Lb
LOADED WT:	62,169 Lb
ENGINES:	475 Isp/7500 Lb (2)

EXPENDABLE CB DELIVERY	SPACE BASED CB DELIVERY

Figure 6.4.2-4 Option 4 - Expendable to SB Man-Rated



AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	40 FT
REDUNDANCY:	NON-MAN RATED
PROP CAP:	52,500 Lb
LOADED WT:	58,283 Lb
ENGINE:	475 Isp/7500 Lb (1)

AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	44 FT
REDUNDANCY:	MAN RATED
PROP CAP:	55,000 Lb
LOADED WT:	62,169 Lb
ENGINES:	475 Isp/7500 Lb (2)

EXPENDABLE
CB DELIVERY

SPACE BASED
ACC DELIVERY

SPACE BASED
CB DELIVERY

Figure 6.4.2-5 Option 5 - Expendable to SB Followed by Man-Rating

GROUND BASED CARGO BAY OTV

VEHICLE DATA

CRYOGENIC PROPELLANT	
TANK SIZE	48434 lbs
DRY WEIGHT	8642 lbs
LOADED WEIGHT	57076 lbs
ASE	5000 lbs
PAD WEIGHT	62076 lbs
SINGLE ENGINE	
THRUST	7500 lbs
ISP	475 sec
AVIONICS: SINGLE FAULT TOLERANT	

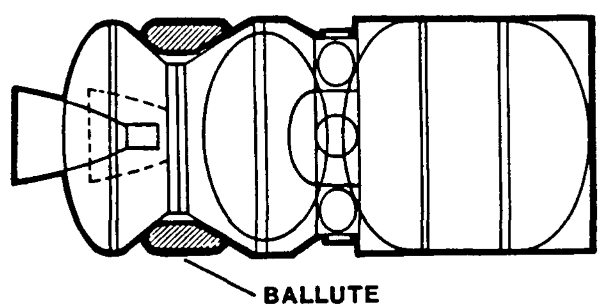
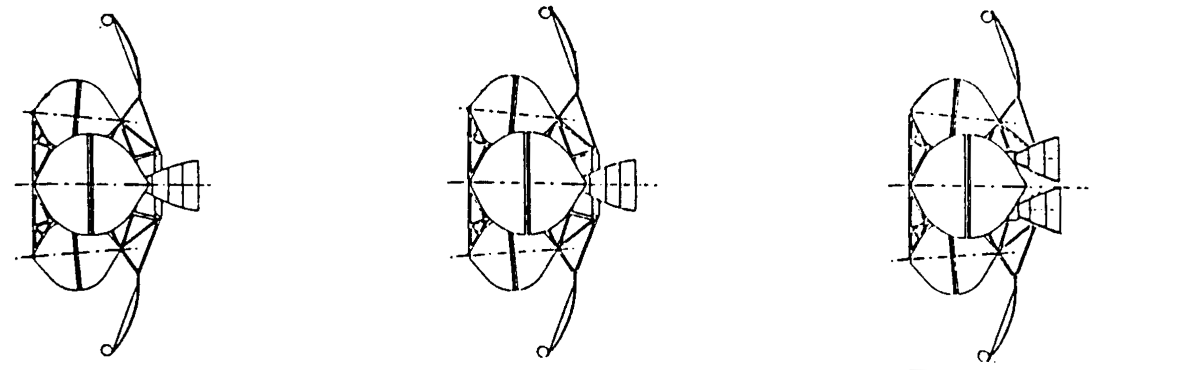


Figure 6.4.2-6 GB Cargo Bay OTV



AVIONICS:	INTEGRAL	AVIONICS:	INTEGRAL	AVIONICS:	INTEGRAL
STRUCTURE:	GRAPHITE EPOXY	STRUCTURE:	GRAPHITE EPOXY	STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	40 FT	AEROBRAKE:	40 FT	AEROBRAKE:	38 FT
REDUNDANCY:	NON-MAN RATED	REDUNDANCY:	NON-MAN RATED	REDUNDANCY:	MAN RATED
PROP CAP:	45,000 Lb	PROP CAP:	54,000 Lb	PROP CAP:	51,000 Lb
LOADED WT:	50,363 Lb	LOADED WT:	59,472 Lb	LOADED WT:	56,925 Lb
ENGINE:	475 Isp/7500 Lb (1)	ENGINE:	475 Isp/7500 Lb (1)	ENGINES:	475 Isp/7500 Lb (2)

GROUND BASED
ACC DELIVERY

GROUND BASED
ACC DELIVERY

GROUND BASED
ACC DELIVERY

Figure 6.4.2-7 Option 7 - All Ground-Based

6.4.3 Program Selection

The OTV program options described in the previous section were compared with a reference 'competition' program to develop benefit and return on investment parameters. That comparison is significant because it provides the reason for embarking on an OTV development in the near term. It is necessary to show an economic advantage for performing near term delivery missions since there are no near term missions that cannot be performed by existing vehicles. Near term capability requirements do not demand development of an OTV. All missions through 1998 in the low mission model can be delivered by existing expendable upper stages. Even after 1998 it is possible, but not likely cost effective, that heavier payloads as well as manned payloads could be captured by an existing or growth expendable upper stage. The competition for the reusable OTV options was constructed as follows.

The competition consisted of PAM D2, IUS, TOS/AMS, Centaur G' and a growth version of Centaur. The growth Centaur had a 75 percent increase in propellant capacity and was presumed man-rated. Each mission in the 'low' Rev. 8 mission model was flown with the least expensive upper stage capable of supporting it. The total life cycle cost of this competitive program was estimated at $25,364M in 1985 dollars, $4,967M in discounted dollars. Using this array of expendable vehicles, 220 STS launches are required to perform the 145 missions in the low mission model. The resulting cost per STS flight is $120.8M in 1985 dollars, $23.7M in discounted dollars.

The first step in our program comparison was to compare Options 2 and 6. The purpose of this trade was to identify and select the preferred method of delivering the OTV to LEO for the 35 ground-based missions identified in the 'low' Rev. 8 OTV mission model. the results of this trade made the pivotal decision as to whether Option 2 or Option 6 would be traded against the remaining options delineated in Figure 6.4.2-1.

Ground-based delivery of OTV and scavenging of shuttle propellants both involve a selection between cargo bay and ACC. This correlation means it is necessary to evaluate both of these factors simultaneously. Table 6.4.3-1 summarizes the ground rules used for this trade study. They are consistent with the OTV ground rules provided by MSFC. the only addition to these standard requirements is the inclusion of an STS benefits factor. The ACC and cargo bay delivery of OTV for ground-basing have different benefits relative to providing additional payload volume and weight delivery capability. In the case of the ACC, it frees 30 feet of cargo bay for other payloads. The cargo bay concept, depending on length and weight, also makes it possible to manifest other payloads on an OTV mission. The inclusion of this benefit is justified since if a concept must pay for a particular development, it has the right to receive all direct and collateral benefits associated with that development.

Table 6.4.3-1 ACC vs Cargo Bay Trade Study Ground Rules and Assumptions

- General
  - Constant fiscal year 1985 dollars excluding fee and contingency
  - Discount rate of 10% per year
- R&T
  - Assumed $100M for AFE flight and $59M for advanced engine technology base for both candidates
- DDT&E
  - Ground test hardware includes STA, GBTA, MPTA and functional test article
  - Dedicated flight test required: Includes STS delivery, ACC and return charges as appropriate.
  - Flight test and GVTA articles refurbed to operational stages
  - GB ACC version includes ACC DDT&E ($163M); DB version includes $27M impact for orbiter bay modifications
  - Both options include DDT&E impacts for P/L clustering structure
- Production
  - Production for both options includes 2 P/L clustering structures 1) operations, 1 spare)
  - No stage production is required due to refurbishment of DDT&E hardware and low flight rates.
- Operations
  - All missions were manifested within the 72K lbs performance and 60' volume constraints of one STS flight
    - Included hardware dry weight, propellant, ASE
    - ACC Weight Included for ACE version per study ground rule
  - STS user charge at $73M per flight (all missions exceeded 755 of orbiter performance): ACC CPF at $2.3M; KSC launch only
  - Low mission model (35 flights, 1994-1999)
  - Ground-based mission ops @ 34 M-yrs/yr
  - Minimum IVA charge due to P/L mating in ACC version (some missions exceed 24 hrs maximum, small IVA charge due to return flight assumed
  - IVA cost @ $16K/hr
  - Aerobrake life = 1 flight
Engine life = 10 flights
Avionics, ECS str life = 40 flights
  - Ground refurbishment of stage based on a percentage of unit cost and analysis of current orbiter crew sizing
- Facilities
  - Facilities costs include
    - Provisions for manufacturing facility for initial stage and refurbishment hardware
    - Dedicated OTV launch processing facility (KSC)
    - Mission operations area at existing KSC facility
- Benefits
  - STS benefits were calculated based on 50% of weight and volume potential after OTV and P/W were manifested

Program cost for four scavenge and OTV delivery options were calculated. The detailed results of the calculations are presented in Volume III. Table 6.4.3-2 summarizes the results. The four options considered are:

1) Cargo Bay OTV Delivery/ACC Scavenging
2) Cargo Bay OTV Delivery/Cargo Bay Scavenging
3) ACC OTV Delivery/ACC Scavenging
4) ACC OTV Delivery/Cargo Bay Scavenging

It is clear that all combinations are viable solutions, but the ACC/ACC approach is far superior. Either ROI/Benefits or ROI/Investment as decision factors would result in choosing ACC/ACC. It is important to note that this conclusion is based on a relatively low STS flight rate. If a more optimistic rate is assumed, the scavenging benefits of the ACC scavenging concept would increase and thus make it even more attractive. These results and conclusions are sensitive to the assumptions concerning the ratio of scavenge flights to OTV flights. If scavenging was not a factor in the trade study, the cargo bay and ACC delivery ROI would be equal. As a consequence of these results, Option 6 was discarded.

The second step in our program comparison was to compare the remaining options delineated in Figure 6.4.2-1. Table 6.4.3-3 shows the ground rules and assumptions used in developing cost data for the economic evaluation of these options. Relatively high fidelity cost estimating was performed using the OTV WBS framework. Details of this estimation are presented in Volume III. The summary cost for each of the five evolutionary options, including interfacing systems, is shown in Tables 6.4.3-4 and 6.4.3-5 in constant dollars and in discounted dollars respectively. The interfacing systems costs -- Space Station, ACC, etc. -- are included as a ground rule requirement. The cost of payload delivery to LEO is also included by ground rule requirement, and adds $34.4M to the cost per flight of each option. The benefits shown include the STS collateral benefits that the ACC provides to each STS flight in terms of available volume and weight that can be used to deliver other cargos. This benefit reduces the cost of each OTV delivery by $8.6M per flight.

Table 6.4.3-2 OTV Delivery/Scavenging Trade Results

ECONOMIC FACTOR	OPTIONS DEL/SCAV			
	CB/ACC	CB/CB	ACC/ACC	ACC/CB
ROI (RATIO)	.66	.51	.77	.64
BENEFITS (M$)	1289	1155	1495	1361
INVESTMENT (M$)	777.7	763.3	843.2	829
SCORES				
ROI	8.5	6.6	10	8.3
BENEFITS	8.6	7.7	10	9.1
INVESTMENT	9.8	10	9.1	9.2

Table 6.4.3-3 OTV Evolutionary Program Trade: Ground Rules and Assumptions

- GENERAL
  - CONSTANT FISCAL YEAR 1985 DOLLARS EXCLUDING FEE AND CONTINGENCY
  - DISCOUNT RATE OF 10% PER YEAR ASSUMED: SPENDING CONSISTENT WITH IOC AND MISSION MODEL REQUIREMENTS

- R&T
  - ASSUMED $100M FOR AFE FLIGHT AND $59M FOR ADVANCED ENGINE TECHNOLOGY BASE FOR BOTH CANDIDATES

- DDT&E
  - GROUND TEST HARDWARE FOR INITIAL STAGE INCLUDES FULL STA, GVTA, MPTA AND FUNCTIONAL TEST ARTICLE: FOLLOW-ON STAGES INCLUDE GROUND TEST HARDWARE AS REQUIRED
  - DEDICATED FLIGHT TEST REQUIRED FOR INITIAL STAGE: INCLUDES DELIVERY AND PROPELLANTS
  - NO DEDICATED FLIGHT TEST FOR FOLLOW-ON STAGES
  - GVTA AND FLIGHT TEST ARTICLE OF INITIAL STAGE REFURBISHED TO MEET OPERATIONS REQUIREMENTS
  - MAXIMUM SHARING OF ENGINEERING AND TOOLING EFFORTS BETWEEN STAGES WAS ASSUMED WHERE APPLICABLE (EVOLUTIONARY APPROACH)
  - ALL OPTIONS INCLUDE DDT&E FOR P/L CLUSTERING STRUCTURE
  - SUPPORTING PROGRAM DDT&E (SPACE STATION ACCOMMODATIONS AND TANK FOR ACC AND PROPELLANT SCAVENGING WERE INCLUDED PER GROUND RULES AS APPLICABLE
    - COSTS WERE INCURRED CONSISTENT WITH BASELINE SCHEDULES AND IOC REQUIREMENTS

- PROVISIONS
  - EACH EVOLUTIONARY STAGE REQUIRES TWO STAGES AT IOC (1 OPERATIONS UNIT, 1 SPARE)
    - REFURBISHED DDT&E HARDWARE CREDITED TO INITIAL OPTION STAGE
    - NO LEARNING ON STAGES ASSUMED DUE TO SMALL PRODUCTION RUN
    - EACH EVOLUTIONARY OPTION STAGE REQUIRES 2 P/L CLUSTERING STRUCTURES (1 OPERATIONS UNIT, 1 SPARE)
    - TRANSPORTATION CHARGES OF PRODUCTION HARDWARE ALLOCATED TO OPERATIONS

- OPERATIONS
  - P/L TRANSPORTATION COSTS INCLUDED FOR ALL OPTIONS ACCORDING TO STS PROGRAM USER CHARGE GUIDELINES
    - 1994-1998 P/L's AND GB OTV STAGES WERE CONSIDERED AN INTEGRAL P/L UNIT AND CHARGED ACCORDINGLY
    - SPACE-BASED PAYLOADS (1999-2010) WERE CHARGED ACCORDING TO USER CHARGE GUIDELINES
    - OPTION 7 (GB EVOLUTIONARY OPTION) P/L's WERE CHARGED IN THE SAME MANNER AS 1999-2020 SB PAYLOADS (P/L GENERALLY CANNOT BE MANIFESTED ON THE SAME FLIGHT AS OTV HARDWARE)

C 3

Table 6.4.3-3 OTV Evolutionary Program Trade: Ground Rules and Assumptions (Continued)

- OPERATIONS (CONTINUED)
  - STS USER CHARGE OF 73M PER FLIGHT, ACC CHARGE OF 2.3M WHERE APPLICABLE
  - LOW MISSION MODEL (145 FLIGHTS)
  - GROUND-BASED MISSION OPS @ 35 M-YR/YR THROUGH OUT OPERATIONS PERIOD
  - EXPENDABLE STAGES (OPTIONS 4 & 5, 1994-1998)
    - OPS COST INCLUDES STAGE CPF AND STS DELIVERY OF STAGE HARDWARE AND MISSION PAYLOAD
  - GROUND-BASED OTV
    - OPERATIONS COST CONSISTENT WITH ACC - CB GB OTV TRADE STUDY
    - OPTION 7 (1999-2010) ASSUMES 1 SHUTTLE FLIGHT PER MISSION FOR OTV STAGE HARDWARE DELIVERY
  - SPACE-BASED OTV
    - SPACE STATION IVA CALCULATED ON A PER MISSION BASIS @ $15K/HR
    - 2 OMV USES PER MISSION COSTS ACCORDING TO STUDY GROUND RULES @ 2 HOURS OUT, 1.5 HOURS BACK AND AVERAGE OF 500 LBS PROPELLANT PER MISSION
    - NO SPACE-BASED MISSION OPS OR EVA REQUIRED
    - STS COSTS INCLUDE DELIVERY OF INITIAL OPERATIONAL UNIT AND SPARES AS REQUIRED
    - ON-ORBIT PROPELLANT COSTS ARE THE COMPOSITE AVERAGE OF SCAVENGED AND STS TANKER DELIVERY COSTS, DETERMINED BY OPTION USAGE ($330 TO $360/LB)
  - OPERATIONS SPARES
    - STS TRANSPORTATION APPLICABLE ONLY TO SB STAGES
    - AEROBRAKE LIFE = 5 FLIGHTS: 0.34 STS FLTS/BRAKE
    - ENGINE LIFE = 10 FLIGHTS: 0.1 STS FLT/ENGINE
    - AVIONICS, EPS, STR LIFE = 40 FLIGHTS: 1.0 STS FLT/REPLACEMENT

- FACILITIES
  - FACILITIES COSTS INCLUDE
    - PROVISION FOR MANUFACTURING FACILITY SPACE FOR INITIAL STAGE AND SPARES PRODUCTION
    - DEDICATED OTV LAUNCH PROCESSING FACILITY (KSC)
    - MISSION OPERATIONS SPACE AT EXISTING KSC FACILITY

- STS BENEFITS WERE CALCULATED BASED ON 50% OF WEIGHT AND VOLUME POTENTIAL AFTER GB OTV AND P/L WERE MANIFESTED

Table 6.4.3-4 Option Cost Summary - Constant Dollars

INTERFACING SYSTEM	OPTIONS 1 GBU/SBM/SBM	2 GBU/SBU/SBM	4 EXU/SBM/SBM	5 EXU/SBU/SBM	7 GBU/GBU/GBM
Space Station	936.00	936.00	936.00	936.00	0.00
ACC	163.20	163.20	163.20	163.20	163.20
Prop Scav	83.00	83.00	83.00	83.00	0.00
P/L Trans	4995.11	4995.11	4995.11	4995.11	4995.11
Subtotal	6177.31	6177.31	6177.31	6117.31	5158.31
OTV					
R&T	153.00	153.00	153.00	153.00	153.00
DDT&E	1351.49	1414.69	1218.70	1257.60	1223.79
Prod.	145.30	251.10	29.90	145.30	242.30
OPS	6408.21	6098.01	8754.00	8443.00	12332.21
Subtotal	8058.00	7916.80	10155.60	9998.90	13951.30
TOTAL	14235.41	14094.11	16332.91	16176.21	19109.61

Table 6.4.3-5 Option Cost summary - Discounted Dollars

	OPTIONS				
	1	2	4	5	7
INTERFACING SYSTEM	GBU/SBM/SBM	GBU/SBU/SBM	EXU/SBM/SBM	EXU/SBU/SBM	GBU/GBU/GBM
Space Station	315.50	315.50	315.50	315.50	0.00
ACC	92.60	92.60	57.53	57.53	92.66
Prop Scav	30.75	30.75	30.75	30.75	0.00
P/L Trans	790.00	790.00	790.00	790.00	790.00
Subtotal	1228.85	1228.85	1193.78	1193.78	882.66
OTV					
R&T	116.94	116.94	72.61	72.61	116.94
DDT&E	692.07	686.32	435.42	421.93	639.90
Prod.	47.28	59.07	8.66	23.33	57.23
OPS	1596.57	1543.63	2416.02	2363.09	2527.33
Subtotal	2452.86	2405.96	2932.71	2880.96	3341.40
TOTAL	3181.71	3634.81	4126.49	4076.74	4224.06

The final five evolutionary strategy options are shown with program costs vs cumulative missions in Figure 6.3.3-1 . At OTV program start, Options 2, 5, and 7 immediately initiate major investments for OTV DDT&E while Options 4 and 5, which begin with expendables, have no initial investment required. At IOC as payback begins from initial flights, the all ground-based Option 7 shows the quickest return with payback at 25 flights. Options 1 and 2 show fast payback until the Space Station accommodations and propellant delivery costs reduce the payback and delay until 42 and 45 flights respectively. Option 2 cost benefits cross over the Option 7 ground-based curve at 66 flights and Option 1 at 71 flights. These curves show how ground-based vs space-based cost trades are impacted by the size and time phasing of the mission model. Options 4 and 5, both using expendables followed by space-based OTVs, delay the break even point to 72 flights for Option 5 and 74 for Option 4.

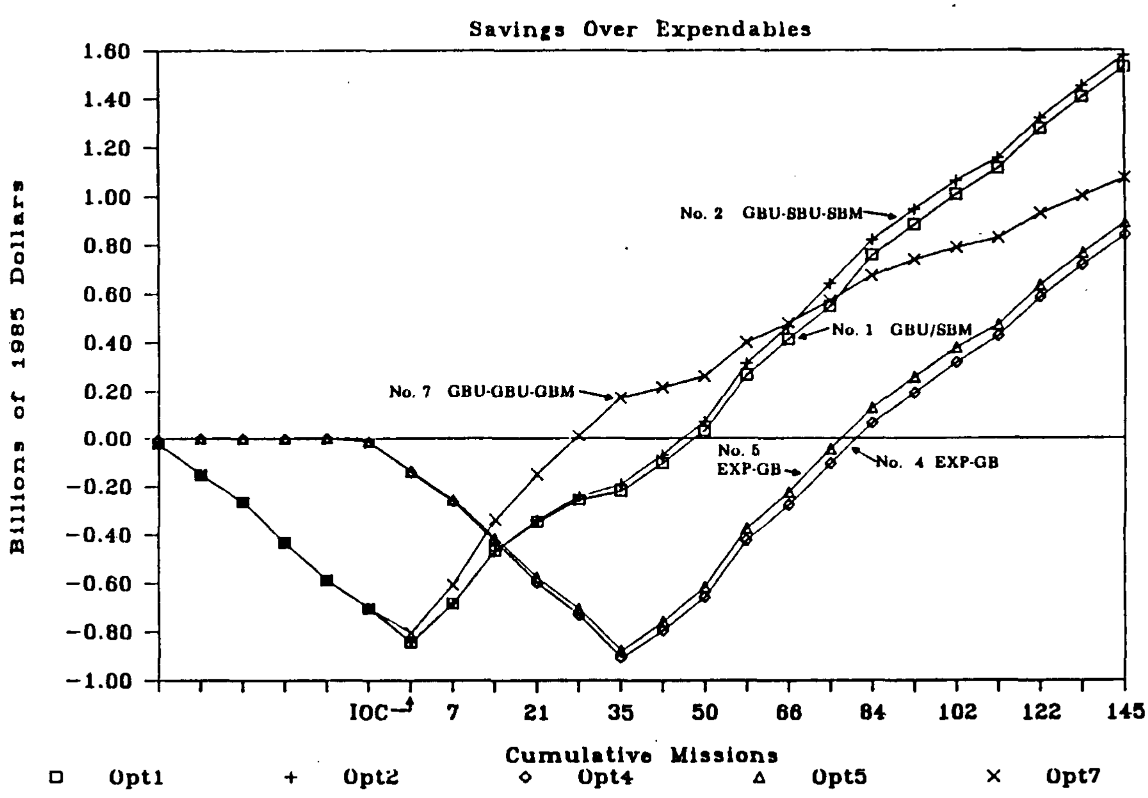


Figure 6.3.3-1 OTV Evolutionary Strategy Comparison

Table 6.4.3-6 shows the principle economic factors for the candidate options along with scoring. The scores are on a base 10. The best candidate is given a ten rating, and all other candidates are assigned proportional scores . No weighting of the factors was made (Investment, Benefits and ROI) however, the combination of ROI and Benefits are considered most important. ROI or Benefits to cost ratio measures the cost effectiveness of an option and Benefits measures the propensity for users to buy the OTV service over existing expendable upper stages. The ratio of any two options shows the relative leverage to attract business away from the expendable upper stages. The chart shows that the three viable candidates are Options 1, 2 and 7. these options have equal ROIs, but when we take ROI and Benefits, Options 1 and 2 are selected with essentially equal scores. Option 7 is attractive when ROI and Investment are considered. This option does carry with it considerable cost risk.

Option 7 is attractive only if the low investment cost, (DDT&E and Production), is real. It is tied directly to the STS user charges. Should the $73M cost to users not be achieved, the attractiveness of the option would be further eroded by a decrease in benefits. As an example, if the user charges were $100M instead of $73, Option 7's benefits would be reduced to $756M (discounted $) which would make the option economically unfeasible. That is, the investment would not be paid back within the 145 missions. Another aspect of Option 7 cost risk is its dependence on the lift capacity of the STS. With the ground ruled 72K lbs capacity, we found that 1.6 Shuttle flights per OTV mission was required. If only 65K lbs is achieved, the benefits over the competition would be reduced to $1625M (discounted $) and the ROI would reduce to 0.79. While still economically profitable, Option 7 would not be as attractive as option 1 and 2. Program considerations are also important reasons for eliminating the all ground--based option. The most important of these is freeing the STS to deliver revenue bearing cargos and to support the Space Station operations. Corollary to this is, if the number of missions for OTV increases, the burden on STS would be significantly increased. On the other hand, the availability of scavenged propellants is key to the cost effectiveness of space basing. If the cost of delivering propellants to the space station degraded to the $1123/pound associated with carrying them in the cargo bay, Option 7 would win over Option 1 by $44M (discounted $). At the present time, the propellant scavenging process is considered to be realistic. Option 1 or 2 is therefore selected over Option 7 because of projected benefits and lower cost risk.

Table 6.4.3-6 OTV Option Results

	OPTION				
DATA	1 GB/SBMR	2 GB/SB/SBMR	4 EXP/SBMR	5 EXP/SB/SBMR	7 GB/GB/GBMR
ROI	1.18	1.21	.93	.98	1.19
BENEFITS	2825.4	2878	1761	1814	1982
INVESTMENT	1295	1301	920	921	907
SCORES					
ROI	9.8	10	7.5	8.1	10
BENEFITS	9.8	10	6.1	6.3	6.9
INVESTMENT	7	7	9.8	9.8	10

The final issue is to select between Options 1 and 2. The major comparative characteristics of these options are:

1) Economics are equal
2) Option 1 maximizes early verification of man-rated reliability
3) Option 1 reduces Space Station operation complexity
   * Only two major program cycles
   * One robotic operation for assembly/disassembly
   * No loss of learning
4) Option 1 provides greater flexibility
   * Earlier heavy payloads capability
   * Earlier manned mission capability
   * Earlier Lunar Mission capability
5) Option 2 has higher cost risk
   * Three major program cycles
   * Avionics repackaging
   * One to two engine transition just prior to manned operation

Considering these differences, the compelling reasons for selection of Option 1 as the preferred OTV evolutionary concept are risk and flexibility. By starting the man-rated concept early, problems will be identified early and eliminated prior to the first manned missions. Starting with a more reliable vehicle will also reduce loss cost. Whenever a system goes through a series of block changes, there is always a cost risk, and DDT&E costs increase when design analyses must be redone. Load, thermal, FMEA, Vibration, Safety, and supporting tests are normally reported when significant changes in configuration are made. Flexibility to accommodate potential mission model changes is also important. By proceeding with a man-rated SBOTV, early heavy GEO missions or earlier manned missions to GEO or the moon could be readily accommodated. The latter would only require increased capacity propellant tanks. Our net conclusion is to recommend Option 1. This option is initially ground-based and transition to a man-rated configuration for space-based operations in 1999. The nature of this selected program is described further in the following sections.

6.5 DEVELOPMENT SCHEDULE

6.5.1 Space Transportation Architecture Summary - An OTV/Space Station/ACC Propellant delivery top level program schedule, shown in Figure 6.5-1, has been prepared to implement the Revision 8 low mission model. The ground-based OTV ATP is January 1998, with PDR in October 1988, and the CDR 9 months later in July 1989. The initial flight OTV is delivered during the third quarter of 1993 for initial flights in 1994. The space-based OTV is shown separately for clarity although an internal part of the same DDT&E program. The man-rated SBOTV begins the DDT&E phase in the first quarter of 1993 with PDR and CDR at 12 and 24 months respectively, leading to delivery in the late 3rd quarter, 1998 for Space Station based and unmanned payload flight in 1999. The main cryogenic rocket engine development would be initiated in 1989 to support the evolution from ground to space-based in 1999.

The dedicated aft cargo carrier ATP is in the first quarter of 1990 with an immediate PDR and CDR 12 months later. Delivery in the third quarter 1993 provides for mating with ET and GBOTV and STS flight in early 1994. Appropriate orbiter and KSC interfaces would be worked through the normal ET/STS/DSC integration organization and are not included in this schedule. The ACC would continue on a parallel schedule with the SBOTV while the propellant scavenging vehicle ATP would be delayed until 48 months prior to SB IOC on January 1995.

This schedule provides the GBOTV and ACC for initial flights in 1994 and SBOTV, ACC and propellant scavenging capability for 1999 SBOTV IOC.

The following four schedules (Figures 6.5.1-2 through 6.5.1-5) provide details for the acquisition of the ground and space-based OTV, the ACC and the scavenging system for the ACC.

The detail schedule for the recommended OTV concept and evolutionary strategy is included as Appendix B in Volume VI, Cost Estimates.

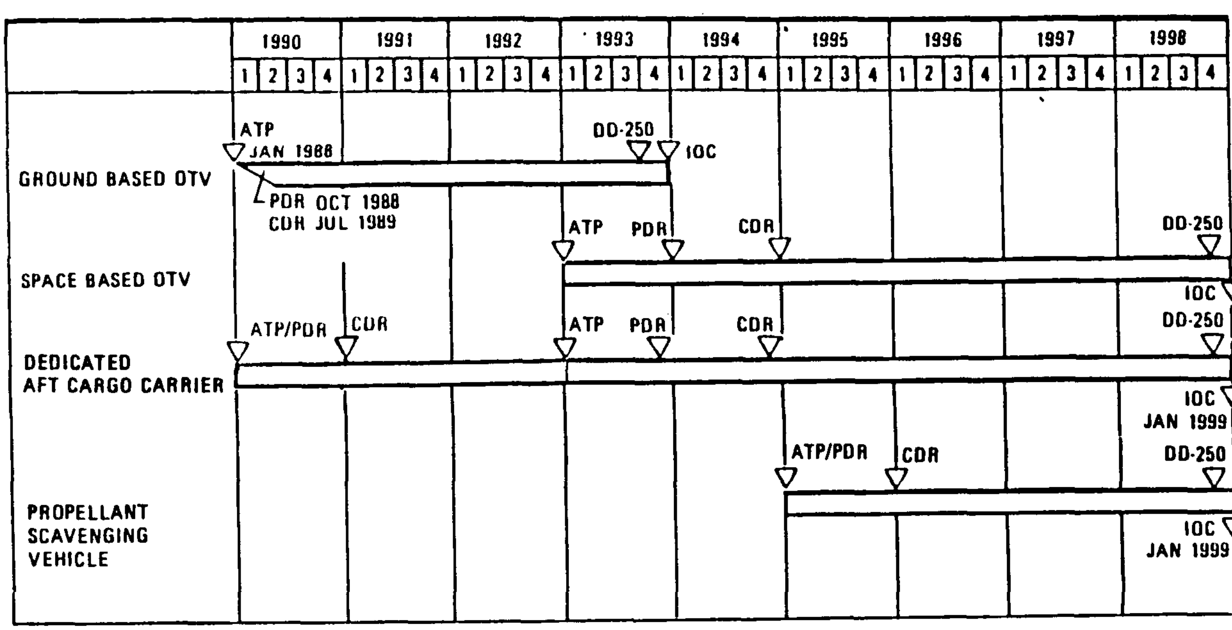


Figure 6.5.1-1 Space Transportation Schedule Summary

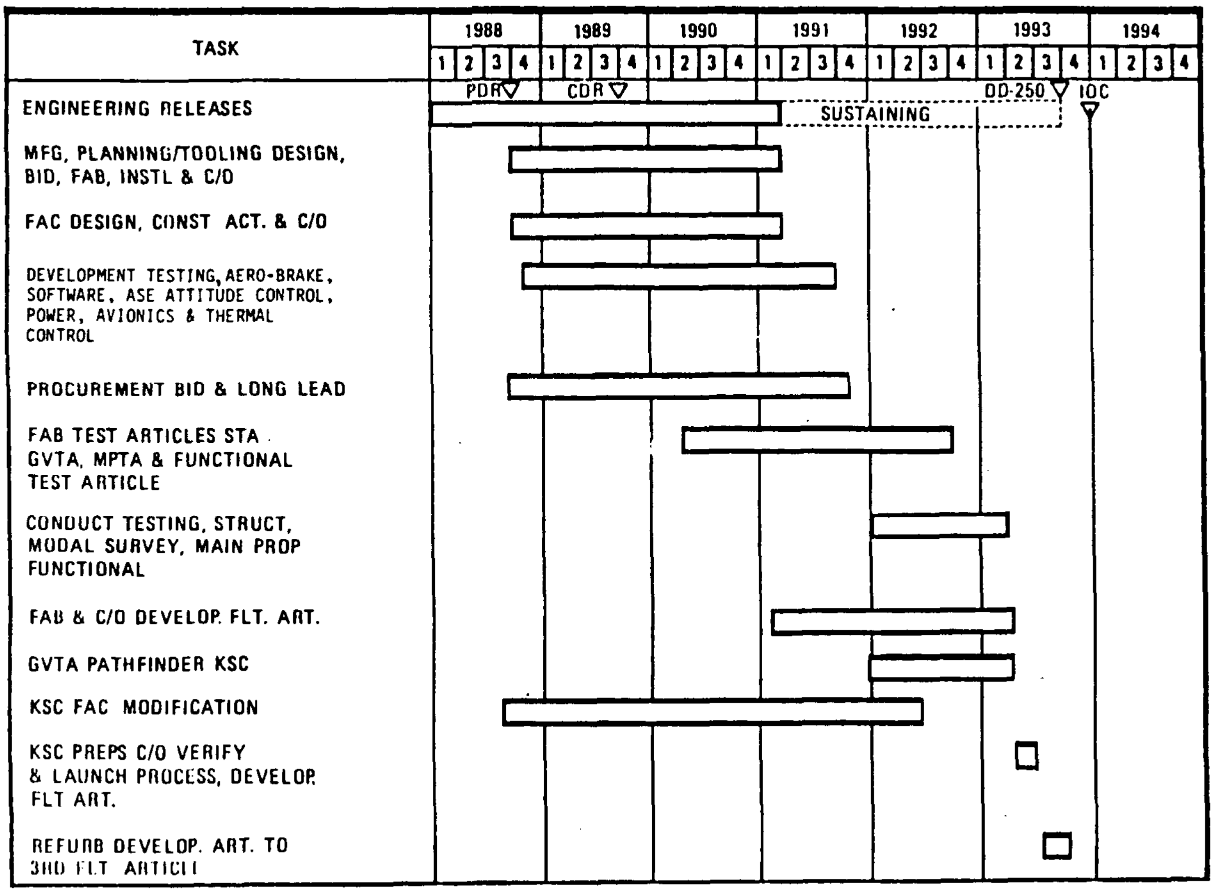


Figure 6.5.1-2 Ground-Based OTV Development

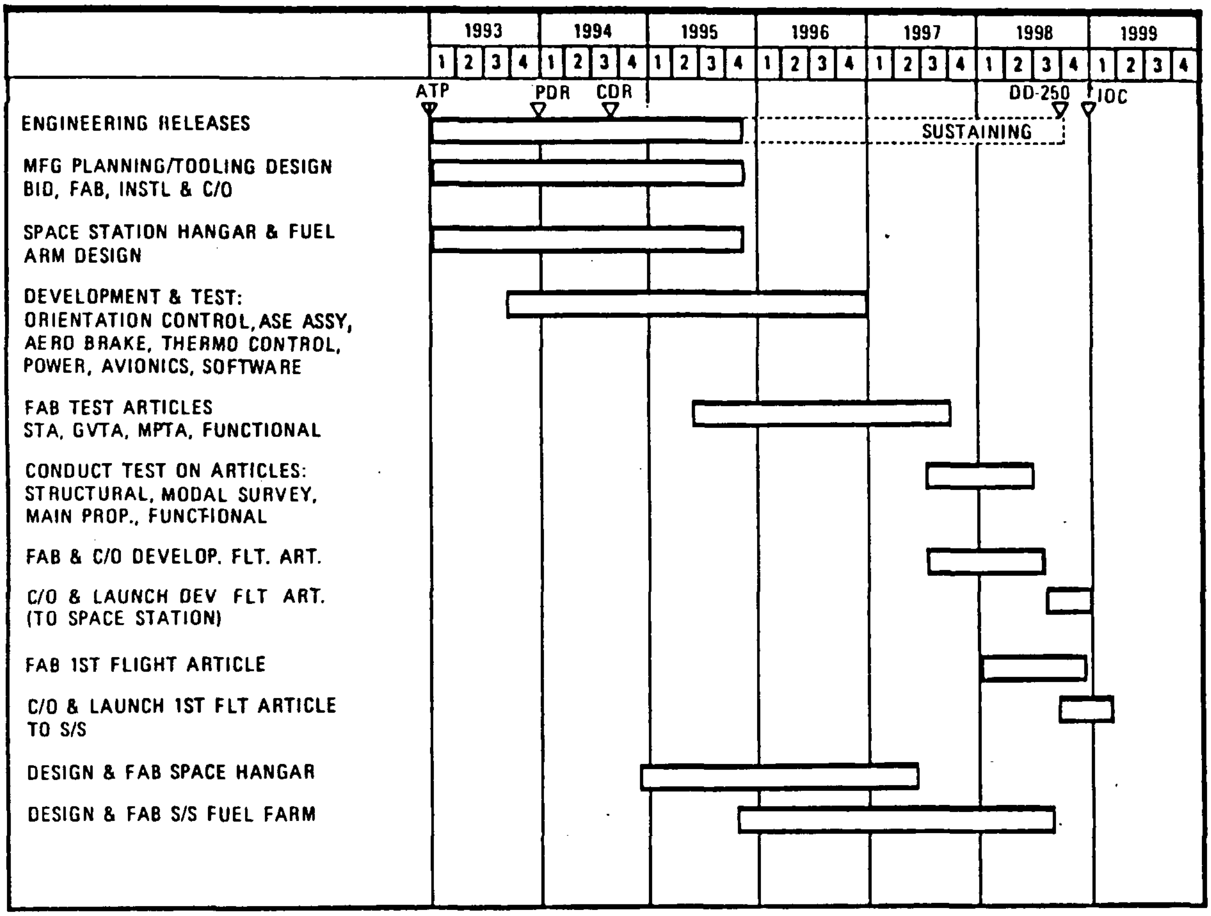


Figure 6.5.1-3 Space-Based OTV Development

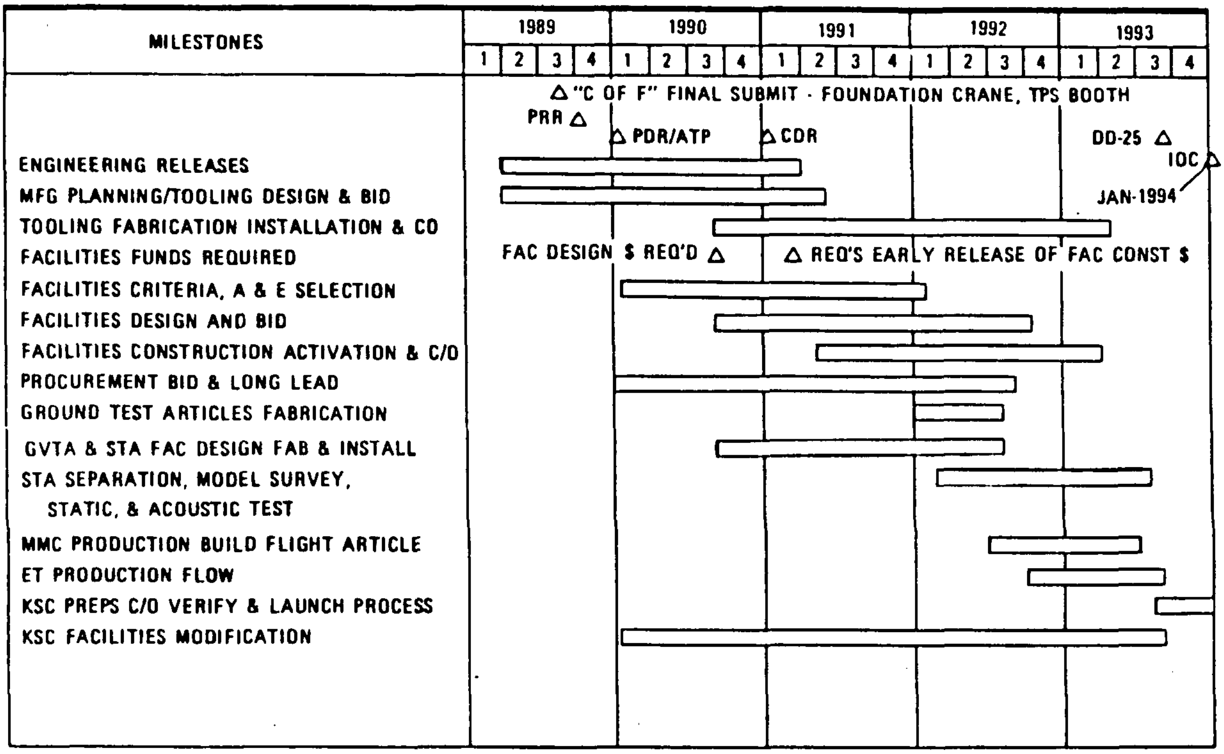


Figure 6.5.1-4 Dedicated ACC Development

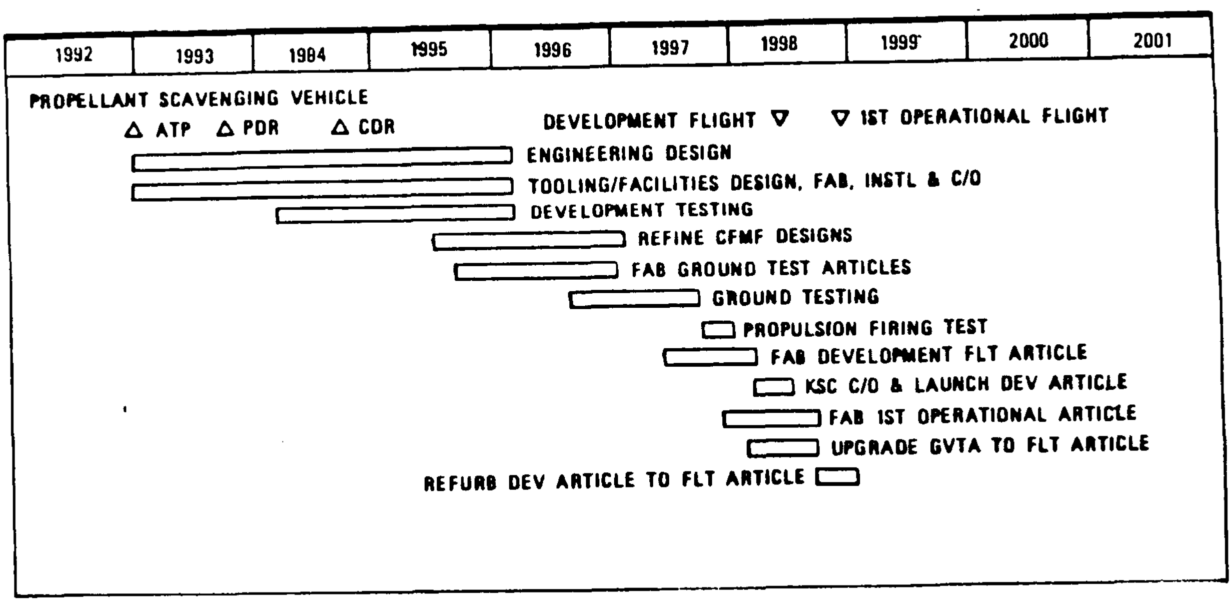


Figure 6.5.1-5 Propellant Scavenging System Development

6.5.2 Space-Based OTV Accommodations Time Phasing by Element - Determination of the space-based OTV accommodations element time phasing is actually not very complicated as accommodations must be available for use in quantum level jumps. The schedule planned is summarized in Figure 6.4.2-1. The propellant tank farm, the servicing and maintenance hangar·with robotics, and the ground support elements must all be in place and operational by the time the space-based OTV is operational. A storage hangar or duplicate servicing and maintenance hangar, and enlargement of the original servicing and maintenance hangar (if necessary) must be in place and operational before the first scheduled 80K Lunar Delivery Mission.

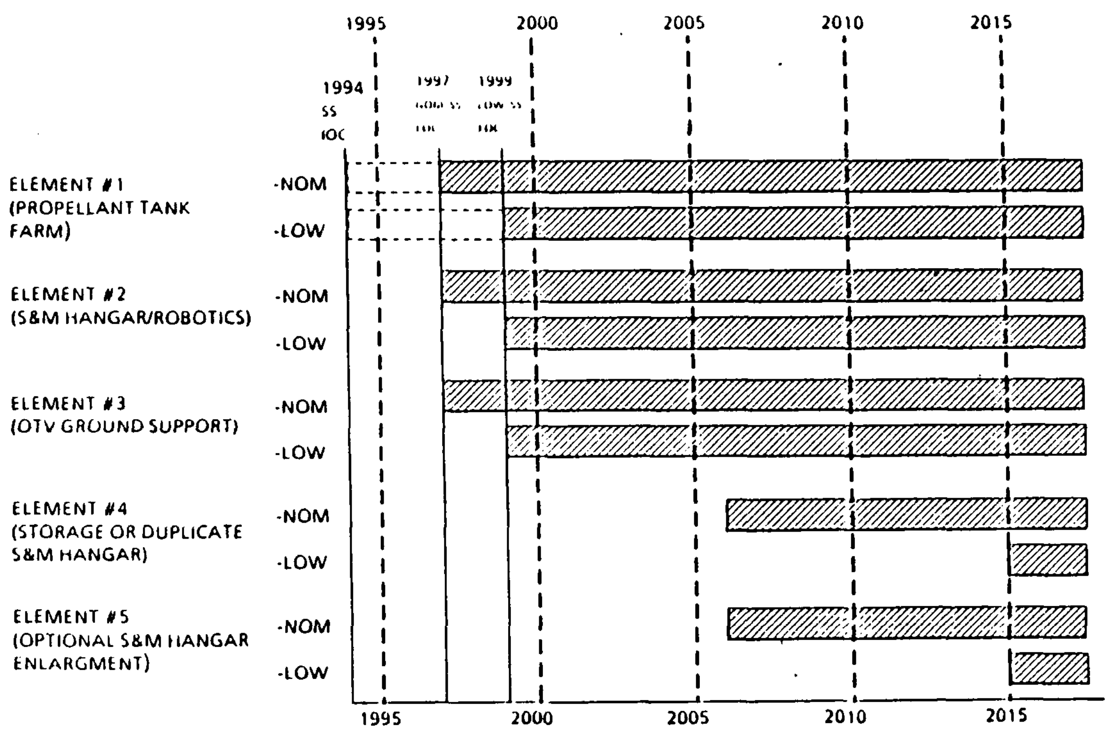
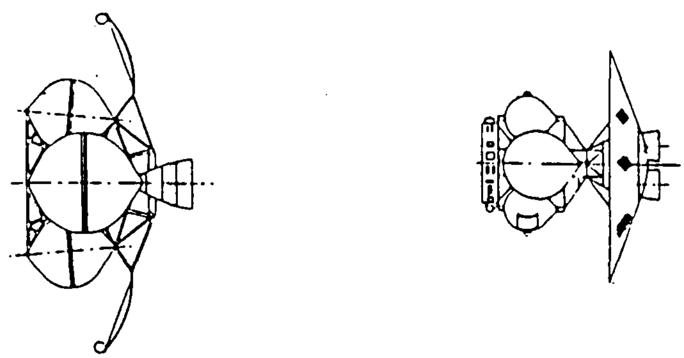


Figure 6.5.2-1 Space-Based Accommodations Time Phasing

6.6 SUMMARY DDT&E AND PRODUCTION COST

6.6.1 Introduction - The evolutionary strategy evaluations resulted in the selection of an ACC ground-based OTV configuration transitioning to a man-rated space-based OTV on 1999. The principle characteristics of this selected option are shown in Figure 6.6.1-1. In this section provides a summary of the DDT&E and production costs.



AVIONICS:	INTEGRAL	AVIONICS:	RING
STRUCTURE:	GRAPHITE EPOXY	STRUCTURE:	GRAPHITE EPOXY
AEROBRAKE:	40 FT	AEROBRAKE:	44 FT
REDUNDANCY:	NON-MAN RATED	REDUNDANCY:	MAN RATED
PROP CAP:	45,000 Lb	PROP CAP:	55,000 Lb
LOADED WT:	50,363 Lb	LOADED WT:	62,169 Lb
ENGINE:	475 Isp/7500 Lb (1)	ENGINES:	475 Isp/7500 Lb (2)

GROUND BASED
ACC DELIVERY

SPACE BASED
CB DELIVERY

Figure 6.6.1-1 Selected Development Option

6.6.2 DDT&E - The premise for development of the DDT&E costs was that the design effort can be accomplished recognizing the planned block change and thereby minimize the cost of transition. Each subsystem's cost was approached on this basis. Table 6.6.2-1 shows a summary of the DDT&E cost broken down by main WBS element. It will be noted that the percentage change in OTV subsystem elements varies from about 1% to 38%. The structural change necessary to accommodate two engines and the avionics ring is the largest. The combination of avionics systems (GN&C, C&DH, electrical power and environmental control) only represents a 15% change. This is because the fundamental circuitry architecture is initially designed for the man-rated mission providing plug in redundancy capability. The major modification to this subsystem is only repackaging into a ring configuration from a structure integral design.

Table 6.6.2-1 shows the total contractor costs for the stage and the payload clustering structure to be $804.7 million to acquire the ground-based capability in 1994 with an additional $241.4 million to man-rate the OTV. Level II costs are those attributable to the NASA functions of management, integration and flight test. Facilities costs include acquisition of KSC and special factory facilities. The flight test costs are for a proto-flight of the ground-based OTV. No additional flight test of the man-rated configuration was deemed necessary since 35 flights of the ground-based OTV will provide sufficient confidence in the design.

Table 6.6.2-1 DDT&E In Constant FY 85 $

	GB ACC	SB MR	TOTAL
D&D	405.9	94.4	500.3
STRUCTURES	15.0	9.3	24.3
TANKS	6.8	5.4	12.2
PROPULSION	8.0	3.5	11.5
ENGINE	175.0	13.7	188.7
ACS	9.1	4.3	13.4
GN&C	81.5	8.6	90.1
C&DH	39.4	5.9	45.3
ELEC PWR	16.6	1.9	18.5
ENV CNTRL	5.5	8.0	13.5
AEROBRAKE	33.2	10.3	43.5
GSE	5.2	.5	5.7
ASE	10.6	2.2	12.8
SSE		20.8	20.8
SE&I	94.0	57.3	151.3
SOFTWARE	61.2	10.1	71.3
TOOLING	19.3	4.3	23.6
TEST HARDWARE	125.2	50.5	175.7
TEST OPS & FIXT	25.0	12.5	37.5
PROG MGT	44.0	12.3	56.3
STAGE DDT&E	774.6	241.4	1016.0
P/L CLUST STRU	30.1		30.1
LEVEL II			
PM,SE&I,TEST	187.0	77.2	264.2
TEST FLT	80.2		80.2
OTV TOTAL	1071.8	318.6	1390.4
FACILITIES	20.0		20.0
DDT&E TOTAL	1091.8	318.6	1410.4

6.6.3 Summary of Production Cost - To minimize front end costs, the acquisition approach is based on refurbishing the ground vibration test article and the flight test article manufactured in the DDT&E phase. Table 6.6.3-1 shows a summary of the ground-based and space-based production costs. The total cost for the ground-based OTV and payload clustering structure is $29.9 million. If new hardware had been built instead of refurbishing test hardware, the cost of two units would have been $128.4 million.

Table 6.6.3-1 also provides the production cost for the space-based OTV at $115.4 million for two units. The low mission model traffic can be captured by one OTV at the Space Station with one back-up on the ground. All subsequent production to support operations has been charged as a part of operations costs.

Table 6.6.3-1 Initial Production Cost (FY 85$)

	GB ACC		SBMB		
	UNIT	PROD	UNIT	PROD	TOTAL PRODUCTION
FLT HARDWARE	38.0		44.6	89.2	89.2
			2.3	4.6	4.6
STRUCTURES	1.4	GND TEST	1.7	3.4	3.4
TANKS	1.6	&	2.1	4.2	4.2
PROPULSION	1.8	FLT TEST	4.0	8.0	8.0
ENGINE	2.0	ARTICLES	1.8	3.6	3.6
ACS	1.3	REFURBED	6.2	12.4	12.4
GN&C	5.7		12.0	24.0	24.0
C&DH	12.0		2.8	5.6	5.6
ELEC PWR	2.6		1.1	2.2	2.2
ENV CNTRL	.7		3.0	6.0	6.0
AEROBRAKE	2.5		7.6	15.2	15.2
A&CO	6.4				
TOOLING & STE	3.6		4.3	8.6	8.6
SUSTAINING ENGR	4.1		4.7	9.4	9.4
SE&I	.8		1.0	2.0	2.0
PROG MGT	2.8		3.1	6.2	6.2
STAGE PROD	49.3		57.7	115.4	115.4
P/L CLUST STRU	14.9	29.9		0.0	29.9
PROD TOTAL		29.9		115.4	145.3

7.0 REFERENCE DOCUMENTS & GLOSSARY

REFERENCE DOCUMENTS

. ICD 80900000025; OTV/Dedicated ACC Interface; 8/1/83; Martin Marietta

2. JSC 07700 Vol XIV (Incl. Attachment 1); Space Shuttle System Payload Accommodations; Johnson Spaceflight Center; Latest Revision

3. Projected STS Lift Capability for Advanced Programs Planning Purposes; August, 1984; Johnson Spaceflight Center

4. TM 82478; Space and Planetary Environment Criteria Guidelines For Use in Space Vehicle Development; Volume 1; 1982 Revision; National Aeronautics and Space Administration

5. NASA TN D-5840; Deployment and Performance Characteristics of 5-foot Diameter Attached Inflatable Decelerators from Mach Number 2.2 to 4.4; H.L. Bohon and Z. Miserentino; August, 1970; NASA Langley Research Center

6. MMC TR-3709014; Viking Aerodynamics Data Book; B.F. Click; December 1970; Martin Marietta

GLOSSARY

The following acronyms and abbreviations are used in the text of this document and are listed here for convenience:

3-DOF	Three Degree of Freedom
A	Area
ACC	Aft Cargo Carrier
ACS	Attitude Control System
ADAM	Aerobrake Deployment Assist Mechanism
AFE	Aeroasisst Flight Experiment
AFRPL	Air Force Rocket Propulsion Laboratory
AKM	Apogee Kick Motor
Al	Aluminum
ALRC	Aerojet Liquid Rocket Company
ATP	Authority to Proceed
ASE	Airborne Support Equipment
C_3	Orbital Energy (km_2/sec^2)
C_D	Drag Coefficient
C&DH	Communications & Data Handling
C&T	Communication and Tracking
C/O	Checkout
CB	Cargo Bay
CDR	Critical Design Review
CFMF	Cryogenic Fluid Management Facility
CG	Center of Gravity
CP	Center of Pressure
CPF	Cost per Flight
CMOS	Complementary Metal Oxide
CPU	Central Processor Unit
CV	Cargo Vehicle
DACS	Data Acquisition and Control System
DAK	Double Aluminized Kapton
DD-250	Material Inspection and Receiving Report (Material Ownership Transfer from Contractor and Government)
DDT&E	Design, Development, Test, and Stet
DMS	Data Management System
DoD	Department of Defense
DRIRU	Dry-Tuned Inertial Reference Unit
DRM	Design Reference Mission
DTG	Dry Tuned Gyro
EKS	Expendable Kick Stage
EPS	Electrical Power Subsystem
EVA	Extra Vehicular Activity
EXP-GB	Expendable Ground Based
FC	Fuel Cell
FMEA	Failure Mode Effects Analysis
FOC	Full Operational Capability
FRCI	Fiber Reinforced Ceramic Insulation
FSI	Flexible Surface Insulation
FTA	Flight Test Article
FY	Fiscal Year
GB	Ground Based
GBM	Ground Based Manned
GBU	Ground Based Unmanned

GN&C	Guidance, Navigation and Control
GH_2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
GPS	Global Positioning System
GVTA	Ground Vibration Test Article
HEO	High Earth Orbit
I.D.	Inside Diameter
I/F	Interface
I/O	Input/Output
IMU	Inertial Measurement Unit
IOC	Initial Operational Capability
IR&D	Independent Research & Development
Isp	Specific Impulse
IRD	Interface Requirements Document
IUS	Inertial Upper Stage
IVA	Intra Vehicular Activity
JSC	Johnson Space Center
Klb	1000 Pounds
KSC	Kennedy Space Center
Kw	Kilowatt
L/D	Lift to Drag Ratio
L	Pound
LCC	Life Cycle Cost
LeRC	Lewis Research Center
LH_2	Liquid Hydrogen
Li	Lithium
LO_2	Liquid Oxygen
LEO	Low Earth Orbit
MAF	Michoud Assembly Facility
MLI	Multi-Layer Insulation
MMH	Mono Methyl Hydrazine
MMS	Multi-Mission Spacecraft
MPS	Main Propulsion System
MPTA	Main Propulsion Test Article
MR	Mixture Ratio (Ox to Fuel, by weight)
MSFC	Marshall Space Flight Center
MTBO	Mean Time
N_2O_4	Nitrogen Tetroxide
nmi	Nautical Mile
NPSH	Net Positive System Head
OMS	Orbital Manuevering System
OMV	Orbital Maneuvering Vehicle
ORU	Orbital Replacement Unit
OTV	Orbit Transfer Vehicle
P&W	Pratt & Whitney
P/L	Payload
PAM	Payload Assist Module
PCDA	Power Control & Distribution Assembly
PDR	Preliminary Design Review
PHASE C/D	Development & Operations Phases
PHI	Pump Head Idle
PIDA	Payload Installation Deployment Aid
PIP	Payload Integration Plan
PSF	Pounds/Ft^2

psia	Pounds per Square Inch Absolute
PU	Propellant Utilization
PV	Present Value
q	Heating Rate (BTU/FT^2 Sec)
q	Dynamic Pressure (Lbs/Ft^2)
Q/D	Quick Disconnect
RAM	Random Access Memory
RCS	Reaction Control System
R/D	Research and Development
REM	Rocket Engine Module
RF	Radio Frequency
RFI	Radio Frequency Interference
RLG	Ring Laser Gyro
RMS	Remote Manipulator System
ROI	Return on Investment
ROM	Read Only Memory
RSI	Reusable Surface Insulation
RSS	Root Sum Square
RTV Sealer	Room Temperature Vulcanizing Sealer
S&M	Servicing and Maintenance
SB	Space Based
SBM	Space Based Manned
SBU	Space Based Unmanned
S/C	Spacecraft
SOFI	Spray on Foam Insulator
STA	Static Test Article
STS	Space Transportation System
T&C	Telemetry and Command
TCS	Thermal Control Subsystem
THI	Tank Head Idle
Ti	Titanium
TLM	Telemetry
TPA	Turbo-Pump Assembly
TPS	Thermal Protection System
TVS	Thermodynamic Vent System
VDC	Volts-Direct Current
W	Weight
WA	Watt of Aerobrake
WBS	Work Breakdown Structure
W_{Dry}	Stage Dry Weight
WLS	Western Launch Site