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Advanced Orbit Transfer Vehicle Propulsion System Study

Final Report

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PREFACE

The purpose of this report is to document the results of the Advanced Orbit Transfer Vehicle Propulsion System Study performed for NASA Lewis Research Center under Contract NAS3-23858. Dr. Larry Cooper of NASA Lewis Research Center and Mr. James Brown of Pratt and Whitney Aircraft provided technical direction to the study. The Martin Marietta Aerospace Program Manager, Mr. Larry Redd, was directly supported by the following personnel:

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Abbreviations

ACC	Aft Cargo Carrier
ACS	Attitude Control System
AI	Artificial Intelligence
APU	Auxiliary Power Unit
Btu	British Thermal Units
cg	Center of Gravity
CCTV	Closed-Circuit Television
DDT&E	Design, Development, Test and Evaluation
DOD	Department of Defense
deg	Degree
dia	Diameter
EMU	Extravehicular Maneuvering Unit
EPS	Electric Power System
EVA	Extravehicular Activity
elec	Electrical
FMEA	Failure Modes and Effects Analysis
fps	Feet per Second
ft	Feet
g	Gravity
GEO	Geosynchronous Equatorial Orbit
GG	Gas Generator
GH_2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
GPS	Global Positioning Satellite
GRD	Ground
Guid/Nav	Guidance and Navigation
H_2	Hydrogen
He	Helium
hp	Horse Power
h	Hour
I&CO	Inspection & Checkout
I/F	Interface
IOC	Initial Operating Capability
IUS	Inertial Upper Stage
IVA	Intravehicular Activity
I_{sp}	Specific Impulse
k	Thousand
L/D	Lift to Drag
LCC	Life-Cycle Cost
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOS	Line of Sight
LRU	Line-Replaceable Unit
lbf	Pounds force
lbm	Pounds Mass
MES	Main Engine Start
MLI	Multilayer Insulation
MMS	Multimission Modular Spacecraft
MMU	Manned Maneuvering Unit

Abbreviations (continued)

MPS	Main Propulsion System
MR	Mixture Ratio
MSFC	Marshall Space Flight Center
mech	Mechanical
min	Minutes
N_2H_2	Hydrazine
NASA	National Aeronautics and Space Administration
NDI	Nondestructive Inspection
NIFR	Nonindependent Failure Rate
nmi	Nautical Mile
O_2	Oxygen
O&S	Operations and Support
OMV	Orbital Maneuvering Vehicle
ORU	Orbit-Replaceable Unit
OTV	Orbit Transfer Vehicle
Ops	Operations
P&W	Pratt and Whitney Aircraft
PU	Propellant Utilization
P/L	Payload
PAM	Payload Assist Module
PIM	Pumped Idle Mode
psf	Pounds per Square Foot
psi	Pounds per Square Inch
QDs	Quick Disconnects
R/R	Remove and Replace
RCS	Reaction Control System
RF	Radio Frequency
RMS	Remote Manipulator System
RN	Retractable Nozzle(s)
Rel	Reliability
SE&I	Systems Engineering and Integration
SEE	Special End-Effector
SN	Standard (Fixed) Nozzle(s)
SS	Space Station
STS	Space Transportation System
s	Seconds
TDM	Technology Development Mission
TDRSS	Tracking and Data Relay Satellite System
THFS	Teleoperator Human Factors Study
THIM	Tank Head Idle Mode
TVS	Thermodynamic Vent System
°F	Degrees Fahrenheit
°R	Degrees Rankine

1.0 SUMMARY

Future NASA and DOD missions will benefit from high-performance reusable orbit transfer vehicles. With the advent of a space station, advanced engine technology, and various new vehicle concepts, reusable orbit transfer vehicles that provide significant economic benefits and mission capability improvements will be realized. Engine and vehicle design criteria have previously lacked definition with regard to such issues as space basing and servicing, manrating and reliability, performance, mission flexibility, and life-cycle cost for a reusable vehicle.

This design study has resulted in the definition of a reusable orbit transfer vehicle concept and subsequent recommendations for the design criteria of an advanced LO_2/LH_2 engine. These design criteria include number of engines per vehicle, thrust level(s) per engine, mixture ratio, nozzle design, etc. The major characteristics of the vehicle preliminary design include a low lift-to-drag aerocapture capability, main propulsion system failure criteria of fail operational/fail safe, and either two main engines with an attitude control system for backup or three main engines to meet these failure criteria. A maintenance and servicing approach has also been established for the advanced vehicle and engine concepts. Design tradeoff study conclusions were based on the consideration of reliability, performance, life-cycle cost, and mission flexibility.

This program was accomplished in two phases. In Phase I tradeoff studies and analyses were conducted to define a baseline LO_2/LH_2 propulsion system at the major component level based on the requirements of mission scenarios for future NASA and DOD space architectures. Major engine assembly characteristics (number, nozzle design, thrust, mixture ratio, throttling range, etc) were determined using engine contractor parametric performance data.

In Phase II subsystem definition, servicing operational procedure description, reliability, and life-cycle cost and sensitivity analyses were performed to assess the design optimization and space basing impacts on the baseline propulsion system established in Phase I. The sensitivity studies provided overall system design criteria and the rationale for choice of engine number, component and subsystem development recommendations, servicing/replacement methods, and recommendations for manrating criteria.

2.0 INTRODUCTION

The design requirements for a space-based, reusable, manrateable orbit transfer vehicle (OTV) are substantially different from the requirements typical of a ground-based expendable upper stage. The mission may still require a delivery capability to geosynchronous equatorial orbit (GEO) of a large payload. However, retrieval of payloads from GEO may be required in addition to returning the vehicle itself so it may be reused. If astronauts are on board, the vehicle must obviously be manrated for that application.

The geometry constraints of ground launch for an upper stage can be relaxed for a space-based OTV. But, depending on the mode of return from a high orbit, geometric concerns are still an issue if the aeroheating effects during an aerobraking maneuver are considered, for example. Because the engine design requirements must be compatible with the operation of an advanced OTV and a suitable approach to servicing and maintaining the vehicle is also essential, we must understand the issues surrounding the design requirements of a space-based reusable OTV and appropriately resolve them.

Previous OTV concept studies have investigated reusable propulsion systems associated with aeroassisted vehicle concepts with moderate and low lift-to-drag (L/D) ratios. There has also been interest in a vehicle that can be packaged into an aft cargo carrier (ACC) on the space shuttle orbiter external tank for delivery to orbit. The trend in overall vehicle evolution is to move from ground-based expendable vehicles to space-based reusable and manned vehicles.

This study focused on the "ultimate" vehicle that would be space-based, reusable, aeroassisted, and manrateable. No attempt has been made to show an evolution from ground basing to space basing (however, initial delivery to the space station was considered). Before proceeding into the conceptual design study, the following previous related studies were reviewed.

Vehicle:

- OTV Concept Definition Studies, General Dynamics Contract NAS8-33533 and Boeing Contract NAS8-33532.
- Future OTV Technology Study, Boeing Contract NAS1-16088.
- In-house activity on ACC OTV, Martin Marietta.

Space Station and Servicing - OTV Servicing Study, General Dynamics Contract NAS8-35039.

- Definition of Technology Development Missions for Early Space Station, Martin Marietta Contract NAS8-35042.

- Space Station Needs, Attributes, and Architectural Options, Martin Marietta Contract NASW-3686.

<u>Aeroassist</u> - System Technology Analysis of Aeroassisted OTVs for Moderate and Low L/D, General Electric Contract NAS8-35096, and Boeing Contract NAS8-35095.

- Technology Identification for Aero-Configured OTVs, Boeing Contract F33615-82-C-3014.

3.0 PHASE I - BASELINE OTV CONCEPT DEFINITION

This section describes the mission ground rules and requirements, system requirements and baseline vehicle/engine identification. The results of vehicle and engine performance and geometry tradeoff studies are also shown.

3.1 GROUND RULES AND ASSUMPTIONS DEVELOPMENT

3.1.1 Mission Model Assessment

Space missions through the year 2000 include the delivery of many large payloads, require manned presence, and will significantly benefit from the use of a reusable advanced orbit transfer vehicle. The MSFC nominal mission model PS-01 (Rev 6) for the period from 1994 through 2000 has been used in this study. The years 1994-2000 were used because 1994 is the earliest projected initial operational capability (IOC) of a space-based OTV. Two of the mission types involving large payloads in the 1994-2000 time frame include:

1) Delivery of 5000 lbm to geosynchronous orbit (GEO) and return of 2000 lbm (8 total missions);

2) Manned sortie with delivery of 13,000 lbm to geosynchronous orbit and return of 13,000 lbm (7 total missions).

The remaining of the 47 LEO/GEO missions concern payload delivery (up to 16,000 lbm) to GEO only.

These large payload missions will require a propulsion system with high total energy capabilities. Other requirements include that the vehicle be manrateable. The space station may provide a suitable location for basing, servicing, refueling, and launching a reusable OTV. The following discussion presents arguments for configuring such a vehicle.

3.1.2 Recommended Vehicle Characteristics

The dry weight of a propulsion system can be expressed by a constant value of hardware weight (independent of total system size) plus a weight of hardware that depends on total propellant loaded. This is expressed as:

$$M_{dry} = A + B\ (M_{prop})$$

where :

M_{dry} = vehicle dry mass
A = constant hardware mass
B = variable hardware factor
M_{prop} = loaded propellant mass

Figure 3.1-1 shows the characteristics of single-stage liquid propulsion vehicles and their limitations. The x-axis is the nondimensional parameter of delta V divided by specific impulse (I_{sp}). The y-axis is the ratio of vehicle and payload initial mass (including propellant) divided by payload mass and vehicle constant hardware mass (A).

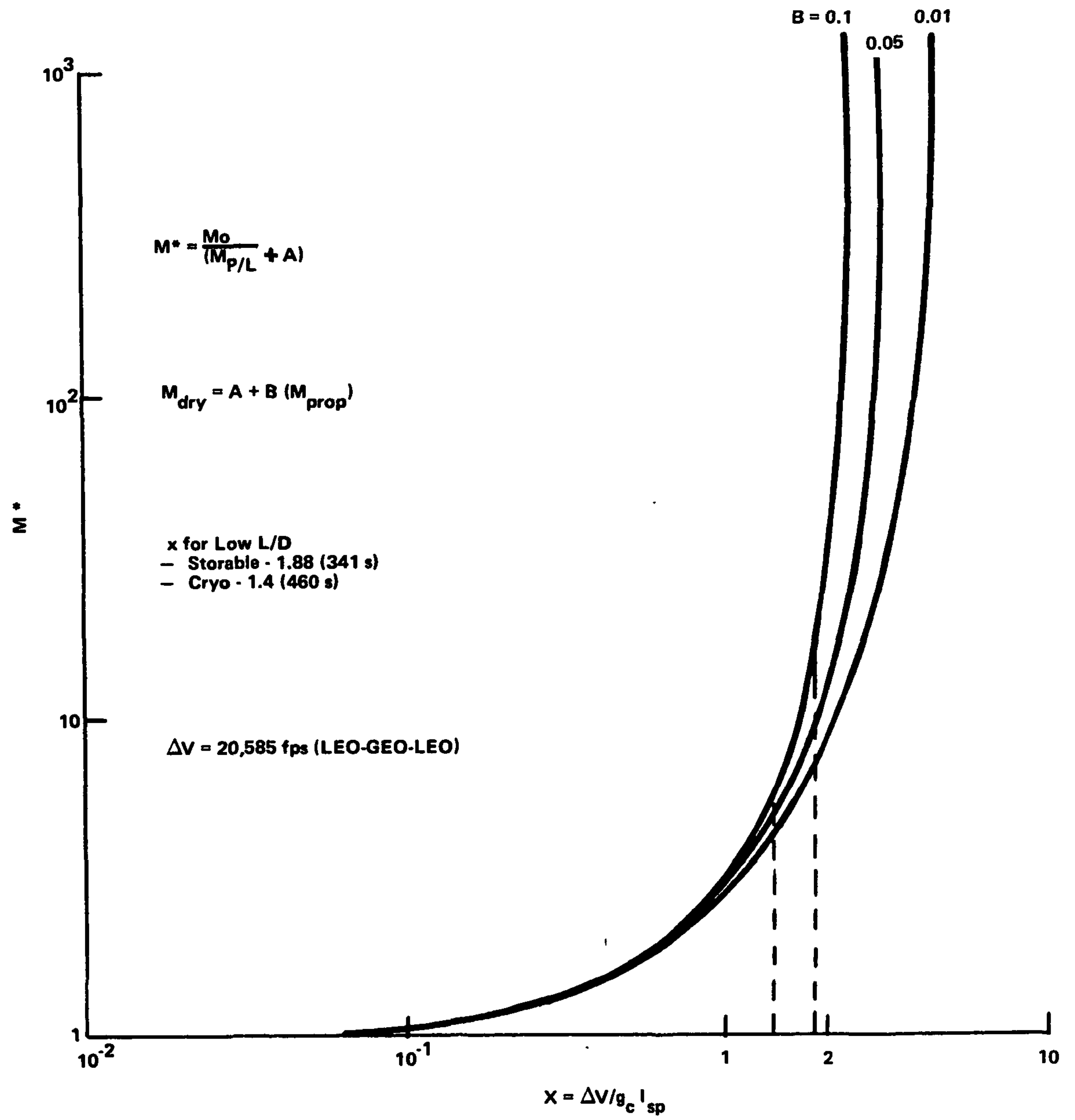

Figure 3.1-1 *OTV Parameters*

Several conclusions may be drawn from the figure. The first is that for a large enough delta V, there is an asymptotic value for a given propellant combination and engine technology (I_{sp}) for which no amount of propellant can deliver any size payload. Therefore, to complete a mission that imposes a delta V above the asymptotic value, either a higher energy propellant must be chosen or the required delta V must be reduced (as with an aerocapture maneuver of some sort). For a low-earth orbit (LEO) to geosynchronous equatorial orbit (GEO) payload delivery with an aerocapture return, the required delta V is about 20,000 fps for a vehicle with a lift to drag ratio (L/D) of zero. Also, a reasonable estimate for B for large propulsion systems is about 0.1 for storable propellants (N_2O_4/MMH) and about 0.125 for LO_2/LH_2 (ref Magnetoplasmadynamic Thruster Definition Study performed for AFRPL, Contract F04611-82-C-0049). So, for a storable propellant combination, the delta V of 20,000 fps places the value of x very near the asymptotic value for total vehicle weight. In other words, small savings in delta V and/or increases in I_{sp} result in very great savings in total vehicle mass for a given payload mass. For example, providing a vehicle with a L/D of 1.0 will result in a possible inclination change capability on LEO to GEO return of 14 deg and a delta V savings of only about 800 fps. This, however, is valuable in terms of reducing storable propellant requirements and total size of the resulting vehicle even with a subsequent increase in dry weight because of the higher L/D ratio. Therefore the higher L/D approach appears suitable for a vehicle with a storable propellant combination.

A LO_2/LH_2 propulsion system with an aerobrake and a L/D of zero provides a delta V savings of about 7500 fps when returning from GEO over an all-propulsive system and results in an x value that is further away from the asymptotic value for total vehicle mass. Therefore the reduction in total vehicle mass is not as dramatic for reductions in delta V by going to L/D ratios greater than zero as for the storable propellant system. This is a result of being on a "flatter" part of the curve relating x and total system mass. For instance, in providing a 800 fps reduction in delta V during return from GEO (via L/D of 1.0), the resulting LO_2/LH_2 total propulsion system mass shows only a reduction of about 5%. This also assumes no increase in dry weight as a result of providing the vehicle with a higher L/D capability. Therefore a mid to high L/D configuration is not recommended for a LO_2/LH_2 vehicle because an inclination change capability does not appear to benefit the system to a great degree. The concept of a low L/D reusable aerobrake, however, is consistent with materials technology evolution for the time frame of interest and the resulting vehicle configuration may be more amenable to component accessibility and servicing.

Figure 3.1-2 shows the relationship between total vehicle system mass (propulsion system dry mass plus propellant) and specific impulse. One can see that in going from an all-propulsive system to one with aerobraking (L/D = 0) the aerobraked system is less sensitive to specific impulse (I_{sp}) in the range applicable to LO_2/LH_2 propellants. Detailed vehicle studies assessing the differences in vehicle dry mass for low vs high L/D ratios would be required before determining the benefits of additional delta V savings.

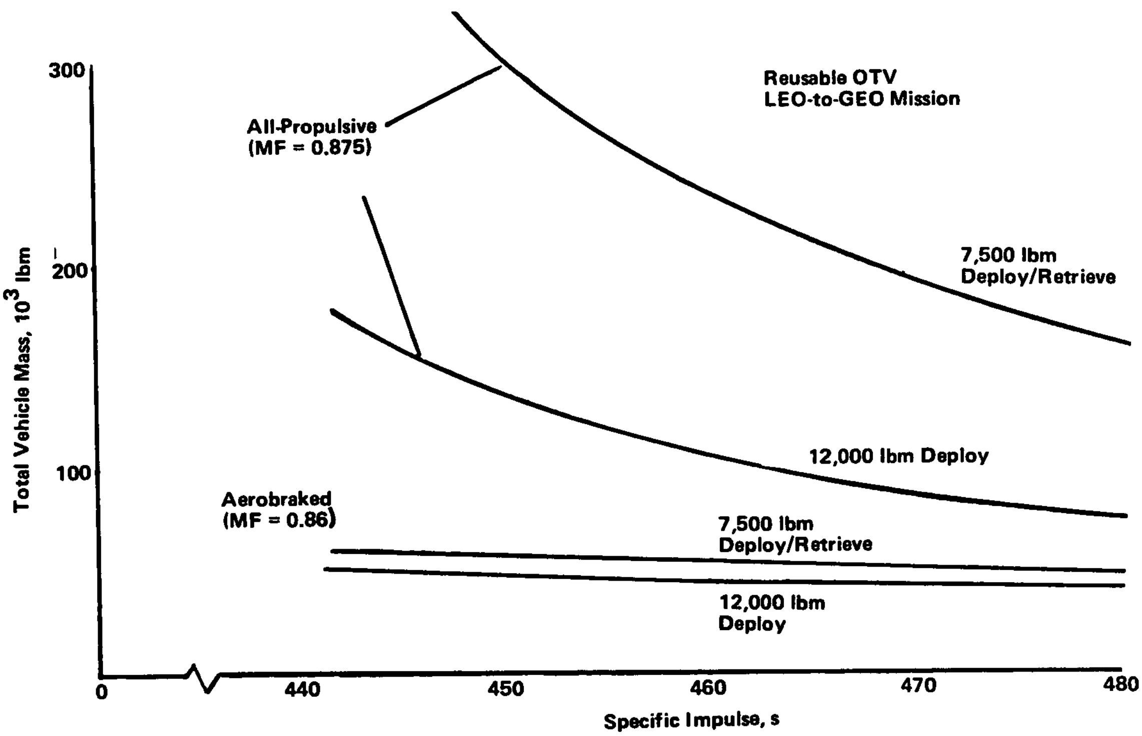

Figure 3.1-2 *Vehicle System Mass vs Specific Impulse*

3.1.3 Propellant Requirements for Rev 6 Mission Model

In addition to the large payloads in the Rev 6 mission model that require a high-energy OTV, numerous missions require much less energy for each payload. This results in a wide range of payload weights and propellant requirements for all missions. A single reusable space-based OTV design for performing all the Rev 6 missions requiring orbit transfer would substantially reduce launch costs because of eliminating the launch of the upper stages for these small payloads. However, a single-stage design is not feasible for performing single payload delivery missions for such a wide range of payload masses.

Figure 3.1-3 (ref Magnetoplasmadynamic Thruster Definition Study, AFRPL Contract F04611-82-C-0049) shows the trend in cost for putting various sizes of payloads into GEO. The two curves show the differences in payload delivery cost depending on whether propellant (LO_2/LH_2) would be available from scavenging external tank residual and surplus propellant and reducing launch costs accordingly. The main conclusion from the figure is that the larger the payload and subsequent delivery vehicle (OTV) the greater the cost savings will be. Therefore, fewer, larger payloads to GEO would be desirable over many small payloads and a few large ones.

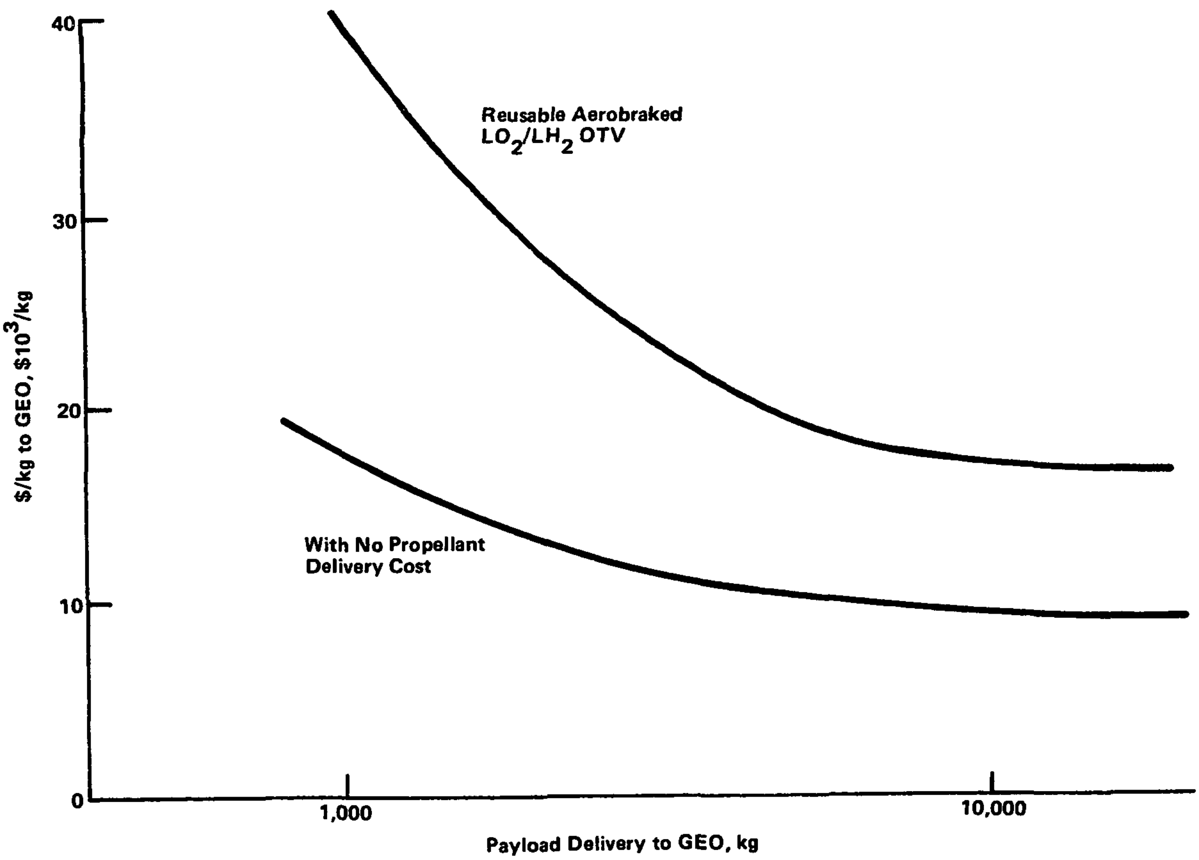

Figure . 3.1-3 *Costs for GEO Payloads*

The small payloads in the Rev 6 mission model (PAM, IUS, and LEASAT class) may lend themselves to grouping for delivery. Distinct groups that are intended to fly in the same month could possibly be grouped and delivered to GEO with the same OTV flight. This would also narrow the range of total propellant requirements over which to design an appropriate OTV capable of performing the Rev 6 missions. Figure 3.1-4 shows the propellant requirements for all (except three small PAM-delivered payloads that could not be grouped by month) of the Rev 6 LEO to GEO missions from 1994 to 2000. The darkened bars represent the large single payload deliveries by an OTV. The hollow bars are the propellant requirements for the "grouped" payload missions. The ground rules for logical grouping of these payloads are as follows:

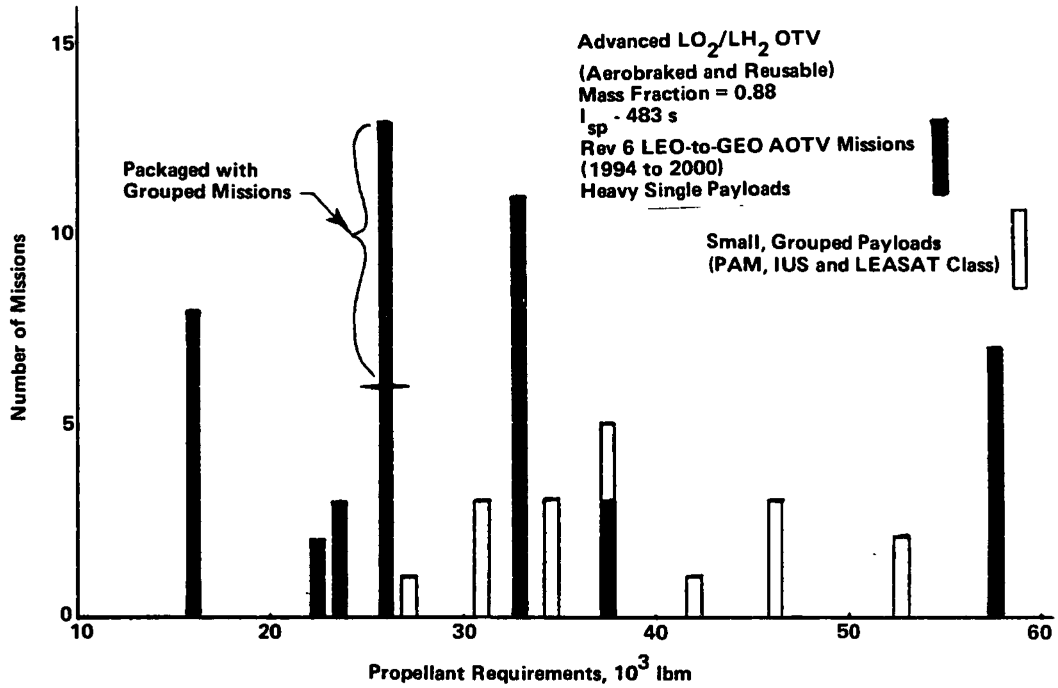

Figure 3.1-4 *Spectrum of OTV Propellant Requirements*

1) All non-DOD;

2) LEO-to-GEO delivery only;

3) Fly during same month.

The groupings of the smaller payloads and subsequent packaged delivery to GEO results in propellant requirements in the same range as those associated with the single large payload delivery missions. However, the wide spectrum of propellant requirements for the single large payloads and for the packaged groups leads to the need for multiple sizes for OTVs, staging, drop tanks, or else a totally modular approach to the design of an OTV.

In comparing these propellant requirements to estimates of propellant available from scavenging residual and surplus space shuttle external tank propellants, it appears that a large percentage of these propellants could be available. For instance, if the total propellant requirement for 47 LEO to GEO missions between 1994 and 2000 were from 1.5 to 2.0 million lbm, as much as two thirds of this propellant could be available from scavenging (Ref STS Propellant Scavenging Systems Study, NAS8-35614). The result here is that for the high-energy OTV missions, the average propellant delivery cost may be significantly reduced by taking advantage of scavengeable propellant.

Based on review of previous study conclusions, the mission model, and appropriate vehicle characteristics, the ground rules and assumptions to be followed during the baseline vehicle development of this study were:

1) Space-based, reusable LO_2/LH_2 vehicle;

2) Space station-berthed and -serviced;

3) Single STS delivery (deliver empty);

4) Aerobraked return (low lift to drag ratio);

5) Manrateable;

6) Able to capture Rev 6 missions (LEO to GEO).

3.2 MISSION DESCRIPTION AND MISSION REQUIREMENTS

3.2.1 Mission Description

The reference mission (Table 3.2-1) is a single payload delivered to GEO and vehicle return empty with an aerocapture maneuver at LEO. The aerobrake maneuver will nominally be a single pass. Variations of this mission will include up to three passes, manned servicing mission to GEO, and unmanned servicing at GEO. Timelines for the mission are shown in the table. Figure 3.2-1 shows the trajectory of the delivery portion and Figure 3.2-2 shows the aerobrake return. For the purpose of system sizing, 325 fps are included as gravity losses and 50 fps contingency for aerospike operation. Additionally 500 fps is included in the perigee raising and phasing scenario to adjust the altitude and inclination of the orbit following atmospheric exit. The velocity taken out by the brake was derived from work done at Martin Marietta on an aft cargo carrier OTV.

The timelines were derived assuming a single shift on the space station is required to load cryogenic propellants and launch the OTV. A Hohmann transfer is used, and the return flight occurs after the second nodal crossing allowing for one day at GEO for operations. The OTV navigation and guidance is updated three times before reentry. The OTV is then safed after achieving a proper phasing orbit before the ACS is used to rendezvous with the space station/OMV. Communications with the space station is used during launch and rendezvous, and communications with the ground through the TDRSS below approximately 6000 nmi is used for actual control of the OTV operations.

Table 3.2-1 Mission Timeline for Reference Mission

Mission Operation	Time, h min from MES-1	System Driver	System Impact	Remarks
Launch Transfer to Loading Area	- 7 30	Launch in one shift after loading cryogens.		
Chill Down and Load LH_2/LO_2	- 7:15	Loading accuracy impact on margins.	Zero-g gage on space station.	
Terminate and Leak Check	- 3:15			
Disconnect Lines	- 3·00	Valve redundancy.	Seals.	Purge lines between valves.
Perform Final Prelaunch Operations	- 2.15	No venting near space station.	Space station and TDRSS links.	Communication links, verify systems "go"
OTV Payload Move from Space Station	- 1 15		ACS sizing.	Phasing orbit for targeting during proximity operations.
Activate Main Engine	- 30	10-mi separation.	Go into tank head idle mode (THIM).	Partial chilldown and loading
Navigation Update Position for Burn	- ·20		ACS sizing.	Ground Monitor
GEO Transfer MES-1 (1st main engine start)			ACS control.	Burn time = 17 min, 7820 fps.
Transfer, Coast and Thermal Rolls		Temperature of insulation and components.	ACS sizing.	0.1 deg/s to 0.5 deg/s.
Navigation Update	3.00	TDRSS below 6000 nmi; NASA 85 ground station above 10,000 nmi.	Attitude adjustment for GPS link acquisition	Ground control monitor OTV
Midcourse	3 15		ACS sizing.	Delta V = 30 to 40 fps.
Position for Burn	5:00		ACS.	
GEO Injection MES-2	5.15			Burn time = 8 min; delta V = 5840 fps.
Spacecraft Deployment Checkout Operations	6:00			Communication links, test.
OTV/Payload Separation	6·15	Control of 50% full OTV.	Collision avoidance—ACS.	
GEO Operations, Thermal Rolls	27:45	Temperature of MLI, components.	ACS sizing nodal crossing every 12 h.	Will use 2nd (24-h) nodal crossing.
Navigation Update	28.50		Ground monitor NASA 85 ground station.	
Position for Burn	29:05		ACS sizing.	
OTV Deorbit Burn (MES-3)	29.20	Aerobrake perigee 45 nmi.		Delta V = 6050 fps; burn time = 2 min.
Guidance/Navigation Update	32·00			
Midcourse No. 1	32·15		ACS.	Delta V = 30 to 40 fps.
Guidance/Navigation Update	34·00			
Midcourse No. 2	34 15		ACS.	Delta V = 30 to 40 fps.
Navigation Update (MES-4)	34.50	Atmospheric variations.	Start engine THIM.	
Atmosphere Entry	35:00	Base heating aerospike.	Structure, TPS, engine throttling, ACS control.	Delta V = 7800 fps, Viking shell, low L/D.
Atmosphere Exit	35.05			Delta V = 50 fps for aerospike
Apogee Boost No. 1 (MES-5)	35:40	Perigee 4 nmi, aerobrake and manrating.	Redundance, space station rendezvous.	Delta V = 250 fps, phasing orbit.
Navigation Update	36:55			
Apogee Boost No. 2 (MES-6)	37:10	Rendezvous with space station orbit.	80-fps allowance for inclin-ation adjustment.	Delta V = 250 fps, circularize and adjust inclination.
Safe OTV	41·00		Backup fuel cells.	4-h allowance.
ACS Moves OTV to Station	41·30	Safe operation, ACS near space station.	ACS sizing.	
Space Station Grapples and Berths OTV OTV	42:00	Safe ACS system.	ACS system design.	

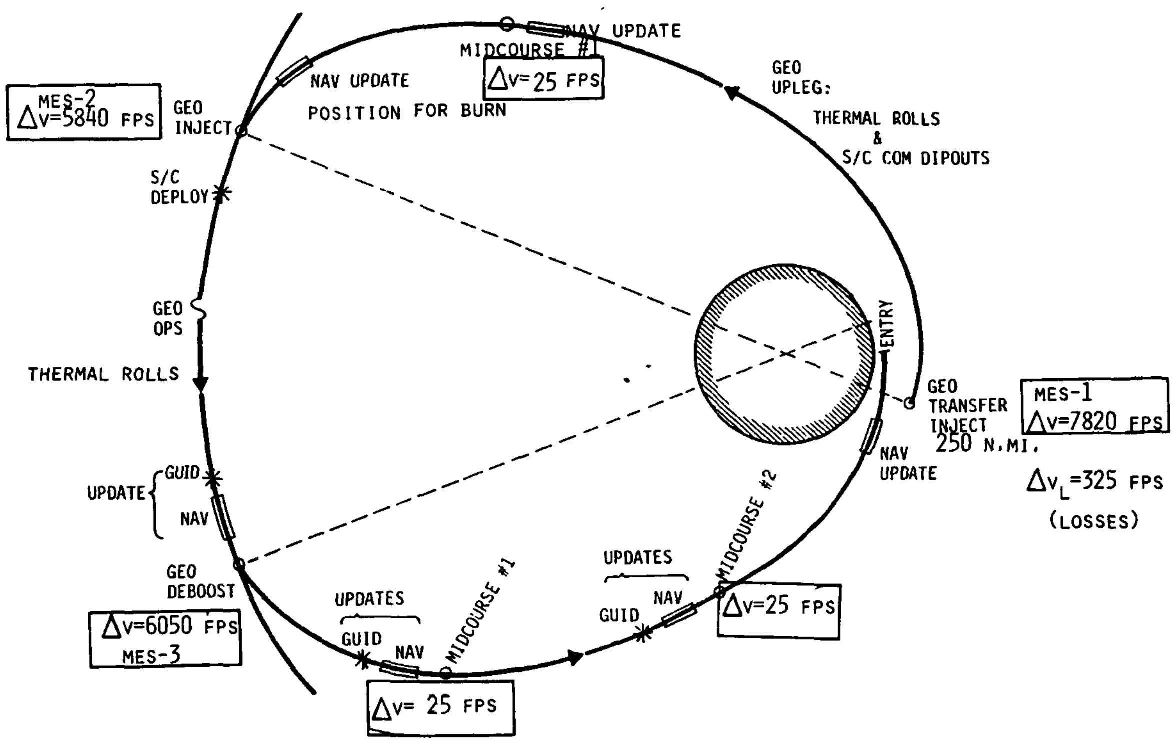

Figure 3.2-1 OTV Trajectory Plot (Delivery)

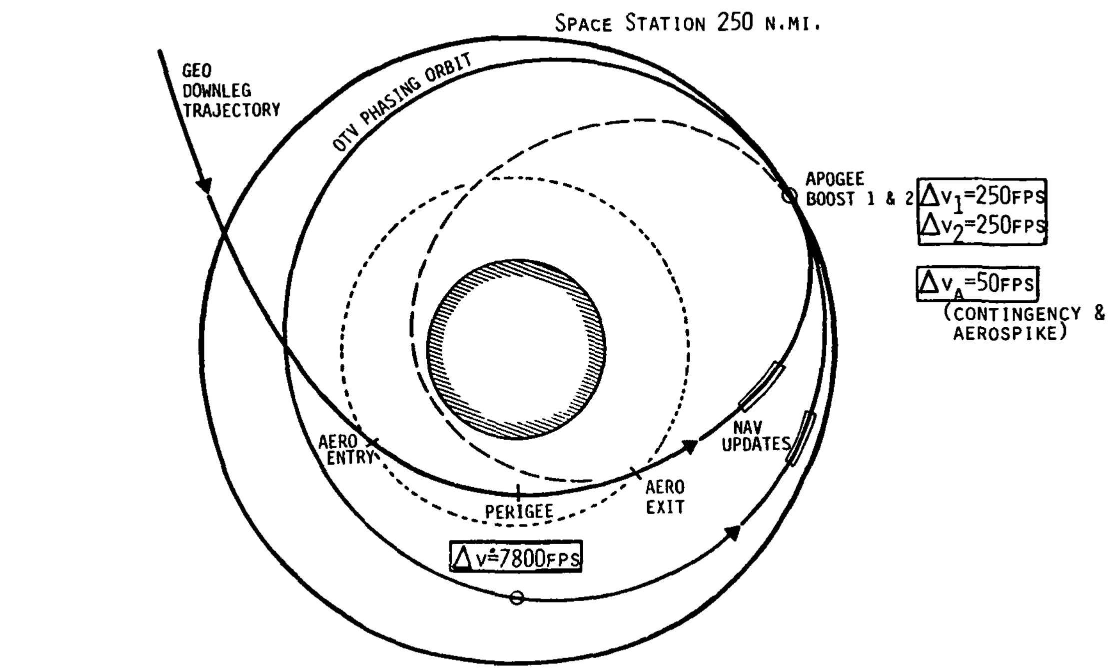

Figure 3.2-2 OTV Trajectory Plot (Aerobrake)

3.2.2 Mission Requirements

Table 3.2-2 shows the mission requirements. These were derived from the reference mission scenario. A maximum time of two weeks at GEO was selected for the manned mission to be compatible with one month between missions and two-week servicing at the space station.

Table 3.2-2 Mission Requirements

Mission Operation	Requirement	Remarks
Delta V (Main Engine)	13,985 fps up, 6,600 fps down.	250-nmi space station; aerobrake = 7800 fps.
Payload Range	16k lbm deliver to GEO. 13k lbm deliver/return (GEO).	Rev 6 September 1982 13k manned sortie.
No. of Burns	6 nominal.	Single perigee and aerospike return.
Throttling	During aerospike or large space system.	TBD.
Maximum Time at GEO	2 weeks.	Manned servicing—one month turnaround (2-week servicing).
Safety—Manned Flight, Space Station	Fail operational/fail safe Shuttle safety.	ICD-2-19001 and NHB 1700.7A.
Design Life	30 missions.	TBD Phase II.
Time Between Overhauls	10 hours.	Engine life paces overhaul.
Aerobrake	Yes (low L/D), with aerospike.	Use Viking configuration 70-deg half-cone (C_D = 1.6).
No. of Aero Passes	Up to 3 (1 nominal).	Minimize transfer time, reduce heating to brake and stage.
Acceleration	3.2 g (maximum).	Shuttle ascent.
Onorbit Thermal Environment	Minimize boiloff.	Thermal roll, 0.1 deg/s to 0.5 deg/s.
Guidance/Navigation	GPS and Ground (and TDRSS) tracking communications.	No control from space station. GPS state vector.
IOC	1994 to 1997.	Space station IOC (advanced engine IOC).
Mission Time	Compatible with 2-week servicing period at space station.	One month between launch.
Initial (T/W)	0.15 (minimum), 0.22 (nominal).	Minimize delta V for single perigee burn.

Near the space station, orbiter safety outlined in NHB 1700.7A was used. Manned missions were assumed to be consistent with fail operational/fail safe. At least in the early stages of this study program, fail operational/fail safe was adopted as a "starting point" in developing an appropriate vehicle design philosophy. This allows the astronauts the option to return after one failure, and the ability to return to the space station, STS, OMV retrieval, etc after sustaining two failures. A 30-mission life is compatible with two OTVs and approximately 60 missions. A 3.2 maximum g-level was selected to be compatible with a shuttle launch. However, the OTV will probably be delivered to the space station empty. Therefore, the loads during orbit transfer (main engine burn and/or the aerobraking loads) will probably be the structural design drivers.

3.3 SYSTEM REQUIREMENTS AND BASELINE SUBSYSTEM SELECTION

3.3.1 System Requirements

Table 3.3-1 shows the OTV requirements for the various vehicle systems. These were derived from the mission requirements, study ground rules, and the statement of work.

Table 3.3-1 System Requirements

Aerobrake Requirements
- Withstand 2500°F and 15 psf
- Compatible with Space Station Servicing
- Reusable
- Compatible with Manned Mission Return

Engine Requirements
- Provide Impulse for GEO Transfer and Return
- Provide Aerospike Capability (TBD)
- Operate At a Mixture Ratio Between 5 and 7
- Reusable
- Provide Main Propellant Tank Pressurization
- Use LH_2/LO_2 Propellants
- Throttlable (TBD)

ACS Requirements
- Provide Positioning and Alignment Capability of OTV During All Mission Phases
- Provide Translation Near Space Station
- Perform Midcourse and Final Rendezvous Maneuver
- Safe Operation Near Space Station
- 10% Margin
- Reusable
- Space-Based

Main Propulsion Requirements
- Provide Subcooled LH_2/LO_2 to the Main Engine System (Engine NPSH Requirements)
- Provide GH_2/GO_2 to Fuel Cell System (or TBD)
- Provide a Fill/Drain System Compatible with Space Station
- No-Vent Operation Near Space Station Except When Provision for Collecting Vapor Is Available
- Provide for Safe System Shutdown 20 mi from Station (or TBD)
- Manrated and STS Safety (Fail Operational/Fail Safe)
- Propellant Utilization (PU) System

Avionics
- Provide Following Capabilities on Orbit
  - Guidance, Navigation and Control Compatible with GPS
  - Telemetry, Tracking and Communications Compatible with TDRSS, Ground Station and Space Station
  - Data Management
  - Safety Status and Control
  - Subsystem Status Indication
  - System Checkout
- Generate, Control and Distribute Electric Power to All OTV Systems
- Minimize Single-Point Failures of Safety- and/or Mission-Critical Functions
  - Provide Safety-Critical Data to Orbiter

Structure
- Ultimate Factor of Safety = 1.40 or Greater
- Pressure Vessels MIL-STD-1522 (Factor of Safety 2.0 on Ultimate)

Thermal Protection
- Provide Protection for Aerobrake
- Provide Protection on Base Heating of OTV
- Provide Thermal Protection of Cryogenic Tanks on Orbit
- Provide Protection of Components, Lines, Avionics, ACS, and Engine on Orbit

3.3.2 Baseline Subsystem Selection

This study was intended to consider the issues of space basing and engine operational design criteria for a reusable OTV. Most of the vehicle subsystems were identified and fixed in order to proceed with the key issues and tradeoff studies of importance. Previous study results were used in subsystem selection. The fixed subsystems and the rationale for their selection follows.

Aerobrake - The baseline aerobrake configuration has the following characteristics:

Shape	Viking type, 70-deg half-cone angle.
C_D	1.6 without aerospike (highest drag loads).
Aerobrake Surface Emissivity, e	0.8
Maximum Temperature (Nextel Synthetic Fabric)	2500°F.
Maximum Loading	15 psf (approx 3.0-g loading).
Maximum Stage Temperature (e = 0.5)	750°F.

Past work at Martin Marietta (Ref 1 and 2) was used to derive reasonable diameter and stage lengths considering the above characteristics and convective heating. The effects of an aerospike were not considered. From Reference 1, the sensitivity of temperature to reentry weight and diameter were considered. Reference 3 indicates that W/C_DA should be constant for a given stagnation point heating. However, given the state of aerobrake analysis for an OTV, further detail was not attempted. With information from References 1 through 3, Figure 3.3-1 was prepared to relate ballistic coefficient and stage/aerobrake geometry. Additional thermal protection could be required or multiple passes could be used to reduce heating and brake diameter. The data in Reference 3 indicate that the stagnation point heat flux can be reduced about 30% with two passes, and up to 50% for three passes.

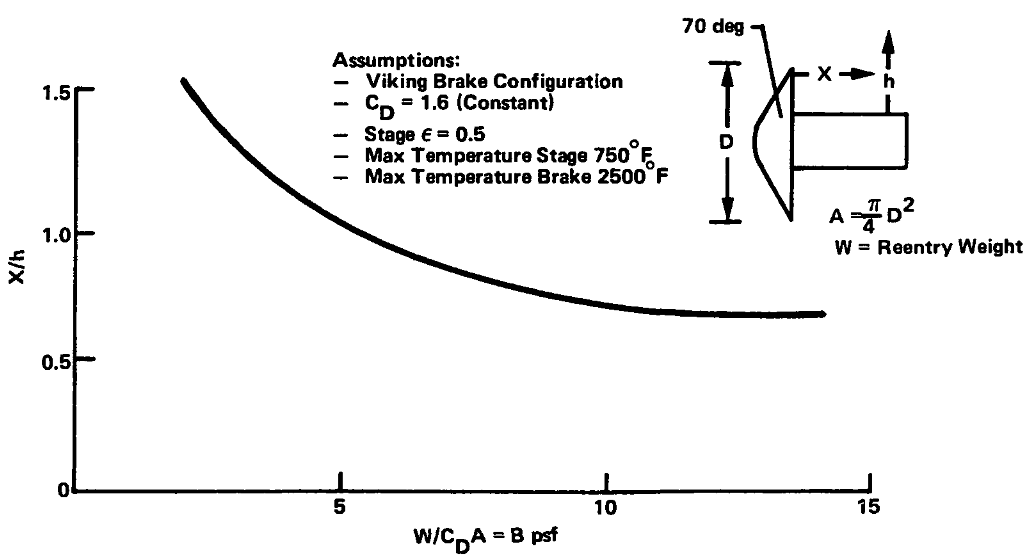

Figure 3.3-1 Aerobrake Design Parametrics

ACS - A GO_2/GH_2 ACS was selected because of ease of servicing and refueling. Commonality with space station attitude control propellant would be a desirable feature for the OTV. Contamination control would make a GO_2/GH_2 ACS attractive for OTV instead of a monopropellant or bipropellant system, especially if the OTV is to fly itself into berth at the space station. Also, as a possible backup for the main propulsion system, the high performance as well as the commonality of propellants make GO_2/GH_2 a logical choice. Sixteen thrusters were chosen for the full three-axis control capability and limited redundancy.

Avionics - A system compatible with the GPS, TDRSS, and space station was baselined. The OTV should be capable of rendezvousing with space station, performing targeting for the aerobraking and phasing maneuvers, and performing midcourse corrections. The electronics on board the OTV will also need to handle the diagnostics information to be recorded or transmitted to the ground for postflight or inflight checkout of the subsystems requiring health monitoring.

Electric Power - The major power source for the baseline OTV was fuel cells with feed from the GO_2/GH_2 accumulator system. Small batteries used as a backup of the fuel cell power system and for use during times when the main propellant tanks are empty were baselined.

Gaging and Propellant Utilization - A point sensor system of gaging during flight operation to monitor the level of propellant in the tanks was baselined. During refill of propellant in low-g at the space station, a flowmeter or other suitable gaging device is recommended to monitor the level of propellants.

A propellant utilization system to minimize the propellant outage was baselined for the OTV design. For a multiple-tank configuration, the outage between tanks must be minimized and the mixture ratio at the engine must be controlled for proper outflow of oxidizer and fuel. Centaur data, space tug studies, and later OTV studies provided a rationale for the performance advantages a propellant utilization system offers a cryogenic vehicle.

Pressurization - An autogenous pressurization system for a LO_2/LH_2 OTV would provide a simple, lightweight means of pressurizing the main propellant tanks with little or no contamination. Therefore, autogenous pressurization was chosen for the study baseline vehicle.

Thermal - Based on the results of a low-thrust chemical orbit-to-orbit propulsion system propellant management study (Contract NAS3-21954); conceptual design and analysis of orbital cryogenic liquid storage and supply systems, (NAS3-22264); and an IR&D D-24S project concerning advanced propulsion concepts (a study performed at Martin Marietta) a multilayer insulation (MLI) thickness of 1/2 in was chosen for the baseline vehicle main propellant tanks. The previous studies show that a thicker insulation is desirable for vehicles that experience boiloff before launch and during ascent from the ground. However, because the baseline vehicle of this study was intended for space operations alone, less than 1 in. of MLI is optimum from a total vehicle performance standpoint.

Venting - A thermodynamic vent system (TVS) with an active component (small pumping unit) may be the lowest weight method of venting for a large cryogenic vehicle as opposed to a passive TVS that which would require a relatively heavy and sizable heat exchanger. However, the active TVS presents a servicing concern for access to the pump unit for inspection and maintenance. The alternative to the two TVS methods mentioned is to simply settle the propellants with the ACS and vent vapor from the tanks.

For this study baseline vehicle, a weight representative of a TVS with pumping units in each tank was defined. Servicing of the unit is addressed later in this report, noting possible modifications and concerns (Section 4.1).

Tankage - A ground rule for this study was that the vehicle would be delivered to orbit in the STS. Although it would be desirable to deliver the space-based OTV to orbit in its fully assembled configuration, assembly on orbit is consistent with the required recurring operations. Other geometry constraints for a space based OTV are associated with the problem of aeroheating during the aerocapture maneuver. In other words, a short stage is desirable to minimize requirements on aerobrake size and thermal protection for the forward end of the stage. Also the space station hangar will certainly not be an unlimited envelope of volume in which to perform vehicle inspection and servicing. Therefore, in addition to the classic shuttle constraints, other constraints drive the OTV toward reasonable compactness.

In addition to the overall geometric considerations for the OTV (tankage and packaging), the concerns include an inspection and replacement capability for many of the components on board the vehicle. Modular arrangement of the tankage would be functional from a contingency replacement standpoint in case of tank damage for one reason or another (meteroid penetration, collision, MLI replacement, etc). A routine interchange of tankage may present problems with regard to the complex interface requirements and handling. Nonetheless, an "open pack" arrangement of all subsystems, including tankage, must be inherent in the design of a space-based OTV.

An alternative to total vehicle modularity (where the vehicle size can be scaled up or down to "fit" the mission/payload) is to have a mixed fleet with at least two different sizes of vehicles. Although these vehicles' tanks would not be intended for routine changeout for mission/configuration flexibility, they would lend themselves to system modularity for inspection, subsystem access, and contingency replacement. The total number of eventual missions, along with vehicle reusability limitations, may warrant at least a two-vehicle fleet. For instance, a two-stage concept capable of performing the large payload LEO-to-GEO missions and also capable of returning each stage to LEO via aerobraking is a possible solution. The two stages may not provide significant performance advantages over a single large vehicle for a large payload (if at all), but would provide a method of performing the wide spectrum of missions with minimum offloaded propellant. In other words, this additional configurational flexibility of two differently sized compact vehicles, along with the ability to stage them, would provide a much more efficient use of propellant for the range of missions by flying with a higher mass fraction than with the earlier vehicle/fleet (single large stage or modular vehicle) concepts mentioned. Also, from a servicing standpoint, the two compact stageable vehicle fleet would simplify the hangar and vehicle handling equipment design because:

1) The two stageable vehicles are the same size (or more nearly) than for the two-vehicle fleet concept consisting of a large vehicle and a small one;

2) The two stageable vehicles are reduced in size from the large single-stage vehicle. Their smaller size is also important from an aeroheating standpoint during the aerocapture maneuver;

3) Two small vehicles may ease turnaround servicing time requirements depending on mission frequency.

3.4 TRADEOFF STUDIES PERFORMED

A number of tradeoff studies were performed to appropriately define a vehicle and engine system that are to be space-based and reusable. The following list itemizes the pertinent issues identified:

1) Mixture ratio vs stage performance;

2) Number of engines;

3) Thrust level(s);

4) Throttling range(s);

5) Gimbaling vs differential throttling;

6) Fixed vs extendable/retractable nozzles;

7) Modular stage(s) vs staged vehicle vs mixed fleet (and corresponding tankage and structure).

The difficulty involving most of these tradeoff studies is that they are interrelated. For instance, the choice of engine number was one of the most important tradeoffs made and directly affected the selection of engine thrust, vehicle servicing, whether gimbaling may be required, packaging concerns with the nozzle(s), and strong consideration of engine reliability requirements. Understanding the synergism of all these factors was important in arriving at a justified conclusion as to the appropriate number of engines for a space-based OTV.

The judgment criteria considered during the performance of these tradeoff studies included:

1) Overall stage performance;

2) Individual subsystem impact;

3) Impact on servicing timeline, equipment, tools, and manned involvement;

4) Probability for mission completion and for safe crew return (reliability and acceptable failure criteria);

5) Mission/configuration flexibility;

6) Life-cycle cost, including all of the above criteria.

3.4.1 Major Configurational Characteristics of a Space-Based, Reusable OTV

Developing an OTV design that will be cost effective in capturing all the future space missions involves more than simply designing a single propulsion stage and providing it with reusability and serviceability. Because of the wide range of mission payload masses and mission requirements (such as deploy only or deploy/retrieve), a single-stage design would not result in an efficient use of propellant compared with other concepts.

Figure 3.4-1 shows the propellant requirement for completing the Rev 6 LEO to GEO missions (47) between 1994-2000 for various vehicle fleet concepts. The single-stage approach to capturing the missions is the most expensive in terms of propellant usage. The modular stage was sized so its half-tankage version (two LO_2 tanks and two LH_2) would hold about 33,000 lbm propellant and deliver 11,000 lbm to GEO. The full-tankage modular version holds twice this amount of propellant and is capable of performing the 13,000 lbm GEO deployment and retrieval mission.

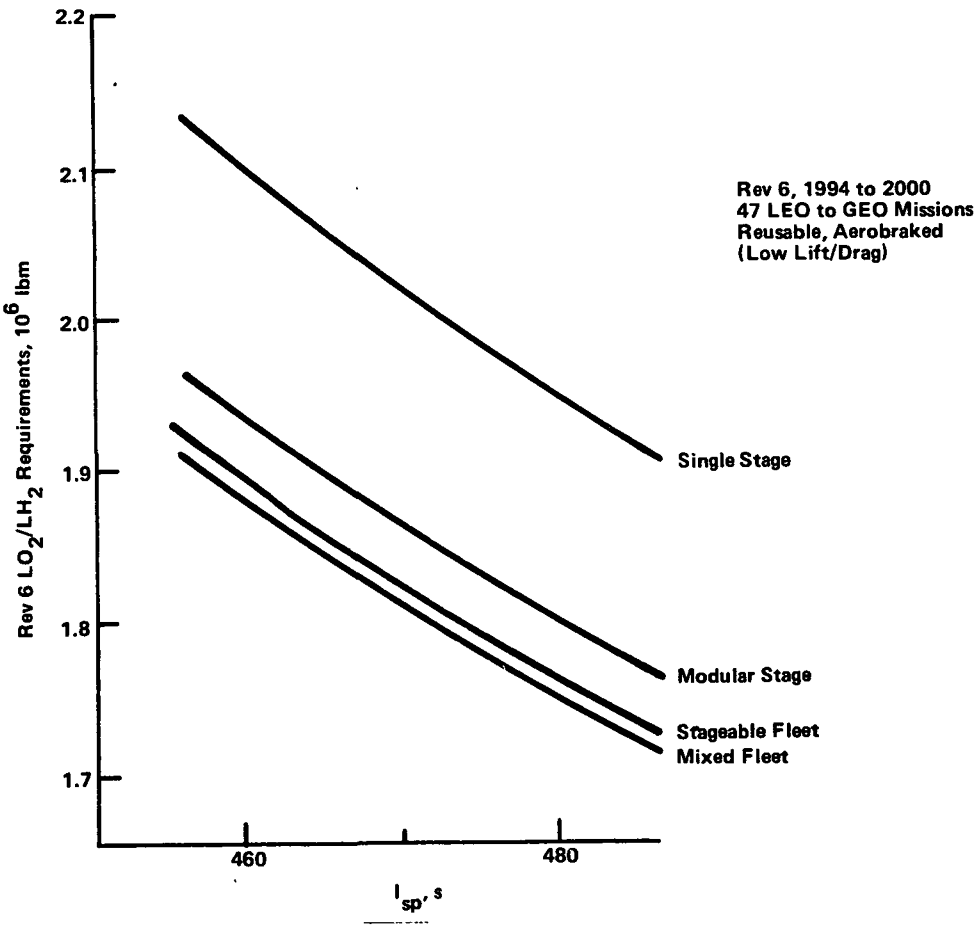

Figure 3.4-1 Vehicle Fleet Comparisons

The stageable fleet was sized so the first stage would deliver 16,000 lbm to GEO. It will also perform the perigee burn for delivery of the 13,000-lbm payload. The first stage would then return to LEO via an aerocapture maneuver. The second stage would perform the GEO circularization burn and then the decircularization burn and aerocapture maneuver for retrieval of the 13,000-lbm payload. The two stages would be used separately for the smaller mission payloads according to their capabilities. The propellant capacities of the first and second stages are about 40,000 lbm and 32,000 lbm respectively. The smaller stage (second stage for the 13,000 lbm deploy/retrieve mission) was sized to deliver 11,000 lbm to GEO because this size of payload and its frequency of flight appeared to be an effective breakpoint for propellant savings. However, both stages being sized to deliver 16,000 lbm to GEO would still provide an efficient means of flying all missions.

The mixed fleet simply consists of a large stage sized for the 13,000 lbm deploy/retrieve mission and a small stage sized for the smaller missions. The smaller stage was sized for a 14,000 lbm delivery to GEO. Unlike the smaller stage of the stageable fleet that was sized for an 11000 lbm delivery to GEO, the smaller stage of the mixed fleet was sized for a larger payload because of the difference in overall stage sizes of the mixed fleet. The breakpoint in propellant capacity for sizing the smaller stage of the mixed fleet is also a result of the large number of payloads of 14,000 lbm.

The main conclusion associated with Figure 3.4-1 is that a single large stage design sized for the largest payload/mission is an inefficient method of performing the missions from a propellant usage standpoint (compared to the possible alternatives). The modular stage, of course, has associated "scar" weight through its modularity that the mixed fleet (two or more stages) does not have. The stageable fleet, of course, would have a scar weight associated with the staging interfaces.

An example of stage performance and character vs service life and servicing requirements is selection of tankage. For instance, one choice would be to provide the vehicle with complete meteoroid protection for the vehicle's design life so the probability of a penetration would be less than that acceptable. This design might also alleviate any need to reproof-check (reverify) the tank(s) between missions (unless an unforeseen incident such as a collision occurred). Also, the removal of MLI for tank inspection would not be required. The effect of dry weight on this particular design would result in noticeably greater propellant usage than for a higher performance (less protective dry weight) design. However, the higher performing vehicle would demand a much higher level of servicing activity. Therefore, the nature of a space-based reusable vehicle is constrained by its impacts on the servicing requirements.

3.4-2 Baseline Configuration Selection

The OTV characteristics ground-ruled early in this study included:

1) Deliverable in STS cargo bay;

2) LO_2/LH_2 propellants;

3) Aerobraked (low lift-to-drag ratio);

4) Space-based and reusable;

5) Manrateable;

6) Able to capture Rev 6 missions.

A desirable characteristic to add to the list is that of high performance to minimize the use of propellants (and therefore their launch cost). However, the amount of onorbit servicing the resulting vehicle would require must be considered. Without prior knowledge of the results of a servicing/performance tradeoff study, a vehicle configuration was chosen that represents a reasonable compromise of performance and service intensity as well as complying with the study ground rules.

The vehicle concepts depicted in Figure 3.4-2 and 3.4-3 are the study baseline vehicle concepts. Schematics of the recommended feed systems are shown in Figures 3.4-4 and 3.4-5. The vehicles were sized to perform the largest Rev 6 delivery mission of 16,000 lbm to GEO. The only missions requiring more total impulse (a significant amount more) are the 13,000-lbm manned deploy/retrieve missions to GEO. The rationale for sizing to the 16,000-lbm delivery to GEO mainly lies in the advantages of performing the profitable missions that comprise most of the mission model.

The low frequency of 13,000-lbm manned missions does not provide the rationale for a dedicated large single stage. The baseline configuration, however, is capable of providing the perigee boost to the 13,000-lbm manned payload and an associated propulsion system (cryogenic or storable). The manned payload with its own propulsion system for GEO insertion and decircularization would seem reasonable from several standpoints. First, if it were a storable system, the boiloff problem might be eliminated for the long onorbit time. Also, if the system were pressure fed, the manrating complexities would be reduced. Because of the low number of projected manned missions, the performance penalties associated with the storable propulsion system concepts previously presented would be minor relative to the overall mission model. This scenario would leave the OTV as simply a delivery workhorse, for the most part, and may even have implications of reducing the operational requirements with regard to manrating.

Let us assume, however, that the 13,000-lbm manned payload will not have its own dedicated propulsion system for GEO insertion, decircularization, and possible reentry with no need to return to the space station. The baseline vehicles shown in Figures 3.4-4 and 3.4-5 will serve well as the larger of two stages for a stageable/mixed fleet. A case exists for a stageable and/or mixed fleet from the standpoints of propellant savings, mission frequency and turnaround time, rescue capability, etc. In any case, for the purposes of this study, the baseline configurations provided reasonable starting points for performing main propulsion system tradeoff studies and determining the implications of space basing and reusability on an OTV.

VEHICLE DEFINITION

Propulsion System: Space Based, Reusable ,
Propellant: LO_2/LH_2 , MR: 6.0
Main Engine: Thrust 15000 lb , ISP: 480 s
Payload Delivery to GEO 16000 lbs, ASE ---- lb

ITEM	WEIGHT, LBS	REMARKS
AVIONICS	972	
STRUCTURES	610	INTERNAL CORE
THERMAL	362	0.5" MLI, TVS
PROPULSION		
FU Tank (2)	418	2219 AL, .025"
OX Tank (2)	217	1.5 NON-OPTIMUM
Propellant Feed	310	INCLUDES PNEUMATICS
Pressurization	63	AUTOGENOUS
Engine Assy (2) 40 lbm - Actuators	738	7500 lbf EACH
ACPS & ACCUMULATORS	350	INCLUDES GAS GEN
AEROBRAKE	1090	44 ft DIA
CONTINGENCY (15%)	770	
DRY WEIGHT	5900	
PROPELLANT, NON-USABLE	640	MARGIN AND RESIDUALS
BURNOUT WEIGHT	6540	
BOILOFF	125	42 HR. MISSION
PROPELLANT LOADED	43318	220 ACS GO_2/GH_2
STAGE INITIAL WT.	49218	
PROPELLANT, USABLE	42333	Main
MASS FRACTION	.860	USABLE/INITIAL

CONFIGURATION SKETCH

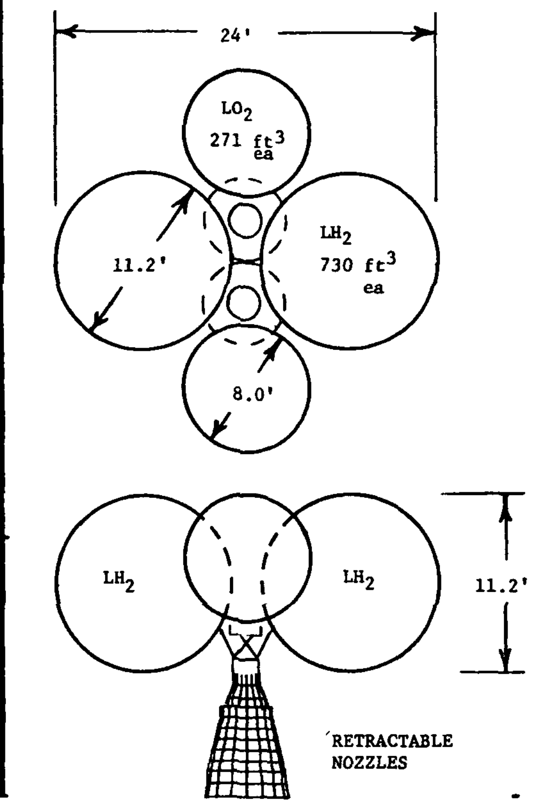

Figure 3.4-2 Space-Based Reusable Vehicle Definition, 2 Engines

VEHICLE DEFINITION

Propulsion System: Space Based, Reusable ,
Propellant: LO_2/LH_2 , MR: 6.0
Main Engine: Thrust 15000 lb , ISP: 480 s
Payload Delivery to GEO 16000 lbs, ASE --- lb

ITEM	WEIGHT, LBS	REMARKS
AVIONICS AND POWER	972	
STRUCTURES	610	INTERNAL CORE
THERMAL CONTROL AND VENT	362	0.5" MLI, TVS
PROPULSION		
FU Tank (2)	410	2219 AL, .025"
OX Tank (2)	213	1.5 NON-OPTIMUM
Propellant Feed & P.U.	434	INCLUDES PNEUMATICS
Pressurization	63	AUTOGENOUS
Engine Assy (4) 46 lbm - ACTUATORS	528	3750 lbf EACH
ACPS & ACCUMULATORS	300	
AEROBRAKE	855	38 FT DIA
CONTINGENCY (15%)	712	
DRY WEIGHT	5459	
PROPELLANT, NON-USABLE	640	MARGIN AND RESIDUALS
BURNOUT WEIGHT	6099	
BOILOFF	125	42 HR MISSION
PROPELLANT LOADED	42087	220 ACS GO_2/GH_2
STAGE INITIAL WT.	47546	
PROPELLANT, USABLE	41102	MAIN
MASS FRACTION	.864	USABLE/INITIAL

CONFIGURATION SKETCH

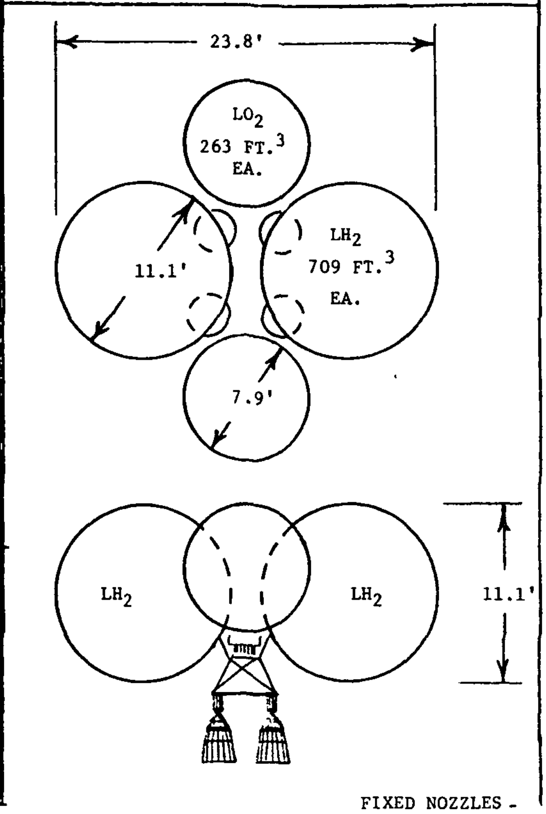

Figure 3.4-3 Space-Based Reusable Vehicle Definition, 4 Engines

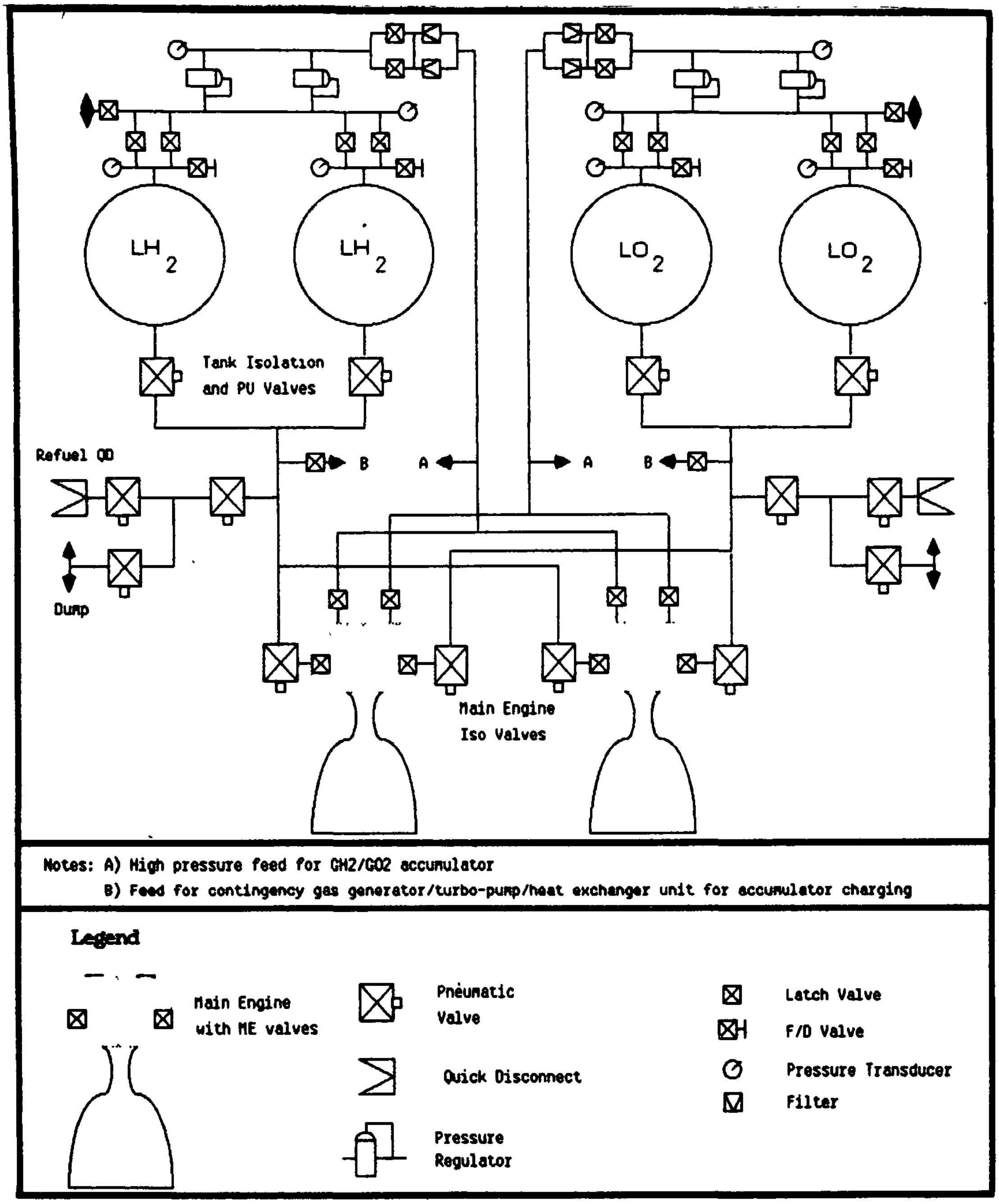

Figure 3.4-4 Main Feed System–2 Main Engine Schematic

Modular spherical tanks were chosen over parallel cylinders because of increased performance and decreased surface area and subsequent propellant boiloff. Also, a shorter length is highly desirable for the vehicle to minimize aerodynamic base heating. The vehicle, however, would not be deliverable in its flight configuration in the STS orbiter bay. It would be assembled after initial delivery to LEO if delivered in the cargo bay. The configuration of modular tankage is based on contingency replacement of tankage only. For example, if a tank suffered a meteoroid penetration that was detected in a routine leak test, the tank would be replaced. The vehicle is not intended for routine configuration modularity for mission flexibility because at this time fluid disconnect technology will not allow such routine freedom. Therefore only contingency replacement of tankage is recommended.

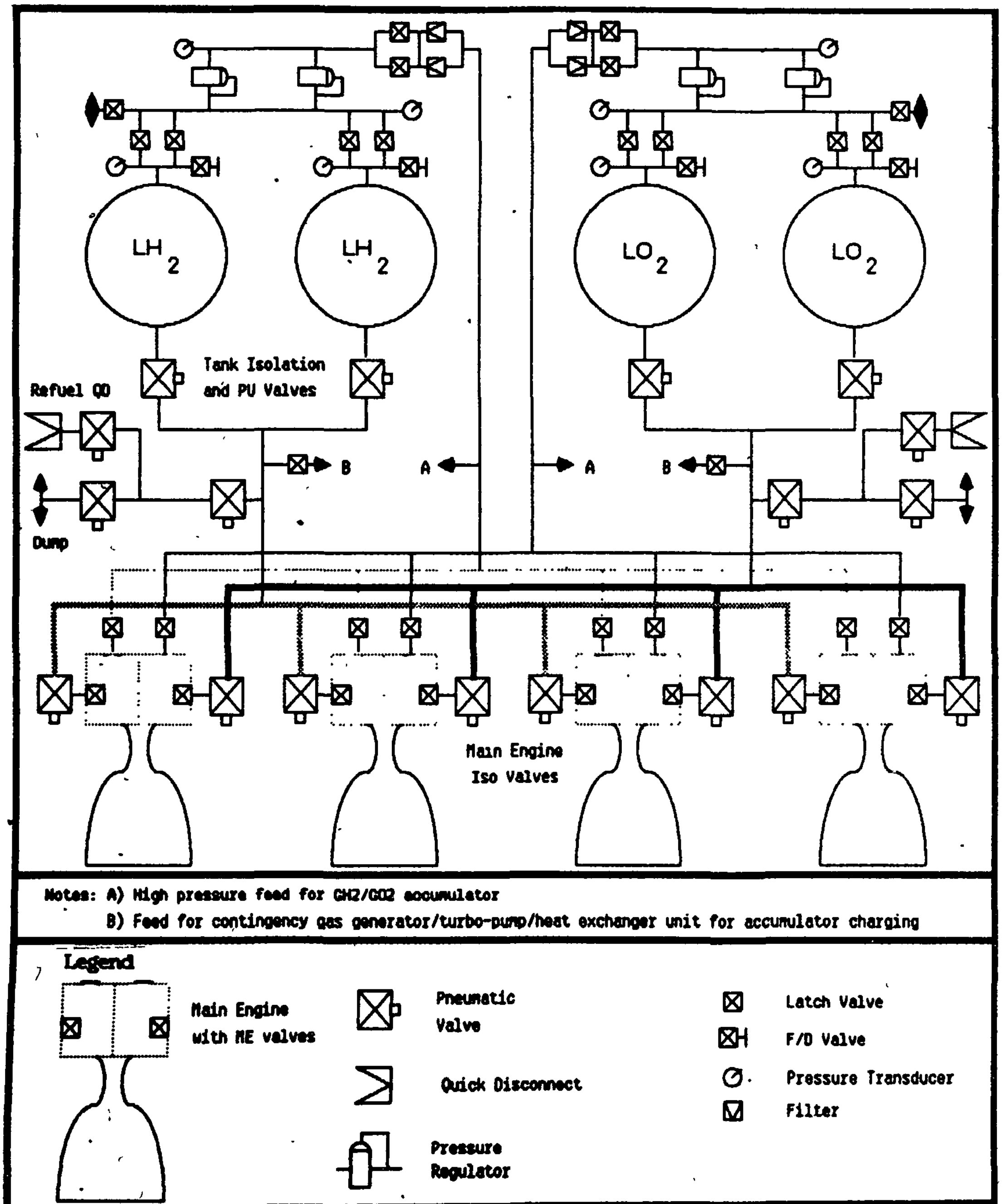

Figure 3.4-5 Main Feed System—4 Main Engine Schematic

Vehicle configurations designed to require no tank modularity for contingency replacement are a possibility. In other words, the exterior structural shell and meteoroid shielding provide the OTV tankage with the protection against probable penetration during the vehicle's design lifetime. Based on current estimates, the weight penalty, however, would significantly increase propellant usage over the baseline configuration and overshadow the costs of periodic tank replacement.

Based on the results of vehicle sizing over a range of mixture ratios, the minimum vehicle weight consistently resulted in a mixture ratio of 6.0. Engine performance vs mixture ratio information from Pratt and Whitney (Fig 3.4-6) was used for the vehicle sizing. Figure 3.4-7 shows the results for various missions.

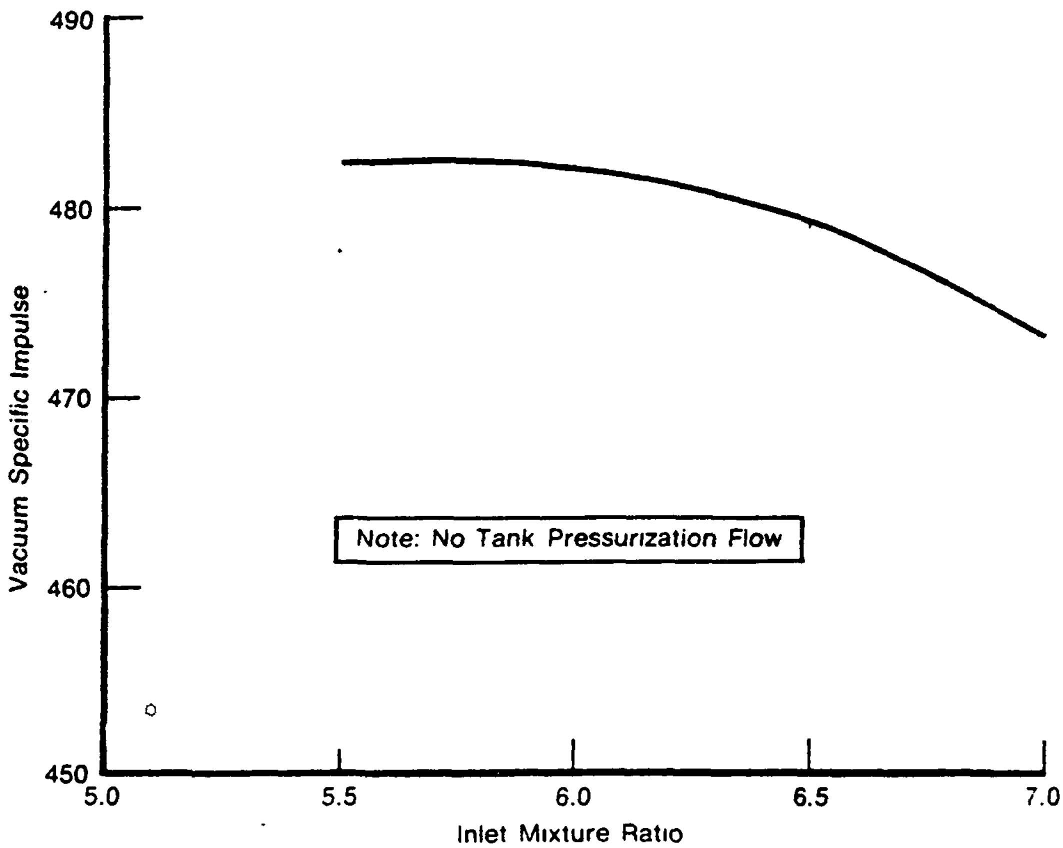

Figure 3.4-6
Estimated Effect of Inlet Mixture Ratio on Vacuum Specific Impulse at Full Thrust

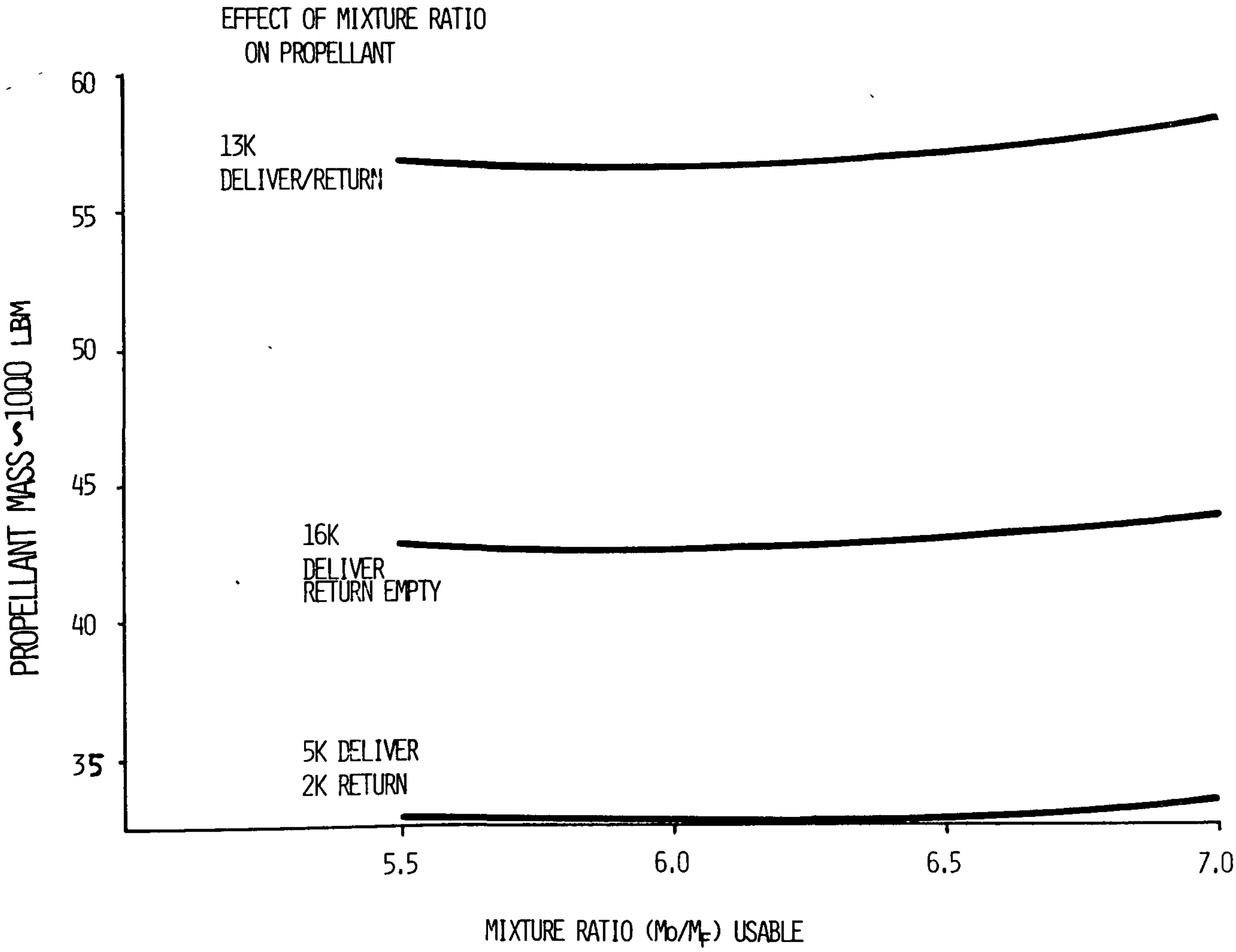

Figure 3.4-7 Effect of Mixture Ratio on Propellant

3.4.3 Engine Design Criteria for Aerocapture

One of the main drivers of engine design criteria for the study baseline vehicle may be the operational requirements during the aerocapture maneuver. Whether aerospike operation is a feasible means of controlling vehicle trajectory during aerocapture remains an open issue.

The design philosophy for the aerobrake and engine system will include several considerations. Minimizing the aeroheating environment of the vehicle is a major concern in keeping the thermal protection system weight for the vehicle to a minimum. Therefore, the aerobrake must be properly designed to withstand the predicted heat load (2500°F) and radiate it. In addition, short vehicle length must be emphasized to minimize the base heating in the wake of the flow around the aerobrake. The aerobrake must also be configured to be compatible with the predicted airloads (up to

15 psf). Also, the aerobrake/vehicle configuration and operation during the aerocapture maneuver should lend themselves to aerodynamic stability to minimize the control authority requirements.

The overall vehicle, aerobrake, and main propulsion system design and subsequent operation must provide adequate control for successful return from high earth orbit during normal main engine system operation as well as during an engine-out condition. Because the aerospike operation (engine operation during the aerocapture maneuver) may be very effective for drag modulation with blunt vehicle configurations as well as help minimize vehicle structural and heat loads, this particular concept of operation and brake design was carried through the main propulsion system tradeoff studies. Based on studies performed to date, a 70-deg Viking-shaped cone aerobrake appears to be a reasonable compromise between drag and stability requirements.

Other aeroassist concept candidates include mechanically actuated surface(s) for either lift capability or surface area variation for drag modulation. A viable concept is one of vehicle cg offset and roll for an angle of attack capability. This concept would most probably provide a significant degree of control with a minimum amount of probable failures.

Because of the wide variations expected in atmospheric characteristics, it is questionable that a single degree of vehicle control will provide total assurance for successful aerocapture on a routine basis. For example, angle of attack (or rather, some degree of lift capability) in addition to drag modulation may be required for an adequate control capability. Possible failure modes of the vehicle must also be considered, particularly for the aerocapture maneuver because of the critical nature of steering through a "thin" corridor and avoiding the obvious hazards of lack of control and augering into the lower atmosphere.

Aerospike operation, particularly with two or more engines, may provide this additional degree of control capability. Martin Marietta Idea Report 83YD59, dated August 11, 1983, suggests that differential throttling of multiple engines may provide lateral motion (via induced angle of attack) by tailoring the bow shock and subsequent streamlines around the vehicle. Whether this degree of vehicle control is necessary or even feasible remains an open issue. Figures 3.4-8 and 3.4-9 depict this method of control for two- and four-engine vehicle concepts respectively.

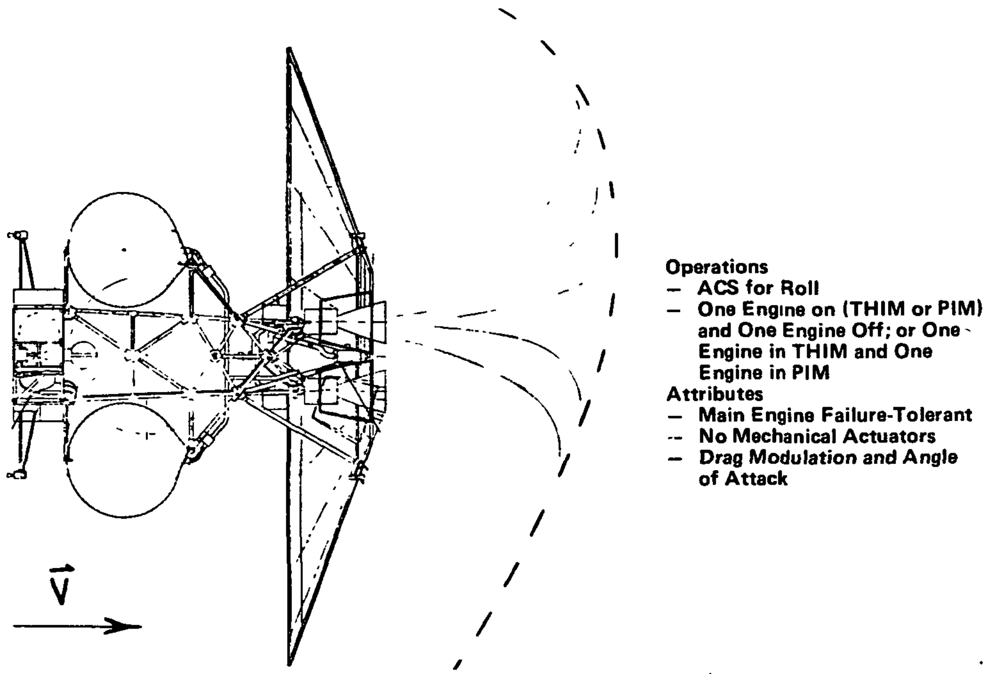

Figure 3.4-8 Aerocapture Maneuver with Two Main Engines

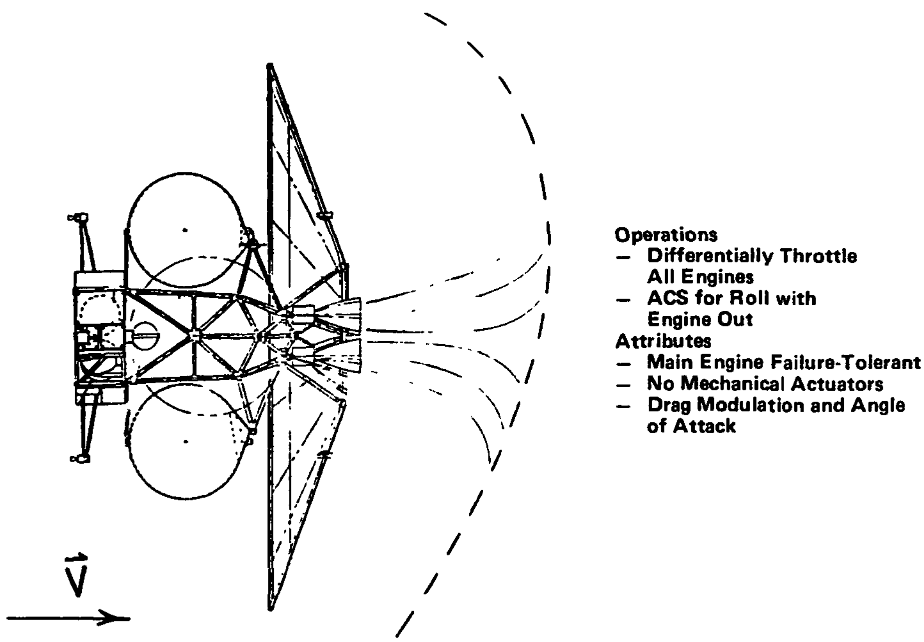

Figure 3.4-9 Aerocapture Maneuver with Four Main Engines

Other open issues with regard to aerobrake design include:

1) Lack of data relative to variation of aerodynamic drag coefficient and flow stability during aerospike operation;

2) Effects of aerospike on reducing aeroheating;

3) Base heating from aerobrake radiation as well as convective heating;

4) Necessity of continuous throttling vs thrust steps for drag modulation during aerospike operation.

The following discussion presents the recommendations for aerobrake/main propulsion system design criteria based on the analyses performed.

As discussed earlier, the aerospike operation has the potential of providing a vehicle control capability in the upper atmosphere (drag modulation), aerothermal shielding for the vehicle, and (in conjunction with drag modulation) loads alleviation on the aerobrake structure. Figure 3.4-10 shows the acceleration levels estimated for various combinations of low-thrust THIM (tank head idle mode) burns and a high-thrust PIM (pumped idle mode) level. Notice that two low-thrust burns before and following pumped idle thrust help minimize the loads that affect the aerobrake structure. The thrust range for aerospike operation is estimated (based on present analyses) to be between 100 and 800 lbf. For thrust greater than 800 lbf, the rocket effect begins to take over and the drag reduction effect is negated.

Two or more main engines have advantages as well as disadvantages with regard to the aerocapture maneuver. Among the advantages are:

1) Shutting down selected engines (or differential throttling them) to add to control capability of vehicle via streamline (bow shock) shaping and possible use of ACS for roll-controlled steering;

2) Achieving several thrust "steps" at low levels for drag modulation by various combinations of engine(s) operation(s) such as off, THIM, or PIM.

The disadvantages of having more than one engine include:

1) Size of opening in aerobrake;

2) Thrust vector offset with engine out;

3) Vehicle control complexity.

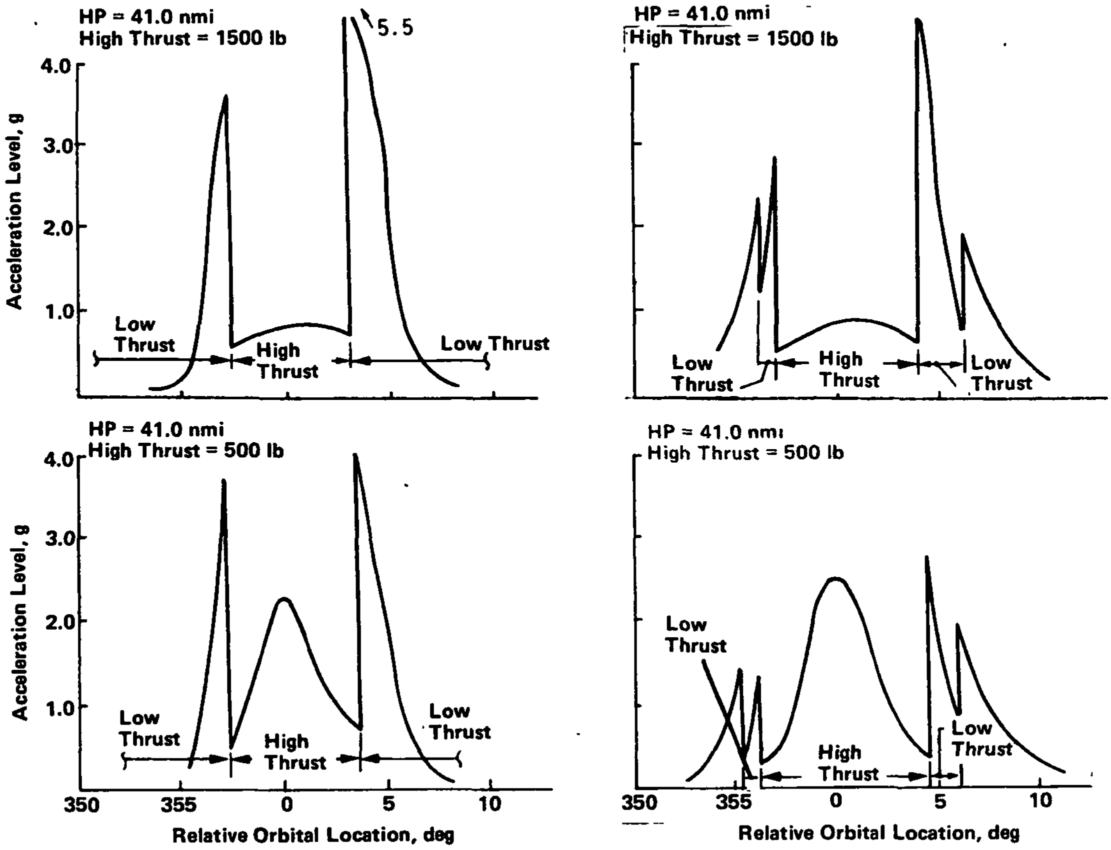

Figure 3.4-10 Trajectory Simulations—Aerospike Control

Contingency operation of performing the aerocapture maneuver by using ACS aft-pointing thrusters to create an aerospike effect may be possible. To target for a multiple-pass (2 or 3 passes) return trajectory without main engine(s) operating, a certain amount of vehicle control is still required. Present estimates suggest that a maximum of 200 to 300 lbf may be adequate for performing a 2- to 3-pass return. Therefore the use of ACS thrusters for the return via multiple passes is a very feasible alternative.

Depending upon the results of vehicle/engine optimization, considering engine length, diameter, and performance, nozzle retraction may be required before entering the atmosphere. With the emphasis on short vehicle length to minimize base heating in the wake of the aerobrake, the long high-performance (high area ratio) nozzles of the main engines will most likely protrude through the aerobrake during the burns before entering the atmosphere (up through GEO decircularization). The phasing and rendezvous burns following the aerocapture maneuver are small and the performance penalty for not reextending the nozzles may not be significant.

3.4.4 Main Propulsion System Tradeoff Studies

The determination of major engine characteristics should depend on several criteria--propulsion system reliability, performance, serviceability, cost, etc. During Phase I of this study, geometry and performance were given highest priority, keeping in mind the implications of cost, manrating, reliability, and serviceability. Cost and serviceability were considered in Phase II (Section 4.0) when a life-cycle cost model of the OTV service life was developed and tradeoff studies were performed. The main propulsion system trade study ground rules were:

1) Total of 15,000 lbf thrust;

2) 1, 2, 3, 4, 5, and 3 inline engines;

3) Fail operational/fail safe included in weight comparisons (exception: single engine that is fail-safe only);

4) Long (retractable) and short (fixed) nozzles for each engine system;

5) Two gimbal angles, 10, 20 deg (except for the 5 and 3 inline engine cases which do not require large gimbal angles);

6) Long engine length chosen where length penalty balances performance gain;

7) Adjacent nozzles separated by 1/2 ft.

The performance sensitivities for the baseline stage are as follows:

$$\frac{2.8\ \text{lbm propellant}}{\text{lbm dry weight}} \quad \text{and} \quad \frac{146\ \text{lbm propellant}}{\text{second Isp}}$$

These were used in comparing propellant requirements and total vehicle weights for various numbers of engines, varying gimbal angles, nozzle design, and engine arrangements. Aerobrake size (and consequently weight) must increase as it is moved aft to package the engine(s) because of the base heating that would otherwise occur at the forward end of the vehicle. Therefore the engine envelope is a strong driver in overall vehicle configuration and performance. (Aerobrake and structure weight sensitivity is 57 lbm/ft further aft.)

Engine lengths for the retractable nozzle cases were chosen so performance gains were balanced by the penalty of longer nozzle length (in retracted configuration). Two axis gimbaling is required for up to four-engine configurations for a two-engine-out capability. The exception is for three engines inline with an ACS for backup for decircularization at GEO and return. In this case, the two outboard engines require one-axis gimbaling and the middle one requires two axes. Five engines provide a two-engine-out capability with only one engine requiring two-axis gimbaling. The ground rule for adjacent packaging of engines required a spacing of 0.5 ft between nozzles.

To provide equivalence in failure criteria (fail operational/fail safe or two-engine-out capability) for the two-engine system vs the multiple engine (3 to 5) cases, the performance penalty of extra loaded propellant for ACS return of the stage only from GEO was calculated. Because the ACS system (GO_2/GH_2) will operate at a mixture ratio of approximately 5 to 1 and a lower I_{sp} than the main engines, extra propellant is required for contingency as a backup mode as opposed to extra engine(s). Figure 3.4-11 shows this weight penalty as a function of main propulsion system performance. For instance, loss of the main propulsion system (MPS) with a performance of 490 s results in greater contingency propellant requirements than for MPS performance of 480 s because of the greater difference in MPS and ACS performance.

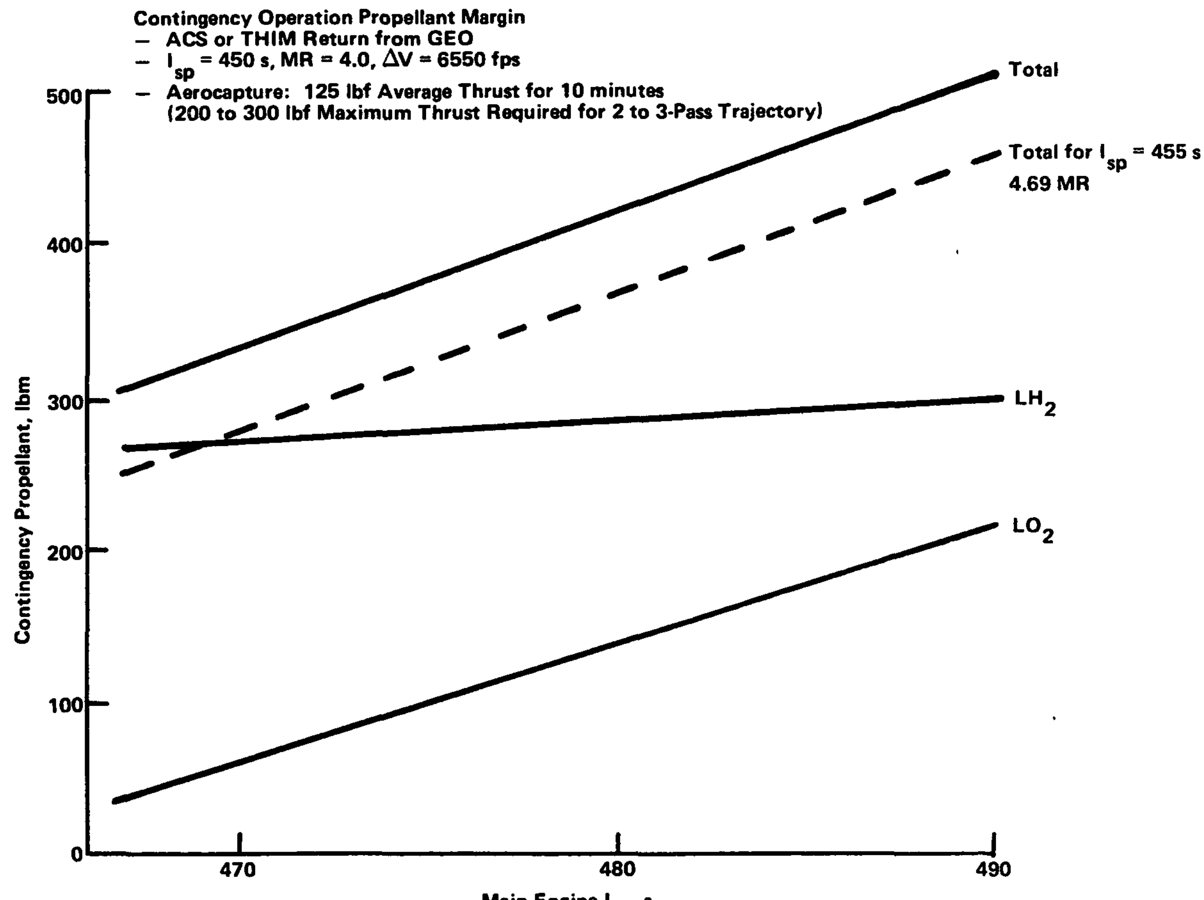

Figure 3.4-11 Contingency Operation Propellant Margin

The parametric vehicle data points were computed using the sensitivities mentioned and engine data from Pratt and Whitney Aircraft and Aerojet Tech Systems. These two sets of engine data appeared to "bound" the variation of engine performance with thrust level. In other words, an attempt was made to include the range of proposed engine concepts (large to small) and their subsequent inherent characteristics and estimated performance. Figure 3.4-12 shows the total vehicle weights resulting from the Pratt and Whitney data and the results for the Aerojet data are shown in Figure 3.4-13.

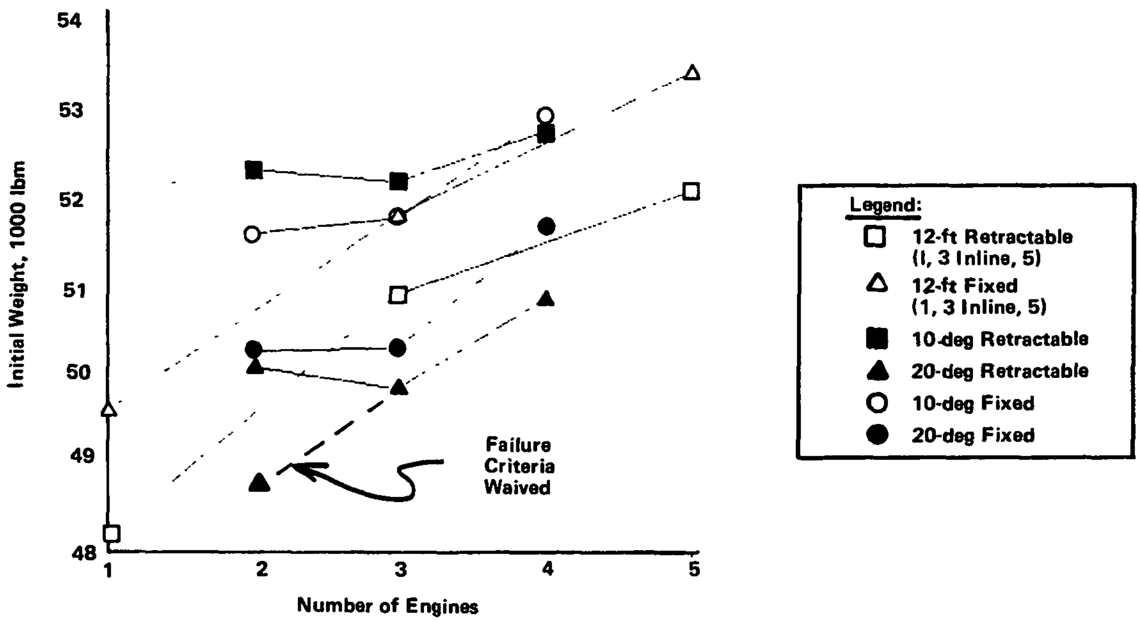

Figure 3.4-12 Initial Weight vs Number of Engines

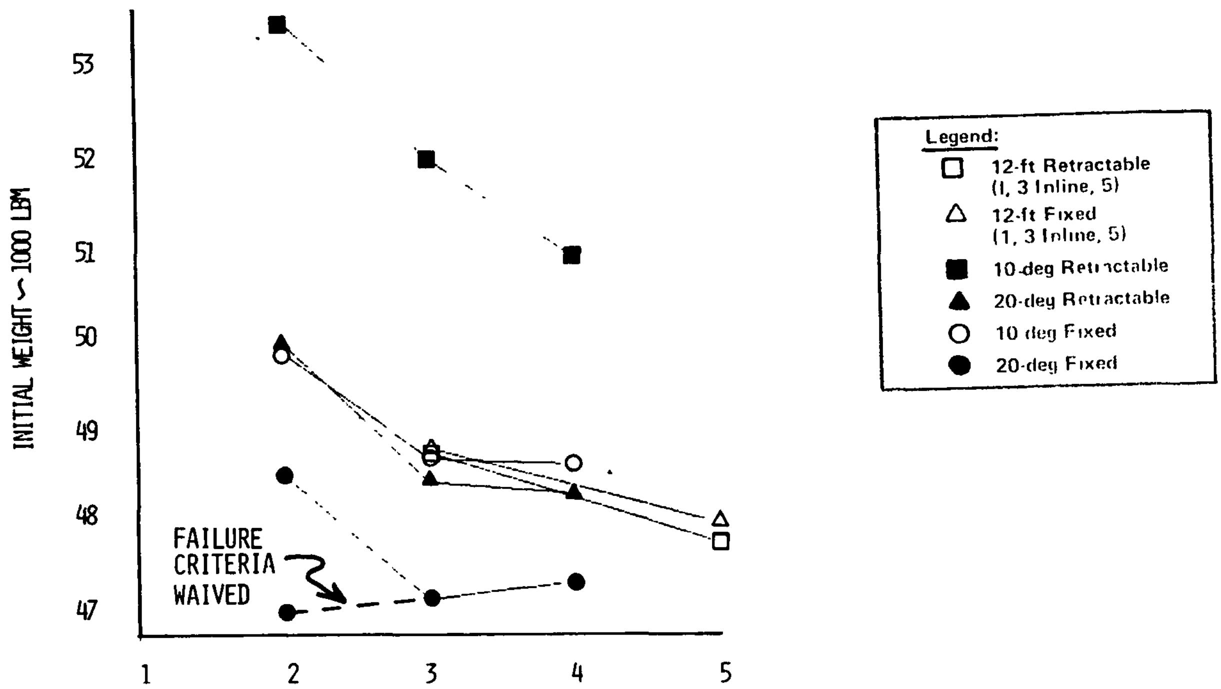

Figure 3.4-13 Initial Weight vs Number of Engines (Aerojet Engine Data)

Example notations for the charts are:

12-ft retractable - 1, 3 inline, or 5 engines that are 12 ft aft of the tankage/accumulator subsystem interface, with retractable nozzles;

10-deg Fixed - Engines allowed to gimbal up to 10 deg for an engine-out capability with fixed (standard) nozzles.

Table 3.4-1 shows the resulting conclusions and engine operational requirements recommended from all Phase I activity.

3.4.5 Baseline Vehicle Configuration

Two baseline configurations were developed (Fig 3.4-14 and 3.4-15) for two engines and four engines respectively. The two-engine configuration is representative of an optimized vehicle incorporating Pratt and Whitney engines with retractable nozzles (as per the main propulsion system tradeoff study). Because the tradeoff study showed short fixed nozzles to be optimum for the Aerojet engine data and less penalty for lower thrust, the four-engine configuration is representative of an optimized vehicle incorporating four Aerojet engines.

Table 3.4-1
Phase I Main Propulsion System Tradeoff Study Conclusions and Engine Operational Requirements

Conclusions

– Results are sensitive to engine performance and geometry data.
– Further investigate 1, 2, 3, or 4 engines in Phase II
  – Vehicle subsystem impacts
  – Implications of reusability/servicing
  – Mission flexibility
  – Life-cycle cost.
– *Nozzle design*
  – Fixed at 60 in. for diameter reasons (Aerojet)
  – Retractable to 60 in. for length reasons (Pratt & Whitney)..
– Benefits of retractable nozzles are more sensitive to available gimbal angle than fixed.
– Differential throttling shows little or no benefit.
  – 5 or 3 inline engines not competitive with other configurations.

Requirements

– Duty cycle
  – 6 burns/mission (21-minute burn time)
– Throttling (or step thrust levels)
  – Flow stability and drag modulation during aerocapture maneuver (aerospike).
  – Reduce thrust after perigee burn to extend engine life (TBD).
  – Low-thrust missions (LSS).
– Gimbal
  – *Up to 20 deg provides significant benefits in total vehicle mass and propellant consumption.*
– Specific impulse
  – Greater than 460 s.
  – Lifetime and reliability implications.
– Mixture ratio
  – 6.0 oxidizer/fuel provides lowest weight stage and propellant usage.

The baseline main propellant tanks are supported at two locations--forward and aft of each tank (Fig. 3.4-14 and 3.4-15). The forward supports react forces in the X and Y directions while the aft supports react against in the X, Y, and Z directions. The structural attachments are designed to be easily detached and all fluid and electrical lines are equipped with quick-disconnects (QDs) to facilitate onorbit construction, servicing and/or replacement of the tanks.

The baseline configuration employs a welded tubular aluminum primary structure. Tubular members were selected on the basis of their high strength-to-weight and stiffness-to-weight ratios. Additional stiffness gains and weight savings may be realized if an all-composite structure is used. While an investigation of this possibility should eventually be instituted, it was outside the scope of this study.

The desire to minimize vehicle length and diameter placed strict limitations on the size and envelope of the primary structure located between the main propellant tanks. To avoid an undesirably heavy structure, the primary launch and aerobrake loads on the tanks in the X, Y, and Z directions are reacted through the tank aft structural attachments. The structure forward of the tank aft ends then need only react tank loads in the X and Y directions in addition to payload and equipment module loads in the X, Y, and Z directions. This concept greatly reduces the size and weight of this forward structure.

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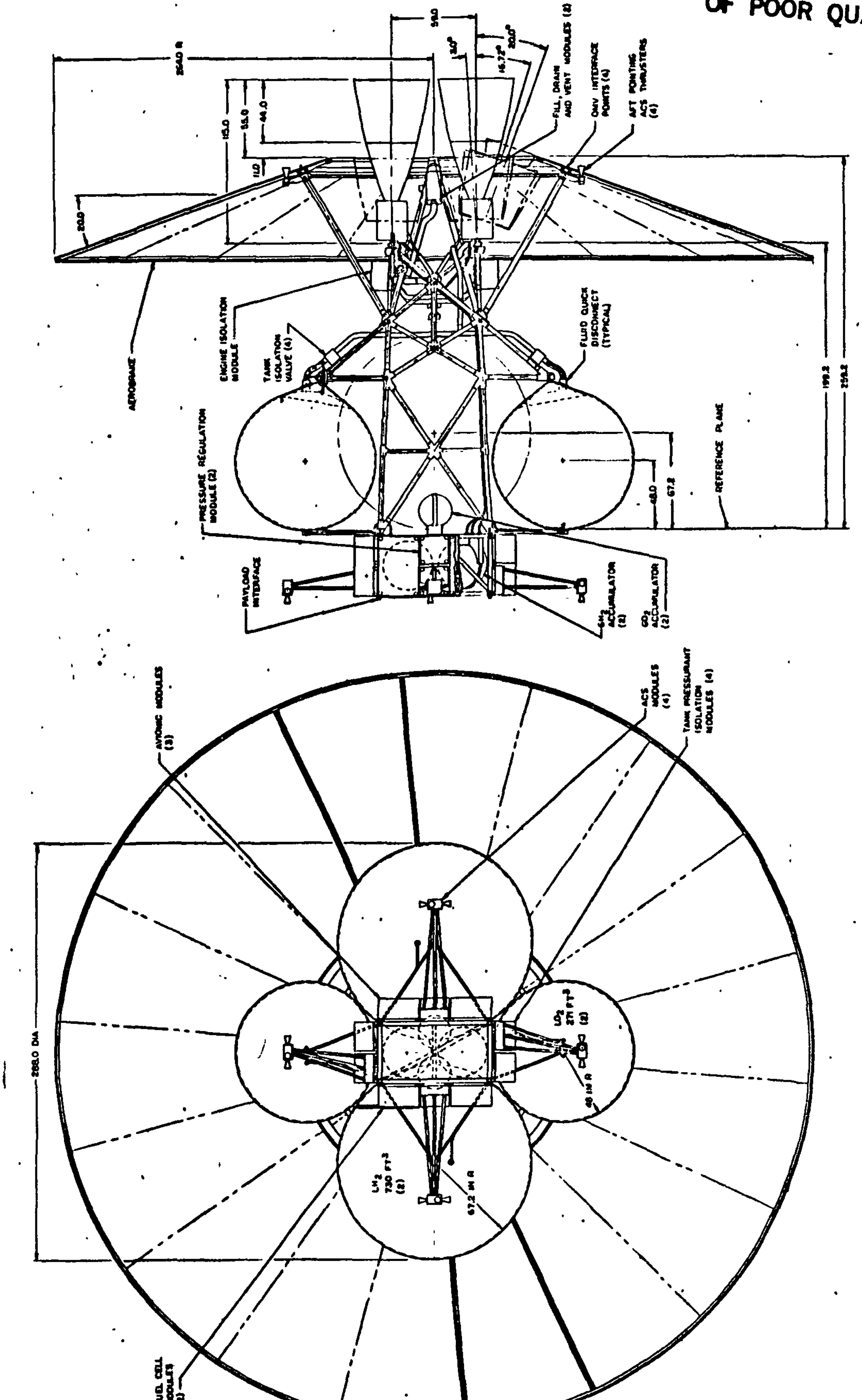

Figure 3.4-14 Two-Engine Baseline Layout

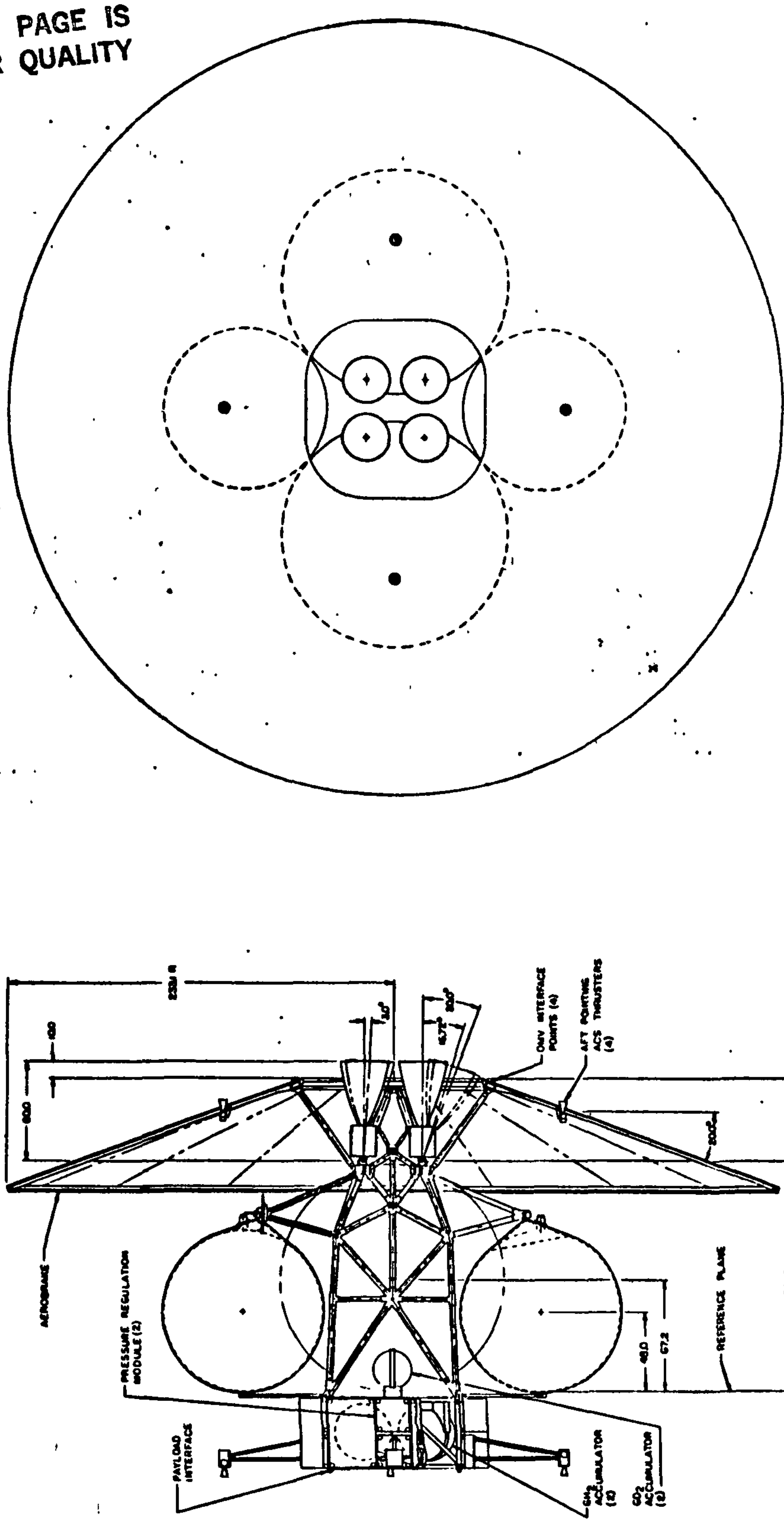

Figure 3.4-15 Four-Engine Baseline Layout

The equipment module is attached to the primary support structure forward of the main propellant tanks. The GH_2 accumulators are internal to the equipment module and the GO_2 accumulators are supported off of the module's aft end. The forward bulkhead of the module serves as the payload interface. Detachable units containing the OTV ACS, avionics, pressurant regulation and control, and power systems surround the equipment module. Each of these detachable units is attached at four structural QD points in addition to any fluid or electrical QDs that may be required. These units are serviced by removal and replacement with a new or refurbished module.

In addition to the ACS units attached to the equipment module, four aft-pointing ACS thrusters are attached to the aerobrake support structure. These aft-pointing thrusters penetrate through the aerobrake, permitting them to provide a limited aerospike in the event of a main propulsion system failure. This thruster location eliminates any thruster efficiency losses that would result from thruster plume impingement on the aerobrake if these thrusters were located forward of the aerobrake (near the payload interface plane).

An OMV is attached to the OTV aerobrake and the aerobrake is attached to its support structure with the same type mechanisms and at the same relative locations. This concept enables the OMV to be attached to the OTV without the aerobrake attached. Berthing of the OTV with the refueling station also employs these same mechanisms, with an RMS accomplishing the actual berthing operation. Propellant handling is accomplished through fluid lines with QDs that could penetrate through the aerobrake if it were left attached.

3.4.6 Space-Based OTV Propulsion Subsystems Design

A space-based OTV will require a unique design to accommodate the needs of onorbit basing. This is necessitated by both space basing and the desire for reusability. These two requirements are beyond those imposed on previous upper stages. The space station will provide the logistics and manned presence. From a servicing standpoint the space station design will also influence the OTV design. Such items as hangar design, resupply accommodations, and the astronauts' servicing capabilities will all influence the ultimate design. Because logistic considerations such as consumables resupply and spares inventory will represent a large portion of the routine operating expenses, the OTV design should account for these requirements. It is also desirable to reduce consumables both in quantity as well as in number. Fewer resuppliable quantities on the OTV will simplify the servicing operations. The subsystems design in the following description will attempt to address these concerns in addition to such "traditional" ones as minimum mass, cost, and maximum performance.

Current LH_2/LO_2 upper stage designs have been highly optimized for the present expendable mode of operation. High performance/low weight were the major design drivers. In addition to the main propellant, previous upper stages have also carried several fluids to be used for various functions, e.g., N_2H_4 for attitude control, and He for purging, pressurization and valve actuation. This represented the best compromise toward the goals of highest attainable performance and reliability. Reusability will shift the optimum breakpoints from one-time usage design life toward periodic replacement for certain high-maintenance or life-limited items such as the main engines. These items would become prohibitively expensive to make last the life of an OTV. Space basing adds yet another constraint in that the number of feasible maintenance tasks is severely limited by the available manpower and support equipment.

It is anticipated that the upcoming space station will develop technology for GH_2/GO_2 thrusters and associated components in the thrust range of interest for an OTV to achieve a GH_2/GO_2 attitude control system (ACS). Once these gases are stored on board, they could be made available to serve other functions as well. The electric power system (EPS) fuel cells could obtain their supply from the GH_2/GO_2 storage bottles (accumulators), eliminating separate storage and servicing. (Assuming fuel cells compatible with propellant-grade cryogenics.) GH_2 could be used in place of He as the actuator pressurant. The GH_2/GO_2 system would be recharged by a simple check-valve arrangement tied to the autogenous pressurant lines between the engine(s) and the tanks. The ACS could also be used as a backup for a failed main engine if a contingency means for charging the GH_2/GO_2 accumulators were provided. On the basis of these arguments, the use of GH_2/GO_2 on a space-based OTV is proposed. The following subsystem descriptions will further illustrate the advantages of a fully integrated cryogenic OTV design with a preliminary point design.

The main propulsion system design is shown schematically in Figures 3.4-4 and 3.4-5 for two and four main engines, respectively. The two-engine and four-engine cases have been selected as the baseline. Only the number of engine isolation valves (propellant and pressurant) and propellant manifolding are affected in comparing the two schematics. A single valve is installed at each tank outlet to provide tank isolation and for use by the propellant utilization (PU) subsystem to control tank-to-tank outage. A pneumatic valve with sufficient metering authority for the PU system will be required. The downstream manifold communicates with the main engine(s), an overboard dump, refueling quick-disconnects, and the feed for a contingency auxiliary power unit (APU) used to charge the GH_2/GO_2 accumulators. Each main engine is provided with isolation valves as a backup for the main engine valves. These valves are also needed to facilitate removal of the main engines. (There may be a blanket pressure in the lines.) A dump system is shown on the liquid side in case the main propellants need to be evacuated (anticipated to be a contingency event). Residuals left at the end of a mission would be recovered. A single QD and valve are shown for the refueling lines. Parallel QDs may be needed depending on the unit's reliability, which cannot be resolved at this point. Also not shown are filters for the main propellants. Presumably, each main engine inlet will have a filter that is serviced each time the main engines are changed out. The propellants should also be highly filtered by the time they are placed on board the OTV. Autogeneous pressurization will be provided by the main engine(s) at a temperature and pressure dictated by the autogenous pressurization and accumulator systems. The gases will be reduced in temperature and pressure by redundant regulators although a simple valve and orifice may be sufficient.

A thermodynamic vent system (TVS) will most probably be needed but is not shown because the exact requirements for the TVS are not known at this time and it should have a design similar to those used on existing stage designs. Recovery of the TVS vent gases is not anticipated because the relatively small amount of mass is not expected to be enough to warrant recovery. A pump would be needed to boost their pressure from near zero to the approximately 2000 psi needed for the accumulators. The mass and power requirements of these pumps are believed to outweigh any benefit gained by recovering the gas mass. However, future design efforts should reexamine this conclusion as the systems become available and more realistic analyses can be made. The TVS will present some servicing complications because most of the components will be mounted inside the propellant tanks. A high degree of reliability and/or access to the tanks will be needed if servicing the TVS components is necessary. Alternative means of controlling tank pressures and temperatures should be investigated.

The gaseous subsystem is shown in Figure 3.4-16. Essentially, four accumulators are fed through check valves from the main tank autogeneous pressurization lines. From these accumulators, gases are regulated and distributed to four "user" subsystems--the ACS thrusters, the EPS fuel cells, the APU, and GH_2 for pneumatic actuators. Four accumulators were chosen so the system could be divided in half, providing full redundancy. It may become desirable to use only half the system for particular missions. The accumulators will be sized by the worst-case mismatch between supply and demand. Isolation valves are provided for either of the two gas sources (autogeneous or APU) so recharging can be done by either. Each tank is provided with its own filter/check-valve unit in full isolation from the other tank. A redundant filter/check-valve unit is provided should either of the primary units fail. Similarly, separate regulators are used downstream from the tanks so the system can be run with only one tank (per gas). These regulator units are manifolded on their downstream sides so they may feed the redundant isolation valve pairs controlling distribution to the four end uses. A common regulated pressure has been specified because this is the simplest and lightest arrangement. The total system is expected to be designed at one time so all the individual units could be optimized together. It is not anticipated that "off-the-shelf" components will be available to drive the design. The EPS fuel cells will probably further reduce the pressure from that supplied by the system shown. A nominal regulated pressure of 500 psi is envisioned. This pressure will likely be determined by the needs of the pneumatic actuators and/or the ACS thrusters. The APU unit depicted between the two tank pairs and will be discussed separately in the following paragraphs.

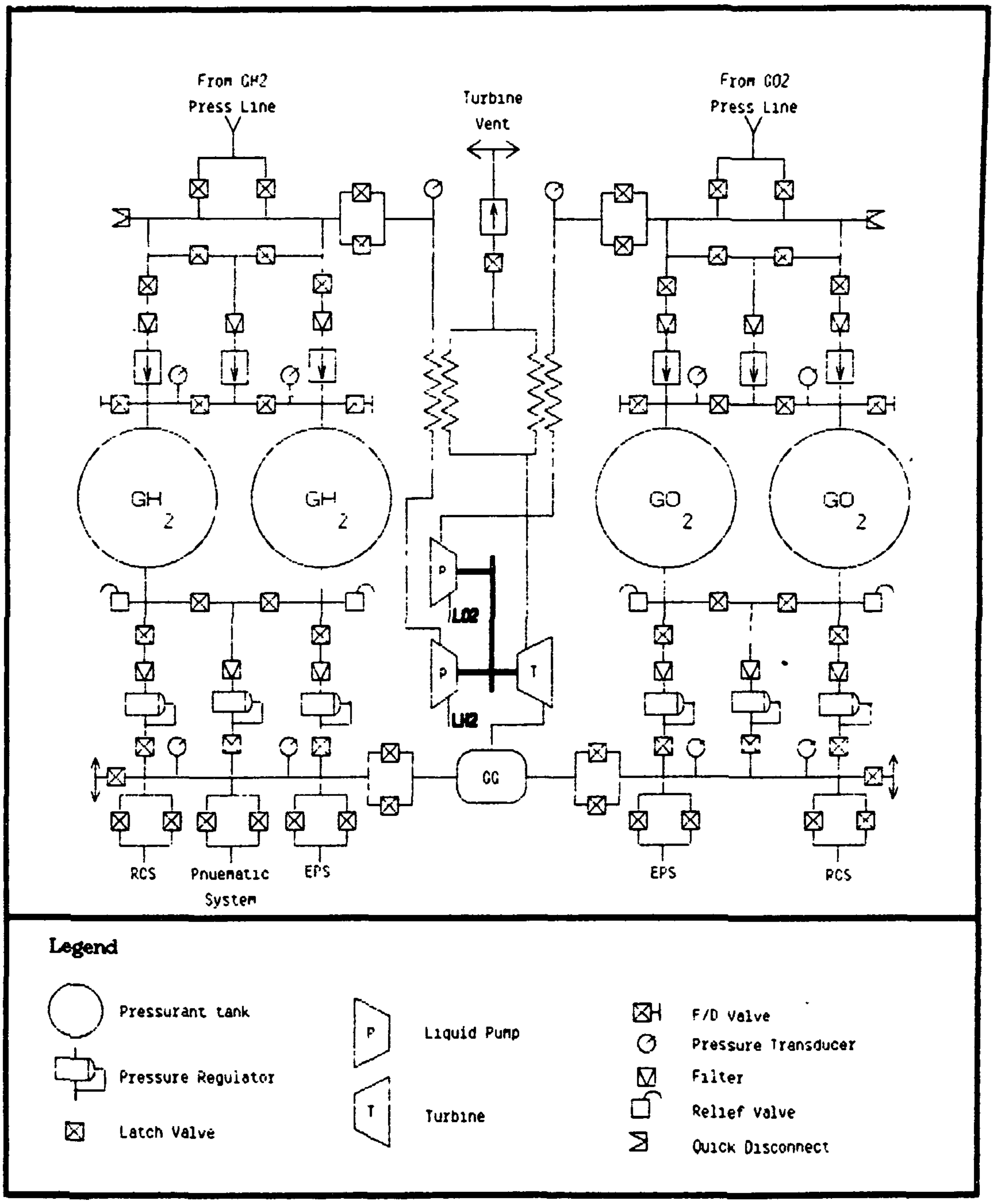

Figure 3.4-16 GH_2/GO_2 Subsystem Schematic

Nominal operation of the accumulator system will be as follows. At the beginning of a mission, at least one of the tanks for each gas will have a pressure sufficient to start the APU and allow activation of the engine valves. Normally, the APU would not be used to charge the accumulators. The firing of the main engines will accomplish this. The APU would be used as a backup for the main engines to sufficiently charge the accumulators to enable returning the OTV. Each main engine firing following the first will keep the accumulators properly charged. A long stay at GEO may require that the APU be started to recharge the accumulators. On returning to the space station, the accumulators could be offloaded to a safe blanket pressure. Alternatively, they could be run down to a safe pressure just before docking with the space station. In either case, they are not expected to require complete evacuation, except when the system needs servicing. Routine OTV turnaround would not require that they be empty, only that they be at a safe pressure. Evacuation after each mission would require a lot of time and energy in handling. In the space environment a small leak will not constitute a safety hazard in itself. Rather, because a small leak will need to be repaired to prevent the loss of propellants, the blanket pressure will provide the gas needed to detect the leak. Once detected, however, the leak repair may require evacuation of the system. For this reason, the space station may require a means of charging the accumulators back to the necessary premission conditions. Potentially the APU could also provide this if the gas generator could bootstrap itself. A schematic for the APU is shown in Figure 3.4-17.

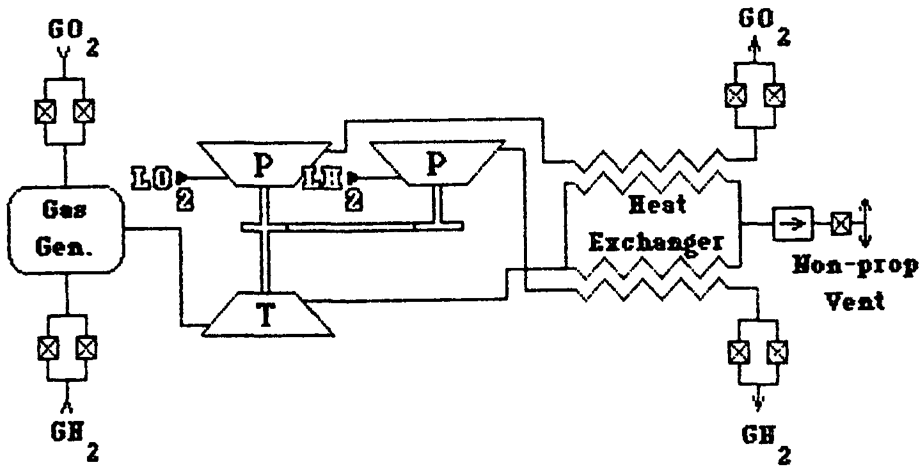

Figure 3.4-17 Auxiliary Power Unit Schematic

Gaseous propellants from the accumulators are combined in the gas generator from where they are expanded through a turbine and twin heat exchangers before being released overboard. Liquid propellants from the main engine feedlines are pumped to a high pressure and vaporized in the cold side of the twin heat exchangers before being introduced into the accumulators. The size of the APU will depend on its intended use. If it will serve as a backup for the main engines, its reliability will need to be high. Redundant APUs may be needed in this case. Steady-state operation vs duty cycle operation will also affect its ultimate size. The expected worst case is where the APU is used with the ACS as a backup for a failed main engine(s). The primary mass use would be for the ACS propellant but additional propellant will be needed to provide tank pressurization and to power the APU. However, because of the difficulty in building a very small turbopump unit, a better compromise between unit mass and efficiency may well be a larger, more efficient unit running at a low duty cycle. The following first-level analysis explores this tradeoff.

The APU is to be used primarily as a backup system for a failed main engine. As such, a premium will be placed on its performance. As currently envisioned, the APU exhausts its products nonpropulsively and thereby "wastes" them. The design must therefore make the most out of what it uses. This means minimum mass use for the APU. However, dry weight must also be kept to a minimum. A balance must be struck between the APU dry mass and the mass consumption. A target weight of 50 lbm was used. This is felt to be a reasonable penalty for the OTV to pay for the flexibility gained by the APU. Approximate masses are shown in Table 3.4-2, based on a steady-state APU design.

Table 3.4-2 APU Mass Estimates

Component	Mass, lbm
Gas Generator and Valves	8
Turbopump Unit	22
O_2 Heat Exchanger (HX)	3
H_2 Heat Exchanger	12
Miscellaneous	5

Individual performance will be covered in detail in the following paragraphs. The design presented below is somewhat conservative because it is based on a first-order analysis. An EPS use rate of 0.85 lbm/h for 48 h plus 78 lbm of ACS propellant at a 5:1 MR was used to size the accumulators. The mixture ratio through the cold side (liquid pumps and heat exchangers) of the APU was sized for steady-state ACS usage. The hot side (GG, turbine and HXs) was sized on a per lbm/s of cold side flow basis so some scaling could be performed.

The APU cold side consists of the liquid pumps, the heat exchangers and the associated valving. Flow rates, heat fluxes, temperatures, pressures and power requirements are shown in Table 3.4-3. For the steady state design point shown, the pumping requirements are quite modest. Liquids at or below the saturated temperatures were used for the inlet conditions to the pumps. Vaporization was assumed to take place entirely in the heat exchangers. Accumulator inlet conditions of 200°R and 735 psi were chosen to minimize the heat flux needed. Efficiencies of 90% were used for the pumps. The JANNAF tables were used to determine the enthalpy needed by the fluids. From this information, the power needed per lbm/s of cold side flow could be determined. As shown in Figure 3.4-17, a single turbine is used to drive the two pumps to facilitate the analysis. This establishes a cold side mixture ratio. Separate turbines and pumps would allow the mixture ratio to the accumulators to be varied at a cost in simplicity. Future analyses may justify the additional complexity this would entail. Also shown is a gear-driven LO_2 pump. Again this was done for ease of analysis. A detailed turbopump design was beyond the scope of this study.

Table 3.4-3 Cold Side Parameters

		Turbopump				Heat Exchanger				
Fuel	Mass Flow, lbm/s	Ti, °R	Pi, psi	Po, psi	Δh, Btu/lb	Ti, °R	To, °R	Pi, psi	Po, psi	q, Btu/lb
O_2	0.833	165	20	750	1.59	167	200	750	740	11.24
H_2	0.167	38	20	750	5.04	42	200	750	740	98.54

The APU hot side consists of the gas generator, the turbine, the heat exchangers and the associated plumbing. Table 3.4-4 shows the hot side parameters. Gas generator operation at 2500°R and 150 psi was used. The operating pressure was chosen to give a good pressure drop for the injector. The temperature was selected as the highest possible without generating excessive thermal material concerns. A higher operating temperature would be desireable to reduce the mass flow needed to condition the cold side fluids. These operating conditions yield a gas generator fuel-rich mixture ratio of 1.35:1. The turbine needs 14.9 hp to drive the pumps, assuming a turbine efficiency of 70%. The heat fluxes shown in Table 3.4-3 size the mass flows to the hot sides of the heat exchangers. The fluxes shown in Table 3.4-4 include 5% extra for losses to the environment from the gas generator and heat exchangers. A heat exchanger exit temperature of 500°R was chosen to avoid freezing the water in the gas generator exhaust products. The resulting heat exchanger effectivenesses are 80% for O_2 and 76% or H_2 based on these temperatures and flows. This implies rather large heat exchangers with both boiling and convective heat transfer. The heat exchanger design was carried no further than needed to estimate weight.

Table 3.4-4 Hot Side APU Performance.

	Gas Generator			Turbopump			Heat Exchanger			
Fuel	Ti, °R	Pi, psi	$\dot{m}$, lb/s	Ti/To, °R	Pi/Po, psi	Δh, Btu/lb	Ti/To, °R	Pi/Po, psi	q, Btu/lb	$\dot{m}$, lb/s
O_2	200	735	0.021	2500 2333	150 113	10.5	2333 500	113 103	11.80	0.004
H_2	200	735	0.016	2500 2333	150 113		2333 500	113 103	103.2	0.034

As mentioned earlier, the design is based on steady-state operation per lbm/s of cold side flow. For four 100-lbf thrusters firing to return the OTV from GEO orbit, a mass flow to the thrusters of 0.847 lbm/s is needed at an I_{sp} of 472 s. The resulting APU mass usage of 0.0315 lbm/s leads to an effective I_{sp} of 455 s at a MR of 4.7. There are several potential means of improving these results. All involve reducing the hot side mass flow. Lowering the accumulator inlet temperature and/or raising the gas generator operating temperature would both be effective. Improving the efficiencies of the pumps and the turbine would also help, of course. However, these have already been chosen close to the physical limits. Only percent improvements could be gained. Running the gas generator oxidizer-rich (MR of 89:1) improves the mixture ratio to 6.4 but reduces the I_{sp} to 376 s. This is a result of the higher molecular mass of oxygen, which may also present material problems. A means of potentially improving the efficiencies and simultaneously increasing the gas generator operating temperature would be to operate the APU at a low enough duty cycle to improve the efficiencies while not incurring transient losses. The accumulators would need to be sized to accommodate this as well as most of the rest of the gaseous subsystem. As a backup mode to a failed engine, using the APU and ACS thrusters offers improved performance (455 s and 4.7 MR vs 450 s and 4.0 MR) over tank head idle mode. Another route to substantial improvement is an alternative means of heating the gases after they have been pressurized. The turbine and pumps use only 8% of the power used to condition the gases. A solar-powered evaporator that supplies 100 Btu/s to the fluids could be used although it would be quite large and massive. The APU will justify itself only as a backup for totally failed main engines (2-engine case) or where the operational advantages previously mentioned become needed.

3.5 PHASE I CONCLUSIONS AND RECOMMENDATIONS

Based on the configuration and performance tradeoff studies of Phase I, the following major conclusions resulted in addition to the main propulsion system tradeoff study conclusions shown in Table 3.4-1:

1) Investigate the range of vehicle/engine systems further (Phase II),
    a) Implication of reusability/servicing,
    b) Life-cycle cost;

2) Spherical tankage with contingency replacement;

3) Retractable nozzles benefit only with large (15- to 20-deg) gimbal angles;

4) GH_2/GO_2 accumulator system supplying ACS, EPS, pneumatics and contingency APU;

5) Choice of engine number and nozzle design strongly depends on engine performance characteristics and manrating failure criteria;

6) Stageable fleet concept should be given further consideration.

These conclusions remain consistent with the philosophy of the baseline vehicle development of Phase I.

4.0 PHASE II - SERVICING, RELIABILITY AND LCC

This section describes the space based OTV servicing scenario, engine configuration reliabilities and life-cycle costs (LCC) for the respective engine configurations.

4.1 MAINTAINABILITY

4.1.1 Baseline Vehicle Servicing Character

The baseline vehicle configuration(s) developed during Phase I were used in defining an appropriate approach to servicing a space-based reusable orbit transfer vehicle. The objective was to develop a reference servicing operational scenario for the life of the OTV that appears reasonable considering life-cycle cost (LCC), logistics, space station facilities, technology availability, etc. Relevant tradeoff studies were then conducted to verify that the reference scenario is the lowest LCC approach.

A reasonable approach to servicing a space-based reusable OTV onorbit involves more than simply modularizing the vehicle so everything imaginable may be easily removed and replaced. Modularization seems to be a key word in the rhetoric surrounding the issue of space servicing; however, the understanding of the word leaves much to be desired. In examining the baseline vehicle configurations, one finds items that should be included in a simplified servicing approach of removal and replacement at the end of a reasonable service life or following a failure of some kind. These items include:

1) Avionics;

2) Batteries or fuel cells;

3) ACS thrusters or cluster modules;

4) Main engine(s).

These subsystems/components lend themselves to "modularization" either because of their inherent geometric peripheral packaging, their character of being at the "edge of things" schematically, or their simple interfaces. Also, and maybe coincidentally, these appear to be items that have shorter expected lifetimes than such items as structure, tankage, isolation valves, etc. So it would seem that providing for easy servicing of these short-life items would be adequate. In further investigation, however, this approach of designing the OTV for long life with the exception of the easy service items makes little sense. For instance, if a component other than those intended for easy serviceability were to fail, the option of returning the "space-serviceable" vehicle to earth for unscheduled servicing does not appear attractive from a cost standpoint. Typical components/subsystems that are not easy-access items for simple remove and replace operations include:

1) Main tankage that may suffer meteoroid penetration and require replacement;

2) Thermodynamic vent system that may include moving parts and require replacement (internal to main propellant tanks);

3) Multilayer insulation that may degrade as a function of time on-orbit or may degrade as function of meteoroid penetrations (that may leave microscopic holes on the exterior but literally shred the underlying layers);

4) Valves, filters, regulators, vents, etc that are part of the main propulsion feed and pressurization systems; and also the accumulator system that feeds the ACS, EPS, and pneumatics (that ideally would be welded into the system).

These typical items are not necessarily buried within the propulsion system in all cases. However, their integration in the propulsion system provides difficulty in arranging them for individual removal/replacement or servicing in the space environment. The ability to detect degradation or failures in each of these items separately is also a significant design problem. Some compromises must be made as to the design of the vehicle for modularization (and proper degree thereof), component redundancy, fault detection and isolation, component design lifetime, and service intensity requirements.

The first step with the baseline vehicle configuration was to examine the design's components and subsystems for estimated lifetimes, failure modes, failure effects, and failure frequency estimates. Grouping the components into reasonable modules for contingency removal and replacement in addition to defining the module interfaces, geometry, and servicing character was essential for further development of the appropriate servicing scenario. Therefore a failure modes and effects analysis was performed for the baseline design as a starting point in addressing the maintainability aspects of the OTV.

4.1.2 Failure Modes and Effects Analysis (FMEA)

A FMEA was performed to analyze the proposed OTV propulsion system concept. Although the data available were not sufficient to perform a complete FMEA, this method of analysis was used to further refine the design by considering the effects of incipient failure modes and to provide input to recommend replacement intervals and component grouping arrangements for remove-and-replace operations.

The FMEA was structured according to the functional block outline illustrated in Figure 4.1-1. Wherever possible and considered appropriate, the OTV propulsion system was broken down to the subassembly level.

1.0	Propulsion System	1.2	ACS and Accumulator	1,3	Main Propulsion	1.4	Avionics and Power
1.1	Aerobrake	1.2.1	ACS	1.3.1	Main Engine Assemblies	1.4.1	Guidance and Navigation
		1.2.2	APU	1.3.2	Propellant Feed	1.4.2	Communications
		1.2.3	Accumulator	1.3.3	Pressurization	1.4.3	Electric Power
				1.3.4	Tank Assemblies		

Figure 4.1-1 OTV Propulsion System Functional Outline

Design features and operating characteristics for the OTV propulsion system were evaluated to identify the various failure modes. The failure modes were then evaluated as to their potential severity and likelihood of occurrence. These classifications are tabulated.

Failure Severity	
Severity Classification	Severity Category
4	Category I—Catastrophic. Failures that may cause death or system loss.
3	Category II—Critical. Failures that may cause severe injury or major system damage that will result in mission loss.
2	Category III—Marginal. Failures that may cause minor injury or minor system damage that will result in delay or mission degradation.
1	Category IV—Minor. Failures not serious enough to cause injury or system damage, but that could result in unscheduled maintenance or repair.
Failure Likelihood	
Failure Likelihood Classification	Description
1	Low probability of occurrence some time during life of system.
2	Moderate probability of occurrence some during life of system.
3	High probability of occurrence some time during life of system..

It is assumed the OTV will require that the system be designed to fail operational in the event of a single failure mode and fail safe in the event of two failure modes. This assumption was used as a criterion for evaluating the compensating provisions for each failure mode and assigning the appropriate severity classifications. Table 4.1-1 lists the results of the FMEA exercise. Table 4.1-2 lists items identified as catastrophic and critical failures and recommends the design considerations to downgrade the severity classifications.

Table 4.1-1 *Failure Modes and Effects Analysis (FMEA)*

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.1	Aero Brake	Provides dissipation of orbital velocity.	1	Torn Nextel Fabric	Less drag and stability in braking.	Vehicle would have to make multiple aero passes.	2	2
				Damaged Structure	Less drag and stability in braking.	Vehicle would have to make multiple aero passes.	2	1
1.2	APU/ACCUMULATOR							
1.2.1	ACS							
1.2.1.1	Isolation Latch Valve	Thruster Valve backup flow control.	16	Fail Open	No immediate effect	Redundant Valve	1	1
				Fail Closed	No immediate effect	Redundant Valve	1	1
				Internal Leak	No immediate effect	Redundant Valve	1	1
				External Leak	Loss of Propellant	Upstream Valve	4	1
1.2.1.2	Thrusters	Provide attitude control thrust.	24	Fail Closed	No immediate effect.	Redundant Thrusters	1	1
				External Leak	Minor vehicle performance degradation.	Redundant Thrusters	2	2
1.2.2	APU	Contingency and long orbital period accumulator charging.	1	Partial Failure	No immediate effect.	Has no effect on mission, providing one engine is operational.	2	2
				Full Failure	No immediate effect.	Has no effect on mission, providing one engine is operational.	2	1

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Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.2.3	ACCUMULATOR	Stores gases for use by ACS, EPS, and pneumatics						
1.2.3.1	Latch Valves	Flow control.	50	Fail Open	No immediate affect	Redundant Valve	1	1
				Fail Closed	No immediate affect	Redundant Valve	1	1
				External Leak	Loss of propellant.	Upstream Valve, (Redundant Systems)	4	1
1.2.3.2	Recharge QD	Access to Accumulator for Space Station Servicing.	2	Blockage	Prevents recharging.		1	1
				External Leak	No immediate affect.	Back up QD Valve.	1	1
1.2.3.3	Filter	Traps particles entrained in fluid.	6	Blockage	No immediate affect.	Monitor Δ P with Health Monitoring System.	1	1
1.2.3.4	Check Valve	One way flow control	8	Fail Open	No immediate effect.	Back up latch valves to isolate back flow	2	1
				Internal Leak	No immediate effect.	Back up latch valves to isolate back flow	2	3
1.2.3.5	EVA Test Port	Provides access to system for EVA testing.	4	Blockage	No immediate effect.	QD Valve	1	1
				External Leak	Potential to depressurize one of the GH_2/GO_2 storage tanks.	Back up latch valves to isolate leak	2	1
1.2.3.6	Pressure transducer	Telemetry pressure signal.	10	Calibration Shift	No immediate effect.		1	3
				Full Failure	No immediate effect.		1	1
1.2.3.7	Relief Valve	Prevents over pressure of Accumulator	4	Blockage	No immediate effect.	Use other relief valve or vent.	1	1

Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.2.3.7	(cont)			External Leak	Potential to depressurize one of the GH_2/GO_2 storage tanks.	Close appropriate latch valves to isolate tank.	2	1
1.2.3.8	Regulators	Provides constant downstream pressure independant of upstream pressure.	6	Fail Open	No immediate effect.	Redundant regulators and close upstream valve.	1	1
				Fail Closed	No immediate effect.	Redundant regulators and close upstream valve.	1	2
				Internal Leak	No immediate effect.	Redundant regulators and close upstream valve.	1	3
				External Leak	No immediate effect.	Redundant regulators and close upstream valve.	1	1
1.2.3.9	Vent Valve	Allows evacuation of system.	2	Blockage	No immediate effect.	Need only at S/S.	1	1
				External Leak	No immediate effect.	Close latch valve	1	1
1.3	MAIN PROPULSION							
1.3.1	MAIN ENGINE ASSEMBLIES	Combines main propellants (LH_2/LO_2) to produce thrust. Also provides GH_2/GO_2 to recharge accumulators and pressurize main tanks.	2	Full	Minor vehicle performance degradation (need to retarget)	Redundant Engine.	2	3
				Partial		Adaptive Controls.	2	3

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Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.3.2	PROPULSION FEED							
1.3.2.1	Tank Isolation Valve/Flow Control	Tank isolation and PU flow modulation.	4	Fail Open	Performance degradation, loss of P. U.	Valve wide open resistances should be matched.	1	1
				Fail Closed	Loss of mission.	Valves would fail at S/S.	4	1
				Internal Leak	No immediate effect.	Downstream valves.	1	1
				External Leak	Loss of propellant and depending on leak, loss of mission.	None.	1	1
1.3.2.2	Auxillary Control Valve	Flow control of main propellants for refuel QD's and main system vents.	6	Fail Open	No immediate effect.	Used at or near Space Station.	1	1
				Fail Closed	No immediate effect.	Used at or near Space Station.	1	1
				Internal Leak	No immediate effect.	Used at or near Space Station.	1	1
				External Leak	No immediate effect.	Used at or near Space Station.	1	1
1.3.2.3	Refuel QD	Connection point for Space Station umbilical for OTV refueling.	2	Fail Open	Loss of propellant.	Used at or near Space Station isolation valve isolates QD.	1	1
				Fail Closed	No immediate effect.	Used at or near Space Station replace valve.	1	1
				External Leak	No immediate effect.	Used at or near Space Station isolation valve isolates QD.	1	1

Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.3.2.4	Main Vent	Allows venting of main propellants.	2	Blockage	No immediate effect.	Used at or near Space Station.	2	1
				Fail Open	No immediate effect.	Used at or near Space Station.	2	1
				Fail Closed	No immediate effect.	Used at or near Space Station.	2	1
1.3.2.5	Engine Isolation Valve	Backup valve for Main Engine valve, Locks in feed system pressure.	4	Fail Open	No immediate effect.	Verified near Space Station.	2	1
				Fail Closed	No immediate effect.	Verified near Space Station.	2	1
				Internal Leak	No immediate effect.	Back up Main engine inlet valves.	2	1
				External Leak	Lose LH_2/LO_2 fuel.	Recommend latch valve upstream to degrade effect to only losing one engine.	4	1
1.3.2.6	APU Cryo Feed Valve	Controls Flow of LH_2/LO_2 to APU pump inlet.	4	Fail Open	No immediate effect.	Close latch valve downstream.	1	1
				Fail Closed	No immediate effect.	Redundant valves in parallel.	4	1
				Internal Leak	No immediate effect.	Close latch valve downstream of pump.	2	1
				External Leak	Loss of propellant and loss of mission.	None.	4	1

Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.3.3	PRESSURIZATION							
1.3.3.1	Engine Isolation Valve		4	Fail Open	No immediate effect.		2	1
				Fail Closed	No immediate effect.	Redundant Engine(s).	2	1
				Internal Leak	No immediate effect.		2	1
				External Leak	Loss of propellant.	Redundant Engine(s).	2	1
1.3.3.2	Control Valve	Regulates temperature and pressure of Auto Pressure gases before introduction to tanks.	4	Fail Open	No immediate effect.	Redundant Control Valves	1	2
				Fail Closed	No immediate effect.	Redundant Control Valves	1	2
				Internal Leak	No immediate effect.	Redundant Control Valves	1	1
				External Leak	Loss of propellant.	None.	4	1
1.3.3.3	Filter	Catches particulate matter in Auto pressure flow.	4	Blockage	No immediate effect.	Monitor Δ P with Health Monitoring System and redundant filters in parallel.	1	1
1.3.3.4	Tank Isolation Valve	Provides isolation of tanks and Auto pressure system.	8	Fail Open	No immediate effect.	Redundant Valve.	1	1
				Fail Closed	No immediate effect.	Redundant Valve in parallel.	1	1
				Internal Leak	No immediate effect.	Redundant Valve in parallel.	1	1
				External Leak	Loss of pressurization and propellant.	None.	4	1
1.3.3.5	Pressure transducer	Telemetry pressure signal.	8	Calibration Shift	No immediate effect.	Redundant Transducer	1	3

Table 4.1-1 (cont)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.3.3.5	(cont)			Full Failure	No immediate effect.	Redundant Transducer	1	1
1.3.3.6	EVA Test Port	Provides access to system for EVA testing.	4	Blockage	No immediate effect.	Used at Space Station.	1	1
				External Leak	Loss of pressurization.	None.	4	1
1.3.3.7	Pressure Vent Valve		2	Blockage	No immediate effect.	Used at Space Station.	1	1
				External Leak	Loss of propellant.	Add upstream latch valve.	4	1
1.3.4	TANK ASSEMBLIES							
1.3.4.1	LO_2 Storage Tank	Provides storage area for LO_2 fuel.	2	Meteroid Penetration	Minor loss of propellant, potentially more serious if not detected.		3	2
				MLI Deterioration	Unknown at this time.		?	?
1.3.4.2	LH_2 Storage Tank	Provides storage area for LH_2 fuel.	2	Meteroid Penetration	Minor loss of propellant. potentially more serious if not detected.		3	2
				MLI Deterioration	Unknown at this time.		?	?
1.3.4.3	Level Sensors (Hot Wire)	Provides fluid level input to PU Computer	4	Full Failure	No immediate effect.	Inputs from other sensors provide adequate data.	1	2
1.3.4.4	Propellant Util. computer	Extrapolates propellant drain data to flow controllers	2	Full	No immediate effect.	Redundant Computer	1	?

Table 4.1-1 (concl)

Control Number	Component Description	Functional Description	Quantity (Q)	Failure Modes	Failure Effect on System	Compensating Provisions	Severity Classification	Failure Likelihood Category
1.3.4.4	(cont)			Partial	No immediate effect.	Redundant Computer	1	?
1.3.4.5	Fluid Pump & Impeller	Mixes fluid in tanks past TVS Heat Exchanger	4	Pump Failure	Loss of propellant bv venting or burning.	Abort mission and return to S/S.	3	3

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Table 4.1-2 Severity Results and Recommendations Based on FMEA

Control Number	Component Description	Failure Modes	Failure Effect on System	Severity Classification	Recommendations To Downgrade Severity
1.2.1.1	Isolation Latch Valve	External Leak	Loss of Propellant	Catastrophic	Back up isolation valves.
1.2.3.1	Latch Valves	External Leak	Loss of propellant.	Catastrophic	Back up valves.
1.3.2.5	Engine Isolation Valve	External Leak	Lose LH_2/LO_2 fuel.	Catastrophic	Recommend latch valve upstream to degrade effect to losing function of only one engine.
1.3.2.6	APU Cryo Feed Valve	Fail Closed	No contingency mode accumulator operation.	Catastrophic	Design should reflect redundant valves in parallel.
		External Leak	Loss of propellant and loss of mission.	Catastrophic	Consider selecting/developing more reliable components or reconfiguring system.
1.3.3.6	EVA Test Port	External Leak	Loss of pressurization.	Catastrophic	Consider selecting/developing more reliable components or reconfiguring system.
1.3.4.1	LO_2 Storage Tank	Meteroid Penetration	Minor loss of propellant, potentially more serious if not detected.	Critical	Tanks should be verified by testing and fracture mechanics analysis. Also, establish a method of leak detection.
		MLI Deterioration	Unknown at this time.	?	Need to determine effects of deterioration over a prolonged period of time.
1.3.4.2	LH_2 Storage Tank	Meteroid Penetration	Minor loss of propellant, potentially more serious if not detected.	Critical	Tanks should be verified by testing and fracture mechanics analysis. Also, establish a method of leak detection.
		MLI Deterioration	Unknown at this time.	?	Need to determine effects of deterioration over a prolonged period of time.
1.3.4.5	Fluid Pump & Impeller	Pump Failure	Loss of propellant by venting or burning.	Critical	Ensure method of servicability or components reliability exceeds tank life.

The FMEA was used as an input source to estimate the number of missions before recommended component replacement and/or overhaul. These estimations were based on the failure modes, likelihood of failure, and best engineering judgment of realistic, obtainable goals for scheduled replacement to ensure safe operation and availability of the vehicle. Evaluation of the estimated replacement intervals disclosed that the missions to replacement fell near 30 mission intervals for several subsystems (actually from about 30 to 50 missions). Examples of estimated lifetimes are (1) main engines - 30-mission life, (2) ACS thrusters - 40 to 55 missions, and (3) main tankage - 100 to 200 missions.

4.1.3 Component Grouping for Servicing

The preliminary OTV baseline design prepared during the Phase I effort has been further refined to identify orbit replaceable units (ORUs) that enable the OTV to be serviced at the space station. A study was performed to determine what varying degrees of serviceability would imply for the modularity of the entire vehicle. Table 4.1-3 shows the results. Fluid system components were divided into logical groups that should minimize the logistic costs of servicing while maximizing the capability to handle unforeseen failures. All major electronic components were left as separate ORUs, there being no great advantage to a larger grouping. Because no single component has an expected lifetime of less than 30 missions, the grouping of similar life components into modules was not needed as earlier anticipated. Figures 4.1-2 and 4.1-3 show the grouping of fluid system components and Table 4.1-4 depicts the module interfaces, geometry, accessibility and operations anticipated.

The seven modules shown in Figure 4.1-2 for the main propulsion system include all fluid components except the main engine, some transducers and the pressurization system vent valves. Vent valves will likely be permanently installed in the plumbing and located remotely for functional reasons. If their reliability proves low, they will be serviced as single components.

Five modules are envisioned for the GH_2/GO_2 subsystem as shown in Figure 4.1-3. The check valves and regulators are expected to have the shortest life. Thirty missions are shown for these components and they are therefore included as part of large modules. If they do prove troublesome they would be isolated into smaller modules so they could be replaced individually after they have failed. Designing for more and smaller modules is undesirable because of the increased number of fluid connectors necessitated (more potential leaks). The APU is the most difficult item to be serviced because of its many and different fluid connections. Despite its rotating machinery, its life is expected to be at least 30 missions because of its contingency nature (backup for failed main engines or possibly during excessively long onorbit hold periods).

Table 4.1-3 Degree of Onorbit Servicing

Level	Type	Cumulative No. of Items Serviced	Typical Items
1	Resupply	2 to 5 Fluids	LO_2, LH_2, He, N_2H_4, GN_2
2	Subsystem-Level Module R/R, Mechanical & Electrical	5 to 15	RCS, EPS, Avionics, Instrumentation, Etc
3	Subsystem-Level Module R/R, Mechanical, Electrical, Fluid	20 to 50	Main Engine, RCS Thrusters, Fuel Cell(s), Tanks, PCA, Etc
4	Major Component Remove/Replace, Mechanical, Electrical, Fluid	60 to 100	Inlet Valves, Tank Valves, TVS, Turbopump Assembly, MLI, Etc
5	Piece Parts	?	Seals, Bearings, PC Boards, Etc

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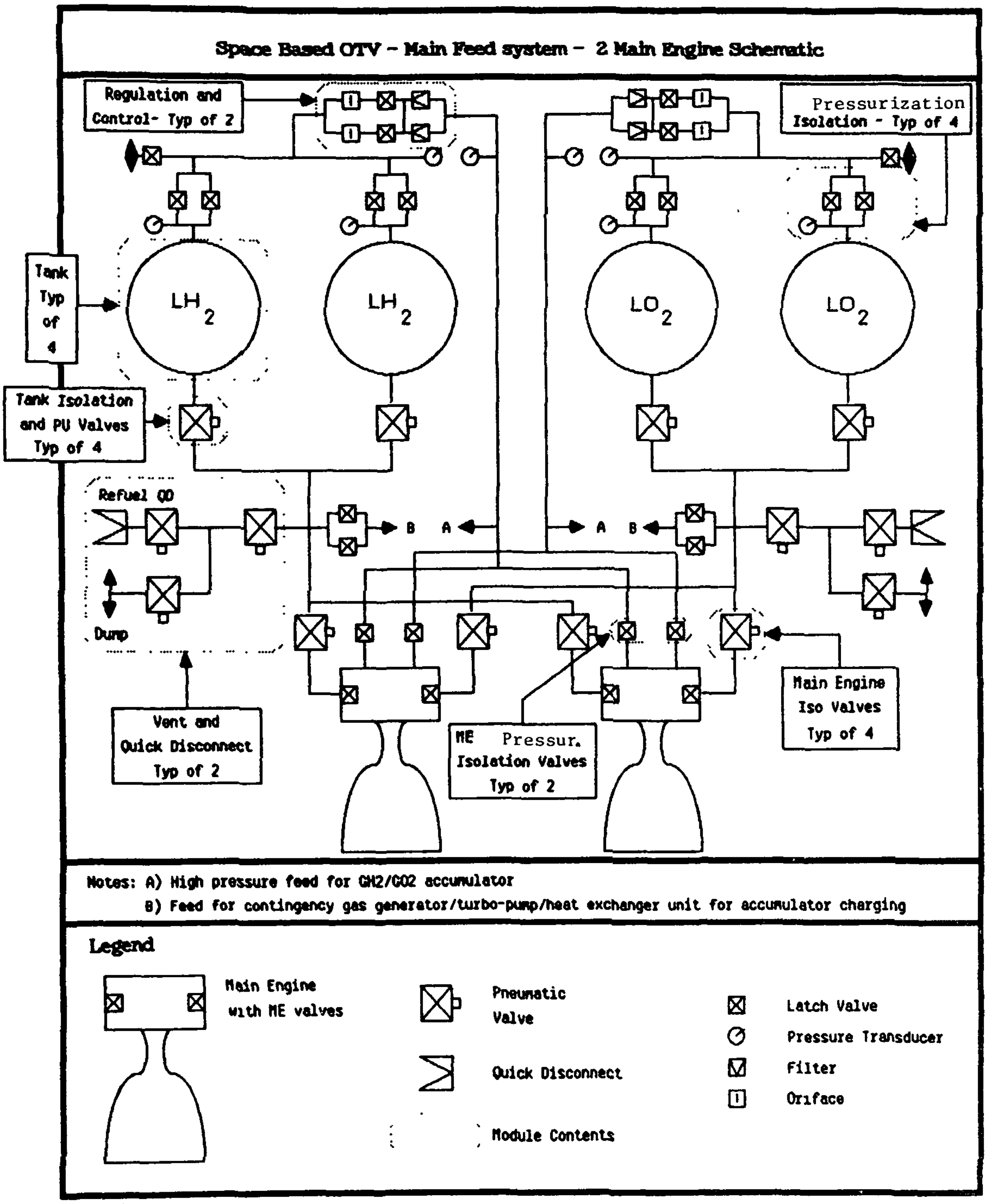

Figure 4.1-2 Main Feed System Modules

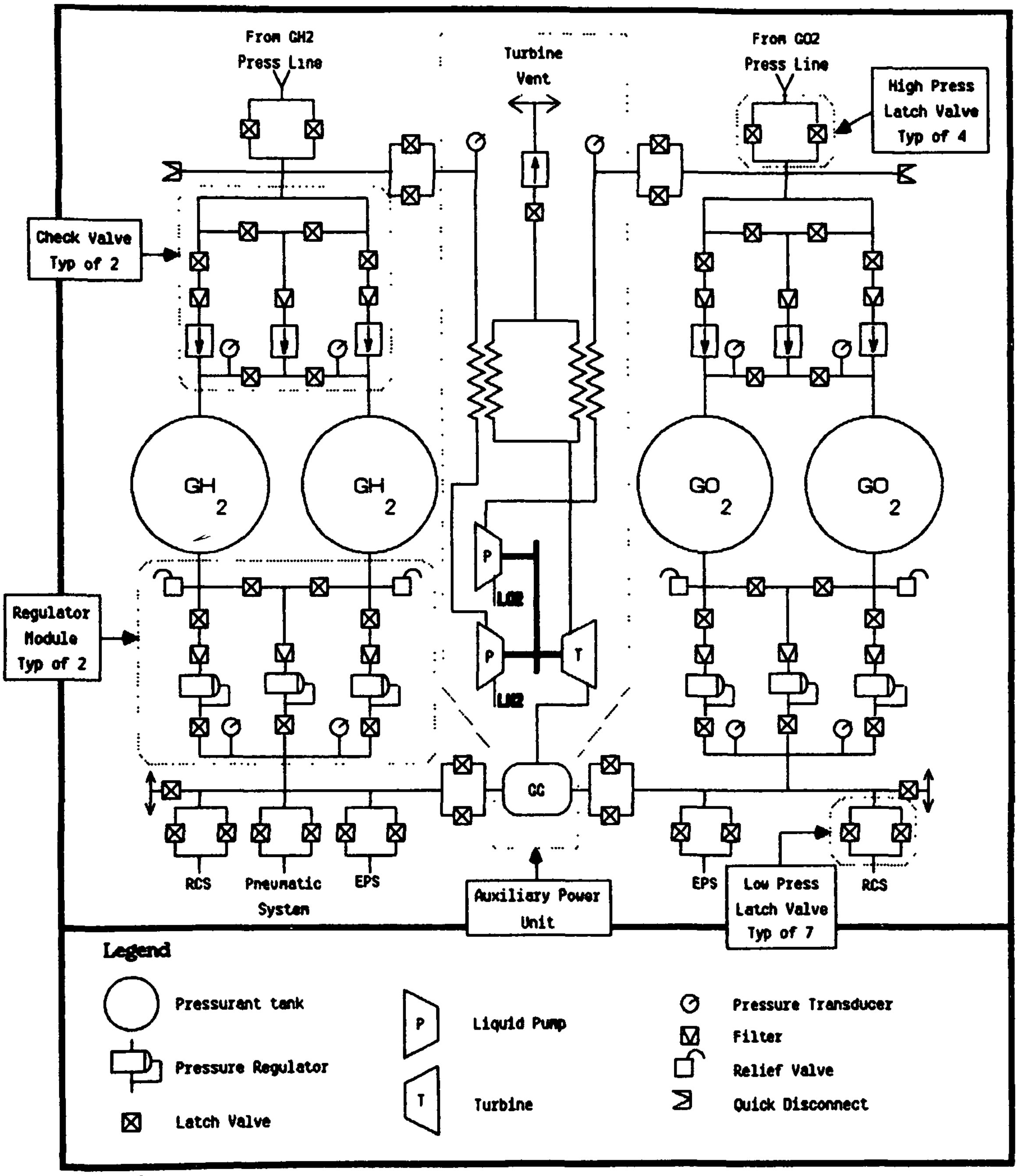

Figure 4.1-3 Gaseous Subsystem Modules

The modules depicted in Figures 4.1-2 and 4.1-3 are further defined in Table 4.1-4. The first column corresponds to the fluid modules identified in addition to the avionics, fuel cells and thruster clusters. The second column defines the number and type of interfaces needed. Rough geometry, access and operations type are shown in the third column. Modules needing frequent servicing such as engines, ACS thrusters and main tanks are given an access rating of 1 (no obstructions) to simplify the remove and replace (R/R) operation as much as possible. The other modules were judged

Table 4.1-4

Table 4.1-4 Vehicle Module Descriptions

Type and Number Of Modules	Type & Number of Interfaces	Geometry, Access,* Packaging, Operations, etc
Main engines (2 to 4)	1 Mech 1 Elec 5 fluid [2 very high pressure for GO_2 and GH_2 autogenous pressures, 1 for GH_2 pneumatics, (high pressure), 2 for LO_2 and LH_2]	60 x 60 in DIA Access - 1 Ops - extensive fluid R/&R
Tanks LO_2 (2) LH_2 (2)	Mech, Elec 1 liquid low pressure, 1 low pressure gas	LH_2 - 11 ft DIA LO_2 - 8 ft DIA Access - 1 Ops - fluid R/&R
ACS (16) 4 valves per thruster	Mech, Elec, 2 high - pressure gas	10 x 5 in DIA Access - 1 Ops - fluid R/&R
APU (1) (gas generator)	2 very high- pressure gas, 2 liquid (low pressure), Elec & Mech	1 ft x 1 ft x 2 ft Access - 2 Ops - extensive fluid R/&R
Accumulator check valve assembly (2)	3 high- pressure gas, Elec & Mech	12 x 12 x 6 in Access - 3 Ops - fluid R/&R
Accumulator pressure regulator assembly (2)	3 high- pressure gas, Elec & Mech	12 x 12 x 10 in Access - 3 Ops - fluid R/&R
Accumulator latch valve set (7)	2 very high- pressure gas, Elec, Mech	6 x 6 x 4 in Access - 3 Ops - fluid R&R
Main pressure control assembly (2)	1 high-pressure gas, 1 low-pressure gas, Mech, Elec	8 x 6 x 12 in Access - 3 Ops - fluid R/&R

* Access:
1 No obstructions
2 Remove access Panel
3 Remove other Items

Table 4.1-4 Vehicle Module Descriptions (Concluded)

Type & Number Of Modules	Type & Number of Interfaces	Geometry, Access,* Packaging, Operations, etc.
Main tank pressure isolation assembly (4)	2 low-pressure gas, Elec, Mech	6 x 6 x 8 in Access - 2 Ops - fluid R/&R
Vent and fill QD assembly (2)	1 low-pressure liquid, 1 high-pressure pneumatic gas, Elec, Mech	3 x 1 ft Dia Access - 1 Ops - fluid R/&R
Tank isolation and PU flow control (4)	2 liquid (low pressure), 1 high-pressure pneumatic gas, Elec, Mech	6 x 8 in Dia Access - 1 Ops - fluid R&R
Engine Isolation valve (4)	2 low-pressure liquid, 1 high-pressure pneumatic gas, Elec, Mech	6 x 8 in Dia Access - 2 (3) Ops - fluid R/&R
Autogenous pressurization isolation valves (2)	2 very high-pressure gas, Mech & Elec	4 x 4 x 12 in Access - 3 Ops - fluid R/&R
Fuel cells (2)	2 high-pressure gas, 1 low-pressure liquid (water), Elec, Mech	1 ft x 2 ft x 2 ft Access - 1 Ops - extensive fluid R/&R
Avionics (4)	Elec, Mech	1 x 1 ft x 6 in Access - 1 Ops - fluid R/&R

* Access:

1 No obstructions
2 Remove access Panel
3 Remove other Items

according to their anticipated physical location. The operation type was determined by considering both the access and interface complexity of the module. Any fluid R/R implies inerting of the plumbing on both sides of the module, the removal and replacement of the module, and finally checkout of the system to verify successful operation. An "extensive fluid R/R" is used for modules affecting more than one fluid circuit such as the APU or main engines.

Determination of the module component groupings basically played the benefits of few fluid connections against the opposing logistics concern to capture the most components with the lowest launch mass, i.e., replace the entire fluid system as opposed to replacing each component. Further considerations were physical proximity, functional similarity, failure frequency, physical size, and such operational factors as requiring venting of the entire fluid system (the OTV is stored with a residual blanket pressure in the tanks and fluid lines to aid in leak checking and to minimize propellant losses). Examples of the above are tank isolation valves, which are are modules because of their physical location and their replacement as individual units requires the evacuation of only one tank (Fig 4.1-2), the vent and fill Q/D modules, which have only two fluid interfaces, contain four components and are located together on the aerobrake (Fig 4.1-3), and the APU, which does not include the control latch valves because this would require the evacuation of parts of both the GH_2/GO_2 and main propulsion systems. In addition the latch valve pairs are grouped as modules. The low-pressure latch valve pairs are shown mainly because of their number and the inability to group them into a larger set (Fig 4.1-3). Their number allows one spare to back up several potential failures. On the other hand, the high-pressure latch valve pairs are identical for the GH_2 and GO_2 sides and therefore lend themselves to grouping into a larger module, which minimizes the number of fluid connections required. This is important because of the this system's high pressure.

It must be emphasized that the fluid system modules shown are preliminary at this point and should be further refined as a more detailed cost analysis is undertaken to understand more fully the logistic influences. This will aid in determining a cost effective means of providing the necessary spares. A more extensive operational analysis will also refine the necessary human factor considerations, space station facility and service requirements, and estimate the time required to complete the module servicing operations. The component failure frequency rates can be further considered as more accurate component life data are developed, presumably when the actual components are in their development phase.

The results of the FMEA and component grouping include:

1) Recommendations for component and subsystem redundancy;

2) Recommendations for fault detection instrumentation and methods of fault isolation;

3) Recommendations for reasonable groupings of components into remove-and-replace modules;

4) Design considerations for both mission success and vehicle serviceability;

5) General maintenance philosophy.

4.1.4 Space-Based OTV Servicing Considerations

Space-based servicing of an OTV has been outlined in sufficient detail to arrive at OTV and support system servicing requirements. The space station facilities needed and their functional requirements have been identified. The impact of logistics and space-serviceable design on the OTV design has been detailed.

Using the space station (SS) as a launch and refueling platform will allow the decoupling of the space transportation system (STS) earth to low earth orbit (LEO) and the OTV LEO to geosynchronous equatorial orbit (GEO) legs of payload delivery to GEO. The shuttle will no longer be forced to launch in a window dictated by the payload delivery, but rather on a periodic basis that would allow optimization of ground resources for routine flow. The burden of meeting the launch window then falls upon the SS/OTV system. This implies the need for a highly dependable OTV and OTV support system if the launch windows are to be reliably met.

The OTV support system will in part consist of SS facilities capable of performing routine maintenance and certain contingency repair procedures. It will also need an efficient logistics function to provide needed spares and consumables in a cost effective, timely manner. Implied by this is a highly developed health monitoring system for the OTV and its subsystems. This system must be capable of diagnosing items in need of attention early enough so the necessary preventive action can be scheduled and lengthy downtimes avoided. All this is made very challenging by the fact that the SS will be able to provide only very limited manned support because of the restricted number of men available, the difficulty of working in the space environment, and the demands of other SS activities.

Because none of the hardware actually exists, it was necessary to make a few assumptions and establish sensible ground rules to allow the study to proceed. These are shown in Table 4.1-5 and are briefly discussed. Because the objective of this study was to identify engine impacts with regard to servicing, detailed design of the SS support facilities, etc was not attempted. For instance, all refueling operations are assumed to be performed on the SS instead of at a remote propellant farm. Operationally, the only impact is on the timeline. The operations to be performed remain similar. The major assumptions show up in Table 4.1-5 while many of the smaller assumptions will be noted in the text as appropriate. The study ground rules were use of the space station as the OTV base, STS shuttle as the launch vehicle, manrating of the OTV, LO_2/LH_2 propellants, and the use of an aerobrake with a low lift-to-drag ratio. From a servicing standpoint, LO_2/LH_2 propellants, manrating, and the aerobrake present the greatest drivers. While the aerobrake itself may not need much servicing, a fixed aerobrake restricts OTV maneuvering about the SS, drives hangar design, and complicates engine servicing. Manrating implies a high degree of reliability/redundancy which in turn affects the integrity of servicing operations. The LO_2/LH_2 propellants have a major impact on the propellant storage and transfer systems and to a lesser extent on the engine servicing requirements. Principally the latter was concerned only with engine changeout implications and the required health monitoring system and its requirements.

As previously mentioned, all space-based OTV servicing was assumed to be at the SS and a means to maneuver the OTV about the SS was to be provided. Specifically, the hangar and refueling depot were assumed attached and controlled from a permanent OTV control station at the SS. The OTV control station will control all OTV-related operations--data handling, refueling, line-of-sight (LOS) proximity operations, maintenance scheduling and procedures [except extravehicular activity (EVA)], and SS inventory control. The OTV was assumed to be under ground control for the LEO-GEO-LEO phase of the delivery missions. Both the baseline Rev 6 mission model and the SS mission model (Ref 5, Vol 3) indicate an OTV launch frequency of one every two weeks to one month. Therefore, a two-week turnaround was used as the ground rule.

Given the above assumptions and ground rules, the general OTV servicing flow was estimated as shown in Table 4.1-6. From this list of operations, those pertinent to engine and OTV servicing were further broken out to allow an operational and functional analysis that revealed the SS facilities needs and the engine servicing impacts. These were used as a baseline against which alternative servicing concepts were explored/evaluated. Also, contingency operations such as unscheduled maintenance are discussed relative to the impact on the baseline functional flow.

Table 4.1-5 Space-Basing Study Assumptions

Space Station to provide up to 4 men/day , 8 hrs/day
Hangar, refueling area, and storage facilities attached to Space Station
OTV ground controled except when within Space Station line-of-sight
2 EVA/1 IVA men per EVA operation
Routine tasks automated as much as possible
Space Station to provide storage for necessary spares
OTV missions to average two week intervals
The OTV to be moved about SS by SS RMS(s)
Times for "routine" operation, not development period
OTV RCS to provide OTV control for "prox ops"
Payload mated to OTV prior to OTV refueling

Table 4.1-6 OTV Mission Flow

Operations	Timeline
Premission Operations	- 7 Days
– Payload and Payload ASE Delivery to Space Station by STS	- 7 Days
– Uplink Mission-Specific Software to Space Station Computer	- 7 Days
Mission Preparation Operations	- 2 Days
– Move Payload to Hangar–Perform Payload Checkout	-24 h
– Move OTV to Hangar	-20 h
– Verify OTV Readiness–Update OTV Mission Computer	-19 h
– Mate OTV and Payload–Verify OTV/Payload Interfaces	-18 h
– Move OTV/Payload to Refueling Area	-17 h
– Secure OTV/Payload and Connect Umbilicals	-16.5 h
– Perform Propulsion System Check	- 8 h
– Chill Down and Fill Main Tanks with Required Propellants	- 7 h
– Resupply RCS and Pressurant (If Necessary)	- 6 h
– Disconnect Umbilicals–Release OTV/Payload/Deploy from Space Station	- 1 h
– "Small" RCS Burn to Separate OTV and Space Station	- 0.5 h
– Pass Mission Control from Space Station to Ground	0.0 h
Perform Mission, Return to Space Station Line-of-Sight	0 to 3 Days
Postmission Processing	3 to 8 Days
– Pass Mission Control from Ground to Space Station	80.0 h
– Safe Main Propulsion at TBD Miles from Space Station	80.5 h
– RCS Burn to Space Station Rendezvous	81.5 h
– Space Station Capture of OTV/Payload, Berth at Refueling Area	81.5 h
– Connect Umbilicals and Offload Propellant Residuals	82.0 h
– Downlink OTV Mission Data to Space Station and Ground Computers	83.0 h
– OTV Exterior Visual Inspection	83.0 h
– Perform Postmission Propulsion System Checks	86.0 h
– Disconnect Umbilicals and Move OTV to Hangar	87.0 h
– Demate Payload (If Attached)	96.0 h
– Prepare Payload for Ground Return (If Necessary)	100.0 h
– Space Station and Ground Computers Return OTV Status and Service Requirements	120.0 h
– Perform OTV Service As Required	122.0 h
– Prepare OTV for Storage and Move to Storage	168.0 h

A "top-down" approach was first used to divide up the nominal two-week turnaround so the maximum available time to do tasks could be delineated. Next, specific individual tasks were considered "bottom-up" in that actual times and equipment needed to perform comparable tasks on the ground were determined. In this fashion, areas of further research were identified. For the purposes of this study, the shorter of the two times was used to assemble the timelines shown. Included with the operational analysis are columns indicating facility needs, intravehicular activity (IVA), EVA and delta time.

Tables 4.1-7 and 4.1-8 indicate tasks, facilities, and time data for the baseline OTV turnaround for vehicle servicing and engine servicing respectively. Complete mission turnaround is shown to take approximately 10 days. This is driven primarily by the LEO-GEO-LEO time and the OTV postmission processing.

Table 4.1-7 OTV Servicing Overview

			Delta Time	Space Station (SS) Manhours		
Operation	Facilities	Tools		IVA	EVA	Total
Main Propellant—Resupply	Resupply Area, RMS Cryo Tanks, Umbilicals	Resupply Software, Control System	7.0	7.0	--	7.0
Avionics—Scheduled Maintenance	SS Hangar	--	6.0	6.0		12.0
– Module Test	SS Computer	Test—Access	3.0	3.0	--	3.0
– Module Replacement	EMU, HPA, Lighting	LRU ASE, Removal	3.0*	3.0	6.0	9.0
– ACS Update	SS Computer	SS & ACS Software	0.5	0.5	--	0.5
Avionics—Health Monitoring Maintenance	SS Hangar	--	--	--	--	--
– Module Replacement	EMU, HPA, Lighting	LRU ASE, Removal	1.0*	1.0	2.0	3.0
– Module Repair	SS Workshop	Electronics	1.0*	1.0	2.0	3.0
Avionics—Mission-Peculiar	SS Hangar	--	--	--	--	--
– Module Replacement	EMU, HPA, Lighting	LRU ASE, Removal	1.0+	1.0+	2.0+	3.0+
– Reconfiguration	EMU, HPA, Lighting	LRU ASE, Removal	2.0+	2.0+	4.0+	6.0+
Tanks—Scheduled Maintenance	Resupply Area	--	2.0	2.0	--	2.0
– External Inspection	CCTV Monitor	RMS + CCTV	1.7	1.7	--	1.7
– PU and TVS System	SS Computer	Test + Access	0.3	0.3	--	0.3
Tanks—Unscheduled Maintenance	SS Hangar	--	--	--	--	--
– Tank Removal	Resupply Area, RMS Console	RMS, CCTV, Tank ASE	2.0	2.0	4.0†	6.0
– Insulation Repair	EMU, HPA, Lighting	Insulation Repair Kit	3.0	3.0	6.0	9.0
– Transducer Replacement	EMU, HPA, Lighting	LRU ASE, Removal	1.0	1.0	2.0	3.0
– PU and TVS System	SS Hangar, EMU, HPA	LRU ASE, Removal	2.0	2.0	4.0	6.0
Tanks—Mission-Peculiar	Resupply Area	--	--	--	--	--
– Tank Reconfiguration	Resupply Area, RMS Console	RMS, CCTV, Tank ASE	2.0	2.0	4.0†	6.0
RCS—Scheduled Maintenance	Resupply Area, Umbilicals	--	1.0	1.0	--	1.0
– Leak Check	SS Computer, Pressure	RCS Software	0.5	0.5	--	0.5
– Transducer Check	SS Computer	RCS Software	0.5	0.5	--	0.5
RCS—Resupply	Resupply Area, Umbilicals	RCS Software	2.0	2.0	--	2.0
RCS—Health Maintenance	SS Hangar	--	--	--	--	--
– Transducer Replacement	EMU, HPA, Lighting	LRU ASE, Removal	2.0	2.0	4.0	6.0
– Thruster Replacement	Hangar, EMU, HPA, Lighting	LRU ASE, Removal	2.0	2.0	4.0	6.0
Structure and Aerobrake—Health Monitoring Maintenance	--	--	--	--	--	--
– Aerobrake Refurbishment	Resupply Area, EMU, MMU	Aerobrake Repair Kit	3.0	3.0	6.0	9.0
– Structure Repair	SS Hangar, EMU, HPA	LRU ASE, Removal	2.0	2.0	4.0	6.0

Note:

* Mission average expected for all avionic modules, 1 hour for contingency.

† Tank replacement only for modular OTV; some EVA assistance anticipated.

Table 4.1-8 Engine Servicing Overview

Operation	Facilities	Tools	Delta Time	Space Station (SS) Manhours IVA	EVA	Total
Engine—Turnaround Maintenance	SS Hangar	--	3.4 h	5.4--	--	5.4
– Analysis of Flight Data	SS and Ground Computer	Engine Software	2 Days	2.0	--	2.0
– Lockup Pressure Decay	SS Computer, Refuel	Engine Software	0.5	0.5	--	0.5
– Engine Valve Operation Check	SS Computer, Refuel	Engine Software	0.5	0.5	--	0.5
– Nozzle Visual Inspection	SS Computer, Refuel	RMS + CCTV	0.6	0.6	--	0.6
– Nozzle Extension Check	SS Computer, Refuel	RMS + CCTV	0.2	0.2	--	0.2
– Gimbal Actuator Check	SS Computer, Refuel	RMS + CCTV	0.2	0.2	--	0.2
– Connect Umbilicals	SS Hangar	RMS	0.3	0.3	--	0.3
– Turbopump Torque Check	SS Computer	Engine Software	0.3	0.3	--	0.3
– Ignition System Check	SS Computer	Engine Software	0.3	0.3	--	0.3
– Instrumentation Checkout	SS Computer	Enginr Software	0.5	0.5	--	0.5
– Solenoid Checkout	SS Computer	Engine Software	0.3	0.3	--	0.3
– Disconnect Umbilicals	SS Hangar	RMS	0.2	0.2	--	0.2
Engine—Periodic Maintenance	SS Hangar	--	4.0	4.0	6.0	10.0
– Setup Operations	SS Hangar	Engine, LRU ASE	0.5	0.5	1.0	1.5
– Turbopump Boroscope	Power, Lights	Boroscope	1.0	0.5	1.0	1.5
– Thrust Chamber Inspection	CCTV Monitor	RMS, CCTV	1.0	1.0	--	1.0
– Engine LRU Replacement	Power, Lights, EMU, HPA	Engine, LRU ASE	2.0	1.5	3.0	4.5
– Tool Stowage	SS Hangar	Engine, LRU ASE	0.5	0.5	1.0	1.5
Engine—OTV Engine Remove and Replace	SS Hangar, RMS, EMU, Foot Restraint, Lighting	Engine Fixture, Engine Disconnect, Protective Covers	5.0	3.8	6.0	9.8
– Setup Tools	--	--	0.5	0.5	1.0	1.5
– Attach Engine Fixture	--	--	0.5	0.5	1.0	1.5
– Disconnect Engine	--	--	0.5	0.5	1.0	1.5
– Move Engine to Storage	--	--	0.2	0.2	0.4	0.6
– Pick Up Replacement	--	--	0.1	0.1	0.2	0.3
– Align and Attach	--	--	0.7	0.7	1.4	2.1
– Check/Verify QDs	--	--	2.0	0.8	0.5	1.3
– Store Tools	--	--	0.5	0.5	0.5	1.0
Engine—Unscheduled Maintenance						
– Repair in Hangar	SS Hangar, RMS, EMU, HPA	Above	2.0+	2.0+	4.0+	6.0+
– Repair LRU in SS	SS Hangar, RMS, EMU, HPA	Above plus LRU ASE	3.0+	4.0+	4.0+	8.0+
– Repair on Ground	SS Hangar, RMS, EMU, HPA	Above plus Engine ASE	2.0	1.7	3.4	5.1

The following discussion will cover the OTV mission flow. The "generic" OTV mission was anticipated to begin early with the mission planning activities and other operations by the payload program. The SS begins its preparations two to three days before the payload is delivered by the STS. The payload is delivered a nominal one week early, principally at the convenience of the shuttle, and is stored onboard the SS awaiting premission processing. Facilities for handling the payload are presumed available. Their exact manifestation is immaterial, but should include a means to mechanically restrain the payload and provide dormant power, data handling, and thermal protection.

A day before the mission the payload is moved into the servicing hangar for final checkout operations. No EVA is anticipated, but could be used if the payload had nonstandard interfaces or required some minor contingency repair. For normal operations all premission payload checkout operations will be handled remotely. The four hours of checkout time are primarily to allow for payload operations that may be more economically performed on the SS than on the ground. For example, payloads could be launched without fluids to relieve designing for launch loads.

Following successful payload checkout, the OTV will be moved to the servicing hangar for mating with the payload. A final health check will be made of the OTV and the mission parameters will be loaded into the OTV main computer. The OTV-to-payload interface (I/F) is assumed to be primarily mechanical with a minimal electrical I/F provided. The electrical I/F would be standardized as well as the mechanical I/F. If nonstandard I/Fs were used, the timeline would need to be modified to allow for OTV I/F modification. No fluid I/Fs are anticipated. Two payload I/Fs are implied--one for the OTV and one for the STS. Once mated and the I/Fs verified, the OTV and payload will be moved to the OTV refueling area.

Refueling is performed as the last major operation in the prelaunch flow to avoid bringing a fully loaded OTV into the hangar and to minimize boiloff. This implies a refueling area capable of accommodating the OTV, aerobrake, and payload. The OTV is docked and refueled on the aft end. A fixed aerobrake will complicate the refueling area design. Presumably a door will be provided in the aerobrake to allow the fluid umbilicals access to the OTV fluid interfaces. The refueling operation itself is the subject of much debate and is simplified here into a tank chilldown operation followed by the bulk fluid transfer. Simultaneous fluid transfer is assumed. Nonhypergolic fluids and "no-leak" quick-disconnects (QDs) should allow this. Also, the attitude control system (ACS) propellants and pressurants are resupplied in parallel with the main propellants. Pressurant needs should be minimized as much as possible because of the inordinate costs of resupplying pressurants.

Following resupply, a final OTV checkout can be performed (gimbal actuators, pressure checks, etc). The OTV and payload are then disconnected from the refueling area and deployed from the SS. The timeline shown assumes that the SS remote manipulator system (RMS) releases the OTV and payload combination with a small delta-V relative to the SS. The OTV uses ACS burn to give additional delta-V of about 3 fps, allowing swifter OTV and SS separation. At a safe distance from the SS the OTV control is passed to the ground and the delivery mission begins. An orbital maneuvering vehicle (OMV) could be used to accomplish the same operation.

Space station control resumes following the return of the OTV to a safe area within LOS of the SS where OTV safing is performed. This may comprise venting the OTV propellant residuals. However, this timeline assumes that the cost of propellants is sufficiently important to warrant recovery. Safing would then primarily entail deactivation of the main engines and the ACS if an OMV is used to recover the OTV. The OTV is returned to the SS following safing either by the OTV ACS or an OMV. The OTV is berthed at the refueling area.

If safing were to entail venting of propellants, it may have a major impact on the OTV. Nonpropulsive vents must be provided with the appropriate valving and controls. Venting through the engines would be possible but could impose undesirable characteristics on the engine. Additionally, the resulting thrust would need to be accounted for. An OMV would not be able to do this because the OMV would likely be mated to the aft end of the OTV so its thrust can act through the OTV/payload center of mass.

Postmission processing is essentially the reverse of the premission flow. The residual propellants are removed after docking at the refueling area. Liquid propellants are returned to the SS cryogen tanks and gaseous propellants are recovered for use by the SS. ACS propellants would also be returned to storage to aid in the accuracy of premission loading (mass measurement errors would otherwise accumulate). It may be desirable to leave a blanket pressure of propellant gases in the tanks for structural reasons.

During propellant offloading, the SS data handling system will downlink mission data from the OTV and return the bulk of these data to the ground where they will be processed. Additional data will have already been sent to the ground during the mission. Some data will also be retained by the SS computer to allow SS personnel to begin postprocessing scheduling. Quick data analysis and turnaround will be essential to efficient OTV servicing. The bulk of the analysis software is assumed to reside on the ground because it isn't cost effective to burden the SS computer or personnel with this task. Two days are allowed for the ground to return a preliminary postmission maintenance schedule to the SS. During these two days, the OTV would be returned to the hangar if it still has a payload attached. Otherwise, the OTV would be moved to its storage area.

Because postmission OTV servicing highly depends on what maintenance needs to be performed, the routine servicing flow will be discussed along with a separate discussion of such major contingency operations as engine removal or aerobrake repair. Because crew time is expected to be a valuable commodity, routine operations will be highly automated. The ground processing of mission data will also perform an optimization of servicing tasks and return a timetable detailing the exact operations to be performed. An approximation is only possible now because both routine (every mission) and contingency operations will be interwoven to effect the optimization. This approximation appears in Table 4.1-6 made up of the scheduled maintenance tasks from Tables 4.1-7 and 4.1-8.

While the OTV is still berthed at the refueling area, a propulsion system check will be performed. This check will be in support of ground analysis of flight data to determine items in need of maintenance and to execute tests designed to isolate any anomalies detected in the flight data. The objective of this checkout is to provide early detection of failures that can be remedied in the OTV maintenance to follow. Also, tests that require pressurants will need to be performed here. If the OTV were equipped with removable tanks, the tank operations would be performed in this area.

All maintenance operations will be performed in the servicing hangar after the schedule has been returned. The first operation will be an overall OTV visual inspection. This could be done via EVA, but will likely be done with a closed-circuit TV (CCTV) and monitor. In this case, sufficient mobility must be given to the CCTV to allow it to reach all areas of the OTV. Most likely, only specific areas will routinely be inspected such as the engine nozzles, aerobrake, and OTV exterior. CCTV movement could then proceed in a preprogrammed manner and the crew would only override to inspect questionable areas.

The servicing hangar is expected to provide for more extensive checkout umbilicals than those provided at the refueling area so specific tests of the avionics can be run. All umbilical actuation will be automated to avoid EVA costs. EVA is anticipated only for nonroutine module changeout operations, nonroutine inspection, and other infrequently performed operations it wouldn't be cost effective to automate. After checkout umbilicals are attached, the avionics will be checked via checkout software and equipment carried for this purpose. Any anomalies will be noted and factored into the maintenance schedule relayed from the ground. Any EVA operations would be performed following schedule finalization. EVA module changeout would be performed on all items so identified in the preceding checks. This assumes that the proper modules are already on board the SS and that the modules were designed for EVA replacement. Both of these assumptions will be discussed more completely later. No modules that will require changeout after every mission have yet been identified. If this were the case, this would likely be accomplished robotically using only one IVA crewman, once again to avoid EVA costs. Table 4.1-9 lists the example EVA-replaceable modules. This table includes estimated times and anticipated interfaces. Because ACS modules may involve fluid disconnects, two operations are shown to illustrate the differences. The fluid QDs lengthen the time because of the additional effort required to assure the crew's safety (installation of spill containment shrouds and checkout following installation).

Table 4.1-9 EVA-Replaceable Modules

Subsystem	Module	Contents	Interfaces	R/R Time, hours
Avionics	Main Computer	CPU, I/O Unit, Memory	Mechanical, Electrical	0.5
	TT&C	Antenna(s)	Mechanical, Electrical	0.2
	C&DH	RF Electronics	Mechanical, Electrical	0.5
	Guidance	Gyros	Mechanical, Electrical	0.4
Reaction Control System	RCS Module	Tanks, Valves, Thrusters	Mechanical, Electrical	0.6
	Thruster	REA Valves, Thrust Chamber Etc	Mechanical, Electrical & Fluid	1.5
Electric Power System	Power Supply	Fuel Cells, Valves, Tanks, Heat Exchanger & Pumps	Mechanical, Electrical	0.5
Structure & Aerobrake	Aerobrake	Aerobrake Module	Mechanical	1.0

Two major contingency operations identified are engine removal (which could also be routine) and aerobrake repair. Aerobrake repair is included at this point as a possibility. It is too early to say exactly what aerobrake repair implies or what type of failure it may suffer. Holes could be repaired either by patching or panel replacement. Aerobrake removal to ease servicing would be desirable but isn't a contingency operation. This would be included in overall processing flow near the end of premission processing and the beginning of postmission processing.

Several levels of engine maintenance are identified as detailed in Table 4.1-8. Two types of scheduled maintenance are shown--operations performed after every flight and those performed every 10 missions. These latter operations are more extensive and performed in addition to the regularly scheduled maintenance. They also include EVA operations (turbopump inspection and orbital replaceable unit (ORU) replacement). The engine removal and replacement operation is detailed as well as three possible unscheduled engine repair operations. Unscheduled maintenance could occur on the engine while it is attached to the OTV. This would involve essentially replacement of an ORU that failed prematurely. A removed engine could have a failed ORU repaired in a SS workshop if future analysis showed this to be feasible and cost-effective. Any major repair of the engine will entail removal and return to earth for repair.

The tasks listed are indicative of the types of operations viewed as feasible. When the engine is further defined, the tasks will need to be reevaluated. The turnaround maintenance tasks are to be fully automated so they may be performed with IVA. Because the inspection tasks will need manned involvement, more manhours are assigned to the tasks. These tasks are listed separately from the OTV tasks previously discussed for ease of discussion. They would be fully integrated with the OTV tasks as part of the ground timeline optimization performed to arrive at the appropriate maintenance schedule. If an engine removal were scheduled, the inspection would be eliminated.

An experienced ground crew under ideal conditions (air-conditioned test cell fully equipped with the necessary tools) can remove an RL-10 in about five hours. The EVA crewmen are expected to replace an engine in four hours in the SS hangar. This short time is a goal to make sufficiently efficient OTV turnaround a possibility. It will be necessary to simplify the OTV/engine interface to enable both the engine removal itself and provide the necessary functional integrity to the interface once the engine has been replaced. For this reason, it is desirable to eliminate pressurant-activated components because this eliminates a gaseous QD from the OTV/engine interface. If the propellant tanks are left with a blanket pressure, a set of valves will be needed on the vehicle side. The main engine valves should remain with the engine so they can be serviced after the engine has been removed (possibly on the SS, likely on the ground). The simplest interface design has all QDs aligned along a plane that separates the engine and the vehicle.

This design type would lend itself to remote engine removal, which is a desirable feature. This approach would likely incur a weight penalty relative to an approach that minimizes weight at the expense of requiring EVA assistance. Cost modeling of OTV servicing scenarios is expected to aid in recommending which approach to use.

The functional and operational analysis just presented identified five basic space station facilities that will be needed to support a space-based OTV. The facilities are shown in Table 4.1-10. While the facilities are treated as separate items dedicated entirely to the OTV, in the actual space station they will be general-purpose facilities designed to support the OTV, OMV and other spacecraft designed for SS servicing. At this point, the facilities are separated more for functional reasons than for hardware reasons. The actual SS facilities will probably recombine the functions into units logically arranged as part of the SS design effort. Therefore, the following facility discussions emphasize the needed functions divided functionally. Possible overlaps are included in the individual discussions.

The servicing hangar will house all the necessary items used for servicing the OTV and other spacecraft. It should be a general-purpose facility with some dedicated items specifically for servicing the OTV and the SS OMV because these two spacecraft will comprise the majority of the servicing requirements. A means of mechanically holding the various spacecraft will be needed. A variety of umbilicals will also be needed, mostly electrical. It may also be desirable to provide a pressurant umbilical. Propellants and other hazardous fluids may be handled at another facility. Power for lighting and power tools should be supplied as well as a means of securing the astronaut, his tools, and any other loose items necessary. One current hangar concept (Fig. 4.1-4) involves a translation mechanism for the crewmen and a rotary carriage for the spacecraft. This would allow the possibility of a quasi-EMU (extravehiclular maneuvering unit) in which the EMU (or spacesuit) shares the SS atmosphere through an umbilical carried with the translation mechanism. In this hangar, total portability would not be necessary because a combination of translation and spacecraft rotation will allow access to all portions of the spacecraft.

Because, as with the servicing hangar, many functions of the SS computer system have already been mentioned, they will only be summarized here. Only a small portion of the SS computers' responsibilities will be represented by the OTV activities. The SS computer will function primarily as a link between the OTV computer, ground facilities, and the SS crewmen. OTV data stored during the mission will be downlinked to the ground through the SS computer with a portion being retained for the SS crewmen to act upon (SS safety-related items, for instance). After ground processing, an estimate of the OTV maintenance schedule will be

Table 4.1-10 Space Station OTV Servicing Facilities

SPACE STATION HANGAR - Provides meteor and thermal protection for OTV and payloads. - Provides power, data, command, and pressurant umbilicals - Storage and use of OTV and payload handling cradles - General purpose RMS's, astronaut foot restraint/positioning aid (HPA), tool and LRU caddies
SPACE STATION COMPUTER - Refers to entire SS C&DH system, including ground and S/C links as appropriate - Stores and executes routine servicing tasks, updating as needed from ground - Assumes major portion of task scheduling operations - Assumes major portion of RMS control and other robotics
OTV CONTROL STATION - Used for OTV-P/L control for LOS operations - Allows mission monitoring while OTV-P/L are under ground control - Used to monitor/control OTV refueling operations
OTV RESUPPLY AREA - Provides all mechanical, electrical, and fluid interfaces for OTV main engine and RCS - Provides propellant storage and fluid transfer control hardware - OTV removable propellant tank handling hardware
OTV STORAGE AREA - Provides mechanical Hold-down , dormant power ,and health monitoring - Could provide thermal and meteoroid protection

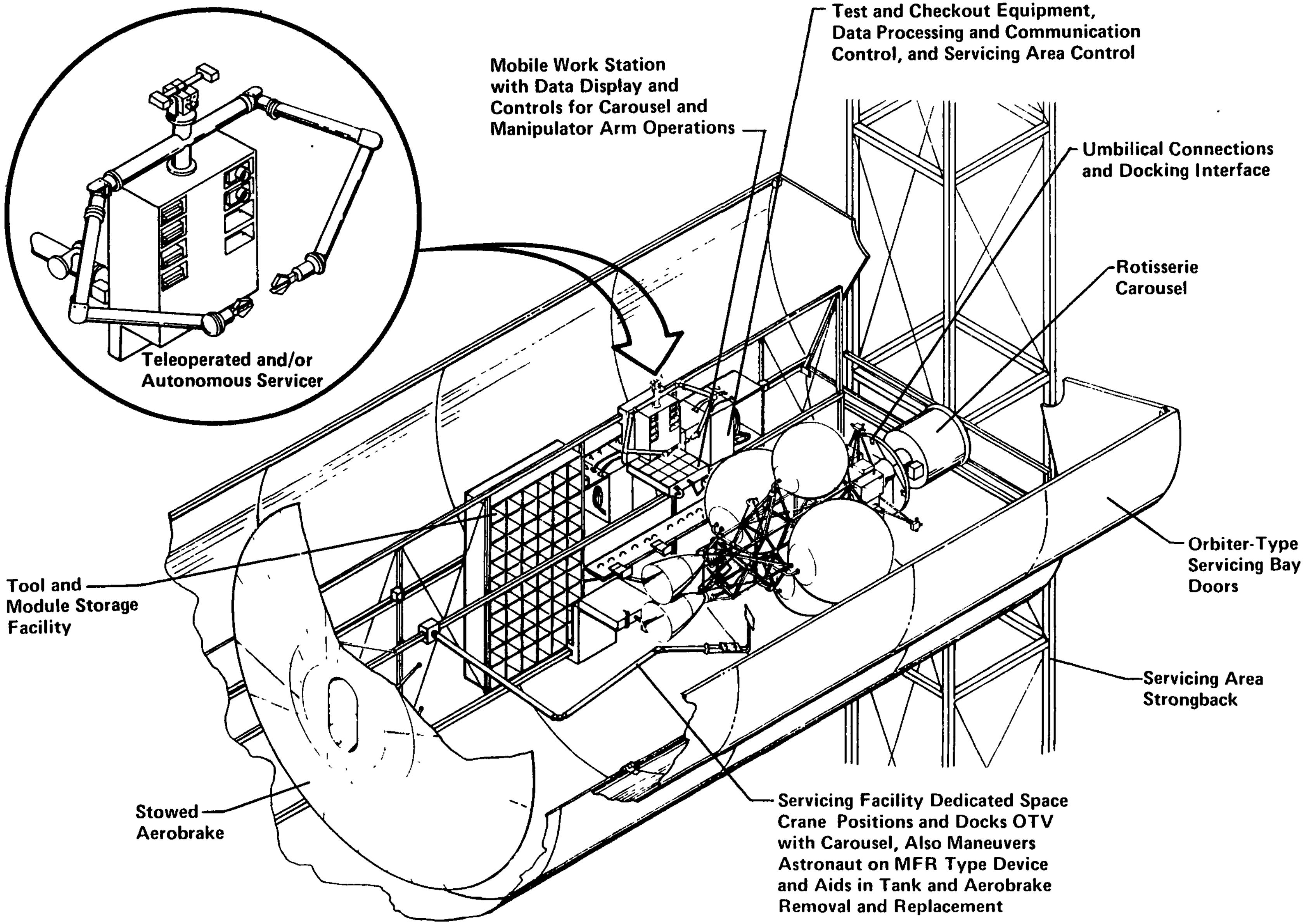

Figure 4.1-4 Space-Based OTV Servicing Facility

returned to the SS. The SS computer will then factor in maintenance tasks discovered during postmission processing of the OTV and prepare a final maintenance schedule. The SS computer will also handle loading of the OTV computer with mission-specific data before the OTV mission. Part of the SS computer will also handle control of the many automated servicing mechanisms. These will include the SS RMS(s), refueling, and CCTV movement.

The above-mentioned functions may more logically be part of the OTV control station. Certainly items that are entirely OTV-specific will be functions of the OTV control station. The major item here is OTV refueling and OTV LOS control. The SS computer will probably just monitor safety-related items so it can respond properly if an emergency should occur. The bulk of the OTV-related software and systems will reside in the OTV control station (functionally at best). The OTV control station will be the primary man/machine link between the OTV and the SS crew. Several OTV display and equipment controllers will be logically arranged to enable efficient IVA control of the various phases of the OTV mission. The OTV control station, as with the servicing hangar, will probably share hardware with other spacecraft. That, however, is a space station issue.

The OTV refueling area will work closely with the control station. The primary function here is, obviously, refueling of the OTV. However, several other propellant- and fluid-related functions will also be accomplished here. Because the refueling area will represent a significant portion of the SS mass, its location will be critical to SS control. The disturbances caused by the propellant transfer will also need to be accommodated.

The refueling area will house the cryogen tanks, an OTV mechanical interface, and the necessary umbilicals to allow refueling of all propellants and pressurants. An electrical umbilical is also necessary to allow control of the OTV and the downlinking of OTV data stored during the OTV mission. It is not envisioned that other spacecraft will be able to employ this hardware for their refueling, mainly because of the physical size of the OTV compared to other spacecraft. Another refueling station will likely be provided by the SS for these smaller spacecraft. (They are also likely to require earth-storable propellants, not cryogens.) Spacecraft wishing to use this facility will accommodate the OTV and not vice versa. All the necessary control hardware will reside here (valves, pumps, plumbing, etc) while the control software will be housed at the OTV control station. One or more CCTVs will be necessary if the refueling area is not visible from the control station.

The space station will need to provide some sort of storage facilities for both the OTV spares and the various payloads. These facilities will at least provide mechanical holddown and minimal power and data interfaces to sustain the vehicles in a dormant mode. Desirable features would be thermal and meteoroid protection. The servicing hangar could provide all of these at a loss in utility. These are, of course, space station issues. However, they are worth some discussion because there are several possible modifications of the baseline timeline. For instance, payload and OTV mating could be performed at the storage area if the proper alignment capability existed. The payload checkout could also be performed there, saving time and minimizing the movement of masses about the SS thereby saving SS propellant.

As an aside, this brings up the subject of the multiple payload interfaces necessary on the payload that it otherwise wouldn't need. The STS interface now involves trunnion fittings and an electrical umbilical. The OTV, on the other hand, would require some sort of axially acting mechanical interface and an electrical umbilical separate from that used for the shuttle. Presumably one of these two interfaces could be used by the space station storage facility. A tradeoff exists between requiring the payload to supply these interfaces and scarring either the shuttle or OTV to eliminate one of the interfaces. Because the payload is launched only once while the STS and OTV make multiple trips, the mass penalty may be best assigned to the payload. This is a subject for further study.

The timelines discussed so far are for a routine mission where no major failure has occurred that requires a delay to allow the STS to bring up the needed spares or, worse yet, return of the OTV to the ground for extensive servicing. Very few missions are likely to be "routine" and may well require delays that affect the baseline timeline. The learning curve is likely to extend through much of the "routine" mission time frame of the early to late 1990s. A fully debugged OTV/SS system by 1994 is unrealistic and an operational OTV by then is an ambitious goal. However, all the mission analyses to date suggest large payoffs for the ability to fly LEO-GEO missions on a two-week schedule. A case for an OTV fleet is emerging.

The other response to downtime impacts is a sufficient spares inventory at the SS to avoid the majority of the delays. Because failures are by nature unpredictable, this implies storing many spares that may never be needed. Unnecessary spares cost both in launch mass for the spares and in the mass of the facilities needed to house them. As a part of the evolving SS and OTV, a comprehensive inventory management effort is recommended that will simultaneously minimize the required mass at the space station and the downtime incurred by the OTV. This would entail a high-reliability OTV coupled with a component-by-component failure analysis to pinpoint likely failures. In addition, grouping the high failure items so they may be replaced as a unit(s) is required. The OTV design must adhere to this modular philosophy to some degree. One spare unit capable of remedying several failures will be very valuable. A reusable space-based OTV cannot be optimized alone; rather the OTV and its support system should be optimized.

4.2 SERVICING OPERATIONS

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4.2.1 Serviceability and Basing of the Baseline Vehicle

The conceptual design philosophy incorporated in the baseline vehicle (Fig. 4.2-1) reflects the considerations for serviceability based on the nature of the vehicle and its usage. This is evidenced by the central core structure, removable tanks, removable aerobrake, easily accessible avionics and propulsion subsystem modules.

Optimum vehicle serviceability also implies consideration of the space station servicing facilities and the operations required in the servicing scenario. Figure 4.2-2 depicts a space station servicing bay dedicated to accommodating the OTV. The orbiter-like doors open for initial berthing of the OTV following a mission. Both routine and contingency turnaround operations, including aerobrake removal for ease of vehicle inspection and servicing, will be conducted here. The doors may also be opened for radial main propellant tank removal/replacement. However, they can remain closed for most remaining turnaround operations to provide the needed meteoroid protection.

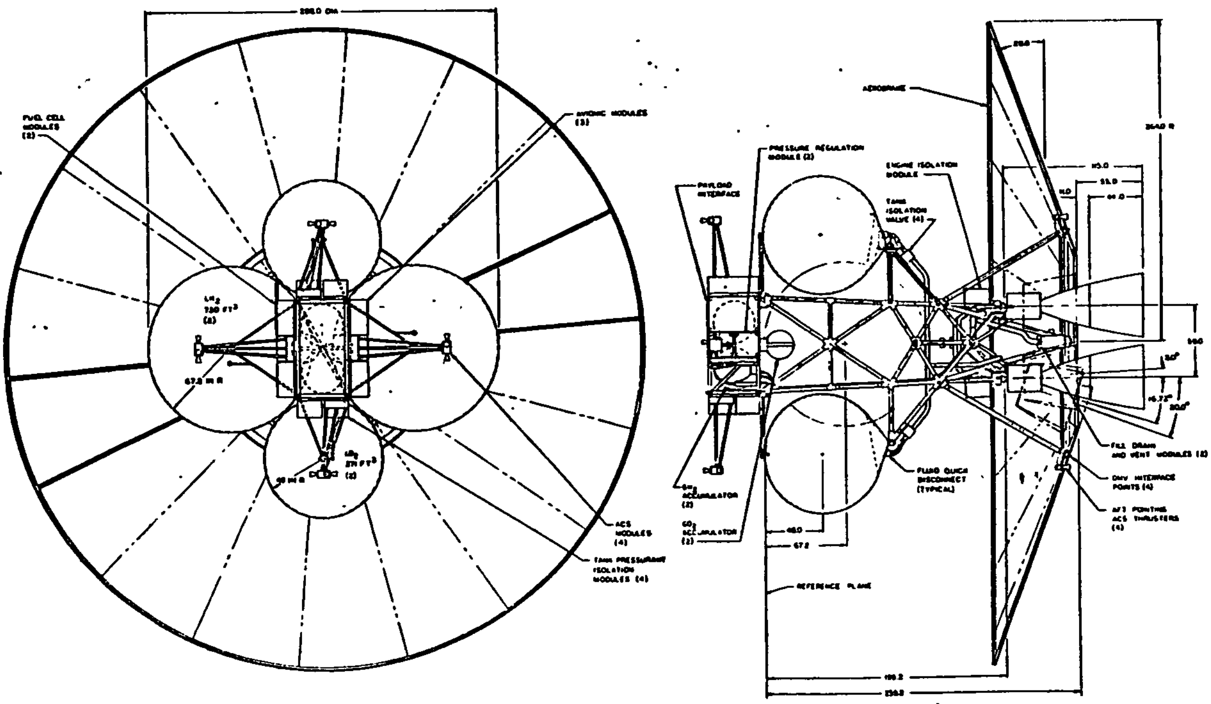


Figure 4.2-1 Baseline (Two-Engine) OTV Layout

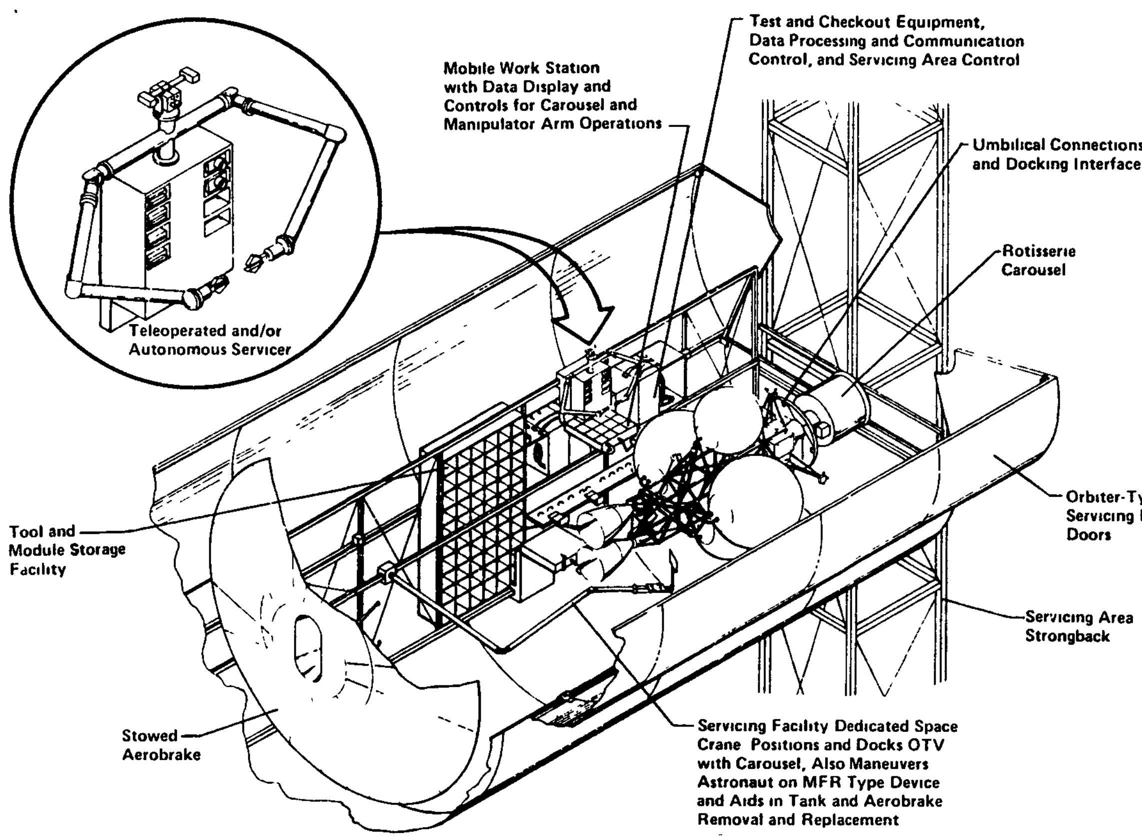


Figure 4.2-2 Conceptual Space Station OTV Servicing Facility

The space crane will be used for:

1) Docking the OTV with the rotisserie carousel;

2) Removal and replacement of tankage and aerobrake (tankage replacement on a contingency basis);

3) Maneuvering an inspection and servicing platform for astronaut use.

Additional intelligent pairs of manipulators for coordinated anthropomorphic operation may be attached to the mobile work platform(s) and translated to areas/subsystems of the vehicle that lend themselves to such automated servicing.

4.2.2 General Servicing Considerations

Servicing will be key to successful space basing of an OTV. Though traditionally defined as "fuel it and fix it," servicing involves much more than this definition suggests, which the succeeding discussion clearly demonstrates.

Technology development and utilization will, in turn, be the keys to successful onorbit servicing. The existing technology is not sufficient in some areas to allow the conduct of servicing operations. New technologies will, however, emerge and, if carefully applied, will support the evolution of satellite servicing capabilities.

Goals and Guidelines for Future Servicing - The goals for OTV servicing in the space station IOC time frame include:

1) Minimization of total system life-cycle cost;

2) Utilization of emerging technology (the state of art is a function of time frame);

3) Development and utilization of systematic technology projections that affect hardware evolution so evolutionary path avoids deadends;

4) Evolution of the servicing mode toward increasingly automated operations.

Other goals will be treated within the context of the detailed discussions that follow. Our reference servicing scenario reflects consideration of all of the above-mentioned goals.

Reference Servicing Scenario Development - Space-based OTV servicing is expected to encompass a wide variety of preflight, postflight, and eventually, inflight servicing operations. These operations have been widely discussed in relevant literature from a variety of viewpoints. Most of the operational descriptions have been rather superficial. Some of the available analyses, however, have attempted to go to a greater level of depth and identify specific service needs and functions for both propulsion vehicles and payloads.

The Teleoperator Human Factors Study (THFS) recently completed for NASA-MSFC, took a different approach and made the assumptions necessary to go to great detail. THFS identified several levels of operational and servicing functions and, through iterative analyses, compiled preliminary lists of low-level (task and subtask) functions we seek to describe as generic, "primitive" tasks. These primitive tasks are the basis for all higher level tasks. The levels of operational and servicing functions identified in THFS are sequence, activity, task, and subtask. These terms are defined in Table 4.2-1.

Table 4.2-1 Levels of Decomposition

Term	Definition	Example
Sequence	A group of goal-directed related activities comprising a major portion of a mission.	Deploy Spacecraft
Activity	A group of goal-directed related tasks that fulfill a limited purpose within a sequence.	Proximity Operations
Task	Basic unit of behavior; smallest logically definable set of perceptions, decisions and responses to complete a task.	Adjust Orientation (Perception of Data, Decision, Control Activation, Feedback of Response Adequacy)
Subtask	Each perception, decision, or response that is a task component.	Initiate Adjustment

The functions at each level were identified by progressive decomposition of scenarios for representative programs selected from the NASA-OAST space systems technology model on the basis of their widely varying natures.

The THFS study and succeeding work will ultimately lead to the identification of a finite set of specific "primitive actions" that, when assembled into complete service operations, can be performed entirely by man, entirely by automation, or by some combination of the two. How the operations will be performed is a function of technology availability; the possibilities range from EVA to IVA, from purely manual to hard automation, teleoperation and flexible automation. Timeline knowledge and knowledge of probable equipment capabilities and configurations, when combined with the knowledge of the composition of the functions, will permit a bottoms-up process of determining the work organization for each servicing function and the overall servicing process.

Figure 4.2-3 presents an overview of this directly applicable hierarchical work organization development approach in an OTV servicing context.

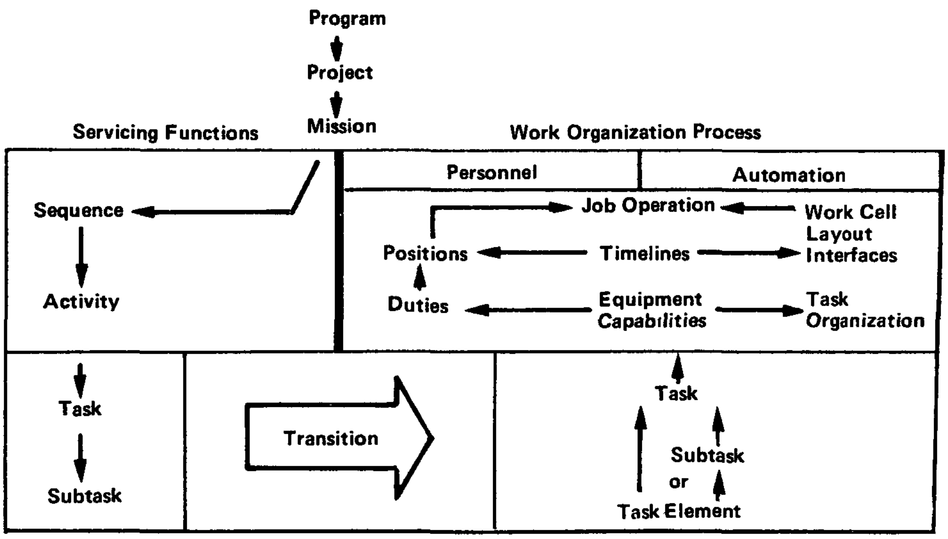


Figure 4.2-3 Work Organization Development for OTV Servicing

The lower level functions, those that go through the transition in Figure 4.2-3, fall into three categories--manipulation, mobility, and housekeeping. These primitive actions are, as stated previously, the building blocks assembled into successively higher level groups--subtasks, tasks, activities, and sequences.

Typical activities are:

1) Capture;
2) Berthing;
3) Leak check;
4) Module R/R;
5) Umbilical connections.

Typical sequences include:

1) Retrieve;
2) Detank;
3) Inspect/examine;
4) Check out;
5) Conduct preventive maintenance;
6) Prepare for deorbit;
7) Repair;
8) Resupply (fluids);
9) Reconfigure;
10) Store;
11) Mate payload;
12) Deploy.

The illustrative top-level scenarios combining these sequences for specific operations are shown in Figure 4.2-4. These illustrative scenarios have been integrated with other scenarios foreseen into a comprehensive top-level turnaround flow for servicing a nominal space-based OTV. Figure 4.2-5 depicts this flow.

4.2.3 Service Operations

The service operations depicted in Figure 4.2-5 are analyzed at some length in the following paragraphs. These operations are decomposed to the level necessary to identify relevant requirements and issues. The requirements, issues and other associated considerations identified in the decomposition process are then organized and discussed separately for each of the operations.

Succeeding sections of this report will discuss the implications of these requirements, issues, and considerations for OTV and space station design. Although these are high-level discussions incomplete at this early stage of OTV design, they represent a first step toward organizing the information that will ultimately result in the identification and justification of systemwide and vehicle-specific design requirements.

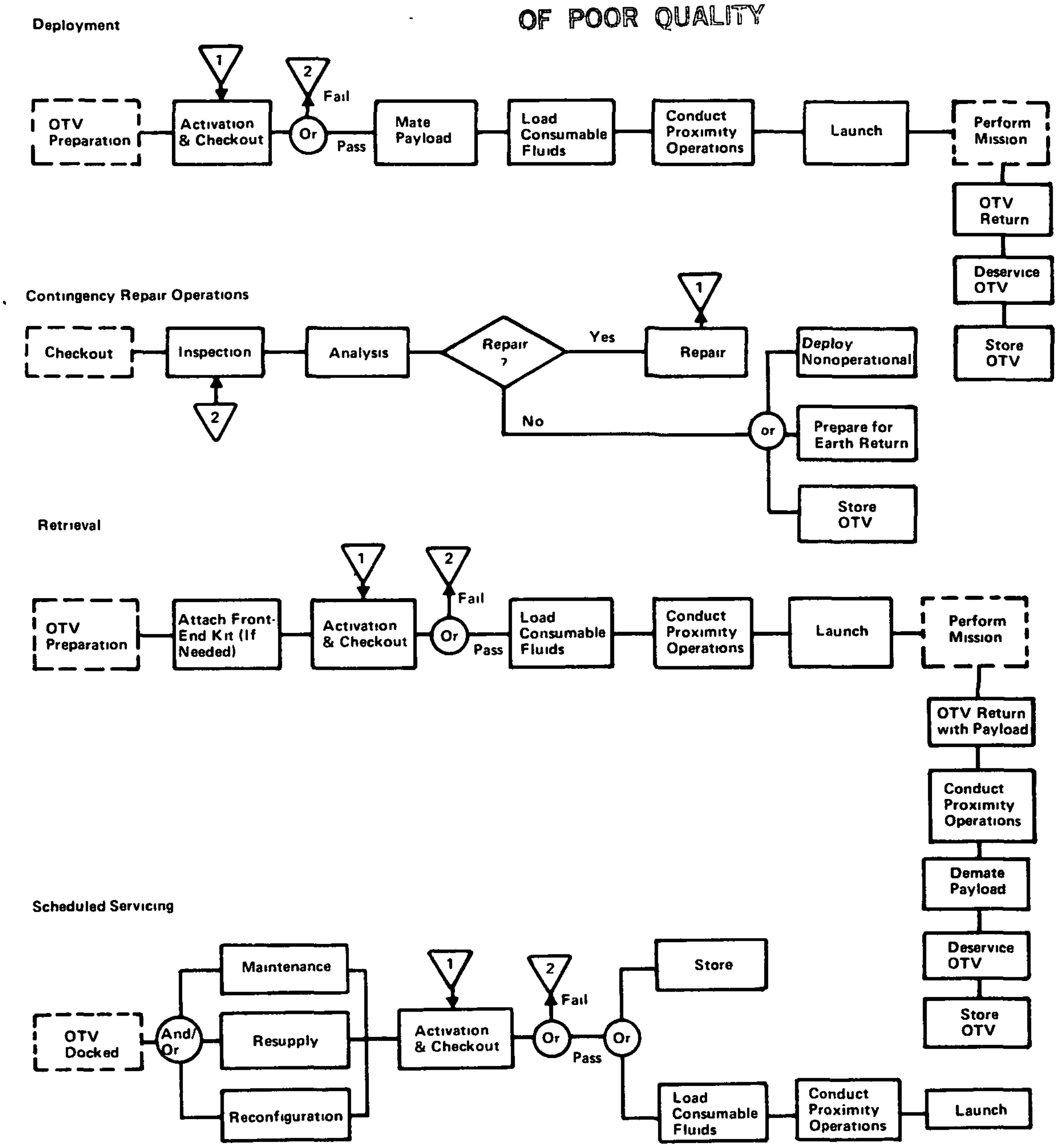


Figure 4.2-4 Representative Servicing Scenarios

OTV Proximity Operations - This process, as nominally conceived, encompasses maneuvering the OTV from its rendezvous and shutdown position to the location where the RMS (or other grappling device such as a "space crane" variation of the RMS) will assume control of OTV movement. It could be preceded or interrupted by the retraction of deployable components. The flow of the activities in this sequence is depicted in Figure 4.2-6. Preceding and succeeding sequences are also shown in this figure for perspective.

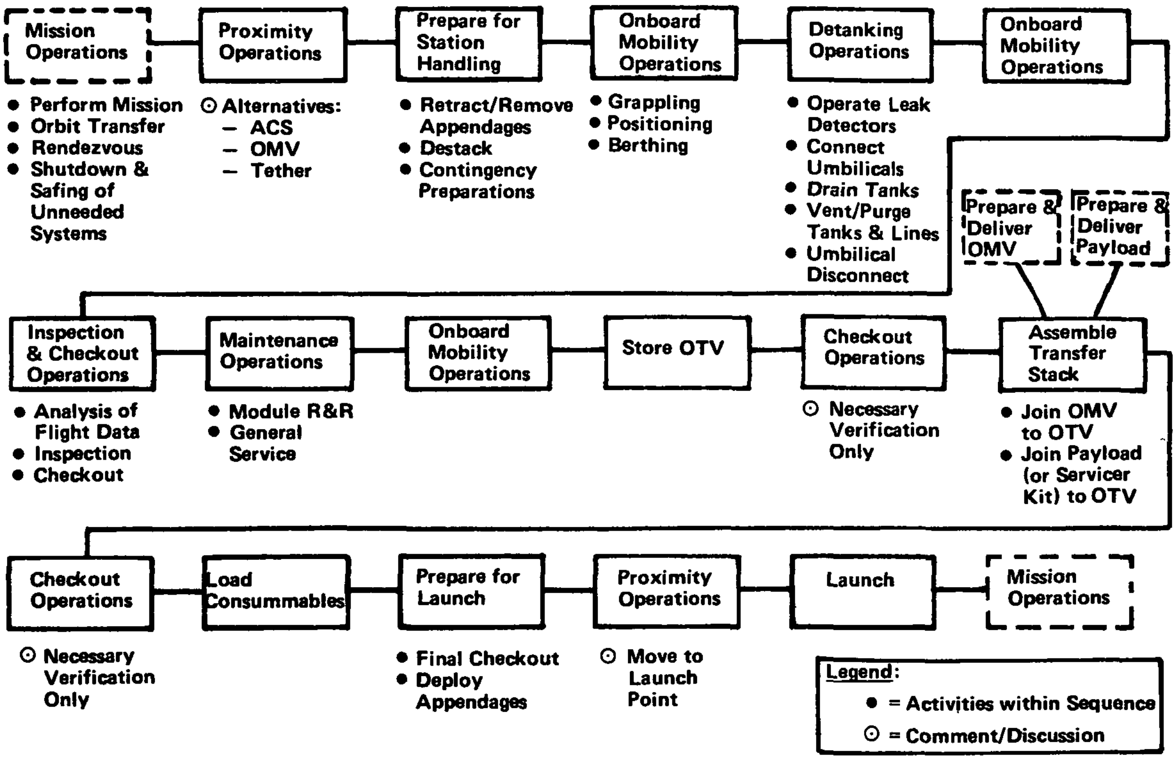


Figure 4.2-5 Top-Level Turnaround Flow for Space-Based OTV

Options for transition of an OTV from free flight to space station control include using its onboard attitude control system (ACS) to maneuver and using an OMV to go out and retrieve the OTV. As can be quickly deduced from Figure 4.2-6, use of the OTV's ACS would be the simpler total system. This is the baseline recommendation. However, if an OMV is to be available for this purpose between commitments to support other activities, it could probably be used to pick up the OTV after shutdown and transfer it to its berth at the detanking facility. Alternatively, the OMV or some other means such as an EVA with the MMU could be used to ferry a cable (tether) out to the OTV so it could be reeled in.

<u>Use of OTV Attitude Control System</u> - This option, as stated earlier, involves the simplest system in terms of equipment, interfaces, and procedures. It would require the OTV design to provide the capability to operate ACS thrusters remotely after all other systems have been shut down. Implementation of this scheme could also require development, installation and support of a remote control station in the space station if tradeoffs indicated local control to be preferable to automated operations and/or ground-based control. But such an integrated teleoperator/

supervisory control workstation would be justified by a number of other applications such as OMV control from the orbiter or the space station and OMV- or OTV-mounted servicer-kit operation--although the latter application would considerably increase the complexity and weight of the workstation. For this reason and others, servicer-kit operations may best be left to ground-based controllers where these operations can be satisfactorily conducted if communication link delays can be overcome.

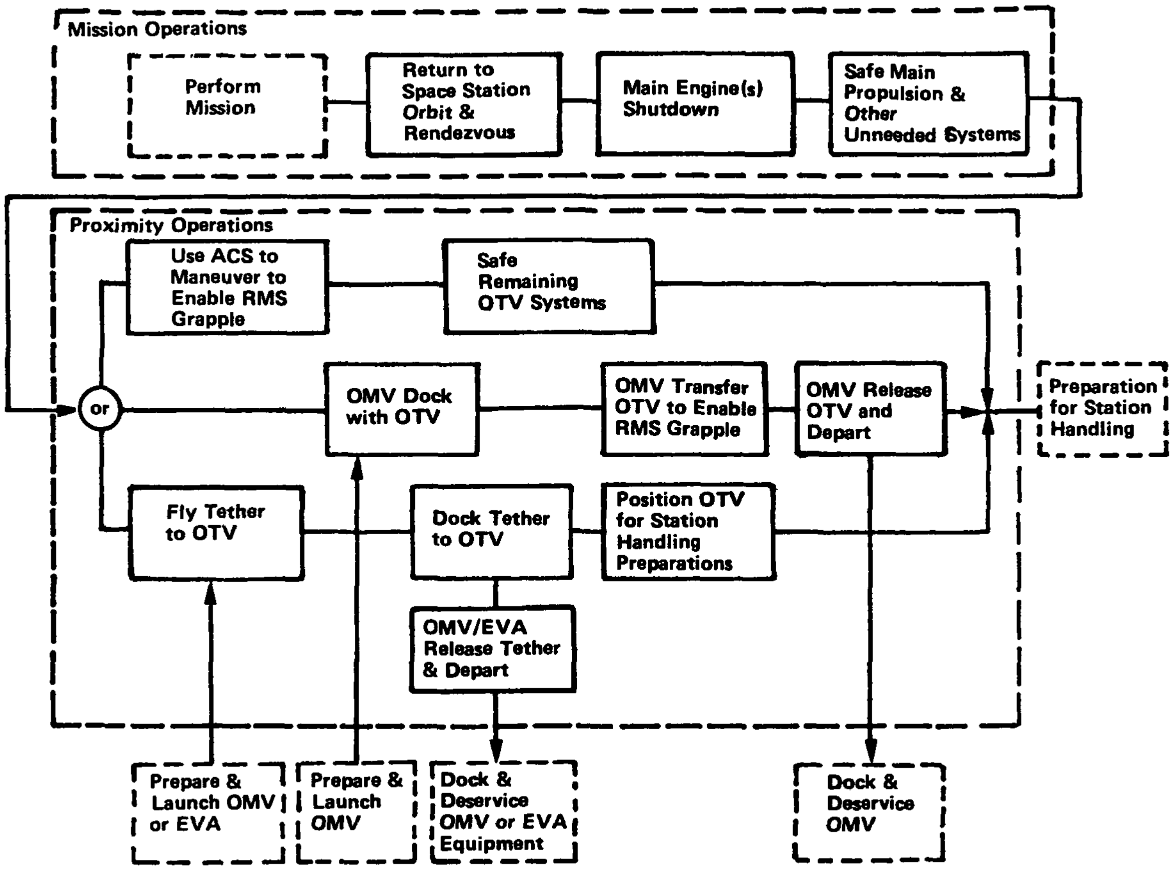


Figure 4.2-6 OTV Proximity Operations

A number of relevant issues are involved in this option. The first is the issue of control location--the determination of the space station's role in the proximity operations process. Though many argue that it is intuitive that local control would be preferable, a formal tradeoff may introduce less apparent concerns that, in the end, argue strongly for ground-based control in which the OTV would be docked with a passive space station or for total automation of proximity maneuvering.

The next issue with a direct impact on design of both the OTV and the space station is control mode. Relevant questions regarding this issue include: Will automated control be feasible? Desirable? Supervisory control? Teleoperator control? What will be the hardware, software, processing time, and other impacts of the mode chosen? A discussion of these points is beyond the scope of this study but they are being treated in a number of ongoing efforts.

Other points relating to this approach must also be considered. The first is the contamination implication of OTV ACS and other system propulsion products, venting and leakage--contributing to a "brown cloud." The resultant system design requirements may affect the development of OTV and space station hardware as well as operational procedures. However, it should be noted that many experts feel the oxygen/hydrogen system selected for the vehicle can be used in the vicinity of the space station with no significant contamination risk.

Another concern is plume impingement. Handling procedures and space station surfaces will have to be designed to prevent damage from ACS plumes if thrusters are used for close-in mobility. Some minor damage to MLI surfaces was noted from MMU plume impingement during the Solar Max rescue mission.

Yet another is illumination management. Obtaining proper levels of lighting during the rendezvous and docking maneuvers may necessitate some stationkeeping activity once proper illumination levels are determined.

Lastly, the additional thruster use for these operations will affect the OTV's ACS life and servicing requirements and the significance of this effect must be determined. It could be that formal tradeoffs will result in clear arguments for or against this option based on such things as reliability impact, servicing timeline optimization, and/or comparative ease of servicing one vehicle or the other.

Use of OMV to Position OTV - Many issues and considerations of this option are, as one should expect, identical to those involving use of the OTV ACS for proximity maneuvering. Should control be automated? Teleoperator controlled? From where: the ground? The space station? It is assumed the OMV would be controlled from the same workstation as the OTV. Will OMV outgassing present a problem? Will plume impingement?

The impact of incremental use is also relevant, but in a broader context than for the previous option. Not only are vehicle life consumed and servicing requirements increased, but scheduling conflicts may be created by the multiple demands on OMV time. Also to be considered are the safety and control issues associated with flying two vehicles simultaneously. While risks may be only nominal, they will exist and will certainly be greater than those associated with managing a single close-in free-flyer.

One possible advantage of using OMV in this application will result from its greater maneuverability. This assumes it will be designed for close-in operations and the OTV will be designed principally as a bulk carrier. The factors that will determine whether this increased maneuverability is really advantageous include whether it is necessary for safety reasons or for positioning before RMS capture of the OTV.

Design requirements that result from this proximity operations concept include:

1) Interface (docking) provisions for joining OMV to OTV;

2) Possible simplification of the OTV ACS system because it will not be employed in the vicinity of the space station.

<u>Use of OMV (or MMU) to Attach Tether to OTV</u> - Tether retrieval would partially overcome the "brown cloud" and plume impingement concerns inherent in close-in thruster operation. However, it would still require outward-bound propulsion and maneuvering as well as the necessary support systems. Because it probably could not be justified solely on the basis of the emission reductions that would result, it is envisioned principally as a candidate for backup mode operations. If, however, it is possible to somehow eliminate the OMV/MMU role in tether retrieval, this option would become clearly superior in terms of the criteria previously noted.

One advantage that could accrue from any tether approach employed would be the elimination of RMS handling of an incoming vehicle. If the tether system was based at the first service berth, it could bring the vehicle directly to its mooring. Of course, such issues as tether maneuvering (control) and deceleration must be satisfactorily resolved before such a system becomes feasible.

An associated benefit is simplification of service bay design. Elimination of the requirement to roll back a portion of the service bay (cylinder) wall for RMS access may become feasible if the vehicle to be serviced could be brought in through the open end of the cylinder by a tether terminating inside.

Station design requirements resulting directly from this retrieval approach include:

1) Addition of the tether system;

2) Accommodation of the tether within the service bay;

3) Integration of the tether system into on-station mobility operations.

No impact on vehicle design is foreseen. The vehicle/tether interface envisioned would employ the docking facilities and operations discussed in the following paragraphs.

Preparation for Station Handling Operations - Though not really a part of the retrieval process, these preparation sequences will be briefly considered at this point because they interrupt the retrieval process here--where they appear in the nominal process flow. The first activity that will be considered is retraction of deployable components. This discussion will be followed by consideration of the destacking operations aspects and what could be called contingency preparations. As illustrated in Figure 4.2-7, any or all of these activities could occur.

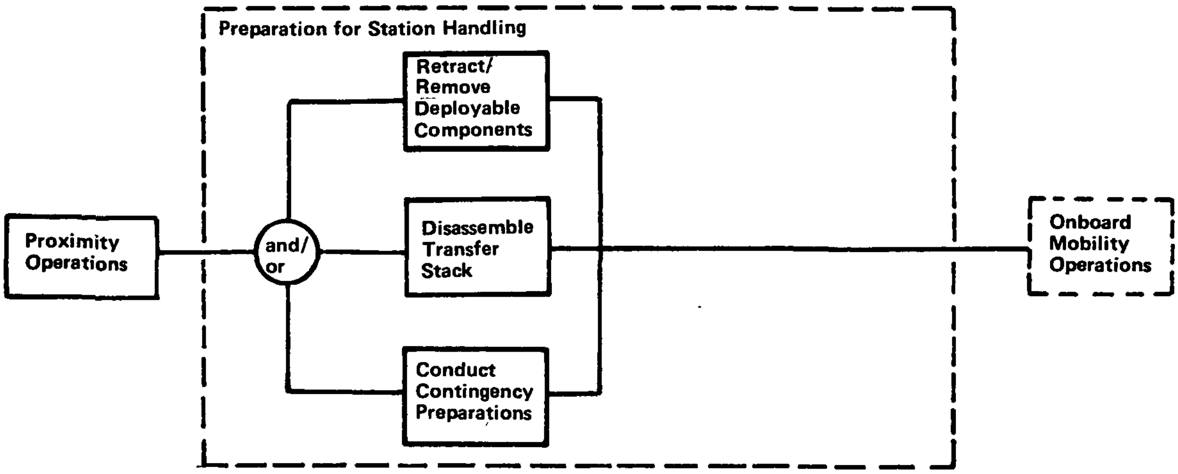


Figure 4.2-7 Preparation Operations

These processes, depending on situation-specific variables, could occur either before or after control over OTV motion is exerted by the RMS.

Retraction or Removal of Deployable Components - This sequence would be necessitated by conflicts between OTV appendages (such as antennas, engine nozzles, and/or the aerobrake) and the clearance at the service berths/bays. The number and complexity of the activities involved would, of course, be a function of system design. With proper forethought, any appendages that could not be eliminated by design could be designed for automated retraction or simplified removal operations. Furthermore, any problems in the retraction process could (again assuming accommodating design) be satisfied by a teleoperator or by an EVA sortie as noted in the contingency preparations discussion.

Destacking Operations - This sequence would be necessitated if a payload is too large to fit within the servicing bay or if safety considerations or physical configuration require separation before detanking. It could become the preferred method of operation, rather than a contingency method, if vehicle/payload control issues can be satisfactorily resolved.

C-2

If it is necessary to remove a payload or decouple the OMV/OTV pair in free space rather than within the confines of the servicing bay, these operations would best be treated as a separate sequence. Whether the operations involve EVA or whether they can be accomplished autonomously or with the intervention/participation of a manipulator, their characteristics and associated issues will differ in a uniform manner from operations inside the service bay. For example, because the OTV (and payload) will probably not have been detanked before these operations, the safety and operational concerns associated with operating and working on fueled vehicles will apply. Control during and after demating operations will be another shared concern. Will the vehicle(s) be free-flying? Tethered? Held by grappling arms?

Contingency Preparations - This sequence includes activities that would be performed by a servicer (EVA or teleoperator) to correct anomalies preventing routine station handling and processing. Examples include:

1) Failure of deployables to automatically retract thus violating envelope constraints;

2) Docking adapter malfunction;

3) Safety-related failures;

4) Attachment of grappling fixtures or other adapter(s) to the payload returned by the OTV.

The issues, options, and alternatives associated with these and other contingency operations would, of course, vary. But certainly the design of hardware and procedures to accommodate multimode operations (EVA, teleoperation, automation) will be a primary concern.

Onboard Mobility Operations - Grappling the OTV is, in the truest sense, a transitional element in the servicing process flow. It both ends free-flight operations and begins onboard mobility operations. This activity and the positioning activities are presently envisioned as the components of the initial onboard mobility sequence. Figure 4.2-8 depicts this sequence.

Grappling Operations - Grappling, with a minor variation (OTV docked at the start instead of in flight), and positioning and berthing will be repeated throughout the servicing process as the OTV is moved from bay to bay and operation to operation around the station. Table 4.2-2 lists the moves that could be involved; each move represents a repetition of the sequence depicted in Figure 4.2-8.

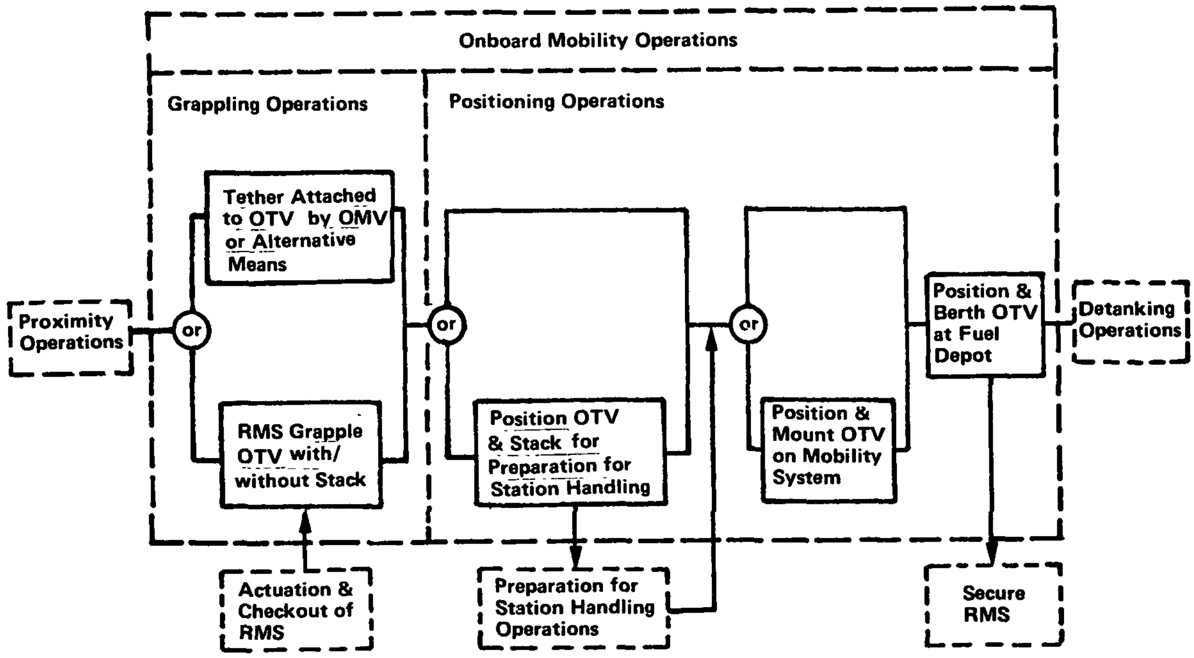


Figure 4.2-8 Onboard Mobility Operations

Table 4.2-2 Possible Onboard Moves

From	To
Capture Point	Detanking Berth
Detanking Berth	Maintenance Bay (for Postflight Maintenance)
Maintenance Bay	Storage Bay
Storage Bay	Maintenance Bay (for Preflight Maintenance)
Maintenance Bay	Stack Assembly Area
Stack Assembly Area	Tanking Berth
Tanking Berth	Launch Point

All grappling activities associated with these moves, except possibly those involving initial capture and those occurring before movement to the launch point, could conceivably disappear if an alternative onboard mobility system such as a rail or cable system is developed. The task flow for this grappling activity would remain unchanged whichever approach is chosen. This flow is shown in Table 4.2-3, which also indicates the nature of the tasks involved.

Grappling will drive a number of requirements in both OTV design and station equipment design. OTV design will be affected in three areas. The first, relatively minor, is the requirement to provide grappling fixtures. Alternative approaches to satisfying this requirement include the schemes illustrated in Figure 4.2-9. The first involves using target fixtures located about the periphery of the vehicle and the second approach employs a probe at the end of a manipulator arm to attach to one of the vehicle's docking fixtures. It is assumed the OTV will have a docking fixture at each end to facilitate stacking operations, OMV docking, and maintenance. The RMS standard grapple fixture described (Fig. 4.2-10) is an example of a concept that could be employed.

Table 4.2-3 Grappling Activity Task Flow

	Task	Nature of Task*
1	Identify Target	C
2	Locate Target	P
3	Select Path to Target Vicinity	C/P
4	Move to Target Vicinity	M
5	Position (Rough) Manipulator(s) to Grasp OTV	M
6	Place (Final) Manipulator(s) to Grasp OTV	M
7	Grasp OTV	M
8	Verify Grasp	P
*C = Cognitive; P = Perceptual; M = Motor		

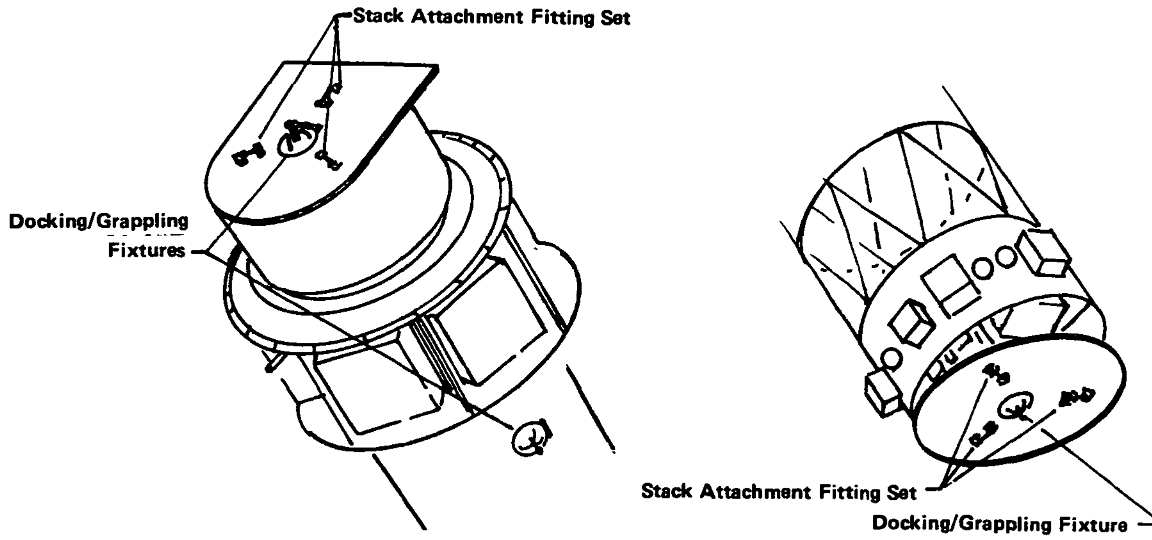


Figure 4.2-9 OTV Grappling Fixtures

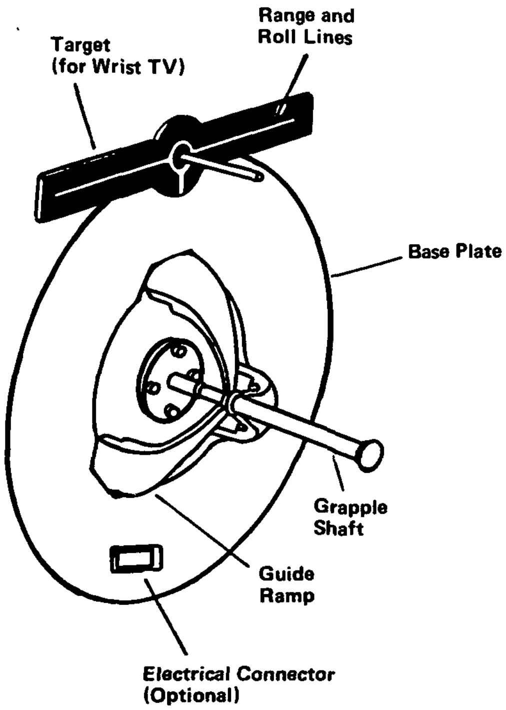


Figure 4.2-10
The RMS Standard Grappling Fixture

The standard grapple fixture consists of a rigid shaft, three alignment cam arms, and a target fixture (Fig. 4.2-10). The rigid shaft that is grappled by the special end-effector (SEE) provides the structural support between the payload and the RMS. The grapple target fixture is sighted by the RMS wrist camera and is used to align the SEE with the grapple fixture before capture. When the grapple fixture is within the capture envelope, the snares of the SEE are closed about the rigid shaft and are withdrawn to the end of the end-effector until a firm connection is made. The grapple fixture cams are fitted into corresponding slots in the SEE to rigidize the payload during manipulation.

Specifications:

Maximum weight: 22 lb
Torsional moment about longitudinal axis of SEE: 450 lb-ft
Bending moment to SEE: 1,200 lb-ft
Shear force associated with bending moment: 50 lb
Maximum payload weight: 32,000 lb

The second OTV design requirement resulting from grappling operations accommodation involves stabilization of the vehicle. Methods of stabilization that minimize rotation must be developed to support grabbing various peripherally located fixtures if this capture approach is chosen/required. The necessary fixtures would already exist because they will be necessary to satisfy other operational requirements such as:

1) Stack stabilization (postdocking);

2) Remote (inflight) servicing;

3) Contingency grappling.

If the probe and docking fixture approach is chosen, the vehicle could either be "normally" stabilized or spin-stabilized and would simply require orientation to facilitate the approach of the arm-mounted probe. Spin stabilization would be possible because the docking fixtures are located at each terminus of the vehicle's longitudinal rotational axis. This approach provides a straightforward contingency capability that seems to contain few risks and that would appear to be easy to perform either with a remote manipulator or with total automation.

The third grappling-imposed requirement on the OTV would be to accommodate the structural loads induced in the capture process. The impact of this requirement can probably be minimized by proper design and procedure development.

The station equipment affected by grappling all relates to the manipulator arm. Specifically, its structure must be designed for the loads involved and its controls must provide the response, accuracy and flexibility necessary to (1) avoid damage to itself and the OTV, and (2) satisfy established timelines. The controls must meet these requirements in autonomous and teleoperator operational modes and for normal and contingency situations.

The key issues involved in control of the grappling process include speed and accuracy of response to control commands and design of the feedback link for teleoperator situations where the human is in the loop. The first two are a function of the evolving state of the art in manipulator design and the situationally imposed requirements (speed and accuracy required). The feedback issue is, simply stated, determination of the video requirements--line of sight, resolution, and 2-D or 3-D--necessary to accommodate consistently satisfactory grappling performance. Future investigations will have to resolve these issues.

<u>Positioning and Mounting/Berthing Operations</u> - As for grappling operations, a fairly set task flow would be involved if an RMS-like system is used for positioning (and mounting or berthing) the OTV stack (Table 4.2-4).

Table 4.2-4 Positioning (and Mounting/Berthing Task Flow

Task		Nature of Task*
1	Clear Locale (Where Grappled)	C/P/M
2	Identify Destination	C
3	Locate Destination	P
4	Select Path to Destination	C/P
5	Move (Transfer) OTV to Destination's Vicinity	M
6	Position (Rough) OTV	M
7	Place (Final) OTV	P/M
8	Wait for (Assume Autonomous) Activation of Locks	–
9	Release Grasp	M
10	Verify Released	P
11	Clear Locale (Where Released)	C/P/M

*C = Cognitive, P = Perceptual, M = Motor.

Tasks 8 through 11 would be eliminated for the "position for destack operations" sequence because it is assumed, for the purpose of the present task flow, that destacking would occur with the OTV "hanging" on the end of the manipulator arm. The remaining tasks would be unchanged. Careful consideration of this sequence may drive a requirement for more than one RMS. On the other hand, if the tether attachment procedure is employed, the Table 4.2-4 steps would be significantly changed--and hopefully simplified. Tether attachment steps would represent the first tasks in rail or cable mobility system operations. Under this scheme of operation some of the tasks identified in Table 4.2-4 would no longer be necessary and the nature of many of the tasks remaining would be changed.

Task 4 in the table, selecting a path, would no longer be necessary; the cable or rail system would be the path from origin to destination. Tasks 1 and 11 would become purely academic because the set path would presumably not contain obstacles. The transition from Task 6 to Task 7 could be simplified to nothing more than a speed adjustment. And, finally, Tasks 8, 9 and 10 could be deleted because no manipulators would be involved.

The remaining tasks (2, 3, 5, 6 and 7) would be simplified enough to permit automation using existing technologies and techniques. A human role would, however, still be necessary until artificial intelligence capabilities advance sufficiently to permit automated task and path planning. This role would, at a minimum, involve selecting and entering through a keyboard the appropriate code to tell the automated system the destination of the vehicle.

The issue of automation, passed over lightly in the treatment of grappling operations, bears further explanation at this time. The three categories of tasks (cognitive, perceptual and motor) identified in Tables 4.2-3 and 4.2-4 will be used to structure the discussion that follows.

All the motor tasks readily lend themselves to full automation. The only reservations that must be mentioned are the possible impacts of control response speed and accuracy limitations imposed by the state of the art in relevant technologies.

Perceptual tasks present an interesting mixture. Some, such as "verify released," can be performed by electromechanical devices. Others, such as "locate destination," require either some sort of built-in and automatically updated position reference map or some ability to interpret visual information. Both would require an extensive computational capability to automate. Further, image understanding is only an emerging discipline and will probably see only limited use over the near term. An alternative technology such as radar may, however, be able to perform many perceptual tasks.

Cognitive tasks are the most complex of the three types. They involve knowing and/or understanding, and often imply choices among alternatives in varying, unstructured situations. Such descriptors as "identify" and "select" are typical of cognitive tasks. There is some overlap between the cognitive and perceptual categories--"identify" typifies this group.

Some degree of intelligence--the ability to cope with a new situation, to learn or understand--is required to perform the cognitive tasks. An advanced understanding of the reasoning process and extensive computational power will be required before these tasks can be automated. Again, as with image understanding, the necessary capabilities (often referred to as artificial or machine intelligence) are beginning to emerge and may see limited application over the near term.

In summary of this brief discussion, it can be said that for highly structured tasks (e.g., "normal" docking), full automation is a reasonable short-term goal. Indeed, it may be preferable to human involvement. However, for less structured tasks (e.g., "contingency" docking or unplanned remove and replace operations) where more complex perceptual and cognitive capabilities are required, man-in-the-loop operation, either supervisory or teleoperator, will be required over at least the short- and intermediate-term goals.

Other considerations relevant to positioning and berthing operations include positioning accuracy requirements and the impact they will have on manipulator and berth design, vision requirements (camera location) and accommodation of any operational contingencies arising from equipment malfunction. The first two subjects were adequately discussed under grappling operations. The last implies accommodation of multimode operations by designing the mobility system (whether a manipulator or a cable/rail system) to readily permit a backup system such as another arm, another tether (possibly with spread bar or sling), or an EVA astronaut to assume control of the OTV and complete the desired transfer, positioning or berthing process.

Detanking Operations - The detanking operations presumed necessary for technical, economic, and probably safety reasons will be the first servicing sequence performed aboard the station. Figure 4.2-11 shows the activities in the process. The issue of whether multiple fluids would be transferred simultaneously or sequentially is beyond the scope of this study; it should be addressed in future fluid transfer studies.

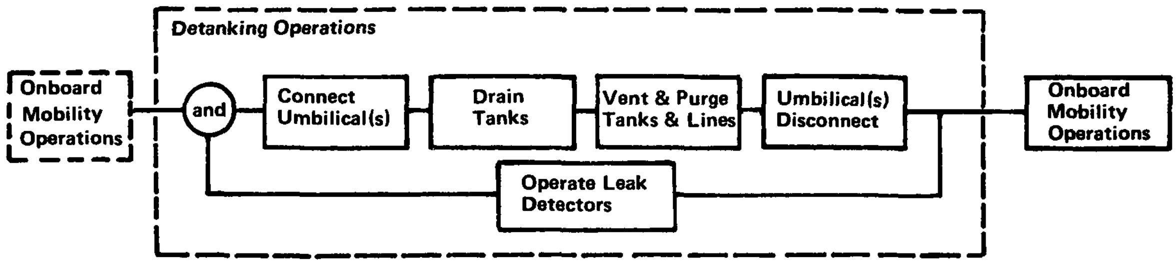


Figure 4.2-11 Detanking Operations

Several fluids could potentially be involved in these detanking operations. However, for this particular OTV design the number will be minimized because the propellant and oxidizer will be used for electric power reactants and for the ACS, the propellant will serve as the pneumatic system fluid, tank pressurization will be autogeneous, and a space station-provided means will be used for purging fluid lines. This approach will permit operations employing only two fluids—liquid hydrogen and liquid oxygen.

Umbilical Attachment - The first of these activities, umbilical attachment, would preferably be a fully automated activity either occurring at the time of docking or initiated on completion of docking. (Though we believe the present space station baseline envisions manual make/break, we foresee a possible change in this position to take advantage of available techniques and capabilities.) This choice will affect both space station and OTV design.

Simultaneous umbilical connection could eliminate some mechanisms from the berthing platform for positioning and inserting the umbilicals and some tasks from the operation. However, it would require precise positioning of the OTV and require the vehicle's propulsion module to employ QD-type connectors and associated plumbing specifically designed for onorbit refueling.

If umbilical attachment is to be initiated after completion of the docking (berthing) activity, a number of additional cognitive, perceptual, and motor tasks would be introduced. The aspects of each type of task were previously discussed. But the specific set of tasks involved cannot be identified until design iterations determine such details as whether access panels will have to be opened, whether connector covers will have to be removed and stored, and so forth.

Table 4.2-5 presents a nominal task set. It ignores questions like those above and focuses on the narrow issue of deploying the umbilical. It is a "worst-case" task set and is described in a manner intended to show adequate detail to illustrate that the individual tasks (and hence the set) could be performed by hard automation, by a sophisticated remotely controlled manipulator or by an EVA individual. For hard automation, Tasks 1 through 3 would be predetermined so they would be dropped from the flow.

Table 4.2-5 Umbilical Attachment Activity Task Flow

Task		Nature of Task*
1	Identify Target (Connection Point)	C
2	Locate Target (Connection Point)	P
3	Select Path to Destination	C/P
4	Free Restraints	M
5	Extend Umbilical (Move to Target Vicinity)	M
6	Position (Rough) Umbilical	M
7	Place (Final) Umbilical (Connect)	M/P
8	Lock (Turn?)	M
9	Verify Locked	P

*C = Cognitive, P = Perceptual, M = Manipulative.

The addition of this activity requires the station to provide and accommodate the manipulator(s) capable of performing the included tasks. Selection of the hard automation alternative to accomplish the tasks would impose weight and maintenance penalties over the flexible automation alternative because hard automation is, by definition, specifically oriented toward a single purpose--it could not be used for anything else. Selection of the flexible automation alternative, in the form of a relocatable, multipurpose, remotely controlled manipulator, would overcome these penalties but impose penalties of its own in terms of operational complexity when compared to simultaneously occurring docking and umbilical attachment.

Whichever attachment method is chosen, some decisions will have to be made about where the attachment(s) will be located. That is, will plumbing within the OTV be used to group all connectors so a single umbilical can simultaneously connect to all of them or will multiple umbilicals be required to minimize OTV plumbing line runs to satisfy weight, maintenance, safety, or technical requirements?

Leak Detection - Activated before or at the time of umbilical attachment, leak detectors would be operated throughout the detanking operations sequence. Presumably they would be tied into caution and warning panels in the station and/or service bay. They could also possibly activate automatic shutdown/recovery sequences (contingency operations) if their reliability were high enough.

Tank Drainage - For the propellants this activity should involve no more than (1) activating (opening) valves to allow the autogeneous pressure to displace them, and (2) closing the valves when the process is complete. Contingency situations, like inoperative valves, would require the OTV either to provide a way to mechanically actuate the valve or to provide a second access route (valve) into each tank to permit depressurization/drainage of the tank.

Inoperative (open) valves would also require some contingency provision for closing the tank before umbilical disconnect to prevent undesired venting. The same solutions, a second valve or a local mechanical means of actuation, would apply. "Design for use" considerations such as location (access), clear identification, and force application requirements must be accommodated in designing and locating these valves. They will have to accommodate all possible contingency-mode operations as well as any changes caused by evolutionary growth.

Thermal management of the tankage and lines throughout this process will also be a concern of note. Because blockage of a line by freezing during the process could prove to be a critical hazard, it must be carefully considered in the design of lines and defueling operations.

Venting and Purging - Clearing residuals from tankage and lines after they are drained will involve two sets of considerations. The first focuses on ensuring the residuals will not be hazardous or contribute to the "brown cloud." Venting directly to space will not be allowed to occur in the vicinity of the space station. Rather, purged residuals will be processed for reuse through a collection system. Whether this policy becomes flexible over time will be a complex function of the seriousness of the contamination problem in the station's environs and of the delivered "cost" of the fluids as well as the weight, complexity, power consumption, etc of a collection system.

The second set of considerations focuses on how the purging will be accomplished. Two distinct phases or types of activity are projected. The first is gas displacement (using autogeneous pressure and, maybe, nitrogen) and the second is vacuum drying. Combined, the two will ensure no residual fluid remains to interfere with the refueling or remove and replace maintenance operations that may become necessary. Both OTV and space station systems will have to be designed to accommodate these processes.

The tasks associated with erecting and retracting the venting masts intended to move the vented gases away from the facility will be included in this venting and purging activity unless it proves preferable and feasible to permanently erect the masts. The only other tasks involved will be opening and closing valves (essentially switching functions) that will be easy to command from a remote site when in a normal operations mode.

Umbilical Disconnect - Once connecting lines are purged, the umbilical(s) will be detached and stowed to complete the detanking operations sequence. Table 4.2-6 shows the tasks that would be involved in this activity.

Table 4.2-6
Umbilical Detachment Activity Task Flow

Task		Nature of Task*
Ref	Vent & Purge Operations	C/P/M
1	Unlock (Turn?)	M
2	Disconnect (Turn?) Umbilical	M
3	Retract Umbilical	M
4	Position (Rough) Umbilical	M
5	Place (Final) Umbilical	P/M
6	Secure Umbilical	M

*C = Cognitive, P = Perceptual, M = Manipulative.

Note that these tasks are all primarily manipulative in nature and none are very complex. This tends to make them amenable to hard automation. Any associated perceptual tasks could easily be accomplished through the use of existing technology, e.g., sensors and switches.

Of course, proper system design would have to allow for manual (EVA) and/or teleoperator task conduct in contingency situations.

Issues and Considerations - A key issue was avoided in the preceding discussion by assuming the OTV tanks will require evacuation. However, a technology goal is to develop systems and procedures that will eliminate this requirement. Two alternative approaches come readily to mind:

1) Refilling partially full tanks without having to first empty them;

2) Removing the "expended" tanks and replacing them with full tanks.

The implications of such servicing alternatives in terms of OTV design, space station design, operational hazards, timeline reduction, etc, will require future investigation.

This discussion also avoids another key issue by not attempting to define where the detanking facility should be located. It would be convenient if detanking could be conducted at the same berth as inspection and checkout (I&CO) and remove and replace (R/R) maintenance functions. However, it is assumed that safety issues and pollution considerations will drive this decision in spite of timeline considerations. For example, safety requirements will impose strict controls to prevent release of propellant and/or oxidant near the orbiter; crew safety and potential contamination of sensitive components are specific concerns. The treatment in the following paragraphs applies whether the facility is remote (tethered or a separate formation flyer) or a part of the main space station structure (a separate bay or a hookup in the main service bay).

A number of components, systems and technologies will be critical to these detanking operation processes. Components that heavily affect design, operation and maintenance include valves, filters, couplings, and connectors. Key systems include drainage interfaces, mass gaging, and component replacement systems. The technologies identified as critical include mass gaging, commodity transfer, slosh control and venting.

Interface management is the next area of concern. As has been alluded to in previous discussions, any given interface must not only accommodate automation, it must also be both EVA- and manipulator-compatible.

Fluid management methods must be developed and definitized for the scenario. Although childown procedures/techniques, particularly for lines, are safety-critical, they have not been well defined. Neither are the methods for measuring fluid levels in the tanks or for handling residuals. Present technology mass and flow gaging techniques are not yet very precise--$\pm$10% is typical. It is presumed that a two-step (gas displacement and vacuum cleaning) residual purging process will be effective but the process and hardware to effect it are unproven.

Finally, an assumption made early in the discussion of this sequence was that EVA avoidance was a relatively high-priority objective, at least for routine task performance. There are two reasons. The first involves timeline impacts on baseline operations. The total service timeline is extensive--prebreathing before each EVA sortie and the manipulative limitations imposed by the suit--are significant contributors to this total. Therefore, minimization of this timeline through avoidance of EVA seems a reasonable goal.

The second reason for assuming the desirability of EVA avoidance involves the potential hazards inherent in management of fluids and, specifically, management of cryogen transfer operations. These operations can be performed by automation; there is no reason to expose an EVA astronaut to the hazards involved under normal operating conditions.

Inspection and Checkout Operations - Very simply stated, the purpose of this sequence is determination of the "fitness" of the OTV for continued operations. It involves two, possibly parallel, activities--inspection and checkout--plus a third, analysis of inflight data. It should occur before initiation of either inspection or checkout. Figure 4.2-12 depicts the sequence.

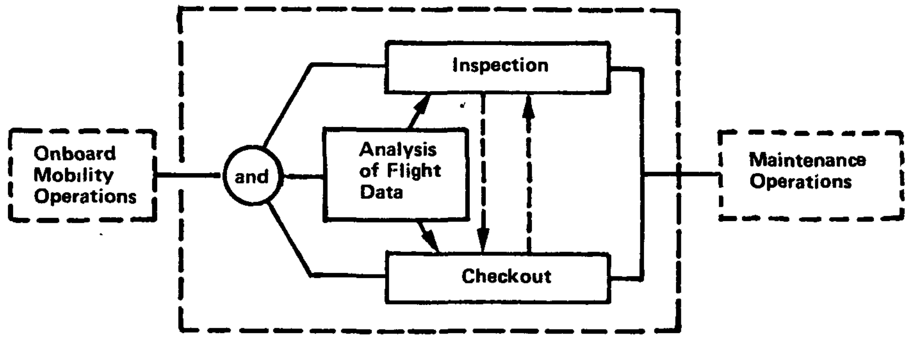


Figure 4.2-12 Inspection and Checkout Operations

Inspection focuses on detection and interpretation ot anomalous physical characteristics through sometimes augmented but typically human sensory (principally visual) means.

Checkout, on the other hand, implies detecting and subsequently isolating systems or components producing anomalous electronic signals during simulated operating conditions if in a service bay, and/or actual operating conditions if in flight. Data may originate from specific sensors or from the capture of enroute signals, depending on the system/component being checked.

These two activities could conceivably be conducted in parallel to minimize timelines if technically compatible techniques are employed. Undoubtedly they will often influence one another; data from one will suggest the need for specific and possibly additional attention to aspects of the other.

Inspection - The requirements for inspection will be a function of the overall maintenance philosophy and its impact on vehicle design. It is safe to assume that whatever the requirements, the techniques employed will be nondestructive. Destructive test techniques will presumably be reserved for unreparable vehicle(s)/components and will, again presumably, be conducted by "depot"-level ground-based activities.

Nondestructive inspection (NDI) techniques that could be considered for use on an OTV at the space station include, but are not necessarily limited to:

1) "Sniffer"-type leak detectors;

2) Unaided vision;

3) Image enhancement;

4) Multispectral imaging (e.g., thermal, electric field);

5) Dye penetrant;

6) X-ray radiographic;

7) Ultrasonic;

8) Eddy current;

9) Magnaflux;

10) Magnetic particle.

Each has advantages and disadvantages that affect its desirability in space environment applications. For example, the magnaflux technique requires inducement of a magnetic field and use of metal flakes, both undesirable. Table 4.2-7 summarizes significant strengths and weaknesses of these techniques in space-based applications.

Table 4.2-7 NDI Technique Advantages and Disadvantages in Space

Technique	Advantages	Disadvantages
Sniffer(s)	– Detect Small Concentrations	– Requires "Sniffable" Medium
Unaided Vision	– Easy to Implement via Multiple Means (EVA or Remote Camera)	– Limited Capability – Limited Consistency
Image Enhancement	– Permits "Seeing" Things That Normally Cannot Be Seen	– Equipment Required
Multispectral Imaging	– Permits "Seeing" Things That Normally Cannot Be Seen	– Equipment Required – Limited Applicability
Dye Penetrant	– Enhances Visibility – Does Not Require Elaborate Equipment	– Fluid Confinement – Detects Only Surface Defects
Radiographic (X-Ray or Gamma Ray)	– Usable for Many Metals & Other Materials – Film Provides Permanent Record – Detects Surface & Subsurface Defects	– Weight of Equipment – Dark Room Required – Film Use – Power Consumption – Potential Safety Hazard
Ultrasonic	– Simple Procedure – Detects Surface & Subsurface Defects – Can Provide a Permanent Record	– Interface Medium (Coupling Agent) Required – Oscilloscope Required – Interpretation Requires Highly Trained Personnel
Eddy Current	– Simple Procedure – Detects Surface & Subsurface Defects	– Electric Current Required – Does Not Give Physical Shape of Discontinuity
Magnaflux (Magnetic Particle)	– Simple Procedure – Equipment is Portable	– Employes Magnetism & Requires Loose Particles – Only Usable on Ferromagnetic Metals – Electric Power Required

Inspection procedures, whatever they turn out to be, will probably be conducted in the service bay. They will have a number of implications for the OTV, for the facility and for support equipment.

Until vehicle design becomes more firm, specific techniques and requirements will be hard to identify but general requirements for all techniques should be considered now. These include such things as provisions for:

1) Access to all areas of the vehicle requiring inspection;

2) Lighting requirements;

3) Storage and handling of inspection aids (tools);

4) Minimization of inspection requirements and timelines.

Checkout - Three levels of checkout operations are envisioned. The first, inflight operations, will focus on (1) identifying the source of anomalous signals (unsatisfactory performance) so the faulty component/system can, if possible, be brought back into tolerance and, if not, be compensated for, and (2) recording the anomaly for future analysis and correction. Compensation would be accomplished by isolation and subsequent use of backup systems or by other workarounds such as alternative approaches to the function.

The second, probably more extensive, level is postflight checkout. These operations will focus on investigation (verification) of inflight anomalies and the conduct of predictive investigations as well as postmaintenance verification.

The third level of checkout operations occurs before operations checkout. It could involve several separate procedures conducted after unscheduled maintenance operations, after removal from storage, after payload mating, after loading consumables and after maneuvering to the launch point. Presumably these operations would be comparatively brief and focus on ascertaining that specific vehicle subsystems are ready for operation.

To be efficient, all these checkout processes must be effectively automated. All three levels of checkout operations will require similar interfaces and support, including:

1) Sensor/test equipment;

2) Communications link;

3) Data analysis and control hardware;

4) Data analysis and control software;

5) Data analysis and control executives.

The aspects of each will be discussed in the following paragraphs.

Automated checkout, particularly for inflight engine condition/performance assessment, will demand substantial upgrading of present diagnostic sensors. Ideally, these advanced diagnostic sensors will be tied into local onboard or other no-delay executives so they will be able to detect impending out-of-limit situations and trigger immediate corrective action.

Diagnostic sensors will also have to be upgraded to support effective automation of pre- and postflight checkout operations. For these applications, attention will be focused on accurate and timely diagnosis. Timeline minimization is seen as a key goal.

The use of these sensors and the issue of where they will reside--built-in, plugged-in, or remotely located at the end of an umbilical or RF communication link--will have a key impact on system design. Overall system implications will be complex.

Turning from the sensors themselves to the handling of the data they will produce, three additional sets of considerations become relevant. These are the link from the sensors to the analytic facility, the analytic facility including hardware, software, and executives, and the process control conducted by the executives and/or human supervisory controllers.

The communications link from the sensors to the analytic facility can be very simple or very complex. A number of "whethers" influence the issue. The first is whether the link will be direct RF or through an umbilical into the space station. The next is whether all diagnostic procedures will reside in programs at one location or in several locations. Another involves whether time delays induced by NASA's block-formatting can be worked around if they are relevant.

Options for analytic facility basing include the ground, the space station, and the OTV itself. Issues such as weight, information processing requirements, equipment reliability and maintenance, and communications (link) management will drive the basing decision. In the end, multimode basing will probably be employed even though it will increase the complexity of the communications system because of the weight and maintenance advantages it will offer.

Process control, management of the checkout process, is seen as presenting the most complex set of considerations. Relevant technologies will still be evolutionary during the space station era. Use of artificial (machine) intelligence techniques will relieve the need for much of the present day's routine process intervention (by humans) at simple decision points as well as permit better use of other techniques such as parallel processing to speed up the automatic checkout process. The human role will evolve over time to that of supervisory control.

These machine intelligence technologies will also be directly applicable to communications management. They will be able to make routine and instantaneous decisions to (1) select the optimum combination of OTV, space station, and ground-based information processing resources, (2) select the optimum data link at a given time and for a given situation, and (3) possibly impact the NASA communication system block format-induced time delays. The end result will be to speed up the checkout process while minimizing OTV and space station information processing requirements.

Maintenance Operations - Maintenance has been defined as a specific operational subset of "servicing" for the purpose of this analysis. It is envisioned as involving assembly/disassembly and upkeep action as taken to prolong system life, enhance system operation, or prepare the system for a specific mission. As a result, only two categories (types) of maintenance operation are envisioned for this space-based reusable OTV. These categories are module removal and replacement and general service operations.

Though there are only two categories of operations, a number of specific (varying) characteristics within each category warrant consideration. For example, module R/R could be scheduled or unscheduled; it could be undertaken to restore operational capabilities, to upgrade the vehicle, or to reconfigure it for a specific purpose; and it could be undertaken by a number of different methods or schemes. Potential onorbit maintenance schemes include intravehicular activity (IVA) in a pressurized compartment, IVA using remote manipulators, extravehicular activity (EVA) using teleoperation, astronaut EVA using advanced high-pressure "tactile" space suits, and IVA-based supervisory control of advanced automation operating in either an IVA or EVA environment. Table 4.2-8 summarizes the full range of spacecraft maintenance method options. Onorbit maintenance approaches and concepts are shown in some detail.

Table 4.2-8 Spacecraft Maintenance Mode, Approach and Concept

Mode	Expendable Spacecraft	Ground-Refurbishable Spacecraft	Onorbit Maintainable Spacecraft					
Approach	No Maintenance	Repair, Refrubish, & Reorbit	Space Station Maintenance		Visiting System Maintenance			
			Service Bay	Inside	Orbiter		OMV/OTV & Front-End Kit	
					Service Bay	Inside		
Concept	N/A	Total Maintenance Capability	Level 1 Capability	Level 2 Capability	Level 1 Capability	Level 2 Capability	Level 1 Capability	Fetch for Orbiter or S/S
			EVA / Auto-mation	Manual	EVA / Auto-mation (TWS)	Manual	Auto-mation	Auto-mation
			Combination		Combination			

Level 1 Maint: Major Module Diagnostics and R&R; Fluids Replenishment; TBD Services (Kick It, Oil It, Lock/Unlock and Reposition . . .)

Level 2 Maint: Submodule Diagnostics and R&R; Submodule Repair (?); Limited Submodule Modification (?)

Because these schemes imply certain operational limitations and impose specific requirements on space station OTV accommodations, they will define maintenance and other service operation methods as well as influence the OTV's configuration and the space station facilities needed to accommodate the processes.

The numerous general service operations envisioned for the vehicle involve many considerations. Some of these operations are wearout prevention, cleaning, adjustment, recharging of energy storage devices and refurbishment of thermal control coatings. Some of the considerations involve access, modularization tradeoffs, definition of advanced automation's role and design for multimode operations.

The many factors inherent in these operations and considerations must be treated in any discussion of OTV maintenance. These will be covered in subsequent paragraphs but first it seems appropriate to briefly introduce Figure 4.2-13 that illustrates the time/process relationships for OTV maintenance operations. This depiction is not intended to be a linear representation of actual time relationships; it does, however, identify and show the time phasing of these operations.

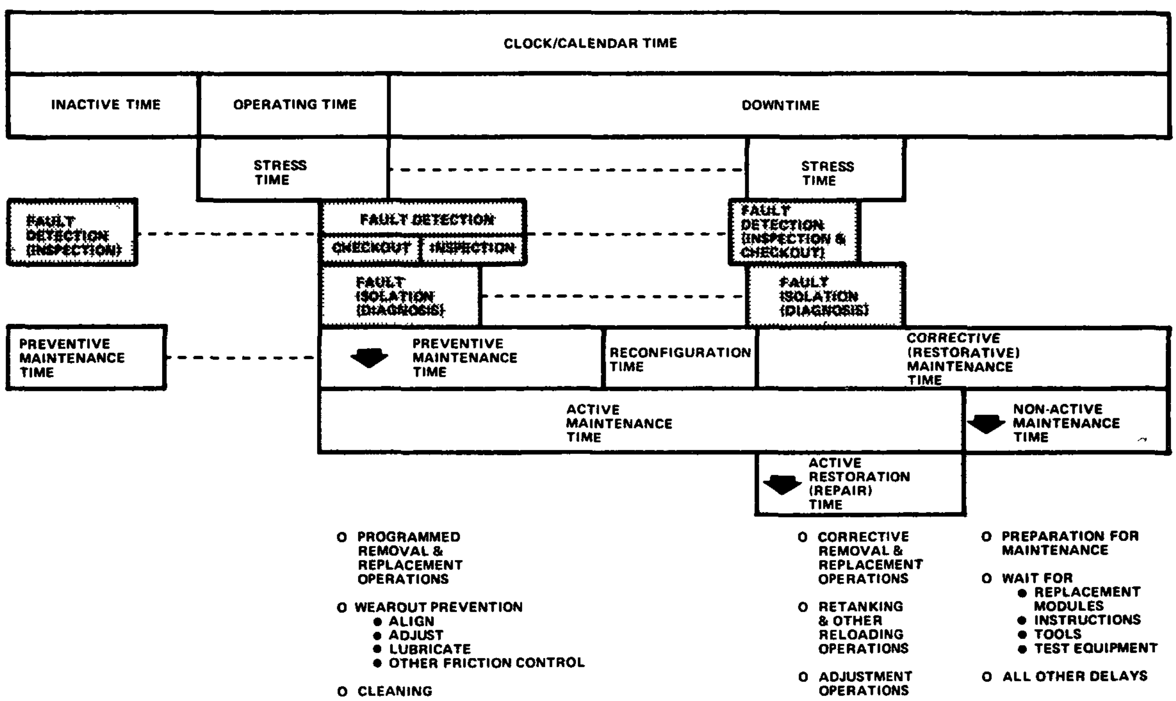


Figure 4.2-13 OTV Maintenance—Time/Process Relationships

The first aspect of OTV maintenance operations that will be discussed is whether they are scheduled or unscheduled. Scheduled servicing includes planned functions conducted to permit continued orbital operation of the OTV. Examples include component replacement to avoid predicted breakdowns (design life limitations) and to replace expendable items that have been consumed. The necessary actions are known well in advance and the service crew will have been familiarized and trained and will have conducted the necessary verification simulations for these events before actual operations. All crew aids/devices/tools and any other necessary support equipment will also be in place and rehearsals will have been conducted to support the scheduled servicing. Unscheduled servicing involves the functions conducted on a contingency basis to restore the OTV to an acceptable level of operational capability. It includes repair or replacement of components that fail randomly (unpredicted breakdowns) or because of "unscheduled events" (accidents). It is also characterized by improvisation and will, by definition, extend timelines. Table 4.2-9 summarizes these points.

Table 4.2-9
Characteristics of Scheduled and Unscheduled Maintenance Operations

- **Scheduled Servicing**
  - **Conducted to Permit Continued Orbital Operation**
  - **Replaces/Refurbishes Design Life Limited Items and Consummable Items**
  - **Programmed Operations**
  - **Crew Well Trained and Verification Simulations Conducted**
  - **Characterized by Use of Specifically Designed Tools and Support Equipment**
- **Unscheduled Servicing**
  - **Conducted to Restore the Vehicle to an Acceptable Level of Operational Capability**
  - **Replaces/Refurbishes Items Which Have Failed Randomly or Due to Unscheduled Events**
  - **Extend or Modify Timelines**
  - **Characterized by Crew Improvization (Tools and Methods)**
  - **Unprogrammed Operations**

Given the tight turnaround schedule expected for the OTVs and the characteristics of unscheduled servicing just listed, it would seem a reasonable goal to seek to minimize unscheduled OTV servicing.

Maintenance Scenarios - The activity scenarios for maintenance operations must be kept as simple and uncomplicated as possible to accommodate present-day EVA processes and, at the same time, promote evolution toward advanced automation. These scenarios will be influenced by OTV characteristics (degree of modularization, interfaces, retention hardware, etc) and by the evolution of onorbit servicing equipment and techniques as we progress from the present manual mode operational capability to increasingly automated methods.

Detailed scenarios for these maintenance operations cannot be developed at this time without a great deal of speculation about the specific aspects of the modules, tools and processes involved. However, experience gained through previous analyses of similar situations (e.g., our remote orbital servicing satellite and teleoperator human factors studies) indicates that, as for other sequences and activities discussed earlier in this report, the maintenance processes will be assembled from the known set of "primitive actions" into higher level processes (subtasks, tasks, activities, and sequences). This same experience also indicates that module and tool design will significantly affect the ultimate process design.

The Role of Automation and Artificial Intelligence - The purpose of the present focused effort to apply advanced automation technology to space station is not to replace man, but to augment or enhance his capability in a technologically complex and sometimes hostile environment. It is analogous to the development of diagnostic tools (expert systems) for the medical profession. In this case, there is no intent to replace the physician; rather the intent is to increase his database and to enhance his diagnostic capability using the experience of others that are "built into" the user-friendly tool. Although we were able to design, build, and operate flying machines long before the advent of the computer, with the computer we have been able to extend our flight regime well beyond the solar system.

A recent aerospace automation industry study group recommended four areas for technology advancement. Three are relevant to OTV servicing efforts at the space station:

1) Spacecraft services self-management (through AI);

2) Ground support automation;

3) Space station robotics for inspection, assembly, servicing and repair, refurbishment and onorbit experiment interaction.

The first area involves the application of artificial intelligence and expert systems to spacecraft services and self-management. Current artificial intelligence research emphasizes the development of self-modifying codes, that is, software that learns, modifies itself, and becomes smarter and better. Useful results are being obtained in this area and some expert systems that have been built perform reasonably well. Some exploratory work is being conducted at NASA Kennedy Space Center on the application of expert systems to ground support operations (ground support automation). The real benefit of expert systems that is directly relevant to onorbit servicing is to reduce the number of human experts needed to do a given job that requires human expert judgment. The human experts cannot be eliminated, but they can be called on less frequently and made more productive. This is the recurring theme for development of automated design and analysis tools.

The following list summarizes potential expert systems applications in space-based OTV servicing:

1) Assist crew in failure detection, diagnostics;

2) Direct test procedures;

3) Chart preferred servicing modes and task sequencing;

4) Assist crew in decision making (autonomous decision making without ground support;

5) Direct safing procedures before crew access;

6) Assist crew in responding to emergencies;

7) Prevent crew-overload from massive data flow;

8) Enable autonomous remote servicing (a long-term prospect).

There is potentially a very large cost impact (savings) in this area because the application is not understood (so no one knows how to plan for it). It is important to understand the potential of this automated technology before the OTV and space station program costs are cast in concrete and budgeted.

Robotics/AI on the space station will be specifically useful for OTV inspection, checkout, assembly, servicing and repair processes. Here, robotics deals with both true robotics (machine intelligence is applied and the mechanisms are autonomous) and telepresence. In telepresence, communication techniques put man in in a remote location so intimately in the loop that he loses "sight" of his remote location and feels that he is physically present at the worksite and is himself doing what his robot proxy is actually doing. Figure 4.2-14 depicts the space shuttle's RMS system. It is an existing teleoperator system that includes necessary telepresence capabilities. Figure 4.2-15 summarizes the relationship of these technologies to each other and to manual operations and total autonomy and Figure 4.2-16 summarizes their characteristics.

It is conceivable under certain scenarios that the use of presently available (state-of-the-art) automated maintenance techniques could eliminate the need for space station personnel to conduct EVA operations in support of many OTV maintenance operations. Of course some contingency situations will always remain that require the versatility and capability to improvise provided by EVA personnel.

For example, it should be possible to use existing teleoperation capabilities to remove and replace (R/R) major assemblies containing multiple submodules. Further, the mechanism could transfer the major module assemblies to an interlock area where the station crew could access the assemblies for submodule repair all without any EVA. Extending this concept, the mechanism could easily be equipped with a camera or other imaging system(s) to enable the station crew to visually inspect the OTV from within the space station.

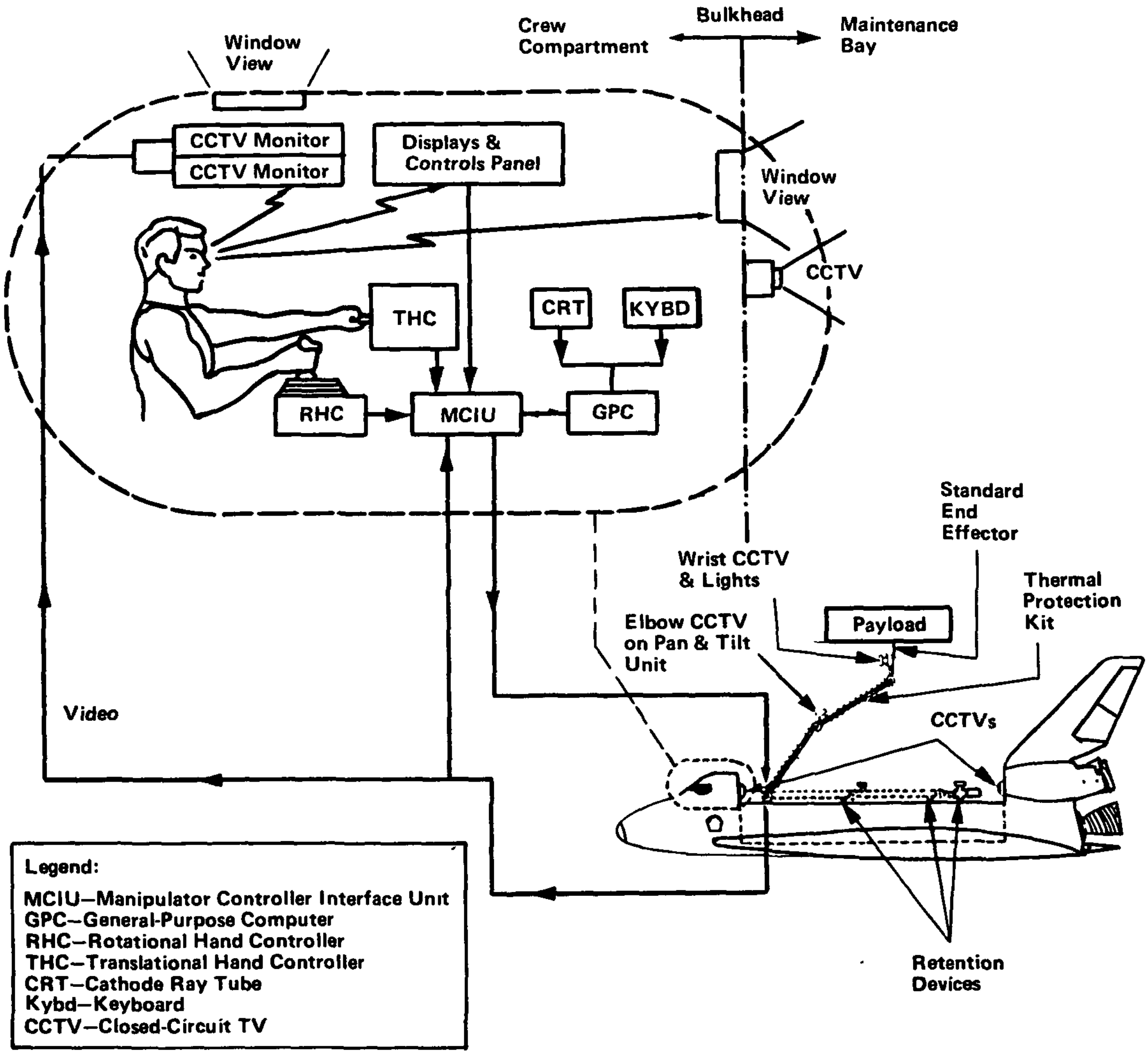


Figure 4.2-14 The Shuttle RMS Teleoperator System

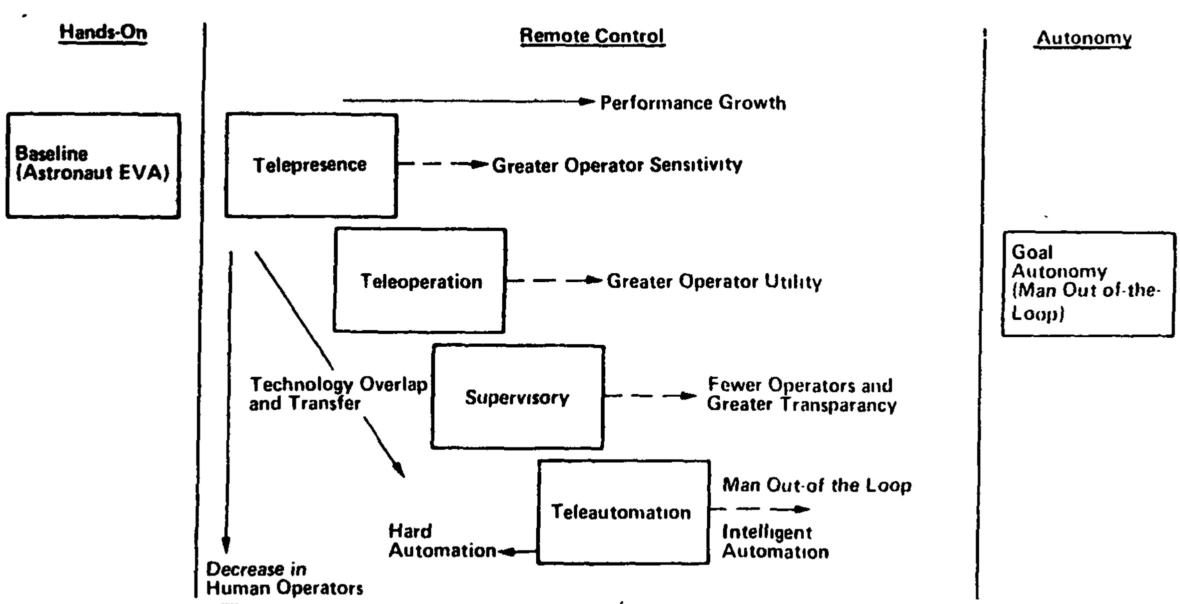


Figure 4.2-15 The Relationship of Servicing Technologies

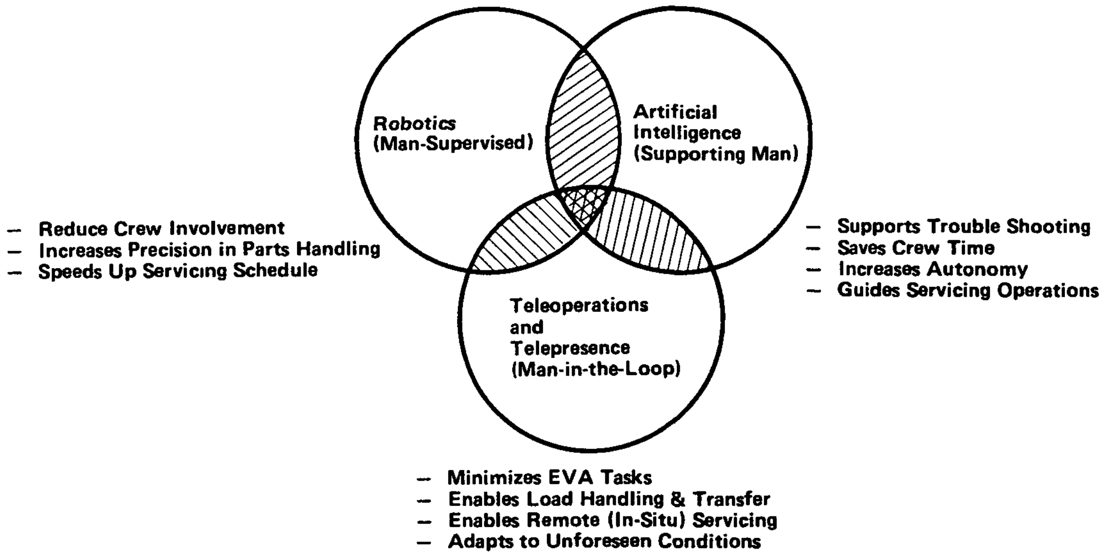


Figure 4.2-16 Characteristics of Automated Tasks

This mechanism would have to be flexible, probably programmable to some degree, and retain the option to conduct remote control operations. If the overall system engineering approach ensures the OTV's design packaging complements the mechanism's capabilities, programming needs can be minimized. For example, if the OTV were designed to break down into five major assembly groups, the maintenance mechanism would only require five program variants to R&R the major assemblies. A potential maintenance scenario employing these capabilities might be something like that described in the following paragraph.

The space station's onboard computer detects a fault during the preoperational checkout sequence and proceeds to isolate the underlying failure to a specific submodule. A space station crew member would subsequently note the error message and determine (if the system had not already) that the submodule is in major assembly group number two. The crew member could then install a preprogrammed disk, tape or chip into the robot command computer or simply authorize execution if the program were already on line. The OTV's berthing platform would then rotate to its proper orientation and the maintenance mechanism would move down its rail and proceed to remove major assembly number two. It would then carry the assembly into the interlock area. The interlock area would then be pressurized so the space station crew could access the assembly to R/R the failed submodule. The repaired major assembly could then be returned to the vehicle and reinstalled by the mechanism. The space station onboard checkout computer could then verify the repair and continue the preoperational checkout sequence.

With respect to the development and application of automation and machine intelligence technology, consider the following concerns:

1) Although we have built and tested expert systems and natural language front-ends for various applications, we have only recently become concerned about the real-time operation aspects of expert systems. Much work is required over the near term if we are to be able to successfully implement automated real-time decision-making processes for a variety of space station-based OTV maintenance applications such as we are doing in the current automated management of power systems (expert systems development) project;

2) With regard to robotics and teleoperators, once again, the emphasis is not on replacing man in space but rather on enabling a greater amount and variety of more dextrous, remote and cost effective activities. To accomplish this, significant laboratory work is required to evaluate such potential techniques as task allocation schema, available control modes, and human interaction optimization. It should be noted that in many cases unique laboratory and space facilities will be required to investigate these potential capabilities and to define specific system requirements.

In summary, with specifically relevant and timely advanced automation-related initiatives, we will be able to take a significant step beyond present capabilities and extend man's reach and utility through the use of integral system intelligence, robotics, teleoperations, sensory perception, and unique processing and packaging concepts (VHSIC). We must, and will, continue to pursue optimized use of man with the goal of increased productivity in space.

Maintenance Concept and Implications - The maintenance concept envisioned for this vehicle was discussed in some detail in Section 4.1. The aspects of the concept that warrant further discussion at this time and in this context are treated in the following paragraphs.

The concept is not firmly set, it is evolutionary and is expected to develop over time as the knowledge (definition) of OTV and space station operations and equipment improve. The philosophy underlying the concept is to limit what is to be done at the vehicle to minimize the complexity of service bay operations and thereby reduce EVA requirements and maximize the near-term use of automation. This philosophy leads to an interactive set of vehicle design approach and maintenance concept considerations.

The first point of the maintenance concept is to perform only major module removal/replacement in the service bay, not onsite component repair or replacement. Such activity would be unnecessarily difficult and time consuming when compared to available alternatives. The next point involves "intermediate"-level maintenance that is defined for this purpose as submodule R/R operations. They are envisioned as possibly being conducted on an as-necessary (contingency) basis inside the space station or in the service bay if the parent module is too big to transit the airlocks. The final point of the concept is to avoid returning the entire OTV to the ground. If the vehicle is to be space-based and if it can be highly modularized, it should only be necessary to deliver and install one or more replacement modules to correct any anomaly that occurs.

Several vehicle design approach considerations have been identified. The first is component grouping (by life, type, and location) into submodules and "major modules"--roughly comparable to today's space-replaceable unit concept. Four types of module-to-module and module-to-structure interfaces are envisioned. These are fluid, electric power, mechanical, and data. Mechanical module interface options (roughly illustrated in Fig. 4.2-17) include multiple-fastener direct, single-fastener direct (like MMS), and indirect through a transition plate. There similarly are a number of alternative options for each of the other module interface types, each with specific advantages and disadvantages.

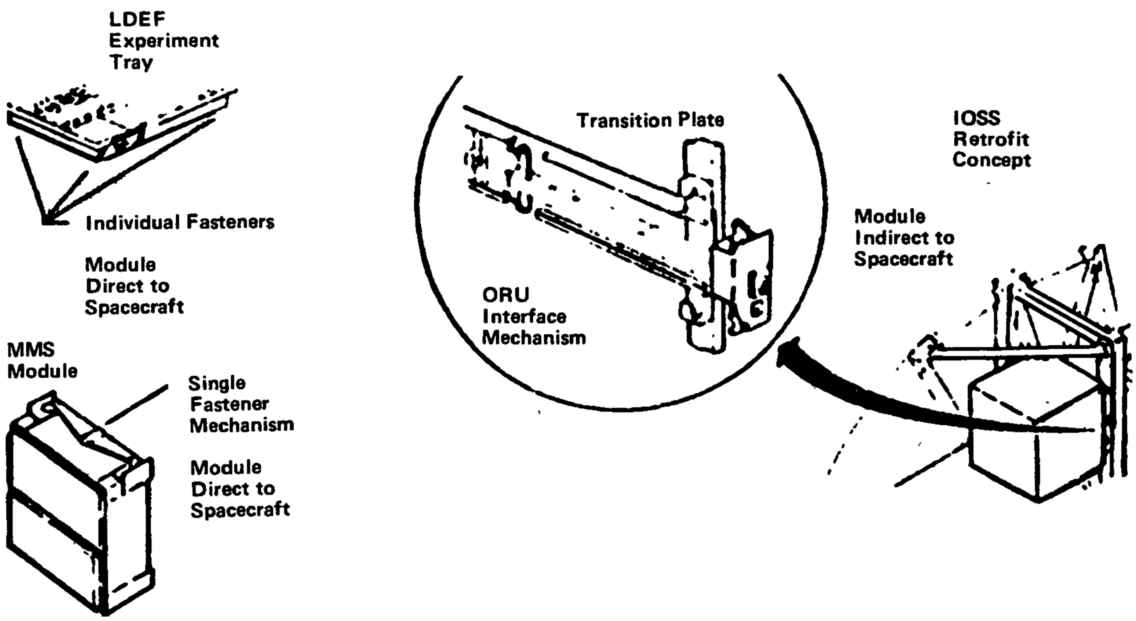


Figure 4.2-17 Mechanical Module Interface Options

Although it will be highly desirable to standardize and minimize the number of interfaces (particularly for the major modules), the design process may possibly mandate that a variety of options be employed for one or more of the interface types because of varied operational and maintenance requirements and characteristics. Figure 4.2-18 shows two concepts for the engine/OTV interface that would both employ a variety of interface types.

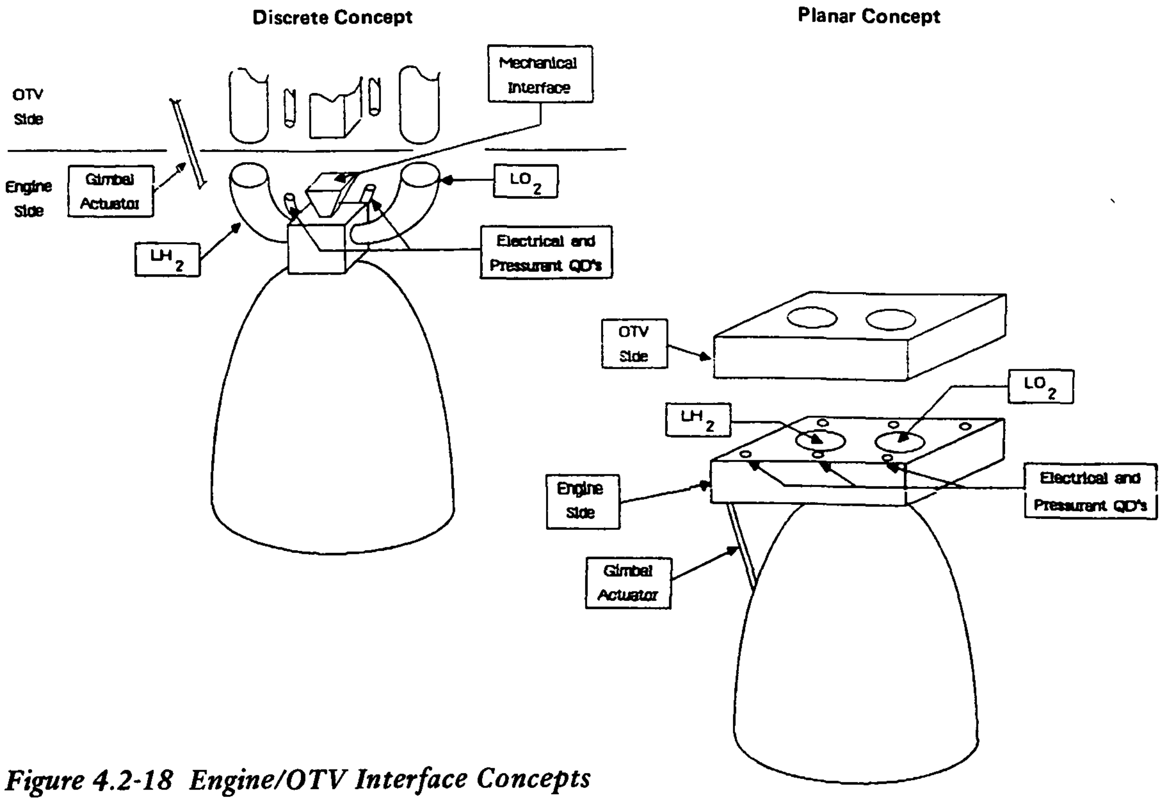


Figure 4.2-18 Engine/OTV Interface Concepts

The implications of this standardization objective extend beyond design of the OTV itself. We will need to carefully develop tasks, processes and tools common to the widest possible variety of space-based servicing operations. A few of the other space station activities that would make use of these common facilities are payload assembly and/or repair, OMV servicing, and expendable propulsion vehicle servicing.

The implications of these points are:

1) To seek common onorbit servicing capabilities and requirements for the widest possible range of applications;

2) To seek OTV, OMV, MMS, and other systems (vehicles and payloads) commonality at the widest possible range of levels—from major module to submodule, and, possibly, to the component level.

The following basic maintainability principles also provide design objectives for OTV system development:

1) Design to minimize the complexity of maintenance tasks such as adjustment, inspections and calibrations;

2) Design for optimum interchangeability and use of standardized (or commercial) items;

3) Design for rapid and positive recognition of equipment malfunction or marginal performance;

4) Design for minimal resource requirements in accomplishing maintenance tasks;

5) Design for rapid and positive identification of the replaceable defective part, assembly or component;

6) Design to require minimum maintenance skills and training;

7) Design to require minimum numbers and types of tools and test equipment (special and standard);

8) Design for optimum access to all equipment and components requiring maintenance, inspection, removal or replacement;

9) Design for maximum safety for both equipment and personnel;

10) Design so the mean time to accomplish scheduled and unscheduled maintenance is sufficiently low to assure the required operational availability of the equipment;

11) Design to enhance and facilitate organizational- and intermediate-level maintenance action;

12) Design to require a minimum number and type of repair parts and assemblies;

13) Provide technical data for easy use (AI maintenance tutors, etc) concurrently with the equipment;

14) Maximize the extent that performance can be verified, malfunctions anticipated and located, and calibration performed.

In regard to the impact of advanced automation on the OTV maintenance concept, a few additional points need to be made. The first is that the OTV modules, support equipment and maintenance procedures should be designed for multimode operations. Whether we baseline EVA activity, automated module replacement, or some intermediate capability, the need to accommodate evolution and contingency operations will remain. As Figure 4.2-19 depicts, it should ideally be possible to conduct these multimode operations with only minimal mode-to-mode change--substitute a proxy for the man but use the same work carriage and module service tool(s).

The next point is that module size and interfaces must be carefully engineered to be compatible with the baseline (1990s) automation capabilities. And the last point, a related one, is that small components/assemblies must be carefully grouped to attain optimum module and submodule size without inextricably confusing the ability to develop reliability calculations and conduct maintenance.

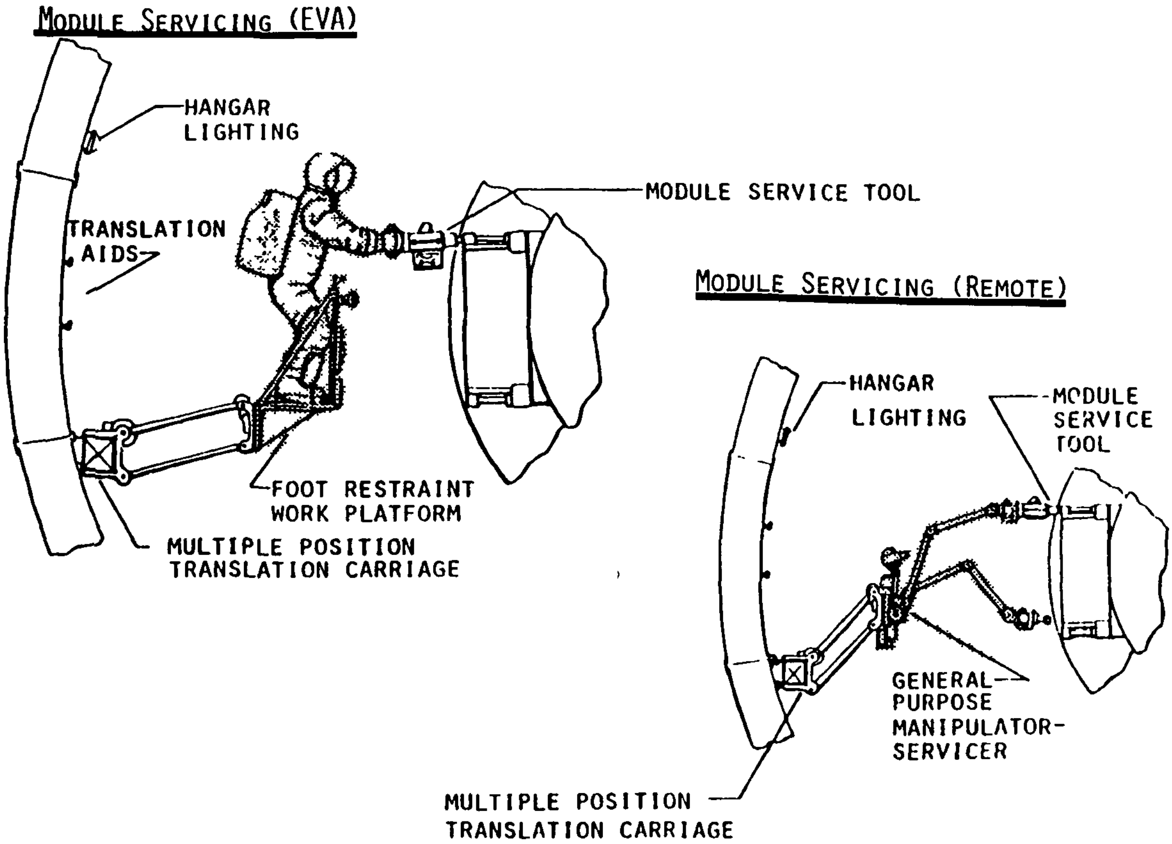


Figure 4.2-19 System Design for Multimode Maintenance Operations

OTV Storage - This sequence (if indeed it can qualify as a sequence) is the transition between postflight and preflight operations. The processes involved are expected to focus only on thermal conditioning of the tanks and possibly conditioning portions of the power system. It could, however, prove advantageous to include provisions for data system access within the storage facility to accommodate automated checkout and reprogramming while the OTV is there. These provisions would permit extended use of such other maintenance facility capabilities as inspection and module changeout.

Transfer Stack Assembly - The transfer stack assembled would always be a function of the mission to be performed. It would include a payload for deployment missions, a servicer-kit and possibly an OMV for service missions and some retrieval missions, and only the OTV itself for many retrieval missions.

The sequence is envisioned as including two distinct activities. The first would be employed only if a scenario using the OMV for close-in mobility operations or remote recovery operations were selected. This activity is mating of the OMV and the OTV. The scenario for the activity is to use available manipulators in the facility to position the two vehicles and hold them while the docking interface was activated. Tasks in the activity would be essentially similar to those in the destacking activity previously discussed.

The second activity, joining the payload (or the servicing kit) to the OTV, would routinely be performed. The single exception would be for recovery missions that would not require front-end kit (manipulator) preparation before orbit transfer. Again, this activity contains essentially the same tasks as the destacking activity. The noticeable difference would probably be in the positioning accuracy required before docking.

Remaining Sequences - The balance of the sequence depicted in Figure 4.2-5 are checkout operations, consumables loading, preparation for launch, proximity operations, and launch (that really begins the mission operations sequence). These will not be treated further because they are either repeats of previously discussed sequences (or portions thereof) or rearrangements of activities treated in previously discussed sequences. They do not introduce any new servicing considerations given the present depth of analysis.

4.3 RELIABILITY ANALYSIS AND EFFECTS OF NONINDEPENDENT FAILURES

A reliability analysis was performed that considers the effects of nonindependent failures (failures that render the entire main propulsion system inoperative) on mission success probability. The purpose of generating main propulsion system mission reliability numbers (other than for the conclusions that can be drawn from them alone) was to incorporate the delta reliability between various engine configurations into the life-cycle cost comparisons of these configurations to account for mission loss costs. This is discussed in Section 4.4 of this report.

A range of engine single-burn reliability from 0.994 to 0.999 was chosen to bound the problem. Also, a nonindependent failure rate from 0 to 5% was selected. Then, for a six-burn mission, the main propulsion system reliability for completing the mission was calculated for various numbers of engines (from 1 to 4) and engine-out capability. The results are shown in Figures 4.3-1 through 4.3-3. The obvious results show that multiple engines (3 or greater) provide the highest mission reliability for low (less than about 2%) nonindependent failure rates. Correspondingly, two engines provide the highest reliability for the higher nonindependent failure rates. The crossover points of highest mission reliability vary as a function of single burn reliability, however. Agreement concerning reasonable estimates of engine single-burn reliability and nonindependent failure rates will have to be reached within the industry before firm conclusions can be drawn. However, the trends shown here should aid in the definition of engine and main propulsion system design decisions.

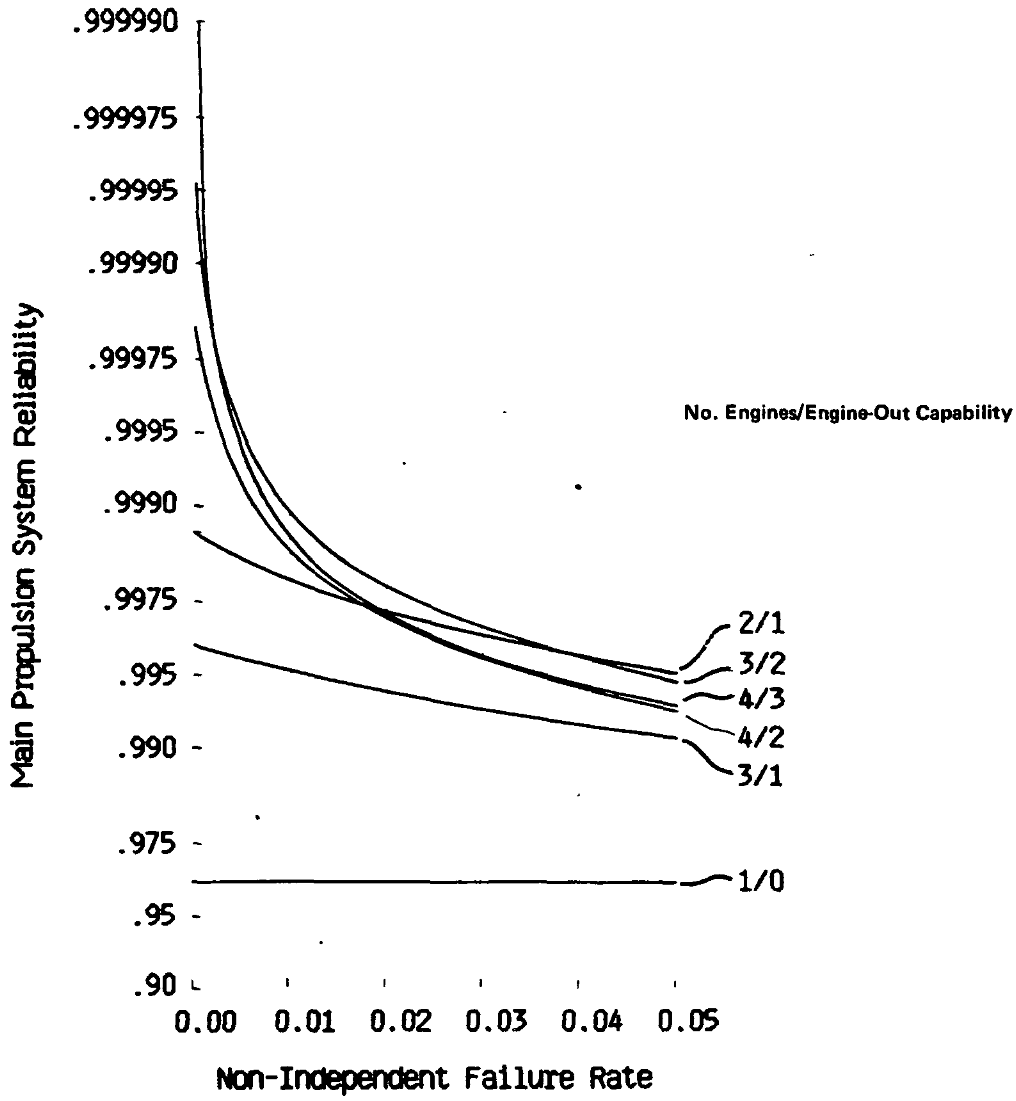


Figure 4.3-1
Main Propulsion System Reliability—Single-Burn Reliability = 0.994

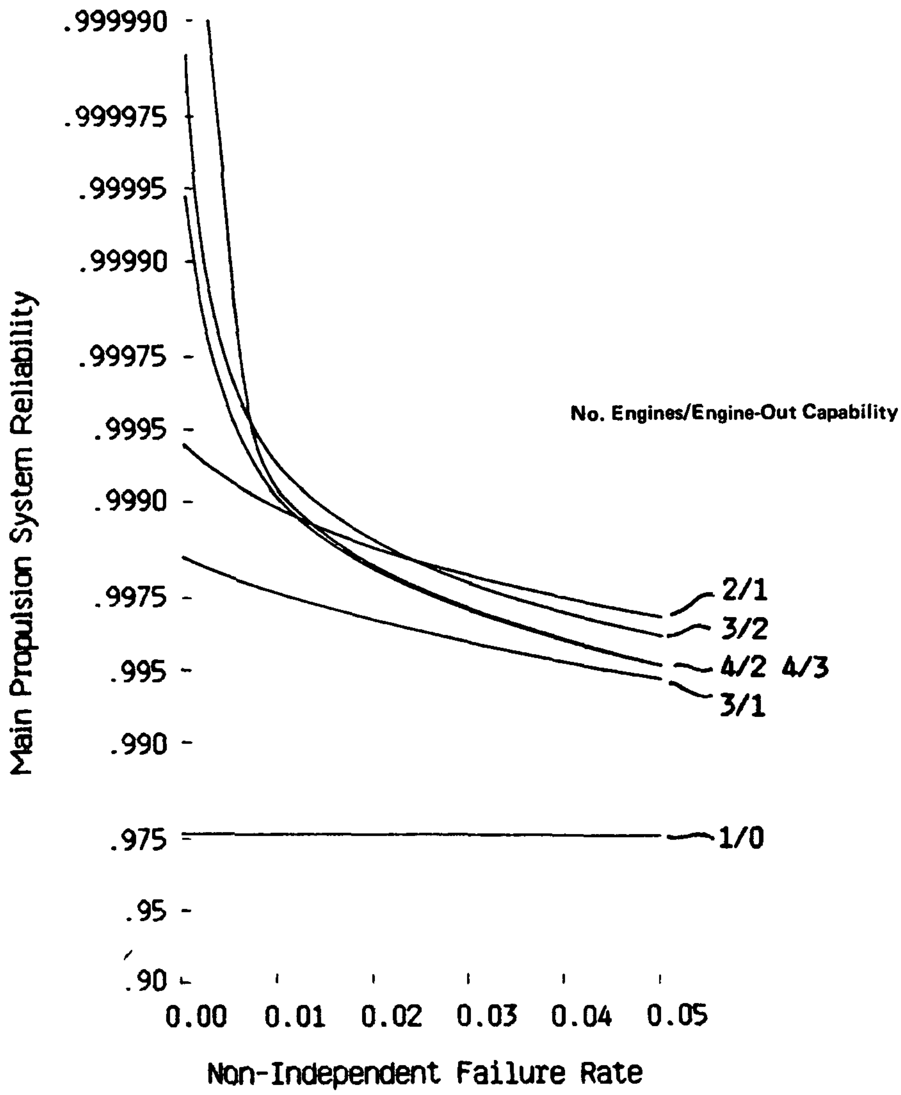


Figure 4.3-2
Main Propulsion System Reliability—Single-Burn Reliability - 0.996

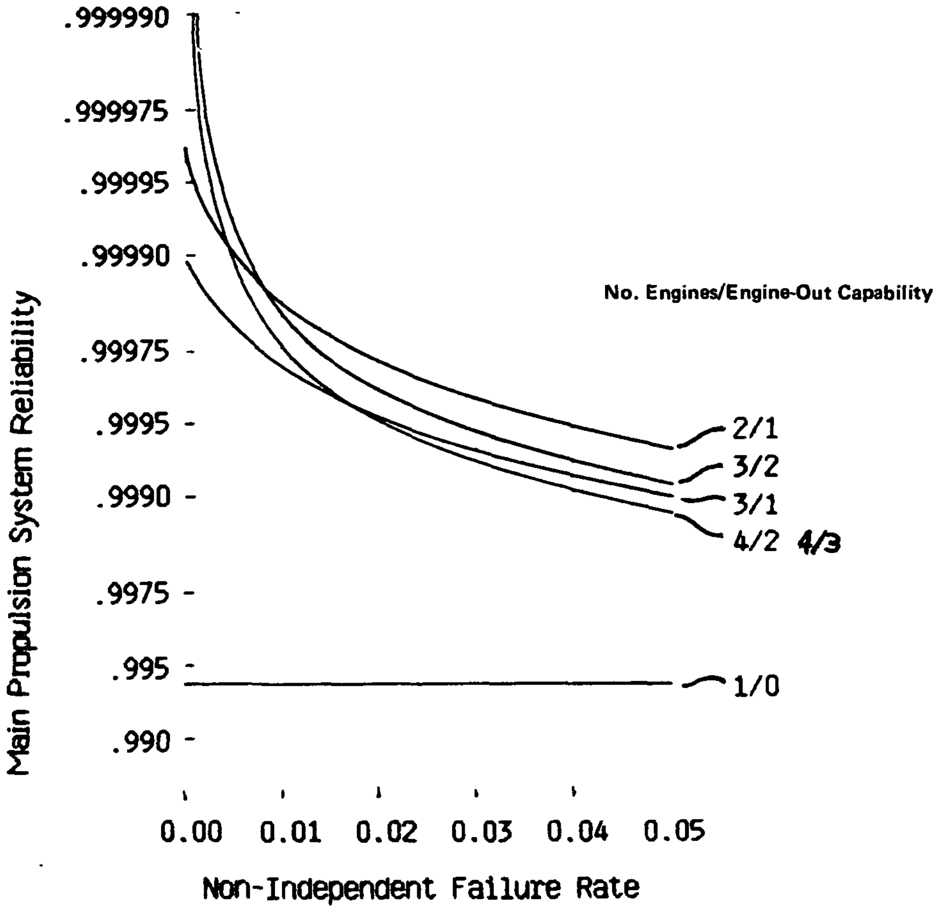


Figure 4.3-3 Main Propulsion System Reliability—Single-Burn Reliability = 0.999

4.4 LIFE-CYCLE COST (LCC) ANALYSIS

The major efforts reported here include the development of a LCC model for the development, production, and operation of the baseline space-based reusable OTV. The cost trends and comparisons that the LCC model was used to generate data for are also included.

4.4.1 LCC Model Development

Figure 4.4-1 depicts the methodology used for development of the OTV LCC model. Table 4.4-1 defines the terms used for each of the model development stages and this is followed by a description of the modeling approach as applicable to the OTV.

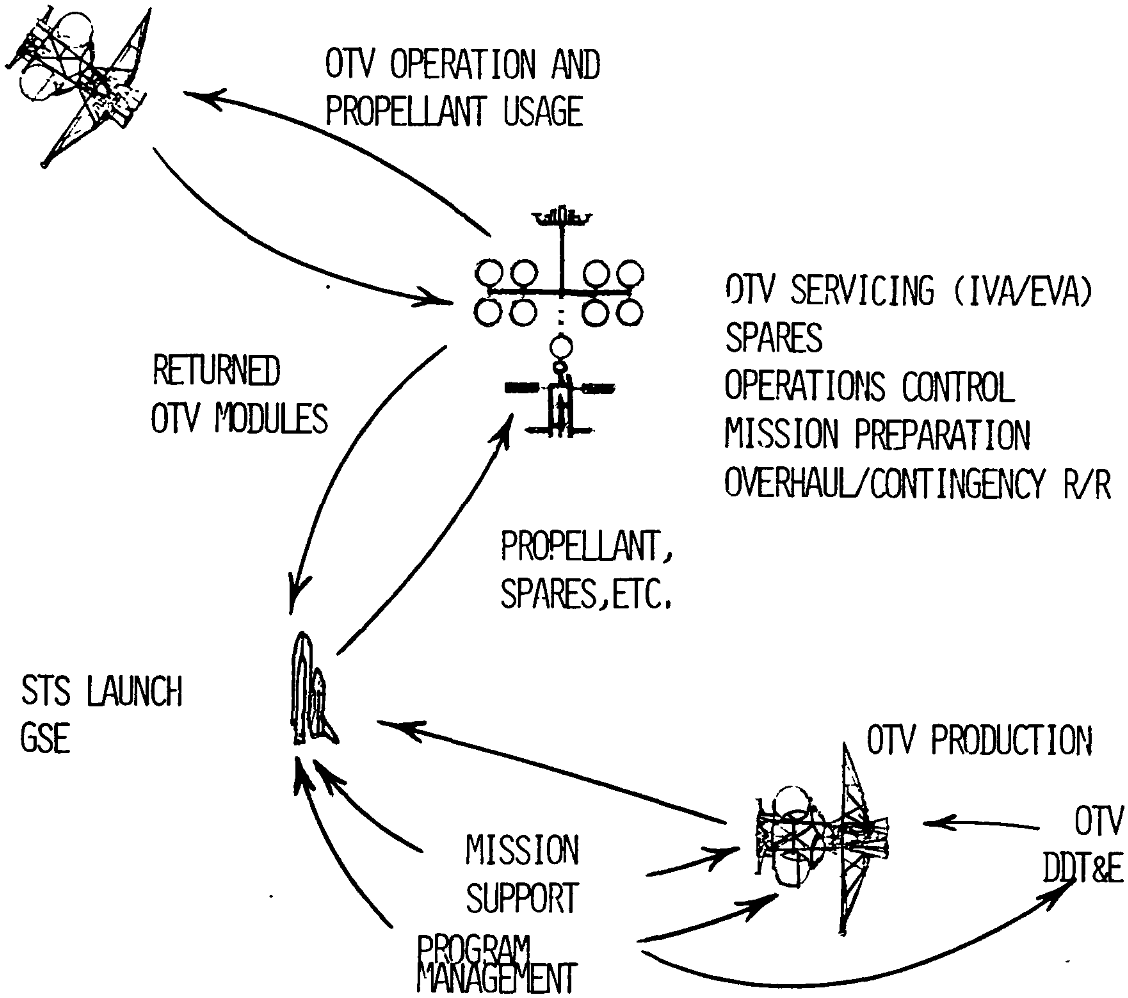


Figure 4.4-1 OTV Life-Cycle Cost (LCC) Model Scenario

Table 4.4-1 LCC Model Development Stages

Stage	Description
Problem Formulation	Identify the problem requiring solution.
System Definition	Determine the boundaries, restrictions and measure of effectiveness to be used in defining the system to be studied.
Modeling	Determine the most suitable form of modeling for solution of the problem.
Model Formulation	Reduce the abstraction of the real system to a logic flow diagram.
Data Preparation	Identify the data needed by the model and reduce the data to an appropriate form.
Model Translation	Describe the model in a language suitable for computer use.
Validation	Assure, to an acceptable level of confidence, that an inference drawn from the model about the real system will be correct.
Experimentation	Execute the model to generate the desired data and then perform sensitivity analyses.
Interpretation	Draw inferences from the data generated by the model.
Implementation	Put the model and results to use.
Documentation	Record the project activities and results and document the model and its use.

Problem Formulation - All the major cost elements associated with the deployment, design, development, test and evaluation (DDT&E), production, operation and maintenance of an orbital transfer vehicle (OTV) were identified. Cost sensitivities were analyzed to assist in obtaining an optimal design in terms of operational capability and life-cycle cost.

System Definition - Figure 4.4-1 depicts the OTV development and operating scenario. Total cost was developed for DDT&E and production of two OTVs, ground preparation and launch via the shuttle. The OTV was assumed to be docked at and operated from the space station. All OTV missions were costed with propellant use and EVA time. Scheduled and unscheduled repair actions, including EVA, IVA, spares, ground resupply etc were costed against the OTV.

Modeling - The most appropriate approach for developing cost predictions for conceptual design studies was through cost estimating relationships (CERs) based on known design parameters and factors from similar programs and reasonably substantiated ground rules and assumptions. The CERs were developed for all major cost elements associated with DDT&E, production and life-cycle operation and maintenance of the OTV. Certain parameters within the CERs may be fixed values, whereas others were variables to enable sensitivity analyses to be performed. These sensitivity analyses were beneficial in achieving the optimal design at the least LCC.

Data Preparation - The following tabulates the baseline ground rules and parameters used in development of the OTV LCC model.

- All costs reported in millions of 1984 $.
- One failure per two OTV missions (one OTV subsystem module R/R).
- $30k per hour per man for EVA.
- $10k per hour per man for IVA.
- Seven missions per year for seven years.
- 30 missions between major overhauls (both modules and engines).
- Four hours EVA R/R time for unscheduled failures.
- Four hours IVA time for unscheduled failures.
- Orbiter fees for OTV (and spares) transportation based on percent of total orbiter cargo weight capacity.
- OTV ground processing factored from STS launch operations.
- Flight operations at JSC $12k/hour.
- STS transportation cost $1476/pound.
- Average module weight 33 pounds.
- 13 module types (52 total modules).
- 50 manhours EVA turnaround time per mission.
- 112 manhours IVA turnaround time per mission.
- One, two, three and four engine configurations.
- Propulsion system total thrust 15,000 lbf.

The following CERs have been developed for inclusion in the OTV life cycle cost model. Each CER algorithm and source is presented, followed by the validity range and a description of the associated rationale.

Structure/Mechanisms (MSFC)

$STRRDM = 0.821 \times STRUC^{0.437}$
$STRUCM = 0.064 \times STRUC^{0.545}$

where: STRRDM = Structure/mechanism DDT&E cost
STRUCM = Structure/mechanism unit cost
STRUC = Structure/mechanism weight in lbm.
Validity range: 60 to 2000/lbm

The structure/mechanism subsystem consists of a primary structure of an aluminum tubular truss design. It includes trunnion fittings and aluminum panels for the mounting of avionic components and assemblies. Docking and interface mechanisms for payloads and the orbiter are also included.

Thermal Control (MSFC)

$THRMRDM = 0.821 \times THERM^{0.437}$
$THRMUCM = 0.064 \times THERM^{0.545}$

where: THRMRDM = Thermal control subsystem DDT&E cost
THRMUCM = Thermal control subsystem unit cost
THERM = Thermal control subsystem weight in lbm.
Validity range: 60 to 2000 lbm

The thermal control subsystem maintains the temperature of the spacecraft, mission equipment, and propellant within allowable limits in certain orbital conditions. The thermal control subsystem includes paint, insulation, radiators, heaters, louver assemblies, temperature sensors, heat pipes, and a thermodynamic vent system. Thermal control can be accomplished either passively or actively. Passive control often means nothing more than a coat of reflective paint. Active control ranges from mechanical shutters to heat pipes running through the structure.

Propulsion (Space Division)

$PROPRDM = 1.668 + (0.126 \times PROP)$
$PROPUCM = 0.43 \times PROP^{0.494}$

where: PROPRDM = Propulsion subsystem DDT&E cost
PROPUCM = Propulsion subsystem unit cost
PROP = Propulsion subsystem weight in lbm
Validity range: 200 to 4000 lbm

The propulsion subsystem provides a reaction force for orbit-keeping, orbit-changing, and attitude control. The propulsion subsystem includes a reaction control system, fuel and oxidizer tanks, pressurization and accumulator system, and flow system hardware. Specifically excluded are costs for an advanced engine and onorbit propellant costs.

Electric Power (MSFC)

$EPSRDM = 0.597 \times EPS^{0.584}$
$EPSUCM = 0.042 \times EPS^{0.784}$

where: EPSRDM = Electric power subsystem DDT&E cost
EPSUCM = Electric power subsystem unit cost
EPS = Electric power subsystem Weight in lbm
Validity Range: 100 to 10,000 lbm

The electric power subsystem stores, converts, regulates, and distributes electrical energy to and between spacecraft components. The electric power subsystem includes batteries, fuel cells, regulators, converters, distribution units and wiring harnesses.

Guidance, Navigation and Control (MSFC)

$GNCRDM = 3.08 \times GNC^{0.516}$
$GNCUCM = 0.5763 \times GNC^{0.494}$

where: GNCRDM = Guidance, navigation and control subsystem DDT&E cost
GNCUCM = Guidance, navigation and control subsystem unit cost
GNC = Guidance, navigation and control subsystem weight in lbm
Validity Range: 200 to 4000 lbm

The guidance, navigation and control subsystem generates control signals that interface with the propulsion subsystem to produce thruster forces for translation and stabilization of the spacecraft orbit. The guidance, navigation and control subsystem includes inertial reference units, accelerometers, range-rate sensors, valve control electronics, and attitude computers.

Data Management and Communications (MSFC)

$DMCRDM = 0.615 \times DMC^{0.653}$
$DMCUCM = 0.053 \times DMC^{0.917}$

where: DMCRDM = Data management and communications subsystem DDT&E cost
DMCUCM = Data management and communications subsystem unit cost
DMC = Data management and communications subsystem weight in lbm
Validity Range: 50 to 1000 lbm

The data management and communications subsystem processes information on spacecraft conditions and mission progress, stores or transmits such data to the ground, and receives commands from the ground and initiates their execution. The data management and communications subsystem consists of processors, memory storage units, coders and decoders, timing units, control consoles, antennas, transmitters, and receivers. The software needed to operate the data collecting and transmitting functions is also included.

Aerobrake (MSFC)

ABRDM = 1.668 x (0.126 x ABWT)
ABUCM = 0.064 x $ABWT^{0.545}$

where: ABRDM = Aerobrake DDT&E cost
ABUCM = Aerobrake unit cost
ABWT = Aerobrake weight in lbm
Validity Range: 60 to 2000 lbm

The aerobrake CERs are the same as those used for structure/mechanisms. As the design matures, an update should be made to account for the aerobrake specifically.

Advanced Engine (Pratt & Whitney)

ENGUNITWT = 28.33 x $THRUST^{0.2825}$
ENGWT = ENGUNITWT x ENG
ENGDD = 306.25 + (0.0029 x THRUST)
ENGUNITCOST = 1.963 + (0.000103 x THRUST)
ENGUC = ENGUNITCOST x ENG

where: ENGUNITWT = The single engine weight in lbm
THRUST = Single engine thrust in lbf
ENGWT = Engine system weight in lbm
ENG = Number of single engines on the OTV
ENGDD = Engine DDT&E cost
ENGUNITCOST = Unit cost of a single engine
ENGUC = Engine system unit cost
Validity Range: 250 to 500 lbm
3000 to 15000 lbf

The engine subsystem weight equations and CERs were developed from data supplied by Pratt & Whitney. They represent the best-fit regression curves of single data points.

Spacecraft DDT&E Cost

DDTE = STRRDM + THRMRDM + PROPRDM + EPSRDM + GNCRDM + DMCRDM + ABRDM + ENGDD

where: DDTE = Spacecraft DDT&E cost
STRRDM = Structure/mechanism subsystem DDT&E cost
THRMRDM = Thermal control subsystem DDT&E cost
PROPRDM = Propulsion subsystem cost
EPSRDM = Electric power subsystem cost
GNCRDM = Guidance, navigation and control subsystem cost
DMCRDM = Data management and communications subsystem cost
ABRDM = Aerobrake DDT&E cost
ENGDD = Advanced engine DDT&E cost

The DDT&E equation gives design, development, test, and evaluation cost of the total OTV. A detailed discussion of each subsystem is given in the preceding pages.

Flight Hardware Cost

FH = STRUCM + THRMUCM + PROPUCM + EPSUCM + GNCUCM + DMCURM + ABUCM + ENGUC

where: FN = OTV flight hardware cost
STRUCM = Structure/mechanism subsystem DDT&E cost
THRMUCM = Thermal control subsystem unit cost
PROPUCM = Propulsion subsystem unit cost
EPSUCM = Electric power subsystem unit cost
GNCUCM = Guidance, navigation and control subsystem unit cost
DMCUCM = Data management and communications subsystem unit cost
ABUCM = Aerobrake unit cost
ENGUC = Advanced engine system unit cost

The flight hardware equation gives the unit cost of a single OTV. Each subsystem is discussed in detail in the preceding pages.

Spacecraft Weight

SCWT = (STRUC + THERM + PROP + EPS + GNC + DMC + ABWT + ENGWT)x1.15

where: SCWT = OTV Weight in lbm
STRUC = Structure/mechanism subsystem weight in lbm
THERM = Thermal control subsystem weight in lbm
PROP = Propulsion subsystem weight in lbm
EPS = Electric power subsystem weight in lbm
GNC = Guidance, navigation and control subsystem weight in lbm
DMC = Data management and communications subsystem weight in lbm
ABWT = Aerobrake weight in lbm
ENGWT = Advanced engine system weight in lbm

The OTV weight equation gives the total weight for a single OTV.

Space Station Crew Cost

$$SMAINT = (SEVA \times EVA + SIVA \times IVA) \times MISYR \times NYO/10^6$$
$$USMAINT = (EVARR \times EVA + IVARR \times IVA) \times FAIL \times MISYR \times NYO/10^6$$

where:
- SMAINT = Scheduled maintenance labor cost
- SEVA = Scheduled EVA time in hours
- EVA = IVA cost per hour
- SIVA = Scheduled IVA time in hours
- IVA = IVA cost per hour
- MISYR = Number of missions flown by OTV per year
- NYO = Number of years of OTV operation
- USMAINT = Unscheduled maintenance labor cost
- EVARR = Unscheduled EVA time in hours
- IVARR = Unscheduled IVA time in hours
- FAIL = Number of failures per OTV mission

The space station crew cost equations represent the cost of EVA and IVA time while on orbit.

Program Support Elements (MSFC)

$$GSE = 0.10 \times DDTE$$
$$SEID = 0.10 \times (DDTE + GSE)$$
$$PMD = 0.05 \times (DDTE + GSE + SEID)$$
$$IACP = 0.20 \times FH$$
$$SEIP = 0.10 \times (FH + IACP)$$
$$PMP = 0.05 \times (FH + IACP + SEIP)$$

where:
- GSE = Ground support equipment cost
- DDTE = OTV DDT&E cost
- SEID = System engineering and integration cost
- PMD = Program management cost for DDT&E phase
- IACP = Installation, assembly, and checkout cost for DDT&E phase for OTV subsystems
- FH = OTV flight hardware cost
- SEIP = System engineering and integration cost for production phase
- PMP = Program management cost and integration cost for production phase

The program support equations were derived from NASA costing information. They represent an average percentage factor of certain elements.

Propellant Cost (Derived from Rev 6 Mission Propellant Requirements)

PROPWT = [BPROP + 148x (SCWT-5096) - 7660 x (ISP - 480)]
PROPLB = SHLC x 10^6 / 65000
PROPCOST = PROPLB x PROPWT / 10^6

where:
PROPWT = Total propellant weight in lbm
BPROP = Baseline propellant weight in lbm
SCWT = Total OTV weight in lbm
ISP = Advanced engine specific impulse in seconds
PROPLB = Propellant cost per lbm
SHLC = Shuttle launch cost in millions of dollars
PROPCOST = Total propellant cost in millions of dollars

The propellant cost equation calculates the total propellant cost based on the basic propellant usage plus the effects of changes in OTV dry weight and engine specific impulse. The cost per pound is calculated assuming a 65,000-lb cargo capacity for the shuttle.

Operation and Support Factors (Engine Replacement and Spares Scenario)

ENGREPCOST = (ENGUNITCOST x ENGREP x MISYR x NYO) +
(MISYR x NYO/MISOV x ENGUC)

SPARES = FAIL x MISYR x NYO
SPARESWT = SPARES x MODWT
OVWT = MODS x MODWT
MODULEWTS = SPARESWT + OVWT
SPARESCOST = (0.22 X MODULEWTS$^{0.44}$) + ENGREPCOST
LOGISTICS = (0.6 x FH) + (0.02 X SPARESCOST x NYO) +
(0.0005 X DDTE x NYO/3)
GC = 15 x NYO

where:
ENGREPCOST = Total cost for engine replacements over the life of the OTV
ENGUC = Advanced engines system weight in lbm
ENGREP = Number of engines replaced per OTV mission
MISYR = Number of spacecraft missions per year
NYO = Number of years of OTV operation
MISOV = Number of OTV missions between scheduled overhauls
SPARES = Number of replacement modules required over the OTV lifetime
FAIL = Number of failures per OTV mission
SPARESWT = Contingency replacement module weight in lbm
MODWT = Average replacement module weight in lbm
OVWT = Replacement module weight, in lbm, per OTV overhaul
MODS = Number of replacement modules on the OTV

MODULEWTS = Total replacement module weight, in lbm
SPARESCOST = Total cost of OTV spares
LOGISTICS = Total cost of training, technical data, inventory control, and sustaining engineering
FH = OTV flight hardware cost
DDTE = OTV DDT&E cost
GC = Ground control cost of OTV operations

The operation and support equations represent the cost, except for labor, to maintain and operate an OTV at the space station.

Mission Loss Cost (two loads propellant and turnaround operations)

MISLOSS = (PROPCOST + SMAINT + USMAINT + SPARESCOST + LOGISTICS) x MPS x 2

where: MISLOSS = Cost to recover a lost mission
PROPCOST = Total propellant cost
SMAINT = Scheduled maintenance labor cost
USMAINT = Unscheduled maintenance labor cost
SPARESCOST = Total cost for spacecraft spares
LOGISTICS = Total cost for training, technical data, inventory control, and sustaining engineering
MPS = 1 - Main propulsion system reliability

The mission loss equation represents the cost to complete failed missions using a second OTV and to retrieve the disabled OTV on the return trip. The number of mission losses is calculated by subtracting the main propulsion system reliability, which is the probability that the system will function properly, from 1.00 (which represents 100% success) and multiplying by the total number of missions flown.

Life-Cycle Cost

LCC = DDTE + FH + SEID + SEIP + GSE + PMD + PMP + SPARESCOST + SMAINT + USMAINT + LOGISTICS + GC + PROPCOST + MISLOSS + IACP

where: LCC = OTV life cycle cost
DDTE = OTV DDT&E cost
FH = OTV flight hardware cost
SEID = System engineering and integration cost for the DDT&E phase
SEIP = System engineering and integration cost for the production phase
GSE = Ground support equipment cost
PMD = Program management cost for the DDT&E phase
PMP = Program management cost for the production phase

SPARESCOST = Total cost of OTV spares
SMAINT = Scheduled maintenance labor cost
USMAINT = Unscheduled maintenance labor cost
LOGISTICS = Total cost of training, technical data, inventory control, and sustaining engineering
GC = Ground control cost for OTV operations
PROPCOST = Total propellant cost
MISLOSS = Cost to recover a lost mission

The life-cycle cost equation represents the entire cost of the OTV from its inception throughout its useful life.

Model Translation - The CERs were translated into BASIC and entered on an IBM personal computer.

Model Validation - Each CER was developed from data obtained from numerous previous and current space programs. Historical data are probably the most valid and accurate from which to develop CERs and these data were used in the preparation of the OTV CERs. Much of the data was contained in the Space Station Cost Model (PRC D-2115) and the remainder were developed through analysis of OTV design requirements.

Model outputs of dollar values for the cost groups of DDT&E, production and operations/maintenance were compared with actuals or other similar complexity programs and found to be of the same nominal order of magnitude.

4.4.2 Life-Cycle Cost Results

Life-Cycle Costs for Engine Configurations - The LCC and propulsion system performance reliabilities were developed for 1, 2, 3, and 4 engines with single-engine burn reliabilities of 0.994, 0.996, and 0.999, and nonindependent failure rates (NIFR) of 0, 0.03 and 0.05. Figures 4.4-2 through 4.4-19 reflect OTV LCC and propulsion system reliability for the respective number of Pratt & Whitney and Aerojet engines. The engines selected for LCC/performance analysis were 1 engine with a 0 engine-out capability, 2 engines with a 1 engine-out capability, 3 engines with a 2 engine-out capability and 4 engines with a 3 engine-out capability.

Figure 4.4-2

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =0)

BURN REL=0.994

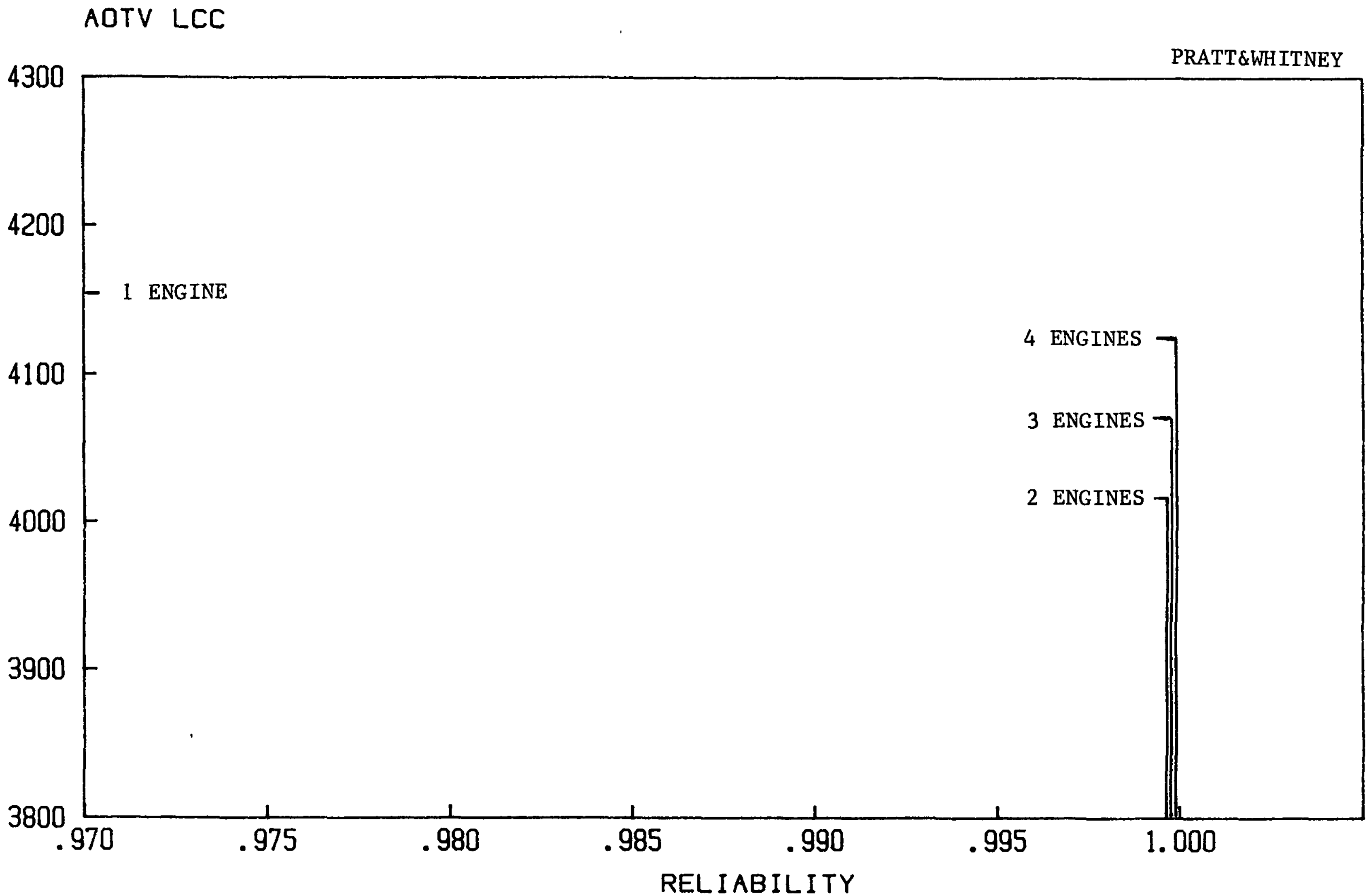


Figure 4.4-3

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.03)

BURN REL=0.994

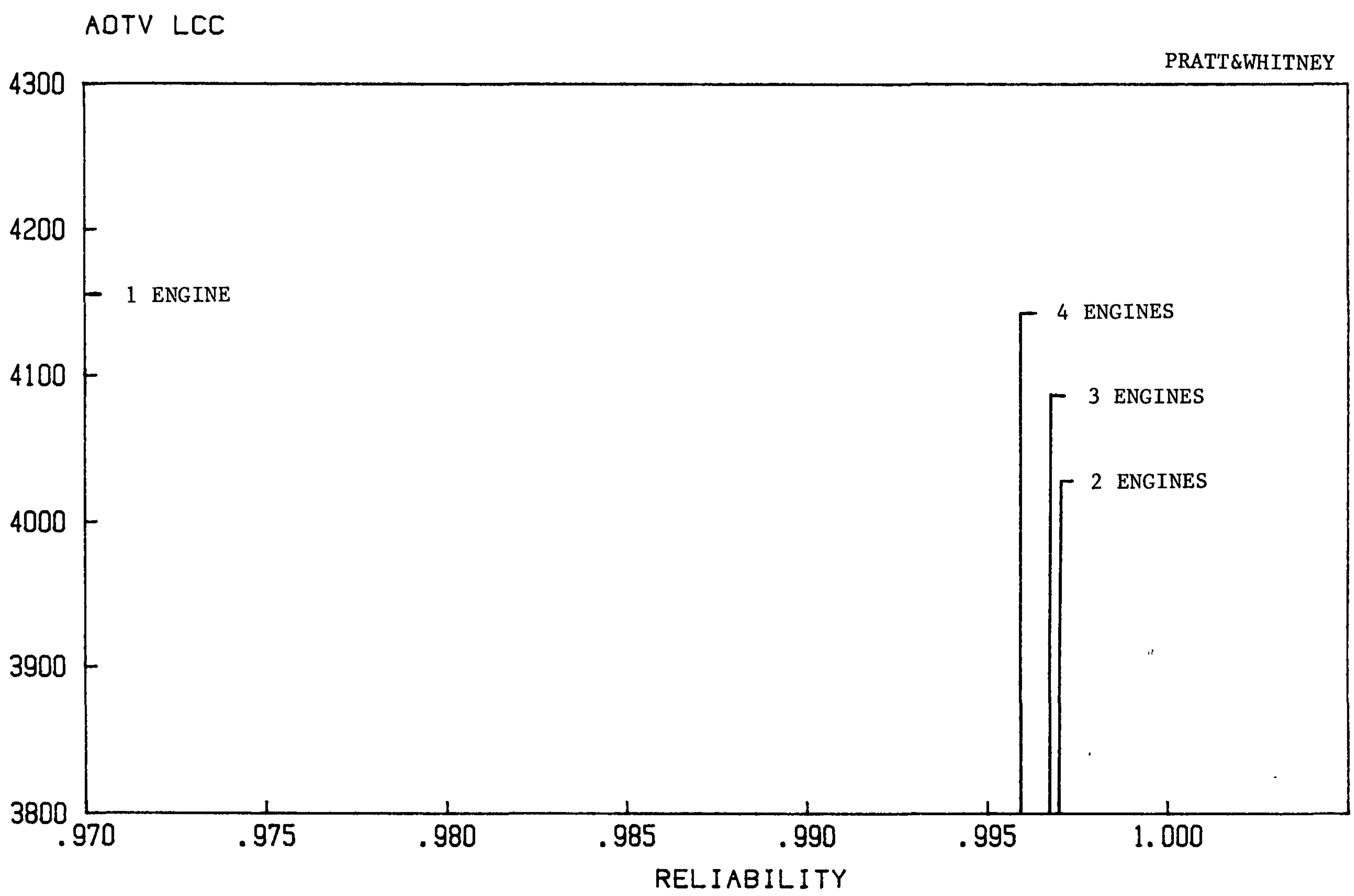


Figure 4.4-4

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.05)

BURN REL=0.994

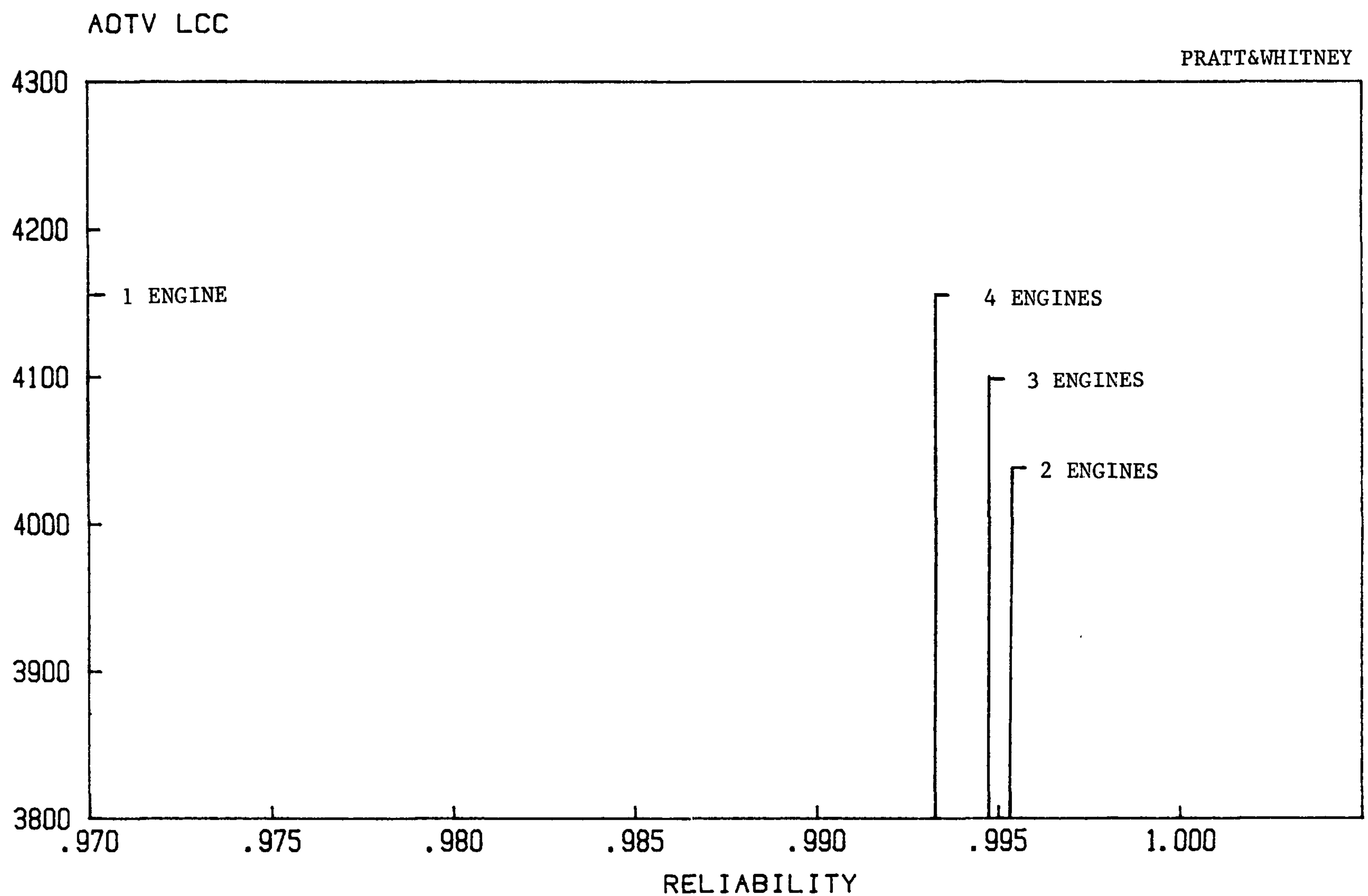


Figure 4.4-5

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =0)

BURN REL=0.996

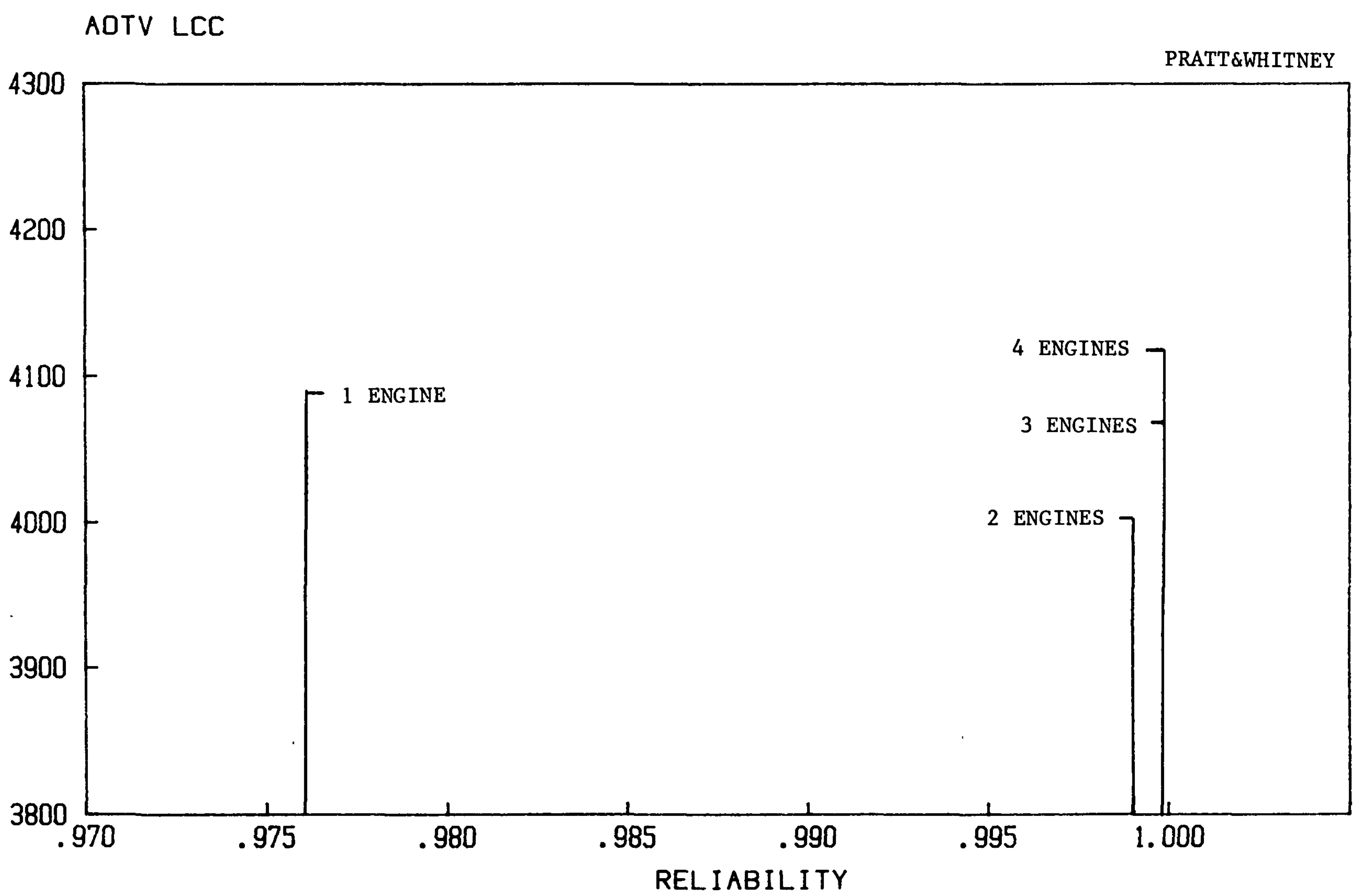


Figure 4.4-6

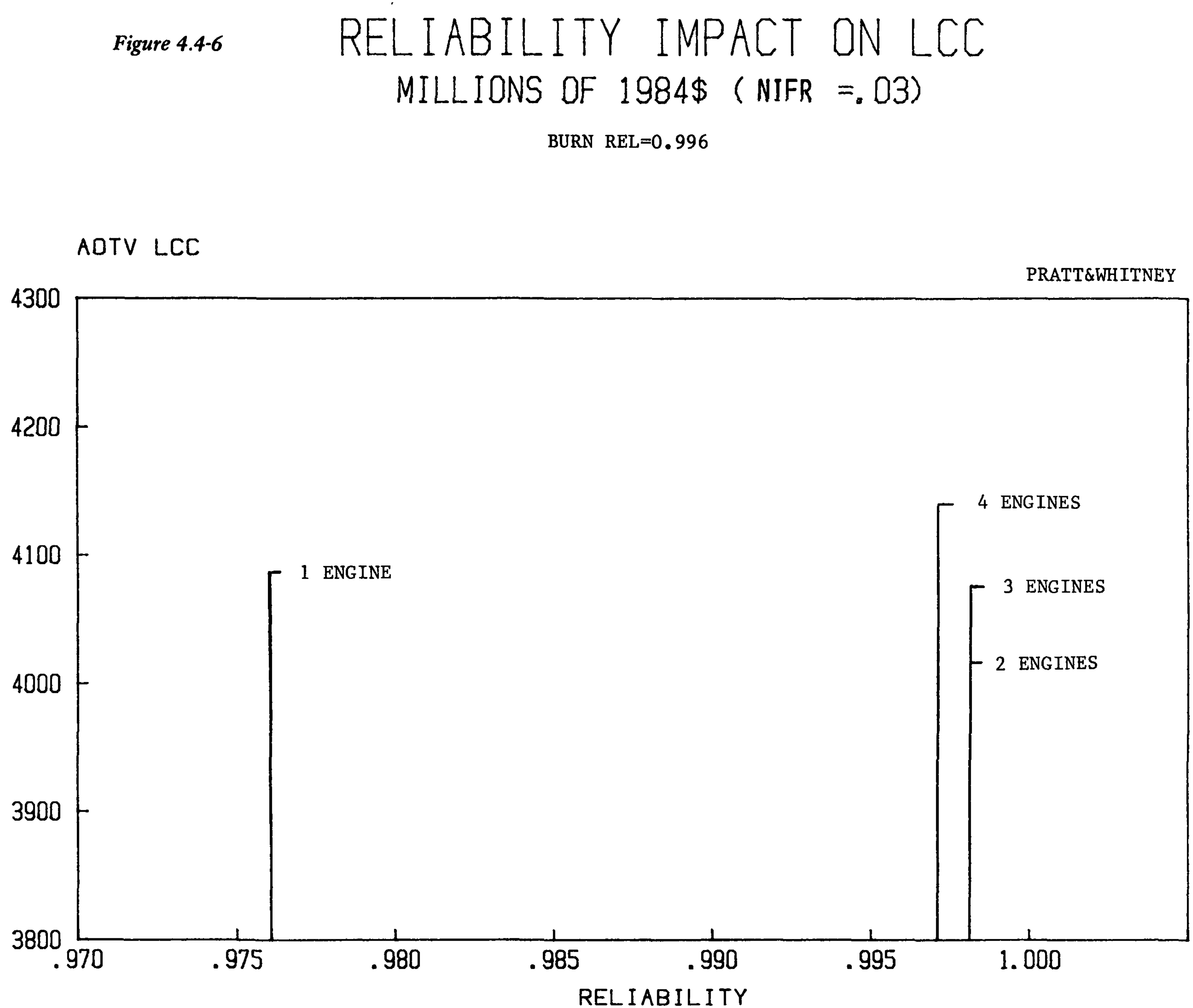

RELIABILITY IMPACT ON LCC
MILLIONS OF 1984$ (NIFR =.03)
BURN REL=0.996
ADTV LCC
PRATT&WHITNEY
4300
4200
4100
4000
3900
3800
.970
.975
.980
.985
.990
.995
1.000
RELIABILITY
1 ENGINE
4 ENGINES
3 ENGINES
2 ENGINES

Figure 4.4-7

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR'=.05)

BURN REL=0.996

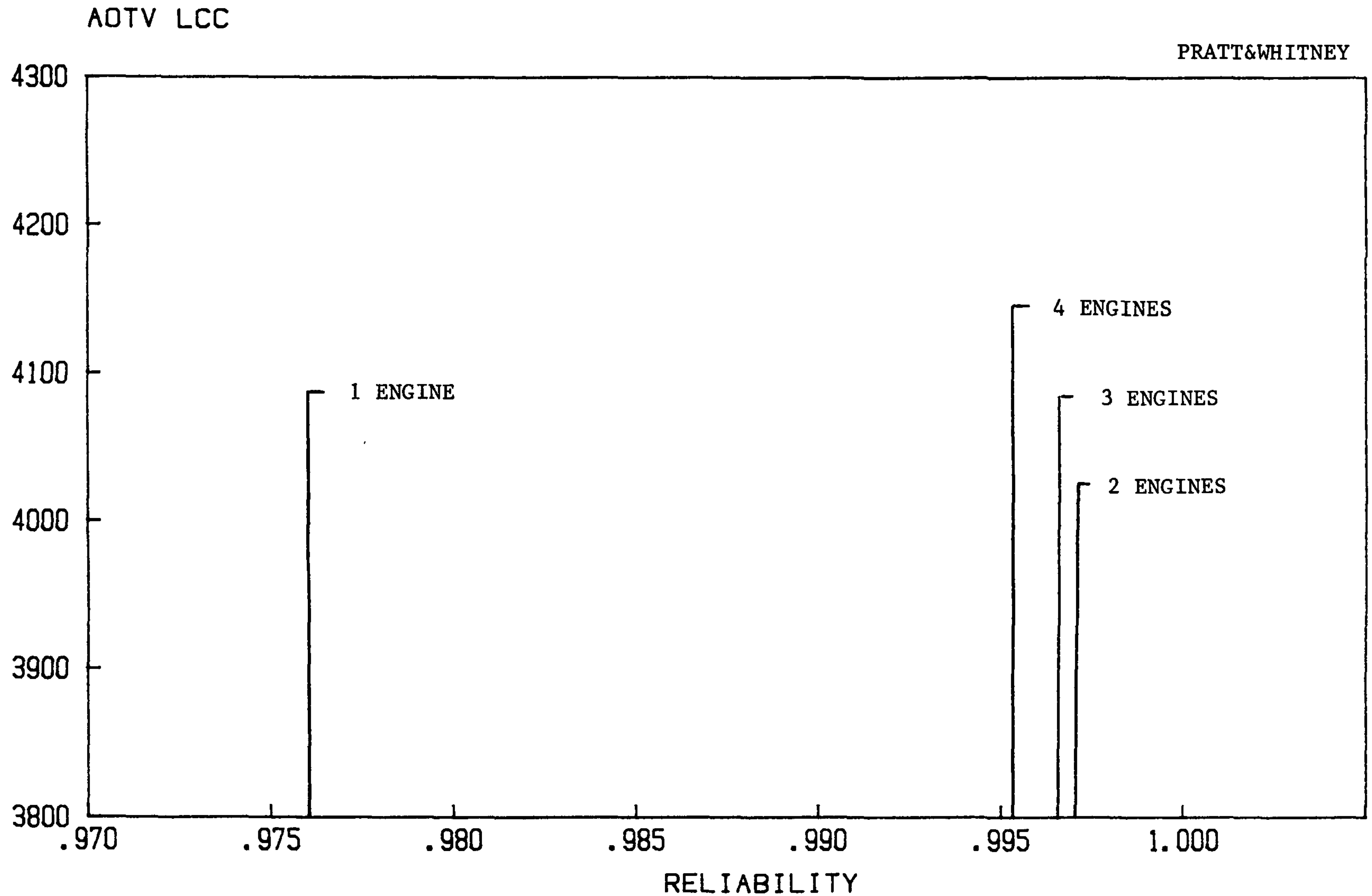


Figure 4.4-8

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =0)

BURN REL=0.999

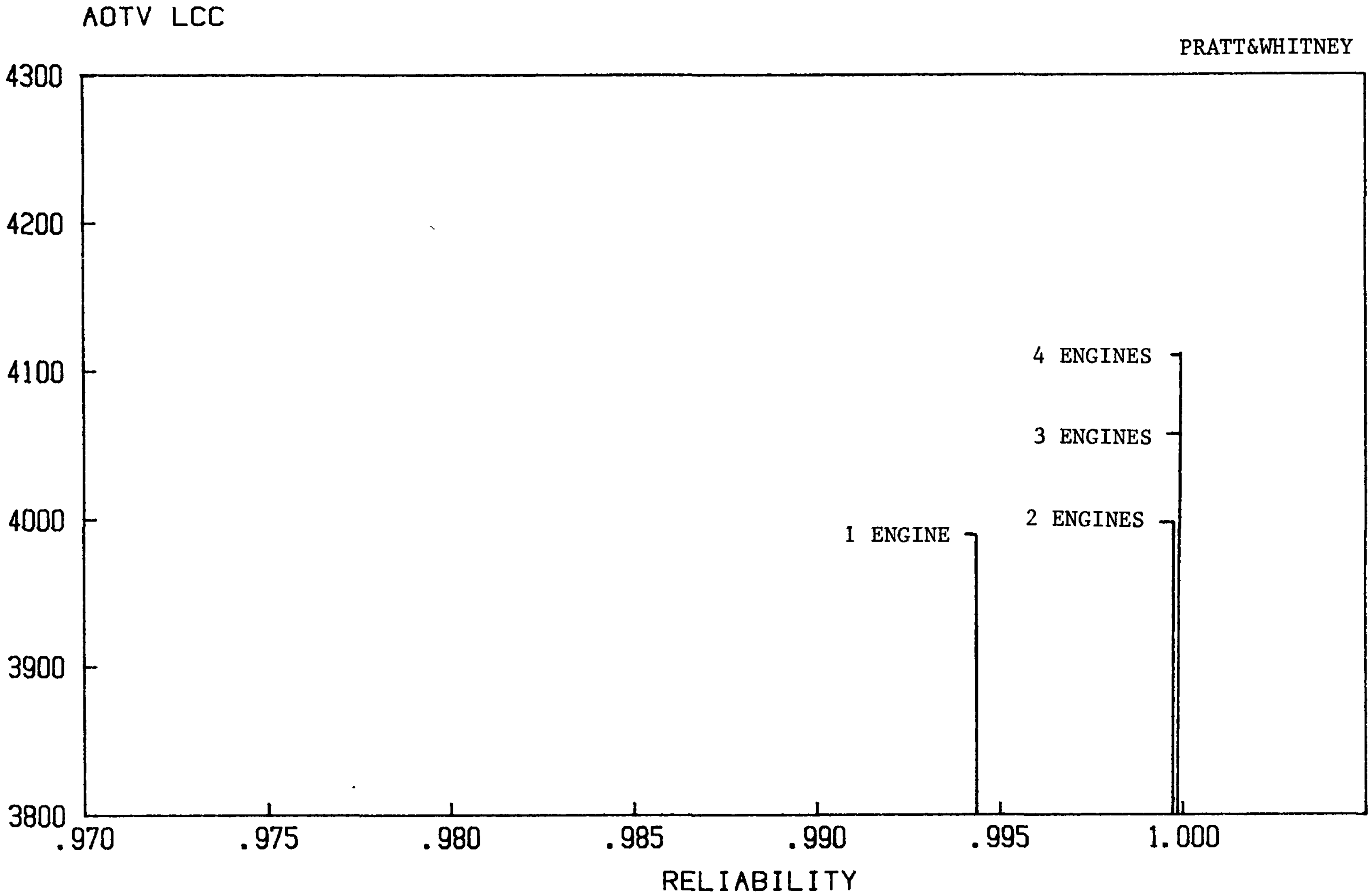


Figure 4.4-9

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.03)

BURN REL=0.999

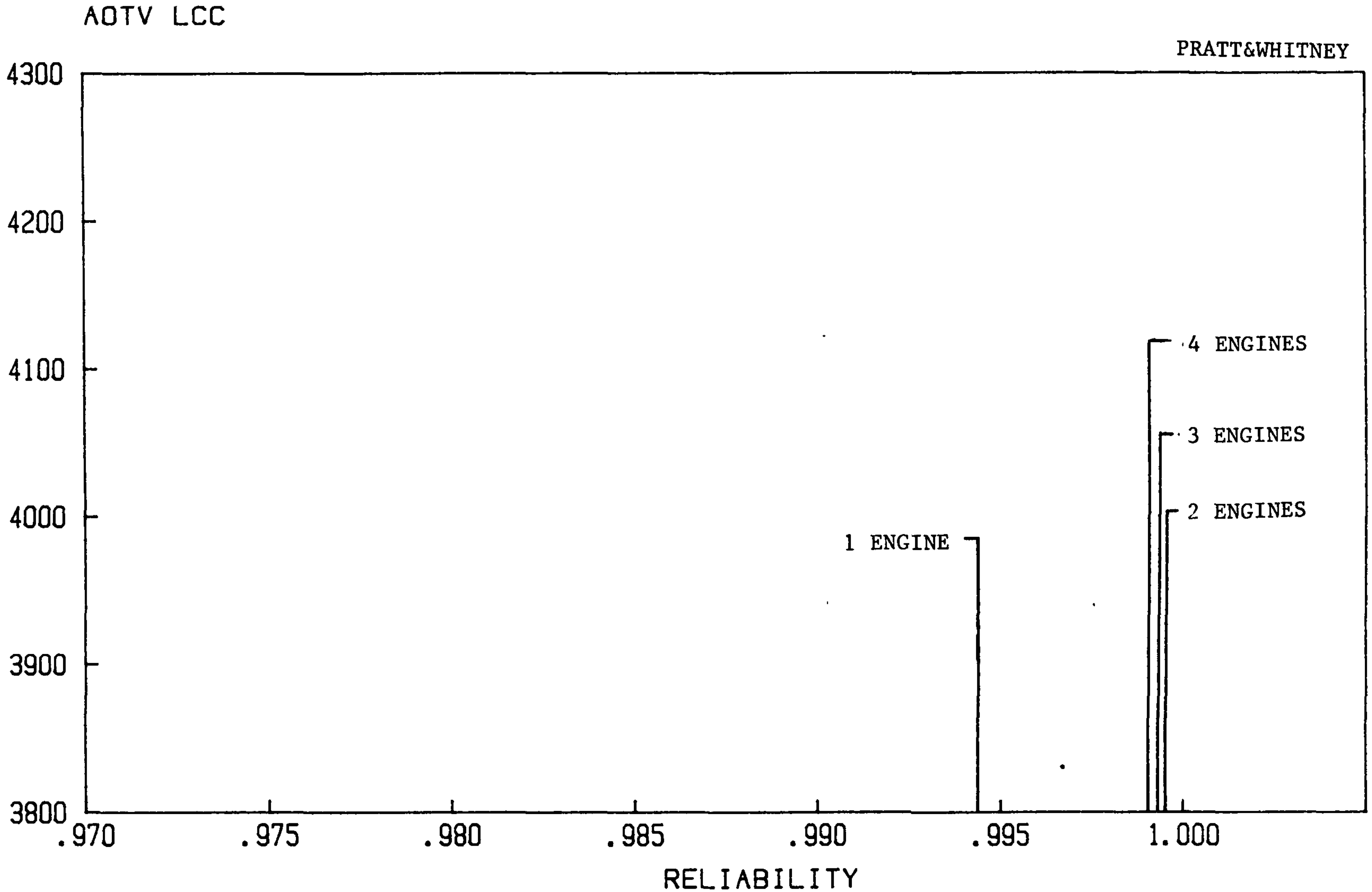


Figure 4.4-10 RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.05)

BURN REL=0.999

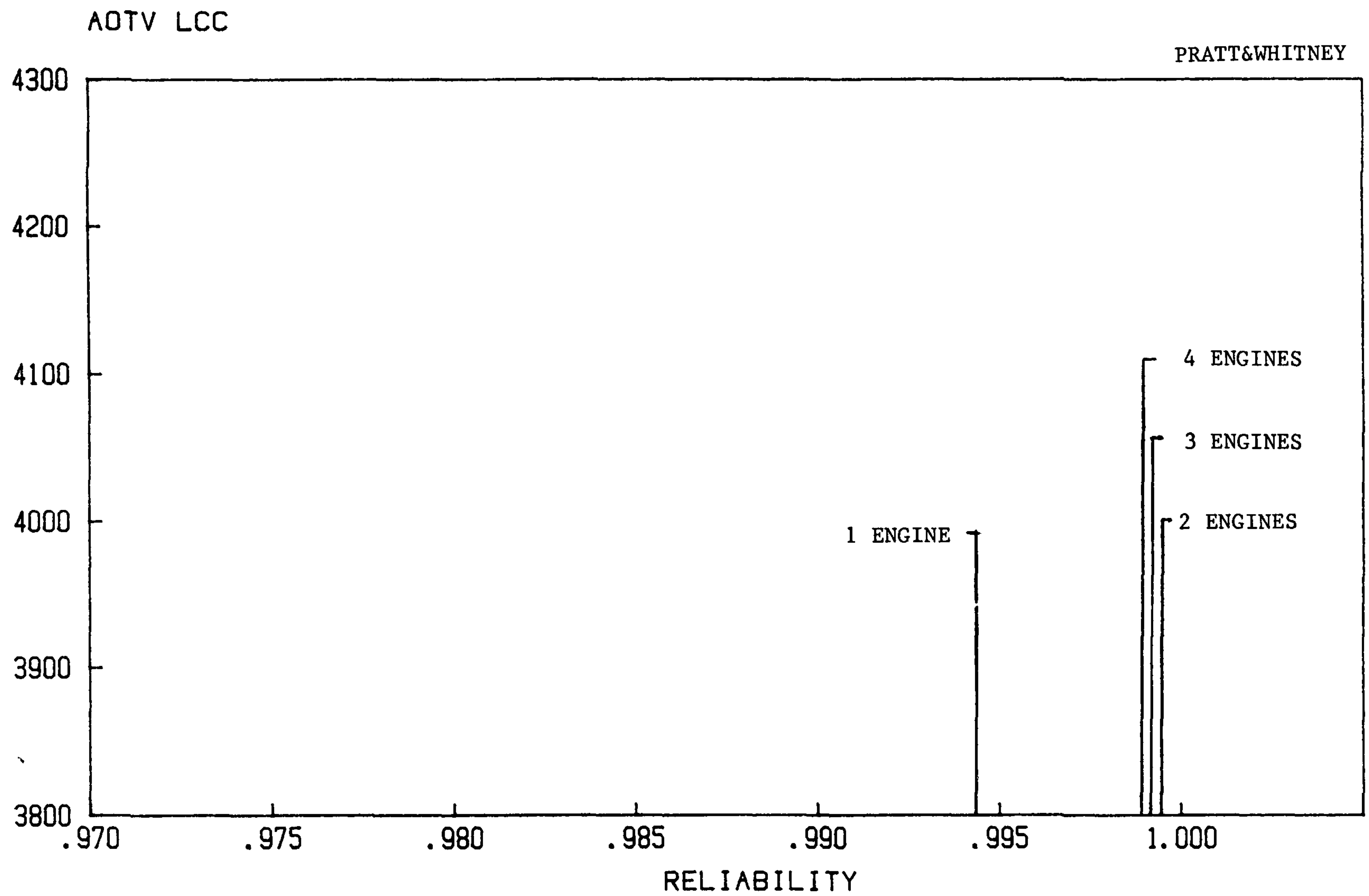


Figure 4.4-11

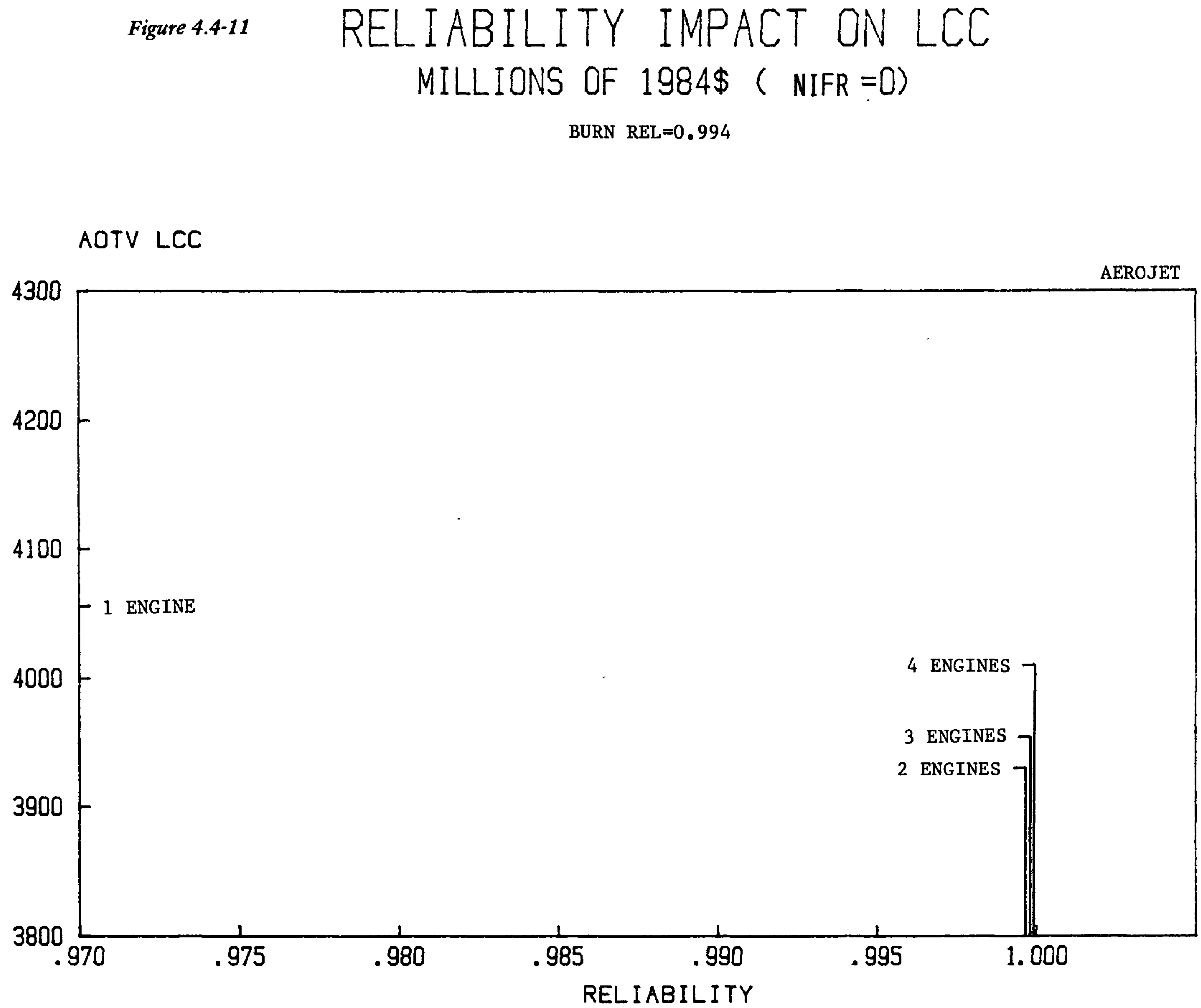

RELIABILITY IMPACT ON LCC
MILLIONS OF 1984$ (NIFR =0)
BURN REL=0.994
AOTV LCC
AEROJET
4300
4200
4100
4000
3900
3800
1 ENGINE
4 ENGINES
3 ENGINES
2 ENGINES
.970
.975
.980
.985
.990
.995
1.000
RELIABILITY

Figure 4.4-12

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.03)

BURN REL=0.994

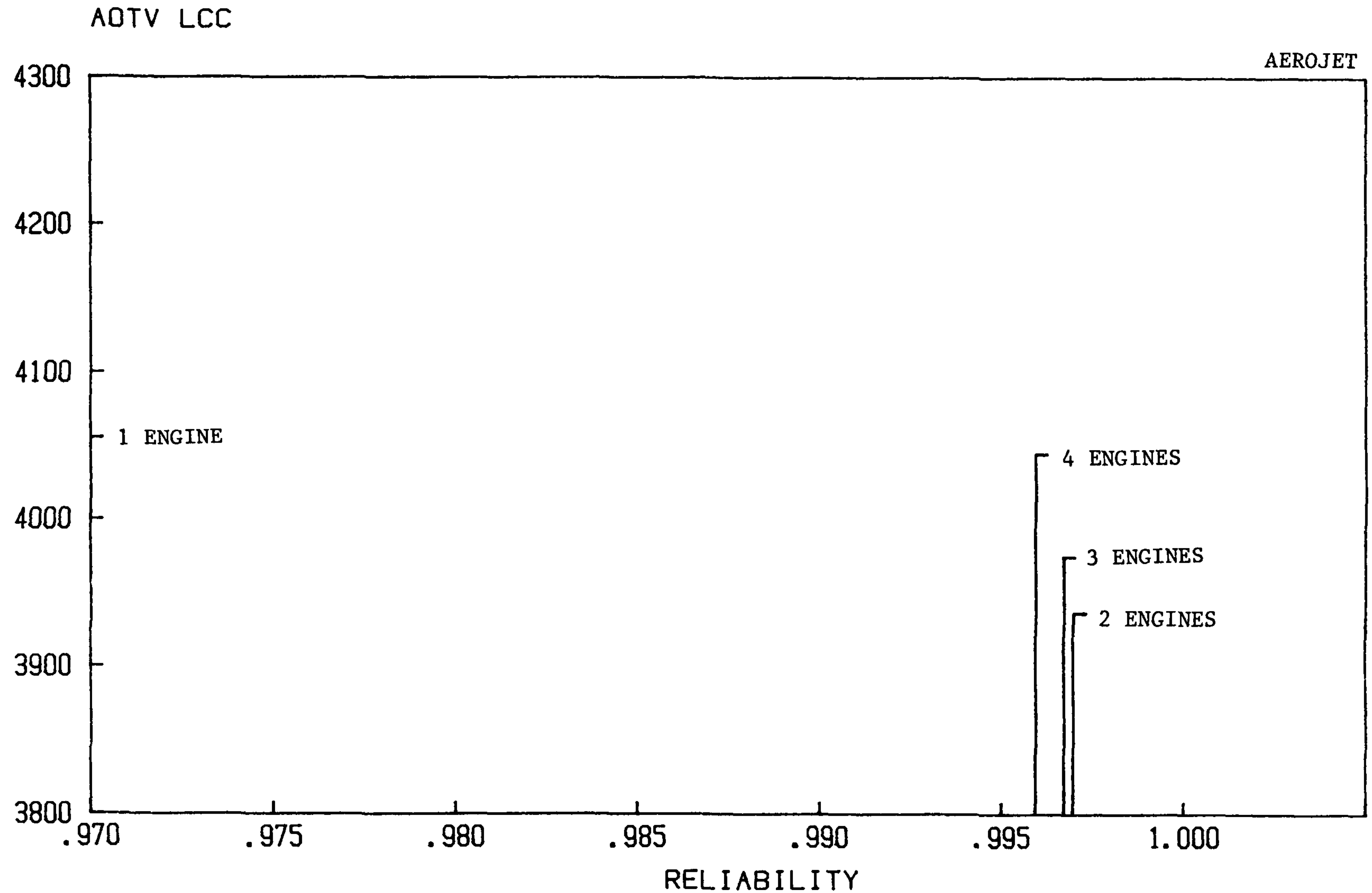


Figure 4.4-13 RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.05)

BURN REL=0.994

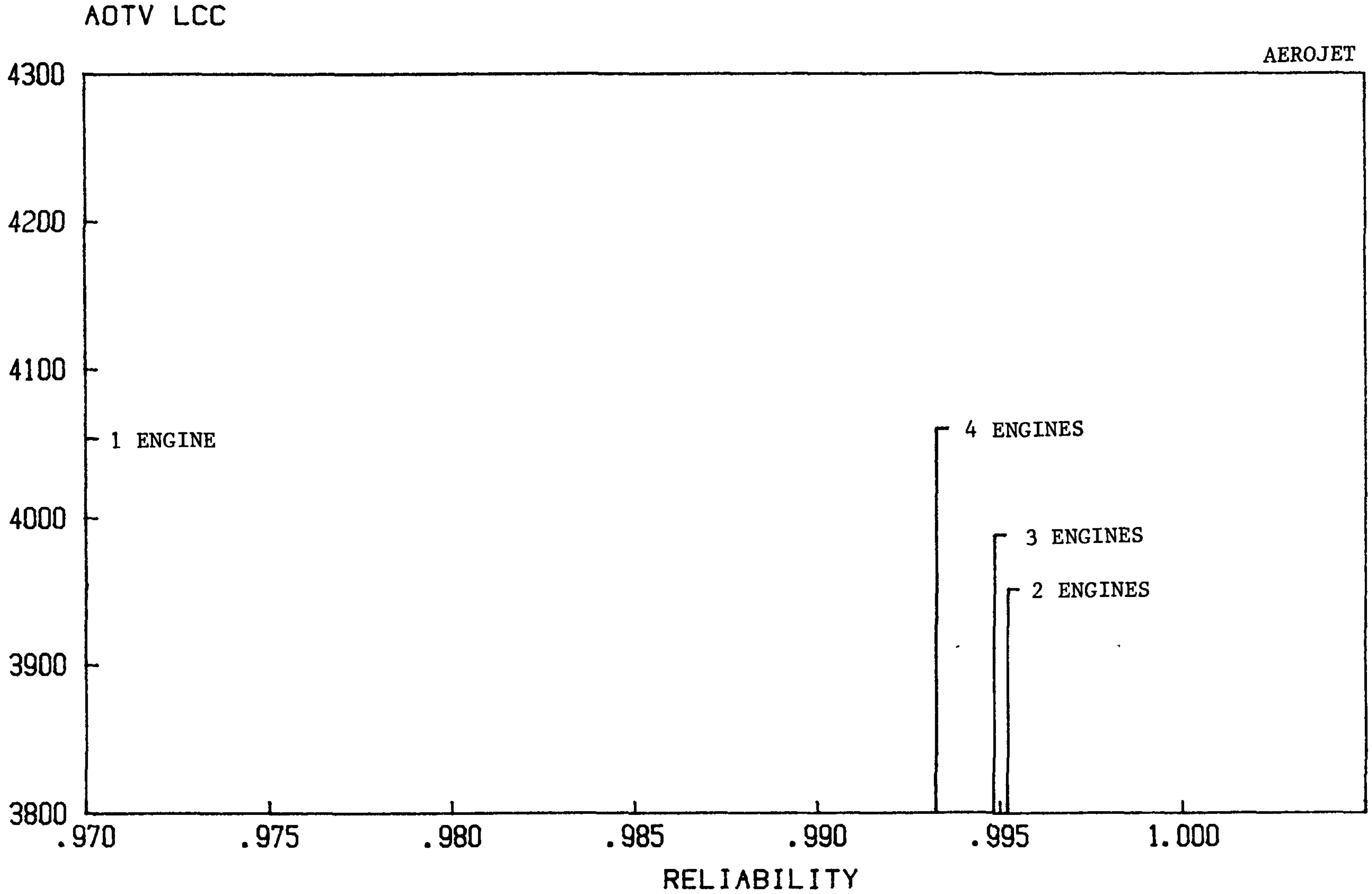


Figure 4.4-14

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =0)

BURN REL=0.996

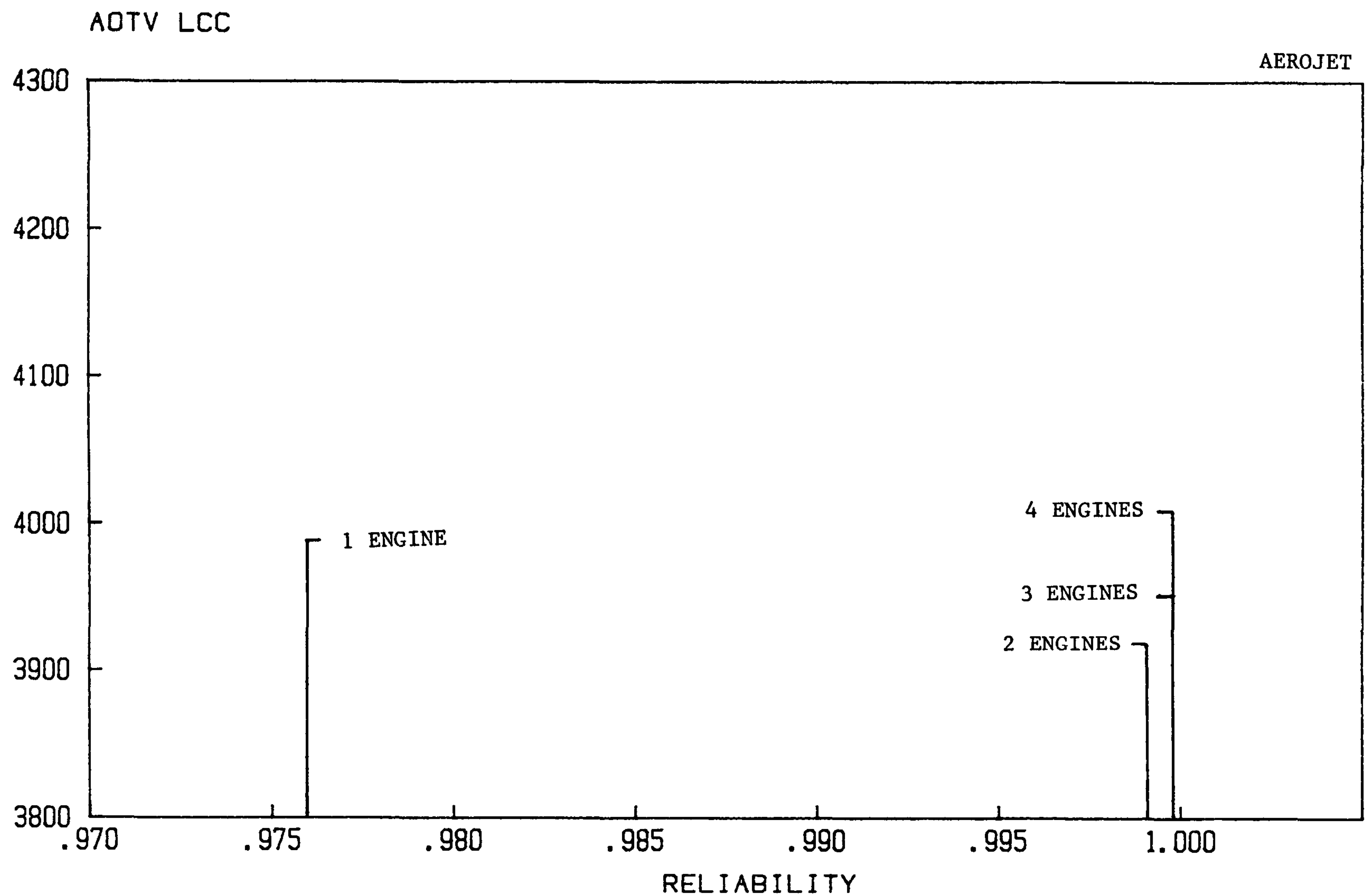


Figure 4.4-15

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.03)

BURN REL=0.996

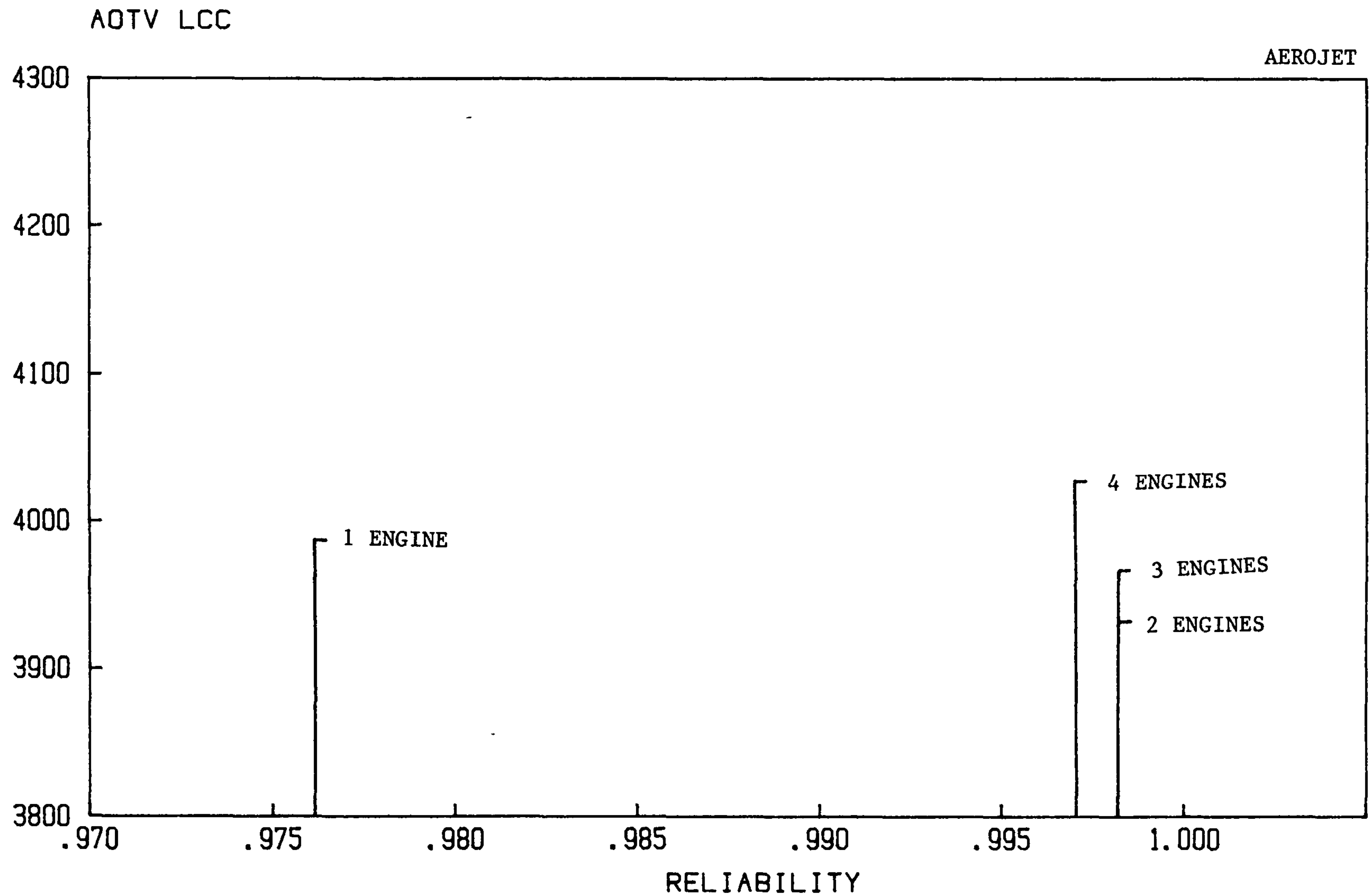


Figure 4.4-16

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR=.05)

BURN REL=0.996

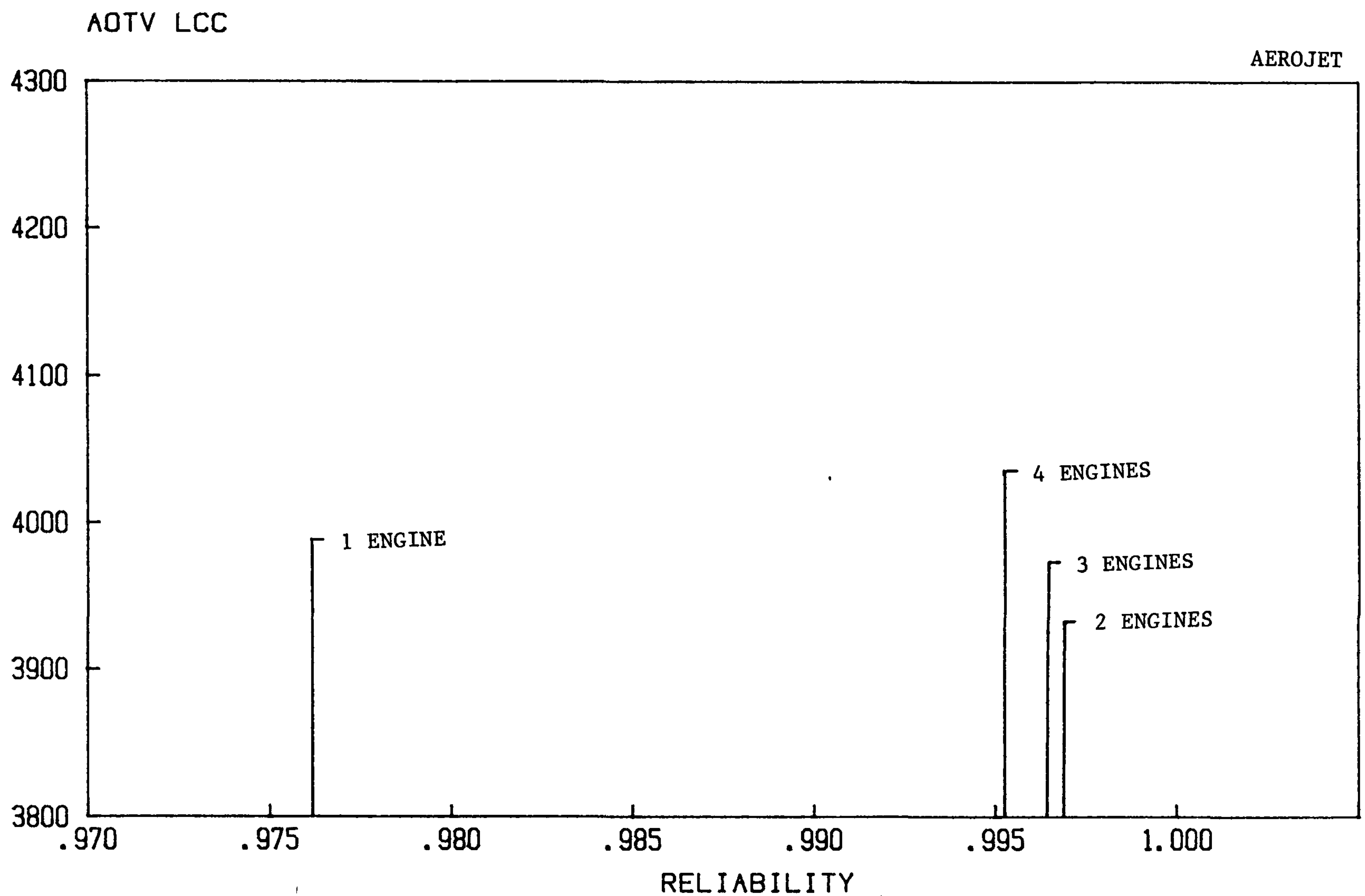


Figure 4.4-17

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =0)

BURN REL=0.999

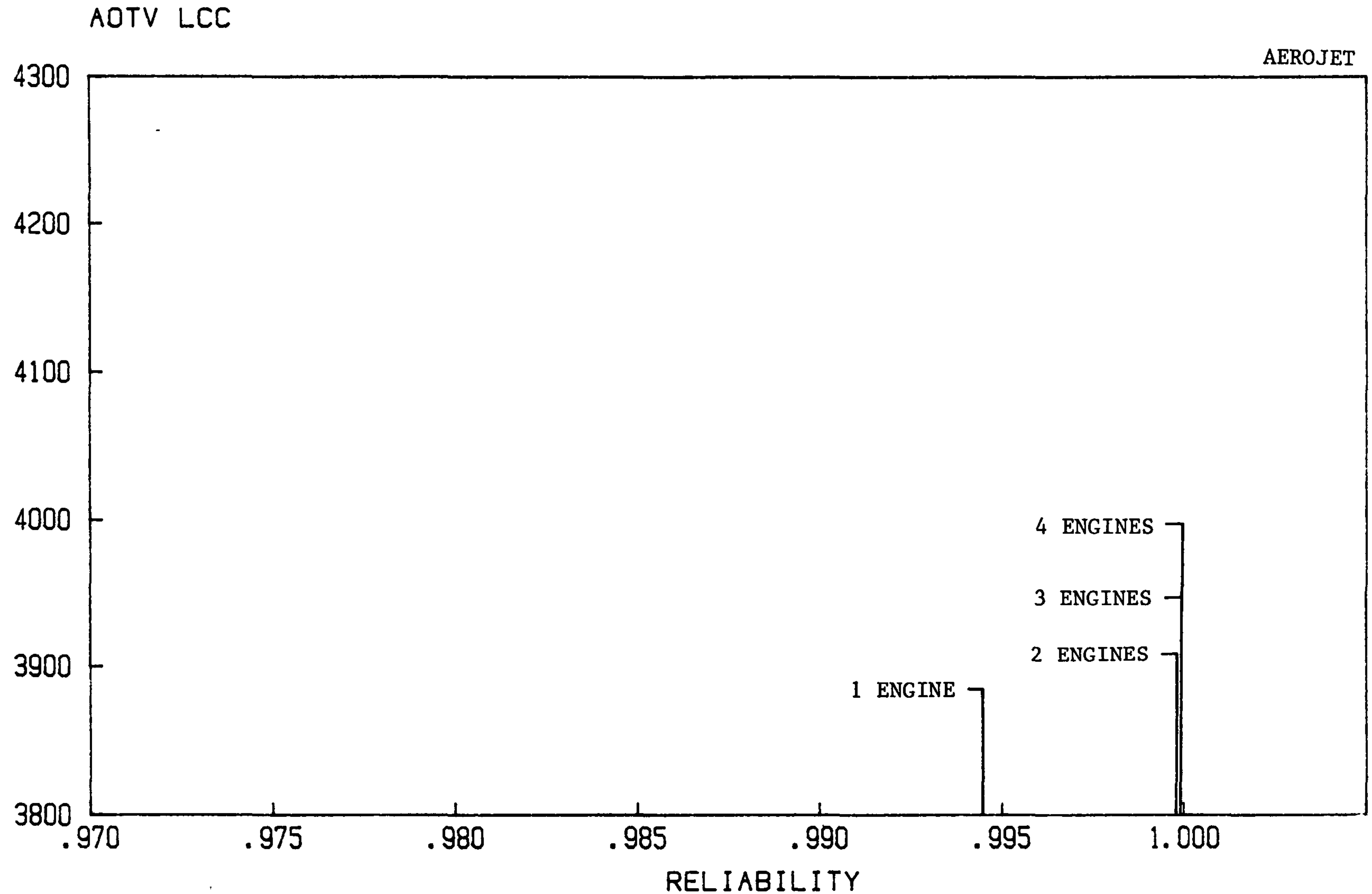


Figure 4.4-18

RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR =.03)

BURN REL=0.999

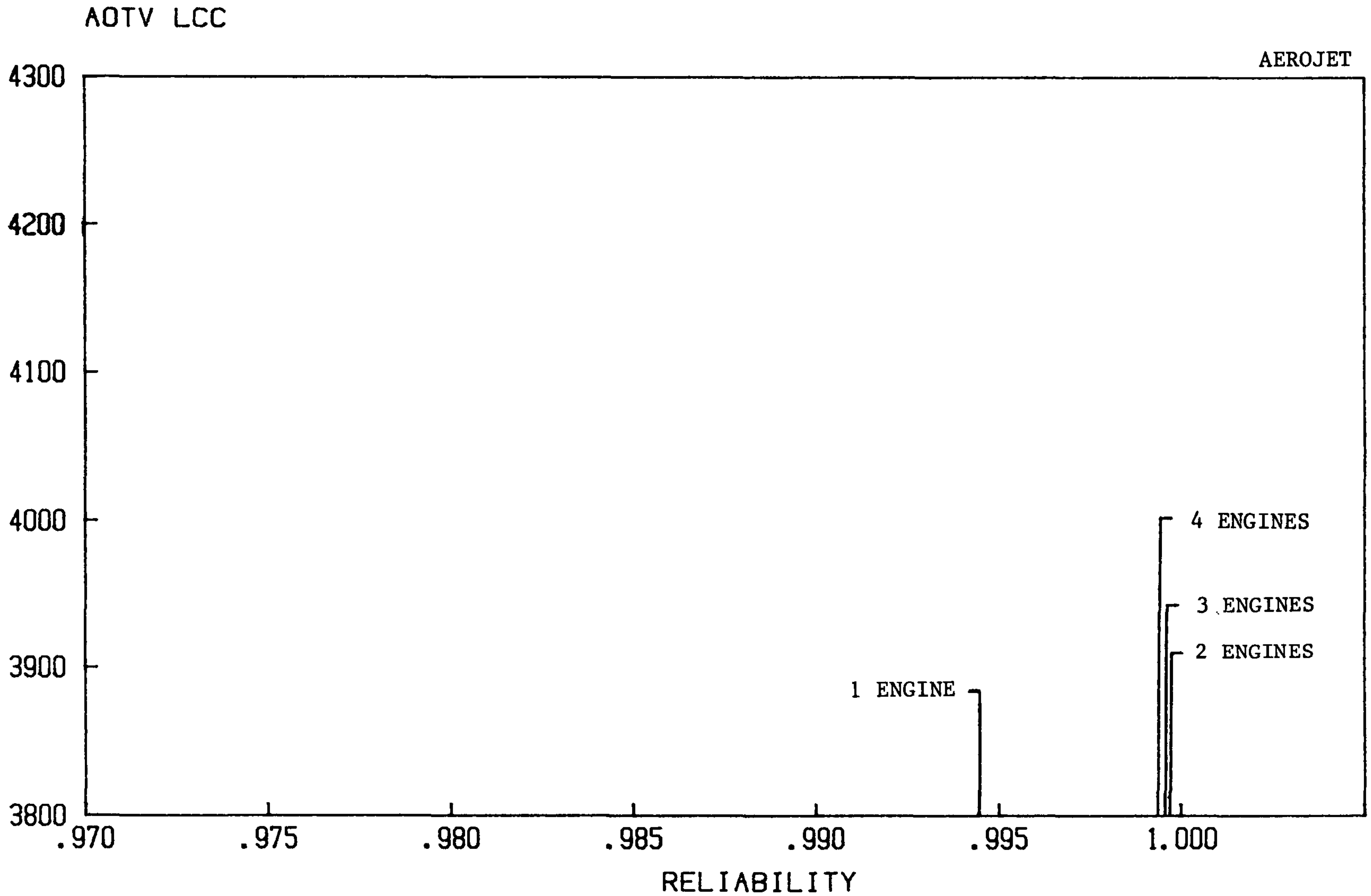
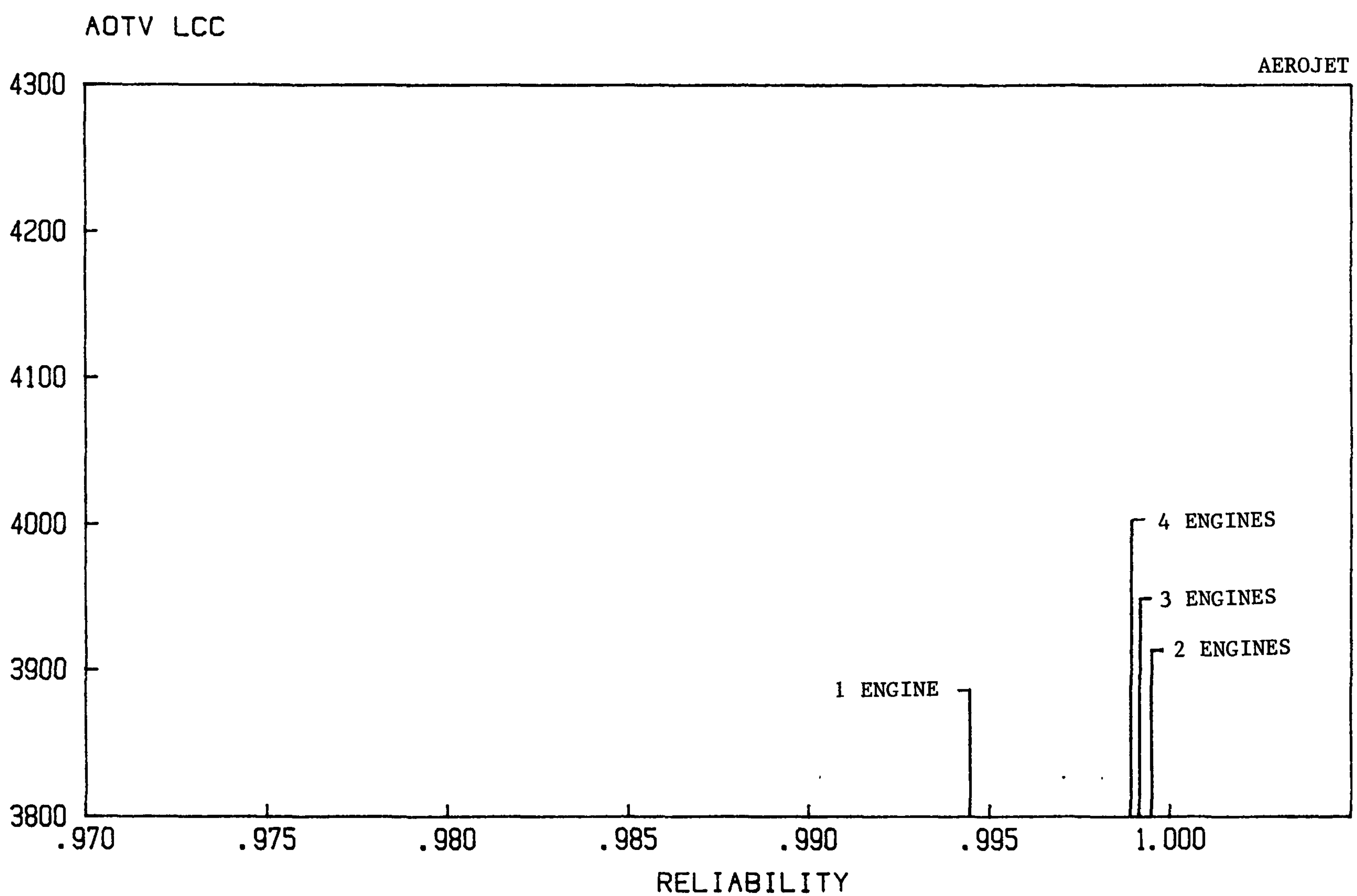


Figure 4.4-19 RELIABILITY IMPACT ON LCC

MILLIONS OF 1984$ (NIFR=.05)

BURN REL=0.999



Based on an engine selection criteria of minimum LCC, Table 4.4-2 reflects the optimum number of engines for all combinations of engine burn reliability and nonindependent failure rate for the Pratt & Whitney engine data.

Table 4.4-2
Pratt & Whitney Optimal LCC Propulsion Configuration

	Nonindependent Failure Factor		
Burn Reliability	0.00	0.03	0.05
0.994	2 Engines	2 Engines	2 Engines
0.996	2 Engines	2 Engines	2 Engines
0.999	1 Engine	1 Engine	1 Engine

An examination of Table 4.4-2 shows two engines to be optimal under the selection criteria for all but the 0.999 single-engine burn situation. Further examination of Figures 4.4-2 through 4.4-10 demonstrates three or four engines to be only marginally superior in reliability and significantly higher in LCC than two engines. Nominal delta LCC values are $M70 for three engines and $M130 for four engines over the two-engine configuration. A single engine shows the least LCC only for a burn reliability of 0.999 (by a delta of nominally $M10) but not for burn reliabilities of 0.994 or 0.996.

These data lead toward the conclusion that across the selected spectrum of single-engine burn reliability (0.994-0.999) and nonindependent failure factor (0.00-0.05), two engines demonstrate near maximum reliability and minimum or near minimum LCC. In addition, in making a two-engine configuration fail operational/fail safe for the return of a manned capsule, an amount of subsequent propellant cost must be added to the two-engine case LCC for contingency operation of a GO_2/GH_2 ACS system. For seven 13,000-lbm manned payloads in the 1994-2000 time period (Ref 6 Mission Model), approximately $20M must be added to the two-engine case to make it fail operational/fail safe. This appears to be an affordable method of providing a high level of manrating reliability and flexibility along with remaining the lowest LCC configuration for the performance of the Rev 6 Mission Model.

Table 4.4-3 reflects the optimum number of engines for all combinations of single-engine burn reliability and nonindependent failure rate for the Aerojet engine data. The results and conclusions are the same as for the Pratt & Whitney data.

Table 4.4-3
Aerojet Optimal LCC Propulsion Configuration

	Nonindependent Failure Factor		
Burn Reliability	0.00	0.03	0.04
0.994	2 Engines	2 Engines	2 Engines
0.996	2 Engines	2 Engines	2 Engines
0.999	1 Engine	1 Engine	1 Engine

In comparing cost element outputs from the LCC model, most cost elements (including DDT&E, unit production, GSE, program management, spares, scheduled/unscheduled maintenance, overhaul labor, and ground control) show only minor delta cost differences between the various engine configurations. However, two cost elements contribute toward major cost deltas between the engine configurations—propellant cost and mission loss cost. Propellant cost is directly related to the propellant usage to achieve a 15,000-lbf thrust for the respective engine configurations. Mission loss cost is related to the propulsion system reliability, which in turn is derived from engine burn reliability, number of engines and the nonindependent failure factor.

For Pratt & Whitney engine data, a greater number of engines results in higher propellant usage and therefore higher LCC. Conversely mission loss cost is highest for a single engine and generally lowest for two engines (depending on burn reliability and nonindependent failure factor - see previous discussion on optimal propulsion system reliability considerations). The costs associated with these two elements tend to be the determining factors in the relative costs of the respective engine configurations. The mission loss costs for two, three, and four engines are relatively low and balance out, whereas one engine generally has a very high mission loss cost. The resultant summation of mission loss cost and propellant cost across the selected ranges of single-engine burn reliability and nonindependent failure factor shows two engines to be the optimal or near optimal choice.

For Aerojet engine data, the propellant cost is less of a cost driver because of the reduced propellant consumption differences between the engine configurations. In view of this, the primary cost driver for the Aerojet propulsion system becomes the mission loss cost. The two-engine configuration demonstrates least mission loss cost for all but the single-engine configuration with a burn reliability of 0.999.

LCC Model Sensitivities - Tables 4.4-4 and 4.4-5 summarize the sensitivities of cost parameter variations on LCC for Pratt & Whitney and Aerojet. The two-engine burn reliability of 0.996 and a nonindependent failure factor of 0.03 was selected as a baseline for developing the cost parameter sensitivities. The sensitivities are generally the same for other engine configurations although the LCC "actual values" differ. Each parameter has been varied over ranges as wide as might be conceived at this time. Figure 4.4-20 shows the sensitivities of LCC to missions per overhaul (average number of missions between remove and replace for the vehicles' subsystems).

Table 4.4-4 LCC Sensitivities Summary (Pratt & Whitney) (for 2 engines)

Parameter	Sensitivity (LCC $M per -)
Propellant Cost	$182M per $100/lb
IVA Cost	$59M per $10k/h
EVA Cost	$28M per $10k/h
Failures/Mission	$11M per failure/mission
Missions/Year	$23M per mission/yr (not including propellant)
Missions/Overhaul	$3.8M to $0.33M per mission/overhaul
Specific Impulse	$11M per second of I_{sp}

TABLE 4.4-5 LCC Sensitivities Summary (Aerojet)
(for 2 engines)

PARAMETER	SENSITIVITY (LCC $M per __)
Propellant Cost	$176M per $100/lb
IVA Cost	$59M per $10k/hr
EVA Cost	$28M per $10k/hr
Failures/Mission	$9.3M per failure/mission
Missions/Year	$23M per mission/yr (not including propellant)
Missions/Overhaul	$4M to $0.33M per mission/overhaul
Specific Impulse	$11M per second of I_{sp}

The sensitivities for the Aerojet data are almost identical to those shown for the Pratt & Whitney data.

Propellant cost variations provide the greatest impact on OTV LCC even for only minor propellant cost deltas. This cost element tends to outweigh all other cost elements considered in this analysis in both actual magnitude and relative LCC sensitivity. Secondary cost sensitivities are IVA, EVA, specific impulse and missions per year. IVA and EVA variations contribute to LCC sensitivity because of the significant amount of IVA and EVA time involved in mission turnaround activity. Specific impulse changes contribute significantly to LCC sensitivity because of the direct relationship between specific impulse and propellant usage. Missions per year changes drive LCC because of the direct relationship with propellant usage, EVA, IVA and all other cost elements. Variations in failures per mission and missions between major engine overhauls are shown to provide minimal impact on LCC.

4.4.3 Life-Cycle Cost Profiles

The two-engine Pratt & Whitney propulsion system configuration with a single-engine burn reliability of 0.996 and a nonindependent failure factor of 0.03 was selected as a baseline for developing LCC cost profiles. Figure 4.4-21 shows the baseline cost profile with operations and support (O&S) costs being the major contributor

Figure 4.4-20 MISSIONS/OVERHAUL IMPACT ON LCC

2 ENGINES/ NIFR =.03/REL=.996

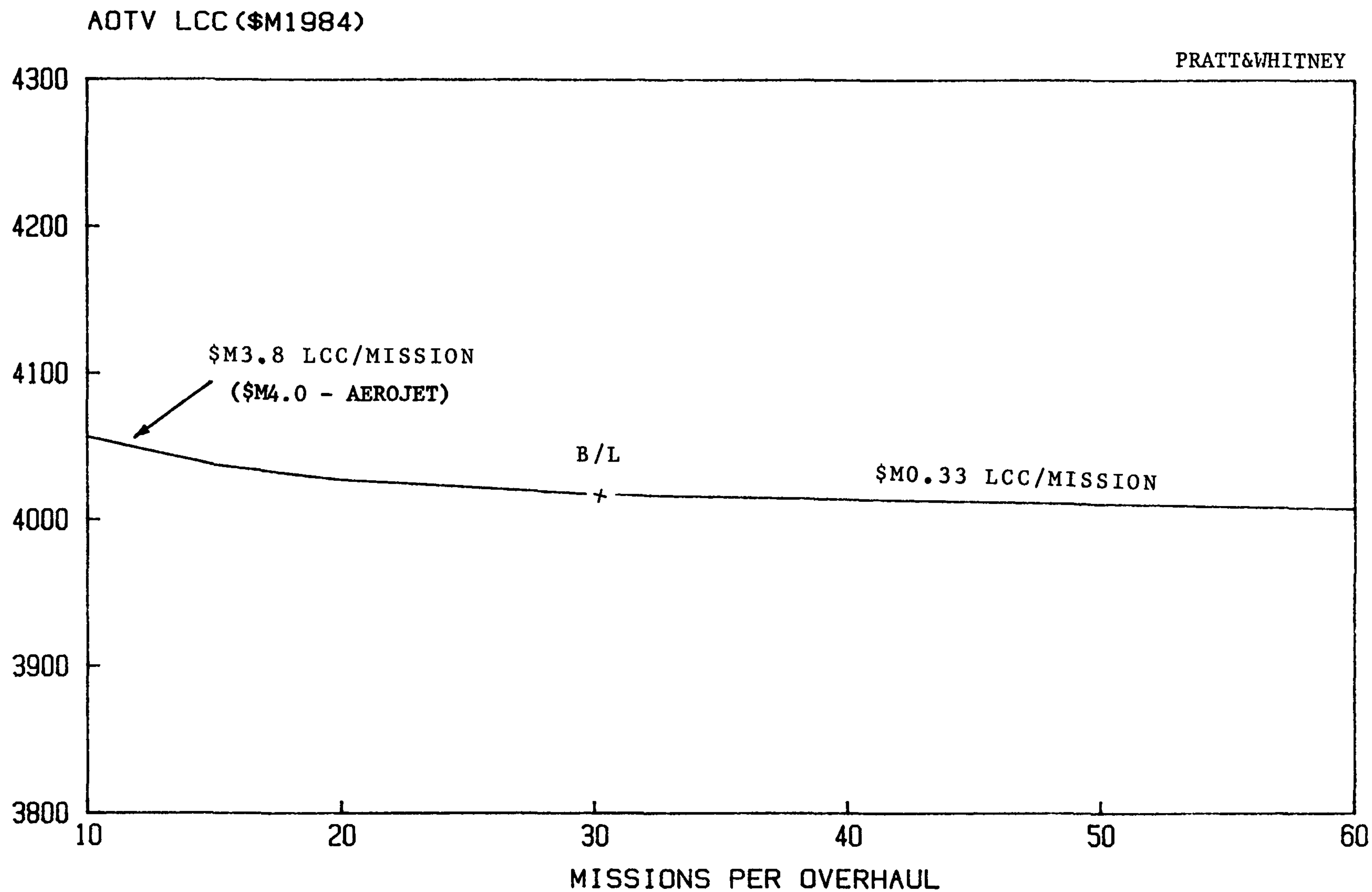
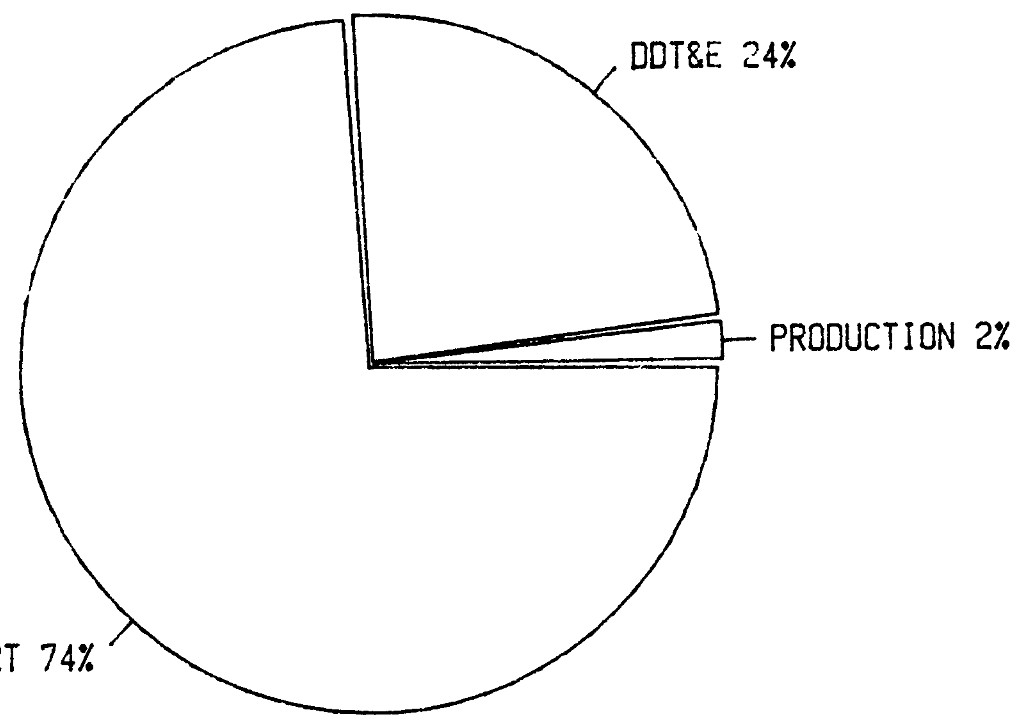


Figure 4.4-21

LIFE CYCLE COST PROFILE

$M4026 (1984)

PRATT&WHITNEY



with 74% of the total LCC. The majority of this O&S cost is attributable to $M2685 propellant cost. Propellant represents 67% of the total LCC and has been removed from consideration in subsequent cost profiles to allow closer examination of the secondary cost drivers.

The most significant LCC element is DDT&E (Fig. 4.4-22) of which engines (34%), propulsion (18%), and aerobrake (14%) contribute 66% to this element.

Operations and support costs (excluding propellant) is profiled in Figure 4.4-23 where scheduled maintenance (mission turnaround) and ground control comprise the majority of this cost element.

The third and least contributor to the LCC is production cost (Fig. 4.4-24), which accounts for only 2% of the total LCC.

Figure 4.4-25 shows the tall poles, or largest of the secondary contributors to LCC, in descending order of magnitude (propellant being excluded).

Figure 4.4-22

DDT&E COST PROFILE

$M964 (1984)

PRATT&WHITNEY

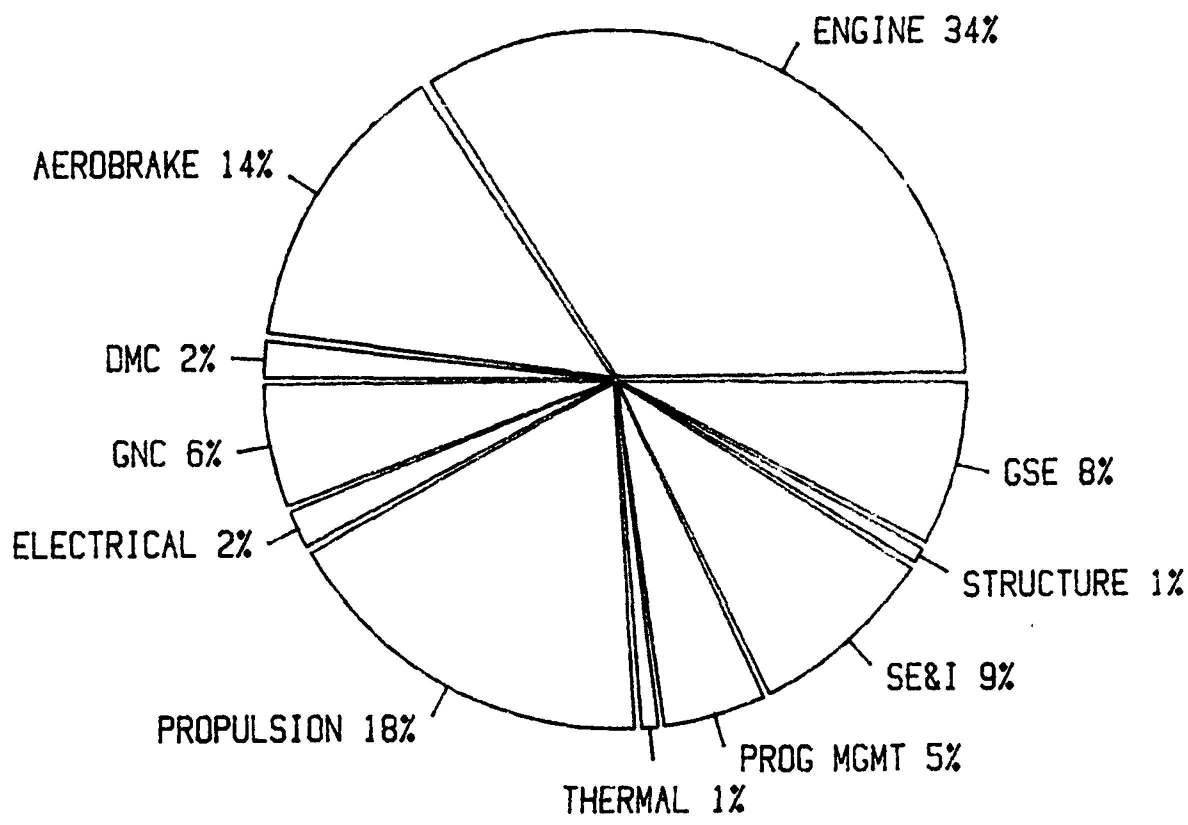


Figure 4.4-23

O&S COST PROFILE

$M312 (1984)

PRATT&WHITNEY

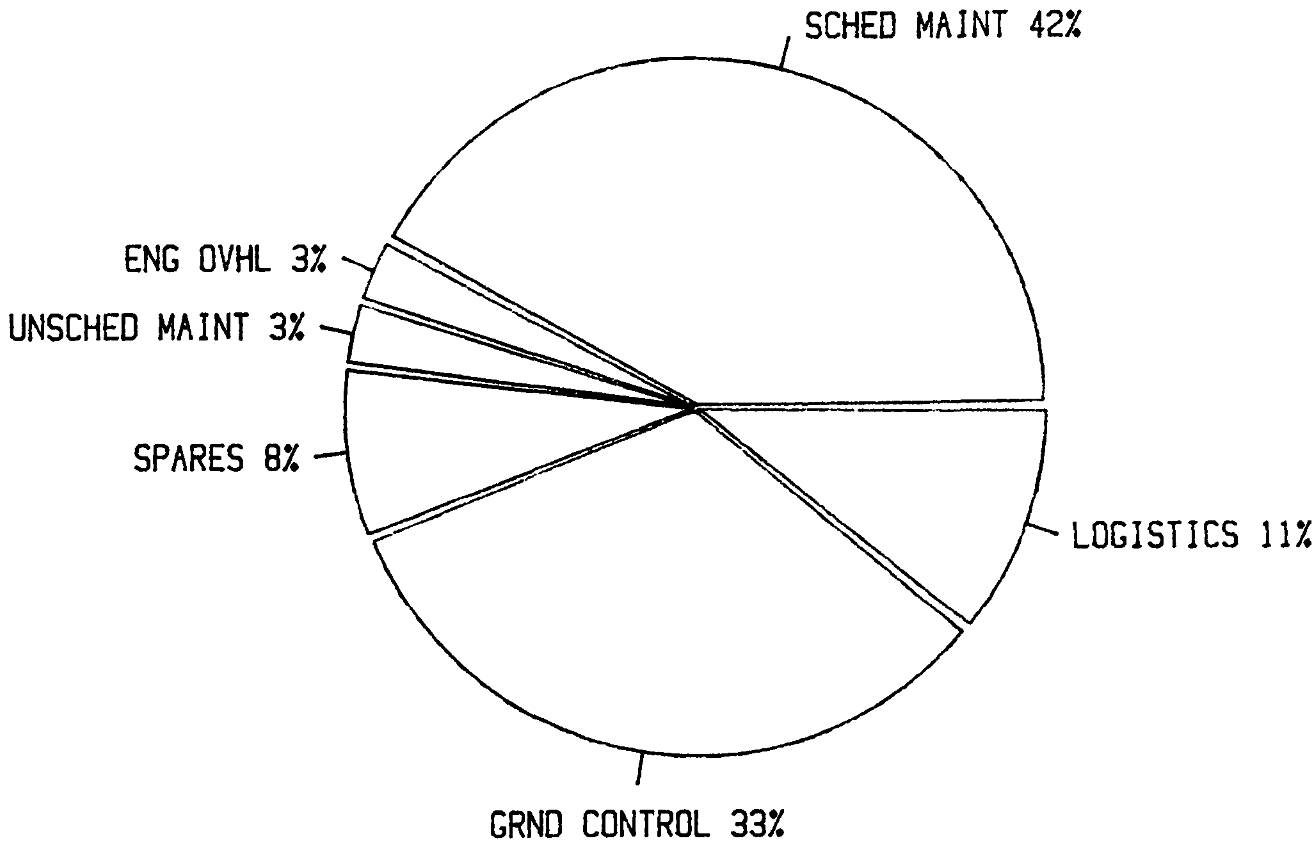


Figure 4.4-24

PRODUCTION COST PROFILE

$M65 (1984)

PRATT&WHITNEY

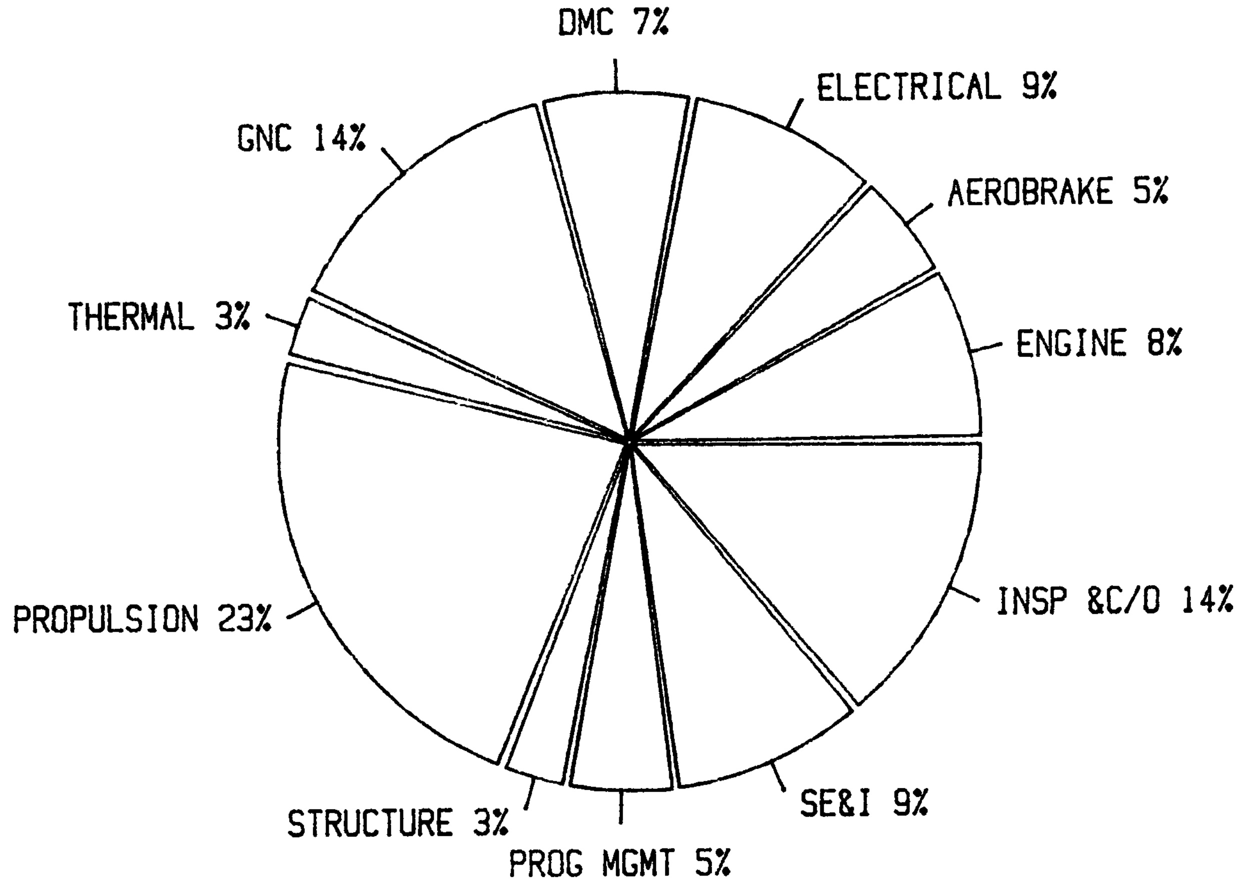
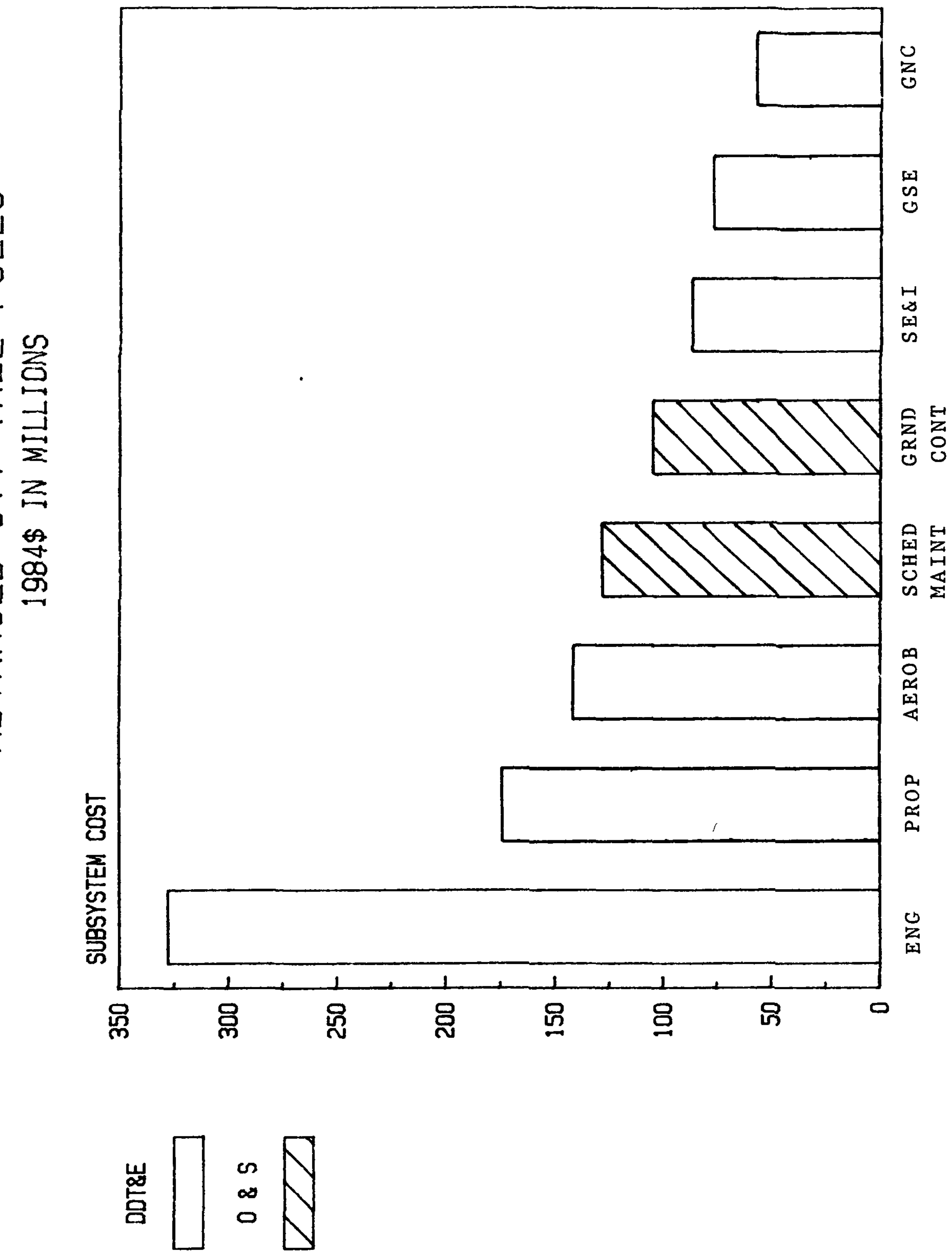


Figure 4.4-25


ADVANCED OTV TALL POLES
1984$ IN MILLIONS
SUBSYSTEM COST
350
300
250
200
150
100
50
0
ENG
PROP
AEROB
SCHED MAINT
GRND CONT
SE&I
GSE
GNC
DDT&E
O & S

4.4.4 Level of Vehicle Modularization

Section 4.1.3 describes the process used in modularizing the baseline vehicle concept. The assessment criteria used in grouping the vehicle components into modules include module size, accessibility, interface complexity, module functional similarity, failure frequency, physical proximity, operational concerns, etc. This section provides a rationale from a LCC standpoint to substantiate the level of serviceability (modularity) selected and discussed earlier.

As demonstrated by the LCC model outputs and sensitivities, the orbiter payload transportation costs are a significant cost driver. Although space station storage costs have not been computed, it is reasonable to expect that this type of storage (for spares and equipment) will be at a premium. A review of Table 4.4-6 indicates a reduced cost for level 3 over the level 1 or level 5 for these two major cost drivers (items A and B). Analysis of the LCC model outputs and sensitivities indicated that items C through G in Table 4.4-6 were not major cost drivers. A cost comparison for items C through F in Table 4.4-6 marginally favors level 1 over level 3 and substantially over level 5. In summary, level 5 probably has the least life-cycle cost. Levels 3 and 5 offer an advantage over the level 1 in that they can incur multiple failures. Further analysis is required to determine which modules require the most support because of their higher failure rate and/or criticality to quantify these tradeoffs. Additional maintainability and design engineering will enhance the location of these modules for easier EVA R/R actions. Module design permits technology upgrade more readily than either full assembly design (via cost) or the subcomponent design (via interfaces).

Based on estimates of the major cost drivers, Figure 4.4-26 illustrates where the probable minimization of LCC will occur with regard to vehicle and engine modularization.

4.5 MAJOR STUDY CONCLUSIONS

Phase II results served to verify the results and conclusions arrived at during the Phase I study effort (see section 3.5). This applies to development of the baseline vehicle(s), engine operational requirements, and design recommendations.

Table 4.4-6 Level of Servicing and LCC

Cost Related Factor	Level 1	Level 3	Level 5
A. Orbiter Transportation	Large volume requirement in orbiter for each maintenance action	Smaller modules result in reduced transport cost & improved orbiter loading opportunity.	Larger volume due to extensive range of spares types.
B. Space Station Storage	Large volume required	Probably less volume if only unit quantities of key modules are selected.	Similar to level 3.
C. EVA Time IVA Time	Handling large units	Handling smaller units but may be additional R&R actions for accessibility. Design for <u>M</u> may produce EVA reductions.	Accessibility increasingly difficult thus increasing EVA time.
D. Training	Limited requirement since only few types of actions.	Required ability to R&R all modules identified as key. Marginally higher requirements	Substantially increased training requirements over level 3.
E. Support Data - Hard copy - Computer	Limited	Marginally higher since the variety of maintenance tasks is increased	Increased data rqmts due to increased complexity.
F. Support Equiment (SE) and Tools	Limited	Possibly marginally higher but standardization would limit proliferation of SE and tools.	Increased SE rqmts due to increased # test points, interconnects.

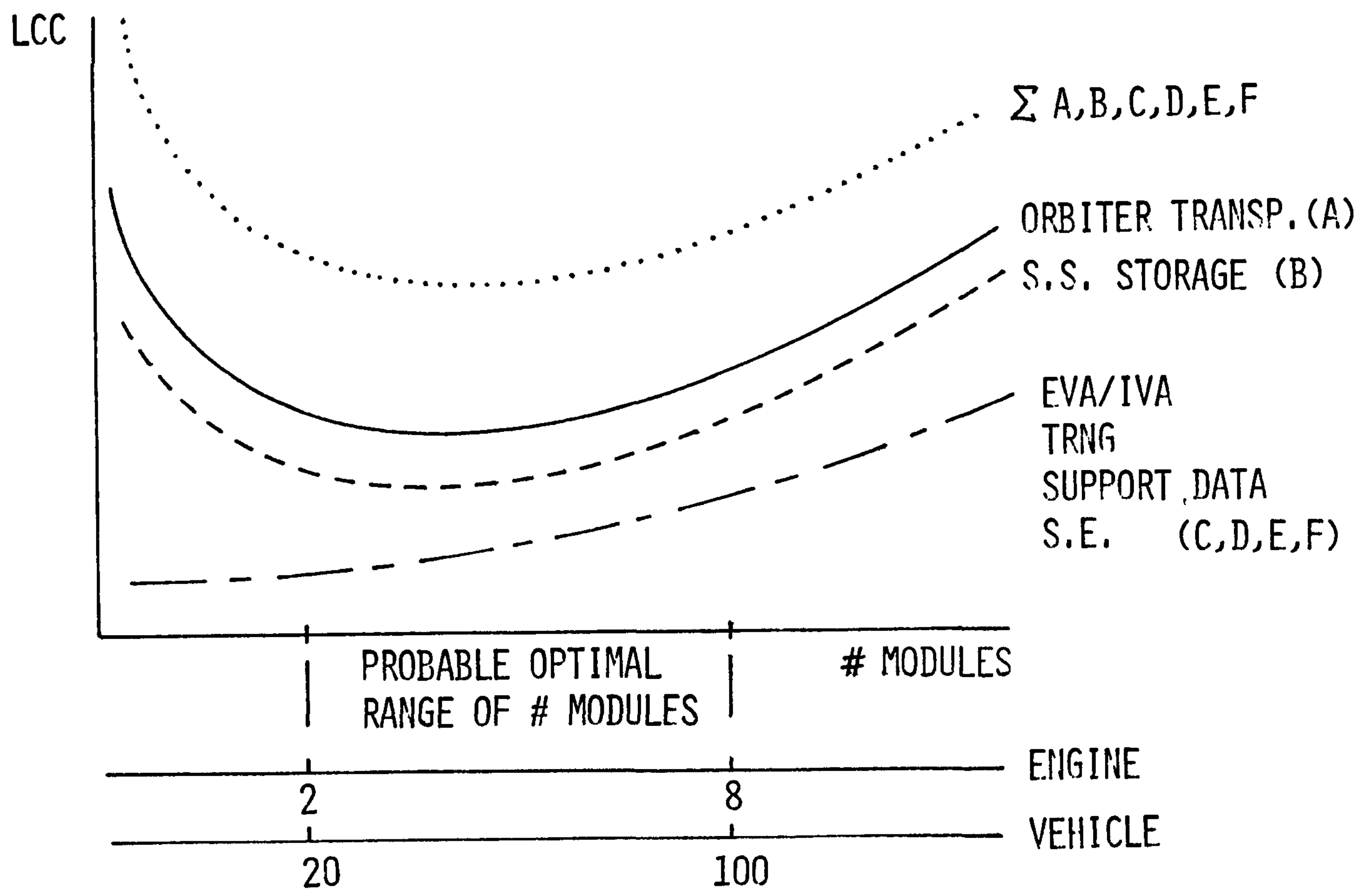


Figure 4.4-26 LCC as a Function of No. Modules

A summary of the study conclusions for Phase II exclusively are tabulated.

- LCC Sensitive to:

  - MPS Reliability (Mission Loss)
  - Vehicle Dry Weight
  - I_{sp}
  - Propellant Cost (amount scavengable, etc)

- LCC Relatively Insensitive to:

  - Number of Missions/Overhaul
  - Failures/Mission
  - EVA and IVA Cost

- Two or Three Engines Recommended Because of:

  - Highest Reliability
  - Minimum LCC - Across Estimated Range of Dependant Failure Rates and Engine Reliability
  - Fail Operational/Fail Safe Capability (affordable manrating criteria)

- Onorbit Servicing is Cost Effective--Proper Degree of Modularity Required:

  - Engine: 2 to 8 modules
  - Vehicle: 20 to 100 modules

5.0 REFERENCES

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5. Space Station Needs, Attributes, and Architectural Options Study. Final Report, Vol. III Mission Requirements. Martin Marietta Aerospace, NASA Contract NASW-3686.