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ORBITAL TRANSFER VEHICLE ENGINE INTEGRATION STUDY

FINAL REPORT

30 November 1984

GENERAL DYNAMICS
Convair Division

GDC-SP-84-050

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FINAL REPORT

30 November 1984

Prepared for
Aerojet TechSystems Company
Sacramento, California

Prepared under
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Prepared by
Advanced Space Programs
GENERAL DYNAMICS CONVAIR DIVISION
San Diego, California

FOREWORD

This report documents the results of contract L-814740, "Orbital Transfer Vehicle Engine Integration Study." This study was conducted by General Dynamics Convair Division (GDC) from March - November 1984 under contract to Aerojet TechSystems Company for NASA-LeRC.

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SUMMARY

NASA-LeRC is sponsoring industry studies to establish the technology base for an advanced engine for orbital transfer vehicles for mid-1990s IOC. Engine contractors are being assisted by vehicle contractors to define the requirements, interface conditions, and operational design criteria for new LO_2-LH_2 propulsion systems applicable to future orbit transfer vehicles and to assess the impacts of space basing, man rating, and low-G transfer missions on propulsion system design requirements.

This report presents the results of a study conducted by GDC under contract to Aerojet for NASA-LeRC. The primary study emphasis was to determine what the OTV engine thrust level should be, how many engines are required on the OTV, and how the OTV engine should be designed. This was accomplished by evaluating planned OTV missions and concepts to determine the requirements for the OTV propulsion system, conducting tradeoffs and comparisons to optimize OTV capability, and evaluating reliability and maintenance to determine the recommended OTV engine design for future development.

Mission analysis resulted in three major mission catagories. GEO Satellite missions accounted for the majority. Low thrust LSS and manned GEO missions are fewer and later, approximately same time as space based OTV IOC and availability of new engine, but more demanding and are, therefore, the discriminators for the OTV propulsion system.

Considering the 7 to 10 year development time for a new engine and the mid-1990s IOC of the LSS and manned missions, the availability of a new space based OTV is expected with advanced engines, composite structure, lightweight tanks, and aerobraking. Although several OTV concepts were considered, an orbiter cargo bay launched, space assembled, symmetrical lifting aerobrake, single stage LO_2-LH_2 OTV was selected for analysis. Substantial performance and economic benefits of advanced engines, lightweight structures, and aeroassist are shown. The characteristics of the advanced engines being considered by Aerojet, Rocketdyne, and Pratt & Whitney were used. Additional parametric data were supplied by the engine contractors for other thrust levels for use in the trade studies. The objective of establishing one engine design required consideration of both the manned and the LSS missions.

The most difficult mission is the manned GEO sortie mission which establishes the maximum vehicle size and the highest thrust requirements, while Large Space Structure (LSS) missions with LEO deployment and checkout determine the minimum thrust requirements. Since these are conflicting requirements for one engine, effort concentrated on resolving this by attempting to determine a design thrust level that would satisfy the manned mission, and with throttling, also

satisfy the LSS missions. This manned mission has previously been assumed to be limited to a high thrust level to reduce gravity losses with a single perigee burn to minimize crew radiation exposure/passes thru the Van-Allen radiation belts. Several recent studies of dedicated OTVs for LSS missions have shown the advantages of multiple perigee burns to minimize gravity losses at the low thrust levels needed to limit acceleration loads on large space structure missions.

Recent work by NASA-LeRC indicates that multiple passes thru the Van-Allen radiation belts would not necessarily incur excessive radiation dosage. Thus, lower thrust could satisfy both the manned mission and the LSS mission. Sensitivity to total thrust for LSS missions was determined showing the advantage of lower thrust levels and multiple perigee burns to obtain the largest LSS diameter; a large symmetrical phased array system was used for analysis. The results indicate that although payload weight capability decreases, the diameter of the payload reaches an optimum at 1000 to 2000 lbf thrust as a result of reduced structural loads. The effects of gravity losses, I_{SP} reduction, and mission transfer losses were included. Sensitivity to total thrust for the manned mission was determined which also shows advantages of lower thrust levels, lighter engines and vehicle systems, and multiple perigee burns to obtain the best payload weight. Optimum total thrust for the manned mission, however, is considerably higher than for LSS missions (6000 to 12000 lbf vs. 1000 to 2000 lbf).

Using radiation data from NASA JSC/LeRC, crew exposure was determined for one, two, and four perigee burns and one week at GEO showing that up to four burns could be tolerated without increasing the current manned module radiation shield thickness. Modified trajectories for further reduced radiation are possible but were not included in this study.

The manned mission requires a very high probability of safe crew return. An overall propulsion system reliability of 0.9997 was selected (based on USA traffic statistics) which would require a single engine of exceptionally high reliability or the need for redundancy. Multiple engines provide for single failure tolerance, eliminating the need for rescue operations, and reduces the number of tests required to demonstrate the needed reliability. A single engine design would have to demonstrate 7600 failure free tests, while a two engine configuration requires only 140 tests. While the ACS (if H_2-O_2) could provide a backup to a single main engine, it is expected that its lower I_{SP} would require additional propellant to be carried. Some OTV missions will be flown prior to the first manned mission, giving the opportunity to help demonstrate the needed reliability. For comparison, the RL-10 engine (based on 69 Centaur flights to date) has a predicted start probability of 0.999797 and failure rate of 509 failures per million hours of operation. Using these numbers, analysis shows that two main engines will attain the desired reliability (0.9997) even with correlation factors, non-independent failure modes, as high as 5 to 10 percent.

Payload optimization for the manned mission was evaluated for the following engine parameters:

- Aerojet, Pratt & Whitney, Rocketdyne
- Thrust, 3000-25000 LBF
- Number of perigee burns, 1-4
- Number of engines, 2-4
- Nozzle area ratio, 600-3000
- Chamber pressure, 1500-2500 PSIA
- Mixture ratio, 5-7
- Fixed, extendible/retractable nozzles

While several vehicle concepts were considered including modular and aft cargo carrier concepts with various aerobrake options, the modular tanks/symmetrical lifting brake concept was selected for the trade studies. Evaluation of the aerobrake/engine interaction determined that doors would be necessary to cover the engines during the aeropass. The interaction of the OTV/Engine/Aerobrake was evaluated. As the engine length increases (function of thrust, area ratio, chamber pressure, fixed vs. extendible nozzles), the aerobrake diameter (weight) must increase to prevent flow stream impingement on the payload. The number of engines and nozzle exit diameter impacts the engine support structure and aerobrake door size. Altogether, these allow trades to determine optimum engine design and sensitivity.

The advantage of lower thrust engines and multiple perigee burns is shown. Additional trades showed the advantages of extendible nozzles, high chamber pressures, and high mixture ratios. A nozzle area ratio of ~ 1000 appeared optimum.

While there is a benefit for designing a long life engine, there appears to be little advantage for reducing the frequency of major overhauls beyond 20 to 30 missions. Major overhaul of the engine for a space based OTV should be done on the ground to reduce cost, while routine maintenance is shown to be advantageous in space for anticipated task manhours.

This study has shown that future OTV engine requirements will be determined by LSS and manned missions. To satisfy the manned reliability requirement, twin engines appear to be needed. The optimum engine thrust level is in the range of

3500 to 6500 lb_f each, depending on the number of perigee burns for the manned mission. Although the lower thrust level is preferable for less vehicle and payload design impact, this is contingent on the acceptance of multiple perigee burns for the manned mission and on the ability of the engine manufacturers to produce a high performance, reliable, maintainable engine at lower thrust with additional starts and longer burn time.

SECTION 1
INTRODUCTION

NASA LeRC is sponsoring industry studies to establish the technology base for an advanced engine for orbital transfer vehicles for mid-1990s IOC. Engine contractors are being assisted by vehicle contractors to define the requirements, interface conditions, and operational design criteria for new LO2-LH2 propulsion systems applicable to future orbit transfer vehicles and to assess the impacts of space basing, man rating, and low-G transfer missions on propulsion system design requirements.

This report presents the results of a study conducted by General Dynamics/ Convair under contract to Aerojet for NASA-LeRC. The primary study emphasis was to determine what the OTV engine thrust level should be, how many engines are required on the OTV, and how the OTV engine should be designed. This was accomplished by evaluating planned OTV missions and concepts to determine the requirements for the OTV propulsion system, conducting tradeoffs and comparisons to optimize OTV capability, and evaluating reliability and maintenance to determine the recommended OTV engine design for future development (Figures 1-1, 1-2).

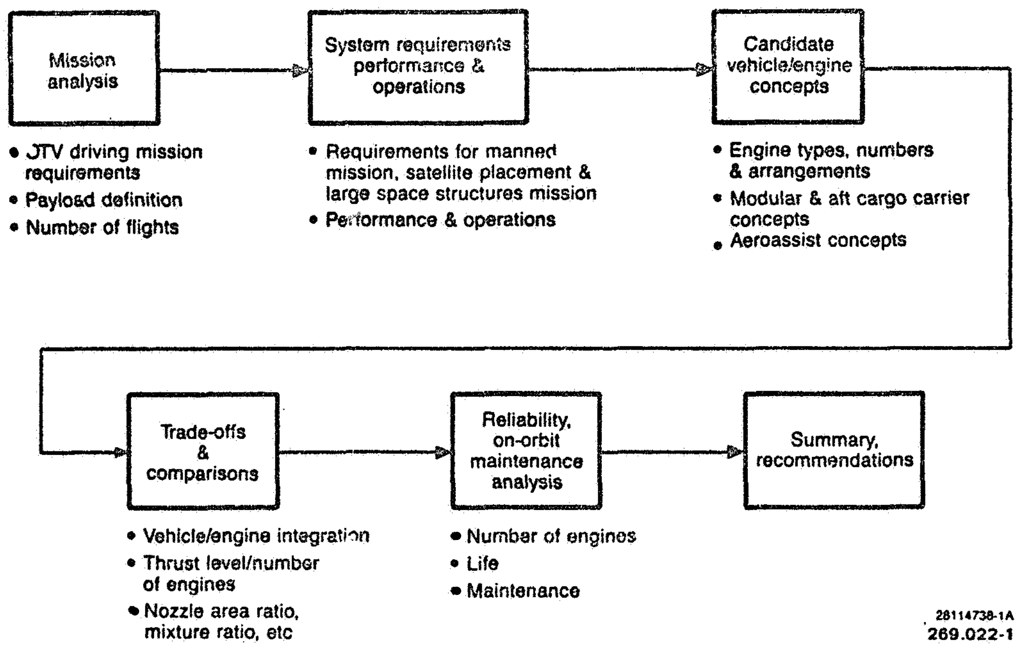

Figure 1-1. OTV Engine Support Study Elements

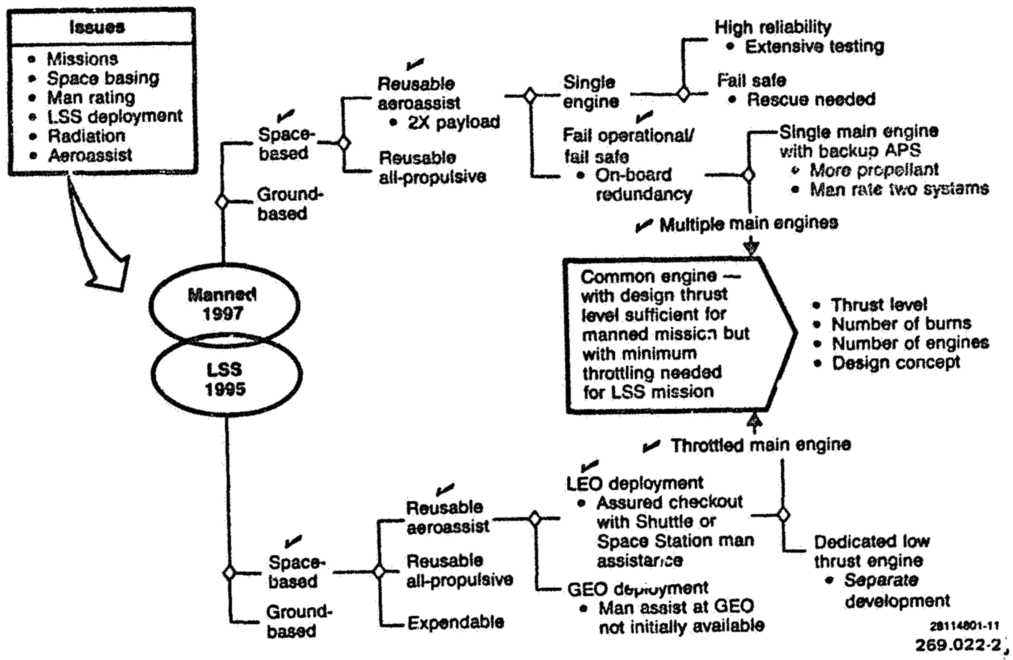

Figure 1-2. Advanced LO_2-LH_2 OTV Engine Definition Approach

SECTION 2

MISSIONS AND REQUIREMENTS

Mission analysis resulted in three major mission categorize (Figures 2-1, 2-2). GEO Satellite missions accounted for the majority. Low thrust LSS and manned GEO missions are fewer and later, approximately same time as space based OTV IOC, and availability of new engine, but more demanding and are, therefore, the discriminators for the OTV propulsion system. The most current NASA mission model and other sources were used to categorize requirements.

- GEO satellite missions
  - 70% commercial & NASA market share — 5 to 7 missions per year (3 to 4 satellites manifested on each mission ≈ 10,000 lb)
  - Servicing — 2 missions per year
  - DoD — 6 missions per year
- Low thrust LSS missions
  - 10,000 to 16,000 lb payload
  - 2 to 4 missions per year
- Manned GEO sortie missions
  - 1 per year
  - 13,000 lb payload round-trip

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Figure 2-1. OTV Missions

MISSION GROUP	WEIGHT (LB) UP/DOWN	PROPELLANT* REQUIREMENT (LB)	MISSION MODEL LOW	MISSION MODEL NOM	IOC
EXPERIMENTAL GEO PLATFORM	12000/0	34800	1	1	1998/1994
OPERATIONAL GEO PLATFORM	20000/0	46300	11	18	2000/1996
UNMANNED GEO PLATFORM SERVICING	7000/4500	33300	8	16	2000/1995
MANNED GEO SORTIE	6500/6500 OR 14000/14000	35200 OR 55700	8	9	2003/1997
GEO STATION ELEMENTS	13000-20000/0	38200 – 46300	2	3	2001/2002
UNMANNED GEO STATION LOGISTICS	10000/2700	35300	19	0	2000/–
MANNED GEO STATION LOGISTICS	16500/8000	52800	0	34	2012/2002
PLANETARY	2000-31000/0	–	12	19	1993/1994
UNMANNED LUNAR DELIVERY	5000-20000/0	32100 – 53400	3	3	2001/2001
MANNED LUNAR SORTIE	80,000/15,000	150,600	3	3	2007/2006
LUNAR BASE ELEMENTS	80,000/0	138,700 **	3	3	2009/2008
LUNAR BASE SORTIE LOGISTICS	80,000/10,000	146,700	2	6	2010/2009
MULTIPLE GEO PAYLOAD DELIVERY	9000-15300/2000	33000 – 42100	19	47	1998/1994
LARGE GEO SATELLITE DELIVERY	10000-20000/0	46300	27	35	1998/1994
UNMANNED GEO SATELLITE SERVICING	7000/4500	33300	9	86	2002/1999
DoD		–	137	137	1993/1993
SDI		–			
MANNED PLANETARY		–			

*LO_2/LH_2 SINGLE STAGE (AEROBRAKED, 485 SEC ISP)

**ASSUMES MULTIPLE OTVs

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Figure 2-2. OTV Mission Requirements

SECTION 3

OTV CONCEPTS AND ENGINES

Considering the 7 to 10 year development time for a new engine and the mid-1990s IOC of the LSS and manned missions, the availability of a new space based OTV (Figures 3-1, 3-2, and Table 3-1) is expected with advanced engines, composite structure, lightweight tanks, and aerobraking. Although several OTV concepts were considered (Figure 3-3), an orbiter cargo bay launched, space assembled, symmetrical lifting aerobrake, single stage OTV was selected for analysis.

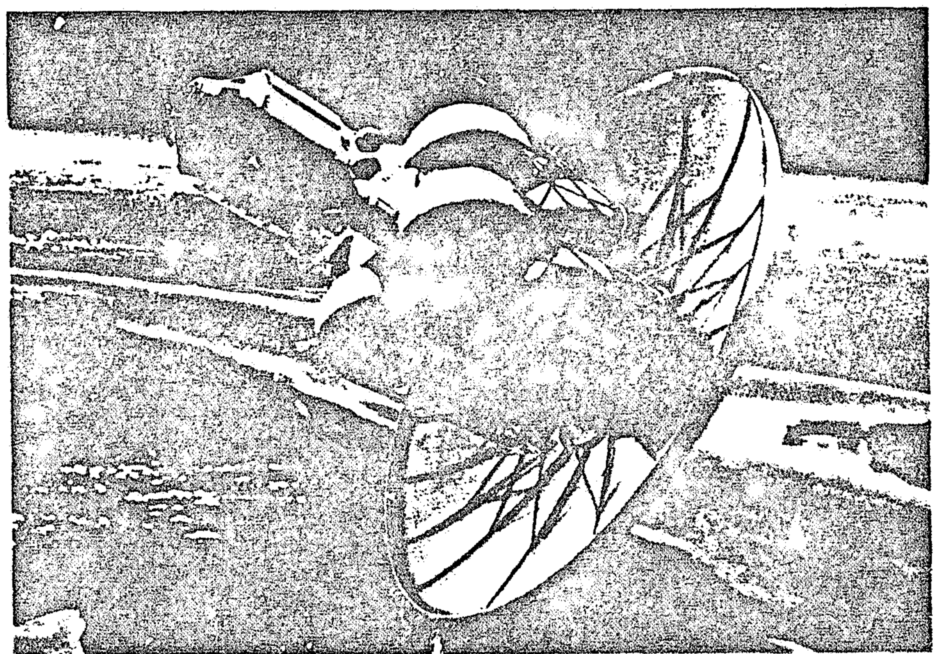

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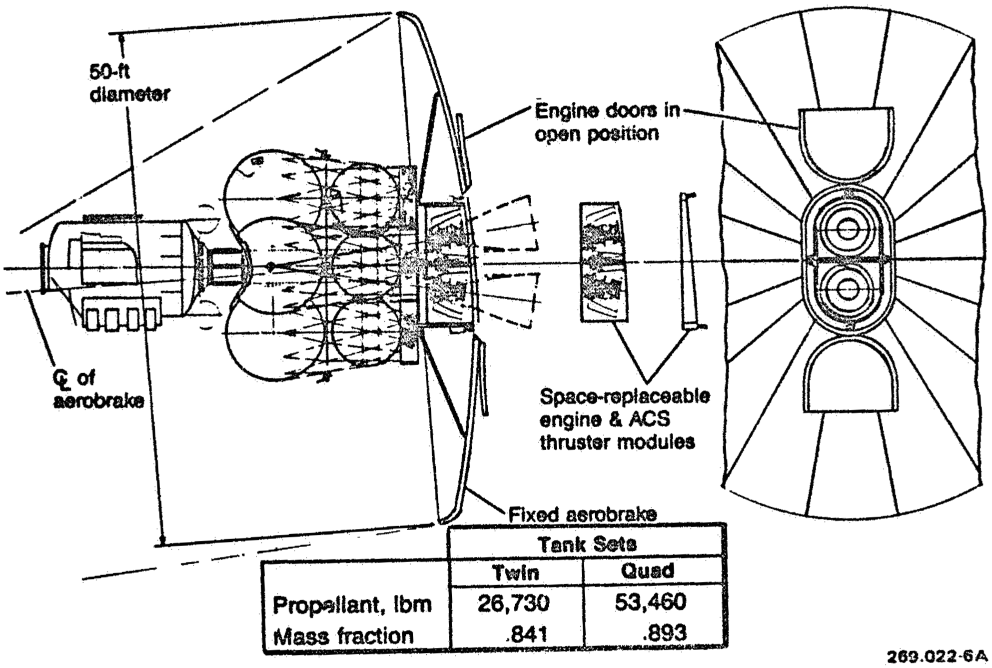
Figure 3-1. Space Based OTV

Besides higher I_{sp} engines, several other technologies have been identified that will make OTV reuse economically beneficial. These include reducing the inert OTV weight and utilizing aeroassist.

Reduced weight can be achieved with advanced structures (composites) by decreasing the loads imposed during launch and powered operation and by reduced tank pressures. Decreased loads are possible by initially launching the OTV from earth without propellant, and by low thrust powered operation.



	Tank Sets	
	Twin	Quad
Propellant, lbm	26,730	53,460
Mass fraction	.841	.893

Figure 3-2. LO_2-LH_2 Space Based OTV

Space basing allows the OTV to be launched initially without propellants and refueled on-orbit. Since the loaded tanks will be exposed only to a vacuum, internal pressures need not exceed those resulting from propellant vapor pressures which can be just above the triple point, possibly not exceeding 3 psia, as opposed to sea-level saturation conditions (> 14.7 psia) for a ground based OTV.

Once fueled, loads can be minimized by use of low thrust during powered operation which will be needed for certain payloads, e.g., Large Space Structures.

Besides inert weight reduction, the technology of aeroassist can reduce the propulsive ΔV requirement for return from GEO by 50 percent (7000 fps versus 14,000). For manned, round trip missions, this results in a 50 percent reduction in OTV propellant required.

To achieve these improvements, technology development is needed in each area.

Table 3-1. Modular Tank Space-Based OTV, Weights Summary

	Tank Sets	
	Quad	Twin
Core assembly	1,647	1,647
• Main engine & TVC	322	322
• Docking system	24	24
• Astrionics	276	276
• Forward & aft service bulkheads	130	130
• Structure	216	216
• Electrical power	291	291
• ACS & tank pressurization	218	218
• Tank module disconnects/attaches	20	20
• Main propellant feed	70	70
• Contingency	80	80
Outrigger tank sets	2,390	1,196
• Propellant tankage & fittings	314	157
• Insulation	348	174
• Propellant acquisition & feed	648	324
• Structure	789	395
• Instrumentation	49	25
• ACS & tank pressurization	87	43
• Tank module disconnects/attaches	35	18
• Contingency (5%)	120	60
Auxiliary fluids	180	100
• ACS usable propellant	90	60
• Fuel cell reactants	90	40
Residuals, boiloff & other losses	220	160
Aerobrake & associated structure	1950	1950
BURNOUT WEIGHT	6387	5053
USABLE MPS PROPELLANT	53460	26730
USABLE MPS PROPELLANT MASS FRACTION	.893	.841

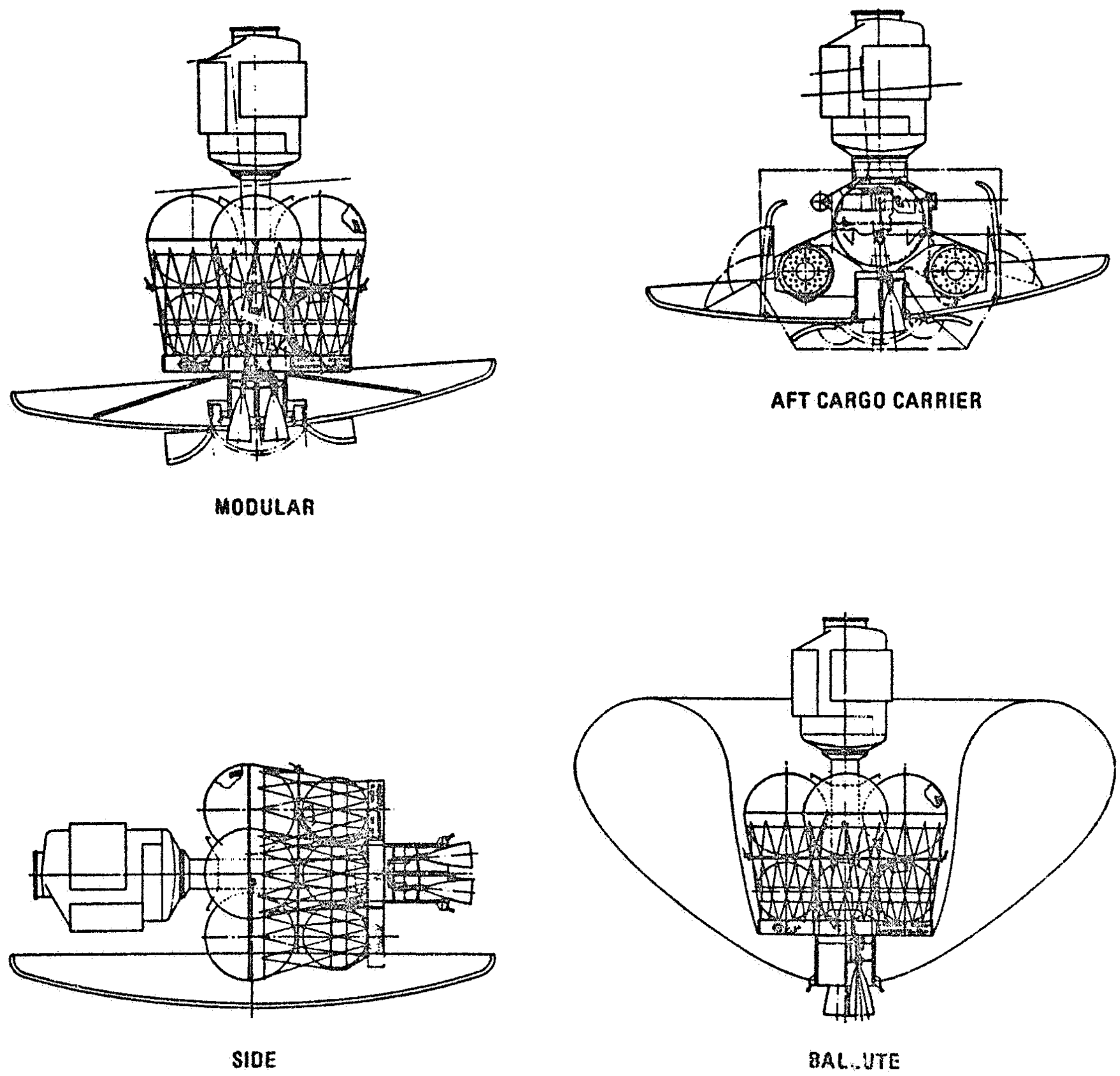


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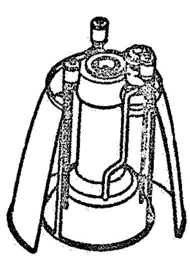
Figure 3-3. Alternate OTV Configurations

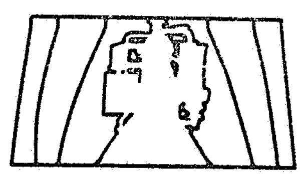
The characteristics of the advanced engines being considered by Aerojet, Rocketdyne, and Pratt & Whitney are shown in Figure 3-4. Additional parametric data were supplied by the engine contractors for other thrust levels for use in the trade studies. The payoff of advanced technologies shows the advantages of advanced engines, lightweight tanks/structure, and aero assist capability (Figure 3-5). Advanced engines and lightweight tanks/structure give high payoff for payload delivery missions. Aeroassist gives high payoff for payload round trip missions, but payload delivery missions are very sensitive to aerobrake weight.

Figure 3-6 shows the substantial economic benefit of a new engine.

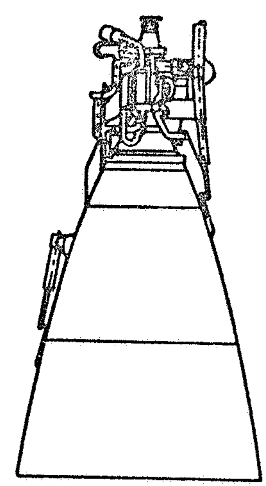
PARAMETER	AEROJET	ROCKETDYNE	PRATT & WHITNEY
THRUST, LBF	3,000	15,000	15,000
CYCLE	EXPANDER H_2-O_2	EXPANDER H_2	EXPANDER H_2
CHAMBER PRESSURE, PSIA	2,000	2,000	1,500
NOZZLE AREA RATIO	1,200:1	1,300:1	640:1
SPECIFIC IMPULSE, LBF-SEC/LBM	>480	>480	>480
TURBOMACHINERY SPEEDS, RPM			
• H_2	200,000	178,000	150,000
• O_2	75,000	56,200	67,390
CONTROL	CLOSED LOOP	CLOSED LOOP	OPEN LOOP
THROTTLEABILITY	30:1	30:1	30:1
• RANGE	2 STEPS	3 STEPS	3 STEPS
• MODE	(15:1 CONTINUOUS)	DISCRETE	DISCRETE
KEY TECHNOLOGIES	GASEOUS OXYGEN DRIVE TURBINE ANNULAR THRUST CHAMBER MULTIPLE ENGINE CONTROL	HYDROGEN PUMP CRITICAL SPEED MULTIVARIABLE CLOSED LOOP CONTROLS HIGH AREA RATIO NOZZLE	HIGH SPEED HYDROGEN COOLED GEARS ADVANCED THRUST CHAMBER MATERIAL HIGH AREA RATIO NOZZLE



AEROJET



ROCKETDYNE



PRATT & WHITNEY

- ADDITIONAL PARAMETRIC DATA WAS SUPPLIED FOR OTHER THRUST LEVELS

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Figure 3-4. Advanced OTV Propulsion System Concepts

- ADVANCED ENGINE & LIGHTWEIGHT TANKS AND STRUCTURE GIVE HIGH PAYOFF FOR PAYLOAD DELIVERY MISSIONS
- AEROASSIST GIVES HIGH PAYOFF FOR PAYLOAD ROUND-TRIP MISSIONS
- PAYLOAD DELIVERY MISSIONS VERY SENSITIVE TO AEROBRAKE WEIGHT

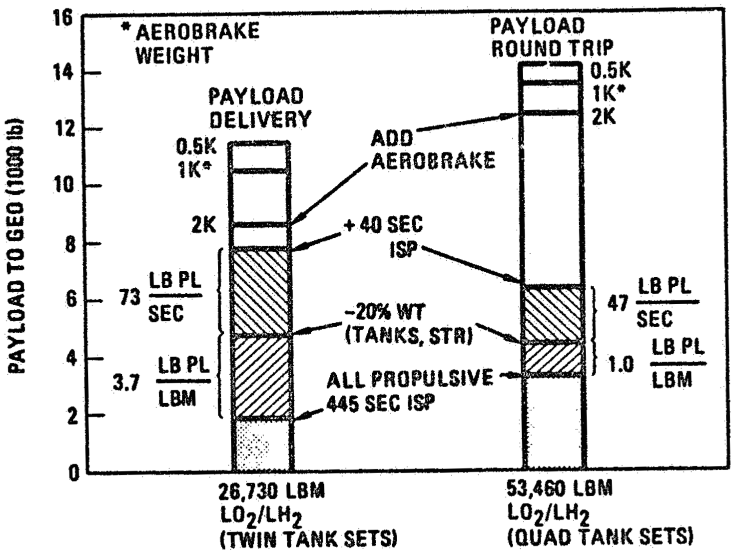

Figure 3-5. Cryogenic, Reusable Space-Based OTV Technology Payoff (Lightweight Tank, Structure, Advanced Engineer, Aeroassist)

$\$/LB_{PL}$	AP[1]	AB[2]	Δ	M$*
RL10 CAT II	6249	5732	517	77.6
NEW ENG	5440	5191	249	37.4
Δ	809	541	1058	
M $*	121	81.1		159

- PAYLOAD DELIVERED TO GEO (NO RETURN)
- OTV ROUND TRIP (LEO-GEO-LEO)
- TWIN TANK SET MODULAR SBOTV (LO_2/LH_2)
- 485 SEC ISP NEW ENGINE
- 1950 LB AEROBRAKE

– NEW ENGINE (+ 25 SEC I_S) SAVES $809/$LB_{PL}$.	$121M/YEAR BENEFIT*
– AEROBRAKING SAVES $517/$LB_{PL}$.	$78M/YEAR BENEFIT*
– NEW ENGINE & AEROBRAKING SAVES $1058/$LB_{PL}$.	$159M/YEAR BENEFIT*

- NEW ENGINE OFFERS 76% OF TOTAL BENEFIT
- AEROBRAKING OFFERS 49% OF TOTAL BENEFIT

1 ALL PROPULSIVE
2 AEROBRAKED

COST ASSUMPTIONS

• OTV TURNAROUND, $M	– 7
• PROPELLANT DELIVERY TO LEO, $/LB-M	– 500
• PAYLOAD DELIVERY TO LEO, $M	– 0.0028 X PAYLOAD WEIGHT

*150,000 LB_M CUMULATIVE PAYLOAD PER YEAR TO GEO (18 OTV MISSIONS PER YEAR)

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Figure 3-6. Economic Benefit of Advanced Engine

SECTION 4

MISSIONS, SYSTEMS, PERFORMANCE INTERACTION AND TRADES

The objective of establishing one engine design required consideration of both the manned and the LSS missions (Figure 4-1). The most difficult mission is the manned GEO sortie mission which establishes the maximum vehicle size and the highest thrust requirements, while LSS missions with LEO deployment and checkout, determine the minimum thrust requirements. Since these are conflicting requirements for one engine, effort concentrated on resolving this by attempting to determine a design thrust level that would satisfy the manned mission, and with throttling, also satisfy the LSS missions.

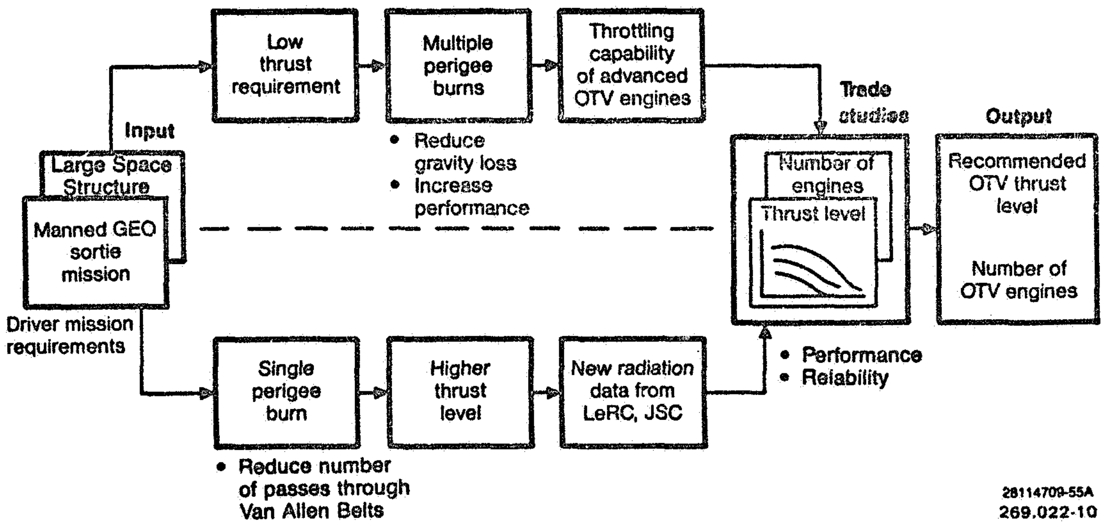


Figure 4-1. Mission, Systems and Performance Analysis Interaction

The manned mission has previously been assumed to be limited to a high thrust level to reduce gravity losses with a single perigee burn to minimize crew radiation exposure/passes thru the Van-Allen radiation belts. Several recent studies of dedicated OTVs for LSS missions have shown the advantages of multiple perigee burns to minimize gravity losses at the low thrust levels needed to limit acceleration loads on large space structure missions.

Recent work by NASA-LeRC indicates that multiple passes thru the Van-Allen radiation belts would not necessarily incur excessive radiation dosage. Thus, lower thrust could satisfy both the manned mission and the LSS mission.

4.1 THRUST LEVEL

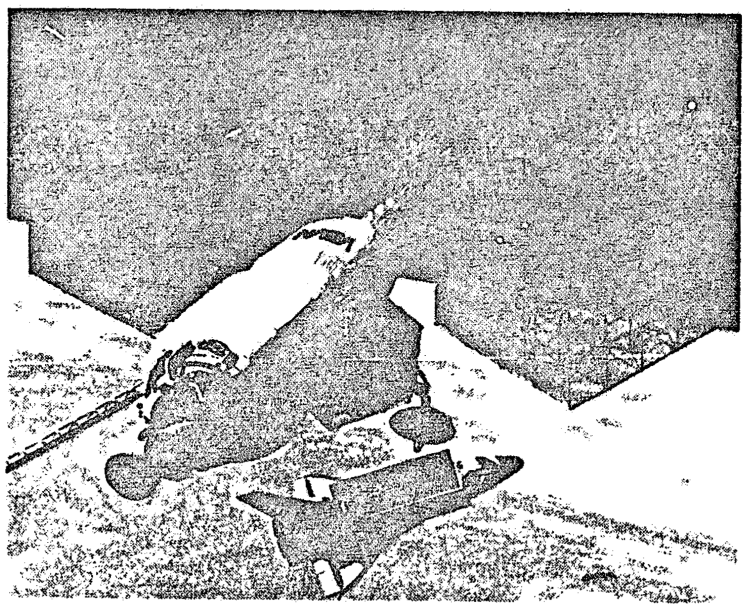
Sensitivity to total thrust, for LSS missions was determined showing the advantage of lower thrust levels and multiple perigee burns to obtain the largest LSS diameter. A large symmetrical phased array system shown in Figure 4-2 was used for analysis. The results indicate that although payload weight capability decreases, the payload diameter reaches an optimum at 1000 to 2000 lbf thrust as a result of reduced structural loads (Figures 4-3, 4-4). Effects of gravity losses, I_{sp} reduction, and mission transfer losses have been included (Figure 4-5).

Sensitivity to total thrust for the manned mission was determined (Figure 4-6) which also shows advantages of lower thrust levels, lighter engines and vehicle systems, and multiple perigee burns to obtain the best payload weight. Optimum total thrust for the manned mission, however, is considerably higher than for LSS missions, 6000 to 12000 lbf vs. 1000 to 2000 lbf.

Using radiation data from NASA JSC/LeRC, crew exposure was determined for one, two, and four perigee burns and one week at GEO, showing that up to four burns could be tolerated without increasing the current manned module radiation shield thickness (Figures 4-7, 4-8, 4-9). Modified trajectories for further reduced radiation are possible but were not included in this study.

4.2 SINGLE AND MULTIPLE ENGINES

The manned mission requires a very high probability of safe crew return. An overall propulsion system reliability of 0.9997 was selected, based on USA traffic statistics, which would require a single engine of exceptionally high reliability or the need for redundancy. Multiple engines provide for single failure tolerance, eliminating the need for rescue operations, and reduces the number of tests required to demonstrate the needed reliability. Figure 4-10 shows that a single engine design would have to demonstrate 7600 failure free tests, while a two engine configuration requires only 140 tests. While the ACS (if H_2-O_2) could provide a backup to a single main engine, it is expected that its lower I_{sp} would require additional propellant to be carried (Figure 4-11). Some OTV missions will be flown prior to the first manned mission (Figure 4-12), giving the opportunity to help demonstrate the needed reliability. For comparison, the RL-10 engine, based on 69 Centaur flights to date, has a predicted start probability of 0.999797 and failure rate of 509 failures per million hours of operation. Using these numbers, analysis shows that two main engines will attain the desired reliability (0.9997) even with correlation factors, non-independent failure modes, as high as 5 to 10 percent (Figure 4-13).



GEO – PLATFORM

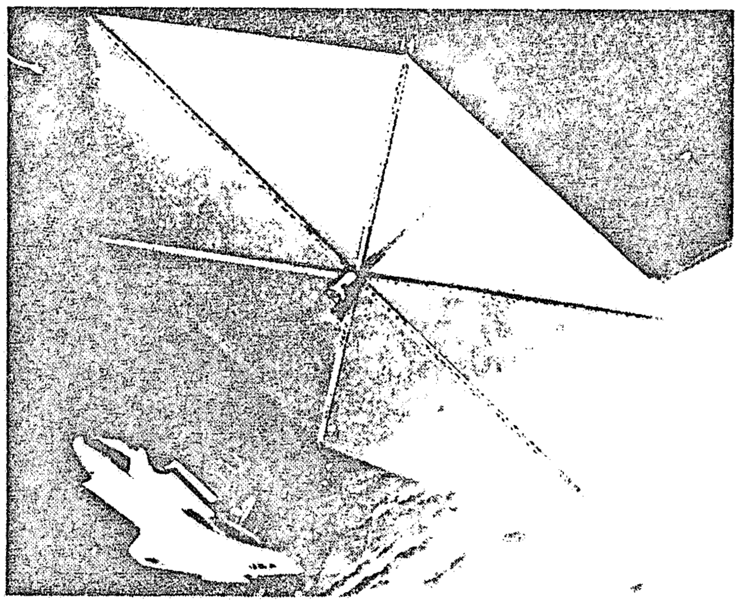


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Figure 4-2. Large Space Structures

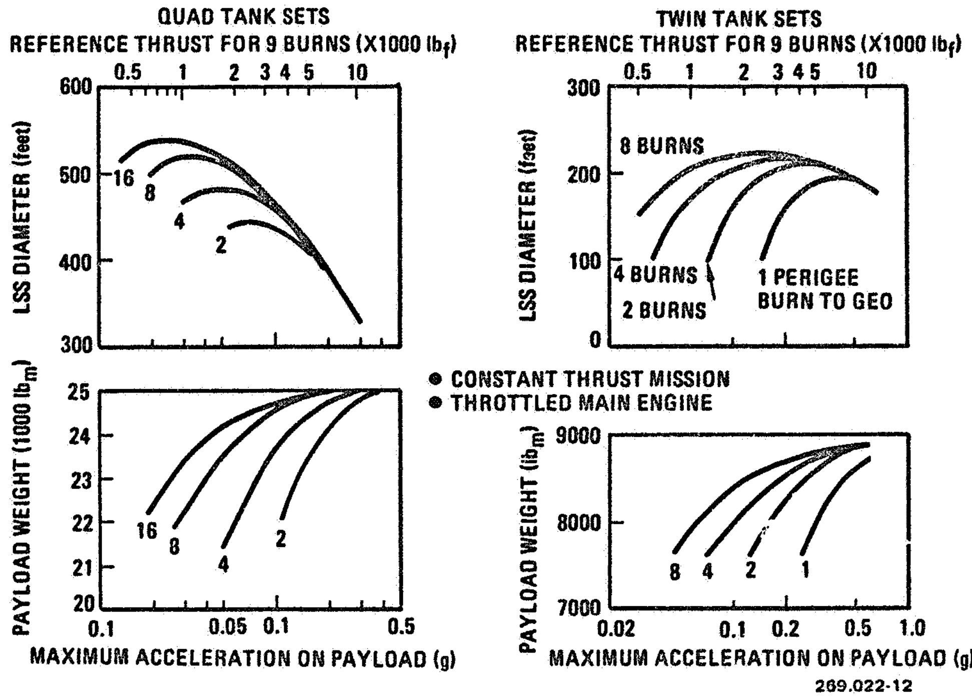

Figure 4-3. Payload Sensitivity to Total Thrust-LSS Missions

Payload weight = 16,000 lbm
Thrust = 2,000 lbf
Isp = 465 s

Burn No.	ΔV (ft/s)	Alt. at Init. of Burn (nmi)	Burn Duration (H:M:S)	Propellant Required (lbm)	Coast Time (H:M:S)
1	781	220	00:12:28	3,086	01:18:00
2	935	311	00:14:08	3,497	01:30:00
3	965	443	00:13:43	3,396	01:48:00
4	1,110	561	00:14:46	3,655	02:18:00
5	1,285	664	00:15:50	3,919	03:00:00
6	1,530	1,006	00:17:14	4,264	04:42:00
7	1,835	1,319	00:18:34	4,592	04:24:00
8	5,572	GEO	00:4[illegible]	11,050	—
Payload separation					
9 (deorbit)	6,095	GEO	00:12:44	3,152	05:15:00
250,000 ft aeropass perigee					
10 (phasing)	500	210	00:00:50	208	—
Total			2:44:57	40,819	

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Figure 4-4. LSS Mission Operations

- ATTITUDE CONTROL — 2 – 3 LB/HR (SYMMETRICAL OTV AND PAYLOAD)
- POWER — 0.5 – 1.0 LB/HR (1 – 2 KW FUEL CELL)
- BOILOFF, LEAKAGE — – (ISOLATED/INSULATED TANKS; ULLAGE EXPANSION COOLING WITH MULTIPLE BURN)

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Figure 4-5. Mission Transfer Coast Period Propellant Losses

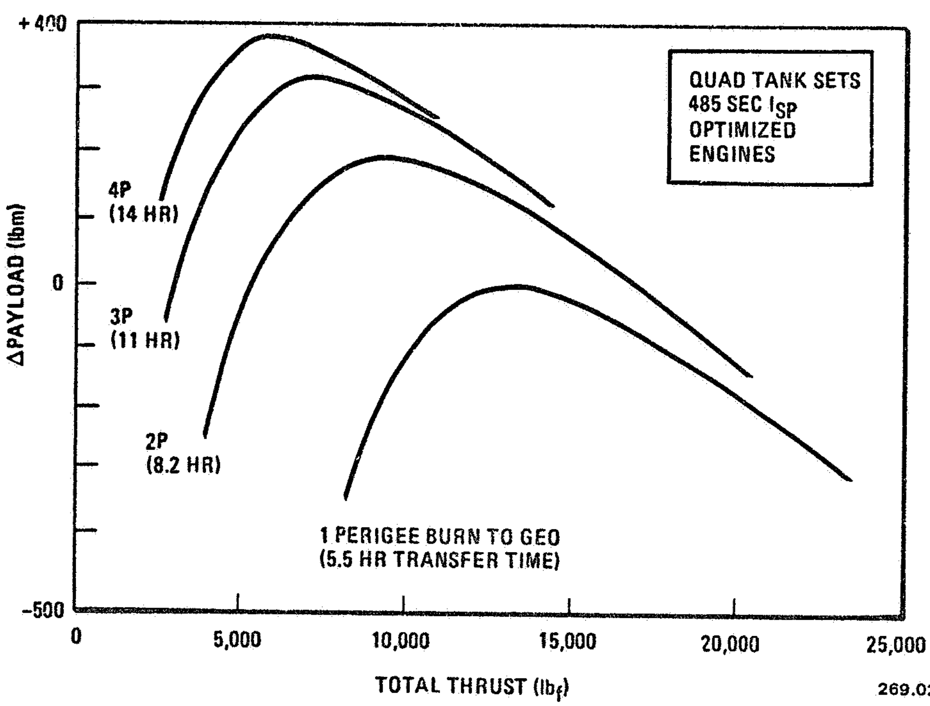

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Figure 4-6. Payload Sensitivity to Total Thrust-Manned Mission

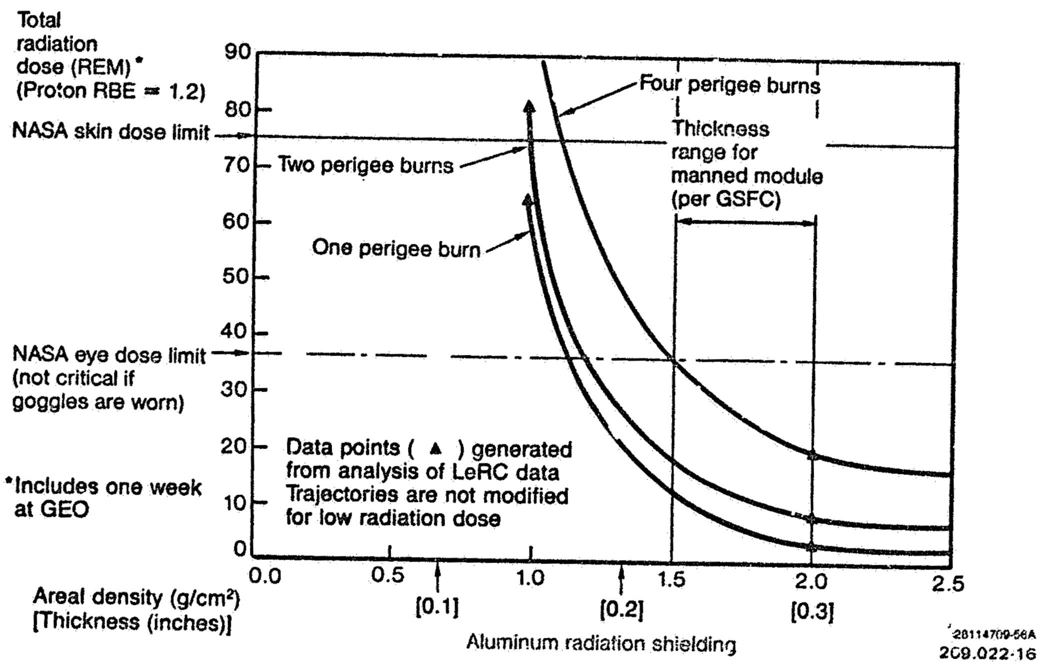

Figure 4-7. Radiation Exposure-Manned GEO Sortie Using Multiple Perigee Burns

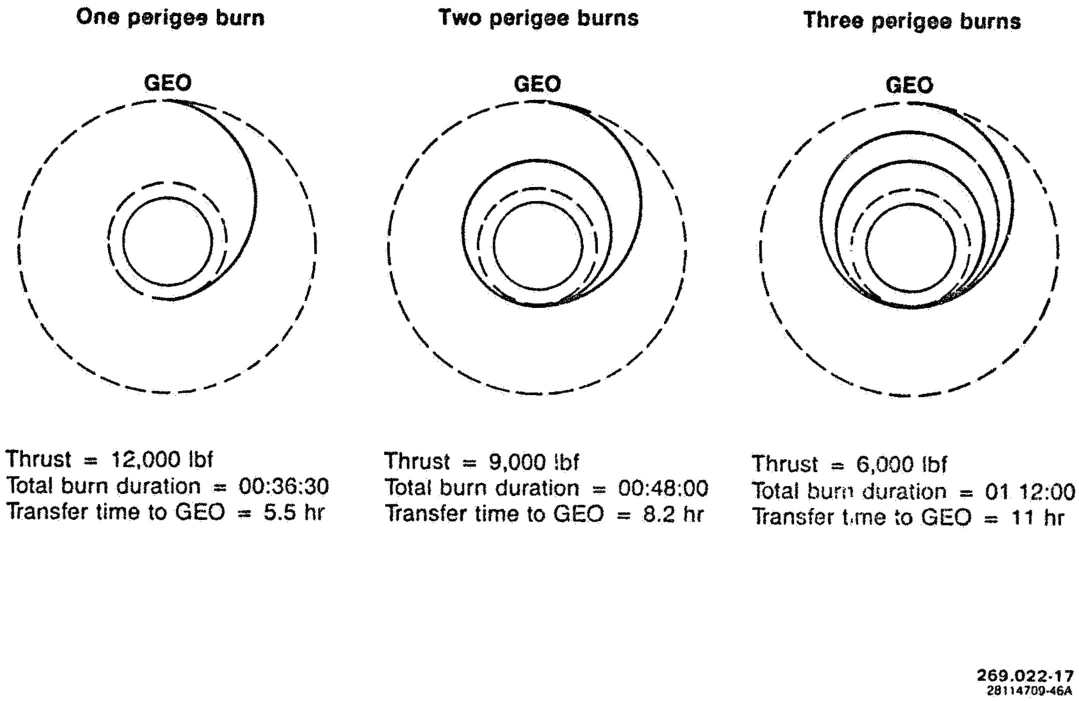

Figure 4-8. Manned GEO Sortie Mission Options

Burn no.	ΔV (ft/s)	Alt. at Init of Burn (nmi)	Burn Duration (H:M:S)	Propellant Required (lbm)	Coast Time (H:M:S)
1	2,368	256	00:13:48	10,248	01:56:24
2	2,656	343	00:13:11	9,788	03:25:04
3	3,032	430	00:12:33	9,315	05:04:11
4	5,870	19,348	00:18:21	13,617	—
5 (deorbit)	6,095	GEO	00:12:59	9,641	05:09:36
250,000 ft aeropass perigee					
6 (phasing)	500	250	00:0:51	636	—
Total	**20,521**		**1:11:43**	**53,245**	

Total thrust = 6,000 lbf
Isp = 485s
Payload weight = 13,000 lbm

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Figure 4-9. Manned GEO Sortie Mission Operations

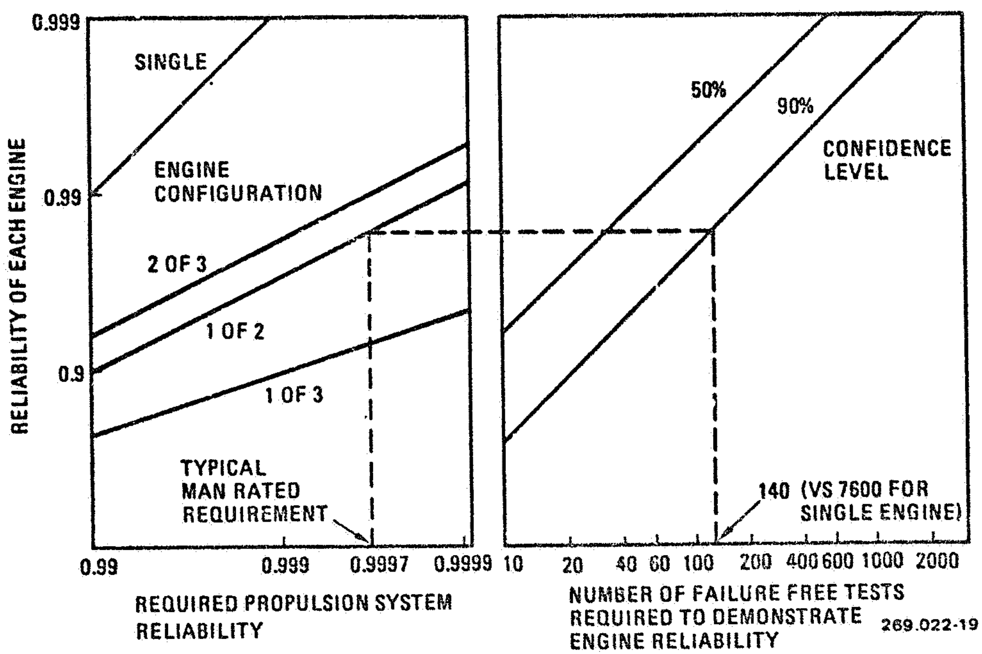

Figure 4-10. Required Number of Engine Tests Needed to Demonstrate Propulsion System Reliability As A Function of Engine Configuration (Zero Percent Correlation)

	2 Main Engines	1 Main Engine + APS	Δ Propellant
I_{sp}, sec	485/485	485/425	
Return propellant, lbm	10,970	12,960	+ 1,990
Ascent propellant, lbm	44,070	46,955	+ 2,885
Total, lbm	**55,040**	**59,915**	**+ 4,875**

13,000 lbm payload round trip to GEO
6,400 lbm OTV burnout weight
Main engine failure at GEO

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Figure 4-11. Additional Propellant Penalty Required for ACS

48-65 OTV missions will be flown prior to Manned GEO Sortie IOC

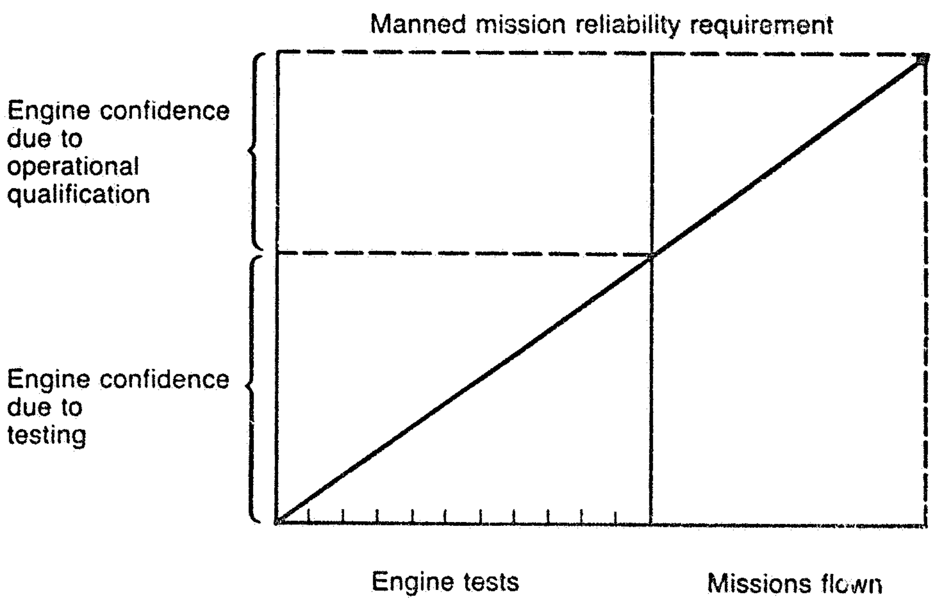

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Figure 4-12. OTV Engine Reliability will Be Assured Through Testing and Operational Qualification

n Total Number of Burns	t Total Burn Time (hours)	Propulsion System Reliability (Goal R = .9997)*			
			Fail Operational (2 Engine) System		
			Correlation Factor		
		Single Engine	0%	5%	10%
4	.60	.998883	.999999	.999887	.999776
5	.79	.998584	.999998	.999857	.999715
6	1.19	.998178	.999997	.999815	.999633

* $R_1 = R_s^n e^{-\lambda t}$: 1 engine

$R_2 = R_1^2 + 2R_1 (1-R_1)(1-C)$: 2 engines

Where R_s = .999797 (start, stop) }
$\lambda = 509 \times 10^{-6}$ failures per hour } RL10 engine
C = 0, .05, 0.1 (correlation)

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Figure 4-13. Manned OTV Propulsion System Reliability

4.3 TRADE-OFFS

Payload optimization for the manned mission was evaluated for the following engine parameters (Appendix I):

a. Aerojet, Pratt & Whitney, Rocketdyne

b. Thrust, 3000 - 25000 LBF

c. Number of perigee burns, 1 - 4

d. Number of engines, 2 - 4

e. Nozzle area ratio, 600 - 3000

f. Chamber pressure, 1500 - 2500 PSIA

g. Mixture ratio, 5 - 7

h. Fixed, extendible/retractable nozzles

While several vehicle concepts were considered, including modular and aft cargo carrier concepts with various aerobrake options, the modular tanks/symmetrical lifting brake concept was selected for the trade studies. Evaluation of the aerobrake/engine interaction determined that doors would be necessary to cover the engines during the aeropass, because of concerns of vehicle stability and control, flow field interactions, engine cooling, and leakage of base gasses to the OTV.

The interaction of the OTV/Engine/Aerobrake is shown (Figure 4-14). As the engine length increases (function of thrust, area ratio, chamber pressure, fixed vs. extendible nozzles), the aerobrake diameter (weight) must increase to prevent flow stream impingement on the payload. The number of engines and nozzle exit diameter impacts the engine support structure and aerobrake door size. Altogether, these allow trade-offs to determine optimum engine design and sensitivity.

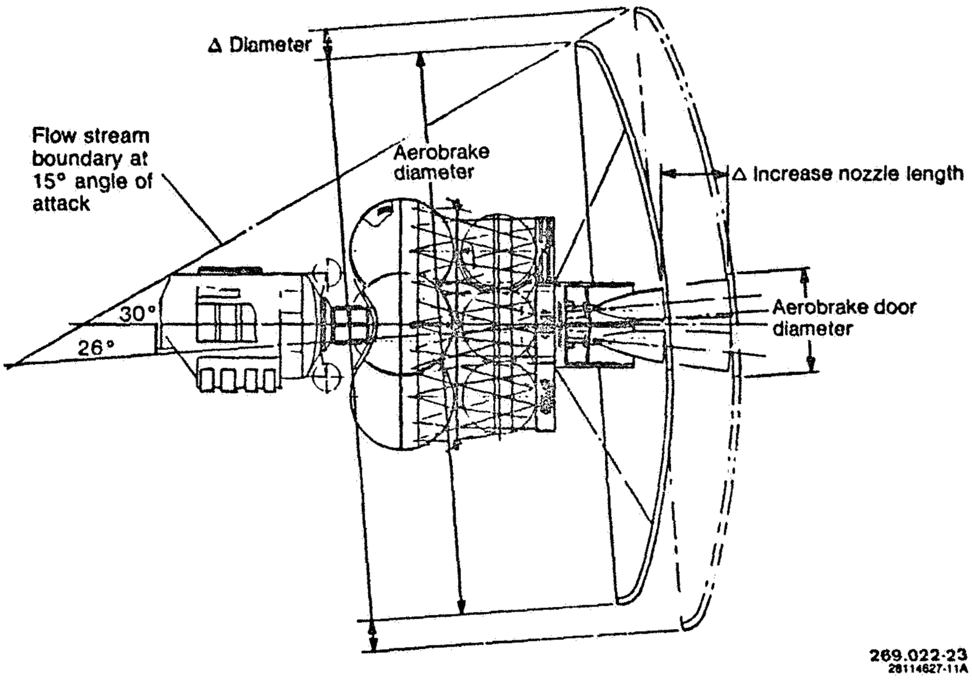

Figure 4-14. Engine Trade Study Impact on Vehicle Design

The advantage of lower thrust engines and multiple perigee burns is shown. (Figure 4-15). A nozzle area ratio of ~ 1000 appeared optimum (Figure 4-16). Additional trades (Figure 4-17 to 4-19) showed the advantages of extendible nozzles, high chamber pressures, and high mixture ratios.

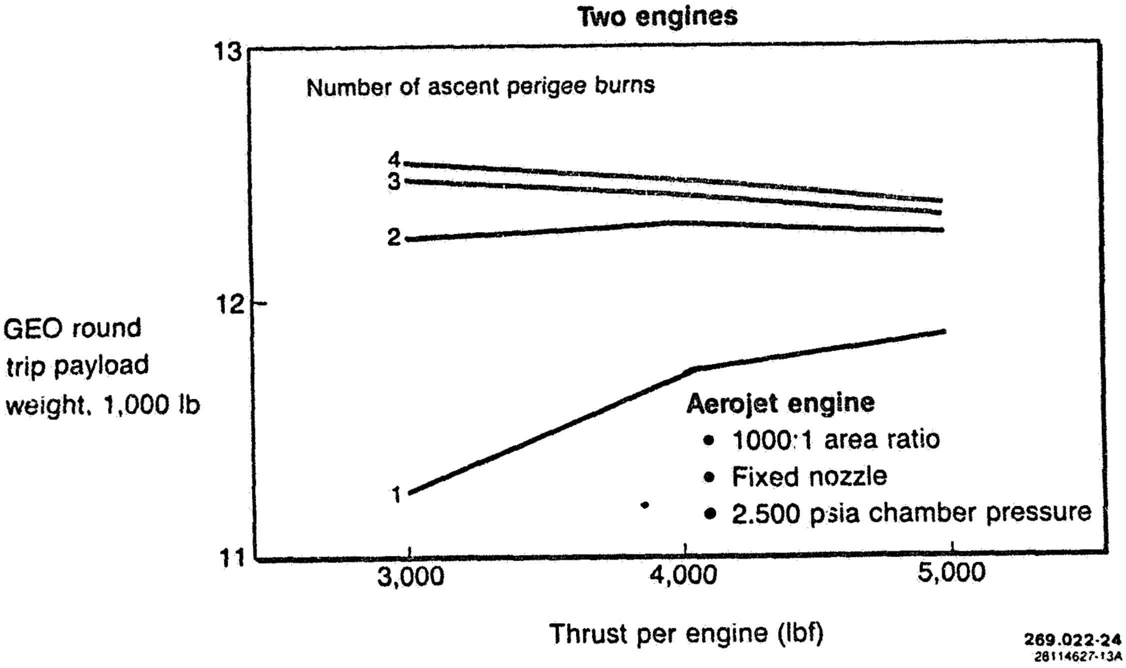

Figure 4-15. Thrust Level, Perigee Burns Trade

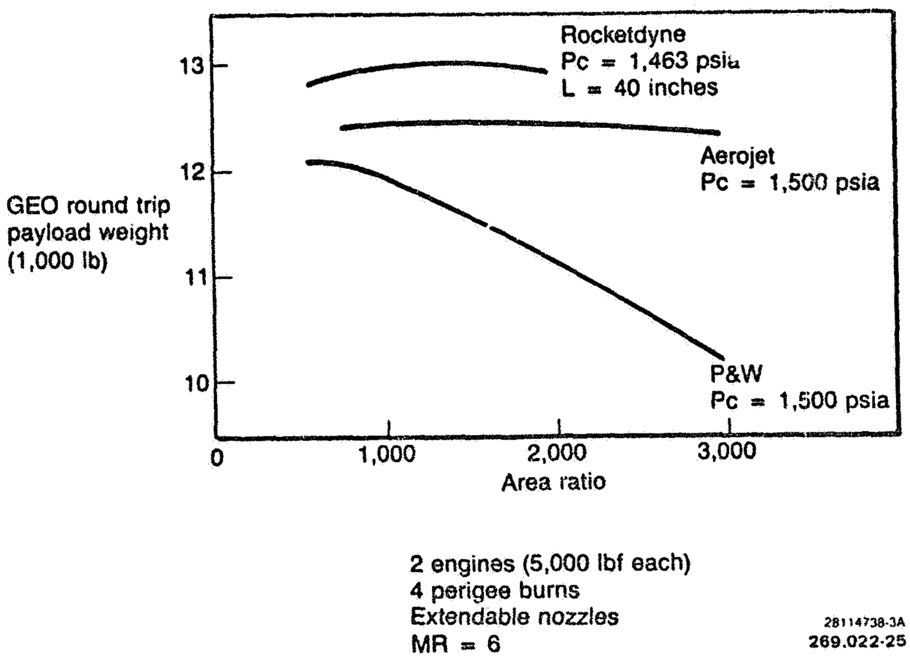

Figure 4-16. Area Ratio Trade

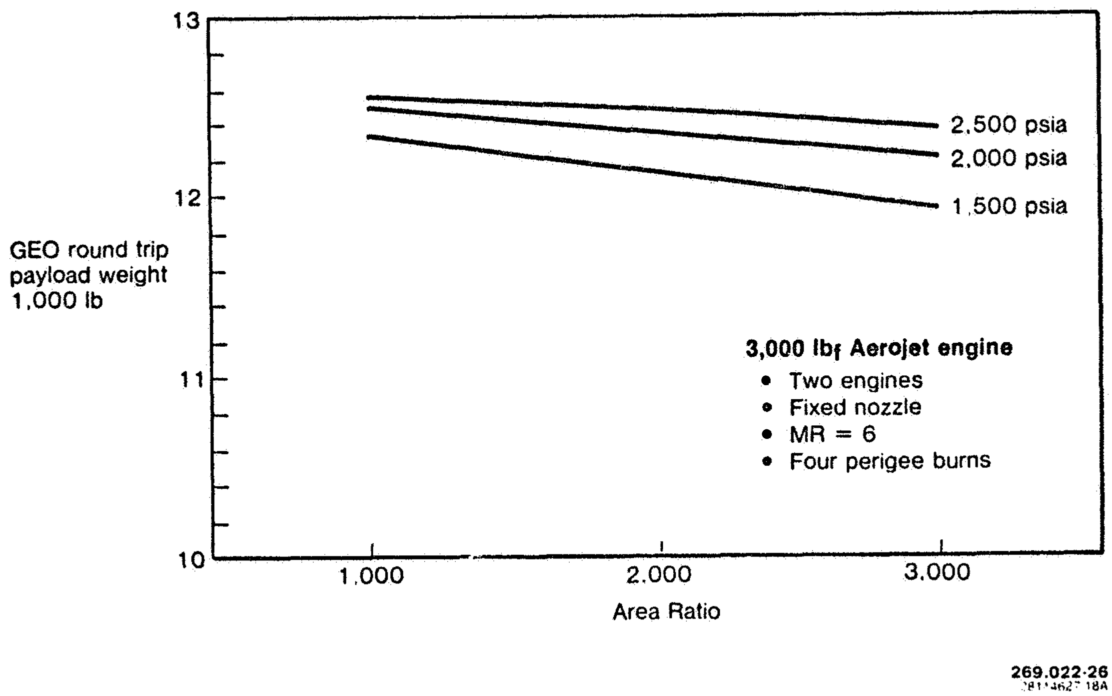

Figure 4-17. Area Ratio, Chamber Pressure Trade

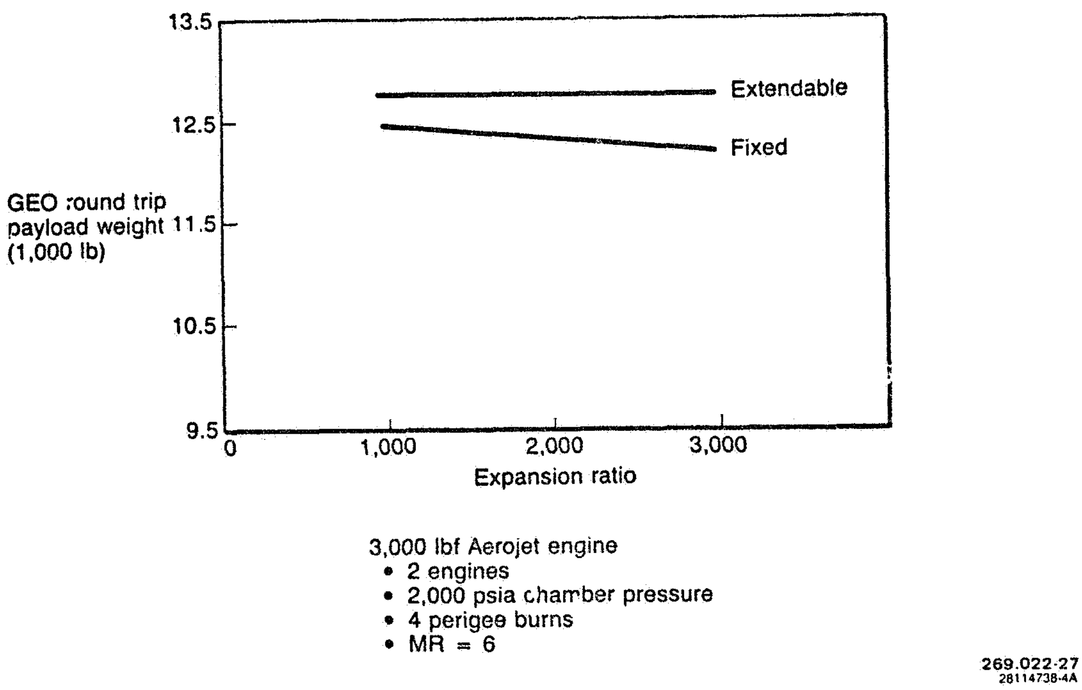

Figure 4-18. Fixed Vs Extendible Nozzles Trade

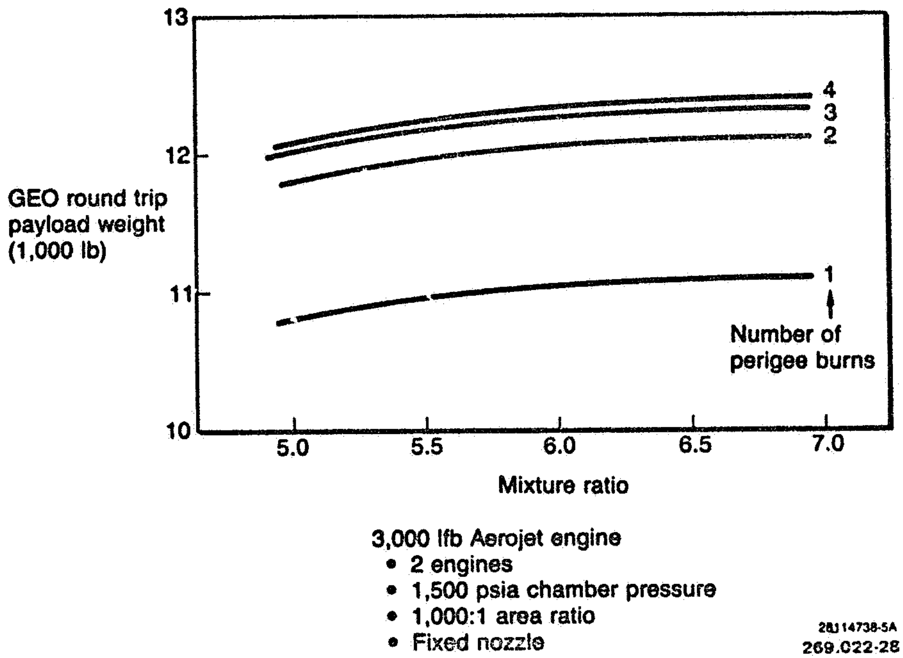

Figure 4-19. Mixture Ratio Trade

SECTION 5

ENGINE DESIGN

5.1 ENGINE LIFE

To define how the engine should be designed, an economic analysis was conducted for engine life and maintenance. While there is a benefit for designing a long life engine, there appears to be little advantage for reducing the frequency of major overhauls beyond 20 to 30 missions.

Figure 5-1 indicates the formula used to determine the benefit of a long-life engine and a description of the parameters used. Figure 5-2 is a sample case generated with these assumptions.

Benefit of long-life engine

- $[D_s + (N_s \times U_s) + N_r \times (T+R)] - [D_L + (N_L \times U_L) + N_f \times (C+T+R)]$

Where

D_s = Development cost of alternative (shorter-life) engine
N_s = Number of units of alternative engine required (over OTV life)
U_s = Unit cost of alternative engine
N_r = Number of short-life engine replacements required over OTV life
T = Cost of transporting one OTV engine to LEO
R = Cost of replacing an OTV engine
D_L = Development cost of long-life engine
N_L = Number of units of long-life engine required (over OTV life)
U_L = Unit cost of long-life engine
N_f = Number of engine refurbishments required (for maintenance of long-life engine over OTV life)
C = Cost of refurbishing a long-life engine

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Figure 5-1. Economic Benefit of Long-Life Engine

Short-life engine
- DDT&E cost = $245M
- Number units required = 22 (10 mission life, 200 missions total, 2 spares)
- Unit cost = $10M

Long-life engine
- DDT&E cost = $350M
- Number units required = 4 (100 mission life, 200 missions, 2 spares)
- Unit cost = $10.42M
- Number of refurbishments required = 6 (every 25 missions)
- Cost of refurbishment = $1M (per refurbishment)

Independent factors
- Cost of transporting one engine to LEO = $4M
- Cost of replacing an engine = $0.8M

Benefit of long-life engine = $124.92M

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Figure 5-2. Sample Case

The life-cycle economic benefit of a long-life (100 mission) OTV engine is closely related to the unit recurring production cost of the alternative short-life 10 mission OTV engine. The long-life engine yields a positive undiscounted benefit over the OTV mission span, two engines in use at all times, if the unit cost of the short-life engine exceeds about $3 million (Figure 5-3). This calculation is based on data provided by Aerojet (Appendix II, Tables I-1 and I-2) which indicate that the long-life engine has a $105 million greater non-recurring cost and a $0.42 million greater recurring cost than the short-life engine.

These data also include the assumption that the long-life engine must be returned to Earth for each major refurbishment every 25 missions. Establishing the capability to do these refurbishing tasks in space could save $4 million in transportation costs per overhaul and hence increase the benefit of the long-life engine by as much as $24 million over the values indicated or this graph. It is expected, however, that the added costs of utilizing the Space Station for engine refurbishment would exceed these transportation cost savings.

The nominal refurbishment rate assumed for the long-life OTV engine, one overhaul per 25 missions, is shown to be in the optimal range (Figure 5-4). Economic benefit of the long life engine drops sharply at higher overhaul rates to $100M at one overhaul per 16 missions, $50M at one overhaul per 9 missions, and to

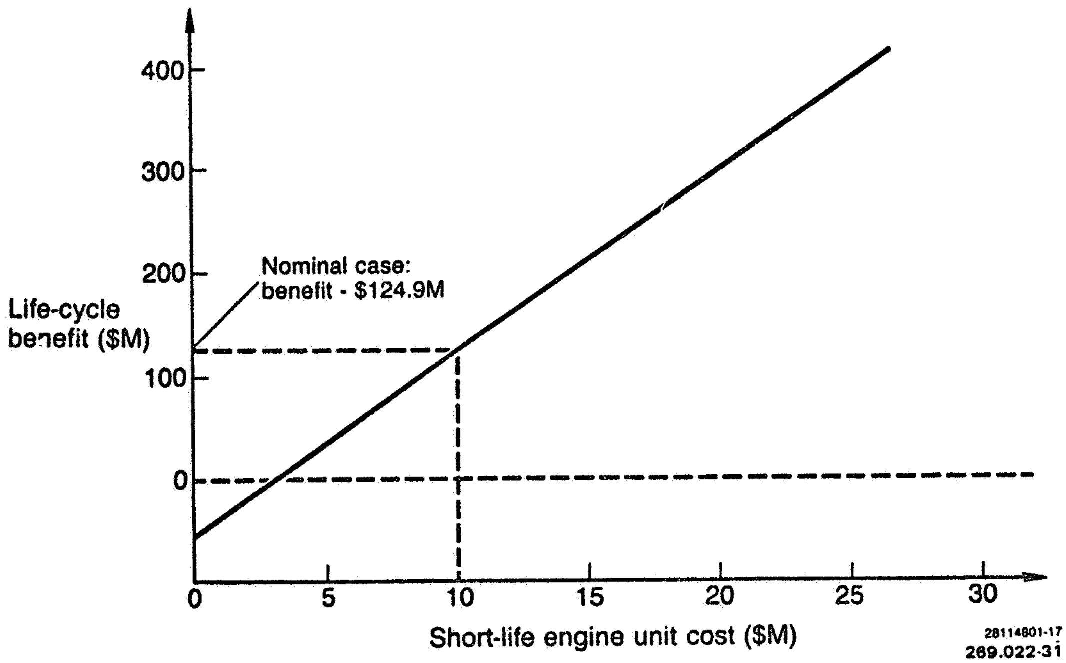

Figure 5-3. Life-Cycle Benefit of Long-Life OTV Engine Sensitivity To Engine Unit Cost

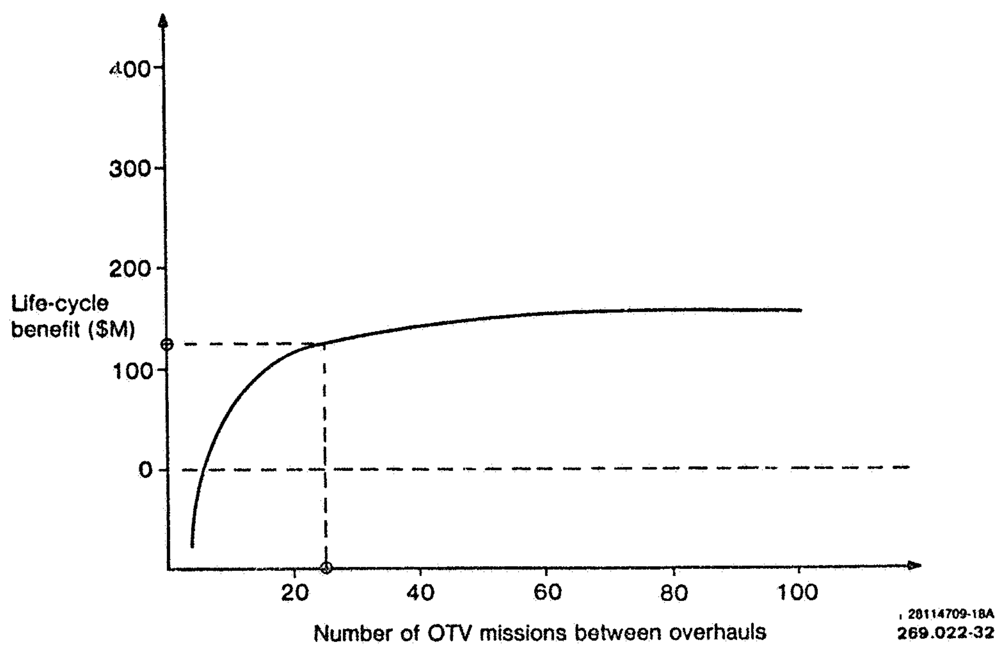

Figure 5-4. Life-Cycle Benefit of Long-Life OTV Engine Sensitivity To Engine Overhaul Frequency

zero at one overhaul per six missions. Benefits of reducing the refurbishment rate below the nominal rate are relatively modest. Doubling the number of missions between overhauls results in a $25 million increase in life cycle benefit; the benefit of further reductions in the refurbishment rate is barely noticeable.

5.2 ENGINE MAINTENANCE

Major overhaul of the engine for a space based OTV should be done on the ground to reduce cost (500 manhours at $100/hr on ground vs. 1000 manhours at $20000/hr in space), while routine maintenance is shown to be advantageous in space for anticipated task manhours (20 on ground, 40 in space).

The benefits and costs of space maintainability versus returning OTV engines to Earth for servicing were evaluated for two cases (Figure 5-5). "Routine maintenance" represents the most frequent and least complex type of servicing, assumed to nominally require 20 man-hours if performed on the ground and 40 hours if performed in space, with a frequency of one event every five missions. Establishment of Space Station facilities to support routine maintenance in space is assumed to cost $10 million more than establishment of similar facilities on Earth. Transportation costs involved in returning engines to Earth for routine maintenance are assumed to be $2.5 million per event.

	Routine Maintenance	Major Overhaul
Man-hours to perform task	Ground-20, Space-40	Ground-500, Space-1,000
Number of times performed (over 200 missions)	38 (every 5 missions)	6 (every 25 missions)
Transportation cost	$2.5M	$4M
Non-recurring cost Δ* for Space Station servicing equipment	$10M (space only)	$100M (space only)
Manpower cost	Ground-$100/hr Space-$20,000/hr	Ground-$100/hr Space-$20,000/hr

* Over cost of establishing same facilities on Earth

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Figure 5-5. OTV Engine Maintenance Baseline Assumptions

"Major overhaul" represents the opposite extreme in engine servicing with 500 hours required if performed on Earth and 1,000 hours if done in space. Frequency of major overhauls is assumed to be once every 25 missions, and development of Space Station facilities to support major overhauls is assumed to cost $100 million more than providing the same capabilities on the ground. Transportation costs are assumed to be $4 million per overhaul. Manpower costs for engine servicing are assumed to be $100/hour on the ground and $20,000/hour in space.

Plotting the costs of major engine overhauls as a function of the man-hours required to perform each overhaul clearly shows that performing major overhauls in space is very unlikely to be economical (Figure 5-6). Although amortization of nonrecurring costs of Space Station support facilities is a major cost factor, performing overhauls in space is over $15 million more expensive per overhaul even when Space Station facility costs are excluded. Performing overhauls in space is only economical if facility costs are excluded and manpower required is less than 200 man-hours to perform the overhaul in space.

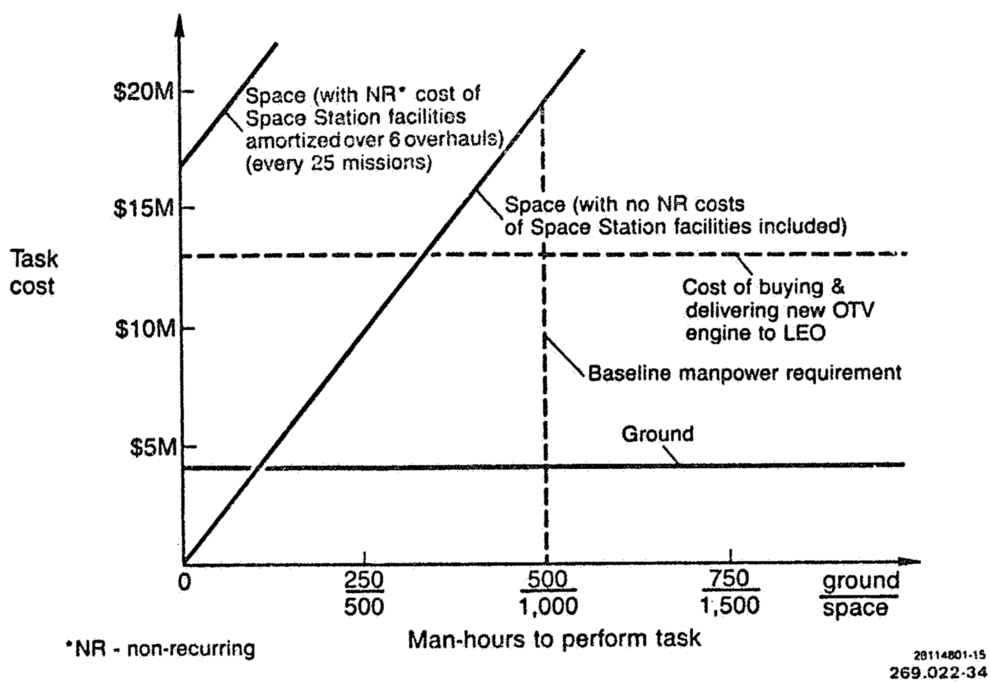

Figure 5-6. Task Costs For Major OTV Engine Overhauls

For routine OTV engine maintenance tasks, servicing in space is shown to be the most economical option (Figure 5-7). Space manpower requirements and costs are much lower than for performance of major overhauls, and Space Station facilities are less expensive and amortized over a greater number of servicing events. Returning the engine to Earth for routine maintenance is cost-effective only if the time required to service the engine in space exceeds 56 manhours. Amortization of Space Station facility costs over fewer events reduces the attractiveness of space servicing, but even at low maintenance frequencies (every 20 missions) the space servicing option is more economical as long as manhours required do not exceed the baseline value (20 hours) by more than 50 percent. In the baseline case, servicing in space is about $1.4 million less expensive per servicing event than ground servicing, or about $280,000 less expensive per OTV mission. The benefit of performing routine engine maintenance in space, calculated at $280,000 per OTV mission, could be considered partially or fully offset by payload weight penalties if any such penalties are incurred by designing the engine for space maintainability and if resultant payl [illegible] capacity reductions are considered to have an economic cost. With payload [illegible]t/lb to GEO (assumed to be $4,000 - $10,000) used as a measure, the weight penalty costs of the space-maintainable engine begin to exceed the benefits of space maintenance when the weight penalty reaches 28 to 70 pounds per mission, depending on the cost/lb to GEO used as a basis for calculation (Figure 5-8). Calculation of the costs associated with payload weight penalties are somewhat subjective, since even with very high manifesting efficiencies the OTV will probably have 500 or more pounds of excess (unused) capability on a typical geosynchronous mission. If, for example, only ten percent of all OTV missions were affected by a weight penalty in the range of consideration, then cost/lb to GEO might be multiplied by 0.10 before being used as a measure of weight penalty costs. With this methodology, weight penalties would need to be in the hundreds of pounds before their costs would approach the benefits of space maintainability.

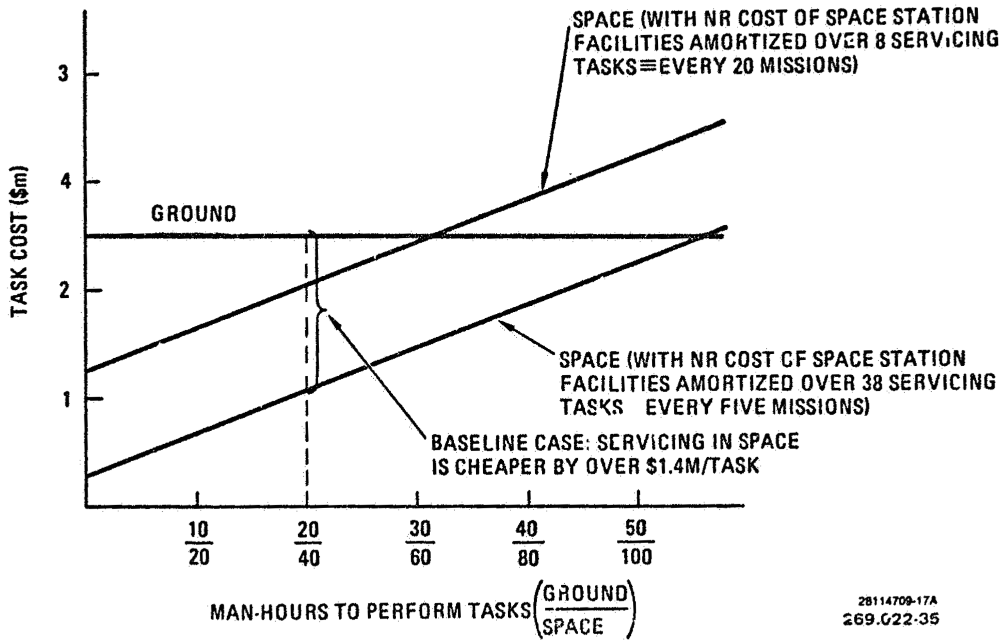

Figure 5-7. Task Costs For Routine OTV Engine Maintenance

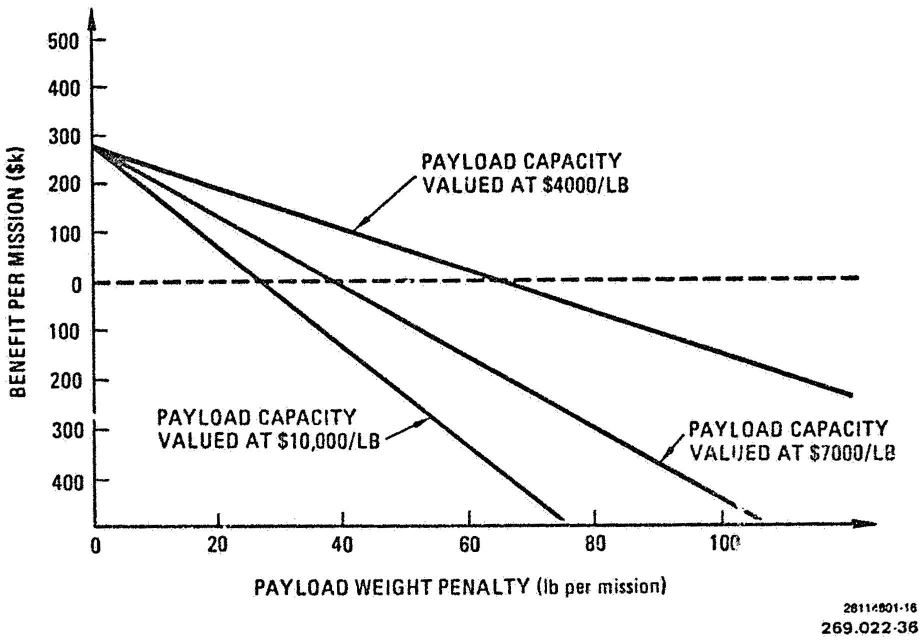

Figure 5-8. Benefit of OTV Engine Space-Maintainability (Routine Maintenance) As A Function of Payload Weight Penalty Incurred).

SECTION 6
CONCLUSIONS

This study has shown (Table 6-1) that future OTV engine requirements will be determined by LSS and manned missions. To satisfy the manned reliability requirement, twin engines are needed. The optimum design thrust level is in the range of 3500 to 6500 lb_f each, depending on the number of perigee burns for the manned mission. Although the lower thrust level is referable for less vehicle and payload design impact, this is contingent on the acceptance of multiple perigee burns for the manned mission and on the ability of the engine manufacturers to produce a high performance, reliable, and maintainable engine at lower thrust with additional starts and longer burn time. It is recommended that further research be planned.

Table 6-1. OTV Engine Study Findings

- Manned GEO mission & LSS mission impose conflicting requirements for one engine design
- Multiple perigee ascent burn trajectories offer optimal performance at low thrust levels needed for LSS missions
- Multiple perigee ascent burns (3-4) can be performed without increasing manned module shielding weight above that required for stay at GEO
- Optimum total thrust level for 13,000 lbm payload manned GEO mission is 6,000-7,000 lbf (3-4 perigee burns) vs. 13000 lb_f (1 burn)
- Optimum total thrust level for 10,000-20,000 lbm payload LSS mission is 1,000-2,000 lbf (8 perigee burns)
- Redundant engines are required for propulsion system reliability needed for mission success & crew safety, and to reduce tests
- High reliability engines will be demonstrated by testing & operational missions before manned requirement occurs
- Backup (O_2-H_2) APS (to a single main engine) results in additional propellant required due to decreased ISP. O_2-H_2 APS may have logistics advantages
- Recommended engine configuration is 2 main engines

Table 6-1. OTV Engine Study Findings, Contd

- Optimum engine design thrust level is 3,000-3,500 lbf each (3-4 perigee burns) vs 6,500 lbf each (1 perigee burn)
- Long-life (100 missions) engine recommended, but little economic benefit indicated for reducing frequency of major overhauls beyond one overhaul per 20-30 missions
- Major overhaul on ground & routine overhaul in space are recommended for Space-based OTV engine
- Further studies are recommended to evaluate multiple perigee burns for manned missions & to define high performance, reliable, maintainable engines at lower thrust levels (3,000-3,500 lbf)

APPENDIX I

ENGINE PARAMETRIC DATA

Aerojet

Pratt & Whitney

Rocketdyne

NOTE: The OTV engine parametric data produced by Rockwell International are considered proprietary information and are hence not included.

Table A-1. Aerojet OTV Dual Propellant Expander Cycle Engine

THRUST LEVEL (LBF.)	CHAMBER PRESSURE PC (PSIA)	AREA RATIO 1000							AREA RATIO 2000							AREA RATIO 3000						
		ISP	80% L	80% W	100% L	100% W	120% L	120% W	ISP	80% L	80% W	100% L	100% W	120% L	120% W	ISP	80% L	80% W	100% L	100% W	120% L	120% W
			PERCENT BELL NOZZLE							PERCENT BELL NOZZLE							PERCENT BELL NOZZLE					
3000	1500	482.4	68	112	81	118	93	124	484.9	89	117	107	124	125	132	486.6	106	122	127	131	149	138
	2000	482.4	61	103	71	108	82	112	484.9	79	107	94	113	110	118	486.6	93	111	112	118	131	124
	2500	482.5	56	99	65	102	75	106	485.0	72	102	86	106	99	110	486.7	85	105	101	110	118	115
4000	1500	482.6	76	135	90	143	104	151	485.1	100	142	121	152	141	162	486.8	119	149	144	160	169	172
	2000	482.6	67	124	80	130	92	136	485.1	89	129	106	136	124	144	486.9	105	134	127	143	148	152
	2500	482.7	61	118	73	123	84	128	485.2	80	122	96	128	112	134	487.0	95	126	114	133	134	140
5000	1500	482.8	83	158	93	168	115	178	485.3	110	167	133	179	156	191	487.1	131	175	159	190	187	204
	2000	482.8	73	144	87	152	101	159	485.3	97	151	117	160	137	169	487.1	115	157	139	168	164	179
	2500	482.9	67	137	79	143	92	149	485.4	88	142	106	149	123	157	487.2	104	147	126	156	148	164

NOTES:

1. ISP IS VACUUM DELIVERED AT 120% BELL (SECONDS).
2. MR = 6.0:1 LO_2/LH_2.
3. L = ENGINE LENGTH (IN.).
4. W = ENGINE WEIGHT (LBM.).
5. NO ENTHALPY PUMPING.

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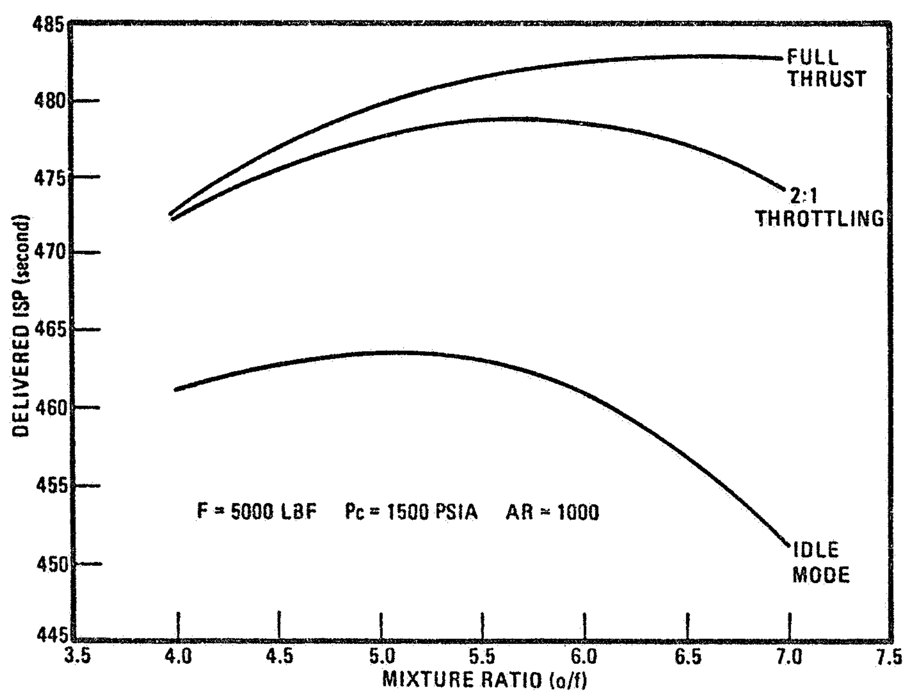

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Figure A-1. Aerojet OTV Dual Expander Cycle Engine Delivered ISP Vs. Mixture Ratio

Table A-2. Pratt & Whitney 3K Thrust Engine

CHAMBER PRESSURE (PSI)	AREA RATIO	DELIVERED SPECIFIC IMPULSE (SEC.)	ENGINE WEIGHT (LB.)
1200	600	473.1	201
1200	800	472.7	208
1200	1000	472.0	216
1200	1500	468.4	236
1200	2000	463.1	255
1200	3000	450.9	294

Table A-3. Pratt & Whitney 5K Thrust Engine

1200	600	473.6	252
1500	600	475.8	243
1200	800	473.2	264
1500	800	476.0	253
1200	1000	472.5	276
1500	1000	475.6	263
1200	1500	468.9	306
1500	1500	473.3	288
1200	2000	463.7	337
1500	2000	469.4	312
1200	3000	451.5	399
1500	3000	460.1	362

Table A-4. Pratt & Whitney 10K Thrust Engine

CHAMBER PRESSURE (PSI)	AREA RATIO	DELIVERED SPECIFIC IMPULSE (SEC.)	ENGINE WEIGHT (LB.)
1200	600	474.2	345
1500	600	476.3	331
1200	800	473.7	368
1500	800	476.4	351
1200	1000	472.9	394
1500	1000	476.2	370
1200	1500	469.3	454
1500	1500	474.0	419
1200	2000	464.4	515
1500	2000	470.0	466
1200	3000	452.2	636
1500	3000	460.7	564

Table A-5. Pratt & Whitney 15K Thrust Engine

CHAMBER PRESSURE (PSI)	AREA RATIO	DELIVERED SPECIFIC IMPULSE (SEC.)	ENGINE WEIGHT (LB.)
1200	600	474.5	419
1500	600	476.5	398
2000	600	478.6	376
1200	800	474.2	454
1500	800	476.7	428
2000	800	479.2	399
1200	1000	473.5	491
1500	1000	476.5	457
2000	1000	479.5	421
1200	1500	469.8	582
1500	1500	474.0	530
2000	1500	478.2	476
1200	2000	464.8	672
1500	2000	470.3	603
2000	2000	476.0	531
1200	3000	452.6	853
1500	3000	461.0	748
2000	3000	469.4	642

APPENDIX II

ENGINE COST DATA

(Aerojet)

Table I-1. Cost Trade: Short vs Long Mission Life

	Cost Delta			
	Short Life (10 Missions)		Long Life (100 Missions)	
Engine	Non-Recurring	Recurring*	Non-Recurring	Recurring
Use of Hydrostatic Bearings				
Development and Qual			5 M	
Additional Fab Complexity, QC, Test				20 K
Health Monitoring - Failure Prevention				
Development and Qual			20 M	
Fab, Instrumentation, Computer				200 K
Health Monitoring - Life Projection				
Development and Qual			80 M	
Fab, Instrumentation, Computer				200 K
Space-Replaceable Engine				
Development and Qual	40 M			
Mission				
Space-Replaceable Engine				
Design of Service Center	100 M			
Assembly of Service Center	250 M			
Space Operations (Engine Replacement)		800 K		
Transfer of Engine from Earth to LEO & Return*		400 K		

*Per Engine

NOTE: Long life is baseline Aerojet design

Table I-2. Cost Trade: Low vs High Specific Impulse (Isp)

Engine	Cost Delta: Low Isp Non-Recurring	Low Isp Recurring*	High Isp Non-Recurring	High Isp Recurring*
Dual Propellant Expander Cycle				
Development and Qual			15 M	
Fab, QC				200 K
Extendible Nozzle				
Development and Qual			10 M	
Fab, QC				50 K
or				
Large Area Ratio Nozzle				
Development and Qual			2 M	
Fab, QC				25 K
Injector Iteration During Development			1 M	
Mission				
Delivery of Additional Propellant (12/480 - 2.5%)	600 M			

*Per Engine

NOTE: Aerojet 3000 lbF engine design with 1200:1 nozzle has projected Isp of 484 lbF sec/lbM and near maximum Isp. Additional gains of 12 lbF sec/lbM are doubtful. Therefore high Isp case is baseline.

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16. Abstract

NASA-LeRC is sponsoring industry studies to establish the technology base for an advanced engine for orbital transfer vehicles for mid-1990s IOC. Engine contractors are being assisted by vehicle contractors to define the requirements, interface conditions, and operational design criteria for new LO_2-LH_2 propulsion systems applicable to future orbit transfer vehicles and to assess the impacts of space basing, man rating, and low-G transfer missions on propulsion system design requirements.

This report presents the results of a study conducted by GDC under contract to Aerojet for NASA-LeRC. The primary study emphasis was to determine what the OTV engine thrust level should be, how many engines are required on the OTV, and how the OTV engine should be designed. This was accomplished by evaluating planned OTV missions and concepts to determine the requirements for the OTV propulsion system, conducting tradeoffs and comparisons to optimize OTV capability, and evaluating reliability and maintenance to determine the recommended OTV engine design for future development.

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