ENGINE SYSTEM ASSESSMENT STUDY USING MARTIAN PROPELLANTS

FINAL REPORT

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FOREWORD

This report was prepared by Science Applications International Corporation (SAIC) in Torrance, California, and contains the results of a study performed for the National Aeronautics and Space Administration (NASA) Lewis Research Center, Space Propulsion Technology Division, as part of contract NAS3-25809, "Manned Lunar and Mars Mission Propulsion System Assessment Studies."

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Other contributors who provided guidance to the study also included Ms. Diane Linne of NASA Lewis Research Center and Mr. Michael Stancati of SAIC. Dr. Robert Mog of SAIC also provided technical assistance associated with the tank meteoroid protection system analysis.

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NOMENCLATURE

%	Percent
%Noz	Nozzle Percent Length
Al	Aluminum
AR	Nozzle Area Ratio
CH_4	Methane
cm	centimeter
cm^3	cubic centimeter
Co	Isentropic Spouting Velocity
CO	Carbon Monoxide
CO_2	Carbon Dioxide
const	constant
E or ε	Nozzle Area Ratio
ELES	Expanded Liquid Engine Simulation
ELM	Earth Launch Mass
EOI	Earth Orbit Insertion
ETO	Earth To Orbit
F	Degree Fahrenheit or Thrust
F_1	Fuel No. 1
F_2	Fuel No. 2
g	gram or gravitational acceleration
GG	Gas Generator
H_2	Hydrogen
hab	habitat
HC	Head Coefficient
in	inches
Inj Dens	Injector Density
Inj Type	Injector Type
Isp	Specific Impulse
ISPP	In Situ Propellant Production
K	Degree Kelvin
kg	kilogram
lbf	pound force
lbm	pound mass
LEO	Low Earth Orbit

LeRC	Lewis Research Center
LEV	Lunar Excursion Vehicle
LH_2	Liquid Hydrogen
Li	Lithium
LLO	Low Lunar Orbit
LMO	Low Mars Orbit
LOI	Lunar Orbit Insertion
LOX or LO_2	Liquid Oxygen
m	meter
m^2	square meter
mm	millimeter
MEV	Mars Excursion Vehicle
MLI	Multilayer Insulation
MOI	Mars Orbit Insertion
MR	Mixture Ratio
MSFC	Marshall Space Flight Center
msn	mission
MTV	Mars Transfer Vehicle
N	Newton
NASA	National Aeronautics and Space Administration
NASP	National AeroSpace Plane
OTV	Orbit Transfer Vehicle
Pc	Chamber Pressure
P_{RF}	Probability of No Penetration
PSDOC	Protective Structures Design Optimization Code
psi	pounds force per square inch
psia	pounds force per square inch absolute
R	Degree Rankine
regen	regenerative
RPM	Revolutions Per Minute
s	second
SAIC	Science Applications International Corporation
Si	Silicon
SOA	State-of-the-Art
SS	Steady-State or Pump Specific Speed
SSME	Space Shuttle Main Engine
SSTO	Single Stage to Orbit

STBE	Space Transportation Booster Engine
STME	Space Transportation Main Engine
t	metric tonnes
Tc	Chamber Temperature
TEI	Trans-Earth Injection
TLI	Trans-Lunar Injection
TMI	Trans-Mars Injection
TPA	Turbopump Assembly
ΔV	Change in Velocity
vac.	vacuum
Wgt	Weight

1.0 INTRODUCTION

Recent studies have shown that there can be substantial advantages in using in situ propellants for fast transfers to, and explorations of, Mars when compared to chemical systems that use Earth-based propellants, see Refs. 1-1 through 1-4. Using vehicles that have propulsion systems that use Martian resources has the potential to greatly reduce Low-Earth-Orbit (LEO) mass requirements as well as potentially increase mobility on the surface of Mars. A single propulsion system that can use two or more candidate propellant combinations, such as LOX/LH_2, LOX/CH_4 and LOX/CO, could best leverage this exploration option. Design of such a propulsion system is challenging due to its requirements that it be inherently compatible with numerous candidate propellants and their by-products, as well as operate efficiently over a large range of conditions.

The objective of this top-level feasibility study was to identify and characterize promising chemical propulsion system designs that use two or more of the following propellant combinations: LOX/LH_2, LOX/CH_4 and LOX/CO. Key results from this study were: 1) identifying the propellant combinations that are best suited for a single multipropellant engine system design, 2) identifying and characterizing promising engine cycles and concepts, 3) determining and characterizing the impact of mission performance on using multipropellant combinations in a given engine design, and 4) identifying and prioritizing enabling and enhancing technologies required to support successful development of such an engine system. The results from this study identify the major engine design and overall mission impact issues associated with the development and use of such engine systems.

The overall study approach integrated both mission and engine system design analyses to address engine system design and performance issues and to determine the impact of such systems on missions performed and In Situ Propellant Production (ISPP) requirements. Based on a recent ISPP study, Ref. 1-4, promising mission scenarios were defined and characterized. Top-level engine system requirements were then identified from these results. In parallel with this effort, a literature review was conducted that addressed key in situ engine system technology areas. These results, then, form the basis for the identification and design assessment of the promising engine system concepts that meet a majority of the mission requirements. These tripropellant, LOX-cooled engine systems for Mars transfer applications, as well appropriate bipropellant design derivatives for lunar and Mars excursion applications, which included both expander and gas generator engine cycle versions of each system, were baselined for the study and examined in detail. Propellant tankage system design considerations and concepts were also examined in a top-level manner for the propulsion systems of interest.

At the conclusion of the study, the initial study mission analysis results were updated for a select number of promising mission scenarios based on the detailed baseline engine system data mentioned previously. For these mission scenarios and engine systems of interest, in addition to characterizing mission performance for a given scenario flight profile, top-level sensitivities of engine system mass, specific-impulse and transfer vehicle propellant staging approach, and their impact on ISPP system requirements are also examined. Additionally, a technology maturation plan was defined that addresses engine system design/ technology issues required to support development of such engine propulsion systems.

Detailed discussions of the study's approach, considerations, assumptions, results, and recommendations are presented in the following sections.

2.0 INITIAL ENGINE SYSTEM REQUIREMENTS

Mission performance was assessed initially to obtain requirements for a space propulsion system that utilizes propellants produced at the Moon and/or Mars for support of manned Mars exploration. These initial requirements provide a starting point for in situ engine design efforts using lunar and/or Mars propellants. Lunar in situ propellants, produced from lunar regolith, are used to fuel the Mars Transfer Vehicle (MTV) for the outbound portion of the Mars mission. Mars in situ propellants, produced from the Martian atmosphere, are used to fuel the MTV for the return leg of the trip.

A major design objective of any space mission is to reduce Earth Launch Mass (ELM) as much as possible without compromising mission objectives. To perform a round-trip, piloted, opposition-class Mars mission (which departs from LEO), the vehicle travels to Mars with a crew and mission payload, and returns to LEO) with conventional LOX/H_2 chemical propulsion requires a vehicle initial mass in LEO of about 1600 metric tonnes (t). This translates into a large amount of mass to be launched from the Earth to LEO for assembly. One option for reducing ELM for a piloted Mars mission that has been proposed in recent studies, see Ref. 1-1, is the use of aerocapture at Mars arrival and at Earth return. This significantly reduces the mission propellant requirements, but the total initial vehicle mass for such a mission is still on the order of 800 t, see Ref. 2-1. Another option for reducing ELM is to set up ISPP plants on extraterrestrial bodies to fuel an MTV in space. This reduces the amount of mission propellant that has to be launched from Earth. While initial plant development, set-up, and supporting infrastructure costs may be high, over the long term, launching some of the MTV propellant from the surface of the Moon up to low lunar orbit (LLO) or from the surface of Mars up to low Mars orbit (LMO) to fuel the MTV might be less costly than launching all of the fuel from the surface of Earth up to LEO at the start of each mission.

This section describes the major assumptions made in determining ISPP requirements and the methodology used for evaluating mission performance. Initial mission performance results are then used to derive top-level engine requirements to serve as a starting point in the design of a space propulsion system that can use multiple in situ propellant combinations.

2.1 In Situ Propellant Candidates and Production Requirements

Many studies have been performed to assess potential benefits of utilizing in situ propellants. In these studies, the ISPP requirements were based on a single processing approach.

The approach used for this study was developed to assess the utility of various in situ propellant combinations and did not attempt to identify an optimal propellant processing scheme. In a previous study, see Ref. 1-4, many processing techniques were reviewed, and ISPP requirement ranges were parametrically characterized to approximate the requirements to obtain a given propellant combination and to encompass the range of requirements presented in the ISPP literature. Promising propellant combinations considered for this study included LOX/H_2, LOX/CH_4, and LOX/CO. Other propellant candidates, such as metallized monopropellants, were not considered because of lack of commonality with bipropellant systems. Although CH_4 and CO can be obtained from the Moon through extraction of solar wind gases, lunar LOX/CO was not considered because of excessive processing requirements to obtain the needed quantities to support a LOX/CO propulsion system. LOX/CO and LOX/CH_4 were chosen as Mars propellant candidates because they are readily available from the Martian atmosphere. Lunar LOX and lunar LOX/CH_4 were chosen as the lunar candidates because they are more compatible with the Mars candidates than are other possible lunar-produced propellants (e.g., metallized monopropellants like LOX/Si or LOX/Al). Earth LOX/H_2 is used for the outbound leg of mission scenarios not utilizing lunar propellant and for boosting the MTV from LEO to LLO for scenarios using lunar propellant. All the candidates are compatible in that they are all used in cryogenic chemical bipropellants with LOX as the oxidizer.

2.2 Mission Description

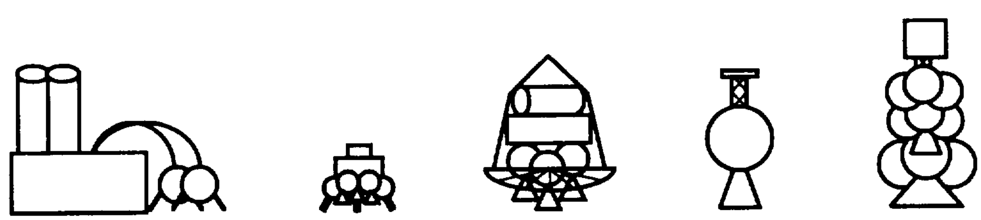
As previously mentioned, the purpose of this assessment is to investigate the application of various in situ lunar and Mars propellants for fueling an MTV that transports crew and payload to Mars to perform a 30-day surface mission and then returns the crew to Earth. Three different propellant combinations (LOX/CO, LOX/CH_4, and LOX/H_2) and three engine types were considered for analysis in different piloted Mars mission scenarios in which some or all of these propellants would be produced and used in situ at the Moon and/or Mars. One proposed engine design burns both LOX/H_2 and LOX/CO. Another design burns both LOX/H_2 and LOX/CH_4. The third one burns both LOX/CO and LOX/CH_4. Seven different scenarios were initially considered, as shown in Table 2-1. Some of the scenarios use both lunar and Mars propellant, and some use only Mars propellant.

The basic infrastructure elements in each scenario are the lunar/Mars propellant production plants, the MTV, the Lunar Excursion Vehicle (LEV), the Mars Excursion Vehicle (MEV), and an expendable booster stage and are schematically shown in Figure 2-1. The LEV and MEV are reusable lunar and Mars-based vehicles that transfer crew, mission payload, ISPP support, and in situ derived propellants between the MTV and the lunar or Mars surface. The expendable booster stage uses high performance LOX/H_2 propulsion and is responsible for transporting the MTV to LLO from LEO in scenarios using lunar-produced propellants. This stage is jettisoned after completing this transfer. The MTV carries the crew, Mars mission payload, and ISPP support to Mars and returns the crew to Earth.

Table 2-1. Initial Mission Performance Assessment Scenarios

Scenario	**Outbound Propellant**	**Return Propellant**	**Mission Profile No.**
1	Earth LOX/H_2	Earth LOX/H_2	Baseline
2	Lunar LOX/Earth H_2 *	Mars LOX/CO	1
3	Lunar LOX/Earth H_2 *	Mars LOX/CH_4	1
4	Lunar LOX/CH_4 *	Mars LOX/CO	1
5	Lunar LOX/CH_4 *	Mars LOX/CH_4	1
6	Earth LOX/H_2	Mars LOX/CO	2
7	Earth LOX/H_2	Mars LOX/CH_4	2

* Earth LOX/H_2 used for trans-lunar injection and lunar orbit insertion



Lunar and Mars ISPP Plants (includes all systems necessary for feedstock collection through propellant storage)

Lunar Excursion Vehicle

Mars Excursion Vehicle

LOX/H_2 Expendable LEO->LLO Stage

Mars Transfer Vehicle

Figure 2-1. Infrastructure Elements

The mission profiles examined are shown in Figure 2-2. The baseline scenario, which uses only Earth supplied LOX/H_2, is used as a point of comparison to evaluate ISPP scenarios. Mission Profile #1 was used for scenarios using both lunar and Mars-produced propellants. Mission Profile #2 was used for scenarios that used Earth-supplied propellant for the outbound leg and Mars-produced propellants for the return trip.

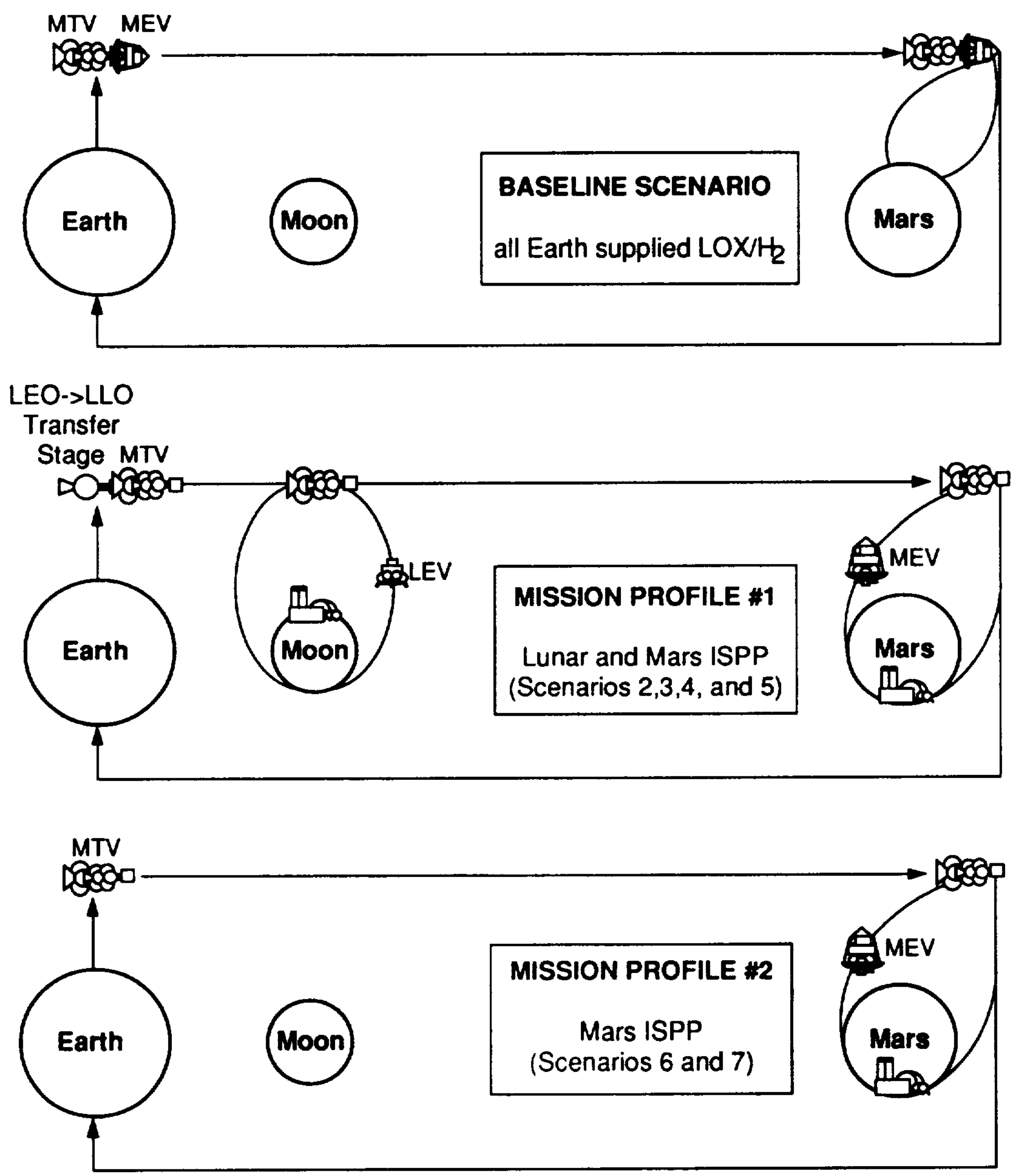


Figure 2-2. Mission Profiles

Scenarios 2-5, where both lunar and Mars propellant are utilized, are described in Figure 2-3. In these scenarios, a plant is set up on the Moon to produce the propellant needed to send the MTV from the Moon to Mars, and the propellant needed by the LEV to transport this MTV propellant up to the MTV in LLO and to carry lunar ISPP plant support to the lunar surface. Additionally, a plant is set up on Mars to produce the propellant needed to send the MTV from Mars back to Earth, and the propellant needed for the MEV to carry the crew, Mars mission payload, and Mars ISPP plant support to the Mars surface. The propellant produced on Mars is also used by the MEV to transport the MTV return trip propellant up to the MTV in LMO. The MTV is brought out to the Moon on an expendable stage, which performs both Earth orbit departure and lunar orbit insertion and then separates from the MTV and is left in LLO. The MTV is fueled up in LLO by the LEV with lunar-produced propellant to make the trip to Mars. At Mars, after the crew performs its surface mission, the MTV is fueled up in LMO by the MEV with Mars-produced propellant for the return trip back to Earth.

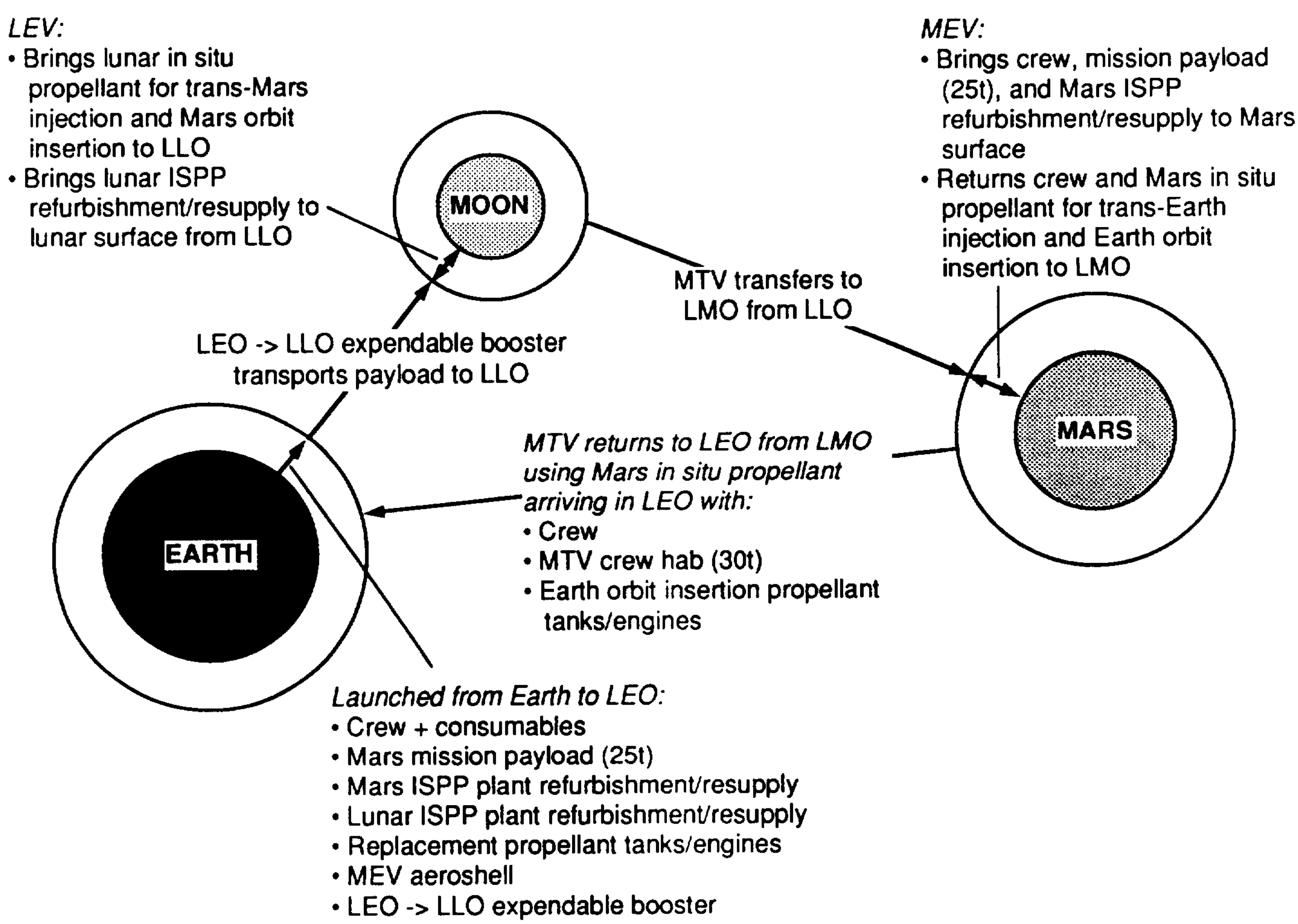


Figure 2-3. Mission Description for Scenarios Using Lunar and Mars ISPP (Mission Profile #1)

Scenarios 6 and 7, where only in situ Mars propellant is used, are described in Figure 2-4. In these scenarios, there is no lunar plant or LEV, and the MTV does not stop at the moon at all. It is injected from Earth orbit onto a Mars transfer trajectory by the expendable booster stage, which is jettisoned upon completion of the Earth departure burn. Several months later, the MTV captures into a Mars orbit, and the crew performs its mission after landing on the Mars surface. After the mission is complete, the MTV is fueled up by the MEV with Mars-produced propellant for the trip back to Earth.

As previously mentioned, Scenario 1 is an all propulsive, all Earth-supplied LOX/H_2 propellant baseline case against which all the other results should be compared. In Scenario 1, no in situ propellants are used and there are no lunar or Mars ISPP plants. All of the propellant utilized by the transfer and excursion vehicles is Earth-supplied LOX/H_2. This case differs from the 90-Day Study chemical propulsion/aerocapture baseline case (see Reference 1-1) in that aerobraking is not employed at Earth or Mars; all maneuvers are performed propulsively.

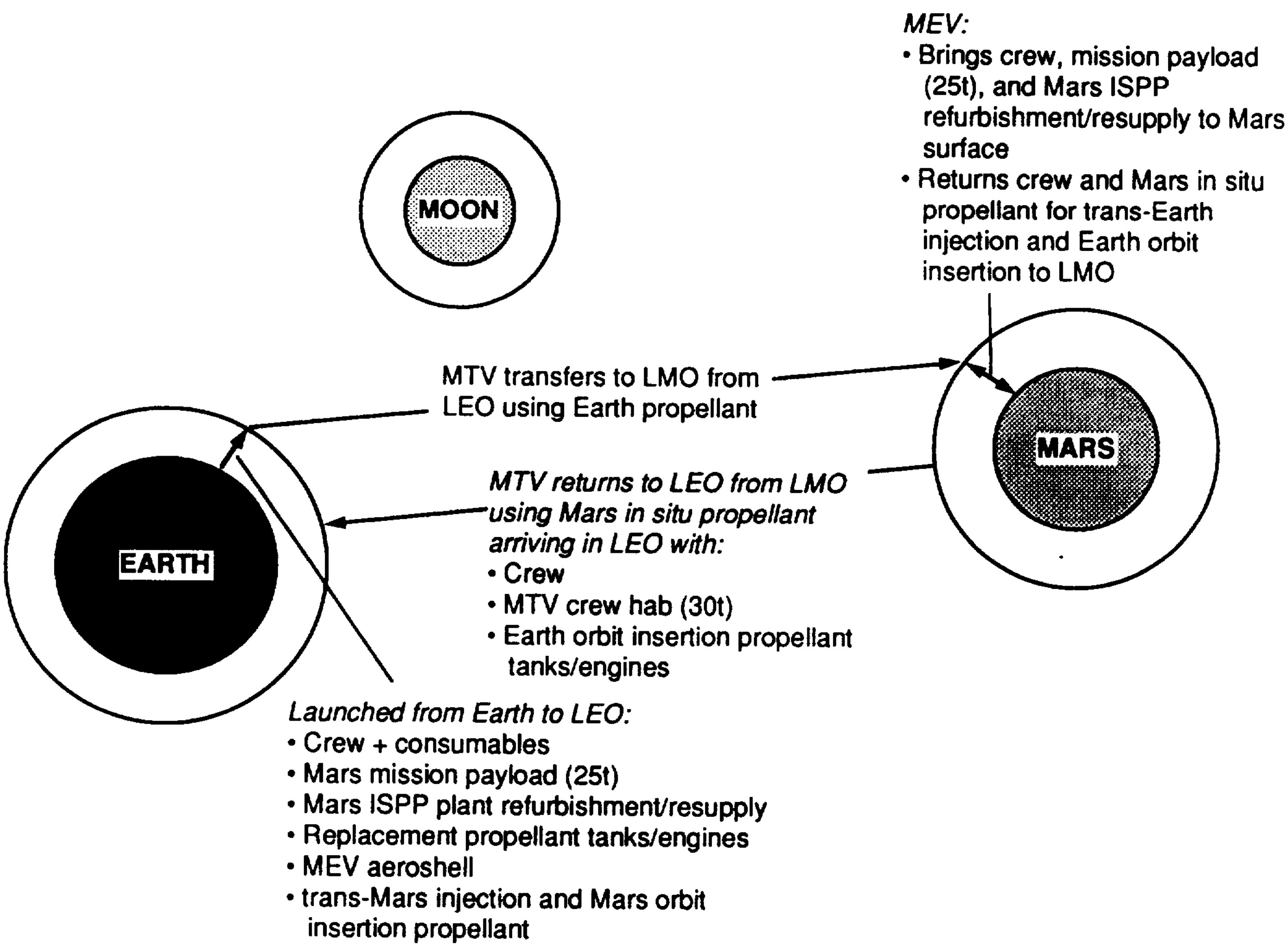


Figure 2-4. Mission Description for Scenarios Using Mars ISPP (Mission Profile #2)

In Scenario 2, a LOX plant is set up on the Moon and a LOX/CO plant is set up on Mars. For this case, an expendable booster using Earth-supplied LOX/H_2 carries the MTV from LEO to LLO. In LLO, the MTV is fueled by a LEV with lunar-produced LOX, which is used with Earth-supplied H_2 to transport the MTV from LLO to LMO. The sole purpose of the LEV is to carry propellant up to the MTV in LLO and bring lunar plant resupply materials back down to the lunar surface. At Mars, the MEV meets the MTV in LMO so that the crew and mission payload can be transferred to the MEV. The MEV then descends to the surface of Mars where it fills up its tanks with propellant for the MTV, while the crew performs their surface mission. When the excursion is complete, the crew return aboard the MEV to LMO, and transfer back into the MTV. The MEV also transfers Mars-produced LOX/CO to the MTV for the return trip to Earth.

Scenario 3 is the same as Scenario 2 except that LOX/CH_4, not LOX/CO, is produced at Mars. In Scenario 4, LOX/CH_4 is produced at the Moon and LOX/CO is produced at Mars. For this scenario, no Earth-produced H_2 is needed for the LLO to LMO leg of the mission. Scenario 5 employs both lunar LOX/CH_4 and Mars LOX/CH_4.

Scenarios 6 and 7 are simpler than Scenarios 2-5 in that no lunar-produced propellant is used. The MTV goes directly from LEO to LMO and back to LEO, using Earth-produced LOX/H_2 for the outbound trip and Mars-produced propellant for the return trip. In scenario 6, Mars LOX/CO is used for the return, while in Scenario 7, Mars LOX/CH_4 is used.

2.3 Mission Performance

Each mission scenario of interest was characterized using SAIC's ISPP Mission Performance Model to determine ΔVs, propellant requirements, vehicle sizes and masses, and flight times for each phase of a given flight profile. From this information, overall propulsion system requirements were derived for each mission scenario.

The methodology used in the mission performance model is depicted in Figure 2-5. This figure shows the steps used to determine steady-state mission requirements. The steady-state requirements assume all ISPP plants to be operational and other associated infrastructure to be established. First, the amount of in situ propellant required to return the MTV to LEO from LMO is determined. This propellant, along with the propellant needed by the MEV to carry the crew, Mars mission payload, and Mars ISPP plant support to the Mars surface from LMO and to carry the MTV's return propellant to LMO from the Mars surface, determine the production rate

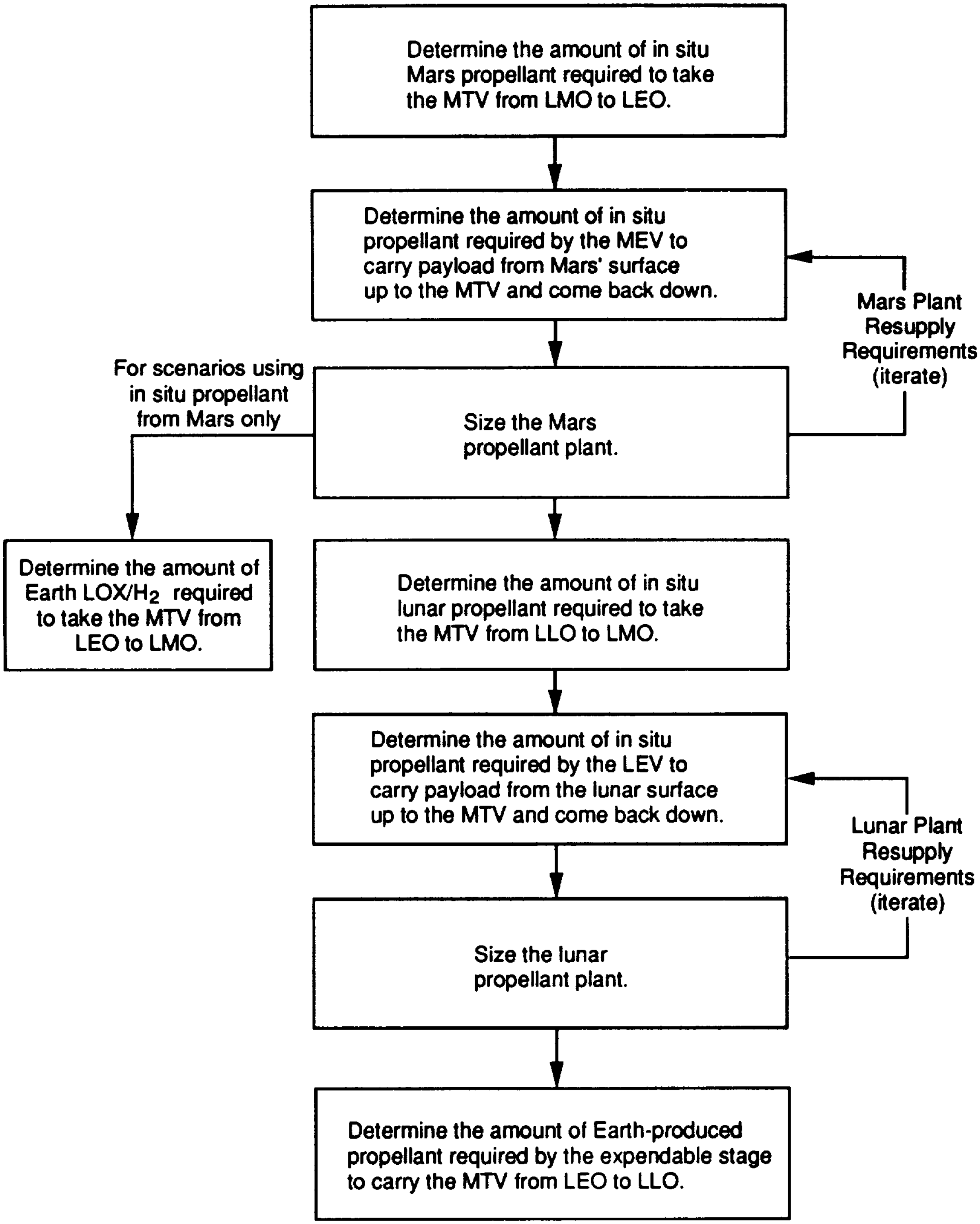


Figure 2-5. Mission Performance Prediction Methodology

and size of the Mars ISPP plant. An iteration is required to estimate the MEV's propellant requirements because each time the MEV's propellant requirement is determined, the size and support requirements for the Mars ISPP plant change, and, therefore, the MEV's payload requirements change. When the iteration is complete, the mass needed in LMO to support a mission is known. If the mission does not use lunar propellants, the MTV is sized to carry this mass from LEO using Earth-supplied propellant. If the mission uses lunar propellant, the same approach used to determine mass needed in LMO is used to estimate the mass needed in LLO to support a mission. An expendable stage is then sized to deliver this mass from LEO to LLO. When these steps are completed, the ELM requirements to support a mission in the steady-state mode are obtained. Also, the masses of the lunar and/or Mars ISPP plants and excursion vehicles and the MTV are determined. The masses of the ISPP plants are representative of the set-up requirements to enable utilization of in situ propellants in a given scenario. The excursion and transfer vehicle masses are representative of the requirements for vehicle change-out or replacement after these vehicles have reached the end of their life cycle. More details on this approach can be found in Ref. 1-4.

Initially, all these scenarios were evaluated using the simple engine mass scaling relations shown in Table 2-2 and the mission performance/vehicle design assumptions presented in Table 2-3. This analysis approach enabled estimation of the thrust requirements for each propulsive maneuver for each of the vehicles in the infrastructure-booster stage, MTV, MEV, and LEV.

Table 2-2. Initial Engine Parameters

Propellant Combination	Specific Impulse - Vacuum, sec.	Thrust/Weight, N/kg (lbf/lbm)	Mixture Ratio (O/F)
LOX/H2	470	765 (78)	6.0
LOX/CH4	380	883 (90)	3.6
LOX/CO	290	961 (98)	0.6

Table 2-3. Mission Performance/Vehicle Design Assumptions

- All maneuvers are done propulsively (no aerobraking)
- Mission ΔVs and flight times are averaged from 6 opposition class opportunities (2015-2030 timeframe):

Scenarios 1, 6, 7 (LEO->LMO->LEO)	Scenarios 2-5 (LEO->LLO->LMO->LEO)
ΔV TMI = 3982 m/sec	ΔV TLI = 3300 m/sec
ΔV MOI = 2590 m/sec	ΔV LOI = 1110 m/sec
ΔV TEI = 2521 m/sec	ΔV TMI = 2005 m/sec
ΔV EOI = 4081 m/sec	ΔV MOI = 2590 m/sec
	ΔV TEI = 2521 m/sec
ΔT Earth->Mars = 250 days	ΔV EOI = 4081 m/sec
ΔT Mars stay = 30 days	
ΔT Mars->Earth = 273 days	ΔT Earth->Moon = 3.5 days
	ΔT Moon stay = 3 days
	ΔT Moon->Mars = 250 days
	ΔT Mars stay = 30 days
	ΔT Mars->Earth = 273 days

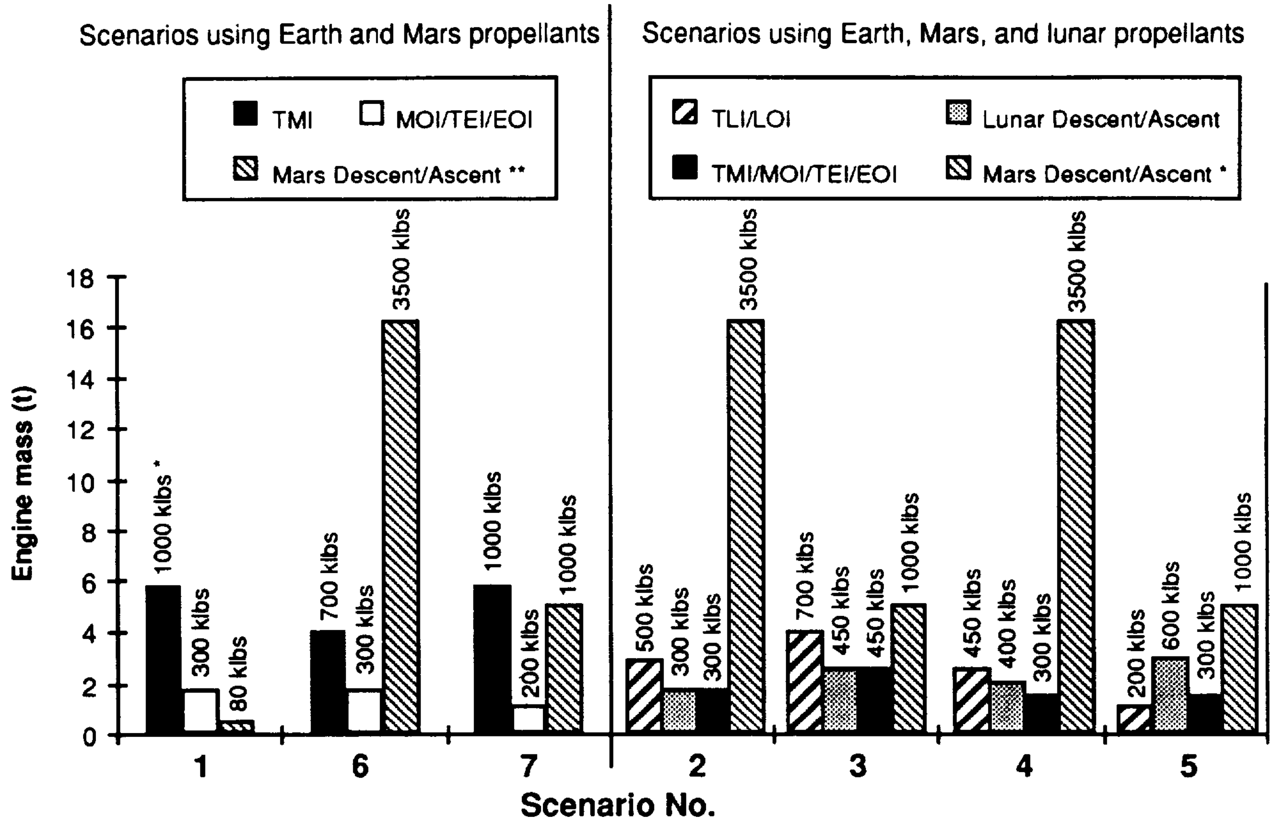
- Earth departure/arrival orbit is 407 km circular
- Mars parking orbit is 250 km x 1 sol
- 4 crew members assumed with consumable rate of 93 kg per person per month
- MTV crew habitation module = 30 t
- 2 MEVs operate simultaneously to bring crew, mission payload (25t), and ISPP refurbishment/resupply down to Mars surface and return crew and Mars in situ propellant for TEI + EOI back to the MTV in LMO
- Vehicle structure mass = 15% of propellant tank dry mass
- Reserve propellant = 2.5% of propellant required
- Propellant tanks are jettisoned after each major burn except for EOI tanks (reused as part of MTV core)
- Empty propellant tanks are brought on the MTV to be filled up at the Moon and also at Mars
- Propellant tank mass = *X*% of propellant mass in the tank (assumes 2% tank ullage):

Cryogen	*X%*
H_2	12
O_2	2
CO	2
CH_4	2

2.4 Engine System Requirements

Detailed mission performance and requirements data for each scenario is given in Appendix A. In Appendix A, for each mission scenario considered, tables summarizing the mission features and assumptions, performance for each mission phase, and overall engine system requirements are given. Figure 2-6 summarizes engine thrust and mass requirements for each burn in each scenario, while Tables 2-4 through 2-10 display the overall propulsion system requirements for Scenarios 1 through 7, respectively.

It should be noted that these initial mission performance predictions are based on rough engine mass scaling relations from which initial overall propulsion system estimates were derived (e.g., thrust requirements and engine burn times). These initial estimates served as inputs to the engine system design effort. This analysis was updated in Section 5 using more accurate engine system data based on detailed engine design analysis to obtain more accurate mass performance results. Scenario 5 was included as a point of comparison to the other alternatives because it was one of the better scenarios in terms of mission performance, see Ref. 1-4. This scenario was not considered for further analysis here because it does not utilize two different propellant combinations for the MTV engine.



Propellant Source		1	6	7	2	3	4	5
Out-bound	Earth*** ->	LOX/H_2	LOX/H_2	LOX/H_2	H_2	H_2		
	Moon ->				LOX	LOX	LOX/CH_4	LOX/CH_4
Return	Earth ->	LOX/H_2						
	Mars ->		LOX/CO	LOX/CH_4	LOX/CO	LOX/CH_4	LOX/CO	LOX/CH_4

* NOTE: Engine thrust levels shown above each bar; each bar represents a single engine or set of engines which perform the indicated burn(s)

** For cases using in situ propellants, Mars Descent/Ascent requirements are shown for one of 2 vehicles required

*** For cases using lunar propellant for the outbound trip, TLI and LOI are performed with Earth LOX/H_2

Figure 2-6. Summary of Initial Engine Masses

Table 2-4. Overall Engine System Requirements Summary

Scenario 1: **BASELINE SCENARIO (NO LUNAR/MARS PROPELLANT): EARTH LOX/H2**

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent		
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Earth LOX/H2	uses Earth LOX/H2			
Propellants Used		Earth LOX/H2		Earth LOX/H2	uses Earth LOX/H2			
Specific Impulse (sec)		470		470	470			
Mixture Ratio (O/F)		6.0		6.0	6.0			
Thrust Level(s) (klbs)		1000 - TMI 300 - MOI		300	80			
Engine Operating Time (%of trip)		0.008%		0.002%	0.9%	0.5%		
Total ΔV (m/sec)		4,075 - TMI 2,638 - MOI		2,527 - TEI 4,087 - EOC	5,329	931		
Total Impulse (x10^6 kN sec)		3.923 - TMI 1.073 - MOI		0.379 - TEI 0.286 - EOC	0.134	0.070		
Maximum Acceleration (g's)		0.757/0.451		1.21/3.14	2.719	0.5319		
Operating Time (sec/mission)		882 - TMI 804 - MOI		284 - TEI 214 - EOC	376	198		
Reusability (# of missions)		5		5	5	5		
Refueling Requirements		refueled in LEO		refueled in LEO	refueled in LEO			

Table 2-5. Overall Engine System Requirements Summary

Scenario 2: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX	uses Mars LOX/CO	uses Mars LOX/CO		uses lunar LOX	
Propellants Used	Earth LOX/H2		lunar LOX + Earth H2	Mars LOX/CO	Mars LOX/CO		lunar LOX + Earth H2	
Specific Impulse (sec)	470		470	290	290		470	
Mixture Ratio (O/F)	6.0		6.0	0.6	0.6		6.0	
Thrust Level(s) (klbs)	500		300	300	3500		300	
Engine Operating Time (% of trip)	0.21%		0.0032%	0.0037%	1.2%	0.019%	1.0%	0.27%
Total ΔV (m/sec)	4,436		4,608	6,644	5,356	930	1,911	2,001
Total Impulse (x10^6 kN sec)	1.430		0.922	1.286	8.329	0.125	0.591	0.155
Maximum Acceleration (g's)	0.919/1.25		0.654/1.19	0.65/2.95	3.02	14.13	0.546	2.2
Operating Time (sec/mission)	643		691	964	535	8	443	116
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

Table 2-6. Overall Engine System Requirements Summary

Scenario 3: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX	uses Mars LOX/CH4	uses Mars LOX/CH4		uses lunar LOX	
Propellants Used	Earth LOX/H2		lunar LOX + Earth H2	Mars LOX/CH4	Mars LOX/CH4		lunar LOX + Earth H2	
Specific Impulse (sec)	470		470	380	380		470	
Mixture Ratio (O/F)	6.0		6.0	3.6	3.6		6.0	
Thrust Level(s) (klbs)	700		450	450	1000		450	
Engine Operating Time (% of trip)	0.25%		0.003%	0.0016%	1.2%	0.04%	1.1%	0.29%
Total ΔV (m/sec)	4,445		4,610	6,609	5,353	930	1,913	2,001
Total Impulse (x10^6 kN sec)	2.323		1.317	0.831	2.322	0.085	0.959	0.250
Maximum Acceleration (g's)	0.79/1.09		0.60/1.10	1.47/4.63	2.33	5.57	0.50	2.05
Operating Time (sec/mission)	746		658	415	522	19	479	125
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

Table 2-7. Overall Engine System Requirements Summary

Scenario 4: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX/CH4	uses Mars LOX/CO	uses Mars LOX/CO		uses lunar LOX/CH4	
Propellants Used	Earth LOX/H2		lunar LOX/CH4	Mars LOX/CO	Mars LOX/CO		lunar LOX/CH4	
Specific Impulse (sec)	470		380	290	290		380	
Mixture Ratio (O/F)	6.0		3.6	0.6	0.6		3.6	
Thrust Level(s) (klbs)	450		300	300	3500		400	
Engine Operating Time (% of trip)	0.16%		0.0037%	0.0037%	1.2%	0.019%	0.011%	0.053%
Total ΔV (m/sec)	4,840		4,612	6,644	5,355	930	1,912	2,000
Total Impulse (x 10^6 kN sec)	0.969		1.066	1.278	8.283	0.125	0.824	0.041
Maximum Acceleration (g's)	1.21/1.66		0.57/1.19	0.65/2.97	3.04	14.19	0.55	11.49
Operating Time (sec/mission)	484		799	958	532	8	463	23
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

Table 2-8. Overall Engine System Requirements Summary

Scenario 5: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX/CH4	uses Mars LOX/CH4	uses Mars LOX/CH4		uses lunar LOX/CH4	
Propellants Used	Earth LOX/H2		lunar LOX/CH4	Mars LOX/CH4	Mars LOX/CH4		lunar LOX/CH4	
Specific Impulse (sec)	470		380	380	380		380	
Mixture Ratio (O/F)	6.0		3.6	3.6	3.6		3.6	
Thrust Level(s) (klbs)	200		300	300	1000		600	
Engine Operating Time (% of trip)	0.61%		0.006%	0.002%	1.2%	0.04%	1.2%	0.06%
Total ΔV (m/sec)	4,635		4,638	6,619	5,350	930	1,914	2,000
Total Impulse (x 10^6 kN sec)	1.649		1.720	0.809	2.264	0.085	1.329	0.067
Maximum Acceleration (g's)	0.34/0.47		0.36/0.75	1.0/3.17	2.39	5.68	0.51	10.84
Operating Time (sec/mission)	1,854		1,289	606	509	19	498	25
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

Table 2-9. Overall Engine System Requirements Summary

Scenario 6: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent		
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Mars LOX/CO	uses Mars LOX/CO			
Propellants Used		Earth LOX/H2		Mars LOX/CO	Mars LOX/CO			
Specific Impulse (sec)		470		290	290			
Mixture Ratio (O/F)		6.0		0.6	0.6			
Thrust Level(s) (klbs)		700 - TMI 300 - MOI		300	3500			
Engine Operating Time (% of trip)		0.004%		0.004%	1.2%	0.02%		
Total ΔV (m/sec)		4,008 - TMI 2,597 - MOI		6,644	5,355	930		
Total Impulse (x 10^6 kN sec)		1.457 - TMI 0.402 - MOI		1.282	8.298	0.125		
Maximum Acceleration (g's)		1.39/1.18		0.65/2.96	3.03	14.16		
Operating Time (sec/mission)		468 - TMI 301 - MOI		961	533	8		
Reusability (# of missions)		5		5	5	5		
Refueling Requirements		refueled in LEO		refueled in LMO	refueled on Mars surface			

Table 2-10. Overall Engine System Requirements Summary

Scenario 7: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent		
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Mars LOX/CH4	uses Mars LOX/CH4			
Propellants Used		Earth LOX/H2		Mars LOX/CH4	Mars LOX/CH4			
Specific Impulse (sec)		470		380	380			
Mixture Ratio (O/F)		6.0		3.6	3.6			
Thrust Level(s) (klbs)		1000 - TMI 200 - MOI		200	1000			
Engine Operating Time (% of trip)		0.006%		0.003%	1.2%	0.04%		
Total ΔV (m/sec)		4,015 - TMI 2,629 - MOI		6,640	5,350	930		
Total Impulse (x 10^6 kN sec)		2.326 - TMI 0.648 - MOI		0.804	2.255	0.085		
Maximum Acceleration (g's)		1.25/0.50		0.68/2.14	2.4	5.7		
Operating Time (sec/mission)		523 - TMI 728 - MOI		904	507	19		
Reusability (# of missions)		5		5	5	5		
Refueling Requirements		refueled in LEO		refueled in LMO	refueled on Mars surface			

3.0 TECHNOLOGY REVIEW

A technology review was conducted to support identification of key technology issues associated with multi-propellant, in situ-based propulsion systems of interest to this study. Additionally, this technology review established a corresponding database that supported the assessment, design, and development of such systems. Key areas of interest in this review included heat transfer/cooling, injection/ignition/combustion characteristics, performance, pumping, materials compatibility and tankage. Technology data compiled in this effort was also used to support engine system characterization and the technology assessment of these systems which are reported in Sections 4.0 and 6.0, respectively.

To support this effort, an extensive literature search was undertaken that focused on rocket engine system technology. The NASA/RECON, Dialog and DTIC literature search database sources were surveyed in key technology/design areas, as well as in other areas such as tripropellant engine systems. Hundreds of literature abstracts were reviewed. From this listing, approximately 30 to 50 technical papers were reviewed indepth that covered the range of technology and design areas of interest. In general, it was found that little of the past work identified in the literature search was directly applicable to integrated multipropellant Mars in situ propellant-based propulsion systems. Most of the literature reviewed addressed technologies associated with LOX/H_2 and LOX/Hydrocarbons engine systems that have some relevance to this effort. Results and supporting rationale associated with this technology review in areas unique to Mars multipropellant, in situ-based propulsion systems are summarized in the following.

3.1 Tripropellant Engine Systems

Tripropellant engine systems have many unique similarities as well as differences with multipropellant Mars in situ-based propulsion systems. These similarities include use of three propellants to support engine operations and integration, design issues such as pumping (multiple fuel systems), control and thrust chamber cooling. It is these similarities that make review of past work in this area of interest to this study.

In considering the applicability of past tripropellant engine studies for this assessment, one must understand the application and operational aspects of these studies and those associated with an in situ multipropellant Mars propulsion system. Past tripropellant engine design and supporting technology investigations focused on Single-Stage-to-Orbit (SSTO) and advanced Earth-to-Orbit (ETO) applications. These engine systems designs stress optimal performance over a typical ETO trajectory with minimal engine system hardware to keep weight at a minimum. Typical tripropellant engine system operation for a dual-throat or dual-expander cycle engines is shown in Figure 3-1. Multimode operation of these engine systems is performed in parallel. During low altitude operation, the LOX/Hydrocarbon and/or LOX/H_2/Hydrocarbon engine segments are operated. In this operating mode, moderate thrust density and performance is achieved. At high altitude, only the LOX/H_2 engine segment is operated which gives low thrust density, but high performance.

These engine system design/operation features are different from those associated with multipropellant Mars in situ-based propulsion systems, where only single bipropellant combinations are operated in series, restartability is required and commonality of hardware, such as a single thrust chamber, is stressed. Though these differences exist, review of past data in this area was considered worth while due to many of the design issues and technology areas they have in common, as previously mentioned.

There is an extensive past database available associated with tripropellant engine systems. Most of the work has been accomplished by Aerojet. They initiated this work in the early 1970's and have been active at a modest level since then. This work has been both IR&D and contract supported. Aerojet has performed numerous engine system and application studies, and supporting technology experimental investigations, see Refs. 3-1 through 3-4. Another past study of interest is one performed by Rocketdyne in 1977, see Ref. 3-5. This study examined the feasibility of modifying the Space Shuttle Main Engine (SSME) for dual mode operation. This is quite different than the other studies conducted in this area because it examined the performance and compatibility issues of a given engine design optimized for LOX/H_2 and operating it with a LOX/Hydrocarbon propellant combination. Such issues and design tradeoffs are typical of the Mars engine systems of interest to this study.

3.2 Heat Transfer/Cooling

Heat transfer and cooling of the thrust chamber was identified as a key issue associated with in situ-based multipropellant Mars engine systems. Key issues associated with this area are: 1) regeneratively cooling thrust chambers using LOX or CO and 2) the design of a regeneratively cooled thrust chamber that can effectively operate with different coolants during different phases of operation associated with a Mars tripropellant engine. Both issues greatly impact the cycle selection and design of this class engine.

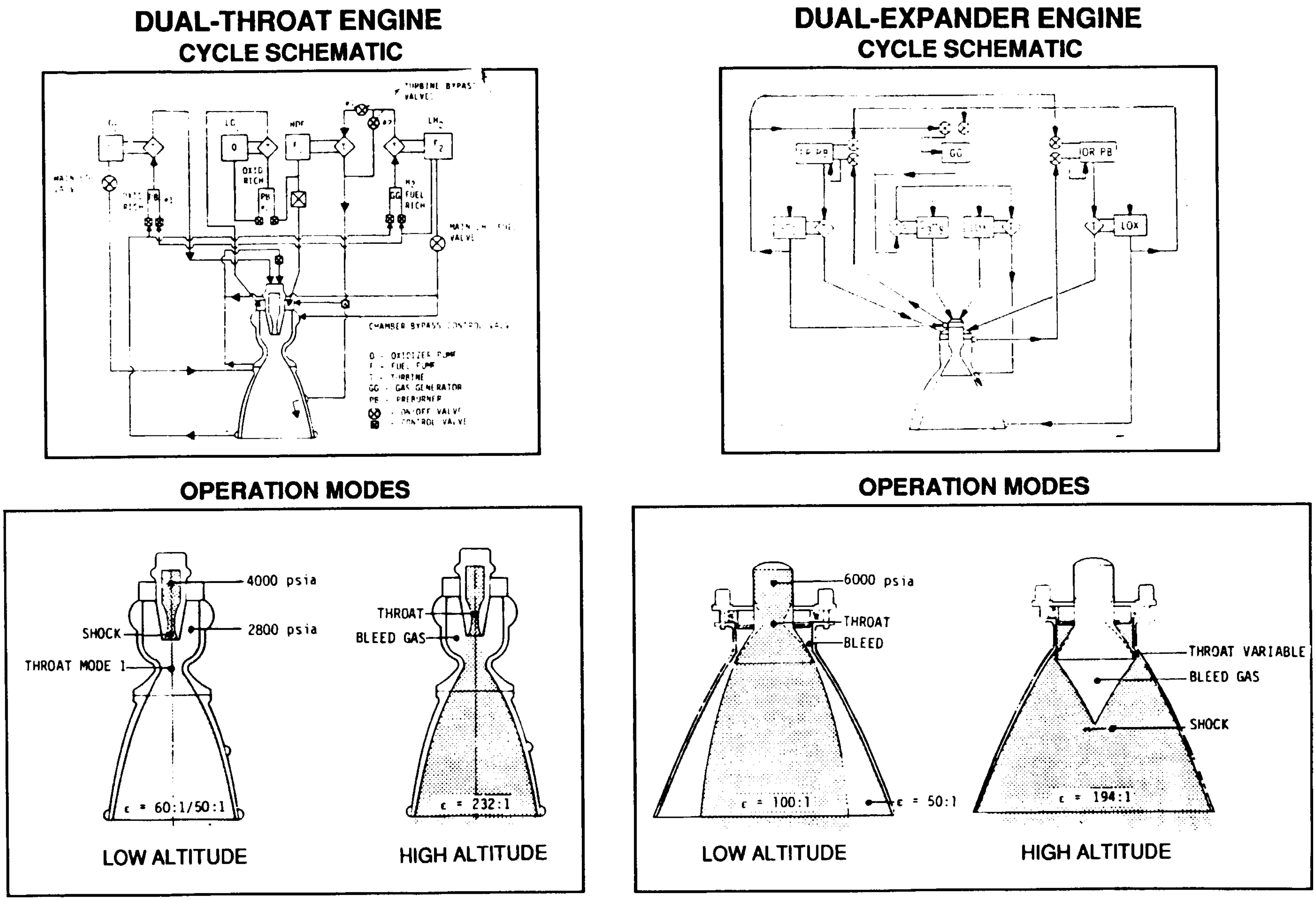


Figure 3-1. Typical Tri-Propellant Engine System Operation

Thrust chamber cooling characteristics for numerous propellants of interest, such as H_2, LOX, and CH_4 plus others, are summarized in Ref. 3-6. Review of the literature indicated that there is extensive data available for using H_2 and Hydrocarbons (CH_4) to cool engine thrust chambers. This area has been extremely active in recent years due to related interest in cooling the SSME, Space Transportation Booster Engine (STBE) and Space Transportation Main Engine (STME), see Refs. 3-7 through 3-9. Fundamental and applicable engine system design data in this area is available.

Some applicable data on the cooling of thrust chambers using LOX is also available. Aerojet's Orbit Transfer Vehicle (OTV) engine concept, see Ref. 3-10, employs a high performance LOX-cooled thrust chamber. Research and development in this area for large engine applications has been conducted for many years, see Refs. 3-11 and 3-12. Additionally, fundamental data associated with LOX cooling is available, see Ref. 3-6.

The literature survey identified no past experimental or analytical work that examined CO as a thrust chamber coolant or supporting fundamental data that would be applicable for such an application. Recent NASA LeRC's work which addressed the use of CO as an engine system coolant, Ref. 3-13, and experimental investigations in this area, Ref. 3-14, were the only relevant items found. It is important that fundamental CO cooling data be established.

Another key result of the technology review in this area was that no literature and/or data was found in the thrust chamber design area that used more than one propellant in series as a coolant. Such an engine system design feature would be highly desirable for Mars in situ-based multipropellant engine systems. It should be noted that past tripropellants engine designs were not required to be cooled in such a manner. They typically operate their various engine modes in parallel and/or use H_2 as a thrust chamber coolant, which is well documented.

3.3 Injection/Ignition/Combustion

A number of issues were investigated in the injection/ignition/combustion technology area. Key technology and/or design issues include: 1) CO injection, ignition and combustion characteristics, 2) gas generator design for a multi-propellant Mars in situ-based tripropellant engines, and 3) multipropellant injector design performance and thrust chamber cooling compatibility.

The literature review indicated that fundamental CO injection, ignition and combustion data is lacking. No past relevant work was found except for the recent ongoing NASA LeRC study efforts examining this area, see Ref. 3-15. Such data is critical in the design and assessment of engine systems employing CO as a propellant.

Due to the multipropellant compatibility and the wide operating range that will likely be required of a Mars in situ-based tripropellant engine system, a conventional gas generator design may not be optimal. Recent work by NASA LeRC, Ref. 3-15, has shown that for ignition of LOX/CO, mixture ratios that are associated with relatively high combustion temperatures for gas generators may be required which will greatly affect the design and reliability of the propellant system's turbopump(s) drive turbine. Recent work by Aerojet on a stoichiometric gas generator concept, Ref. 3-16, addresses many of these issues. It is an attractive design option for inclusion in a candidate Mars in situ-based tripropellant engine system. This concept employs a small core flow at stoichiometric combustion (high temperature) conditions that is diluted downstream by the addition of propellant to a lower temperature, before it enters the turbine drive region.

Advanced ignition devices technologies, such as laser igniters, are other technology options that should be considered for Mars in situ-based tripropellant engine systems. They are relatively lightweight, reliable and have the potential to perform the ignition function for a number of propellant combinations over a wide range of operating conditions. This technology is maturing rapidly and is currently being developed for solid motor and National Aerospace Plane (NASP) applications.

Little literature or supporting data was found that addressed the issues and/or design of a single injector for more than one combination of propellants. Aerojet's past tripropellant engine design efforts did not address this issue because they employ separate embedded combustor(s) or outer ring combustor designs, see Figure 3-1. Rocketdyne's past tripropellant SSME study effort, Ref. 3-5, showed that using a single injector design for more than one propellant combination, LOX/H_2 and LOX/CH_4, was a major problem. In addition to performance issues, stability and thrust chamber cooling compatibility over a wide range of operating conditions are other issues that need further study.

3.4 Pumping

Key technology/design areas associated with pumping technology of Mars in situ-based tripropellant engine systems are: 1) Warm O_2 and oxidizer-rich driven turbopumps, 2) the pumping of CO, and 3) multipropellant capable, single turbopumps designs.

Warm O_2 and oxidizer-rich driven turbopump designs have been examined in the past that have applicability to the design and assessment of Mars in situ-based engine systems of interest to this study. Such a turbopump is incorporated in Aerojet's OTV engine design, Ref. 3-17. R&D has been performed in this area for many years and some supporting fundamental data is available. Design issues associated with this class of turbopump are well understood.

Little data was found to be available in the literature on the pumping of CO. It is believed that the best source for this data may reside in the petroleum/chemical industry, Ref. 3-18, but no effort was undertaken in this study to substantiate this claim. NASA LeRC has performed some recent work, Ref. 3-13, that addresses CO pumping requirements and performance for applicable engine systems of interest. This work is preliminary in nature and needs to be substantiated by the development of a fundamental database in this area.

The literature survey showed that design issues associated with multipropellant capable, single turbopump designs are well understood, but little demonstrated capability or supporting data is available in this area. The Rocketdyne tripropellant study, Ref. 3-5, which examines the use of SSME turbopump hardware for multipropellant usage does address this issue. No substantial turbopump design and/or test work has been done in this area.

3.5 Materials Compatibility

The compatibility of a propellant and/or its by-products (after it is burned with another propellant) with which the engine material interfaces is critical for all the major subsystems/components, such as the propellant tank(s), fuel line(s), valve(s), turbopump(s), thrust chamber, and nozzle of any liquid propulsion system.

The multipropellant capability, wide operating range, and the maximum use of common hardware for engine systems of interest in this study, stress the material options and technologies available to support its development. Key design and technology issues examined in this area were: 1) Warm O_2 and oxidizer-rich turbine materials that are compatible, 2) O_2, CH_4 and CO compatible materials for thrust chamber applications, and 3) materials that are all compatible with CO, CH_4 and H_2 for common fuel propellant tank applications.

The literature survey identified some fundamental data on warm O_2 and oxidizer-rich turbopump turbine materials. Aerojet has been active in this area for many years. An example of the data available, depicted in Table 3-1 and discussed in Ref. 3-19, shows compatibility data for candidate O_2 driven turbopump materials. Review of the literature in this area has shown that design issues associated with this area are well understood but that more data is required to properly design such systems with a high degree of confidence.

Table 3-1. Example Propellant/Material Compatibility Data
– Candidate Burn Resistant Materials for Oxygen-Driven Turbopumps* –

Material	Burn Factor	Observations
Zirconium Copper	35	No Ignition in Any Test (790/1800°F)**
Nickel 200	550	Ignition Above 2200°F in FRT Only (825/220°F)
Silicon Carbide	1145	No Ignition in Limited Testing (850/—°F)
Monel 400	1390	Ignition Above 1200°F FRT Only (800/1200°F)
K Monel-500	2090	Ignition Above 1500°F FRT (750/1500°F)
Inconel 600	3226	Ignition Above 1100°F (—/1000°F)
316 Stainless Steel	4515	Ignition in All Tests (450/800°F)
Invar-36	5444	Ignition in All Tests (675/340°F)
Hastelloy-X	7160	Ignition in All Tests (725/750°F)

* L. Schoenman, AIAA Journal of Propulsion and Power, Volume 3, No. 1, Jan-Feb 1987, Pages 46-55.
** Temperatures from particle impingement test friction rubbing test (FRT) at 1000 psi and 17,000 rpm.

Materials compatibility data for CO was found to be lacking. Little was found in the open aerospace literature. Only one document in this area was found to be relevant, Ref. 3-20, but was classified and could not be reviewed. A discussion with an expert in this field, Ref. 3-19, indicated that the petroleum/chemical industry is probably the best source for this information, but no effort in this study was undertaken to substantiate this claim. Additionally, this expert claimed that for a first approximation, to support preliminary design efforts, that materials which are compatible with CH_4 would likely be compatible with CO except for materials that have iron content. Fundamental data needs to be established in this area.

Little data was found to be available that addresses the common compatibility of a number of propellant of interest in this study (O_2, CH_4, H_2 and CO), with material candidates that are used in thrust chambers, propellant tanks, lines, and valves. Some fundamental data was found to be available for many specific propellant/material combinations. Data needs to be established experimentally in this area to address the commonality issue. Based on the literature, Table 3-2 presents a "top-level" preliminary propellant/material compatibility screening summary for many of the materials and propellants of interest to Mars in situ-based propulsion systems.

Table 3-2. Example Propellant/Material Compatibility Data
– Propellant/Material Compatibility Screening Summary –

PROPELLANT \ MATERIAL	ALUMINUM	Al ALLOY	BRASS	CHROMIUM	Cr-Ni STEEL	COPPER	Cu ALLOY	INCONEL	IRON	IRON OXIDE	LEAD	MAGNESIUM	Mg ALLOY	NOMEL	NICKEL	Ni ALLOY	STAINLESS STEEL	STEEL	TITANIUM	Ti ALLOY	ASBESTOS	FLUORINE FILM	FLUOROCARBON	FREON	GLASS	POLYETHYLENE	POLYVINYLCHLORIDE	PYREX	RUBBER	RUBBER (RESISTANT)	TIN	ZINC
Oxygen	●	●	●	●	●	●	●	●						●	●		●				●	●	●							●		
Hydrogen	○	●				○	○	●						●	●		●				●		●									
Methane	●	●	○	●	●	○	○	●	●		●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●		●		
Carbon Monoxide*	●	●	○	●	●	○	○	●			●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●		●		

● Compatible With One or More Materials in the Category

○ Compatible Under Limited Conditions

* Based on Similar Material Compatibility Characteristics With Hydrocarbon Propellants (Estimated)

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4.0 PROPULSION SYSTEM DESIGN

The principal goal of this study effort portion was to characterize promising systems that can efficiently use the multiple propellant combination of interest to this study, $LOX/H_2/CO$, $LOX/H_2/CH_4$ and $LOX/CO/CH_4$. This effort focused on defining representative engine systems that meet the overall mission requirements such as performance, weight, thrust level, throttling and operation mode (series operations), for many of the scenario options discussed in Section 2.0. Additionally, these representative engine systems were configured to: 1) use the maximum amount of common engine system hardware, while attempting to minimize engine system mass, and 2) exhibit high performance for each engine operating mode and range of interest.

Major tripropellant engine system elements considered for commonality are shown in Figure 4-1. These engine system elements included the fuel propellant tank, oxidizer feed system, injector, thrust chamber, and nozzle. For the initial study effort, common fuel feed systems were not considered due to the inherent difference in pumping requirements for the fuels considered. Such requirements would produce a common fuel turbopump design that would operate inefficiently over the range in which it would be required to operate. This design issue was addressed in a preliminary manner in the latter portion of this study. Additionally, in a latter portion of this effort, propellant tank system sizing and commonality issues are also addressed.

To perform this effort, top-level engine system requirements were established from the initial mission analysis results discussed in Section 2.0. Promising engine system concepts were then identified for further study. A baseline technology/design database was then established for each engine system concept. The database drew on results from the initial technology review that is discussed in Section 3.0. These candidate engine system concepts were then analyzed by using SAIC's version of the Expanded Liquid Engine System (ELES) analysis code, see Refs. 4-1 and 4-2. Using ELES, numerous design sensitivity analyses were performed to determine the influence of key engine system parameters such as: mixture ratio, chamber pressure, nozzle area ratio, injector pattern density and type, turbine bypass, regenerative cooling channel bypass, turbine inlet temperature, and thrust chamber channel design geometry. From these sensitivity studies, representative engine systems were identified. Propellant tank system requirements were established, and design and sizing of representative candidate systems using the ELES analysis code was then performed at the conclusion of this study effort.

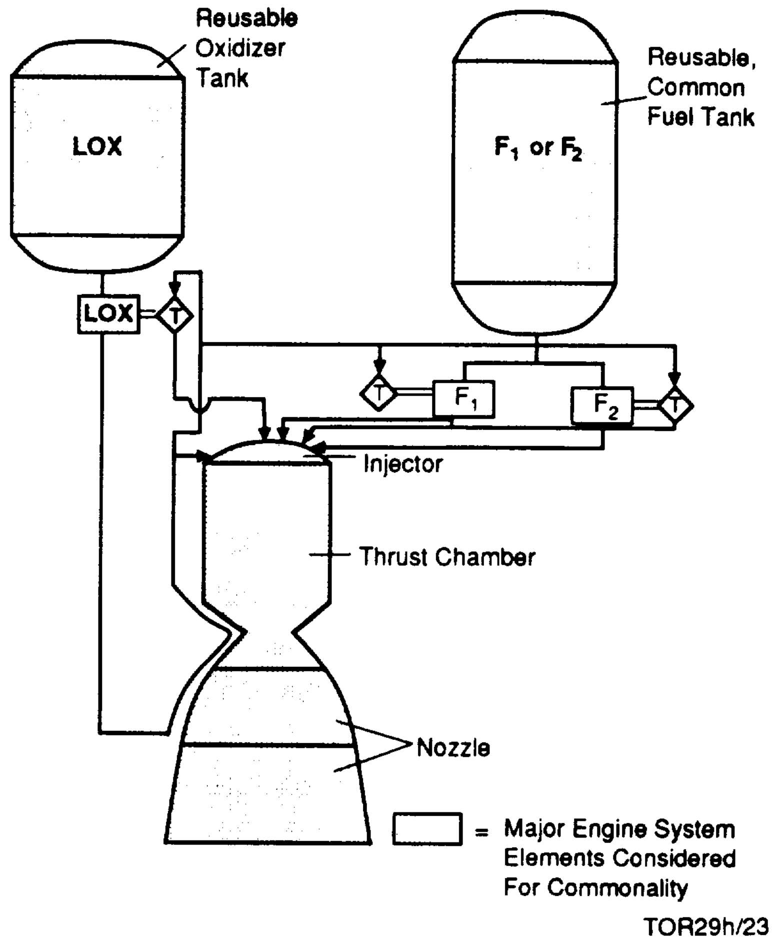

Figure 4-1. Representative Tripropellant Engine System Common Hardware Elements

The following discussion highlights the engineering assumptions and rationale and results in characterizing representative common tripropellant propulsion system candidates to support in situ propellant-based Mars missions.

4.1 Engine System Requirements Concepts

Engine system design requirements were derived from the initial mission analysis assessment discussed in Section 2.0. From these requirements, top-level baseline representative engine system concepts were identified that addressed a large portion of the mission scenarios considered in this study. These engine system concepts were then defined and characterized in more detail in the engine assessment portion of the study, see Section 4.2. The following sections address the development and rationale of the engine system design requirements and the identification of the baseline engine system concepts.

4.1.1 Identification of Requirements

Figure 2-2 shows the two basic mission profiles that were considered for this study that use ISPP resources. Mission Profile No. 1, which corresponds to Scenarios 2, 3, and 4, uses some form of in situ propellants from both the Moon and Mars, and Mission Profile No. 2, which correspond to Scenarios 6 and 7, employs only Mars in situ-produced propellants. Review of the initial mission analysis results and their corresponding requirements indicates that the in situ engine system commonality would best be leveraged for the transfer vehicle outbound and inbound mission elements. Little differences in engine system requirements, such as for thrust level and acceleration profiles, were found for these mission elements. Likewise, it was noted that a high proportion of the overall mission delta-v is associated with these mission segments. Large differences in excursion vehicle engine system requirements, such as thrust levels and acceleration profiles, were also observed. From this initial assessment of requirements it was concluded for further study that: 1) the baseline engine system(s) be based on transfer vehicle requirements, and 2) that these baseline engine system(s) and/or their hardware be used only where possible to meet excursion vehicle requirements.

Other engine system design assessment requirements specified are that the baseline engine system examined should be easily scalable in terms of thrust level and address key functions, design issues, and technologies that are representative of such systems. Due to the nature of the deep space missions considered, high reliability and reusability (five missions) would be required. This was addressed in the study by employing one or both of the following approaches: 1) sizing the propulsion system with engine out capability and/or 2) operating at a derated power level for most of the mission operation profile. Because of man-rating considerations, a maximum vehicle acceleration level of 3 g's was assumed which is directly related to an engine system's throttling requirements. A conservative limit of 2.8 g's was used in the requirements analysis.

Considering many of the just mentioned engine system requirements and reviewing the initial mission analysis results, top-level requirements for baseline engine system candidates were derived which are displayed in Table 4-1. These candidate engine systems address a large portion of mission scenario trade space as shown in Table 4-2. At least one engine concept shown in Table 4-1 applies to all deep space transfer and excursion mission segments which employ multiple fuels to perform the mission. The LEO → LLO transfer mission phase is not addressed by any of the engine system concepts because the transfer vehicle uses an expendable LOX/H_2 stage that is a more conventional engine system, which is not of interest to the study. Likewise, the engine system concepts do not address Scenarios 1 and 5 because they use only conventional single-propellant combinations, LOX/H_2 and LOX/CH_4, respectively. Table 4-3 shows the number of engines, the percent power rating level, and engine out capability, if specified, by each mission segment for each applicable engine system concept.

Table 4-1. Top-Level Requirements for Engine System Candidates

Concept No:	1	2	3
Propellants	$LOX/H_2/CO$	$LOX/H_2/CH_4$	$LOX/CH_4/CO$
Thrust Level (lbf):	175,000	250,000	175,000
Throttling Range:	5:1	2.2:1	6:1

4.1.2 Engine System Cycle Considerations/Recommendations

After initial sizing of the baseline engine systems was completed, engine system options and their applicability to meet the tripropellant Mars in situ propellant engine system requirements, were then addressed. Table 4-4 lists the numerous candidate engine cycles considered. Assessment factors used in evaluating these engine cycles are given in Table 4-5. These factors are highly coupled to overall requirements unique to the missions of interest. Table 4-5 also shows how these factors impact engine cycle design characteristics. A top-level comparison of these engine cycle candidates is shown in Table 4-6. Major advantages and disadvantages of each engine cycle option are presented as well as a qualitative assessment of its applicability to meet in situ propellant-based Mars evaluation factors.

The staged combustion cycle maximizes performance for a given engine size by eliminating secondary flow losses and by maximizing the energy available to drive the turbine. The turbomachinery is subjected to high-pressure operating conditions because the turbine drive gases are injected into the main combustion chamber at its stagnation chamber pressure level. This exposes the main injector to high-temperature turbine gases. Though it exhibits good performance and thrust-to-weight traits, it has marginal reliability and multipropellant capability qualities because of its inherent complexity.

The gas generator cycle is a simplified system that maximizes the independence of the components, which is done by placing the turbine gas flow path in parallel with the thrust chamber gas flow path. It also lends itself to independent component experimental development that helps ensure high initial system reliability. The gas generator cycle, due to its simplicity and operational maturity, meets all assessment factors positively except for performance which is marginal.

Table 4-2. Engine System Assessment Trade Space Summary

VEHICLES / SCENARIOS	TRANSFER				EXCURSION				COMMENT(S)
	EXP. STAGE	MTV			MEV		LEV		
	LEO→LLO	LEO→LMO	LLO→LMO	LMO→LEO	ASCENT	DESCENT	ASCENT	DESCENT	
1. Baseline Case (No Lunar/ Mars Propellant): Earth LOX/H_2									Single Propellant Combination (LOX/H_2). Extensive Technology Data Base Available. Not Recommended for Study.
2. Lunar LOX (Earth H_2) for Outbound and Mars LOX/CO for Return			1*	1	1	1	1	1	
3. Lunar LOX (Earth H_2) for Outbound and Mars LOX/CH_4 for Return			2	2	2	2	2	2	
4. Lunar LOX/CH_4 for Outbound and Mars LOX/CO for Return			3	3	3	3	3	3	
5. Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 for Return									Single Propellant Combination (LOX/CH_4). Extensive Technology Data Base Available. Key Design Issues Addressed in Concept No. 2 and 3 Assessments. Not Recommended for Study.
6. Earth LOX/H_2 for Outbound and Mars LOX/CO Return		1		1	1	1			
7. Earth LOX/H_2 for Outbound and Mars LOX/CH_4 for Return		2		2	2	2			
COMMENT(S)	Single Propellant Combination (LOX/H_2), No Reuse Capability Required. Extensive Technology Base Available, Not In Situ Propellant Mission Driven. Not Recommended for Further Study.								

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* *Engine Concept Number*

Table 4-3. Number of Engines and Power Rating Summary

VEHICLES / SCENARIOS	TRANSFER				EXCURSION			
	ESP. STAGE	MTV			MEV		LEV	
	LEO→LLO	LEO→LMO	LLO→LMO	LMO→LEO	ASCENT	DESCENT	ASCENT	DESCENT
1. **Baseline Case (No Lunar/ Mars Propellant): Earth LOX/H_2**								
2. Lunar LOX (Earth H_2) for Outbound and Mars LOX/CO for Return			2/.86*	2/.86	20/1.00 22/.91**	20/1.00 22/.91**	2/.86	2/.86
3. Lunar LOX (Earth H_2) for Outbound and Mars LOX/CH_4 for Return			2/.90	2/.90	4/1.00 6/.67**	4/1.00 6/.67**	2/.90	2/.90
4. Lunar LOX/CH_4 for Outbound and Mars LOX/CO for Return			2/.86	2/.86	20/1.00 22/.91**	20/1.00 22/.91**	3/.76	3/.76
5. Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 for Return								
6. Earth LOX/H_2 for Outbound and Mars LOX/CO Return		TMI 4/1.00 MOI 2/.86		2/.86	20/1.00 22/.91**	20/1.00 22/.91**		
7. Earth LOX/H_2 for Outbound and Mars LOX/CH_4 for Return		TMI 5/.80 MOI 1/.80		1/.80	4/1.00 6/.67**	4/1.00 6/.67**		

* *No. of Baseline Engines/Percent Rated Power Level ($x10^2$)*

** *Engine Out Capability*

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Table 4-4. Candidate Engine Cycles

- Staged Combustion
- Gas Generator
- Expander
- Hybrid Staged Combustion
- Augmented Expander
- Dual Expansion
- High Pressure, Low Pump Discharge
- Thrust Chamber Tapoff
- Full Bleed Cycle

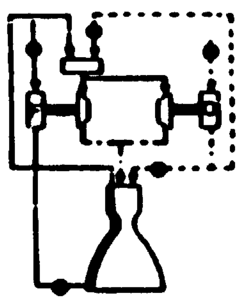
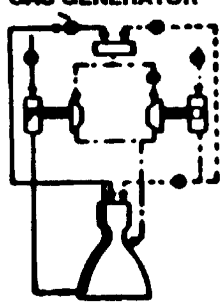
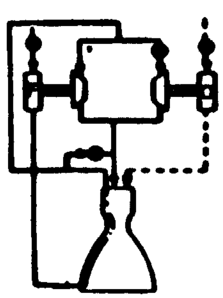
Table 4-5. Key Engine Cycle Assessment Factors for In Situ Propellant-Based Mars Missions and Their Impact on Engine Cycle Design

- High Performance
- High Reliability → Simple Design
- High Thrust/Weight → High Pressure Operation, Compact Packaging
- Throttleability → Controllable, Simple Design
- Multi-Propellant Capability → Simple Design/Operation
- Maturity

Maximum performance can be obtained by employing an expander cycle for a given engine complexity by eliminating both the secondary flow losses and the need for a hot-gas preburner. It is the most benign system for the turbomachinery, but is limited to maximum chamber pressure operation by the available energy to drive the turbines. This results in a relatively low chamber pressure that translates into low thrust-to-weight and large engine systems. Like the staged combustion cycle, it is a high coupled, complex system. Its applicability is for low-thrust and high-altitude (orbit transfer) engines. Though rating high on performance, reliability, and operational maturity, it exhibits low thrust-to-weight and marginal throttleability and propellant compatibility characteristics.

The remaining engine cycles considered in Table 4-6 are derivatives and/or combinations of the basic three-cycle types just mentioned. These remaining engine cycle options exhibit little in terms of positive features to meet the engine assessment requirements.

Table 4-6. Candidate Engine Cycle Top-Level Comparison

CYCLE TYPE	ADVANTAGES	DISADVANTAGES	IN SITU PROPELLANT-BASED MARS ENGINE ASSESSMENT					
			PERFORMANCE	RELIABILITY	THRUST/WEIGHT	THROTTLEABILITY	MULTI-PROPELLANT COMPATIBILITY	MATURITY
STAGED COMBUSTION 	• Highest Performance of All Engine Cycles for Given Engine Size • Eliminates Secondary Flow Losses • Maximizes Energy Available to Drive Turbine	• Most Complex Single Cycle • Largest Number of Components • High Thrust to Weight	+*	–	+	0	–	+/0
GAS GENERATOR 	• Simpler Cycle Than Staged Combustion Cycle • Minimizes Interdependence of Engine Components • Allows Independent Component Experimental Development	• Lower Performance Than Staged Combustion Cycle • High Thrust to Weight	+/0	+	+	+	+	+
EXPANDER 	• Simplest Cycle • Smallest Number of Components • Eliminates Secondary Flow Losses • Eliminates Need for Hot Gas Preburner • Most Benign Turbomachinery Operating Conditions	• Maximum Operating Chamber Pressure Limited by Available Energy to Drive Turbines • High Performance and Thrust at Attainable Chamber Pressure • Relativity "Slow" Start Up Transient	+	+	–	0	0/–	+

* "+" = *Positive Feature,* "0" = *Neutral,* "–" = *Negative Feature*

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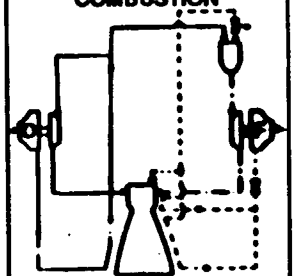
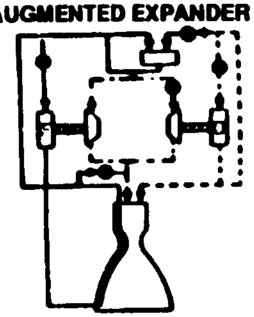
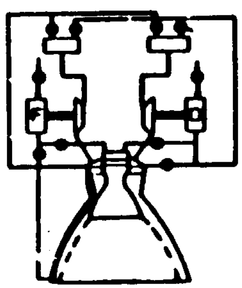
Table 4-6. Candidate Engine Cycle Top-Level Comparison (Cont.)

CYCLE TYPE	ADVANTAGES	DISADVANTAGES	IN SITU PROPELLANT-BASED MARS ENGINE ASSESSMENT					
			PERFORMANCE	RELIABILITY	THRUST/WEIGHT	THROTTLEABILITY	MULTI-PROPELLANT COMPATIBILITY	MATURITY
HYBRID STAGED COMBUSTION 	• Increases Total Turbine Energy Available by Combining Staged Combustion and Expander Cycles • Eliminates Need for LOX Turbopump Purge Seal	• Highly Complex Cycle • Large Number of Components	+*	—	+	—	—	—
AUGMENTED EXPANDER 	• Turbine Inlet Temperature Higher Than Conventional Expander Cycle Through Use of Prebumer in Parallel With Turbine Drive Flow	• Highly Complex Cycle • Large Number of Components • Less Reliable Than Regular Expander Cycle	+	—	0/—	—	0	—
DUAL EXPANSION 	• High Performance at All Altitudes • Increases Total Turbine Energy Available by Utilizing Dual Staged Combustion Cycles • Eliminated Need for LOX Turbopump Purge Seal	• Most Complex Combined Cycle • Large Number of Components	+/0	—	0	0	0	0

* "+" = *Positive Feature* "0" = *Neutral,* "–" = *Negative Feature*

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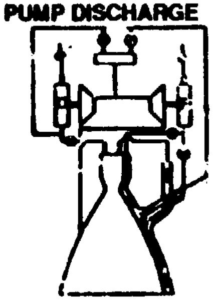
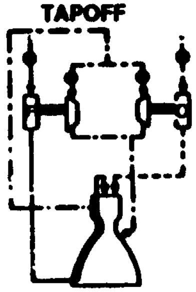
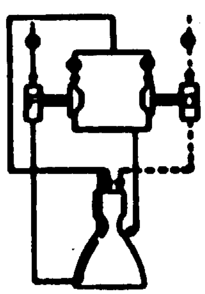
Table 4-6. Candidate Engine Cycle Top-Level Comparison (Cont.)

CYCLE TYPE	ADVANTAGES	DISADVANTAGES	IN SITU PROPELLANT-BASED MARS ENGINE ASSESSMENT					
			PERFORMANCE	RELIABILITY	THRUST/WEIGHT	THROTTLEABILITY	MULTI-PROPELLANT COMPATIBILITY	MATURITY
HIGH PRESSURE, LOW PUMP DISCHARGE 	• Pump Discharge Pressure Reduced by Using Only Gas Generator Available Fuel to Regeneratively Cool Main Combustion Chamber	• Complex Cycle • Large Number of Components	+/0*	—	0	—	0	—
THRUST CHAMBER TAPOFF 	• Eliminates Need for Gas Generator by Obtaining Turbine Drive Gas From Main Combustion Chamber	• Highly Complex Cycle • Susceptible to Injector Tapoff Characteristics as a Function of Chamber Pressure • May Require Fuel Injection Bleed for Temperature Stabilization	+/0	+/0	0	0	0	+/0
FULL BLEED CYCLE 	• Simple Cycle • Combines Features of Both Expander and Gas Generator Cycle	• Low Performance • Relativity "Slow" Start Up Transient	+/0	+/0	0/-	0	0	+/0

* "+" = *Positive Feature,* "0" = *Neutral,* "–" = *Negative Feature*

TOR29v/16b

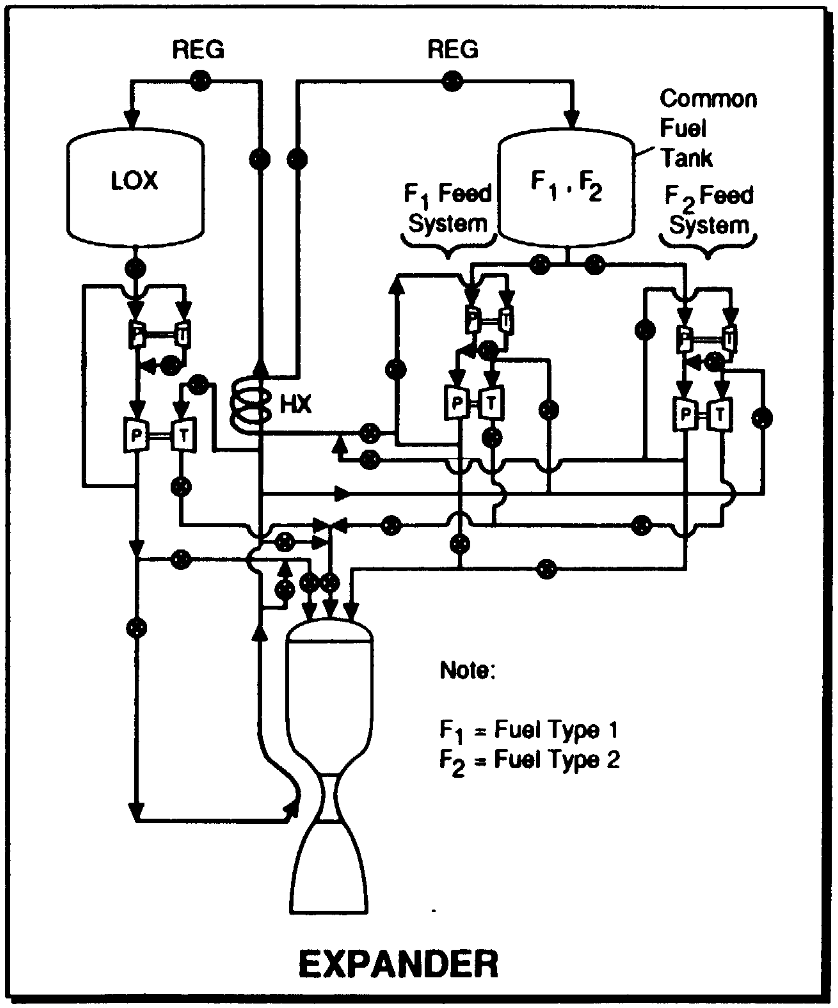
Based on this assessment, expander and gas generator engine cycles were selected for further study. The cycles demonstrate many key engine features, shown in Table 4-7, that are typical of Mars in situ propellant-based engine design options. By examining both engine cycles one bounds, from a technical perspective, the range of available options. The expander cycle, which is high performance, complex, and exhibits low thrust-to-weight, represents one class of engine system designs, while the gas generator cycle, which is simpler, with moderate performance and high thrust-to-weight characteristics, represents an engine class substantially different than the expander cycle. Both engine cycles have been demonstrated in operational systems and have been shown to be highly reliable.

Table 4-7. Engine Cycles Which Demonstrate Many Key Engine Features of Interest

- Expander
  - High Performance
  - Low Thrust/Weight Ratio
  - Coupled Design/Operation
- Gas Generator
  - Moderate Performance
  - High Thrust/Weight Ratio
  - Decoupled Design/Operation
- Both
  - Highly Reliable
  - Demonstrated Maturity

Another key result of the assessment was that for all the engine systems to be investigated, all of them are to be cooled with LOX through all modes of their operation. This engine system design feature was selected because: 1) oxygen is a common lunar/Mars in situ propellant resource, and 2) it eliminated multipropellant cooling design issues that were discussed in more detail in Section 3.0.

The generic tripropellant engine system cycles selected for detailed study are displayed in Figure 4-2. For these LOX-cooled systems, note that a common multipropellant-compatible fuel tank, LOX tank and feed system, autonomous pressurization system, injector, thrust chamber, and nozzle are used in all operating modes. Each fuel has its own independent feed system, as previously mentioned.



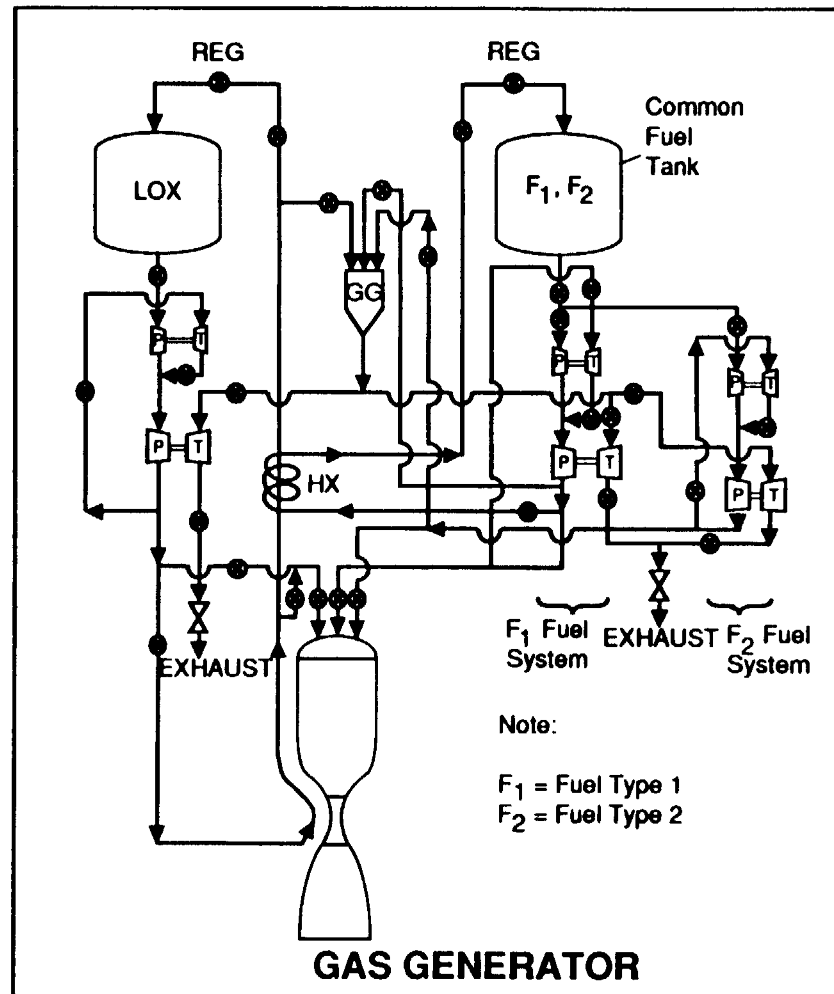


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Figure 4-2. Generic Engine Cycles Studied

4.2 Engine System Assessment

Based on initial engine system requirement/concept definition results discussed in Section 4.1, many candidate baseline propulsion system configurations were defined and analyzed in detail with SAIC's version of the ELES analysis code. Numerous engine system design sensitivity trades were conducted on the candidate baseline engine concepts. From these results, baseline tripropellant MTV and bipropellant engine system designs were identified and characterized. These engine system designs were then used to update overall mission performance, and to identify critical technology and design issues that are discussed in Sections 5.0 and 6.0, respectively. The following sections discuss the analysis approach, assumptions, and results associated with the assessment of the engine system designs.

4.2.1 Assessment Approach and Assumptions

Numerous baseline engine systems were defined and characterized. Three tripropellant engine systems for MTV applications and many bipropellant engine system versions of these engines for LEV and MEV applications were assessed. Expander and gas generator engine versions of each engine option were evaluated. Table 4-8 summarizes these baseline engine system options. This translates into a family of engines for each engine concept, as is shown in Figure 4-3.

Table 4-8. Baseline Engine Systems Defined

- Three (3) Engine Concepts (Propellant Combinations):
  - MTV Engine Options:
    - -- $LOX/H_2/CO$ – 175,000 lbf Thrust
    - -- $LOX/H_2/CH_4$ – 250,000 lbf Thrust
    - -- $LOX/CH_4/CO$ – 175,000 lbf Thrust
  - LEV and MEV Engine Options:
    - -- LOX/H_2
    - -- LOX/CO
    - -- LOX/CH_4

    Many Engine Versions as a Function of Engine Concept
  - Expander and Gas Generator Engine Cycle Versions Were Evaluated for Each Engine Option Listed Above

As previously mentioned, SAIC's version of the ELES analysis code was used to characterize the baseline engine systems. ELES, see Ref. 4-1 and 4-2, is an industrial standard analysis code that designs and determines operational parameters and performance of liquid propulsion systems. It employs empirical and mechanistic design approaches to predict overall propulsion system and subsystem dimensions, weights, operating characteristics, and performance. It has the capability to model a wide range of engine cycles, cooling options, engine and tankage configurations, system component parameters, and construction materials. Additionally, it has the capability to perform vehicle stage and tank system designs. ELES has been verified extensively against real operational propulsion systems, see Ref. 4-3.

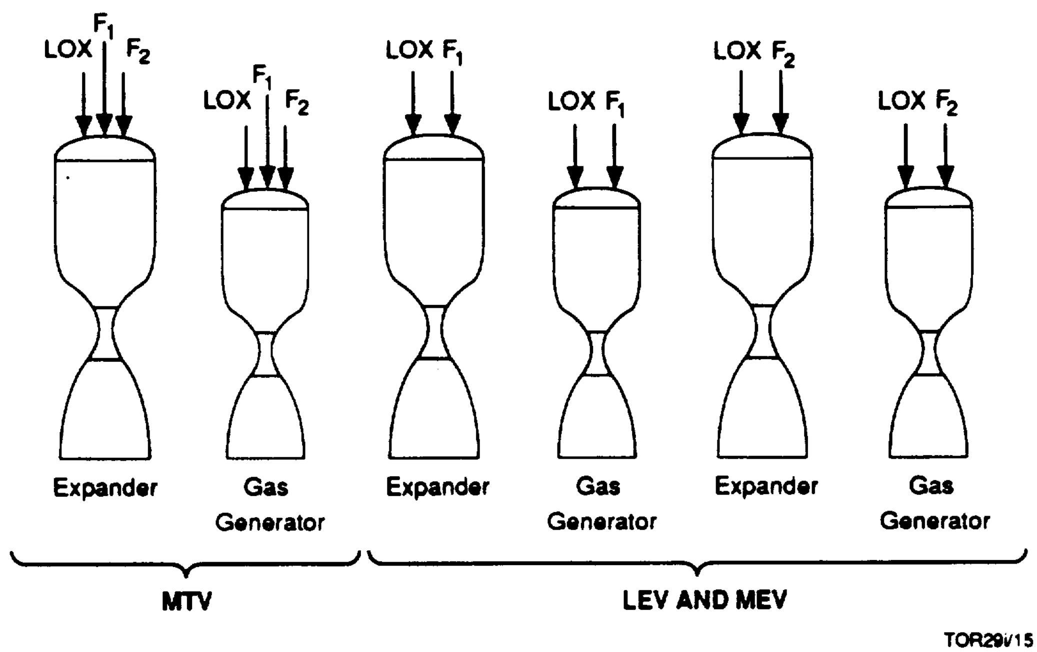


Figure 4-3. Definition of an Engine Family for Each Engine Concept

To perform the engine system analysis, a CO propellant properties library and an off-design engine operation analysis capability were incorporated in ELES. The off-design analysis capability is an essential requirement to characterize the tripropellant engine options. This is because once an engine system hardware design is established for one operational mode using one bipropellant combination, it then must be characterized for a different operational mode that uses, possibly, a different bipropellant combination. Appendix B summarizes the modifications that were performed to ELES to provide the off-design analysis capability.

In performing the many engine design sensitivity trade studies, numerous parameters were investigated. Major parameters examined are listed in Table 4-9. Key screening criteria used in evaluating the trade study results are given in Table 4-10. All screening criteria were considered and sound engineering practice was applied in assessing the results. Specific impulse, engine system weight, size and operating conditions, and their comparison to state-of-the-art (SOA) technology limits were primary evaluation considerations; the impact of an engine design parameter on in situ architecture infrastructure requirements was given secondary importance. Engine parameter ranges and design features that produce engine systems which exhibit one or more of the following engine system traits: 1) high specific impulse, 2) low engine system weight, 3) small size, 4) do not stress the design technology, and 5) reduce in situ infrastructure requirements are features that would be considered for inclusion in a baseline engine system design.

Table 4-9. Major Engine System Design Parameters Examined

• Chamber Pressure
• Mixture Ratio
• Regen Chamber Bypass
• Turbine Bypass
• Area Ratio
• % Nozzle Length
• Chamber Length
• Injector Type
• Injector Density

Table 4-10. Key Screening Criteria Used

• Specific Impulse
• Engine System Weight
• Size
• Operating Conditions All Within State-of-the-Art Limits
• Effect on In Situ Architecture Infrastructure Requirements

Other design assessments and comparisons were also conducted in this study effort. These included: evaluating, translating nozzle design packaging and its associated weight and performance impact for expander engines, turbine material effects on expander engine cycle operation, and the feasibility of using a common duel fuel turbopump feed system in the baseline tripropellant engine designs.

All the engine system designs considered in this analysis incorporated SOA materials and rocket propulsion system design practices, where appropriate. Table 4-11 summarizes these technology level considerations. Additionally, weight savings and possible gain in performance associated with the use of SOA robust engineering design analysis tools were incorporated in the analysis, and where possible the legacy of a given design assumption is shown.

Table 4-11. Engine System Design Technology Level Considerations

- Use SOA Material Technology Where Appropriate
  - Nozzle and Its Extension
  - Turbopump Turbine
  - Electronics
  - Thrust Mount

- Incorporate SOA Rocket Design Practices
  - Efficient/Stable Injectors/Injection
  - High Chamber Pressure
  - High Chamber Temperature
  - High Heat Flux Nozzle
  - High Turbopump Turbine Inlet Temperatures and Speeds
  - High Pump Discharge Pressure
  - Fast Response, Integrated Controls Available

The baseline engine designs feature a three-section thrust/chamber design shown in Figure 4-4. It uses a Rao nozzle contour (90% length) that incorporates a slotted, cooper regenerative LOX-cooled thrust chamber nozzle section to a downstream area ratio (E) of 6:1, an inconed LOX cooled tube construction segment from E of 6:1 to an E of 25:1 where a radiation cooled carbon-carbon extension is attached. The extension extends to an E of 400:1 or as specified. Some large low pressure expander engine designs incorporate a nozzle extension that translates. Chamber length in the study is defined from the injector to the nozzle throat.

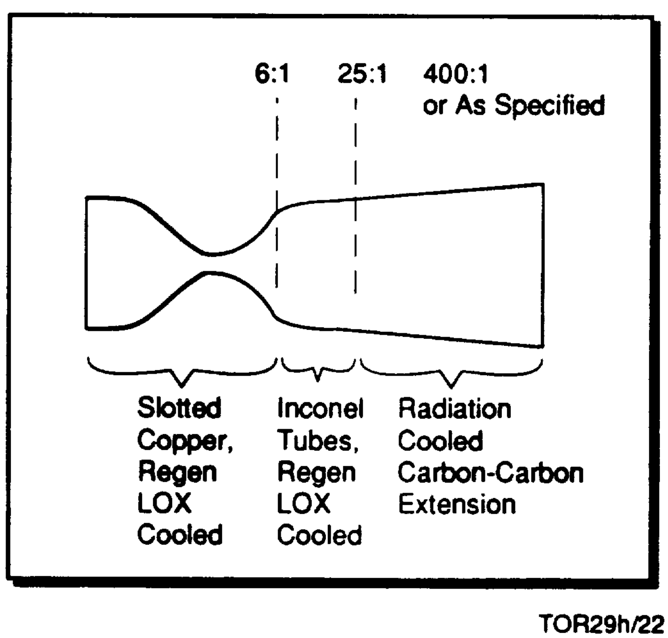


Figure 4-4. Baseline Thrust Chamber Nozzle Design Features Assumed

Other design analysis factors and assumptions are presented in Table 4-12. The LOX/CO and LOX/CH_4 thrust chamber wall temperature limits are based on SOA materials compatibility data discussed in Section 3.0. The LOX/H_2 wall temperature limits are based on SSME experience. The turbopump limits have been well demonstrated by the SSME and the OTV, the technology demonstration engine. Minimum nozzle thickness is determined by quality control uncertainty associated with the manufacturing of a large high area ratio composite nozzle. Another key design analysis assumption is that associated with the impact of engine system weight as a function of engine throttling requirements. The ELES default weight multiplying correlation was assumed, which is shown in Figure 4-5. This correlation is based on past Lunar Excursion Module propulsion system design studies, see Ref. 4-1. Table 4-13 shows the safety factors assumed in the analysis. These safety factors are similar to those used in the SSME design. Thus, reusable, long life design margin is considered inherent in the design analysis.

Table 4-12. Other Key Design Analysis Factors/Assumptions

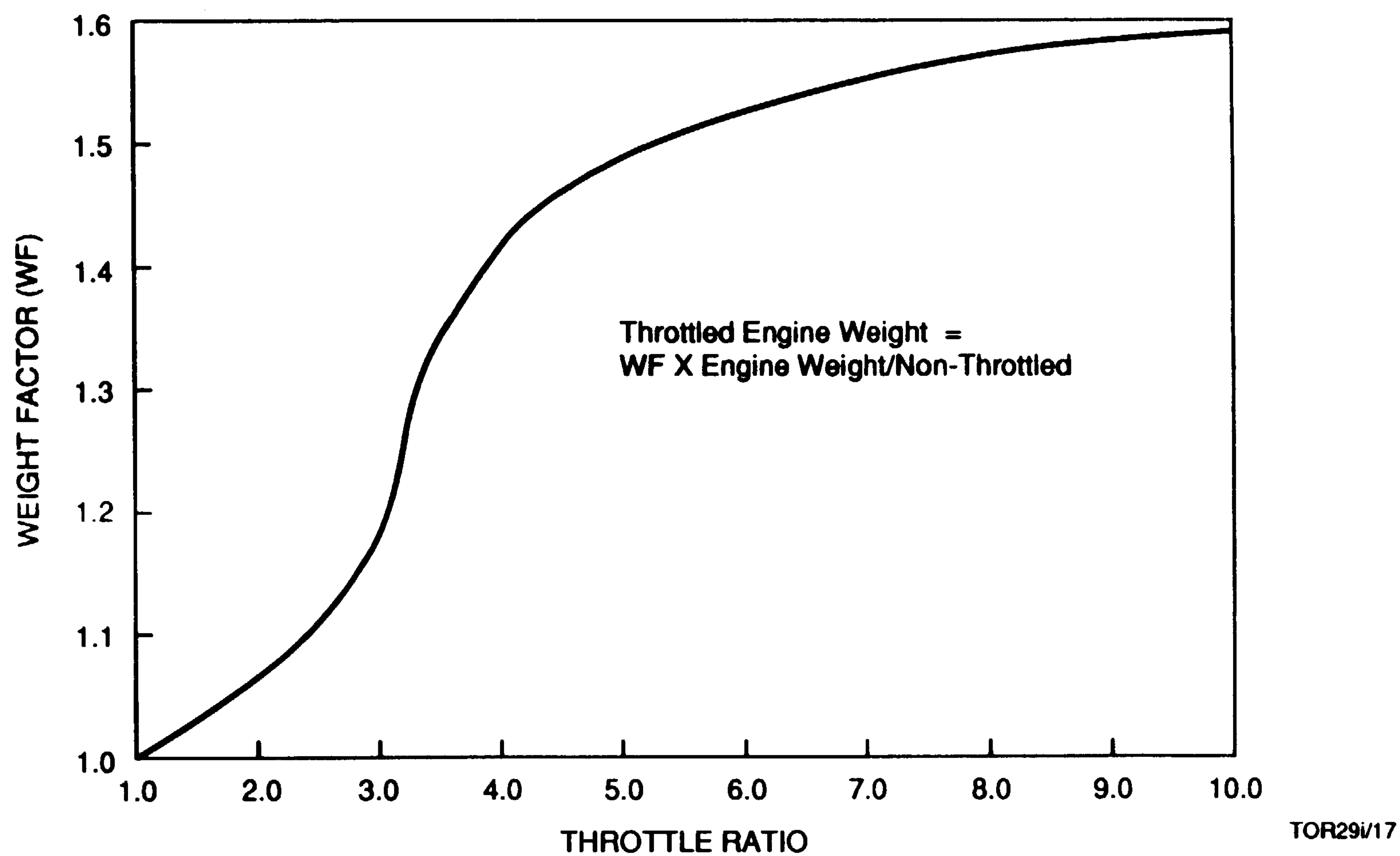
- Thrust Chamber Wall Temperature Limits
  - LOX/CO = 700°K
  - LOX/CH_4 = 778°K
  - LOX/H_2 = 778°K

- Turbopump Limits
  - Turbine Inlet Temperature ≤ 950°K
  - Speed ≤ 60,000 RPM
  - Outlet Pressures ≤ 7,000 psia

- Minimum High Area Ratio Nozzle Extension Exit Thickness = 2.5 mm (0.1 in.)

- Lightweight Carbon-Carbon Nozzle Translation Mechanism Assumed

- Baseline Tank Used for Engine System Analysis
  - 68,050 kg Total Propellant
  - Run Time Range: 220-400 Seconds
  - Diameter: 457 cm
  - Length Range: 560-685 cm



**Based on Past Lunar Excursion Module Studies*

Figure 4-5. Throttling Engine Weight Correlation Assumed
– ELES Default Correlation* –

Table 4-13. Safety Factors Assumed

- All Components, Except Lines:
  - 1.1 of Yield
  - 1.4 of Ultimate
- Lines – 2.0 Ultimate

Major engine component materials and design approaches employed in all the engine designs evaluated are summarized in Table 4-14. All materials and design approaches considered have a strong operational and/or development base legacy. Likewise, the materials selected for each design should be compatible with the propellants and combustion products, as well as the operating conditions to which they are exposed.

Table 4-14. Major Engine Component Materials and Design Approaches Assumed

Component(s)	Material	Design Approach	Comment(s)
Injector	Inconel	• High Density, Co-Axial Design	• Used on SSME • Extensive R&D Base
Thrust Chamber and Upstream Nozzle - ε|attachment = 6:1	Copper Alloy	• High Heat Flux Thin Slotted Wall Construction • LOX Cooled	• Used on SSME • Extensive R&D Base
Nozzle - ε|downstream = 6:1 to ε|downstream = 25:1	Inconel	• Tube Constructed • LOX Cooled	• Used on SSME and Many Other Engines
Nozzle Extension - ε|downstream = 25:1 to ε|downstream = 400:1 or as specified	High Temperature Carbon-Carbon With Oxidation Resistant Coating - Renium or Nirobium Coating Candidates - Translating Nozzle Design, if Specified	• Radiation and/or Film Cooled	• Based on Solid Propulsion, NASP, and R&D Technology Bases
Main Fuel and Oxidizer Valves	Inconel	—	• Material Used in SSME
Low Pressure Fuel and Oxidizer Turbopumps	Inconel	• Bootstrap Boost Pump	• Material Used in SSME
High Pressure Fuel and Oxidizer Turbopumps - Pumps - Turbine - Housing	• Inconel • Monel Alloy (500) • Inconel	• Direct Drive Turbopumps - Axial Turbine - Centrifugal Pump	• Used in SSME • R&D Base and OTV Technology Dev. • Used in SSME
Gas Generator	Inconel	• Uses Multi-Propellants • Low Pressure • Low Mixture Ratio	• Used in SSME
Propellant Lines/ Valves/Supports	Inconel	—	• Used in SSME

The overall engine system trade space evaluation process is displayed in Figure 4-6. In defining a tripropellant engine system, an optimum or near-optimum design would be established first for one bipropellant combination. Then, the other bipropellant combination is analyzed through the fixed engine design to determine its performance and operational characteristics. The expander cycle engine designs were established first; these were followed by the gas generator cycle engine designs. During the analysis, as optimal design parameter(s) or feature(s) were identified for a given propellant combination and design type, they were then baselined for similar engine design concepts.

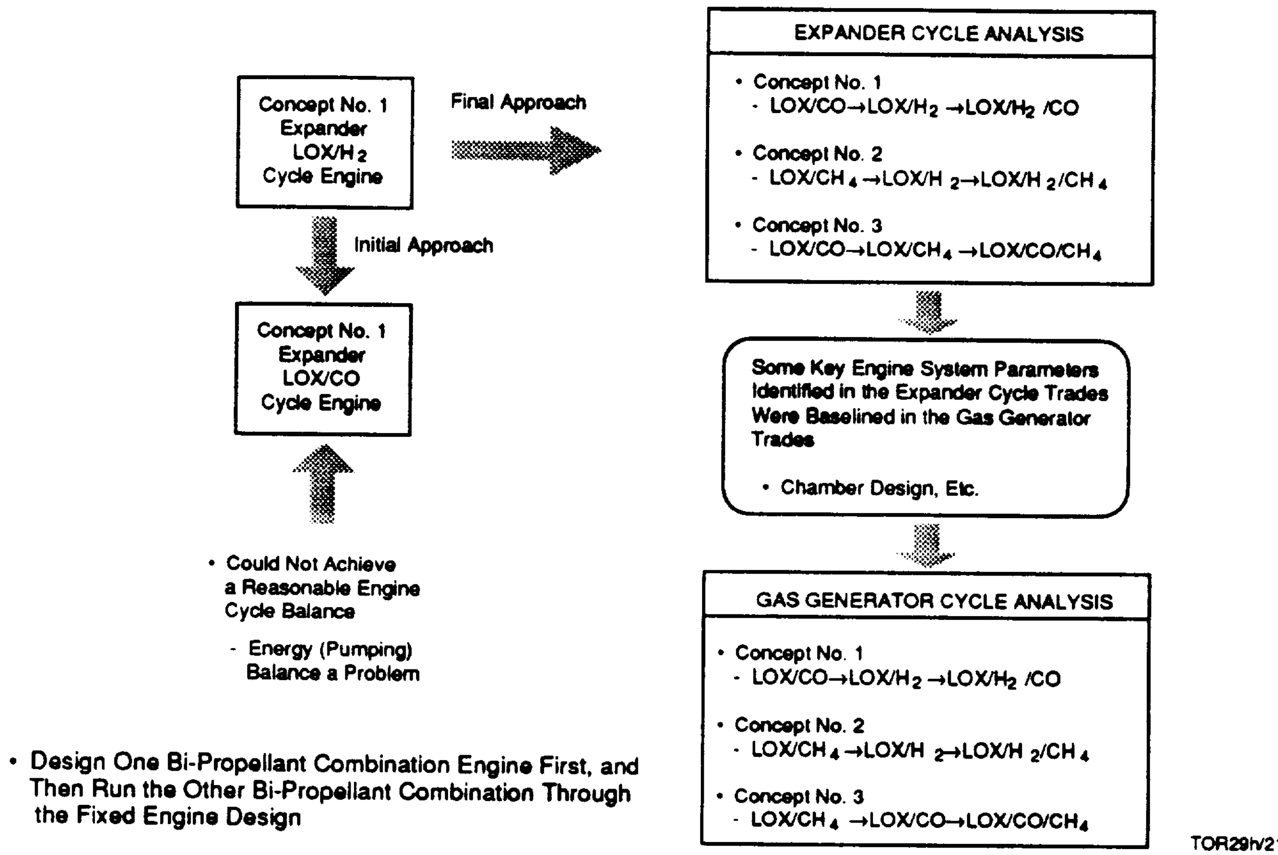


Figure 4-6. Overall Engine System Trade Space Evaluation Process

At the beginning of the analysis it was felt that chamber heat loading and propellant pumping would be the key cycle balance driving factors. As shown in Figure 4-6, the Concept No. 1 expander cycle LOX/H_2 engine design was defined initially. This initial starting attempt addressed heat load issues associated with expander cycle LOX/H_2 engine designs. It was then found for Engine Concept No. 1 that the LOX/CO operation mode could not achieve a reasonable engine cycle balance. It was then determined that propellant pumping requirements drove the operation of such engines. Hence, the higher pumping requirement engine operation mode was designed first. It was also found during the engine system evaluation process that the pumping requirements are much more coupled for the expander cycle engines than for the gas generator cycle engines.

4.2.2 Design Sensitivity Trades

Engine system trades were conducted in accordance with the overall process summarized in Figure 4-6 and the assumptions previously discussed. From these sensitivity trades, optimal or near-optimal design features and operating characteristics for each engine design concept were identified. Based on these results, baseline engine systems were established for each of the three concept categories, which are presented in Section 4.2.3. Detailed engine system sensitivity trade results for each engine system concept are depicted graphically in Appendix C.

In the process of identifying optimal engine system design features, performance, weight, size and operational technology limitations were considered equally. Sound fundamental engineering judgment was also incorporated in the evaluation process.

The initial sensitivity trades were performed on representative Engine Concept No. 1, an expander cycle engine system that operated in a LOX/H_2 propellant combination mode. Key observations and results from this effort were: 1) that a nozzle area ratio greater than 200:1 would be required to achieve the desired performance to support its intended mission, 2) that the use of turbine and chamber regenerative cooling bypass had little effect on engine system performance and weight, and 3) that for the tripropellant in situ engine designs of interest, the heat loading associated with an engine operating in the LOX/H_2 mode at low chamber pressure, $P_c < 3000$ psia, should not be an issue. From these observations it was directed for the reminder of the trade study that: 1) a nozzle area ratio of 400:1 be baselined, 2) further turbine and regenerative bypass trade be omitted, and 3) the tripropellant engines initially be defined by the operating mode that drives pumping requirements (LOX/CO or LOX/CH_4 operating modes), as previously discussed.

After this initial trade assessment effort, detailed trades were then conducted for the candidate expander and gas generator engine concepts, respectively. Appendix C summarizes the results of these key trades. Key engine system design parameters and features identified from these trades are shown in Tables 4-15 and 4-16 for the expander and gas generator engine designs, respectively.

Table 4-15. In Situ Propellant Expander Cycle Engines
– Key Engine Design Parameters and Features, Baselined –

Parameter/Feature	LOX/CO	LOX/CH_4
Chamber Pressure (psia)	550	700
Mixture Ratio	0.55	3.60
Injector Density (Elements/in^2)	10	10
Injector Type (Co-Axial)	3.0	3.0
Turbine Bypass (%)	0.0	0.0
Chamber Regen Bypass (%)	0.0	0.0
Chamber Length (cm)	91.4	66.0
Area Ratio(s)	400:1/165:1	400:1/140:1
Percent Nozzle (%)	90.0	90.0

Table 4-16. In Situ Propellant Gas Generator Cycle Engines
– Key Engine Design Parameters and Features, Baselined –

Parameter/Feature	LOX/CO	LOX/CH_4
Chamber Pressure (psia)	2,000	2,000
Mixture Ratio	0.55	4.0
Chamber Length (cm)	91.4	66.0
Gas Generator Mixture Ratio	0.05	0.4
Area Ratio(s)	400:1	400:1

Both the LOX/CO and LOX/CH_4 expander engine designs operate at low chamber pressures, < 700 psia, and at mixture ratios that produced near-optimum performance. Both engines incorporate well-proven moderate element density, co-axial injector designs. No turbine or chamber regenerative bypass are included in the designs. The chamber length of the LOX/CO engine (91.4 cm) is approximately 30% longer than that associate with the LOX/CH_4 engine, 66.0 cm. Engine system performance, length, weight, and thrust chamber regen cooling pressure drop were considered in the selection of the chamber length of each engine. The baseline nozzle on both engines systems uses 90 percent length Rao contour nozzles which were found to be a good compromise in terms of packaging, weight, and performance.

Due to the low operating pressures associated the expander cycle class of engines, they were found to be somewhat heavy in terms of weight and extremely large. Because of their size, each baseline engine system had two baseline versions — one which incorporated a nozzle area ratio of 400:1 and another which had a nozzle exit diameter limited to 457-cm diameter. The 457-cm diameter is based on the maximum usable diameter of the Space Shuttle's payload bay. As shown in Table 4-15, baseline systems which considered this packaging constraint, translated into nozzle area ratio of either 165:1 or 140:1 for the expander engine systems.

The gas generator cycle engine systems incorporated many of the same features as those associated with the expander engine systems. These operate at substantially higher chamber pressure, Pc=2000 psia, than that characteristic of the expander cycle engines. These higher chamber pressure engines are more compact and do not require truncated or translating nozzle designs. The selection of the gas generator mixture ratio was based on the compromise between overall engine system performance, weight, and turbine inlet temperature.

4.2.3 Baseline Engine Systems

Based on the engine sensitivity trade assessment, just discussed, baseline expander and gas generator engine system designs were established. The baseline expander and gas generator cycle engine designs are summarized in Tables 4-17 and 4-18, respectively. Key overall engine system parameters and features are given by each engine operating mode. As previously noted, each baseline expander cycle engine system comes in two design versions: one for a nozzle area ratio of 400:1 and the other with a specified area ratio, which was previously discussed. The 400:1 nozzle expander cycle engine system design version assumes that a lightweight translating nozzle is used. Note that engine system design Versions C and D, which are bipropellant design derivations of the tripropellant engines that support LEV and/or MEV applications, include only the hardware required to support bipropellant operation. Thus, only one fuel feed system is included in its weight budget compared to two fuel feed systems for the tripropellant engine designs. Likewise, for the lighter LEV and MEV engine system design the support hardware is resized.

Detailed descriptions and data associated with the baseline engine designs are given in Appendix D. Features and descriptions for all of the baseline expander and gas generator engine system designs at full rated power and at throttled (off-design) conditions are presented in Appendix D. Typically, engine operating conditions, chamber/coolant, and chamber/injector design compatibility characteristics are given.

Table 4-17. Baseline Engine Summary – Expander Cycle Engines

Concept No. - Propellant Combination	1-$LO_2/CO/H_2$				2-$LO_2/CH_4/H_2$				3-$LO_2/CH_4/CO$			
Vehicle Types	MTV		LEV or MEV		MTV		LEV or MEV		MTV		LEV or MEV	
Parameters / Version	A	B	C	D	A	B	C	D	A	B	C	D
Propellants	LO_2/H_2	LO_2/CO	LO_2/H_2	LO_2/CO	LO_2/H_2	LO_2/CH_4	LO_2/H_2	LO_2/CH_4	LO_2/CH_4	LO_2/CO	LO_2/CH_4	LO_2/CO
Rated Thrust (Vac.) - lbf	175,000				250,000				175,000			
Throttling Range	5.0:1				2.2:1				6.0:1			
Rated Specific Impulse (Vac.) - sec	470.0/ 457.2*	293.2/ 283.2	470.0/ 457.2	293.2/ 283.2	472.3/ 456.5	389.9/ 373.8	472.3/ 456.5	389.9/ 373.8	387.4/ 374.3	293.2/ 283.2	387.4/ 374.3	293.2/ 283.2
Propellant Flow Rate - kg/sec	168.9/ 173.6	270.7/ 280.2	168.9/ 173.6	270.7/ 280.2	240.0/ 248.4	290.8/ 303.3	240.0/ 284.4	280.8/ 303.3	204.9/ 212.1	270.7/ 280.2	204.9/ 212.1	270.7/ 280.2
Mixture Ratio	6.0	0.55	6.0	0.55	6.0	3.6	6.0	3.6	3.6	0.55	3.6	0.55
Chamber Pressure - psia	585/ 580	550	585/ 580	550	735/ 730	700	735/ 730	700	555/ 550	550	555/ 550	550
Area Ratio	400/165:1				400/140:1				400/165:1			
Weight - kg	4420.1/3050.6		4297.2/ 2922.9	4339.5/ 2966.4	3915.0/2712.5		3807.0/ 2596.9	3725.0/ 2512.0	4515.7/3088.2		4434.4/ 3004.0	4461.0/ 3031.5
Thrust/Weight - lbf/lbm	18.0/26.0		18.5/ 27.2	19.3/ 26.8	29.0/41.8		29.8/ 43.7	30.4/ 45.1	17.6/25.7		17.9/ 26.4	17.8/ 26.2
Dimensions - Length (m)	11.64/7.07				12.15/6.72				11.64/7.07			
- Diameter (m)	6.97/4.57				7.42/4.57				6.97/4.57			

* For Area Ratio of 400:1/For Specified Area Ratio

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Table 4-18. Baseline Engine Summary – Gas Generator Cycle Engines

Concept No. - Propellant Combination	1-$LO_2/CO/H_2$				2-$LO_2/CH_4/H_2$				3-$LO_2/CH_4/CO$			
Vehicle Types	MTV		LEV or MEV		MTV		LEV or MEV		MTV		LEV or MEV	
Parameters / Version	A	B	C	D	A	B	C	D	A	B	C	D
Propellants	LO_2/H_2	LO_2/CO	LO_2/H_2	LO_2/CO	LO_2/H_2	LO_2/CH_4	LO_2/H_2	LO_2/CH_4	LO_2/CH_4	LO_2/CO	LO_2/CH_4	LO_2/CO
Rated Thrust (Vac.) - lbf	175,000				250,000				175,000			
Throttling Range	5.0:1				2.2:1				6.0:1			
Rated Specific Impulse (Vac.) - sec	457.2	289.7	457.2	289.7	463.0	384.7	463.0	384.7	383.3	292.3	383.3	292.3
Propellant Flow Rate - kg/sec	171.0	292.9	171.0	292.9	241.2	317.6	241.2	317.6	221.8	267.4	221.8	267.4
Mixture Ratio	6.0	0.55	6.0	0.55	6.0	4.0	6.0	4.0	4.0	0.55	4.0	0.55
Chamber Pressure - psia	2200	2000	2200	2000	2190	2000	2190	2000	2000	2035	2000	2035
Area Ratio	400:1											
Weight - kg	1922.4		1832.0	1703.3	2249.1		2107.9	1946.5	1940.0		1841.1	1850.8
Thrust/Weight - lbf/lbm	41.3		43.3	46.6	50.4		53.8	58.2	40.9		43.1	42.9
Dimensions - Length (m)	7.02				7.88				6.89			
- Diameter (m)	4.57											

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Baseline engine system thrust-to-weight is compared to other operational, development, conceptual engine designs in Figure 4-7. These engine systems exhibit a substantially lower thrust-to-weight ratio when compared to other engines in their thrust class. Most of these other engine are expendable designs with little or no throttling capability and some operate at higher chamber pressures than those associated with baseline engine designs, and are optimized for ETO operation which may imply low nozzle area ratio designs. The differences in these design features give some insight into their thrust-to-weight disparity. The thrust-to-weight ratio of the baseline engine system is in the same range or a little higher than those associated with lower thrust OTV engine systems. Though somewhat lower in thrust, the OTV engine systems have many similarities with the baseline engine system designs. These similarities include that many of these engines are throttleable and that they are optimized for performance, which implies large-area-ratio nozzle designs. The baseline tripropellant engine designs exhibit lower thrust-to-weight than Aerojet designs because they operate at substantially higher thrust levels and chamber pressures. Likewise, the baseline tripropellant engine designs have low thrust-to-weight because they include the weight of two independent feed systems. The baseline gas generator cycle engine system designs also have a substantially greater thrust-to-weight ratio than those characteristic of the baseline expander cycle engine systems, as shown in Figure 4-7.

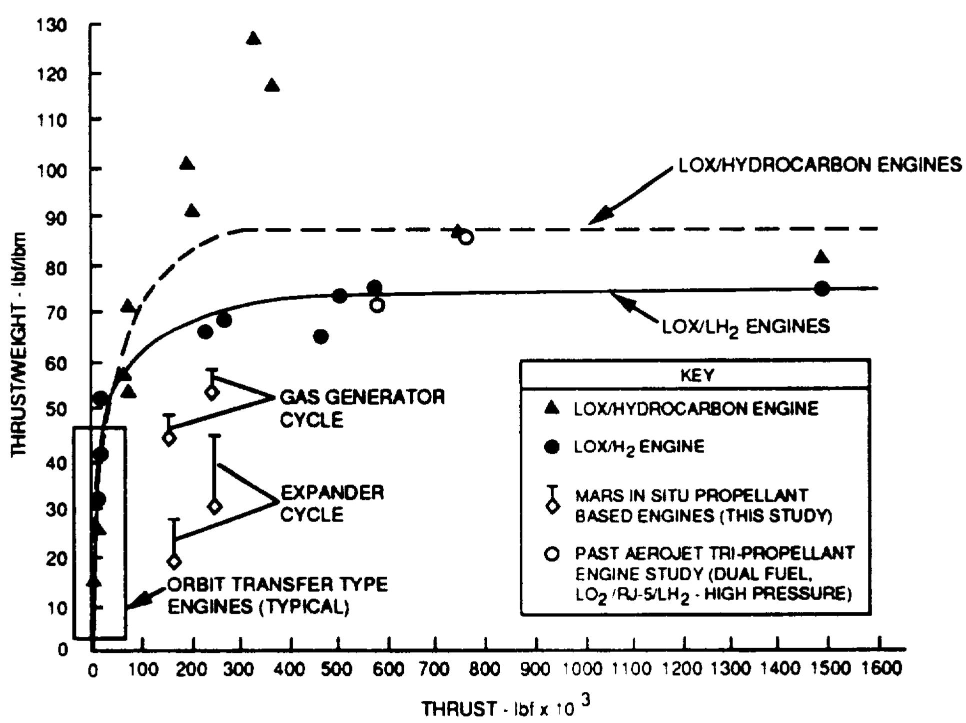


Figure 4-7. Engine Thrust-to-Weight as a Function of Thrust
– Comparison –

4.2.4 Other Engine Design Comparisons

Top-level engineering assessment studies were also performed that addressed some of the key design issues that were identified during the definition and evaluation effort associated with the baseline engine designs. These studies addressed: 1) the use of translating nozzles in terms of packaging and weight for expander cycle engines, 2) the influence of turbopump turbine blade strength on the maximum chamber pressure for expander cycle engines, and 3) the feasibility/ compatibility of multifuel-compatible feed systems for the tripropellant engine systems considered. The following discusses these studies in more detail.

4.2.4.1 Translating Nozzle Assessment

A translating design nozzle concept was studied for the baseline tripropellant expander cycle engine systems to determine its impact on packing and weight. Due to large size of the low chamber pressure expander cycle engines, see Section 4.2.3, packaging the engine into a launch vehicle could be difficult. The nozzle incorporates a lightweight, screw rod translating design which moves its carbon-carbon high area ratio extension into position, where it locks in place. Three screw rods are placed 120 degrees apart about the periphery of the engine. It is made of a lightweight carbon, composite structure. Before the nozzle extension is deployed, it is stowed around the outer portion of the engine.

The results of this assessment are shown in Table 4-19. For each tripropellant engine design, the packaging length is reduced substantially (approximately 29%). The overall engine system weight is increased substantially by incorporating a translating nozzle for each baseline engine design considered. The weight is increased by approximately 45% for the baseline $LOX/CO/H_2$ and $LOX/CO/CH_4$ engine systems while the weight is increased by 76% for the $LOX/CH_4/H_2$ baseline engine system design. As was previously mentioned, it is felt that the packaging is a major issue with the expander cycle baseline engines and a translating nozzle was incorporated in their design.

4.2.4.2 Turbine Blade Strength Assessment

It was observed during engine system sensitivity trades evaluation effort that the selection of the turbopumps turbine blade strength had a major influence on the maximum chamber pressure achievable for the expander cycle engines. These engines designs assumed warm O_2 driven turbopumps to feed the propellants through the engine. Because these engine designs use warm

O_2 to drive their turbopumps, turbine material options are limited because of chemical compatibility considerations. MONEL 500 was selected as the turbine material for all the engine systems considered in this study because it is compatible with warm O_2 and has adequate yield stress (>80,000 psi) at the operating conditions of interest.

Table 4-19. Translating Nozzle Effects in Terms of Packaging and Weight
– Expander Cycle Engines –

Engine Concept Number	Propellants	Stowed Length AR = 400:1 (m)	Total Deployed Length (m)	Engine Weight w/o Translating Nozzle (kg)	Engine Weight w/Translating Nozzle (kg)
1	$LOX/CO/H_2$	8.27	11.64	2963.2	4420.1
2	$LOX/CH_4/H_2$	8.81	12.15	2227.7	3915.0
3	$LOX/CO/CH_4$	8.27	11.64	3058.1	4515.0

This assessment was performed to give some insight into the inherent design margin associated with the selection of MONEL 500 as the turbine blade material. A LOX/CH_4, expander cycle engine design was used in this evaluation that operated at a mixture ratio of 3.6, a thrust level of 250,000 lbf, and incorporates a nozzle area ratio of 400:1. The minimum turbine blade yield stress was varied and the maximum operating chamber pressure was identified. These results are shown in Table 4-20. It is concluded from these results that turbine blade materials with only a minimum yield stress of 40,000 psi can adequately support operation of the baseline engine systems of interest. Hence, the selection of MONEL 500 as the turbine material has a substantial design margin for its intended application in the low chamber pressure baseline engine systems.

Table 4-20. Turbine Blade Strength Influence on Chamber Pressure

Maximum Chamber Pressure (psia)	Minimum Turbine Yield Stress (psi)*
400	30000
700	40000

* Ultimate Stress = 1.20 x Yield Stress in Analysis

4.2.4.3 Common Fuel Turbopump Assessment

This assessment addressed the feasibility of using one single common fuel turbopump (feed system) for the baseline tripropellant engine systems that incorporate two independent fuel systems. If found feasible, such a design approach has the potential to reduce tripropellant engine system weight and increase its simplicity which translates into higher reliability. All baseline engine systems were evaluated in this assessment. It was found that using a baseline engine CH_4 or CO turbopump for pumping H_2 was not possible. This result is not surprising due to the large density difference between the fuels. For the baseline LOX/CH_4/CO gas generator cycle engine, it was found that a single turbopump design could adequately pump both CH_4 and CO. Table 4-21 shows the design and operational characteristics for such an engine over a large thrust level range. All the other baseline engine system designs that incorporated a common fuel turbopump design were found not to be feasible.

4.3 Propellant Tank Design Assessment

Top-level engineering design assessment of candidate propellant tankage for Mars in situ-based propulsion/vehicle system was performed to investigate key design issues and to identify promising design options. This assessment was based on the results of the initial mission requirements discussed in Section 2.0 and used the baseline engine system designs presented in Section 4.2.3. Tankage systems for MTV applications were examined because they showed the potential for a substantial weight savings due to using common propellant tanks through all or some phases of their mission flight profile.

The preliminary design analysis of candidate tank design options was performed using SAIC's ELES program, see Refs. 4-1 through 4-3, and the PSDOC (Protection Structures Design Optimization Code) model, see Ref. 4-4, which defined meteoroid protection system requirements. Trade studies were conducted that addressed: 1) in situ multipropellant tank commonality/compatibility issues such as sizing, materials compatibility and pressurization, 2) boiloff and 3) meteoroid protection system requirements and design. Results from these trades were compared to comparable SOA LOX/H_2 tank systems. The design assumptions, considerations, and key results associated with this assessment are presented in the following sections.

4.3.1 Design Requirements/Considerations

In addition to tank size, which is a strong function of ΔV for a given mission segment, other tankage system requirements must be characterized to accurately design a propellant tank system. These other key requirements are the propellant exposure (storage) time in space, the thermal environment, the space debris environment/protection requirements, acceleration loading, and geometric envelope constraints, which are usually dictated by the ETO launch system.

Table 4-21. $LO_2/CH_4/CO$ Gas Generator Engine Common Fuel Turbopump Design Characteristics

	COMMON TURBOPUMPS												
	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		58%		58%		16.60%		16.60%	
	Propellant Combination=	LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO	
	Cycle Type	Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	FEATURES												
Chamber	Copper Chamber weight	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg
	- Includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CH4		LO2/CO		LO2/CH4		LO2/CO		LO2/CH4		LO2/CO	
	Mixture Ratio	4		0.55		4		0.55		4		0.55	
	Chamber Diameter	36.4	cm	36.4	cm	36.4	cm	36.4	cm	36.4	cm	36.4	cm
	Chamber Length	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm
	Chamber Temperature	3711	deg K	3629	deg K	3597	deg K	3518	deg K	3391	deg K	3314	deg K
	Chamber Pressure	2000	psia	2000	psia	1185	psia	1190	psia	345	psia	345	psia
	Inconel Injector weight	145.5	kg	145.5	kg	145.5	kg	145.5	kg	145.5	kg	145.5	kg
	Propellant Mass Flow	221.8	kg/s	267.3	kg/s	120.0	kg/s	157.9	kg/s	35.1	kg/s	46.3	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg
	*Nozzle - Inconel, regen tubes	86.6	kg	86.6	kg	86.6	kg	86.6	kg	86.6	kg	86.6	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	199.6	kg	199.6	kg	199.6	kg	199.6	kg	199.6	kg	199.6	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm
	Exit Diameter	364.5	cm	364.5	cm	364.5	cm	364.5	cm	364.5	cm	364.5	cm
	Deployed Nozzle Length	508.3	cm	508.3	cm	508.3	cm	508.3	cm	508.3	cm	508.3	cm
	Delivered Vacuum Isp	383.34	sec	292.48	sec	379.78	sec	288.52	sec	368.78	sec	279.71	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	105000	lbf	105000	lbf	35000	lbf	35000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	13.6	kg	13.6	kg	13.6	kg	13.6	kg	13.6	kg	13.6	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3240.0	psia	3224.0	psia	1909.2	psia	1909.7	psia	546.7	psia	538.5	psia
	Pump Speed	35484	rpm	28023	rpm	26928	rpm	21093	rpm	13676	rpm	10655	rpm
	Pump Diameter	16.6	cm	16.6	cm	16.6	cm	16.6	cm	16.6	cm	16.6	cm
	Pump Horsepower	4606	HP	7521.65	HP	1624.4	HP	2714.2	HP	148.3	HP	244.5	HP
	Pump Efficiency	0.716		0.821		0.688		0.796		0.621		0.73	

Table 4-21. $LO_2/CH_4/CO$ Gas Generator Engine Common Fuel Turbopump Design Characteristics (Cont.)

Main Oxidizer	Main Oxidizer Pump weight	6.1	kg	6.1	kg	6.1	kg	6.1	kg	6.1	kg	6.1	kg
Pump	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3341.3	psia	3092.9	psia	1885.7	psia	1802.9	psia	506.7	psia	498.3	psia
	Pump Speed	34760	rpm	32140	rpm	25678	rpm	24017	rpm	12693	rpm	12026	rpm
	Pump Diameter	11.4	cm	11.4	cm	11.4	cm	11.4	cm	11.4	cm	11.4	cm
	Pump Horsepower	4561	HP	2559.6	HP	1552.4	HP	911.1	HP	131.5	HP	80.8	HP
	Pump Efficiency	0.805		0.766		0.789		0.741		0.738		0.68	
Fuel Turbine	Fuel Turbine weight	16.1	kg	16.1	kg	16.1	kg	16.1	kg	16.1	kg	16.1	kg
	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	35484	rpm	28023	rpm	26928	rpm	21093	rpm	13676	rpm	10655	rpm
	Turbine Efficiency	0.7		0.708		0.708		0.675		0.493		0.418	
	Turbine Diameter	17.1	cm	17.1	cm	17.1	cm	17.1	cm	17.1	cm	17.1	cm
Oxidizer	Oxidizer Turbine weight	16.8	kg	16.8	kg	16.8	kg	16.8	kg	16.8	kg	16.8	kg
Turbine	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	34760	rpm	32140	rpm	25678	rpm	24017	rpm	12693	rpm	12026	rpm
	Turbine Efficiency	0.7		0.665		0.706		0.705		0.471		0.474	
	Turbine Diameter	17.5	cm	17.5	cm	17.5	cm	17.5	cm	17.5	cm	17.5	cm
Fuel Boost Pump	Fuel Boost Pump weight	8.6	kg	8.6	kg	8.6	kg	8.6	kg	8.6	kg	8.6	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	485.2	psia	344.2	psia	169.3	psia	204.4	psia	33.8	psia	58.6	psia
	Pump Speed	13838	rpm	16498	rpm	8917	rpm	10994	rpm	10694	rpm	7652	rpm
	Pump Diameter	11.8	cm	11.8	cm	11.8	cm	11.8	cm	11.8	cm	11.8	cm
	Pump Horsepower	218	HP	461.4	HP	73.9	HP	145.8	HP	6	HP	11.9	HP
	Pump Efficiency	0.772		0.683		0.772		0.757		0.777		0.764	
Oxidizer Boost	Fuel Boost Pump weight	16.7	kg	16.7	kg	16.7	kg	16.7	kg	16.7	kg	16.7	kg
Pump	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	558	psia	231.1	psia	237.4	psia	135.8	psia	36.9	psia	38.2	psia
	Pump Speed	5953	rpm	4240	rpm	3840	rpm	2817	rpm	4406	rpm	4369	rpm
	Pump Diameter	16.1	cm	16.1	cm	16.1	cm	16.1	cm	16.1	cm	16.1	cm
	Pump Horsepower	231.8	HP	125.3	HP	77.3	HP	45.8	HP	6.1	HP	3.5	HP
	Pump Efficiency	0.809		0.798		0.809		0.753		0.816		0.809	

Table 4-21. $LO_2/CH_4/CO$ Gas Generator Engine Common Fuel Turbopump Design Characteristics (Cont.)

Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	55.6	kg	55.6	kg	55.6	kg	55.6	kg	55.6	kg	55.6	kg
	Engine Lines	98.2	kg	98.2	kg	98.2	kg	98.2	kg	98.2	kg	98.2	kg
	Main Valve	21.4	kg	21.4	kg	21.4	kg	21.4	kg	21.4	kg	21.4	kg
	Gimbal System	30.5	kg	30.5	kg	30.5	kg	30.5	kg	30.5	kg	30.5	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	121.9	kg	121.9	kg	121.9	kg	121.9	kg	121.9	kg	121.9	kg
	Gas Generator	15.7	kg	15.7	kg	15.7	kg	15.7	kg	15.7	kg	15.7	kg
	Gas Generator Features:												
	*Mixture Ratio	0.4		0.05		0.4		0.05		0.4		0.05	
	*Temperature	924.3	deg K	562.8	deg K	924.3	deg K	562.8	deg K	924.3	deg K	562.8	deg K
	*Pressure	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia
	*Mass Flow Rate	18.0	kg/s	22.7	kg/s	6.2	kg/s	7.9	kg/s	0.5	kg/s	0.7	kg/s
Subtotal	Engine Weight	1294.4	kg		kg	1294.4	kg		kg	1294.4	kg		kg
	Throttling Factor Weight	510.6	kg		kg	510.6	kg		kg	510.6	kg		kg
	Margin (2%)	36.1	kg		kg	36.1	kg		kg	36.1	kg		kg
Total Engine	Weight	1841.1	kg		kg	1841.1	kg		kg	1841.1	kg		kg
System	Length	688.6	cm		cm	688.6	cm		cm	688.6	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

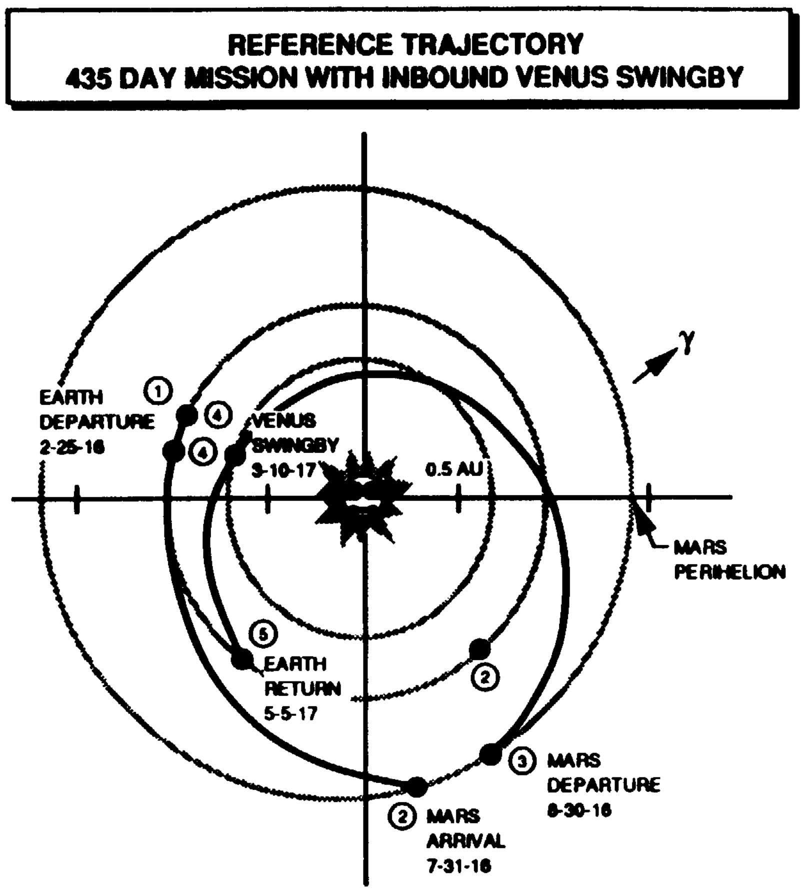
Typically, the propellant exposure time and thermal environment (distance from the sun) greatly influence the boiloff characteristics/requirements of propellant tankage system. For this study, a typical 435-day Mars mission was used in the assessment which is shown in Figure 4-8. Due to the nature of this mission the propellant tankage system must be able to survive a dynamic space debris environment. Key tankage space debris conditions/design considerations by mission segment are summarized in Table 4-22. General tankage systems features and requirements were identified and are shown in Table 4-23.

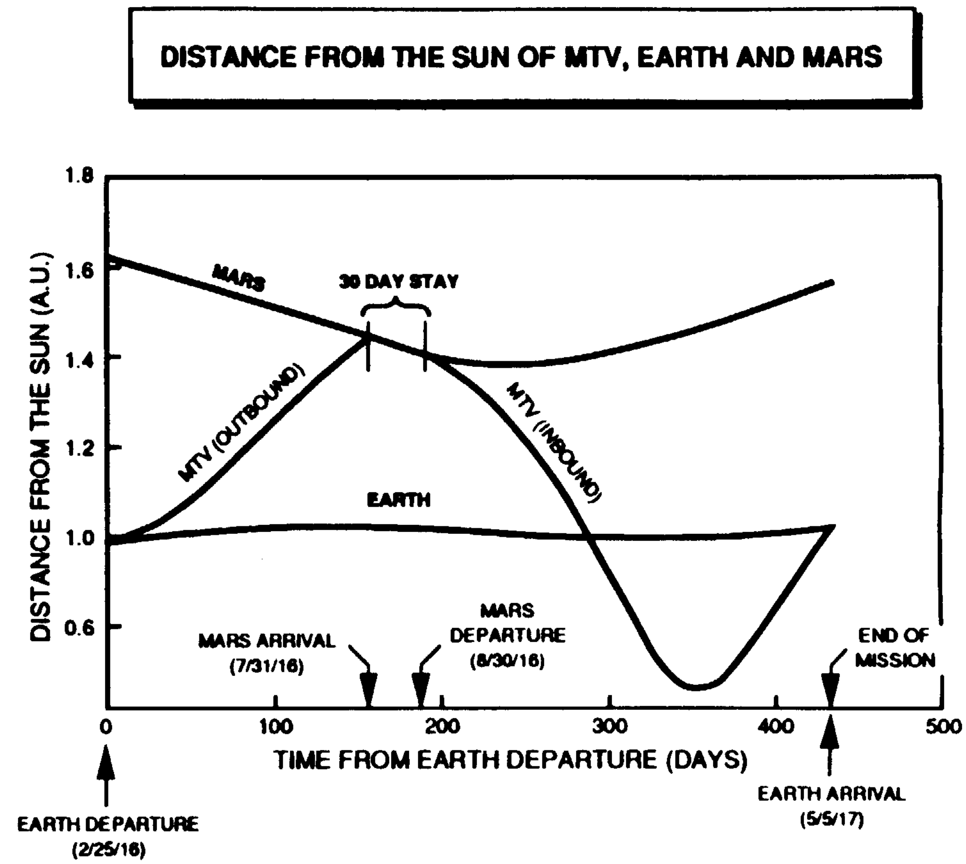
Table 4-22. Key Space Debris Tank Design Considerations by Mission Segment

Mission Segment	Conditions/Design Considerations
LEO	Earth-Orbital Space Debris, Cometary Meteoroids, Earth Shielding, Gravitational Defocusing, Altitude, Inclination, Configuration
Transit	Asteroidal and Cometary Meteoroids, Trajectory and Schedule, Configuration
Mars Orbit	Asteroidal and Cometary Meteoroids, Mars/Phobos/Deimos Shielding, Gravitational Defocussing, Altitude, Configuration
Martian Surface Excursion Supply/Surface Vehicles	Asteroidal and Cometary Meteoroids, Surviving Particle Mass to Surface, Primary Impacts on Surface, Secondary Ejecta, Configuration

To support the Mars transportation systems considered in this study, an ETO launch system based on a growth version of the Advanced Launch System, discussed in Ref. 1-1, was assumed. Figure 4-9 shows this ETO launch system with its key payload performance and geometric features listed.

Additionally, an assessment of tank system sizing was performed by scenario type and mission segment to identify common propellant tank volumes. This was based on the initial mission requirements, see Section 2.0, as previously discussed. Table 4-24 shows the tank system sizing assessment results. Based on these results, the other design considerations and issues, and the overall assessment goal to examine candidate tank designs that best display design differences and issues, propellant tank designs for the following mission scenarios were evaluated. They are: 1) Scenario No. 2 - Lunar LOX, Mars LOX/CO, and 2) Scenario No. 4 - Lunar LOX/CH_4, Mars LOX/CO. In addition to these, tank system designs associated with the all Earth LOX/H_2 based system (Scenario No. 1) were also evaluated so that the in situ propellant-based tank designs could be compared.





- Key Tankage System Requirements:
  - $\Delta V \Rightarrow$ Tank Size
  - Propellant Exposure (Storage) Time to Space } Boil-Off Characteristics
  - Thermal Environment
  - Space Debris Environment/Protection Requirements
  - Acceleration Loading

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Figure 4-8. Tankage Requirements Are Based on a 435-Day Mars Reference Mission

Table 4-23. General Tankage System Features/Requirements
– Assumes Representative Mission Scenario –

VEHICLE/ SEGMENT / MISSION APPROACH	TRANSFER				EXCURSION			
	EXP. STAGE	MTV			MEV		LEV	
	LEO→LLO	LEO→LMO	LLO→LMO	LMO→LEO	ASCENT	DESCENT	ASCENT	DESCENT
NO. 1 LOX/H_2 BASELINE CASE		TMI • ET[1] ≤ 7 Days • D[2] ~ 1 AU • MPR[3] = L* • BOC[4] = L-M* MOC • ET ~ 300 Days • D[2] ~ 1 to 1.5 AU • MPR = H • BOC = H		TEI • ET ~ 340 Days • D ~ 1 to 1.5 AU • MPR = H • BOC = H EOC • ET ≤ 600 Days • D ~ 1 to 1.5 AU • MPR = H • BOC = H	• ET ~ 380 Days • D ~ 1 to 1.5 AU • MPR = H • BOC = H	• ET ~ 340 Days • D ~ 1 to 1.5 AU • MPR = H • BOC = H		
NOS. 2, 3, 4 AND 5	TLI • ET ~ 3 Days • D ~ 1 AU • MPR = L • BOC = L LOI • ET ~ 4 Days • D ~ 1 AU • MPR = L • BOC = L		TMI • ET ≤ 14 Days • D ~ 1 AU • MPR = H[a] • BOC = L-M MOC • ET - ~ 300 Days + - < 300 Days ++ • D ~ 1 to 1.5 AU • MPR = H • BOC - H+, X - L - M++	TEI • ET ≤ 14 Days • D ~ 1 to 1.5 AU • MPR = L • BOC = L EOC • ET ~ 300 Days • D ~ 1.5 to 1 AU • MPR = H • BOC = L - M	• ET ~ 3 Days • D ~ 1.5 AU • MPR = L • BOC = L	• ET ~ 10 Days • D ~ 1.5 AU • MPR = L • BOC = L	• ET ~ 3 Days • D ~ 1 AU • MPR = L • BOC = L	• ET ~ 10 Days • D ~ 1 AU • MPR = L • BOC = L
NOS. 6 AND 7		TMI • ET ≤ 7 Days • D ~ 1 AU • MPR = H[a] • BOC = L - M MOC • ET ~ 300 Days • D ~ 1 to 1.5 Days • MPR = H • BOC = H		TEI • ET ≤ 14 Days • D ~ 1.5 AU • MPR = L • BOC = L EOC • ET ~ 300 Days • D ~ 1.5 to 1 AU • MPR = H • BOC = L - M	• ET ~ 3 Days • D ~ 1.5 AU • MPR = L • BOC = L	• ET ~ 10 Days • D ~ 1.5 AU • MPR = L • BOC = L		

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[1] ET = Exposure Time of Propellant in Tank to Space, [2] Distance From Sun (in Astronautical Units (AU)), [3] MPR = Metorite Protection Requirement,
[4] BOC = Boil-Off Characteristic, H = High, M = Medium and L = Low, + Mission Case No. 2 and 3, ++ Mission Cases No. 4 and 5,
* H_2 Tanks Drive Requirements, [a] Driven By TEI Reuse Requirement

Table 4-24. Initial Tank System Sizing Results

VEHICLE/ MISSION PHASE	SCENARIO CASE / TANK TYPE	No. 1 Baseline Earth LOX/H_2	No. 2 Lunar LOX Mars LOX/CO	No. 3 Lunar LOX Mars LOX/CH_4	No. 4 Lunar LOX/CH_4 Mars LOX/CO	No. 5 Lunar LOX/CH_4 Mars LOX/CH_4	No. 6 Earth LOX/H_2 Mars LOX/CO	No. 7 Earth LOX/H_2 Mars LOX/CH_4
BOOSTER STAGE								
TLI (2)*	OXIDIZER	--	LOX/3x14.88/100**	LOX/3x23.72/163	LOX/3x10.3/68	LOX/3x17.26/117	---	---
	FUEL	--	H_2/7.6x7.69/267	H_2/7.6x11.36/434	H_2/7.6x5.78/181	H_2/7.6x8.68/312	---	---
LOI (2)	OXIDIZER	--	LOX/3x3.38/19	LOX/3x5.04/31	LOX/3x2.53/13	LOX/3x3.57/20	---	---
	FUEL	--	H_2/4.6x4.12/51	H_2/4.6x6/82	H_2/4.53x3.2/34	H_2/4.6x4.34/54	---	---
MTV								
TMI (2)	OXIDIZER	LOX/3x46.9/327	LOX/3x6.89/44	LOX/3x10.74/71	LOX/3x9.17/60	LOX/3x14.4/97	LOX/3x17.65/193	LOX/3x28.08/327
	FUEL	H_2/7.6x21/872	H_2/4.6x8.1/117	H_2/4.6x12.47/189	CH_4/4.6x3.83/46	CH_4/4.6x5.53/74	H_2/7.6x8.92/516	H_2/7.6x3.17/872
MOC (2)	OXIDIZER	LOX/3x13.5/91	LOX/3x5.46/34	LOX/3x8.42/55	LOX/3x6.42/40	LOX/3x9.89/65	LOX/3x5.51/55	LOX/3x8.45/91
	FUEL	H_2/4.6x15.6/242	H_2/4.6x6.48/96	H_2/4.6x9.84/146	CH_4/4.33x3.09/31	CH_4/4.6.x4.06/50	H_2/4.6x6.54/146	H_2/4.6x9.87/242
TEI (2)	OXIDIZER	LOX/3x5.25/32	LOX/3x7.93/51	LOX/3x7.38/47	LOX/3x7.88/51	LOX/3x7.22/46	LOX/3x7.91/46	LOX/3x7.20/32
	FUEL	H_2/4.6x6.24/86	CO/4.6x8.47/123	CH_4/4.6x3.25/36	CO/4.6x8.42/122	CH_4/4.56x3.23/35	CO/4.6x8.45/35	CH_4/4.56x3.22/86
EOC (1)	OXIDIZER	LOX/3x7.65/49	LOX/3x7.75/50	LOX/3x4.41/62	LOX/3x7.70/49	LOX/3x9.19/60	LOX/3x7.72/59	LOX/3x9.11/49
	FUEL	H_2/4.6x8.96/131	CO/4.6x8.29/120	CH_4/4.6x3.91/47	CO/4.6x8.24/119	CH_4/4.6x3.24/46	CO/4.6x8.26/45	CH_4/4.6x3.81/131
LEV	OXIDIZER	--	LOX/7.02x4.97/128	LOX/8.25x5.83/208	LOX/7.68x5.43/168	LOX/9.01x6.37/271	---	---
	FUEL	--	H_2/9.74x6.89/342	H_2/10x9.42/555	CH_4/7.02x4.46/128	CH_4/8.23x5.82/207	---	---
MEV	OXIDIZER	LOX/3x5.67/35	LOX/10x15.46/1029	LOX/10x8.3/467	LOX/10x15.38/1023	LOX/10x8.16/456	LOX/10x15.42/1026	LOX/10x8.13/35
	FUEL	H_2/4x839/94	CO/10x33.87/2475	CH_4/9.87x6.98/356	CO/10x33.68/2460	CH_4/9.79x6.92/348	CO/10x33.76/2467	CH_4/9.78x6.91/94

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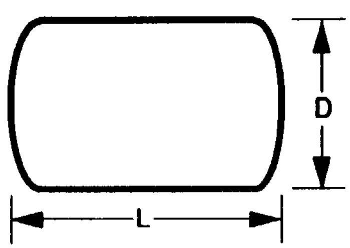
* *Number of Tanks Per Mission Phase for Each Propellant*

** *(Propellant Type/Tank Dimensions (Dx L) in Meters/Tank Volume in Cubic Meters)*



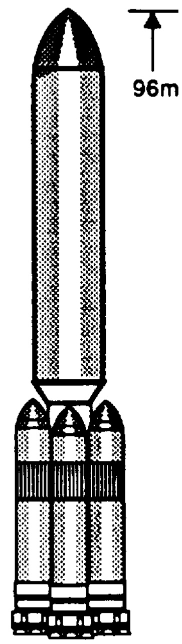


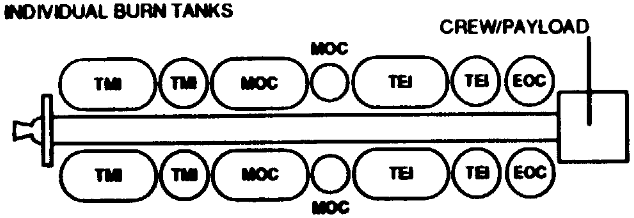
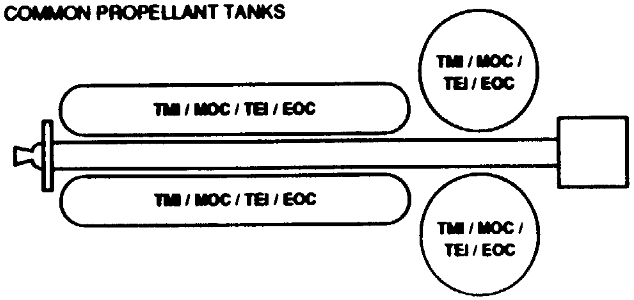
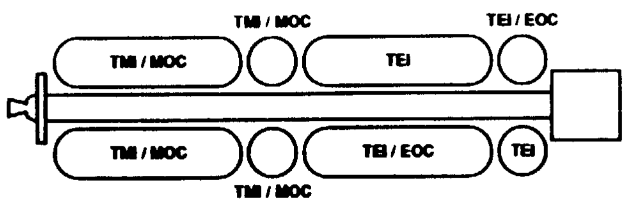
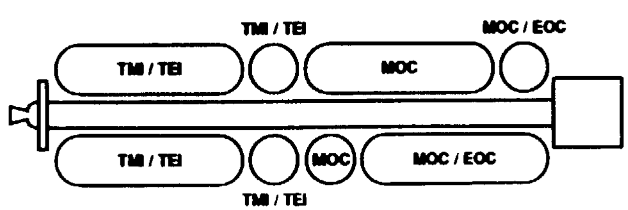
Figure 4-9. Growth Version of the Advanced Launch System

Scenario No. 2 was selected because it addressed the influence of employing the in situ propellant CO on the tank design compared to a conventional Earth H_2 tank design. The potential of ISPP, reduced boiloff, and cryogenic (LH_2)/storable (CO) propellant compatibility on MTV tank design were key reasons to examine this mission scenario. Scenario No. 4 demonstrates a tankage system that uses only in situ propellants.

For these scenarios, MTV vehicle tankage systems were selected for the design assessment because it was felt that such systems had the highest potential to reduce weight over other mission segment vehicles (LTVs and MTVs) by employing common propellant tanks. MTV vehicle tankage configuration strategies considered in the assessment were: 1) individual burn tanks, 2) common propellant tanks, 3) mission segmented common tanks, and 4) common/mission segmented propellant tanks. These configuration strategies are summarized in Table 4-25.

Other tank system design approaches considered in the assessment were: identify tank system design which made the maximum use of LOX tankage throughout the mission; modular tank sizing; and performing a complete change out of tanks at the Moon and/or Mars. For the latter

Table 4-25. MTV Tank Configuration Strategy Options Summary

NO.	CONCEPT	FEATURES	ADVANTAGES	DISADVANTAGES
1	INDIVIDUAL BURN TANKS 	• Vehicle Propellants Are Distributed Into Individual Tanks for Each Mission (Propulsion Burn) Phase • Appropriate Tanks Are Jettisoned After Each Mission Phase Is Complete	• Highly Efficient in Terms of Overall Vehicle Performance for Vehicles That Use All Their Propellants From Earth • Optimal Tank Sizing Possible • No Tank Design Compatibility Issues	• Many Tanks of Possibly Different Sizes • Extensive Tankage Manifolding Required • Vehicles That Use In Situ Propellants Are Required to Carry Empty Tanks During the Early Phases of the Mission - Possible Weight Penalty
2	COMMON PROPELLANT TANKS 	• All Propellants Are Stored in Common Tanks for All Mission Phases • No Staging of Tanks Takes Place	• Minimum Number of Tanks • Maximum Use of Tanks for Vehicle That Use In Situ Propellants	• Tank Sizing Could Be Difficult • Possible Tank Design Compatibility Issues • Tanks Only Partially Full Over Most Phases of Flight - Carry Around Large Partially Full/Near-Empty Tanks - - Weight/Sizing Issue • Extremely Large Tanks for Vehicles That Do Not Use In Situ Propellants
3	MISSION SEGMENTED COMMON TANKS 	• Vehicle Propellants Are Distributed into Common Tank Sets by Mars Transit and Earth Transit Mission Segments • After Each Major Mission Segment Appropriate Tanks Are Jettisoned	• Reduces Number of Tanks • Potential Weight Saving for Vehicles That Do Not Use In Situ Propellants • Staging Mars Transit Tanks Could Increase Overall Vehicle Performance • No Tank Design Compatibility Issues	• Vehicles That Use In Situ Propellants Are Required to Carry Empty Tanks During the Early Phases of the Mission • Does Not Use Common Propellant Tanks - No Tank System Weight Savings Foreseen for Using In Situ Propellants
4	COMMON/MISSION SEGMENTED PROPELLANT TANKS 	• Vehicle Uses Common Mars and Earth Injection and Orbit Capture Tank Sets • Appropriate Tanks Are Jettisoned After MOC and TEI	• Reduces Number of Tanks • Potential for Substantial Weight Savings for Vehicles That Use In Situ Propellants - Modest Weight Saving Fore-seen for Vehicles That Do Not Use In Situ Propellants • Better Compatibility In Terms of Tank Sizing • Staging Tanks Could Increase Overall Vehicle Performance	• Empty TMI and TEI Tanks Carried to Mars for In Situ Propellant-Based Vehicles • Possible Tank Design Compatibility Issues

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approach tank production from in situ materials would be extremely attractive because tanks associated with MTV return propellant would not have to be reused or carried from Earth. This approach would have a major impact on in situ material production infrastructure requirements. A major tradeoff assessment would be required to quantify the impact of these requirements as compared to the life cycle saving possible for the MTV transportation system.

Inflatable propellant tanks, shown in Figure 4-10, may also be another attractive option to store in situ propellants. Weight savings may be possible with such a tankage concept because of its reduced susceptibility to meteoroid penetrations while in its stored, folded position during a portion of the flight. There are many technology issues associated with such a concept. An example of such an issue is the chemical compatibility of a highly flexible material with the propellants at the operating conditions of interest.

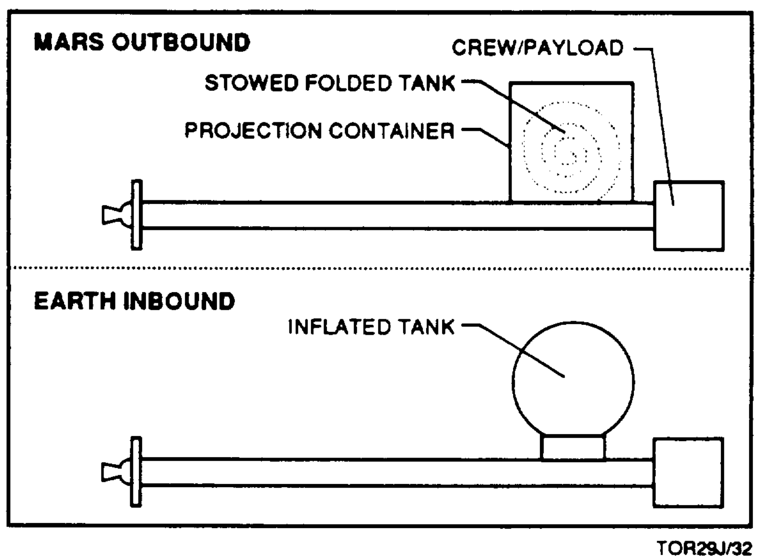

Figure 4-10. Inflatable Tanks May Be Attractive to Store In Situ Propellants

4.3.2 Analysis Approach and Results

Tank design analysis was performed using the ELES design program, as previously discussed. The ELES program characterized the tank design in terms of its boiloff characteristics, but meteoroid shield protection system design analysis could not be performed. The tank meteoroid shield design evaluation was performed using SAIC's PSDOC which was recently developed for NASA MSFC, see Ref. 4-4.

The PSDOC model incorporates probabilistic space environment debris characteristics that includes deterministic hypersonic impact predictor models. It models many of the key meteoroid protection factors that drive the design of a protection system. These factors include: the space debris environment; spacecraft operational period; spacecraft exposure area and orientation; and mission altitude and inclination. In this evaluation a 7.8 g/cm^3 average debris mass was assumed, which is typical of a meteoroid with high iron content. This is the typical asteroid/meteoroid debris environment associated with the transit to and from Mars and its surface. A bumper shield meteoroid protection system was assumed in the evaluation. Figure 4-11 shows this concept and the basic tank geometry modeled. Candidate meteoroid impact shielding materials were also identified and assessed. This assessment is summarized in Table 4-26. Aluminum alloys were baselined in the evaluation because of their well defined properties. Though some of the other material options showed potential to produce a weight savings, more impact and space environment compatibility characterization testing is required for these candidates.

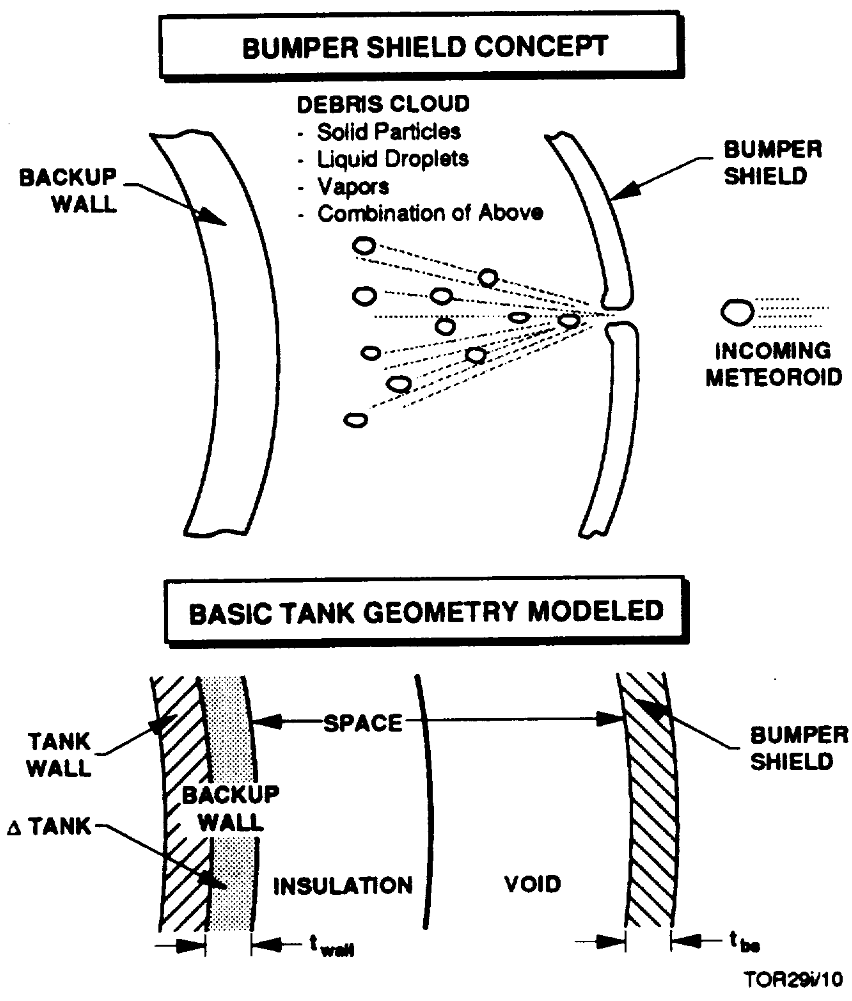

Figure 4-11. The Tank Meteoroid Shield Penetration Concept Evaluated

Table 4-26. Impact Shielding Materials/Options Considerations

MATERIAL	PROS	CONS
Aluminum Alloys* (e.g. 6061-T6, 2219-T87)	Well-Known, Well-Tested, Good All-Around Properties	Wide Variance in Impact Resistance Among Alloys, May Not Be Optimal
Titanium Alloys	Well-Known, Good Properties, Some Alloys Appear Superior to Best Aluminum Alloys	Not as Well-Tested for Impacts. Potentially Wide Variances
Metal Matrix Composites (e.g. Graphite Aluminum)	Greater Flexibility for Tailoring, Potentially Weight Efficient	Not Well-Studied for Impacts. Potential Problems for Other Space Environments
Graphite Epoxies	Greater Flexibility for Tailoring, Potentially Weight Efficient	Not Very Well-Studied. Potential Problems for Other Environments, Particularly for Epoxy Materials
Ceramic Composites	Well-Tested for DoD Applications, Good Impact Resistance	Potential Weight Problems

**Selected for Initial Functional Screening Analysis*

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The evaluation parametrically characterized the baseline meteoroid shield design concept in terms of: 1) mission duration, 2) tank size (surface area) and 3) probability of no penetration (P_{RF}). The results of this evaluation are shown in Figures 4-12 and 4-13. As shown in Figure 4-12, the P_{RF} for deep space missions will likely be greater than 0.980 because inspection, maintenance, and repairs will be unlikely for such missions. For the results displayed in Figure 4-13 P_{RF}=0.990 was assumed. The meteoroid protection system weights for the tank designs were extrapolated from these results.

The overall analysis approach used to assess common tank designs is presented in Figure 4-14. The general tank design features assessed in the analysis are summarized in Table 4-27. The tank designs examined in this assessment are displayed in Table 4-28. By evaluating these tank designs for a given mission scenario and mission segment, a large number of tank design comparisons can be made. These tank design comparisons are presented in Table 4-29. A large number of sizing compatibility and technology options are addressed in this evaluation trade space.

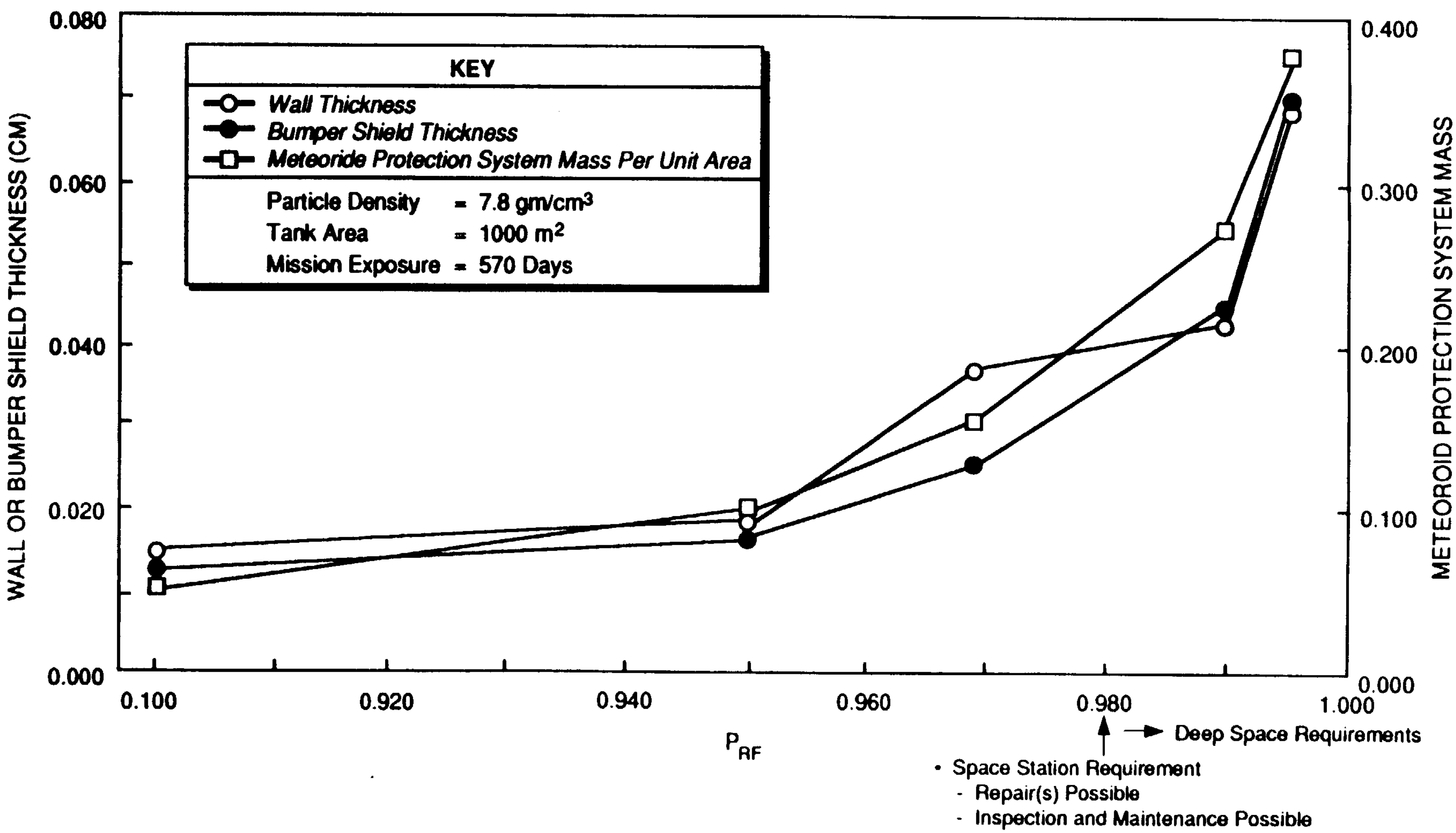

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Figure 4-12. Typical Tank Meteoroid Protection System Characteristics as a Function of Probability for No Tank Penetration (P_{RF})

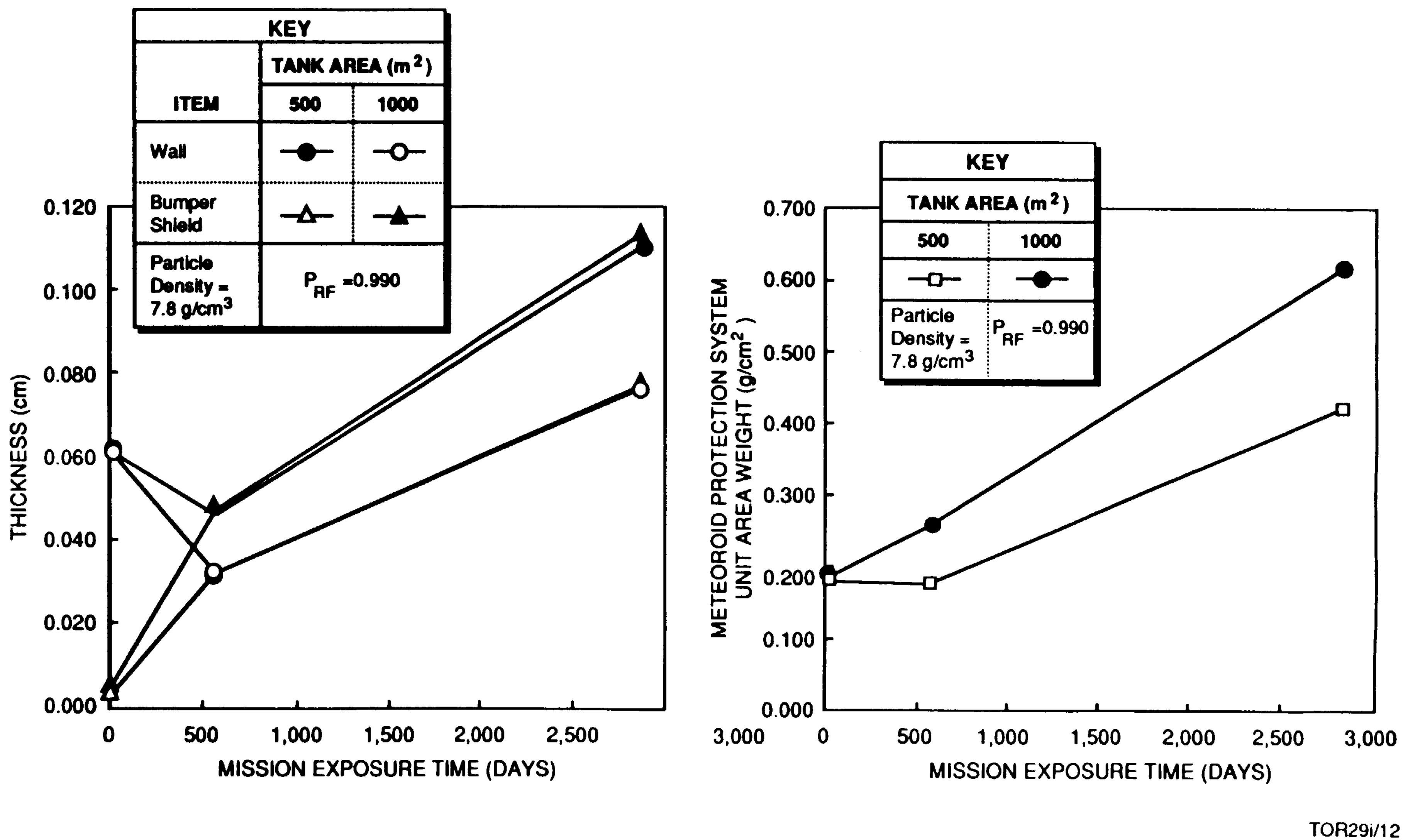


Figure 4-13. Typical Tank Meteoroid Protection System Characteristics as a Function of Mission Exposure

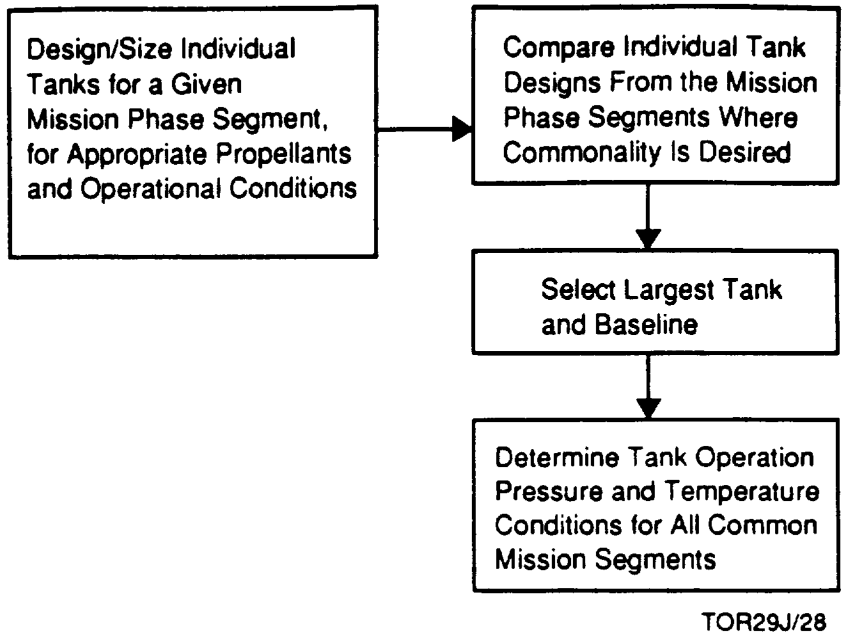


Figure 4-14. Common Tank Sizing Analysis Approach

Table 4-27. General Tank Design Features Assumed

- Suspended Nonloading Tankage Design with External Lines
- Separated Dome Tanks (where applicable)
- Spherical Tanks Except Where Tank Diameter Would Be Greater Than 11.8 meters (Compatible with Launch Vehicle Payload Envelope Constraints). Otherwise, Cylindrical Tanks with Elliptic Tank Domes Assumed
- 2% ullage
- Surface Tension Propellant Acquisition Devices
- Tank Materials
  - Weldalite*
  - Al 2219-T87
- Insulation
  - SUPERFLOC*
  - Conventional MLI
- Meteoroid Shield Material
  - Al 2219-T87 (Conventional Aluminum Alloy)
- Autonomous Pressurization
- Helium Startup Pressure System (2 tanks per stage)
- Average Propellant/Space Exposure Time Used
- Average Distance and Worst-Case Radiation Exposure Assumed

* Trademark

Table 4-28. Tank Design Systems Evaluated by Mission Segment

Scenario No./Vehicle/Tank Configuration	MISSION SEGMENT			
	TMI	MOC	TEI	EOC
1. Baseline Earth LOX/H_2				
- Individual Burn Tanks	●*	●	●	
2. Lunar LOX, Mars LOX/CO				
- Individual Burn Tanks	●		●	
- Common/Mission Segmented Propellant Tanks	●		●	
4. Lunar LOX/CH_4, Mars LOX/CO				
- Individual Burn Tanks	●	●	●	
- Common/Mission Segmented Propellant Tanks	●	●	●	

* Complete Tank Set (Fuel and Oxidizer)

Table 4-29. Tank Design Comparison Rationale

Comparison Case No.	Tank System Elements	Rationale/Insight
1	(1/TMI)*, (1/MOC), (1/TEI)	• Establishes baseline LOX/H_2 cryogenic tank system design
2	1/(TMI) vs. (2/TMI) vs. (4/TMI)	• Evaluates the impact of in situ propellant tank designs vs. cryogenic propellant tank designs for TMI • A direct comparison of the lunar LOX/H_2 vs. lunar LOX/CH_4 for TMI
3	(1/TMI) + (1/TEI) vs. (2/TMI) + (2/TEI)	• Compares a tank design for TMI and TEI for Scenario 2 against the conventional LOX/H_2 baseline system
4	(1/TMI) + (1/TEI) vs. (4/TMI) + (4/TEI)	• Compares a tank design for TMI and TEI for Scenario 4 against the conventional LOX/H_2 baseline
5	(2/TMI) + (2/TEI) vs. (4/TMI) + (4/TEI)	• Comparison of tank designs between Scenarios 2 and 4
6	(1/MOC) vs. (4/MOC)	• Comparison of boiloff effects on tank design used for long propellant storage for the conventional cryogenic LOX/H_2 system and the storable in situ-based LOX/CH_4 system
7	(1/TEI) vs. (2/TEI)	• Comparison of tank designs for long-term exposure to space for the conventional LOX/H_2 system and the in situ-based LOX/CO system
8	(1/TMI) + (1/MOC) + (1/TEI) vs. (4/TMI) + (4/MOC) + (4/TEI)	• Comparison of tank designs for an all cryogenic LOX/H_2 system versus an all in situ propellant-based LOX/CH_4 and LOX/CO system
9	(1/MOC) and (4/MOC)	• Tank insulation type varied, effect of boiloff compared
10	(1/TMI) and (4/TEI)	• Tank material varied, system weight compared
11	(4/TMI)	• Tank pressure varied, system weight compared

* (Scenario No./Mission Segment)

Detailed results for the tank designs analyzed, see Table 4-28, are given in Appendix F. Tables 4-30, 4-31 and 4-32 summarize the tank design comparison results. Detailed mass tank design weight comparisons results are shown in Table 4-30. Substantial reductions in the dry weight of the tank systems which employ LOX/CO and LOX/CH_4 can be realized when compared to systems that use Earth-based LOX/H_2. Use of SUPERFLOC insulation and Weldalite tank materials reduces tank system weight substantially when compared to conventional tank materials and insulations. Tank pressure over the range investigated had little effect on overall tank system mass.

Total tankage system mass fractions are summarized in Table 4-31 for the tank designs evaluated. Tankage systems which store LOX/H_2 have total tankage system mass fractions greater than 0.020. Those tank systems which hold the in situ-based propellant combinations of LOX/CO and LOX/CH_4 exhibit mass fractions in the range of 0.011 to 0.016. The high mass fractions associated with the LOX/H_2 tank design are attributed to the large size from increased boiloff and the low density associated with H_2. These mass fraction shown in Table 4-31 are considered highly representative of such systems and should be considered for incorporation in future top-level mission and vehicle design studies.

Table 4-32 presents the estimated dry tankage system weight savings by employing common propellant tanks for the MTV for each mission scenario considered. Employing a common tank MTV can reduce tank system weight by approximately 40% compared to using individual tanks for the in situ-based scenario. Developing tank technologies to support such common tank designs would have a high payoff.

Table 4-30. Tank Design Comparison Results Summary

Comparison Case No.	Mass Comparisons (lbm)	Comment(s)
1	• 1/TMI* - Propellant Carried = 1,099,183 - Oxidizer Tank = 4,603.9 - Fuel Tank = 16,643.8 - Other = 5,186.3 - Total (wet) = 1,125,608 • 1/MOC - Propellant Carried = 605,699 - Oxidizer Tank = 2,862.6 - Fuel Tank = 6,716.3 - Other = 2,869.5 - Total (wet) = 618,147.4 • 1/TEI - Propellant Carried = 202,832 - Oxidizer Tank = 1,386.6 - Fuel Tank = 3,378.8 - Other = 1,147.4 - Total (wet) = 208,739.8	—
2	• 1/TMI - Same as Comparison Case No. 1 • 2/TMI - Propellant Carried = 273,022 - Oxidizer Tank = 1,698.0 - Fuel Tank = 3,998.1 - Other = 1,393.3 - Total (wet) = 280,111.4 • 4/TMI - Propellant Carried = 384,128 - Oxidizer Tank = 1,991.8 - Fuel Tank = 1,896.8 - Other = 1,680.9 - Total (wet) = 389,697.5	• Substantial reduction in tankage system weight is possible using in situ-based propellants for the TMI mission segment - Scenario No. 1 vs. Scenario No. 2 – Oxidizer tank = 63.1% reduction – Fuel tank = 76.0% reduction – Total dry weight = 73.2% reduction - Scenario No. 1 vs. Scenario No. 4 – Oxidizer tank = 56.7% reduction – Fuel tank = 88.6% reduction – Total dry weight = 78.9% reduction • Scenario No. 4 TMI tankage system dry weight is 21.4% lighter than that associated with Scenario No. 2 • Major differences in total wet weight for all 3 scenarios - Influenced by mission approach, propellant density, and engine specific impulse effects
3	• 1/TMI - Same as Comparison Case No. 1 • 1/TEI - Same as Comparison Case No. 1 • 2/TMI - Same as Comparison Case No. 2 • 2/TEI - Propellant Carried = 325,607 - Oxidizer Tank = 1,053.9 - Fuel Tank = 2,148.5 - Other = 1,570.6 - Total (wet) = 330,380	• Using the tank design approach for Scenario No. 2 for the MTV transit flight phases reduces total tankage system dry weight 63.3% - Not influenced by boiloff

* Tankage Concept No. (see Table 4-25)/Mission Segment

Table 4-30. Tank Design Comparison Results Summary (Cont.)

Comparison Case No.	Mass Comparisons (lbm)	Comment(s)
4	• 1/TMI - Same as Comparison Case No. 1 • 1/TEI - Same as Comparison Case No. 1 • 4/TMI - Same as Comparison Case No. 2 • 4/TEI - Propellant Carried = 731,017 - Oxidizer Tank = 1,789.0 - Fuel Tank = 3,251.4 - Other = 3,258.1 - Total (wet) = 739,315.5	• Using the tank design approach for Scenario No. 4 for the MTV transit flight phases reduces total tankage system dry weight 63.3% - Not influenced by boiloff
5	• 2/TMI - Same as Comparison Case No. 2 • 2/TEI - Same as Comparison Case No. 3 • 4/TMI - Same as Comparison Case No. 2 • 4/TEI - Same as Comparison Case No. 4	• Total dry tankage system weight for Scenario 2 is only reduced 14.2% when compared to Scenario 4
6	• 1/MOC - Same as Comparison Case No. 1 • 4/MOC - Propellant Carried = 261,979 - Oxidizer Tank = 1,546.0 - Fuel Tank = 1,477.6 - Other = 1,229.9 - Total (wet) = 266,232.5	• Scenario 4 MOC dry tankage system weight is 65.8% less than that associated with the comparable baseline LOX/H_2 tankage system - Boiloff, engine performance and propellant density influence this result
7	• 1/TEI - Same as Comparison Case No. 1 • 2/TEI - Same as Comparison Case No. 3	• Using LOX/CO for TEI reduces the dry tankage system approximately 19.2% compared to the LOX/H_2 scenario baseline
8	• 1/TMI - Same as Comparison Case No. 1 • 1/MOC - Same as Comparison Case No. 1 • 1/TEI - Same as Comparison Case No. 1 • 4/TMI - Same as Comparison Case No. 2 • 4/MOC - Same as Comparison Case No. 6 • 4/TEI - Same as Comparison Case No. 4	• MTV dry tankage system weight can be reduced by 59.5% by using all in situ propellant scenarios (Scenario No. 4) compared to an all Earth LOX/H_2 system

Table 4-30. Tank Design Comparison Results Summary (Cont.)

Comparison Case No.	Mass Comparisons (lbm)	Comment(s)
9	• 1/MOC - Same as Comparison Case No. 1 • 1/MOC - Propellant Carried = 611,696 - Oxidizer Tank = 3,151.8 - Fuel Tank = 7,626.8 - Total (wet) = 625,342.3 • 4/MOC (Baseline) - Same as Comparison Case No. 6 • 4/MOC - Propellant Carried = 263,764 - Oxidizer Tank = 1,694.5 - Fuel Tank = 1,623.6 - Other = 1,227.9 - Total (wet) = 268,310	• Use of conventional MLI for the MOC baseline LOX/H_2 tankage system increases its total dry weight by 56.7% compared to a system which uses SUPERFLOC • Using MLI or SUPERFLOC has little effect on the Scenario 4 MOC tankage system. Only a 6.4% increase in weight is predicted by using SUPERFLOC
10	• 1/TMI (Baseline) - Same as Comparison Case No. 1 • 1/TMI - Propellant Carried = 1,099,189 - Oxidizer Tank = 6,582.2 - Fuel Tank = 24,558.8 - Other = 5,209.7 - Total (wet) = 1,135,539.7 • 4/TEI (Baseline) - Same as Comparison Case No. 4 • 4/TEI - Propellant Carried = 731,012 - Oxidizer Tank = 1,832.2 - Fuel Tank = 3,890.2 - Other = 3,257.7 - Total (wet) = 739,992.1	• Employing Al 2219-T87 tank materials increases the Scenario 1 TMI dry tankage weight by 27.3% • Employing Al 2219-T87 tank materials increases the Scenario 4 TMI dry tankage weight by only 7.6%
11	• 4/TMI (Baseline) - Same as Comparison Case No. 2 • 4/TMI - Propellant Carried = 368,289 - Oxidizer Tank = 1,992.1 - Fuel Tank = 1,904.3 - Other = 1,753.3 - Total (wet) = 391,938.7 • 4/TMI - Propellant Carried = 388,476 - Oxidizer Tank = 1,995.2 - Fuel Tank = 1,918.0 - Other = 1,754.4 - Total (wet) = 394,143.6	• Pressure ranges examined - Oxidizer tank: 22.8 to 62.8 psia - Fuel tank: 35.0 to 52.5 psia • Increasing tank pressure had little effect on tankage system dry weight (<1.7%)

Table 4-31. Summary of Total Tankage System Mass Fractions

Scenario No.	Mission Segment	Total Tankage System Mass Fraction*
1	TMI	0.024
	MOC	0.020
	TEI	0.028
2	TMI	0.025
	TEI	0.014
4	TMI	0.014
	MOC	0.016
	TEI	0.011

* Baseline design assumptions assumed; individual burn tank design approach used.

Table 4-32. Summary of Potential Tankage System Weight Savings by Employing Common Propellant Tanks for MTV Earth-Mars-Earth Mission Segments

Scenario No.	Tank Type	Mission Segment Which Drives Tank Commonality	Estimate of Dry Tankage System Weight Savings (%)
1	Oxidizer	TMI	18.3
	Fuel	TMI	
2	Oxidizer	TMI	40.2
	Fuel	TMI	
4	Oxidizer	TMI	42.0
	Fuel	TEI	

5.0 MISSION PERFORMANCE AND COMPARISON

Mission performance was reassessed using the baseline multipropellant engine designs described in Section 4.2.3. Details of the approach and assumptions used in this updated analysis, except as noted, are the same as those used in the initial mission analysis effort described in Section 2.0. This section compares candidate mission scenarios and engine cycles, and describes the results of trade studies defining sensitivity of mission performance to engine design parameters such as mass, Isp, and nozzle area ratio. Also discussed is an assessment of alternative propellant tank reuse/staging strategies. A summary of all scenarios described in this section is shown in Table 5-1. All figures in this section refer to the scenario designations from this table. Scenario 5, which was included in the initial mission assessment effort (see Section 2.0) for comparison to the other candidate scenarios, was excluded from these final performance assessments because it does not require a multipropellant engine.

For these final performance calculations, more refined tank sizing assumptions were also employed. In the initial calculations, mass was simply computed as a percentage of the propellant inside the tank. For the final calculations, a specific Al/Li alloy is assumed for the tank wall material. On top of this alloy, a layer of foam is sprayed, and MLI insulation, a vapor-cooled shield, and a micrometeoroid shield are added (see Table 5-2). For tanks containing the TMI propellant, only 5 cm of MLI is assumed, since these tanks have a much shorter space storage time than the other tanks.

For each scenario, vehicle and plant mass were calculated for expander and gas generator engines of 400:1 area ratio for all vehicles (booster stage, MTV, LEV, and MEV). A 165:1 expander engine was also assessed for Scenario 6, along with trades investigating the effect of higher or lower engine mass and higher or lower Isp for a 400:1 area ratio engine. Additionally for Scenario 7, trades were performed for alternatives in which tanks and/or engines would be reused within the same mission or from one mission to the next.

The final mission performance tables in Appendix E provide the propellant requirements for each mission burn, showing the mass of the vehicle immediately prior to each burn, ΔV requirements, engine masses, Isp's, thrust levels, and engine thrust/burn times.

Table 5-1. Mission Performance Assessment Scenarios

Scenario	Outbound Propellant	Return Propellant	Engine Thrust (klb)	Engine Cycle	Options
1A	Earth LOX/H_2	Earth LOX/H_2	250	Expander	400:1 area ratio
1B	Earth LOX/H_2	Earth LOX/H_2	250	GG	400:1 area ratio
2A	Lunar LOX/ Earth H_2	Mars LOX/CO	175	Expander	400:1 area ratio
2B	Lunar LOX/ Earth H_2	Mars LOX/CO	175	GG	400:1 area ratio
3A	Lunar LOX/ Earth H_2	Mars LOX/CH_4	250	Expander	400:1 area ratio
3B	Lunar LOX/ Earth H_2	Mars LOX/CH_4	250	GG	400:1 area ratio
4A	Lunar LOX/ CH_4	Mars LOX/CO	175	Expander	400:1 area ratio
4B	Lunar LOX/ CH_4	Mars LOX/CO	175	GG	400:1 area ratio
6A	Earth LOX/H_2	Mars LOX/CO	175	Expander	400:1 area ratio
6B	Earth LOX/H_2	Mars LOX/CO	175	Expander	165:1 area ratio
6C	Earth LOX/H_2	Mars LOX/CO	175	GG	400:1 area ratio
6D	Earth LOX/H_2	Mars LOX/CO	175	Expander	400:1 area ratio +10% eng. mass
6E	Earth LOX/H_2	Mars LOX/CO	175	Expander	400:1 area ratio -10% eng. mass
6F	Earth LOX/H_2	Mars LOX/CO	175	Expander	400:1 area ratio +10% Isp
6G	Earth LOX/H_2	Mars LOX/CO	175	Expander	400:1 area ratio -10% Isp
7A	Earth LOX/H_2	Mars LOX/CH_4	250	Expander	400:1 area ratio
7B	Earth LOX/H_2	Mars LOX/CH_4	250	Expander	400:1 area ratio MTV MOC tanks reused for TEI+EOC
7C	Earth LOX/H_2	Mars LOX/CH_4	250	Expander	400:1 area ratio No tank/engine staging
7D	Earth LOX/H_2	Mars LOX/CH_4	250	GG	400:1 area ratio
7E	Earth LOX/H_2	Mars LOX/CH_4	250	Expander	400:1 area ratio 2 MOC tank sets: 1 MOC set reused for TEI and then staged; 1 MOC set sized for EOC propellant (reused for EOC)

Table 5-2. Propellant Tank Mass Allocations

Layer	Thickness (cm)	Areal Density (kg/m^2)
Tank Wall	0.4	10.95
Foam	1.27	0.55
SUPERFLOC MLI	5 (60 layers)	1.115
Vapor-Cooled Shield	-	1.27
SUPERFLOC MLI	5	1.115
Micrometeoroid Shield	0.05	2.80
Total Areal Density (kg/m^2) = **17.8**		

5.1 Expander vs. Gas Generator Cycle Engine Assessment

Figure 5-1 shows a comparison of lunar and Mars propellant plant mass for each scenario for vehicles using both expander and gas generator cycle engines that use a 400:1 nozzle area ratio. These plant masses are representative of the front-end investment required to support a given scenario. The plant masses required for scenarios employing expander-type engines are consistently higher than those that employ gas generator cycle engines. Although the expander cycle engines have slightly higher Isp's than the gas generator engines, the performance advantage of the higher Isp expander engine is overshadowed by its significantly higher engine mass, and, therefore, requires more propellant and a larger ISPP plant. The greatest plant mass difference occurs for the Mars LOX/CO propellant plant of Scenario 4, where the plant required for the gas generator engine scenarios is 16.4% lighter than that for expander engine scenarios. The smallest plant mass difference occurs for the lunar propellant plant of Scenario 3, where the plant required for gas generator engines is 1.7% lighter than that required for expander engines.

The Mars LOX/CO plant mass used in Scenarios 2, 4, and 6 is substantially greater than any of the other plant masses, as depicted in Figure 5-1. This is due mainly to the refrigeration requirement to separate CO from a CO-CO_2 gas mixture obtained during processing of the Mars atmosphere. Alternative technologies for this separation are currently under investigation by several researchers and may enable production of Mars LOX/CO with much smaller ISPP plant sizes.

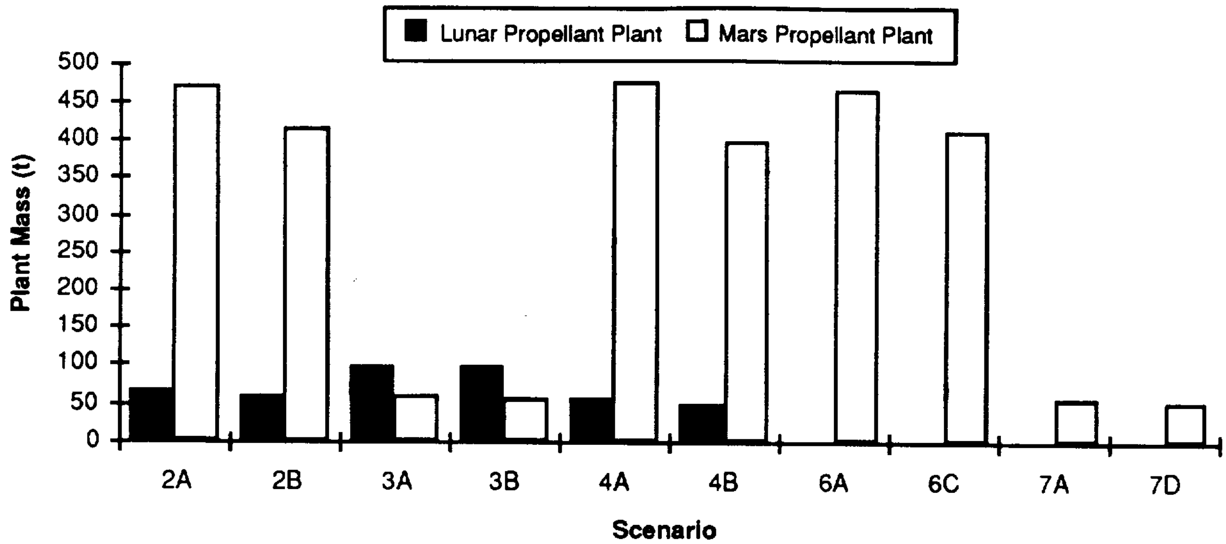
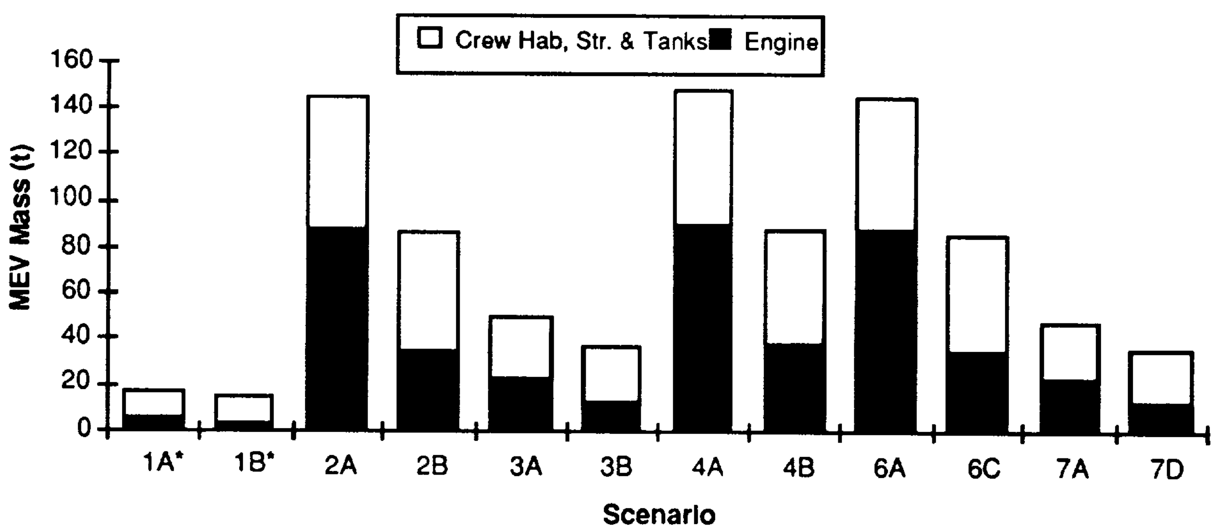


Figure 5-1. ISPP Plant Mass Comparison: Expander vs. Gas Generator Cycle Engine Assessment

Figure 5-2 shows MEV dry mass for each scenario for both expander and gas generator engines. The vehicles using the expander engines are consistently heavier than those using the gas generator engines, since the gas generator engines are anywhere from 43% to 61% lighter than the expander engines (see Tables 4-15 and 4-16). The MEV mass in scenarios where Mars LOX/CO is used is markedly higher than that in scenarios using Mars LOX/CH_4. This difference is because LOX/CO propellant has an Isp of about 290 seconds, compared to LOX/CH_4 which has an Isp of about 390 seconds. Therefore, much more LOX/CO propellant is needed to perform the mission than LOX/CH_4 propellant.



* These masses are for a single MEV, while all the other scenarios (bars) refer to the combined masses of 2 MEVs.

Figure 5-2. MEV Mass Comparison: Expander vs. Gas Generator Cycle Engine Assessment

LEV dry mass is shown in Figure 5-3 for the in situ scenarios in which lunar propellant is used. Similar to the MEV case, the LEVs using expander cycle engines are heavier than those using gas generator engines. The LEVs in Scenarios 2 and 3 use lunar-produced LOX in combination with Earth-produced H_2. The vehicles carry lunar-produced LOX up to LLO and transfer it into the MTV tanks. The MTV makes the trip from LLO to LMO using this lunar LOX along with Earth H_2. In these two scenarios, the LEV not only transports oxygen plant resupply materials down to the lunar surface, but it also has to carry down the Earth-produced H_2 it needs to perform the next surface-to-LLO-to-surface mission. This H_2 is brought out to the Moon on the expendable booster and is transferred to the LEV in orbit.

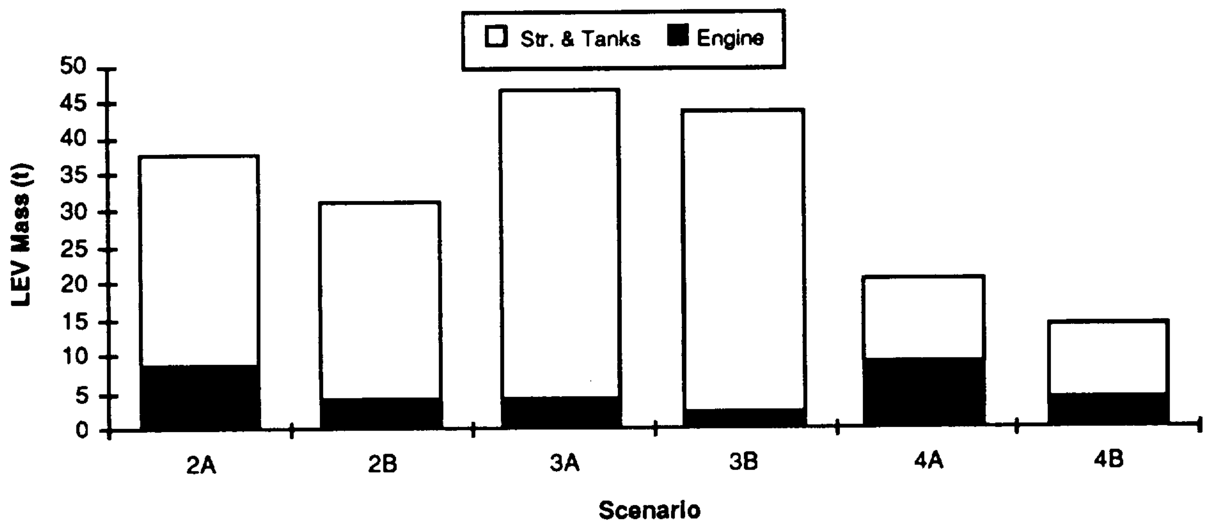


Figure 5-3. LEV Mass Comparison: Expander vs. Gas Generator Cycle Engine Assessment

In Scenario 4, the LEV uses lunar-produced LOX/CH_4 for propellant. Here, all of the propellant used by the LEV is lunar-produced LOX/CH_4. The dry mass is lower here than in the cases using lunar LOX/Earth H_2, since it does not have to carry Earth-produced propellant back down to the surface.

Figure 5-4 shows a comparison of MTV dry mass for all the scenarios for expander vs. gas generator engines. As expected, the vehicles with expander engines have higher mass than those with gas generator engines. The shaded portion of each bar is the MTV engine mass. Again, the heavier expander engines' performance is slightly improved (higher Isp) over the gas generator engines, but results in a higher vehicle weight. The white portion of each bar represents the combined mass of a 30 t crew habitat module, core EOI propellant tanks, and structure. The masses shown here do not include the mass of the crew and consumables (totaling approximately 7 t).

C-2.

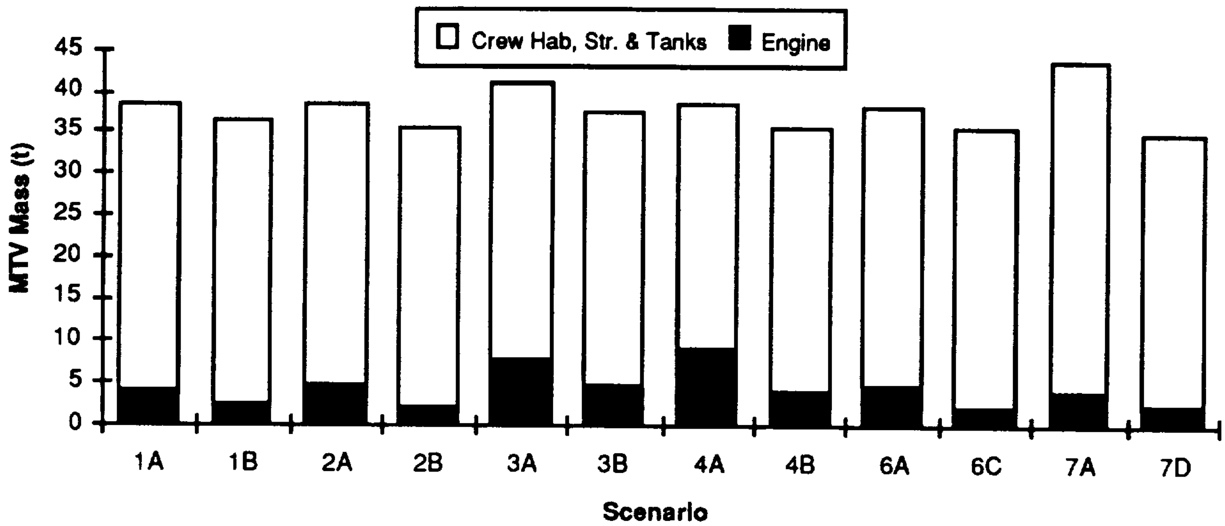


Figure 5-4. MTV Mass Comparison: Expander vs. Gas Generator Cycle Engine Assessment

Steady-state Earth Launch Mass (ELM) per mission is displayed in Figure 5-5 for each scenario for both expander and gas generator engine types. The legend at the top of this figure shows the elements that comprise the steady-state ELM and include (from top of each bar down): 1) the 25 t Mars mission payload; 2) the 4 crew members and their consumables; 3) the MEV aeroshell used for decelerating the MEV during descent to the Mars surface; 4) the engines that are staged during the mission; 5) the staged propellant tanks; 6) propellant supplied from Earth; 7) refurbishment and consumable resupply for the Mars ISPP plant; and 8) refurbishment and consumable resupply for the lunar ISPP plant used only in Scenarios 2, 3, and 4.

The significance of Figure 5-5 is that it shows the launch mass savings achievable per mission over the long term by employing in situ propellant production at the Moon and/or Mars. Scenarios using expander cycle engines (1A, 2A, 3A, 4A, 6A, 7A) depict the potential ELM savings as great as 81% (Scenario 4A) over the baseline chemical propulsion scenario (1A), which uses no in situ propellant. The major mass savings is in reduction of the amount of Earth-sourced propellant required to perform the mission. In Scenarios 2 and 3, Earth-supplied LOX/H_2 is needed by the expendable booster to transport the MTV from LEO to LLO, and Earth-supplied H_2 is needed to fuel the LEV and the MTV for the LLO to LMO leg of the trip. In Scenario 4, Earth-supplied LOX/H_2 is needed only by the booster to carry the MTV from LEO to LLO, while in Scenarios 6 and 7, Earth-supplied LOX/H_2 is used for the LEO to LMO leg of the MTV trip. The scenario using the least Earth-supplied propellant is Scenario 4.

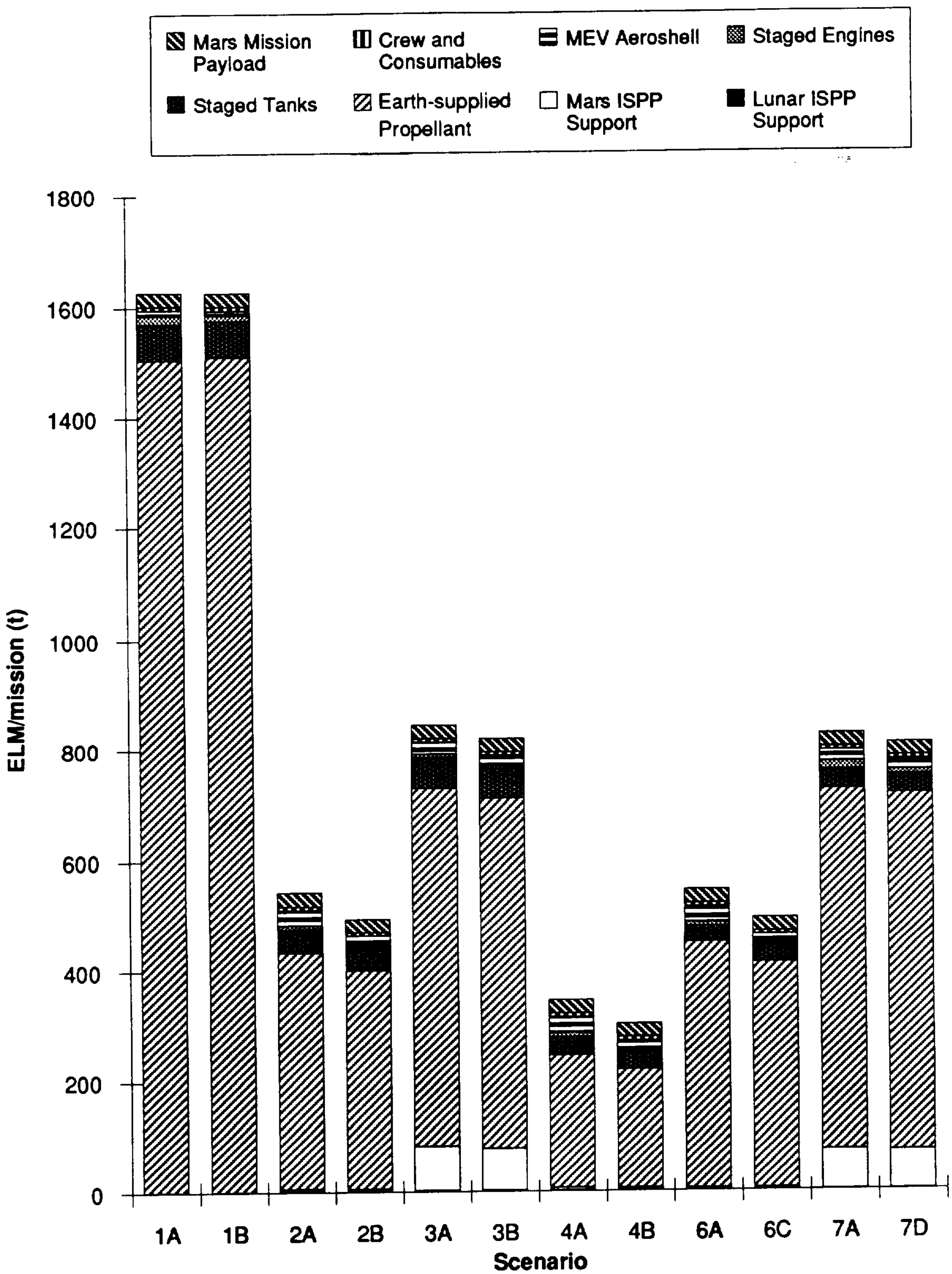


Figure 5-5. Steady-State Earth Launch Mass per Mission Comparison: Expander vs. Gas Generator Cycle Engine Assessment

An interesting observation is that even though the Mars plant mass is substantially higher in Scenarios 2, 4, and 6 than it is in Scenarios 3 and 7, the ELM is lower because no reagent resupply is needed by the Mars LOX/CO plant, see Figure 5-2. For the LOX/CH_4 plant, however, over 70 t of Earth-produced H_2 is needed for reagent resupply. This necessity increases ELM substantially.

Note that Figure 5-5 shows ELM per mission in the steady-state operation, after the plants have been constructed at the Moon and/or Mars. Figure 5-5 does not show the ELM required for the first few missions that emplace the infrastructure elements. The infrastructure elements are: 1) the fully operational lunar and Mars ISPP plants; 2) the surface excursion vehicles (LEVs and MEVs) needed to transport propellant from the plants up to the MTV and to bring crew, mission payload, and plant resupply down to the surface; and 3) the MTV. The masses represented by each bar are the masses of elements that are resupplied for each mission. These elements are shown in the legend at the top of Figure 5-5.

5.2 Engine Design and Tank Reuse Trades

To better understand the sensitivity of the mission performance assessment to engine design parameters, several trades were performed for Scenarios 6 and 7. In Scenario 6, these trades included investigations of mission performance using different engine mass and Isp values, and using an engine with a lower nozzle area ratio. Additionally, in Scenario 7, three propellant tank reuse strategies were assessed to identify potential savings by using a tank for more than one burn. All other scenarios staged tanks after being emptied and carried empty tanks for fuel obtained from the Moon or Mars.

Results for the engine design and tank reuse trades are characterized by three key elements. The first is the mass of the ISPP plant required on the Mars surface to enable the production levels needed for the return trip to Earth. This comparison is shown in Figure 5-6. The second element is the mass of the transfer and excursion vehicles used and is representative of the requirements for vehicle replacement missions. These results are shown in Figure 5-7. The third element is the ELM requirements for steady-state operation. These requirements are shown in Figure 5-8 and can be compared to the case using all Earth propellants, which requires 1,627 t delivered to Earth orbit for support of a single mission. A discussion of these results follows.

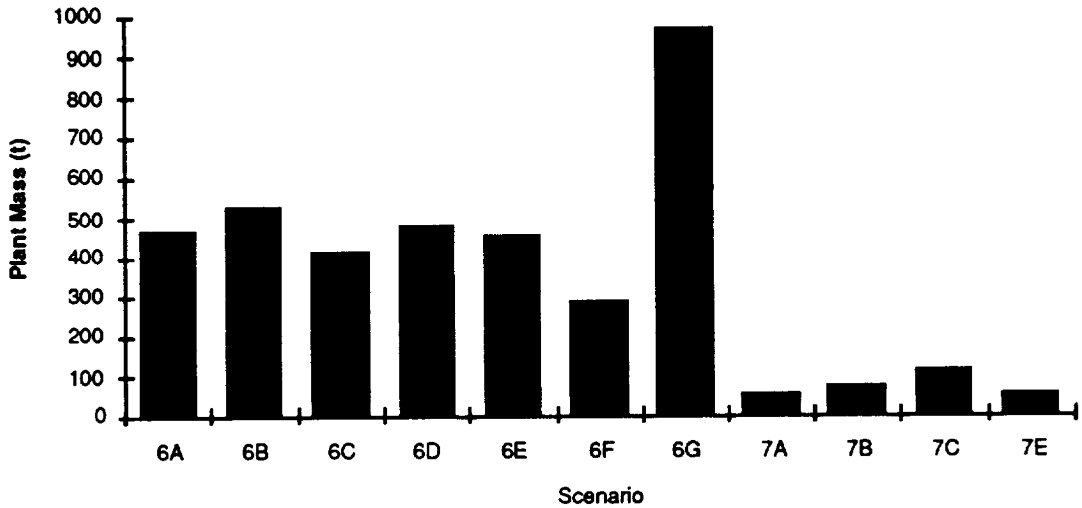


Figure 5-6. Mars Plant Mass Comparison: Engine Design and Tank Reuse Trades

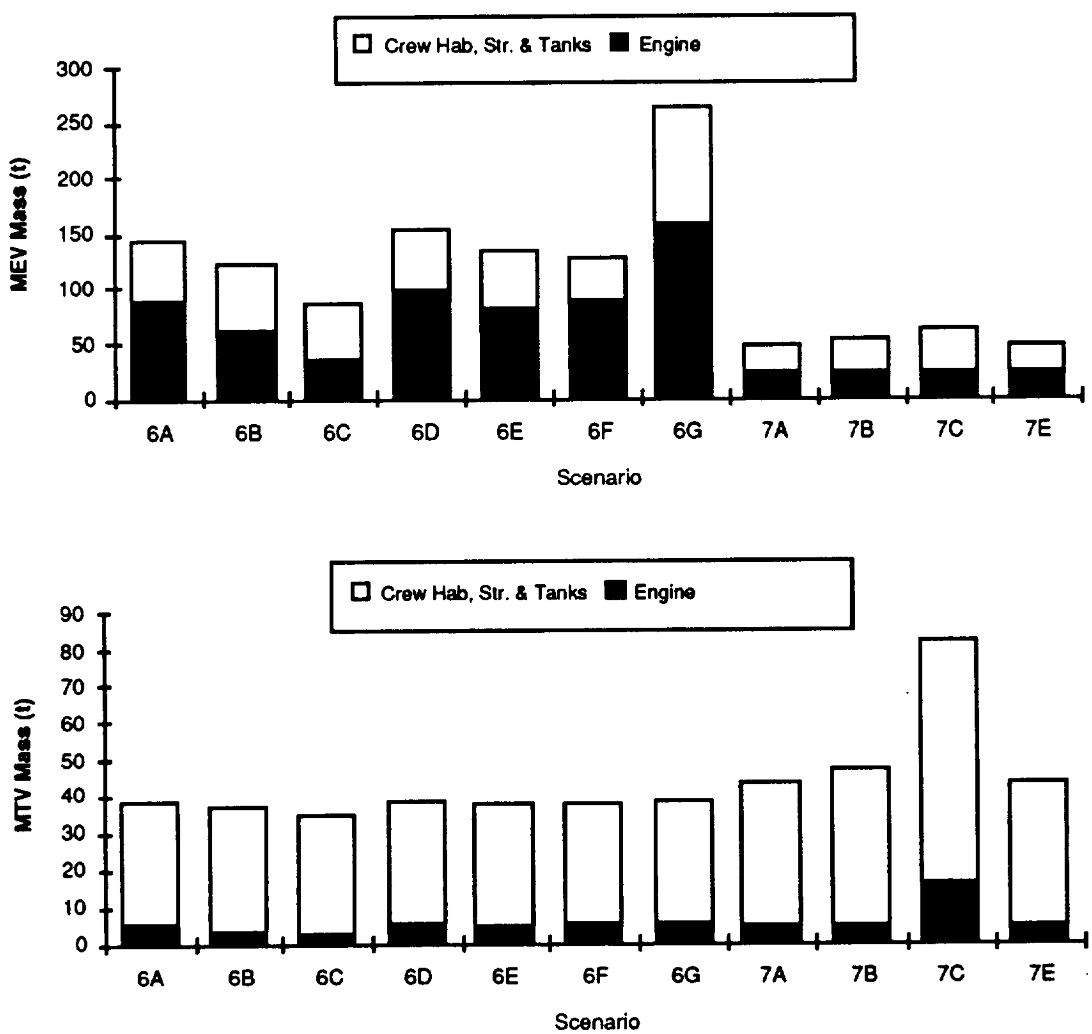


Figure 5-7. MEV and MTV Mass Comparison: Engine Design and Tank Reuse Trades

5.2.1 Engine Mass

Engine mass sensitivity analyses were performed for the case that departs Earth orbit with Earth LOX/H_2 for the outbound trip and refuels with LOX/CO produced at Mars for the return trip (Scenario 6). The engine design used is the LOX/CO/H_2 expander cycle engine with a 400:1 area ratio. The results are shown in Figures 5-6 through 5-8 and refer to Scenarios 6D and 6E. Scenario 6A uses the engine design obtained from the engine system assessment portion of this study described in Section 4.2.3. Scenario 6D adds 10% to the engine mass from 6A. Scenario 6E uses an engine with 10% less mass than in 6A. Comparing the results of Scenarios 6D and 6E to 6A shows low sensitivity of mission performance results to a ±10% change in engine mass. The impacts of this change in engine mass on the masses of the Mars ISPP plant, MEV, MTV, and steady-state Earth launch requirements to support one mission are shown in Figure 5-9. Although the change in steady-state Earth launch mass requirements is not more than ±3% with a ±10% change in engine mass, the reduction of ELM with a -10% change in engine mass is twice the increase of ELM with a +10% change in engine mass. This suggests that further reductions in engine mass, without a loss of performance, may yield even greater savings in ELM requirements.

5.2.2 Engine Performance

Engine performance sensitivity analyses were performed for the same case and with the same engine design as described above. These results are shown in Figures 5-6 through 5-8 and refer to Scenarios 6F and 6G. Scenario 6F adds 10% to the Isp used for 6A and Scenario 6G reduces the Isp from 6A by 10%. These results are summarized in Figure 5-10. The sensitivity of mission performance to engine Isp appears significantly higher than the sensitivity to engine mass. Because engine Isp directly affects propellant requirements, which in turn affect the Mars ISPP plant mass and support requirements, which affect the size of the payload transported to Mars, mission performance is strongly impacted. The steady-state Earth launch mass penalty for a -10% change in Isp is over 60%, although a +10% change saves only about 20%. This sensitivity may not be as great working with a different engine design with a higher Isp (LOX/CO Isp for the return trip is only 293 sec). These results suggest that if engine Isp can be increased with only a small increase in engine mass, additional Earth launch mass savings may be attainable.

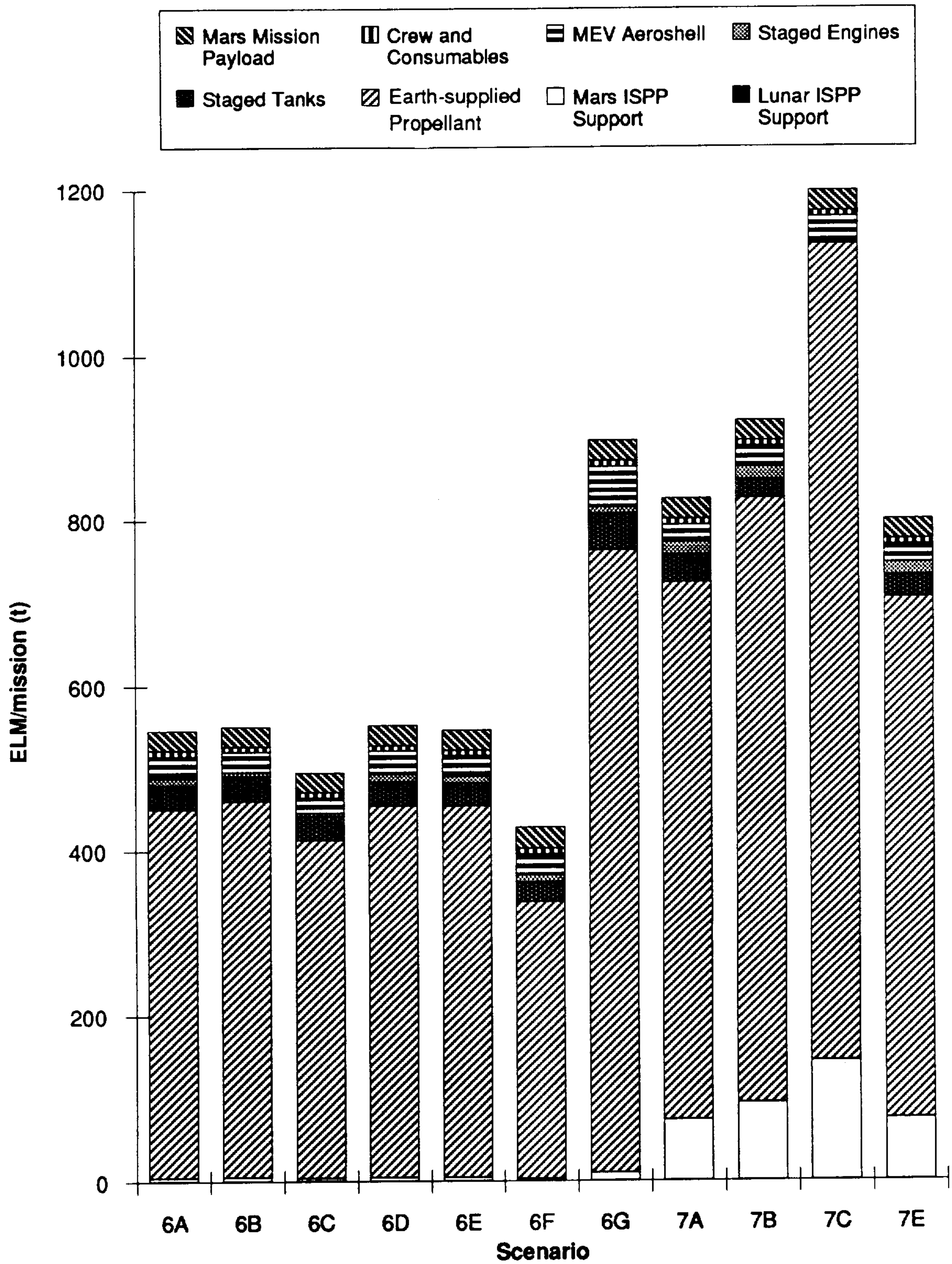


Figure 5-8. Steady-State Earth Launch Mass per Mission Comparison: Engine Design and Tank Reuse Trades

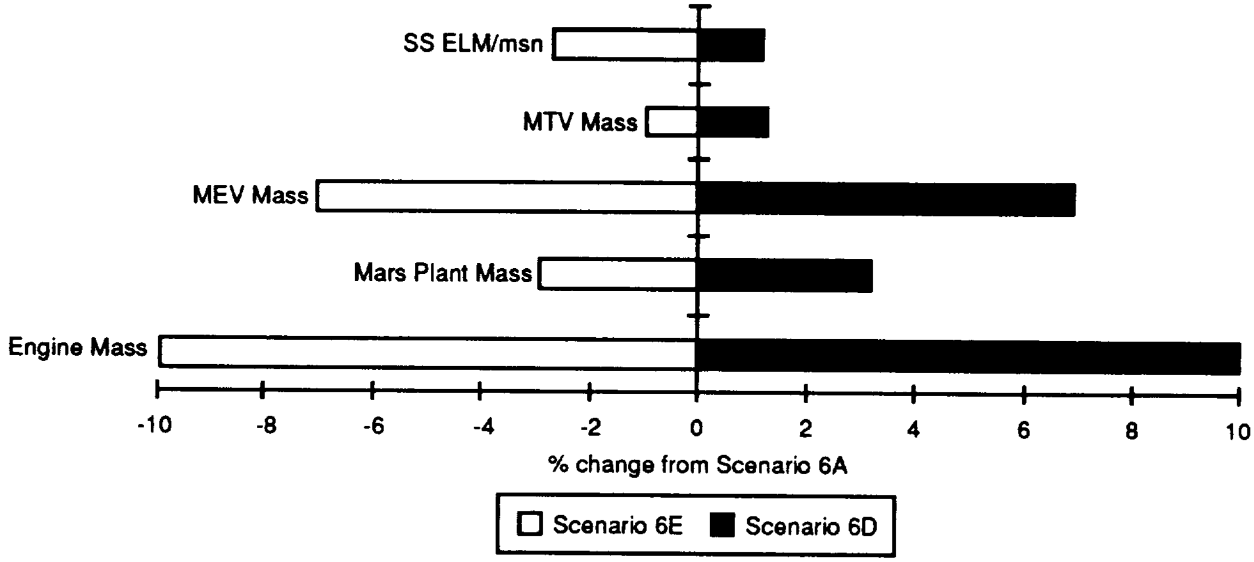


Figure 5-9. Results of Engine Mass Trade Study

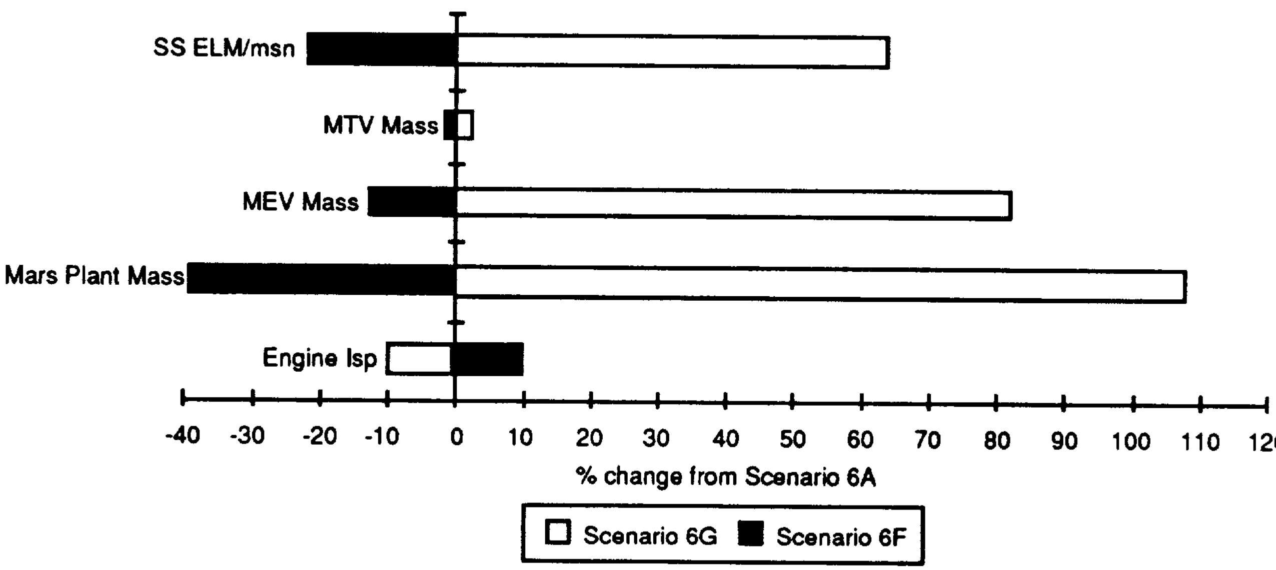


Figure 5-10. Results of Engine Isp Trade Study

5.2.3 Nozzle Area Ratio

The effect of using an engine with an nozzle area ratio of 165:1, versus 400:1, for the same scenario and engine concept as in the engine mass sensitivity analyses was investigated and is shown as Scenario 6B in Figures 5-6 through 5-8. The effect of reducing the area ratio resulted in about a 30% decrease in engine mass with only about a 3% decrease in engine Isp.

Impacts on the Mars ISPP plant mass, MEV, MTV, and steady-state Earth launch mass are shown in Figure 5-11. The result of the lower engine mass and Isp is less than a 1% increase in steady-state ELM required. The masses of the transfer and excursion vehicles will reduce requirements for vehicle replacement missions, but the higher mass of the Mars ISPP plant will drive up the front-end costs of emplacing the needed ISPP plant and push back the time to the ELM break-even point. One advantage of using the lower area ratio engine that is not shown in the mission performance analysis is that this engine should be easier to package in the cargo bay of an Earth-launched vehicle.

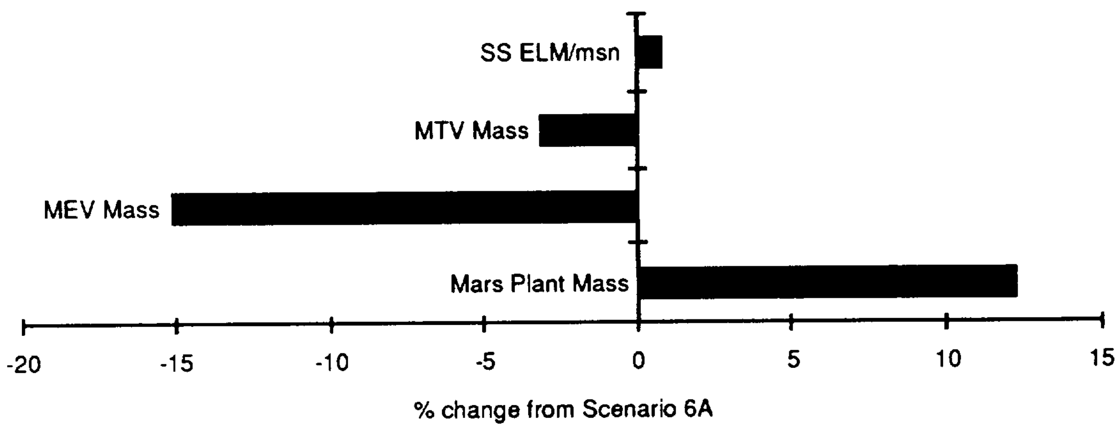


Figure 5-11. Results of Engine Nozzle Area Ratio Trade Study

5.2.4 Tank Reuse Strategies

For Scenario 7, several tank reuse strategies were investigated to identify effects on mission performance. Baseline mission performance does not reuse tanks, except for the core MTV tanks holding EOI propellant, and carries empty tankage to fill at Mars for the return trip. The MTV basically consists of a core with tanks, engine(s), and crew habitat module and several sets of stageable tanks which jettison after TMI, MOC, and TEI burns. In Scenario 7, the MTV uses Earth LOX/H_2 for the outbound trip and returns with Mars LOX/CH_4. The engine concept used is the expander cycle LOX/CH_4/H_2 engine with a 400:1 area ratio. This case was chosen because tank volumes needed for the outbound trip with LOX/H_2 were anticipated to be close to the volumes needed for the return trip with LOX/CH_4.

The strategies investigated are shown schematically in Figure 5-12 and are depicted in Figures 5-6 through 5-8 as Scenarios 7B, 7C, and 7E. In Scenario 7B, TMI tanks are staged after TMI and the tanks used for MOI are sized to hold the propellant for the return trip and are carried with the MTV back to LEO. In Scenario 7C, no tanks are staged. These strategies were selected to reduce the steady-state ELM by minimizing the mass of replacement propellant tanks needed for a mission. The approach used for Scenario 7E attempts to minimize the mass of empty tankage carried through Earth departure and Earth return ΔVs. In this scenario, the TMI tanks are staged after TMI and the MOI tanks are separated into two sets. One MOI tank set is sized for EOI so that no empty tankage would be carried through this ΔV. The other MOI tank set is sized to hold the remainder of the MOI propellant, which occupies a volume slightly greater than the TEI propellant requires. This second MOI tank set is staged after TEI, leaving a full tank set holding the EOI propellant that is reused for the next mission. A summary of the tank reuse/staging strategy analyses is shown in Figure 5-13. All alternative staging strategy scenarios required an increase in Mars ISPP requirements because empty tankage is carried on the return trip in each of these scenarios. However, the increase is relatively minimal for Scenario 7E, where the strategy focused on minimizing the acceleration of empty tankage. Of these scenarios, only 7E achieved a lower steady-state ELM than the baseline scenario, 7A, although this savings is small (approximately 3%).

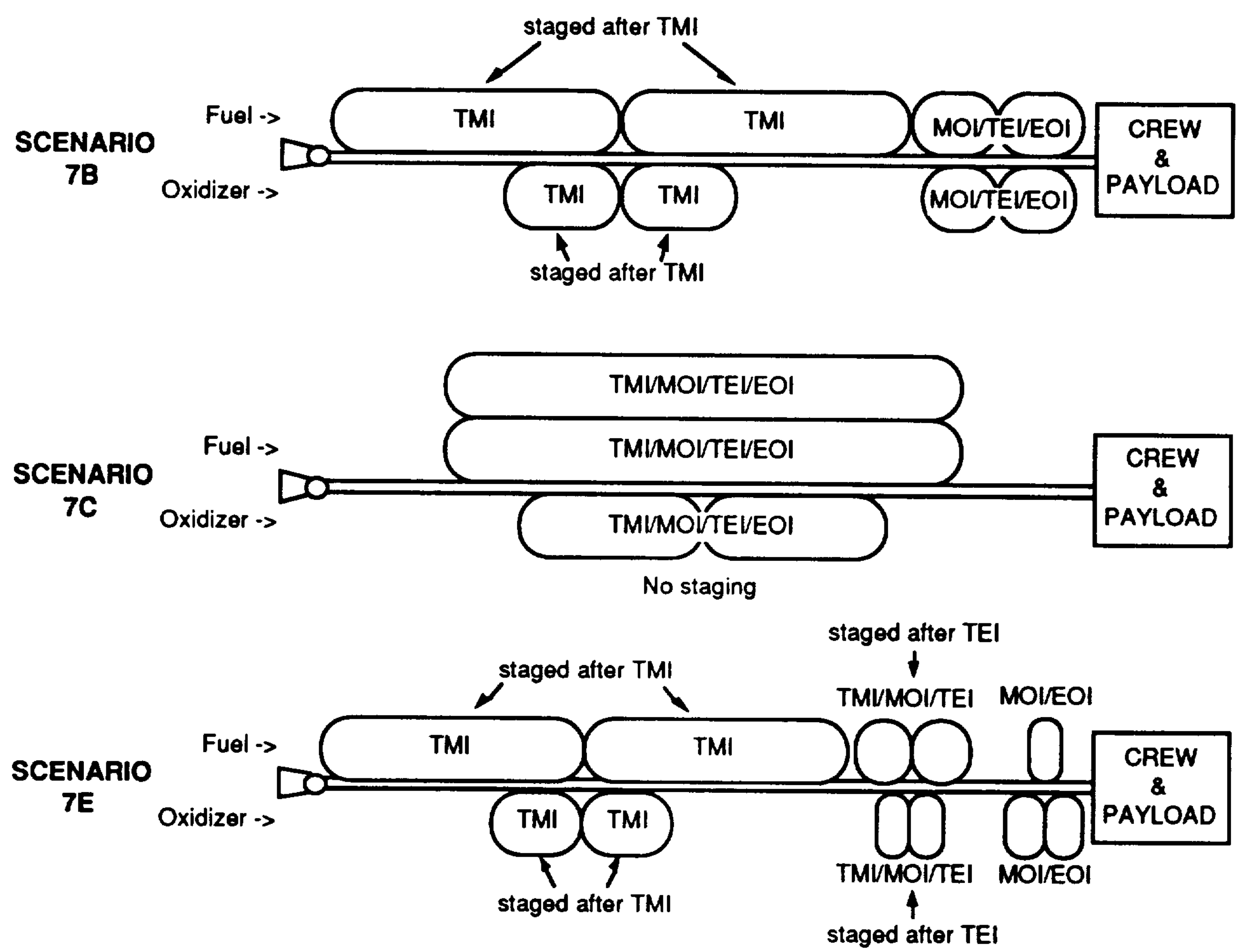


Figure 5-12. Alternative Tank Reuse/Staging Strategies

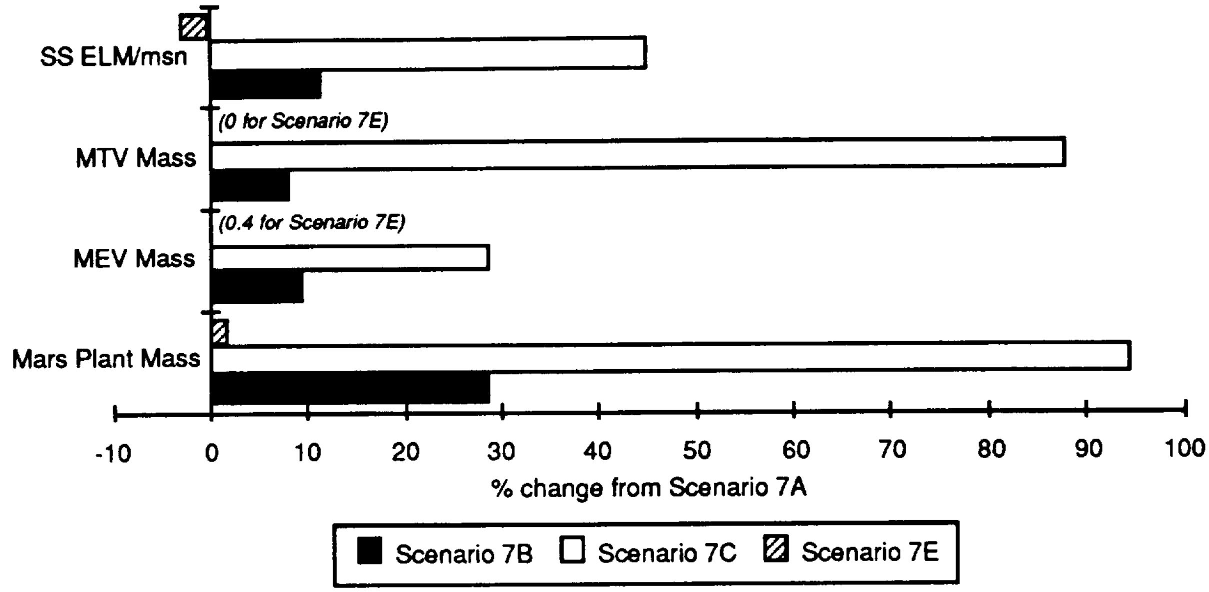


Figure 5-13. Results of Tank Reuse/Staging Strategy Analyses

5.3 Mission Performance Conclusions and Recommendations

From the final mission performance predictions, summarized in Table 5-3, steady-state ELM is reduced substantially if in situ lunar and Mars propellants are used to fuel the MTV and MEV. Plant masses, propellant masses, vehicle masses, and ELM are all lower in scenarios that utilize gas generator cycle engines rather than expander cycle engines, due to the substantially lower mass of the gas generator engines. For the LOX/CO/H_2 expander engine, going from a 400:1 nozzle area ratio to a 165:1 ratio does not significantly affect steady-state ELM. The mission performance assessments for Scenario 6 indicate that a 10% change in engine Isp has a greater performance impact than does a 10% change in engine mass. Propellant tank reuse can reduce ELM if the tanks are sized such that acceleration of empty tank volume is minimized as much as possible. However, completely reusing all propellant tanks for the entire mission (i.e., no tank staging), can significantly increase ELM. In terms of reducing steady state ELM, the most favorable scenario is Scenario 4, which utilizes lunar LOX/CH_4 and Mars LOX/CO. For all the scenarios, Earth-supplied propellant comprises a majority of ELM requirements.

It is recommended for further study that a comprehensive year-by-year performance assessment be performed that includes propellant plant set-up missions and vehicle change-out missions to characterize multimission performance. While propellant plant masses and vehicle masses were calculated, the requirements for emplacing these elements were not evaluated. This

Table 5-3. Summary of Final Mission Performance Data

I. ISPP System Mass Comparison (t)

	2A	2B	3A	3B	4A	4B	6A	6B	6C	6D	6E	6F	6G	7A	7B	7C	7D	7E
Lunar Propellant Plant	65	60	99	97	54	48	0	0	0	0	0	0	0	0	0	0	0	0
Mars Propellant Plant	472	417	60	56	476	398	467	525	412	482	453	285	972	55	71	107	53	56

II. Vehicle Mass Comparison

Transfer Vehicle (t)

	1A	1B	2A	2B	3A	3B	4A	4B	6A	6B	6C	6D	6E	6F	6G	7A	7B	7C	7D	7E
Engine	3.9	2.2	4.4	1.9	7.8	4.5	4.5	1.9	4.4	3.1	1.9	4.9	4	4.4	4.4	3.9	3.9	15.7	2.2	3.9
Crew Hab, Str. & Tanks	34.5	34.5	34.1	33.9	33.2	33	34.1	33.9	33.7	33.8	33.6	33.7	33.7	33.1	34.6	39.6	43.3	66.1	32.5	39.6

Mars Excursion Vehicle (t)

	1A	1B	2A	2B	3A	3B	4A	4B	6A	6B	6C	6D	6E	6F	6G	7A	7B	7C	7D	7E
Engine	3.8	2.1	86.8	34	22.4	11.6	89.2	37	86.8	59.4	34	95.4	78.2	86.8	156.2	22.4	22.4	22.4	11.6	22.4
Crew Hab, Str. & Tanks	13	12.7	58.2	52.8	26.6	25.4	58.6	50.8	57.6	63.2	52	59	56	39.2	107.2	25.2	29.8	39	24.4	25.4

Lunar Excursion Vehicle (t)

	2A	2B	3A	3B	4A	4B
Engine	8.6	3.7	3.8	2.1	8.9	3.7
Str. & Tanks	29.1	27.4	42.5	41.6	11.9	10.8

III. Steady-State Comparison (t)

	1A	1B	2A	2B	3A	3B	4A	4B	6A	6B	6C	6D	6E	6F	6G	7A	7B	7C	7D	7E
Lunar ISPP Support			1.6	1.5	2.5	2.5	1.1	1												
Mars ISPP Support			4.7	4.2	80.1	75.5	4.8	4	4.7	5.2	4.1	4.8	4.5	2.9	9.7	73.6	93.8	143.8	71.4	74.4
Earth-supplied prop.	1505	1510	427.1	395.7	646.1	632.5	239.8	214.1	444.5	453.8	408.9	448.4	433	337.2	753.4	649.8	731.6	989.6	643.3	630.5
Staged Tanks	65.1	65.5	42.2	40.1	54	52.8	32.3	29.8	29.2	30.2	27.7	29.4	28.7	23.5	44	33.4	22.2	0	33.2	26.7
Staged Engines	15.7	9.8	8.8	3.8	7.8	4.5	8.8	3.8	8.8	6.1	3.8	9.7	8	8.8	8.8	15.7	15.7	0	9	15.7
MEV Aeroshell	9	8.4	27.2	18	22	19.6	27.6	18.2	27	24	18	28.6	25.4	23.2	48	20.8	24.6	32.8	19	21
Crew and Consumables	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7	7
Mars Mission Payload	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25	25
TOTALS ->	1627	1626	543.6	495.3	844.5	819.4	346.4	302.9	546.2	551.3	494.5	552.9	531.6	427.6	895.9	825.3	919.9	1198	807.9	800.3

is a key consideration in evaluating in situ propellant use because, although the ISPP steady-state mission ELM may be considerably reduced over the non-ISPP case, the set-up and vehicle replacement requirements may be substantial and will affect the number of missions to ELM payback and savings over the course of multiple missions.

Other sensitivity analyses may improve insight into understanding the impacts on mission performance of ISPP requirements, vehicle design, and mission design. Trades can be run to investigate the effect of lower lunar and Mars ISPP plant masses on required ELM. Also, possible engine improvements that may increase Isp without significantly increasing engine mass should be investigated. Tank sizing and staging strategies should also be more closely examined, including the possibility of using common-sized tanks for all the vehicles. Also, the use of aerocapture at Mars and Earth should be considered. Other possibilities for improved performance would be to base the MTV in LLO, so that is does not have to be boosted out of LEO for each mission, or to transport lunar propellant to LEO, so that the MTV would not have to go to the Moon at all. Most importantly, enhancement of our understanding of the ISPP requirements, through laboratory studies on Earth and technology investigations on the lunar and/or Mars surface, is necessary to more accurately define mission performance improvements.

6.0 TECHNOLOGY MATURATION PLAN

A technology maturation plan has been established that addresses the development and demonstration of critical technologies and systems required to support a decision at the turn-of-the-century (year 2000+) to develop an operational Mars in situ propellant-based propulsion system. The technology research and development plan, as well as the technology assessment and major assumptions that support it, are discussed in the remainder of this section.

It was assumed that development of a Mars in situ propellant-based propulsion system would draw upon ongoing cryogenic space propulsion system technologies, see Ref. 6-1 and 6-2, and on technologies that address unique technology and design issues of such systems. This development consideration is displayed in Figure 6-1. The technology plan established in this study addresses only the technology and design developments required that are unique to Mars in situ propellant-based propulsion system. Many of the technologies and design issues for deep space cryogenic engines are also similar to those associated with engine systems of interest to this study. An example of this is the generic engine system characteristics associated with space-based engine systems, shown in Table 6-1, which are applicable to both cryogenic and Mars in situ propellant-based engine systems.

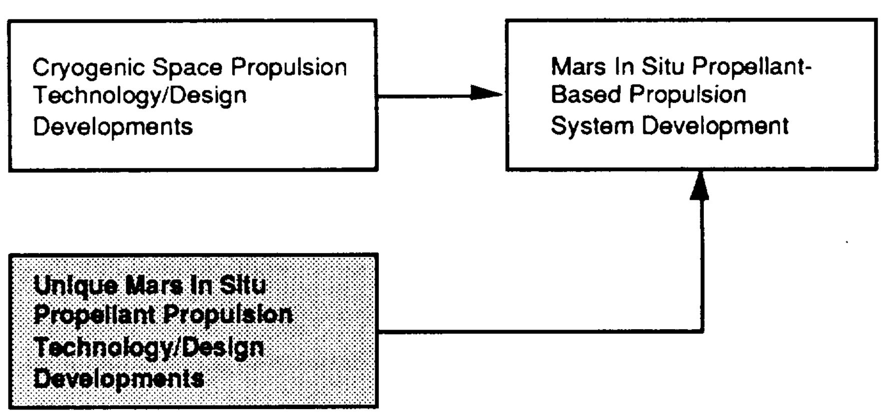


Figure 6-1. Development of a Mars In Situ Propellant-Based Propulsion System

Table 6-1. Engine System Characteristics to Meet Space Basing Requirements

• Automated Pre-Mission Checkout
• Real-Time Safety Monitoring
• Incipient Failure Mode Detection
• Post-Firing Trend Monitoring
• Long Duration Space Exposure
• Minimum Maintenance
• Engine Servicing in Space
• Replaceable Modular Systems/Robotic Engine Changeout
• Minimize Fluid Requirements

A technology readiness assessment was conducted in four fundamental engineering areas associated with development of Mars in situ propellant-based engine systems. The areas assessed involved: 1) materials compatibility, 2) cooling, 3) ignition/combustion and 4) pumping. The assessment was based on results associated with the technology review and engine system design analysis discussed in Sections 2.0 and 4.0, respectively, and by applying the NASA technology readiness level definition given in Table 6-2. Results of this assessment are presented in Table 6-3. For engine systems that use more conventional bipropellants such as LOX/H_2 and LOX/CH_4 technology readiness is very high. This is based on the extensive research and development experience associated with LOX/H_2 and LOX/CH_4 launch and upper stage/space engines over the past 30 years, as well as operational experience with LOX/H_2 engines systems. Bipropellant LOX/CO and tripropellant engine systems lack a strong experience base and are rated low (1 to 3) in terms of technology readiness in all of the key engineering areas.

Based on the propulsion system assessment reported in Section 4.0, an evaluation was performed by each major propulsion system, subsystem, or component to identify the technology improvements that may be required. These improvements were then rated in terms of their confidence to achieve the required goal. The results of this evaluation are presented in Table 6-4. The relative confidence rating is based on the probable difficulties to achieve the goal.

From the previous two assessments, just mentioned, key research and development issues were then identified and categorized. Table 6-5 summarizes these issues. These key issues are unique to Mars tripropellant propulsion systems. The issues are categorized as either being enabling or enhancing. An enabling issue is one that must be addressed and successfully demonstrated by one or more solutions to ensure the feasibility of a Mars in situ propellant-based

Table 6-2. Technology Readiness Levels

Basic Technology Research	LEVEL 1	BASIC PRINCIPLES OBSERVED AND REPORTED
Research To Prove Feasibility	LEVEL 2	TECHNOLOGY CONCEPT AND/OR APPLICATION FORMULATED
	LEVEL 3	ANALYTICAL & EXPERIMENTAL CRITICAL FUNCTION AND/OR CHARACTERISTIC PROOF-OF-CONCEPT
Technology Development	LEVEL 4	COMPONENT AND/OR BREADBOARD VALIDATION IN LABORATORY ENVIRONMENT
Technology Demonstration	LEVEL 5	COMPONENT AND/OR BREADBOARD VALIDATION IN RELEVANT ENVIRONMENT
	LEVEL 6	SYSTEM/SUBSYSTEM MODEL OR PROTOTYPE DEMONSTRATION IN A RELEVANT ENVIRONMENT (Ground or Space)
System/Subsytem Development	LEVEL 7	SYSTEM PROTOTYPE DEMONSTRATION IN A SPACE ENVIRONMENT
System Test, Launch and Operations	LEVEL 8	ACTUAL SYSTEM COMPLETED AND "FLIGHT QUALIFIED" THROUGH TEST AND DEMONSTRATION (Ground or FLight)
	LEVEL 9	ACTUAL SYSTEM "FLIGHT PROVEN" THROUGH SUCCESSFUL MISSION OPERATIONS

Table 6-3. Technology Readiness of Fundamental Research Issues Associated With In Situ Mars Propellant-Based Engines

Propellant Combinations	TECHNOLOGY READINESS LEVEL			
	Materials Compatibility	**Cooling**	**Ignition/ Combustion**	**Pumping**
Bipropellants				
LOX/H_2	9	9	9	9
LOX/CO	1-2	1-2	1-2	1-2
LOX/CH_4	5-6	5-6	5-6	5-6
Tripropellants				
$LOX/H_2/CO$	1	1	1	1
$LOX/H_2/CH_4$	3	3	3	2
$LOX/CO/CH_4$	1	2	1	1

Table 6-4. Propulsion System Subsystem/Component Evaluation

Subsystem or Component	Options	Relative Confidence to Achieve Improvements(s)*
Propellant Tankage	• Lightweight structure, and meteoroid shell, high performance insulation	High
	• Common fuel tankage	Medium
	• Common propellant tankage - Fuel and oxidizer	Low
	• Lightweight, inflatable propellant tankage	Low
	• Integrated, high performance tank/refrigeration	Low
Feed System	• Lightweight, reliable, highly throttleable turbopumps	Medium
	• Common fuel turbopumps	Medium
	• Common turbopumps - Fuel and oxidizer	Low
	• Lightweight, common propellant lines, valving - Compatible composite structures/materials	High
	• Turbine drive systems using multiple fuel-rich, high-temperature gases	Low
	• Stoichiometric gas generator	High
	• High temperature turbine materials for oxygen-rich drive gases	Medium
Injector	• Common, high performance multiple propellant injector design - At design and throttled conditions	Low

* Low = Difficult; Medium = Moderate Difficulty; High = Low Difficulty

Table 6-4. Propulsion System Subsystem/Component Evaluation (Cont.)

Subsystem or Component	Options	Relative Confidence to Achieve Improvements(s)*
Thrust Chamber	• High performance, oxygen cooled thrust chamber	Medium
	• High performance and high chamber pressure cooled thrust chamber	Low
	• Common multiple fuel cooled thrust chamber	Medium
	• Common fuel or oxygen cooled thrust chamber	Low
Nozzle	• Lightweight nozzle extension	High
	• Lightweight translate nozzle extension	Medium
Control System	• Lightweight, radiation environment compatible	High
	• Highly robust, adaptive control system to support multimode engine operation with various propellant combinations	High
	• Sensors compatible with more than one propellant	Medium
Mounts and Support	• Lightweight thrust mounts and supports	High
	• Highly integrated feed system/thrust mount support system design	Medium

* Low = Difficult; Medium = Moderate Difficulty; High = Low Difficulty

Table 6-5. Key Research and Development Issues

Issues	Rationale/ Comments	Type (Enabling or Enhancing)
• Materials		
- Compatibility		
– CO	Little data available on CO at high temperature and pressure conditions	Enabling*
– LOX	Additional research required to identify materials that are compatible with LOX at temperature higher than present day options	Enhancing
	- Turbine materials - Thrust chamber materials } Improved Performance	
- Common Multipropellants	Little or no data available	Enhancing
– CO/H_2 – H_2/CH_4 – CO/CH_4	- Tank materials which support common tank designs - Common pumping/ cooling engine systems } Reduced Weight	

* Impacts Mission Scenarios 2, 4, and 6 Only.

Table 6-5. Key Research and Development Issues (Cont.)

Issues	Rationale/ Comments	Type (Enabling or Enhancing)
• Cooling - CO	Little fundamental data available on CO cooling at high heat flux and pressure conditions	Enabling*
• Ignition/Combustion - LOX/CO	Little fundamental data available on the ignition and combustion of LOX/CO at the conditions of interest	Enabling*
• Pumping - CO	Little fundamental data available on pumping of CO at the conditions of interest	Enabling*
• Common Multipropellant Injector Design	Little design data available associated with main injector and gas generator (preburner) designs that can operate with more than one propellant combination of interest over a wide operating range (required for throttling)	Enabling
• Common Multipropellant Feed System/Turbopump Design	Design database lacking to support design of a common pump-fed (including turbopump) feed system that can efficiently pump more than one fuel of interest over a wide operating range	Enhancing
• Common Thrust Chamber Design	Design database lacking to support design of a common thrust chamber that is cooled by more than one propellant over the operating range of interest	Enabling
• Ignition/Gas Generator Design - LOX/CO	Little data available associated with design and operation of a LOX/CO gas generator at low temperature and pressure operating conditions	Enhancing*
• Common Control/Health Monitoring System	Little experience available associated with the design and operation of control/health monitoring system for an engine system that uses different propellant conditions during various operating modes	Enabling
• Common Propellant Tank Design and Supporting Operations	Little experience/design database available on the design and operations (such as refilling in space) of tanks that can store more than one propellant of interest	Enhancing
• Lightweight, Compact High Area Ratio Nozzle Design	Low chamber pressure in situ Mars propellant-based engines may require high weight translating high area ratio nozzle or an alternative design due to packaging constraints	Enhancing*

* Impacts Mission Scenarios 2, 4, and 6 Only.

propulsion system. If an engineering solution cannot be found for a given issue, development of the propulsion system will not be possible. An enhancing issue addresses area(s) of possible improvements, over the state-of-the-art engineering solution, that can produce a high payoff typically in areas of performance, mass savings, and/or mission flexibility, for example. Some of the issues identified in Table 6-5 are associated with propulsion systems that employ only CO as a fuel. Many of the issues address common multipropellant combustion hardware component design that is critical for the proposed MTV propulsion systems.

It should be noted that there are many research and development issues which are characterized as enabling in Table 6-5. This should not be interpreted that high-risk technology breakthroughs are required in these areas to develop a Mars tripropellant propulsion system. Presently, many of these issues lack an adequate technology base. These issues can be successfully addressed by implementing focused technology development programs in these areas.

A technology development plan was then defined that addresses the key technology/design issues given in Table 6-5 as well as demonstrates the feasibility of the Mars tripropellant engine system concept employing extensive common engine system hardware. Tables 6-6 and 6-7 list the major planning assumptions and key areas to be addressed, respectively, which are associated with the technology development plan. As previously mentioned, the technology development plan draws on ongoing space propulsion technology developments and only addresses technology and design issues associated with Mars tripropellant propulsion systems.

Table 6-6. Major Assumptions in Defining Technology Development Plan

- Development decision associated with Mars in situ propellant propulsion systems will be made at the turn of the century (year 2000)
- Technology available from other propulsion areas (such as advanced LOX/H_2 space engines) will be available to support development of Mars in situ propellant-based propulsion systems
- Existing United States and possibly world propulsion system development testing facilities will be available to support development of Mars in situ propellant-based propulsion system
  - No new major testing facilities required, only modification/ upgrading of current facilities will be required

Table 6-7. Key Areas to Be Addressed by the Technology Development Plan

• Establish fundamental database associated with candidate propellant and material options
• Investigate feasibility of common propulsion system hardware design approach
• Demonstrate overall in situ Mars propellant engine system feasibility
• Assess the impact of engine system technology capabilities on overall mission architecture and vehicle design

The technology development plan is comprised of four major phases. They are: 1) fundamental research, 2) exploratory development, 3) breadboard engine system demonstration, and 4) system engineering studies. Table 6-8 summarizes these major phases. The first three phases focus on propulsion system technology/design issues, while the other provides the overall systems engineering/integration development function. In this development phase emerging mission, vehicle and engine system designs are identified and assessed as new technology data becomes available from the other technology plan development phases. Figure 6-2 shows the overall technology development plan process, which would last for 7 years from go-ahead. If the initial program go-ahead were approved for Government Fiscal Year 1993, a flight system development decision at Fiscal Year 2000 could be supported by the proposed technology development plan program. For each technology plan development phase, programs addressing key technology/design issues were defined. Table 6-9 summarizes these programs. Detailed descriptions of each technology development plan program element are given in Appendix G. Figure 6-3 provides an overall technology development plan schedule and the estimated required funding by program element and fiscal year to accomplish it.

The overall funding required for the 7-year maturation plan is approximately $104 million. The initial program funding requirements for the first 2 years is a little over $3 million per year which focuses on the fundamental research aspects of development. At the conclusion of this development phase if major fundamental research issues are still outstanding, the Mars common tripropellant propulsion system approach should be completely be reassessed. If after this development phase, results look encouraging, an exploratory development and a breadboard engine system demonstration would then be initiated, as shown in Figure 6-3. Yearly funding requirements would then increase (ramp up) accordingly to a maximum of $26.5 million in the fifth year of the technology maturation plan. At the conclusion of this program, necessary data should be available to establish the feasibility of Mars in situ propellant-based propulsion systems and provide the insight to make a knowledgeable decision to develop an operational flight system.

It is estimated that 5 to 7 years would be required to develop and certify a flight engine system if the development decision is approved. Based on the technology plan just discussed, the earliest initial operational capability of such an engine system would be in FY2005.

Table 6-8. Summary of Goal(s) and Activities by Development Phase

Development Phase	Goal(s)	Activities
Fundamental Research	Establish fundamental material, thermal-hydraulic and combustion databases to support definition and evaluation of component, subsystem and propulsion system concepts	Fundamental experimental and theoretical studies are performed in the areas of: • Materials compatibility • Cooling • Combustion/ignition • Pumping • Other(s)
Exploratory Development	Demonstrate promising technologies and designs (components and/or subsystems) that can support development of high performance, lightweight, reliable engine system(s) that use Mars in situ propellants	Design manufacturing and component/ subsystem testing: • Injector(s) • Turbopump(s) • Thrust chamber(s) • Ignition/gas generator design(s) • Control/health monitoring system(s) • Common tankage system(s) • Translating high area ratio nozzle(s)
Breadboard Engine System Demonstration	Demonstrate one or more complete engine system concepts	Design, manufacture and test one or more promising engine system concepts that use Mars in situ-based propellants; tests will examine the following areas: • Thrust range (throttling) • Duty cycle compatibility • Startup/shutdown/throttling characteristics • Performance • Life • Multipropellant compatibility
Systems Engineering	Provide propulsion system requirements and guidance in identifying critical technologies and design concepts, and their impact on the overall mission and vehicle design	Mission and vehicle system design studies as well as assessments of emerging propulsion system concepts and their supporting technologies

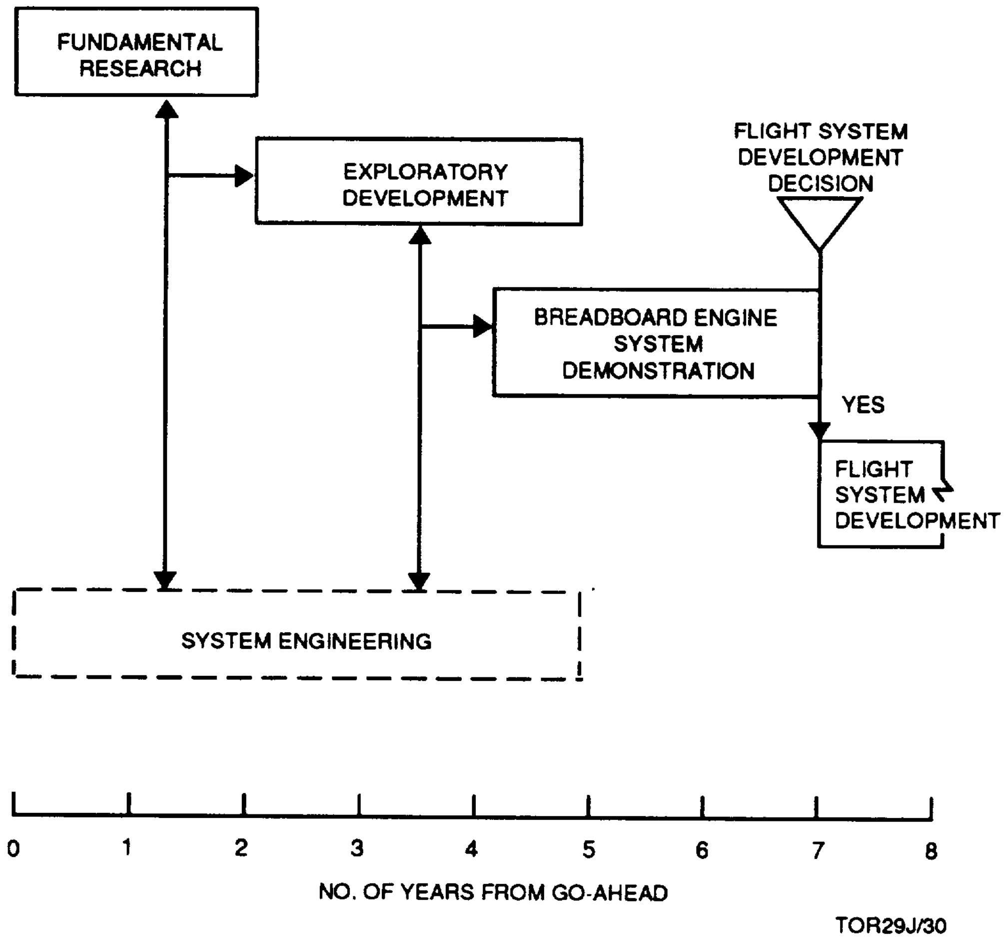


Figure 6-2. Overall Process to Support Development of an In Situ Mars Propellant-Based Propulsion System

Table 6-9. Summary of Technology Development Plan Program

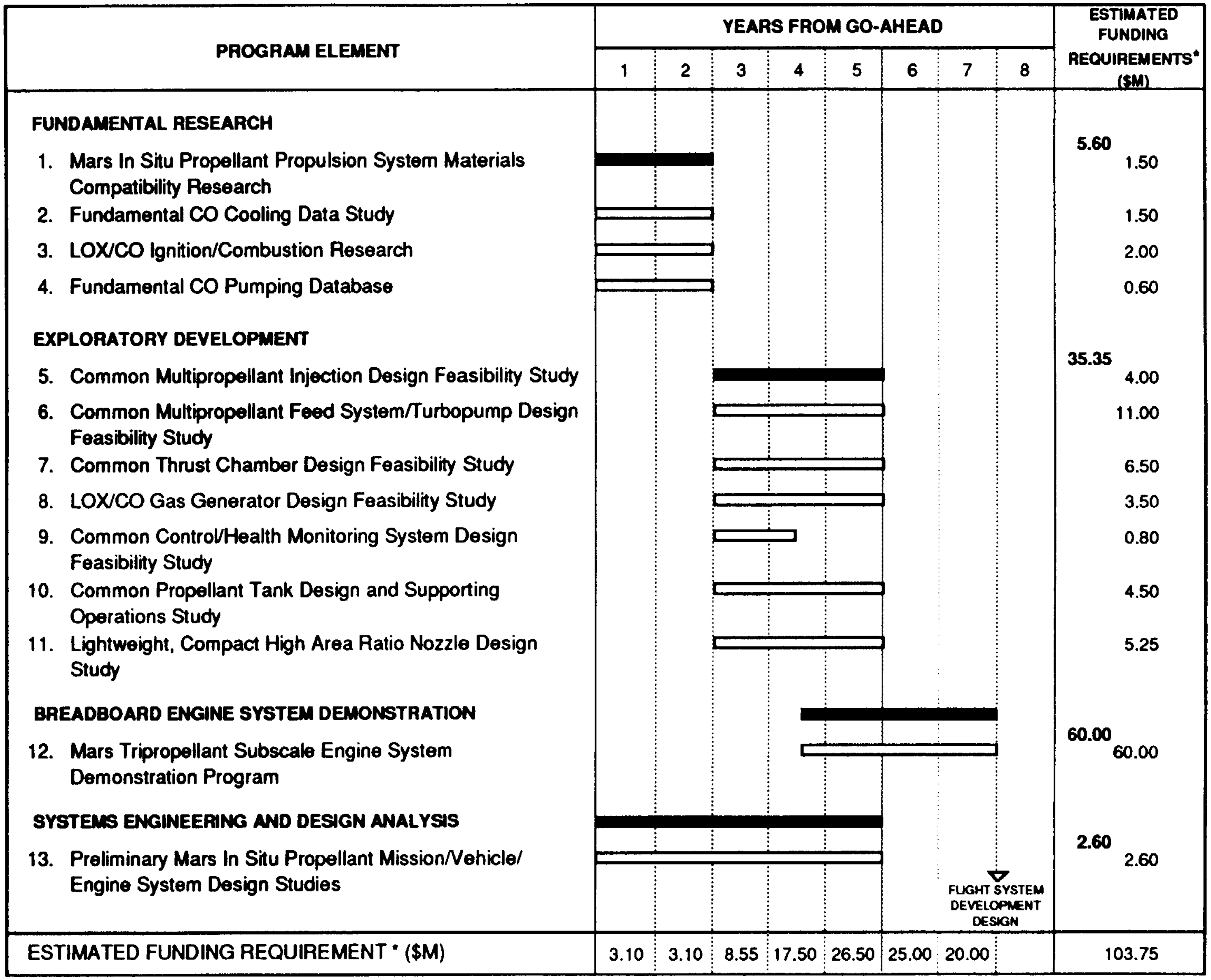
Program No.	Title	Development Phase*	Objective(s)
1	Mars In Situ Propellant Propulsion System Materials Compatibility Research	FR	Identify propulsion system material candidates that are compatible with potential Mars in situ propellants and/or propellant combinations. Propellants and/or propellant combinations for which material compatibility should be investigated include: CO/LOX, CO/H_2, H_2/CH_4, CO/CH_4
2	Fundamental CO Cooling Data Study	FR	Establish a fundamental database associated with CO cooling for conditions that are typical of thrust chambers and turbopumps

* FR = Fundamental Research; ED = Exploratory Development; BED = Breadboard Engine Demonstration; SE = System Engineering

Table 6-9. Summary of Technology Development Plan Program (Cont.)

Program No.	Title	Development Phase*	Objective(s)
3	LOX/CO Ignition/ Combustion Research	FR	Establish a fundamental database associated with LOX/CO injection and combustion for conditions typical of an engine system
4	Fundamental CO Pumping Database	FR	Establish a CO pumping database for the range of conditions typical of a LOX/CO engine
5	Common Multipropellant Injector Design Feasibility Study	ED	Establish feasibility and identify promising injector design(s) that can operate with more than one Mars in situ-based propellant combination over a wide operating range. Main injector and gas generator designs are to be investigated
6	Common Multipropellant Feed System/Turbopumps Design Feasibility Study	ED	Establish feasibility and identify promising feed system/turbopump design(s) that can operate efficiently with more than one Mars in situ-based fuel over a wide operating range
7	Common Thrust Chamber Design Feasibility Study	ED	Establish feasibility and identify promising thrust chamber design(s) that can operate with more than one Mars in situ-based propellant over a wide operating range
8	LOX/CO Gas Generator Design Feasibility Study	ED	Establish feasibility and identify LOX/CO gas generator design(s) that can operate over a wide range of operating conditions
9	Common Control/Health Monitoring System Design Feasibility Study	ED	Establish feasibility and identify promising common control/health monitoring system(s) that can operate with numerous in situ Mars propellant combinations for various engine system operating modes
10	Common Propellant Tank Design and Supporting Operations Study	ED	Establish feasibility and identify common propellant tank design(s) and supporting operation requirements and design approach(es), such as for resupply. Identification of high payoff alternative tank designs will also be considered
11	Lightweight, Compact High Area Ratio Nozzle Design Study	ED	Identify lightweight, compact high area ratio nozzle designs for Mars in situ tripropellant engine systems employing LOX/CO as one of its two propellant combinations
12	Mars Tripropellant Subscale Engine System Demonstration Program	BED	Successfully demonstrate and establish feasibility of a subscale (15,000-60,000 lbf thrust level) candidate Mars in situ propellant-based tripropellant engine system design concept
13	Preliminary Mars In Situ Propellent Mission/Vehicle/ Engine System Design Studies	SE	Assess the impact of engine technology data as it becomes available, on evolving Mars in situ propellant-based mission, vehicle and engine system designs

* FR = Fundamental Research; ED = Exploratory Development; BED = Breadboard Engine Demonstration; SE = System Engineering



* Assumes 1992 Dollars

TOR29J/35

Figure 6-3. Overall Technology Development Plan Schedule and Required Funding

7.0 CONCLUSIONS

A top-level feasibility study was conducted that identified and characterized promising chemical propulsion system designs which use two or more of the following propellant combinations: LOX/H_2, LOX/CH_4 and LOX/CO. The engine systems examined emphasized the usage of common subsystem/component hardware where possible. In support of this study, numerous mission scenarios were characterized that used various combinations of Earth, lunar and Mars propellants to establish engine system requirements to assess the promising engine system design concept examined, and to determine overall exploration leverage of such systems compared to state-of-the-art cryogenic (LOX/H_2) propulsion systems. Initially in the study, critical propulsion system technologies were assessed. Candidate expander and gas generator cycle $LOX/H_2/CO$, $LOX/H_2/CH_4$ and $LOX/CO/CH_4$ engine system designs were parametrically evaluated. From this evaluation baseline, tripropellant MTV LOX cooled and bipropellant LEV and MEV engine systems were identified. Representative tankage designs for a MTV were also investigated. Re-evaluation of the missions using the baseline engine design showed that in general the slightly lower performance, smaller, lower weight gas generator cycle-based engines, required less overall mission Mars and ISPP infrastructure support compared to the larger, heavier, higher performing expander cycle engine systems.

Additionally, the study identified key technology and design issues that must be addressed to ensure the technical feasibility of such engine systems. A 7-year technology maturation plan was established that would address these issues in an efficient manner.

It is recommended in the near-term, that additional tripropellant engine system design studies be undertaken that consider propellants other than LOX as the engine system coolant. By assuming LOX as the coolant in engine systems examined in this study, chamber pressure was limited. Engines that employ the candidate fuel as their coolant may have the potential to operate at higher chamber pressures, hence possibly reducing the engine's size and weight substantially, for a given thrust level. In parallel with this effort, it is recommended that a robust fundamental research program in the areas of materials compatibility, cooling, ignition/combustion and pumping be initiated as discussed in the technology maturation plan. This data is critical in the assessment of candidate tripropellant engine systems. Due to highly coupled interrelationship of the propulsion system, which uses in situ-derived lunar and/or Mars propellants, with the vehicle and ISPP infrastructure, additional mission/vehicle design studies are also recommended at this time.

8.0 REFERENCES

1-1. "Report of the 90-Day Study on Human Exploration of the Moon and Mars," NASA Publications, November 1989.

1-2. "Report of the Synthesis Group on America's Space Exploration Initiative," U.S. Government Printing Office, May 1991.

1-3. Linne, D. L. and Meyer, M. L., "Technical Prospects for Utilizing Extraterrestrial Propellants for Space Exploration," IAA-91-669, presented at the 42nd Congress of the International Aeronautical Federation, Montreal, Canada, October 5-11, 1991.

1-4. Stancati, M. L., Jacobs, M. K., Cole, K. J., and Collins, J. T., "In Situ Propellant Production: Alternatives for Mars Exploration," NASA CR 187192; SAIC Report No. SAIC-91/1052, October 1991.

2-1. "Space Transfer Concepts and Analysis for Exploration Missions – Third Quarterly Review," Boeing Aerospace and Electronics Co., NASA Contract No. NAS8-37857, June 22, 1990.

3-1. Beichel, R. and O'Brien, C. J., "Dual-Fuel Engine for Single-Stage Shuttles," AIAA Paper No. 75-1249, presented at the AIAA/SAE 11th Propulsion Conference, Anaheim, California, September 29-October 1, 1975.

3-2. O'Brien, C. J., "Dual-Fuel, Dual-Throat Engine Preliminary Analysis," Aerojet Liquid Rocket Company, Report 32967F, Contract NAS-8-32967, August 1979.

3-3. O'Brien, "Dual-Nozzle Design Update," AIAA-82-1154, presented at the AIAA/SAE/ASME 18th Joint Propulsion Conference, Cleveland, OH, June 21-23, 1982.

3-4. Beichel, R., O'Brien, C. J. and Swiss, J. D., "Single-Stage-to-Orbit Advanced Propulsion," Aerojet Liquid Rocket Company, IR&D Program 8776-06, Report No. 8776-06-F, November 1976.

3-5. Wheeler, D. B., "Tripropellant Engine Study," NASA-CR-150482, December 9, 1977.

3-6. LaBotz, R. J. and Ewen, R. L., "Thrust Chamber Technology for Advanced Earth-to-Orbit Rocket Engines," Aerojet Technology, Vol. 1, No. 1, 1983.

3-7. Schoeman, L., "High-Pressure Propulsion – Advanced Concepts for Cooling," Acta Astronautica, Vol. 14, 1986.

3-8. Visek, W. A., "Future LOX/Hydrocarbon Booster Engines," SAE Paper No. 861813, presented at the Aerospace Technology Conference and Exposition, Long Beach, CA, October 13-16, 1986.

3-9. Worlund, A. L., Monk, J. C. and Bachtel, F. D., "NLS Propulsion Design Considerations," AIAA-92-1181, presented at the 1992 Aerospace Design Conference, Irvine, CA, February 3-6, 1992.

3-10. Cooper, L. A., "Propulsion Issues for Advanced Orbit Transfer Vehicles," NASA TM 83624, 1984.

3-11. Spencer, R. G., Rousar, D. C. and Price, H. G., "LOX-Cooled Thrust Chamber Technology Developments," Journal of Spacecraft, Vol. 17, No. 1, Jan-Feb 1980.

3-12. Dederra, H. and Kirner, E., "High Pressure Rocket Engine Liquid Oxygen Technology," IAF-76-174, presented at the 1976 International Astronautical Federation Conference, Anaheim, CA, October 10-16, 1976.

3-13. Roncace, E. A., "In Situ Propellant Rocket Engines for Mars Mission Ascent Vehicle," AIAA-91-2445, presented at the AIAA/SAE/ASME/ASEE 27th Joint Propulsion Conference, Sacramento, CA, June 24-27, 1991.

3-14. Armstrong, E. S., "Cooling of In Situ Propellant Rocket Engines for Mars Mission," NASA TM 103729, January 1991.

3-15. Linne, D. L., Roncace, R. and Groth, M. F., "Mars In Situ Propellants: Carbon Monoxide and Oxygen Ignition Experiments," AIAA-90-1894, presented at the AIAA/SAE/ASME/ASEE 26th Joint Propulsion Conference, Orlando, FL, July 16-18, 1990.

3-16. Morgan, D. and Beichel, R., "Stoichiometric Gas Generator – A Strategic Departure," AIAA-91-2584, presented at the AIAA/SAE/ASME/ASEE 27th Joint Propulsion Conference, Sacramento, CA, June 24-26, 1991.

3-17. Buckmann, P. S., Shimp, M. R., Viteri, F. and Proctor, M., "Design and Test of an Oxygen Turbopump for a Dual Expander Cycle Rocket Engine," AIAA-89-2305, presented at the AIAA/ASME/SAE/ASEE 25th Joint Propulsion Conference, Monterey, CA, July 10-12, 1989.

3-18. Kurth, R., Private Communication, SAIC, June 1991.

3-19. Schoenman, L., "Selection of Burn -Resistant Materials for Oxygen-Driven Turbopumps," Journal of Propulsion and Power, Vol. 3, No. 1, Jan-Feb 1987.

3-20. Daniels, N. H. G., Syrell, B. C. and Jones, R. L., "Materials Requirements for Advanced Energy Systems – New Fuels, Volume 1, Interactions of Materials with New Fuels – Final Report," DARPA Contract No. DAHC15-73-C-0313, Stanford Research Institute, Menlo Park, CA, July 1974.

4-1. Taylor, C. E., "Expanded Liquid Engine Simulation Computer Program – Technical Information Manual," Aerojet Tech Systems Company, Report No. ELES-1984, August 1984.

4-2. Taylor, C. E., "Expanded Liquid Engine Simulation Computer Program – Programmer's Manual," Aerojet Tech Systems Company, Report No. ELES-1984, December 1984.

4-3. Taylor, C. E., "Expanded Liquid Engine Simulation Computer Program – Advanced User's Manual," Aerojet Tech Systems Company, Report No. ELES-1984, December 1984.

4-4. Mog, R. A. and Price, D. M., "Optimization Techniques Applied to Passive Measures for In-Orbit Spacecraft Survivability – Final Report," NASA Contract No. NAS8-37378, June 1991.

6-1. Zurawski, R. L., "Chemical Propulsion Technologies for Space Transfer Vehicles," presented at the AIAA/SAE/ASME/ASEE 26th Joint Propulsion Conference, Orlando, FL, July 16, 1990.

6-2. Stone, J. R., Shaw, L. M. and Aukerman, C. A., "Plans for the Development of Cryogenic Engines for Space Exploration," AIAA-91-3438, presented at the AIAA/NASA/OAI Conference on Advanced SEI Technologies, Cleveland, OH, September 4-6, 1991.

APPENDIX A

INITIAL MISSION REQUIREMENTS DATA

APPENDIX A
INITIAL MISSION REQUIREMENTS DATA

This appendix contains summary data of the results from the initial mission performance analysis. Three outputs characterize each of the seven scenarios investigated:

1. Mission Description and Assumptions – describes the sequence of mission events, identifies required infrastructure elements and steady-state Earth launch requirements, and states major assumptions made.
2. Mass ΔV, Specific Impulse (vacuum), Thrust, and Burn Time Summaries Arranged by Burn.
3. Engine Requirements Arranged by Vehicle.

These requirements provided a starting point for the engine system design effort and used rough engine performance and mass estimates. Section 2.0 summarized these efforts, and Section 5.0 contained the mission performance results using the specific propulsion system designs described in Section 4.0.

SCENARIO 1

Baseline Scenario (No Lunar/Mars Propellant): Earth LOX/H_2

Mission Requirements for Engine System Assessment Study
Scenario 1: Baseline Scenario (No Lunar/Mars Propellant): Earth LOX/H2

Mission Description

1 - MTV departs LEO loaded with propellant for the round trip, the Mars mission payload and crew, and propellant for the MEV

2 - the MTV rendezvous with the MEV in LMO and transfers crew, mission payload, and propellant to the MEV

3 - the MEV descends to the surface, performs the mission (approximately 30 day stay time), and returns the crew to the MTV in LMO

4 - the MTV returns to LEO

Required infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Propellant for: MTV and MEV round trips

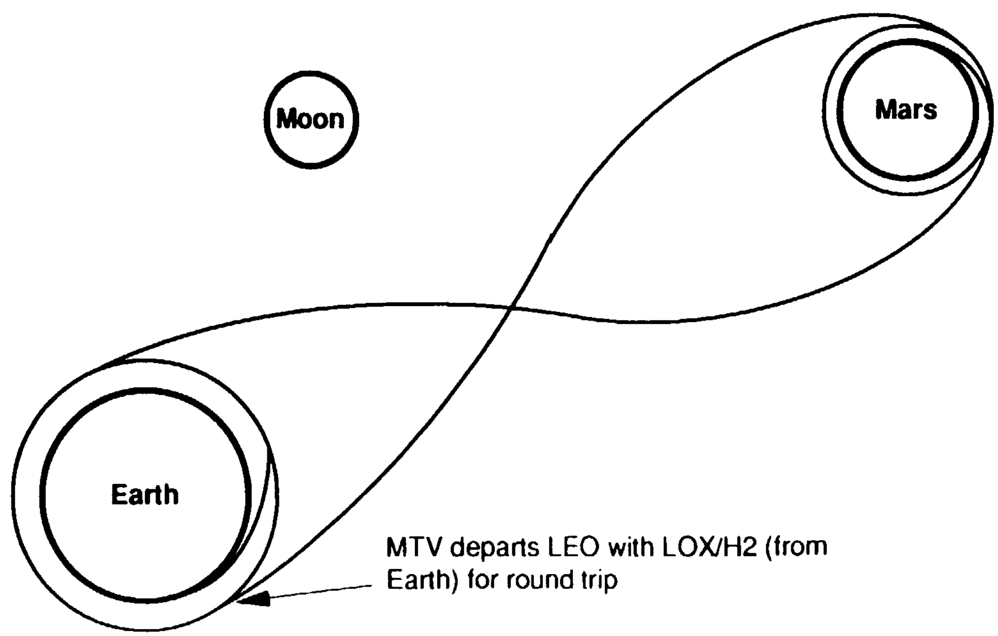
Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of the 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) 6.0

Scenario 1: **BASELINE SCENARIO (NO LUNAR/MARS PROPELLANT): EARTH LOX/H2**

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	MTV	LEO	1450	869	3982	93	4075	5814	470	1000	882
MOI	MTV	LMO	534	241	2590	48	2638	1744	470	300	804
Mars ascent	MEV*	Mars Surf.	42	31	5300	29	5329	465	470	80	376
Mars desent	MEV*	LMO	83	16	930	1	931	465	470	80	198
TEI	MTV	LMO	195	85	2521	6	2527	1744	470	300	283
EOI	MTV	LEO	105	65	4081	6	4087	1744	470	300	214

* These numbers are for each of 2 MEVs



MTV departs LEO with LOX/H2 (from Earth) for round trip

Scenario 1: BASELINE SCENARIO (NO LUNAR/MARS PROPELLANT): EARTH LOX/H2

	TRANSFER VEHICLES				EXCURSION VEHICLES		
	Expendable Stage	MTV			MEV		LEV
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Earth LOX/H2	uses Earth LOX/H2		
Propellants Used		Earth LOX/H2		Earth LOX/H2	uses Earth LOX/H2		
Specific Impulse (sec)		470		470	470		
Mixture Ratio (O/F)		6.0		6.0	6.0		
Thrust Level(s) (klbs)		1000 - TMI 300 - MOI		300	80		
Engine Operating Time (%of trip)		0.008%		0.002%	0.9%	0.5%	
Total ΔV (m/sec)		4,075 - TMI 2,638 - MOI		2,527 - TEI 4,087 - EOC	5,329	931	
Total Impulse (x10^6 kN sec)		3.923 - TMI 1.073 - MOI		0.379 - TEI 0.286 - EOC	0.134	0.070	
Maximum Acceleration (g's)		0.757/0.451		1.21/3.14	2.719	0.5319	
Operating Time (sec/mission)		882 - TMI 804 - MOI		284 - TEI 214 - EOC	376	198	
Reusability (# of missions)		5		5	5	5	
Refueling Requirements		refueled in LEO		refueled in LEO	refueled in LEO		

SCENARIO 2

Lunar LOX (Earth H_2) for Outbound + Mars LOX/CO for Return

Mission Requirements for Engine System Assessment Study
Scenario 2: Lunar LOX (Earth H2) for Outbound + Mars LOX/CO for Return

Mission Description

1 - MTV departs LEO on an expendable LOX/H2 stage to LLO carrying refurbishment/resupply for the lunar and Mars propellant plants, hydrogen for the outbound leg, the Mars mission payload and crew

2 - the LEV meets the MTV in LLO, transfers LLOX to the MTV, and obtains refurbishment/resupply for the lunar plant before returning to the lunar surface. The LEV also gets fueled with Earth supplied H2 from the MTV.

3 - the MTV leaves LLO and transfers to LMO, is met by the MEV in LMO, and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant

4 - after completing the mission, the MEV delivers the crew and MLOX/CO for the return leg to the MTV in LMO and the MTV returns to LEO

Required infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant
- Lunar Excursion Vehilce (LEV)
- Lunar Propellant Plant
- LEO → LLO Expendable Booster

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Lunar Plant Refurbishment/Resupply
- Propellant for: LEO→LLO, LLO→LMO and LEV round trip (hydrogen and tanks only)

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

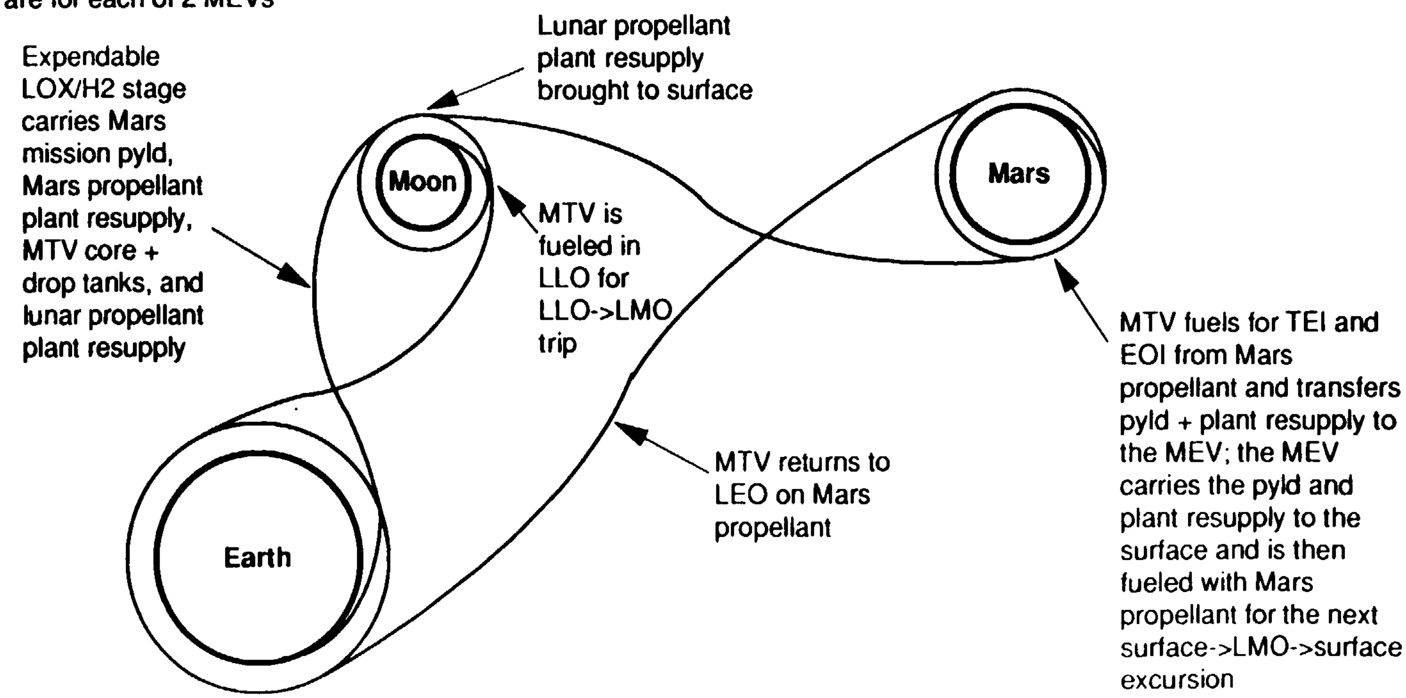
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

  LOX/CO: Thrust/Weight = 98 lbf/lbm; Isp = 290 sec; Mixture Ratio (O/F) = 0.6

Scenario 2: **LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CO FOR RETURN**

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	508	267	3300	26	3326	2907	470	500	541
LOI	Exp. stg.	LLO	230	50	1110	0	1110	2907	470	500	102
Lunar ascent	LEV	Lunar Surf.	378	135	1900	11	1911	1744	470	300	443
Lunar descent	LEV	LLO	95	35	2000	1	2001	1744	470	300	116
TMI	MTV	LLO	322	116	2005	6	2011	1744	470	300	393
MOI	MTV	LMO	202	89	2590	7	2597	1744	470	300	298
Mars ascent	MEV*	Mars Surf.	3454	3082	5300	56	5356	16196	290	3500	535
Mars desent	MEV*	LMO	156	46	930	0	930	16196	290	3500	8
TEI	MTV	LMO	513	310	2521	30	2551	1744	290	300	648
EOI	MTV	LEO	194	151	4081	12	4093	1744	290	300	316

* These numbers are for each of 2 MEVs



Scenario 2: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX	uses Mars LOX/CO	uses Mars LOX/CO		uses lunar LOX	
Propellants Used	Earth LOX/H2		lunar LOX + Earth H2	Mars LOX/CO	Mars LOX/CO		lunar LOX + Earth H2	
Specific Impulse (sec)	470		470	290	290		470	
Mixture Ratio (O/F)	6.0		6.0	0.6	0.6		6.0	
Thrust Level(s) (klbs)	500		300	300	3500		300	
Engine Operating Time (% of trip)	0.21%		0.0032%	0.0037%	1.2%	0.019%	1.0%	0.27%
Total ΔV (m/sec)	4,436		4,608	6,644	5,356	930	1,911	2,001
Total Impulse (x10^6 kN sec)	1.430		0.922	1.286	8.329	0.125	0.591	0.155
Maximum Acceleration (g's)	0.919/1.25		0.654/1.19	0.65/2.95	3.02	14.13	0.546	2.2
Operating Time (sec/mission)	643		691	964	535	8	443	116
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

SCENARIO 3

Lunar LOX (Earth H_2) for Outbound + Mars LOX/CH_4 for Return

Mission Requirements for Engine System Assessment Study

Scenario 3: Lunar LOX (Earth H2) for Outbound + Mars LOX/CH4 for Return

Mission Description

1 - MTV departs LEO on an expendable LOX/H2 stage to LLO carrying refurbishment/resupply for the lunar and Mars propellant plants, hydrogen for the outbound leg, the Mars mission payload and crew

2 - the LEV meets the MTV in LLO, transfers LLOX to the MTV, and obtains refurbishment/resupply for the lunar plant before returning to the lunar surface

3 - the MTV leaves LLO and transfers to LMO, is met by the MEV in LMO, and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant

4 - after completing the mission, the MEV delivers the crew and MLOX/CH4 for the return leg to the MTV in LMO and the MTV returns to LEO

Required Infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant
- Lunar Excursion Vehilce (LEV)
- Lunar Propellant Plant
- LEO → LLO Expendable Booster

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Lunar Plant Refurbishment/Resupply
- Propellant for: LEO→LLO, LLO→LMO and LEV round trip (hydrogen and tanks only)

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

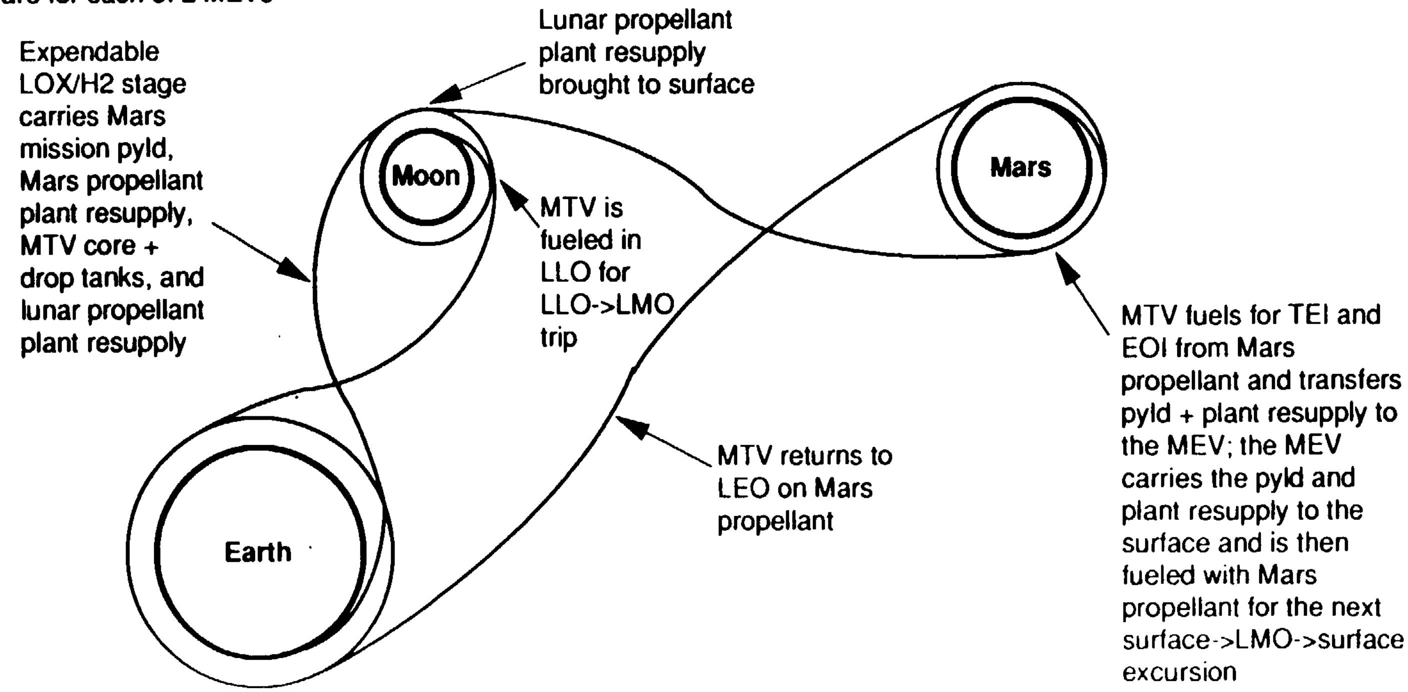
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

  LOX/CH4: Thrust/Weight = 90 lbf/lbm; Isp = 380 sec; Mixture Ratio (O/F) = 3.6

Scenario 3: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	824	433	3300	35	3335	4070	470	700	628
LOI	Exp. stg.	LLO	372	81	1110	0	1110	4070	470	700	118
Lunar ascent	LEV	Lunar Surf.	612	219	1900	13	1913	2616	470	450	479
Lunar descent	LEV	LLO	154	57	2000	1	2001	2616	470	450	125
TMI	MTV	LLO	522	189	2005	7	2012	2616	470	450	426
MOI	MTV	LMO	325	145	2590	8	2598	2616	470	450	232
Mars ascent	MEV*	Mars Surf.	817	656	5300	53	5353	5040	380	1000	522
Mars desent	MEV*	LMO	104	24	930	0	930	5040	380	1000	19
TEI	MTV	LMO	274	137	2521	4	2525	2616	380	450	251
EOI	MTV	LEO	132	90	4081	3	4084	2616	380	450	164

* These numbers are for each of 2 MEVs



Scenario 3: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX	uses Mars LOX/CH4	uses Mars LOX/CH4		uses lunar LOX	
Propellants Used	Earth LOX/H2		lunar LOX + Earth H2	Mars LOX/CH4	Mars LOX/CH4		lunar LOX + Earth H2	
Specific Impulse (sec)	470		470	380	380		470	
Mixture Ratio (O/F)	6.0		6.0	3.6	3.6		6.0	
Thrust Level(s) (klbs)	700		450	450	1000		450	
Engine Operating Time (% of trip)	0.25%		0.003%	0.0016%	1.2%	0.04%	1.1%	0.29%
Total ΔV (m/sec)	4,445		4,610	6,609	5,353	930	1,913	2,001
Total Impulse (x10^6 kN sec)	2.323		1.317	0.831	2.322	0.085	0.959	0.250
Maximum Acceleration (g's)	0.79/1.09		0.60/1.10	1.47/4.63	2.33	5.57	0.50	2.05
Operating Time (sec/mission)	746		658	415	522	19	479	125
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

SCENARIO 4

Lunar LOX/CH_4 for Outbound + Mars LOX/CO for Return

Mission Requirements for Engine System Assessment Study

Scenario 4: Lunar LOX/CH4 for Outbound + Mars LOX/CO for Return

Mission Description

1 - MTV departs LEO on an expendable LOX/H2 stage to LLO carrying refurbishment/resupply for the lunar and Mars propellant plants, the Mars mission payload and crew

2 - the LEV meets the MTV in LLO, transfers LLOX/CH4 to the MTV, and obtains refurbishment/resupply for the lunar plant before returning to the lunar surface

3 - the MTV leaves LLO and transfers to LMO, is met by the MEV in LMO, and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant

4 - after completing the mission, the MEV delivers crew and MLOX/CO for the return leg to the MTV in LMO and the MTV returns to LEO

Required Infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant
- Lunar Excursion Vehilce (LEV)
- Lunar Propellant Plant
- LEO → LLO Expendable Booster

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Lunar Plant Refurbishment/Resupply
- Propellant for: LEO→LLO

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

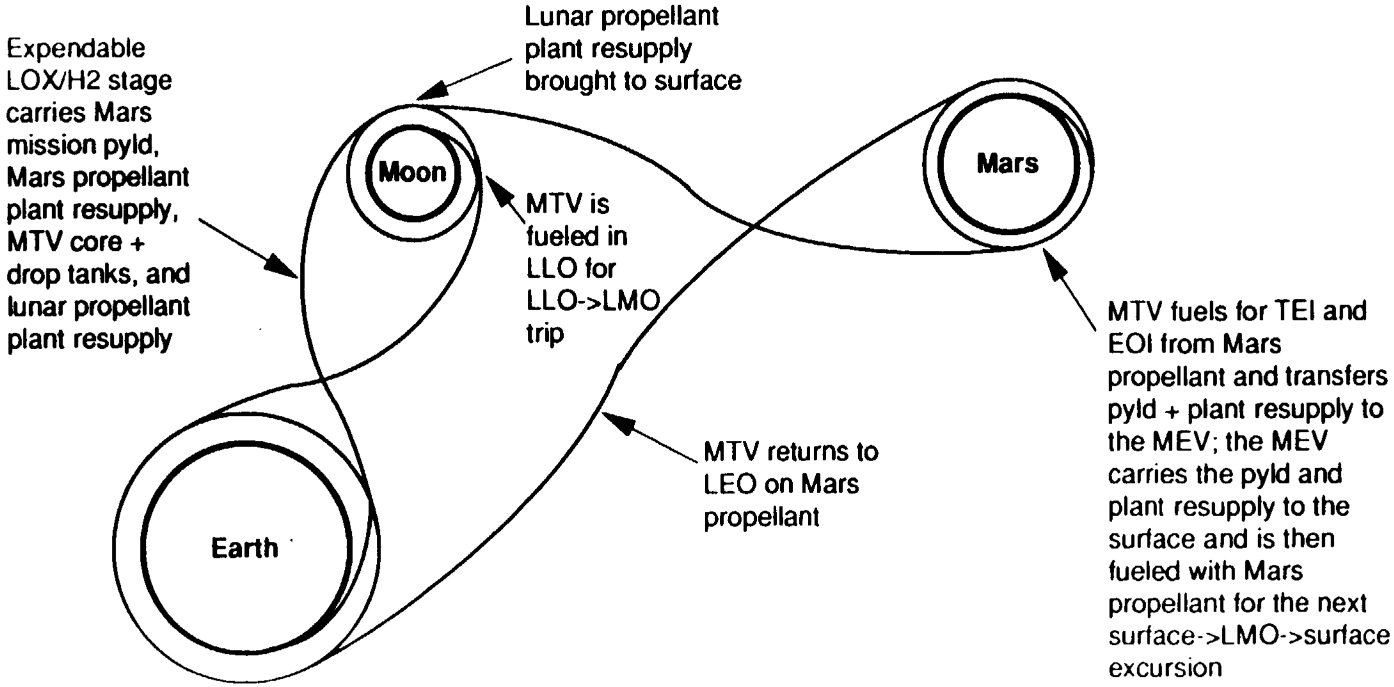
  LOX/CH4: Thrust/Weight = 90 lbf/lbm; Isp = 380 sec; Mixture Ratio (O/F) = 3.6

  LOX/CO: Thrust/Weight = 98 lbf/lbm; Isp = 290 sec; Mixture Ratio (O/F) = 0.6

Scenario 4: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CO FOR RETURN

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	345	181	3300	15	3315	2616	470	450	407
LOI	Exp. stg.	LLO	157	34	1110	0	1110	2616	470	450	77
Lunar ascent	LEV	Lunar Surf.	550	233	1900	12	1912	2016	380	400	463
Lunar descent	LEV	LLO	27	12	2000	0	2000	2016	380	400	23
TMI	MTV	LLO	409	174	2005	9	2014	1512	380	300	477
MOI	MTV	LMO	230	118	2590	8	2598	1512	380	300	322
Mars ascent	MEV*	Mars Surf.	3433	3064	5300	55	5355	16196	290	3500	532
Mars desent	MEV*	LMO	155	46	930	0	930	16196	290	3500	8
TEI	MTV	LMO	510	308	2521	30	2551	1512	290	300	644
EOI	MTV	LEO	193	150	4081	12	4093	1512	290	300	314

* These numbers are for each of 2 MEVs



Scenario 4: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX/CH4	uses Mars LOX/CO	uses Mars LOX/CO		uses lunar LOX/CH4	
Propellants Used	Earth LOX/H2		lunar LOX/CH4	Mars LOX/CO	Mars LOX/CO		lunar LOX/CH4	
Specific Impulse (sec)	470		380	290	290		380	
Mixture Ratio (O/F)	6.0		3.6	0.6	0.6		3.6	
Thrust Level(s) (klbs)	450		300	300	3500		400	
Engine Operating Time (% of trip)	0.16%		0.0037%	0.0037%	1.2%	0.019%	0.011%	0.053%
Total ΔV (m/sec)	4,840		4,612	6,644	5,355	930	1,912	2,000
Total Impulse (x 10^6 kN sec)	0.969		1.066	1.278	8.283	0.125	0.824	0.041
Maximum Acceleration (g's)	1.21/1.66		0.57/1.19	0.65/2.97	3.04	14.19	0.55	11.49
Operating Time (sec/mission)	484		799	958	532	8	463	23
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

SCENARIO 5

Lunar LOX/CH_4 for Outbound + Mars LOX/CH_4 for Return

Mission Requirements for Engine System Assessment Study
Scenario 5: Lunar LOX/CH4 for Outbound + Mars LOX/CH4 for Return

Mission Description

1 - MTV departs LEO on an expendable LOX/H2 stage to LLO carrying refurbishment/resupply for the lunar and Mars propellant plants, the Mars mission payload and crew
2 - the LEV meets the MTV in LLO, transfers LLOX/CH4 to the MTV, and obtains refurbishment/resupply for the lunar plant before returning to the lunar surface
3 - the MTV leaves LLO and transfers to LMO, is met by the MEV in LMO, and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant
4 - after completing the mission, the MEV delivers crew and MLOX/CH4 for the return leg to the MTV in LMO and the MTV returns to LEO

Required infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant
- Lunar Excursion Vehilce (LEV)
- Lunar Propellant Plant
- LEO → LLO Expendable Booster

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Lunar Plant Refurbishment/Resupply
- Propellant for: LEO→LLO

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

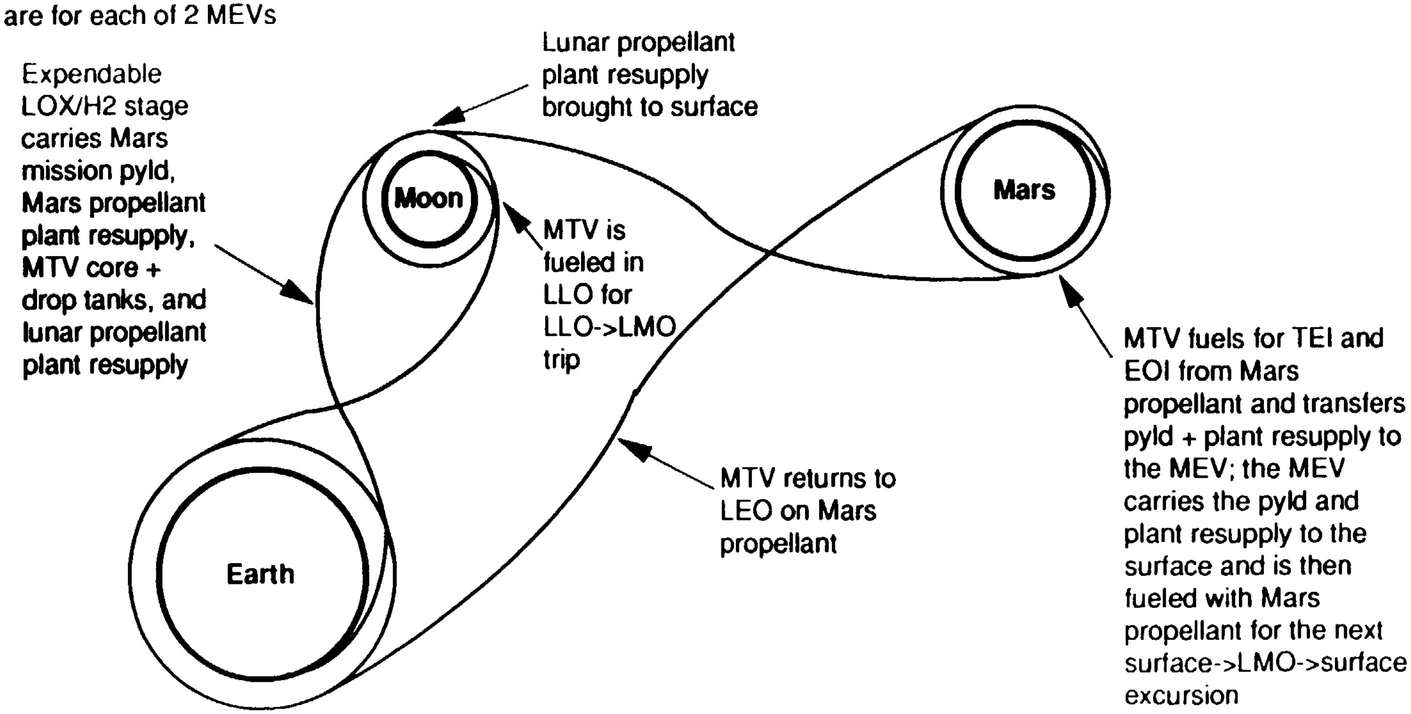
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

  LOX/CH4: Thrust/Weight = 90 lbf/lbm; Isp = 380 sec; Mixture Ratio (O/F) = 3.6

Scenario 5: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.	LEO	571	311	3300	224	3524	1163	470	200	1581
LOI	Exp. stg.	LLO	246	54	1110	1	1111	1163	470	200	273
Lunar ascent	LEV	Lunar Surf.	887	376	1900	14	1914	3024	380	600	498
Lunar descent	LEV	LLO	43	19	2000	0	2000	3024	380	600	25
TMI	MTV	LLO	658	282	2005	23	2028	1512	380	300	772
MOI	MTV	LMO	368	189	2590	20	2610	1512	380	300	517
Mars ascent	MEV*	Mars Surf.	798	640	5300	50	5350	5040	380	1000	509
Mars desent	MEV*	LMO	103	24	930	0	930	5040	380	1000	19
TEI	MTV	LMO	267	134	2521	10	2531	1512	380	300	367
EOI	MTV	LEO	128	87	4081	7	4088	1512	380	300	239

* These numbers are for each of 2 MEVs



Scenario 5: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent	ascent	descent
Mission Leverage Feature(s)	transfers MTV+pyld from LEO to LLO		uses lunar LOX/CH4	uses Mars LOX/CH4	uses Mars LOX/CH4		uses lunar LOX/CH4	
Propellants Used	Earth LOX/H2		lunar LOX/CH4	Mars LOX/CH4	Mars LOX/CH4		lunar LOX/CH4	
Specific Impulse (sec)	470		380	380	380		380	
Mixture Ratio (O/F)	6.0		3.6	3.6	3.6		3.6	
Thrust Level(s) (klbs)	200		300	300	1000		600	
Engine Operating Time (% of trip)	0.61%		0.006%	0.002%	1.2%	0.04%	1.2%	0.06%
Total ΔV (m/sec)	4,635		4,638	6,619	5,350	930	1,914	2,000
Total Impulse (x 10^6 kN sec)	1.649		1.720	0.809	2.264	0.085	1.329	0.067
Maximum Acceleration (g's)	0.34/0.47		0.36/0.75	1.0/3.17	2.39	5.68	0.51	10.84
Operating Time (sec/mission)	1,854		1,289	606	509	19	498	25
Reusability (# of missions)	1		5	5	5	5	5	5
Refueling Requirements	none		refueled in LLO	refueled in LMO	refueled on Mars surface		refueled on lunar surface	

SCENARIO 6

Earth LOX/H_2 for Outbound + Mars LOX/CO for Return

Mission Requirements for Engine System Assessment Study
Scenario 6: Earth LOX/H2 for Outbound + Mars LOX/CO for Return

Mission Description

1 - MTV departs LEO carrying refurbishment/resupply for the Mars propellant plant, the Mars mission payload and crew
2 - the MTV is met by the MEV in LMO and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant
3 - after completing the mission, the MEV delivers the crew and MLOX/CH4 for the return leg to the MTV in LMO and the MTV returns to LEO

Required infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Propellant for: LEO→LMO

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

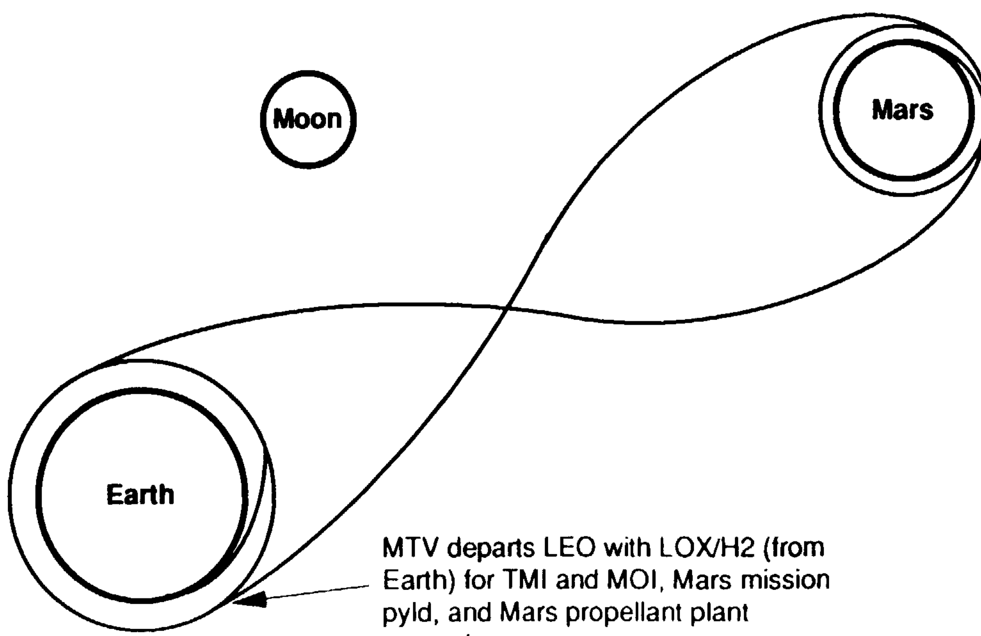
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

  LOX/CO: Thrust/Weight = 98 lbf/lbm; Isp = 290 sec; Mixture Ratio (O/F) = 0.6

Scenario 6: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	MTV	LEO	544	322	3982	26	4008	4070	470	700	468
MOI	MTV	LMO	202	90	2590	7	2597	1744	470	300	301
Mars ascent	MEV*	Mars Surf.	3442	3072	5300	55	5355	16196	290	3500	533
Mars desent	MEV*	LMO	155	46	930	0	930	16196	290	3500	8
TEI	MTV	LMO	512	309	2521	30	2551	1744	290	300	646
EOI	MTV	LEO	194	151	4081	12	4093	1744	290	300	315

* These numbers are for each of 2 MEVs



MTV fuel for return trip from Mars propellant and transfers pyld + plant resupply to the MEV; the MEV carries this pyld to the surface and is then fueled with Mars propellant for the next surface->LMO->surface excursion

MTV departs LEO with LOX/H2 (from Earth) for TMI and MOI, Mars mission pyld, and Mars propellant plant resupply

Scenario 6: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent		
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Mars LOX/CO	uses Mars LOX/CO			
Propellants Used		Earth LOX/H2		Mars LOX/CO	Mars LOX/CO			
Specific Impulse (sec)		470		290	290			
Mixture Ratio (O/F)		6.0		0.6	0.6			
Thrust Level(s) (klbs)		700 - TMI 300 - MOI		300	3500			
Engine Operating Time (% of trip)		0.004%		0.004%	1.2%	0.02%		
Total ΔV (m/sec)		4,008 - TMI 2,597 - MOI		6,644	5,355	930		
Total Impulse (x 10^6 kN sec)		1.457 - TMI 0.402 - MOI		1.282	8.298	0.125		
Maximum Acceleration (g's)		1.39/1.18		0.65/2.96	3.03	14.16		
Operating Time (sec/mission)		468 - TMI 301 - MOI		961	533	8		
Reusability (# of missions)		5		5	5	5		
Refueling Requirements		refueled in LEO		refueled in LMO	refueled on Mars surface			

SCENARIO 7

Earth LOX/H_2 for Outbound + Mars LOX/CH_4 for Return

Mission Requirements for Engine System Assessment Study
Scenario 7: Earth LOX/H2 for Outbound + Mars LOX/CH4 for Return

Mission Description

1 - MTV departs LEO carrying refurbishment/resupply for the Mars propellant plant, the Mars mission payload and crew
2 - the MTV is met by the MEV in LMO and transfers crew, Mars mission payload, and refurbishment/resupply for the Mars propellant plant
3 - after completing the mission, the MEV delivers the crew and MLOX/CH4 for the return leg to the MTV in LMO and the MTV returns to LEO

Required infrastructure elements

- Mars Excursion Vehicle (MEV)
- Mars Transfer Vehicle (MTV)
- Mars Propellant Plant

Elements launched from Earth to support steady-state operation

- Crew and consumables
- Mars Mission Payload (25 t)
- Mars Plant Refurbishment/Resupply
- Propellant for: LEO→LMO

Assumptions

- Mars trajectory ΔV's are averaged from 6 opposition class opportunities in the 2015 to 2030 timeframe
- All-propulsive capture to Mars orbit
- Earth departure/arrival orbit is 407 km circular
- Mars arrival/departure orbit is 250 km x 1 sol
- Allowances made for 4 crew members and 93 kg consumables per crew member per month
- MTV mass at Earth return is 30 t + engine(s) + core propellant tanks
- MEV requirements shown are for 1 of 2 vehicles required
- Propellant boiloff only accounted for in hydrogen tanks
- Initial Engine Parameters:

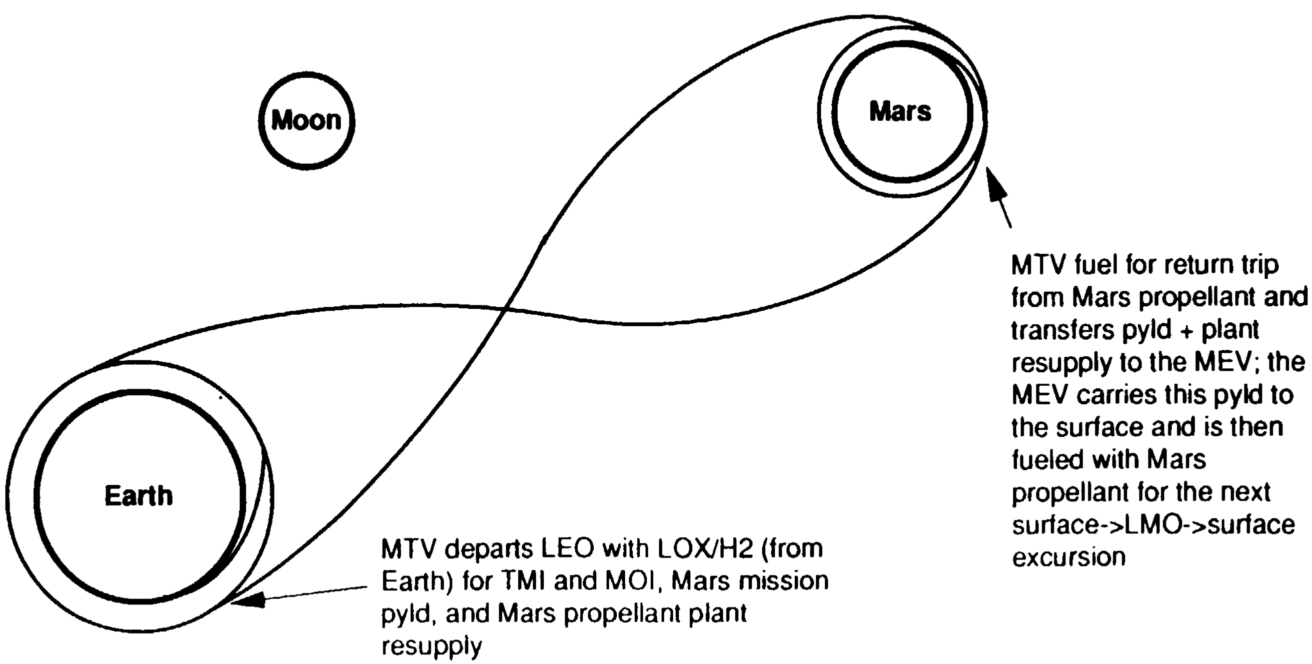
  LOX/H2: Thrust/Weight = 78 lbf/lbm; Isp = 470 sec; Mixture Ratio (O/F) = 6.0

  LOX/CH4: Thrust/Weight = 90 lbf/lbm; Isp = 380 sec; Mixture Ratio (O/F) = 3.6

Scenario 7: **EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN**

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	MTV	LEO	867	515	3982	33	4015	5813	470	1000	523
MOI	MTV	LMO	323	145	2590	39	2629	1163	470	200	728
Mars ascent	MEV*	Mars Surf.	794	637	5300	50	5350	5040	380	1000	507
Mars desent	MEV*	LMO	102	24	930	0	930	5040	380	1000	19
TEI	MTV	LMO	265	134	2521	22	2543	1163	380	200	549
EOI	MTV	LEO	127	87	4081	16	4097	1163	380	200	355

* These numbers are for each of 2 MEVs



Scenario 7: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN

	TRANSFER VEHICLES				EXCURSION VEHICLES			
	Expendable Stage	MTV			MEV		LEV	
	LEO->LLO	LEO->LMO	LLO->LMO	LMO->LEO	ascent	descent		
Mission Leverage Feature(s)		uses Earth LOX/H2		uses Mars LOX/CH4	uses Mars LOX/CH4			
Propellants Used		Earth LOX/H2		Mars LOX/CH4	Mars LOX/CH4			
Specific Impulse (sec)		470		380	380			
Mixture Ratio (O/F)		6.0		3.6	3.6			
Thrust Level(s) (klbs)		1000 - TMI 200 - MOI		200	1000			
Engine Operating Time (% of trip)		0.006%		0.003%	1.2%	0.04%		
Total ΔV (m/sec)		4,015 - TMI 2,629 - MOI		6,640	5,350	930		
Total Impulse (x 10^6 kN sec)		2.326 - TMI 0.648 - MOI		0.804	2.255	0.085		
Maximum Acceleration (g's)		1.25/0.50		0.68/2.14	2.4	5.7		
Operating Time (sec/mission)		523 - TMI 728 - MOI		904	507	19		
Reusability (# of missions)		5		5	5	5		
Refueling Requirements		refueled in LEO		refueled in LMO	refueled on Mars surface			

APPENDIX B

OFF-DESIGN ELES ANALYSIS METHODOLOGY

APPENDIX B
OFF-DESIGN ELES ANALYSIS METHODOLOGY

The user-defined turbomachinery option of ELES allows evaluation of fuel and oxidizer pump and turbine performance at off-design operating characteristics and with a variety of propellants. The parameters input to define the TPA for off-design evaluation are detailed in the worksheets following, and include number of stages for all pumps and turbines, pump and turbine diameters, turbine annulus area, turbine admission fraction, and various gas generator parameters.

ELES calculates pump head rise and volumetric flowrate, and turbine horsepower, mass flowrate, and pressure ratio based on cycle balance requirements. Using these values, the pump rpm is calculated as a function of input pump diameter. To perform this calculation, a correlation had to be developed for pump head coefficient as a function of specific speed (standard cases interpolate this coefficient from a data table), and is of the form:

$$HC = const * SS^{x}$$

where

HC = head coefficient
SS = pump specific speed

For example, the main pump correlation is:

$$HC = 3.7852 * SS^{-0.28786}$$

This correlation is different for main pumps and boost pumps. The specific speed is a function of pump rpm, head rise, and volumetric flowrate, as is shown below:

$$SS = RPM * SQRT(\text{volumetric flowrate})/(\text{pump head rise}^{0.75})$$

The pump diameter is calculated as:

$$Dia = (720/pi*RPM) * SQRT(32.2*\text{pump head rise/head coefficient})$$

Substituting the head coefficient and specific speed equations into the equation for pump diameter and rearranging gives an equation for pump rpm's as a function of input pump diameter only. Once the rpm's are known, the specific speed, efficiency, and horsepower are easily found from the standard ELES equations.

The user-defined TPA version of ELES calculates the required turbine mass flowrate and horsepower and then evaluates the user input turbine to see how well it performs in meeting these requirements. The first step is to calculate the isentropic spouting velocity (Co) based on the number of turbine stages. Then, the ratio of turbine blade tangential velocity (U) to Co based on input turbine diameter (U/Co) is calculated and checked to determine whether this ratio is within the accepted range of 0.2-0.6. If U/Co is not within an acceptable range, a warning is printed. Next, the user-defined TPA version of ELES calculates the turbine inlet Mach number and checks whether it is below the accepted maximum value of 1.7. Finally, turbine specific speed, efficiency, and horsepower is calculated. The horsepower provided is then compared with the horsepower required and if not within 3%, a new turbine pressure ratio is selected and the entire process is repeated.

When a gas generator cycle is being evaluated, the user can also input values for GG bleed efficiency, turbine/GG inlet temperature and pressure, Isp of GG bleed, and turbine and bleed nozzle flowrates.

APPENDIX C

ENGINE SYSTEM DESIGN SENSITIVITY TRADE RESULTS

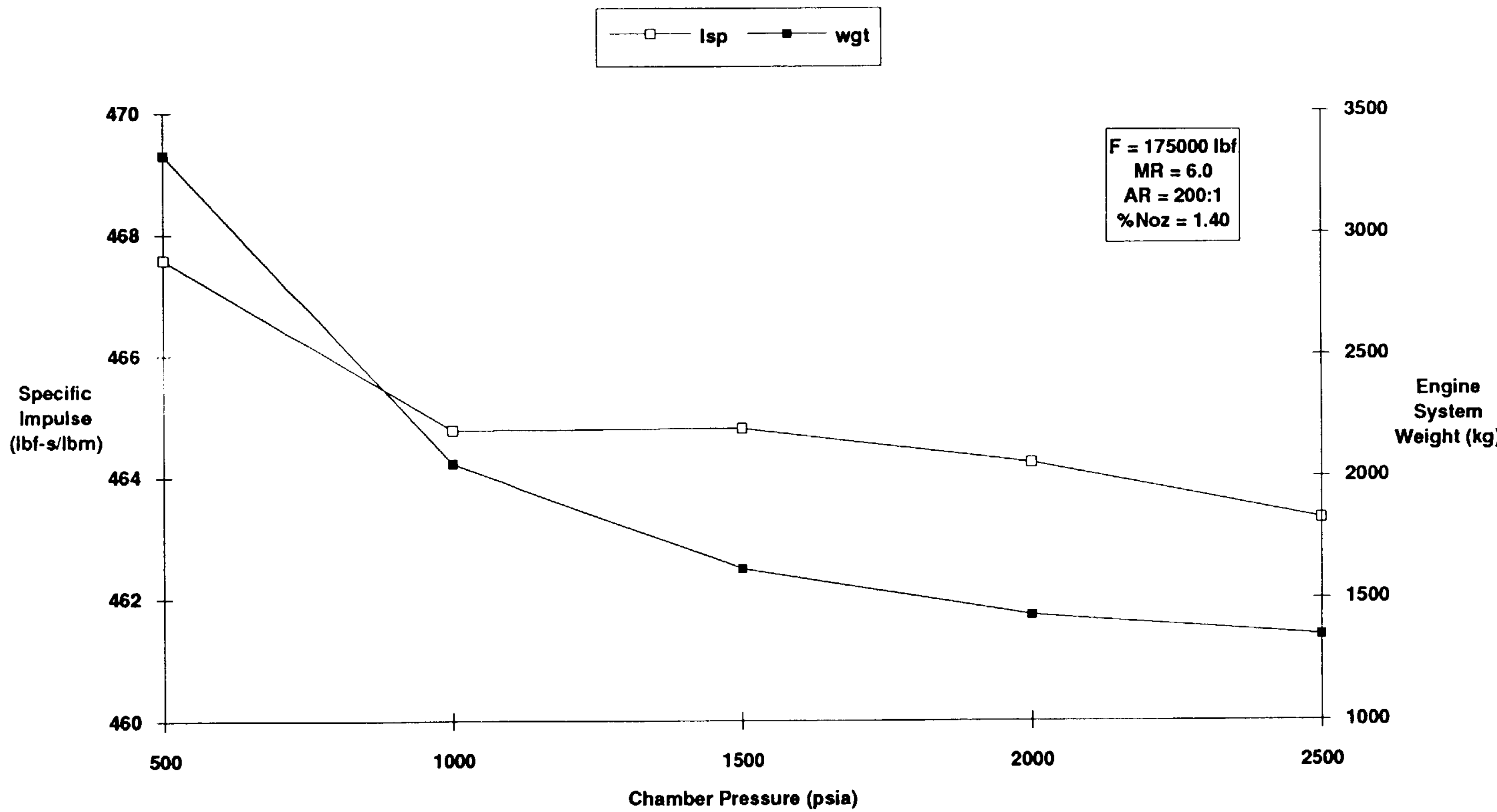
APPENDIX C
ENGINE SYSTEM DESIGN SENSITIVITY TRADE RESULTS

A detailed summary of engine system design sensitivity trade results are presented in this appendix. Numerous trades are presented for both expander and gas generator cycle engine systems using LOX/H_2, LOX/CO and LOX/CH_4 propellant combinations. It is based on the assessment of this data presented herein that optimum or near-optimum engine system design operation conditions and features were identified. These are discussed in Section 4.2.2.

ENGINE SYSTEM SENSITIVITY TRADE RESULTS

– LOX/H_2 Expander Cycle Engine –

L02/H2 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Chamber Pressure –



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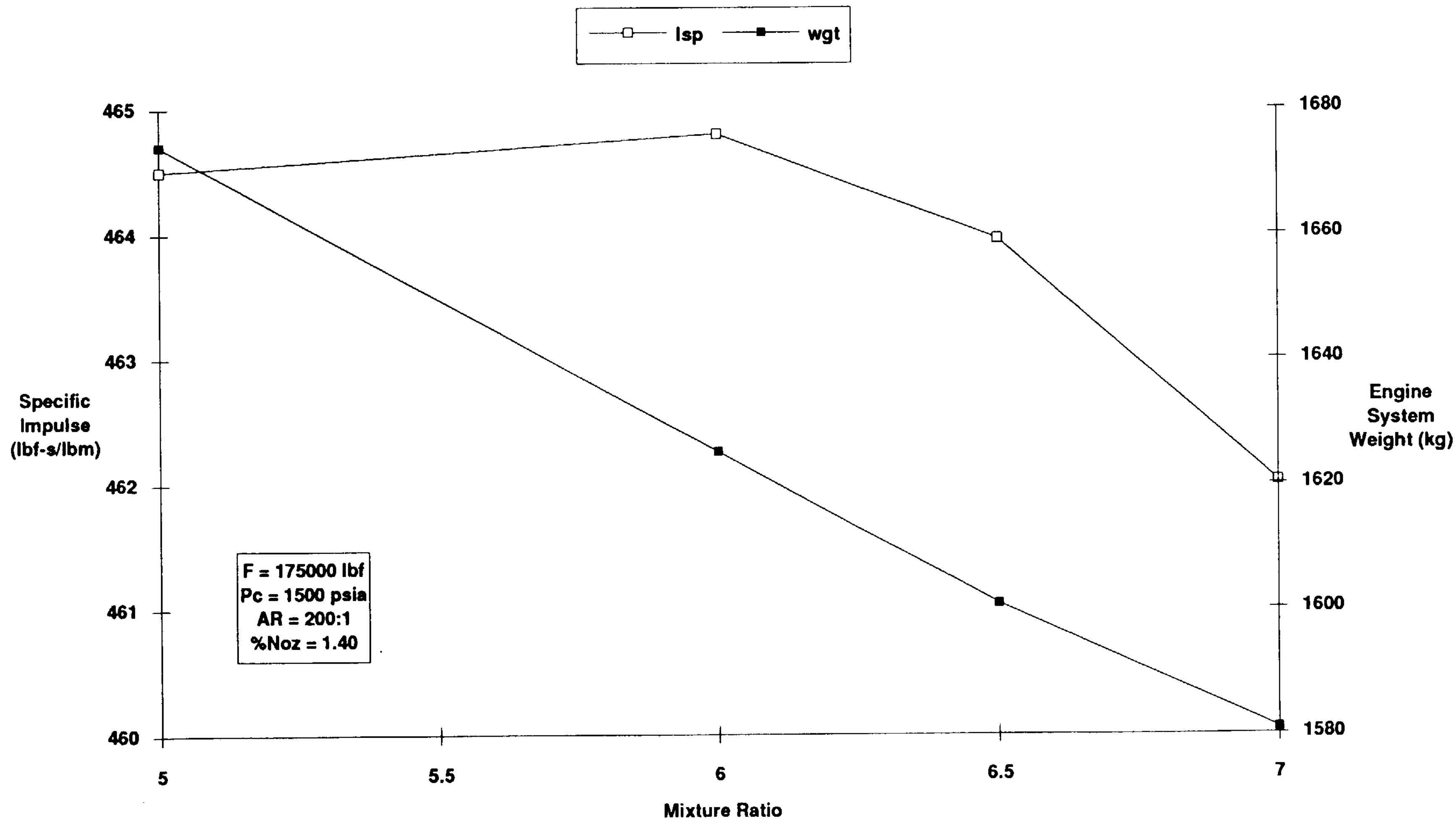
L02/H2 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Mixture Ratio –



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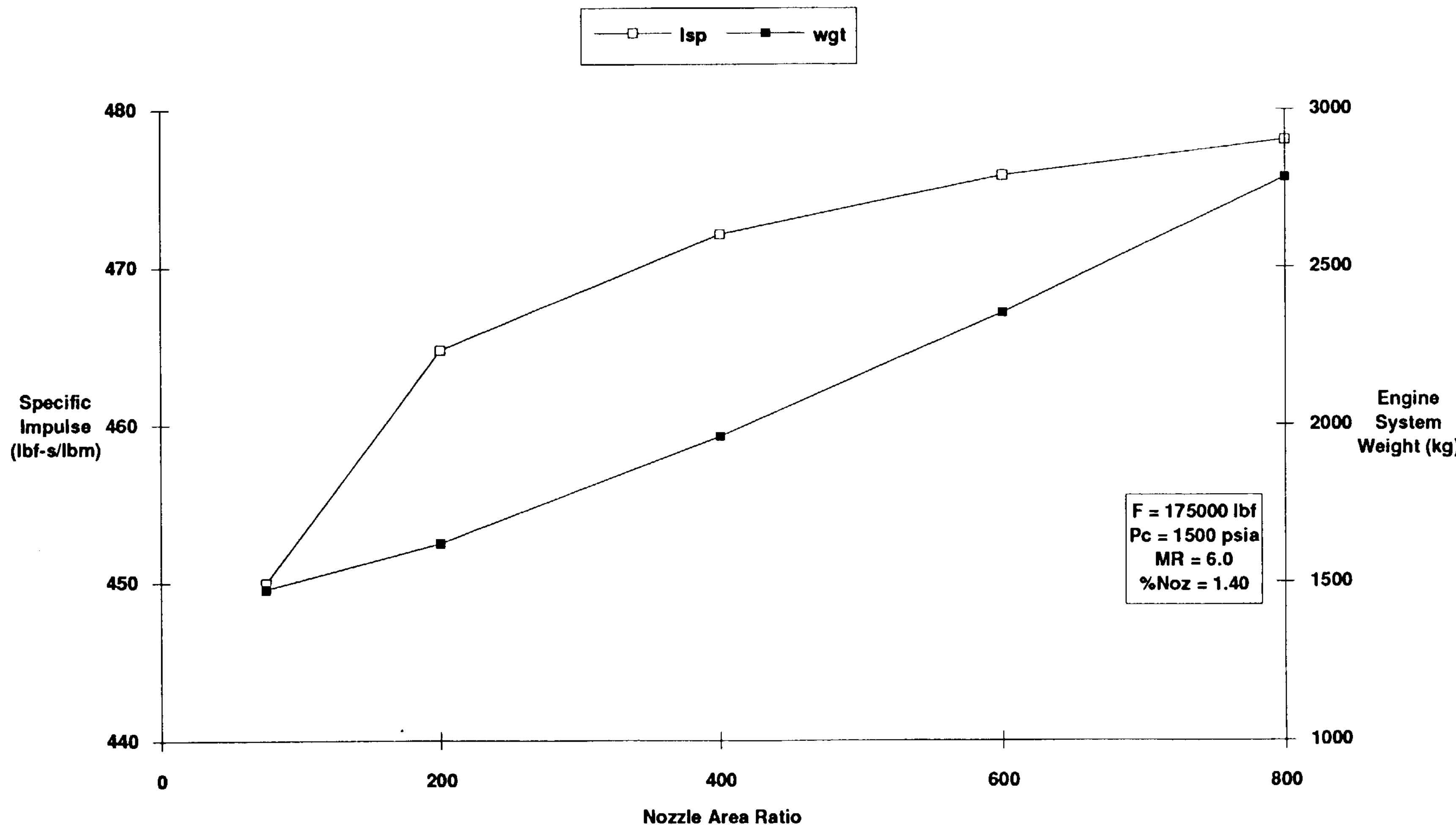
L02/H2 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Nozzle Area Ratio –



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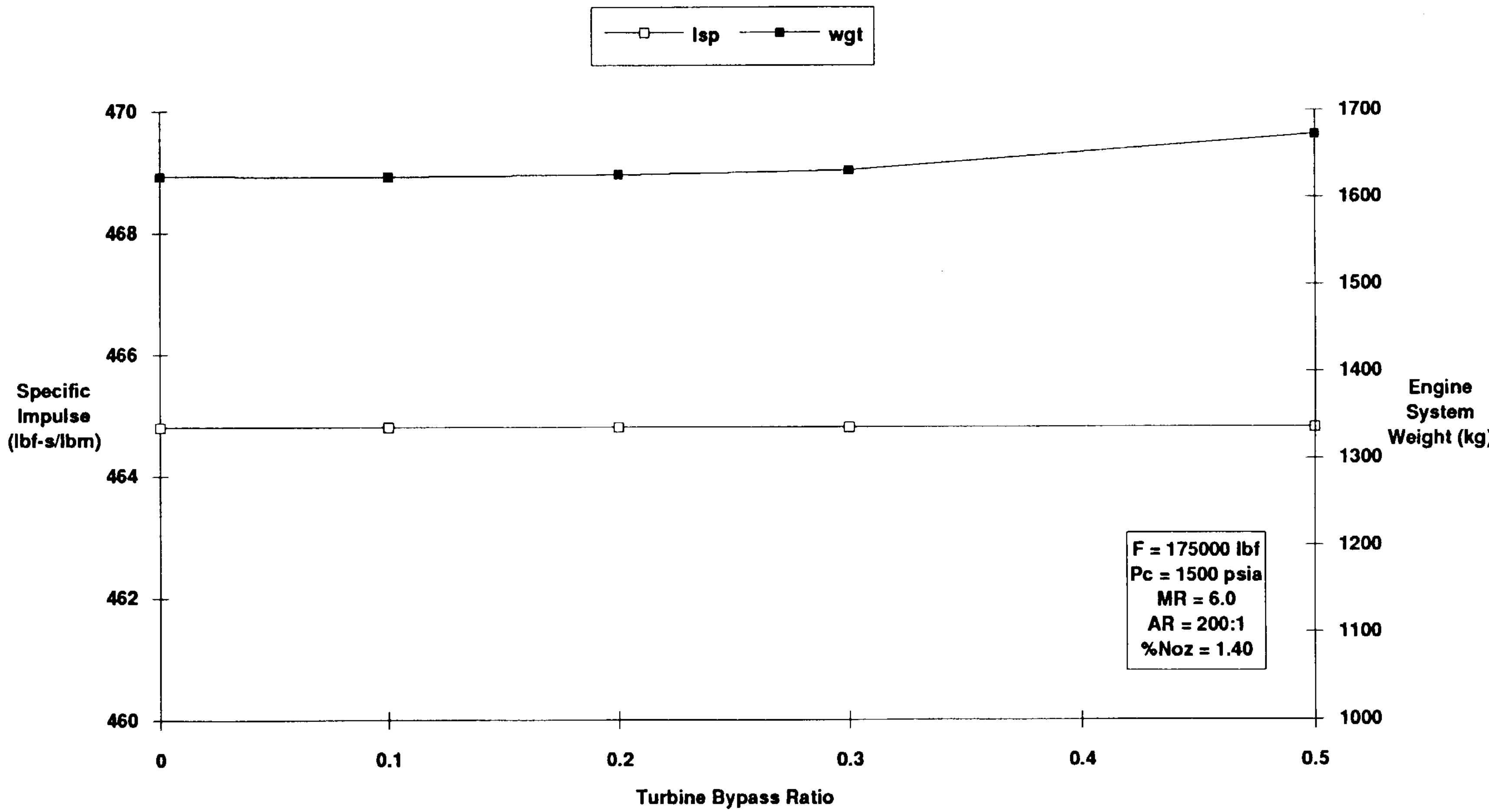
LO2/H2 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Turbine Bypass Ratio –



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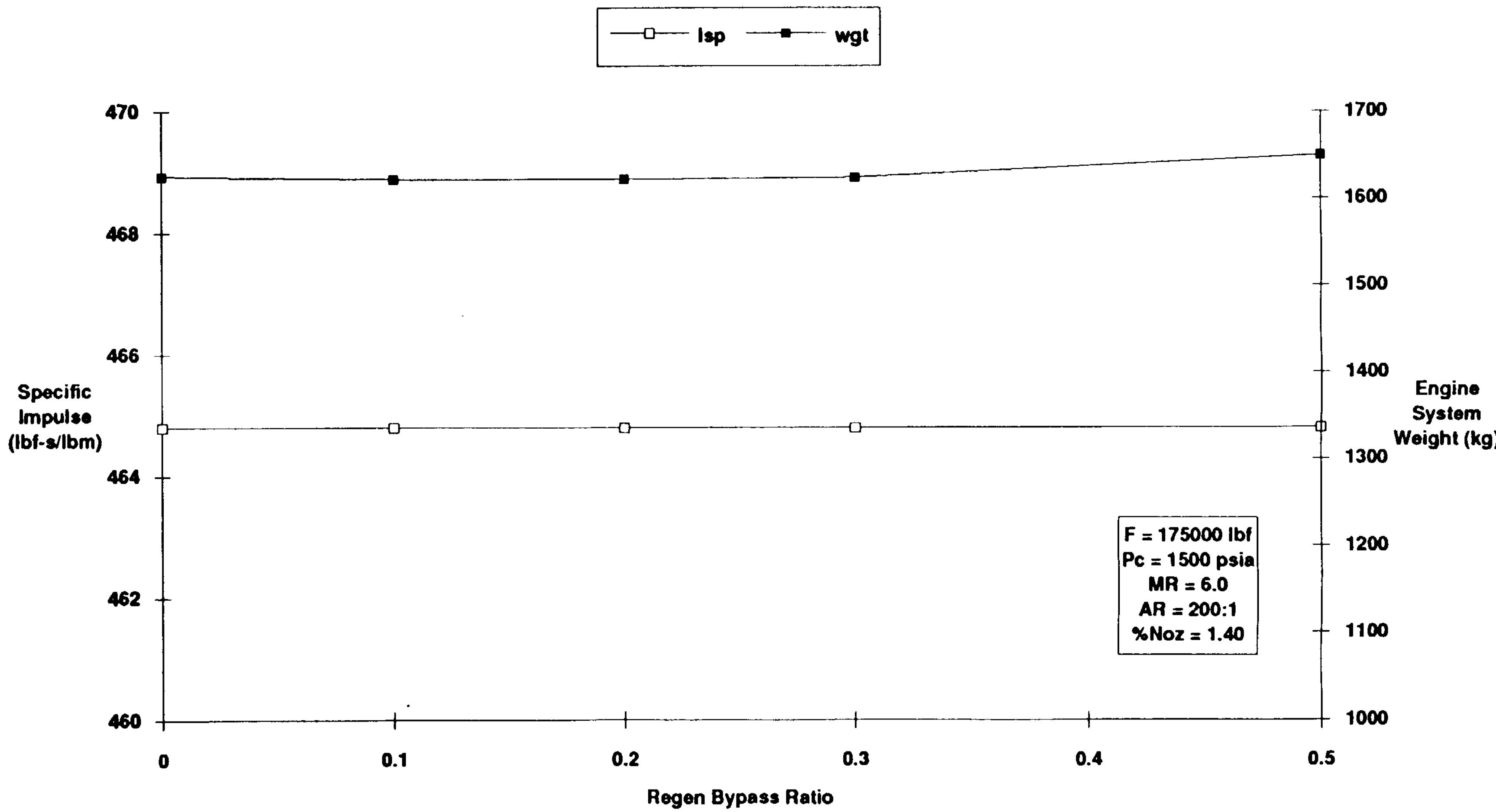
LO2/H2 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Regen Bypass Ratio –



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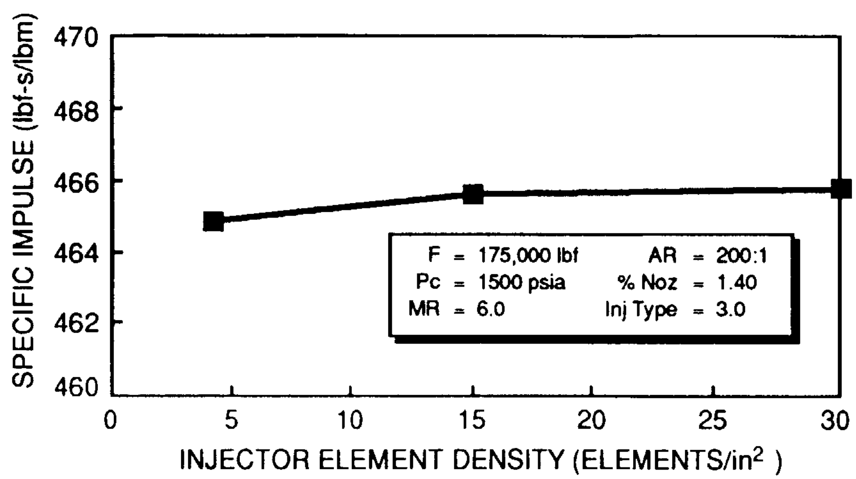
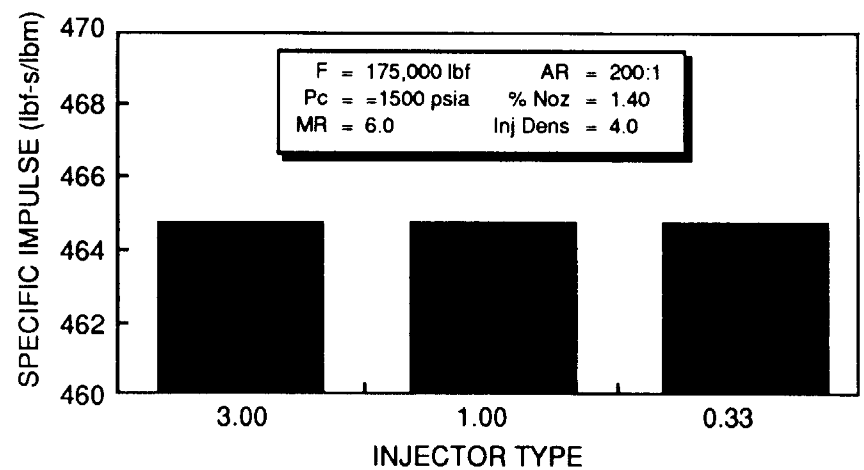
LO_2 /H_2 EXPANDER CYCLE ENGINE
– Performance as a Function of Injector Design –

INJECTOR DENSITY INFLUENCE



INJECTOR ELEMENT TYPE INFLUENCE



NOTE:

Injector Density (Elements/in^2):

1.0 COARSE | 4.0 NOMINAL | 15.0 PLATELETS PATTERN TYPE | 40.0 HYPERTHIN PLATELET

Injector Element Type:
3.00 Showerhead, Shear Co-Axial
1.00 Likes-Doublets, Splash Plate, X Doublet, V Doublet, Pre-Atomized Triplet
0.50 Vortex, Swirl Co-Axial
0.33 Unlike Triplet, Unlike Doublet

INCREASING ATOMIZING EFFICIENCY

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ENGINE SYSTEM SENSITIVITY TRADE RESULTS

– In Situ Propellant Expander Cycle Engines –

-- LO_2/CO --
-- LO_2/CH_4 --

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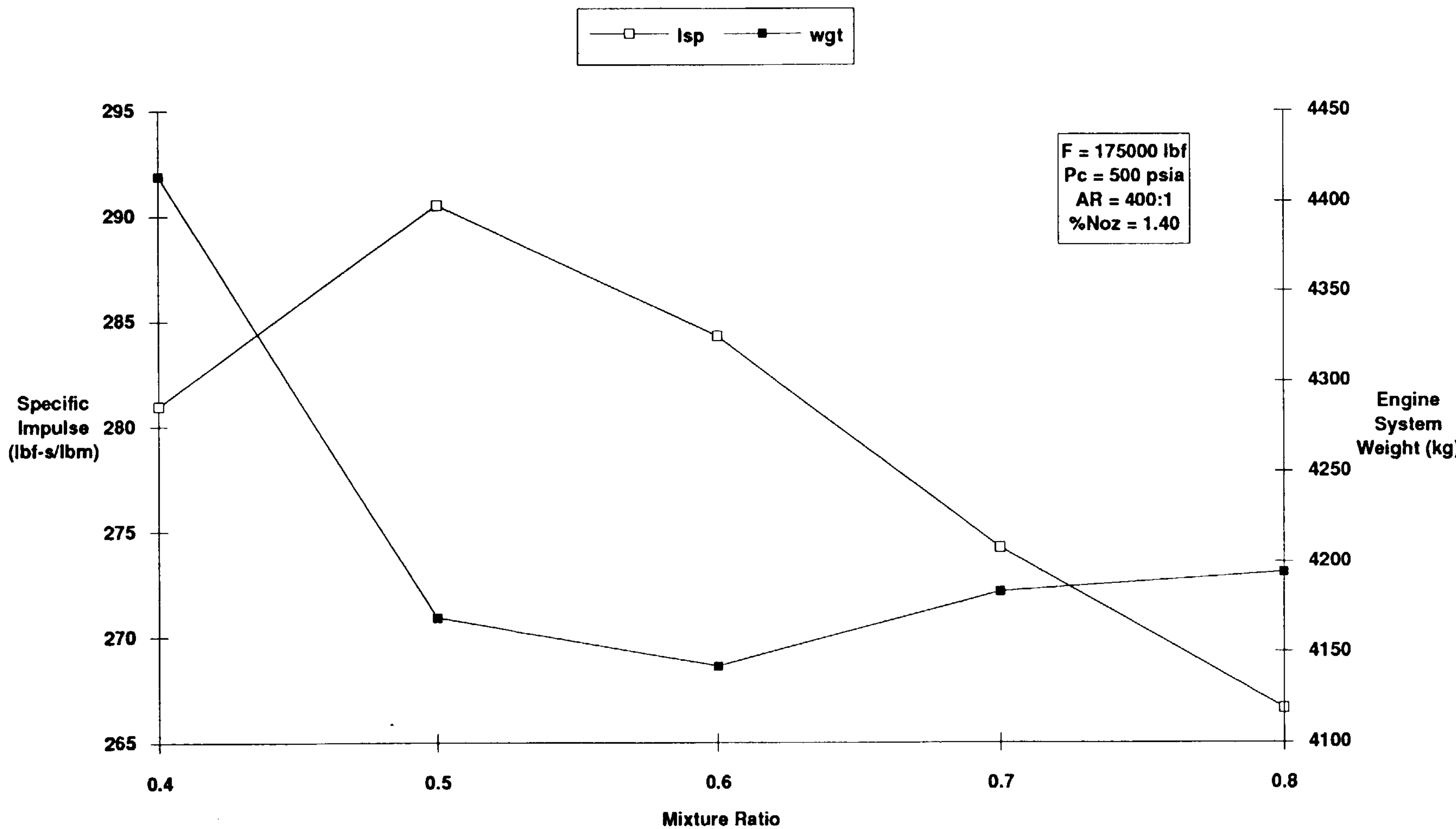
LO2/CO EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Mixture Ratio –



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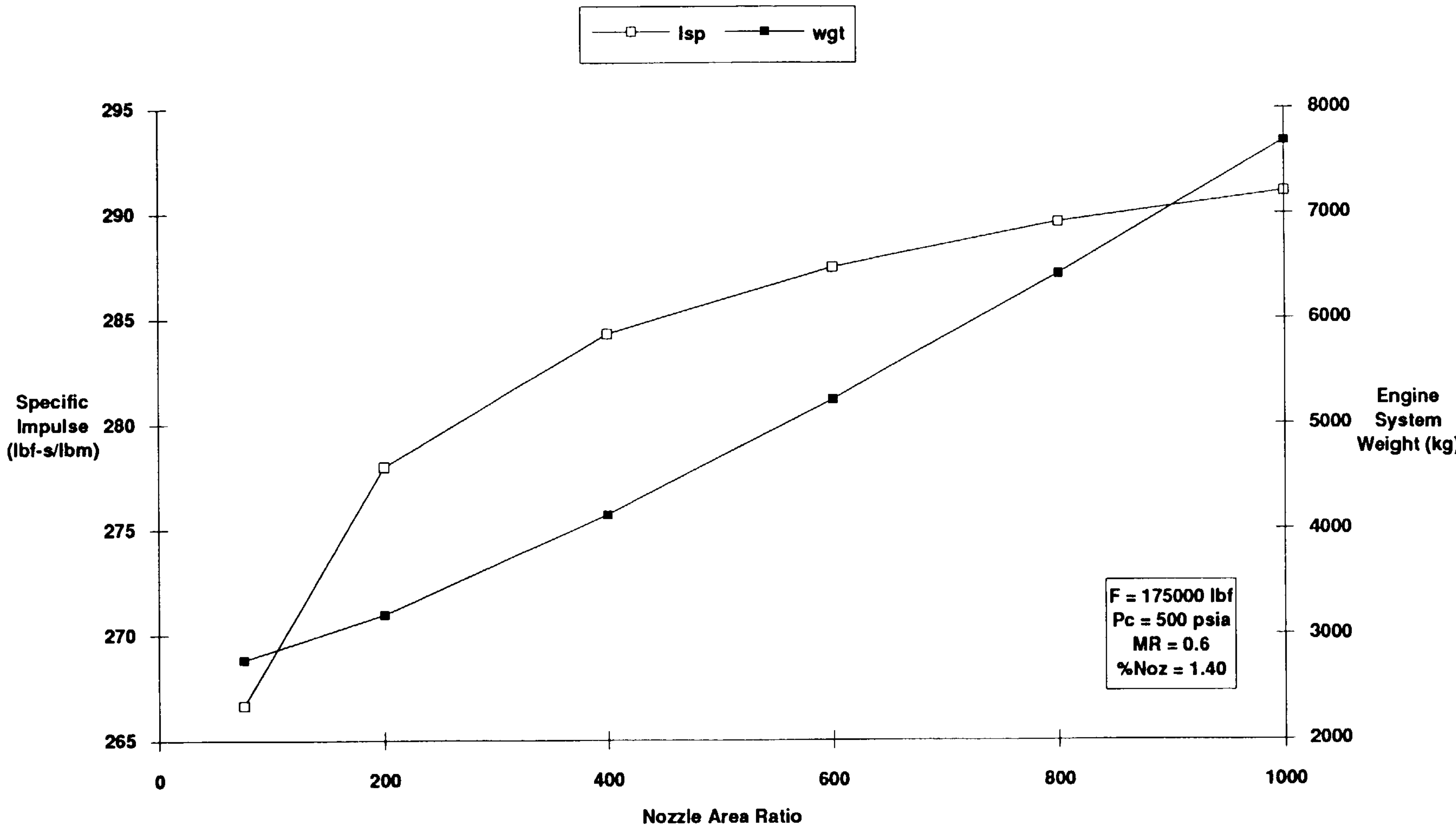
LO2/CO EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Nozzle Area Ratio –



PE/01.7

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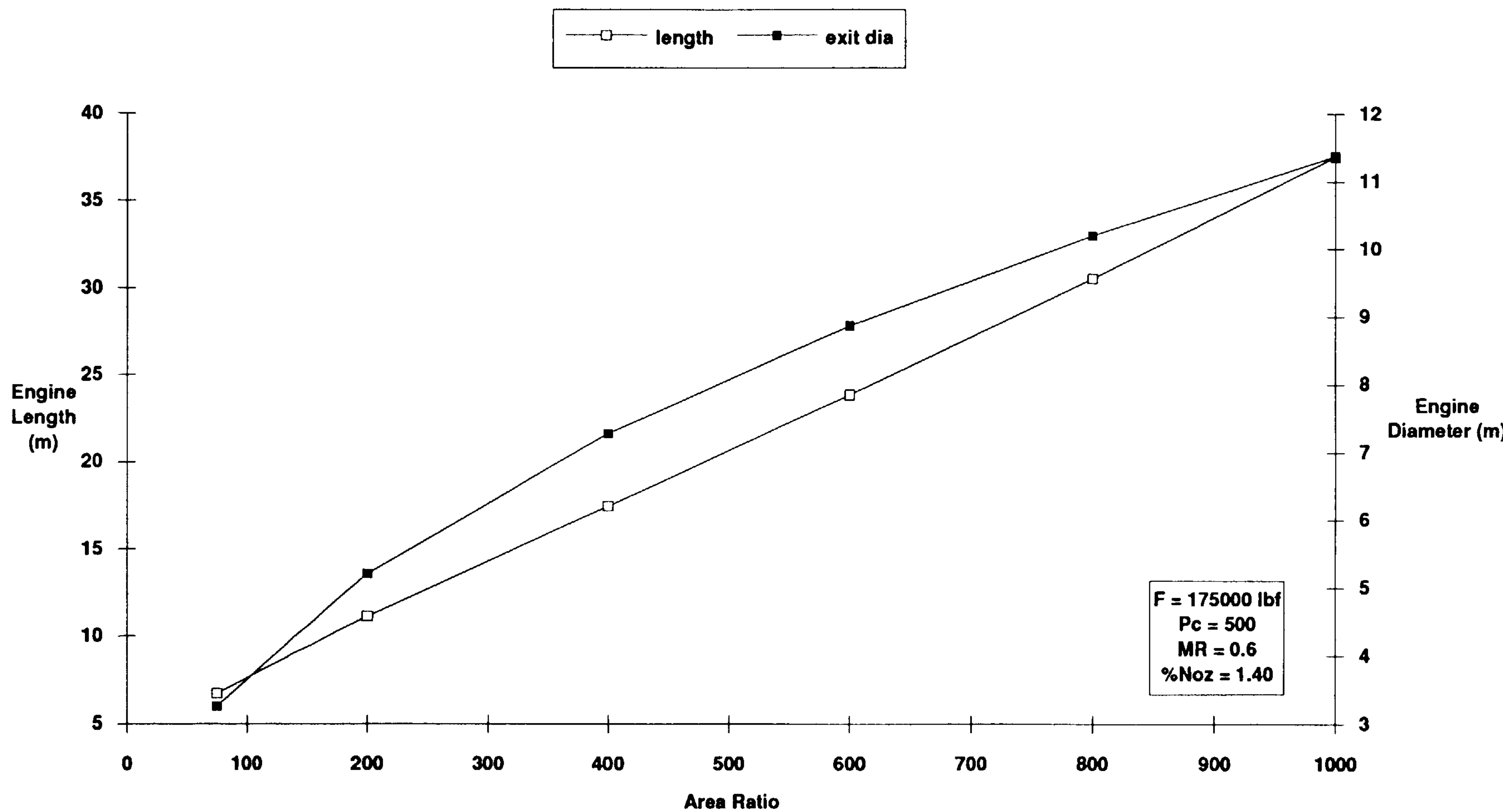
LO2/CO EXPANDER CYCLE ENGINE
– Engine Length and Diameter as a Function of Nozzle Area Ratio –

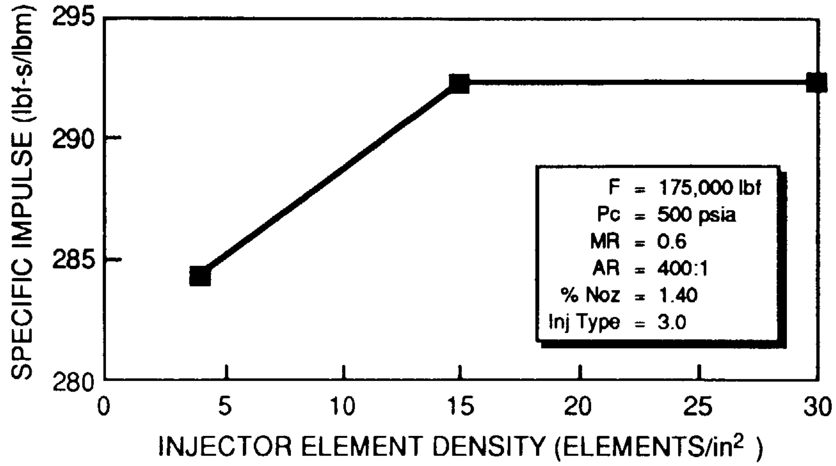


PE01.24

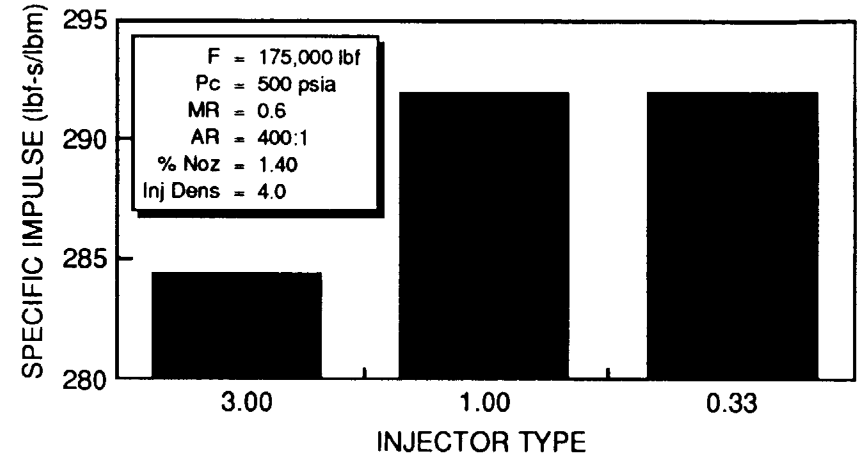
Science Applications International Corporation

LO_2 /CO EXPANDER CYCLE ENGINE
– Performance as a Function of Injector Design –

INJECTOR DENSITY INFLUENCE



INJECTOR ELEMENT TYPE INFLUENCE



NOTE:

Injector Density (Elements/in^2): 1.0 COARSE — 4.0 NOMINAL — 15.0 PLATELETS PATTERN TYPE — 40.0 HYPERTHIN PLATELET

Injector Element Type:
3.00 Showerhead, Shear Co-Axial
1.00 Likes-Doublets, Splash Plate, X Doublet, V Doublet, Pre-Atomized Triplet
0.50 Vortex, Swirl Co-Axial
0.33 Unlike Triplet, Unlike Doublet

INCREASING ATOMIZING EFFICIENCY

TOR29J/24

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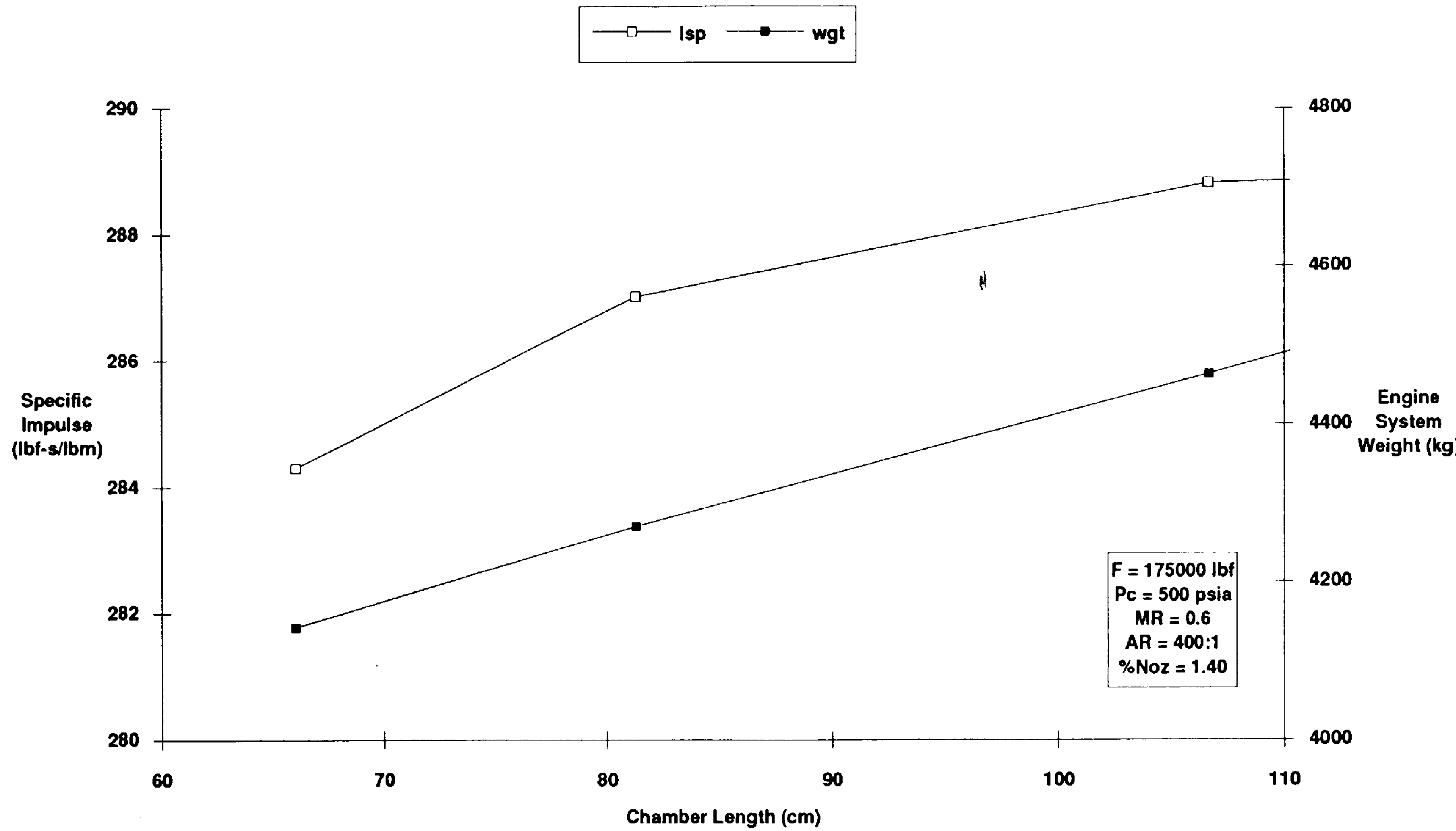
LO2/CO EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Chamber Length –



PE01.8

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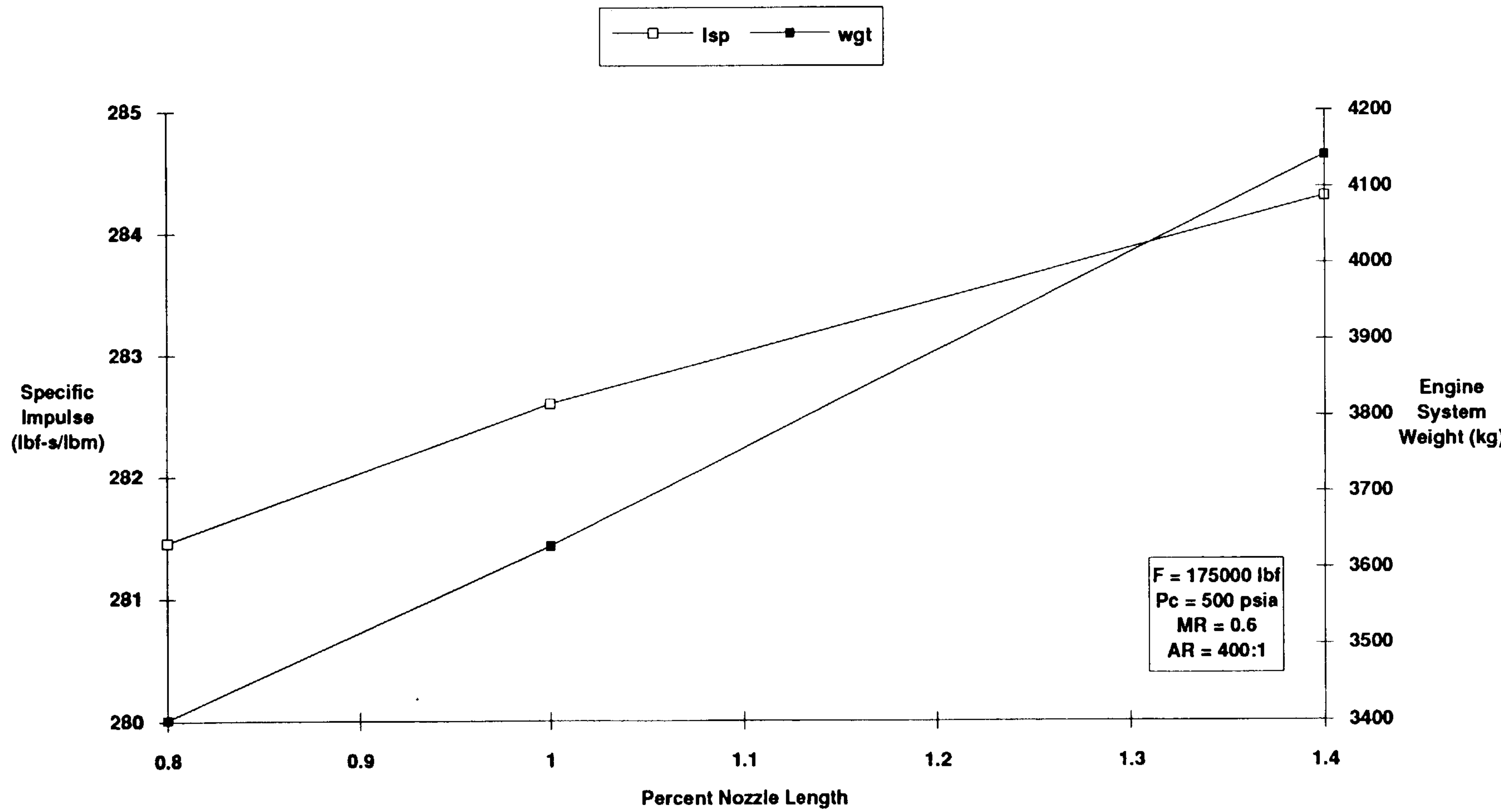
LO2/CO EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Percent Nozzle Length –



PE01.26

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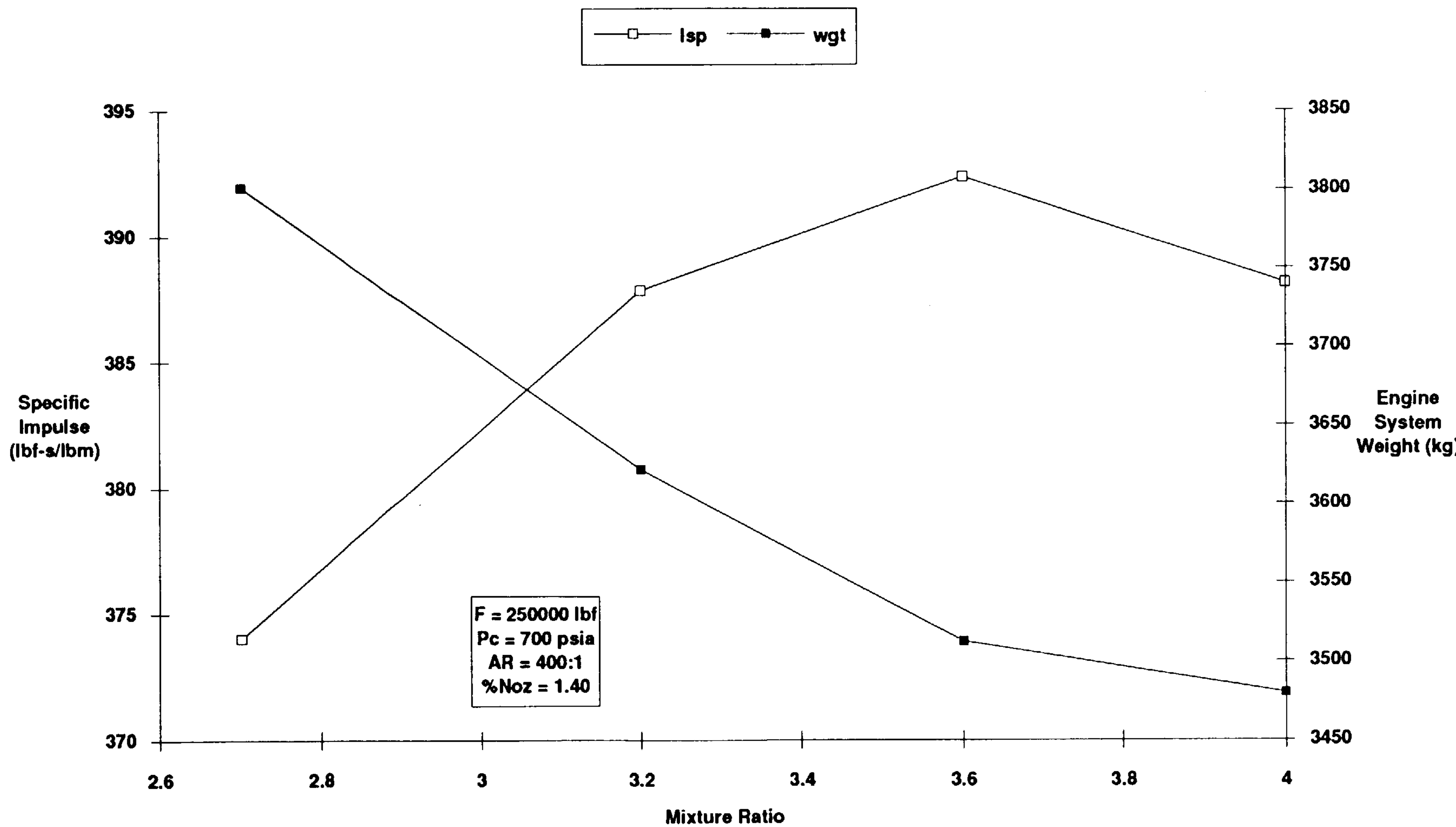
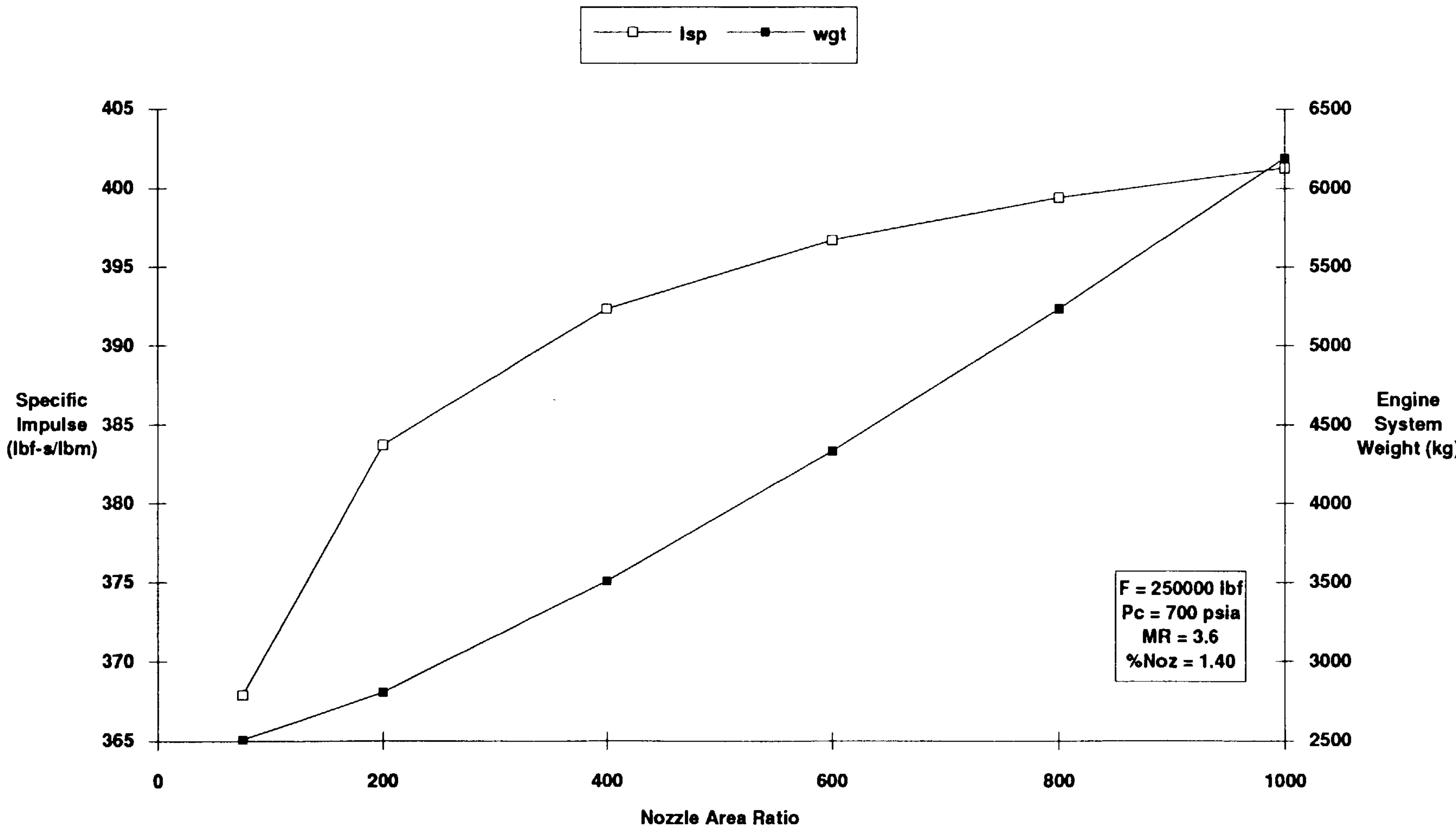
LO2/CH4 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Mixture Ratio –



PE01.9

Science Applications International Corporation

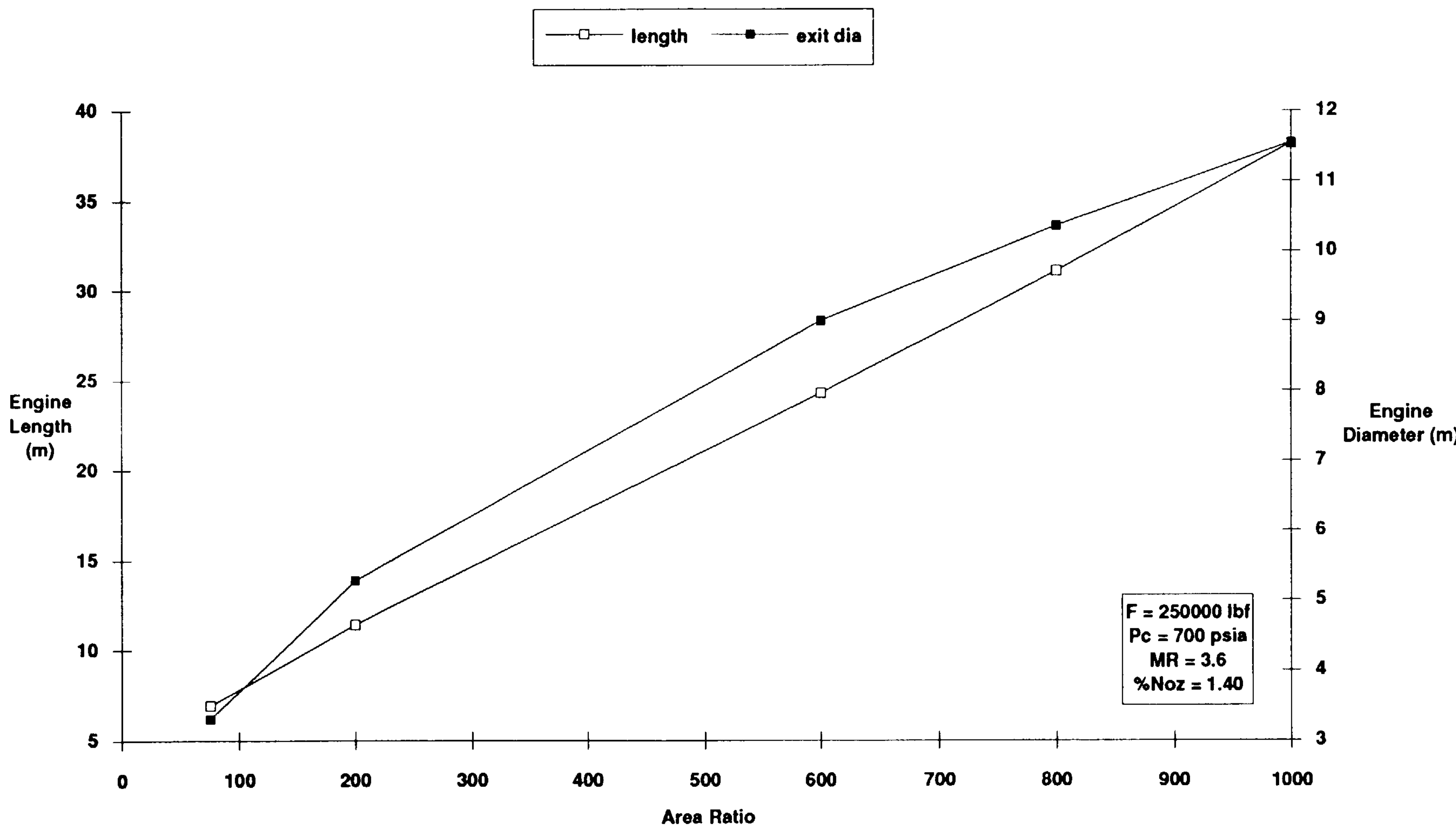
LO2/CH4 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Nozzle Area Ratio –



PE01.10

Science Applications International Corporation

LO2/CH4 EXPANDER CYCLE ENGINE
– Engine Length and Diameter as a Function of Area Ratio –

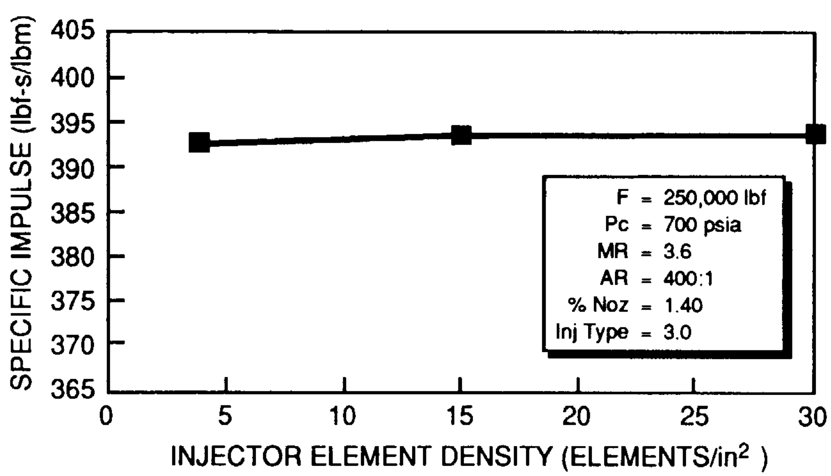


PE01.25

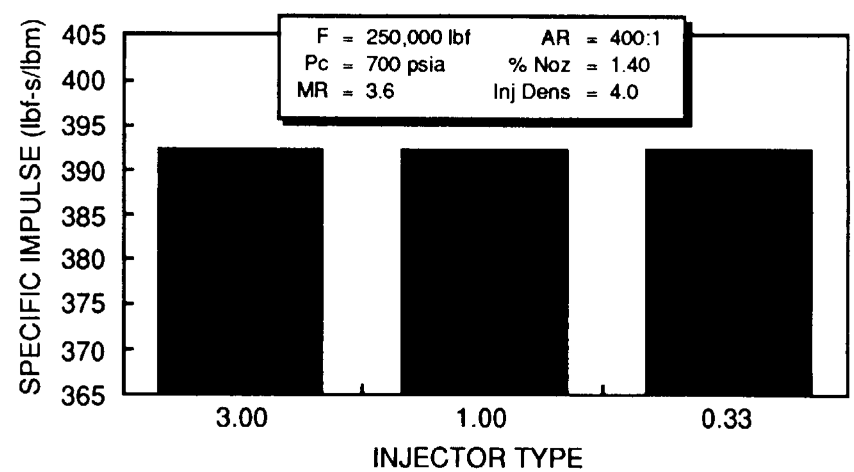
Science Applications International Corporation

LO_2/CH_4 EXPANDER CYCLE ENGINE
– Performance as a Function of Injector Design –

INJECTOR DENSITY INFLUENCE



INJECTOR ELEMENT TYPE INFLUENCE



NOTE:

Injector Density (Elements/in^2): 1.0 COARSE — 4.0 NOMINAL — 15.0 PLATELETS PATTERN TYPE — 40.0 HYPERTHIN PLATELET

Injector Element Type: 3.00 Showerhead, Shear Co-Axial
1.00 Likes-Doublets, Splash Plate, X Doublet, V Doublet, Pre-Atomized Triplet
0.50 Vortex, Swirl Co-Axial
0.33 Unlike Triplet, Unlike Doublet

INCREASING ATOMIZING EFFICIENCY

TOR29J/25

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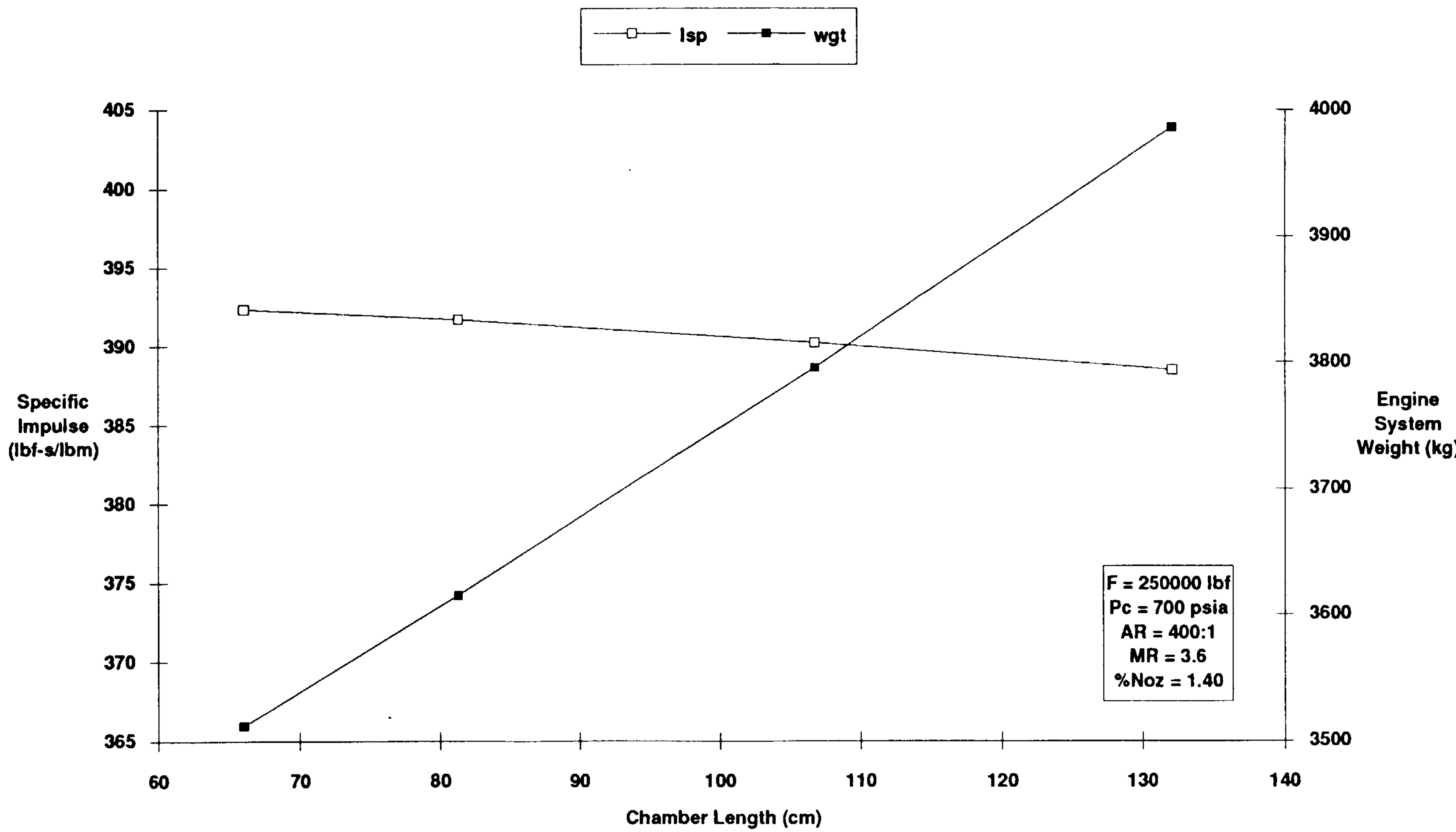
LO2/CH4 EXPANDER CYCLE ENGINE
– Performance and Weight as a Function of Chamber Length –



PE01.11

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ENGINE SYSTEM SENSITIVITY TRADE RESULTS

– In Situ Propellant Gas Generator Cycle Engines –

-- LO_2/CO --
-- LO_2/CH_4 --

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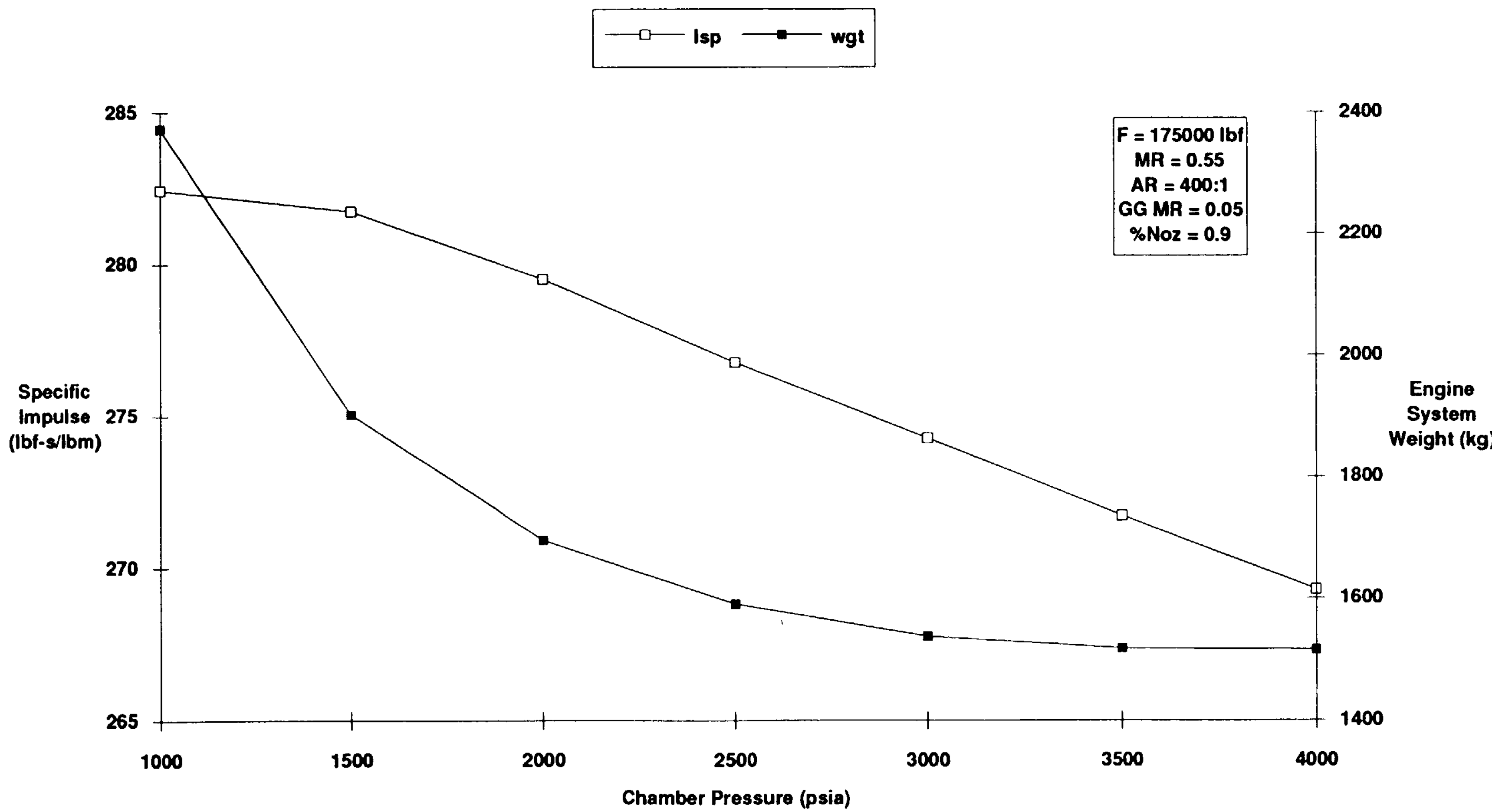
LO2/CO GAS GENERATOR CYCLE ENGINE – Performance and Weight as a Function of Chamber Pressure –



PE01.12

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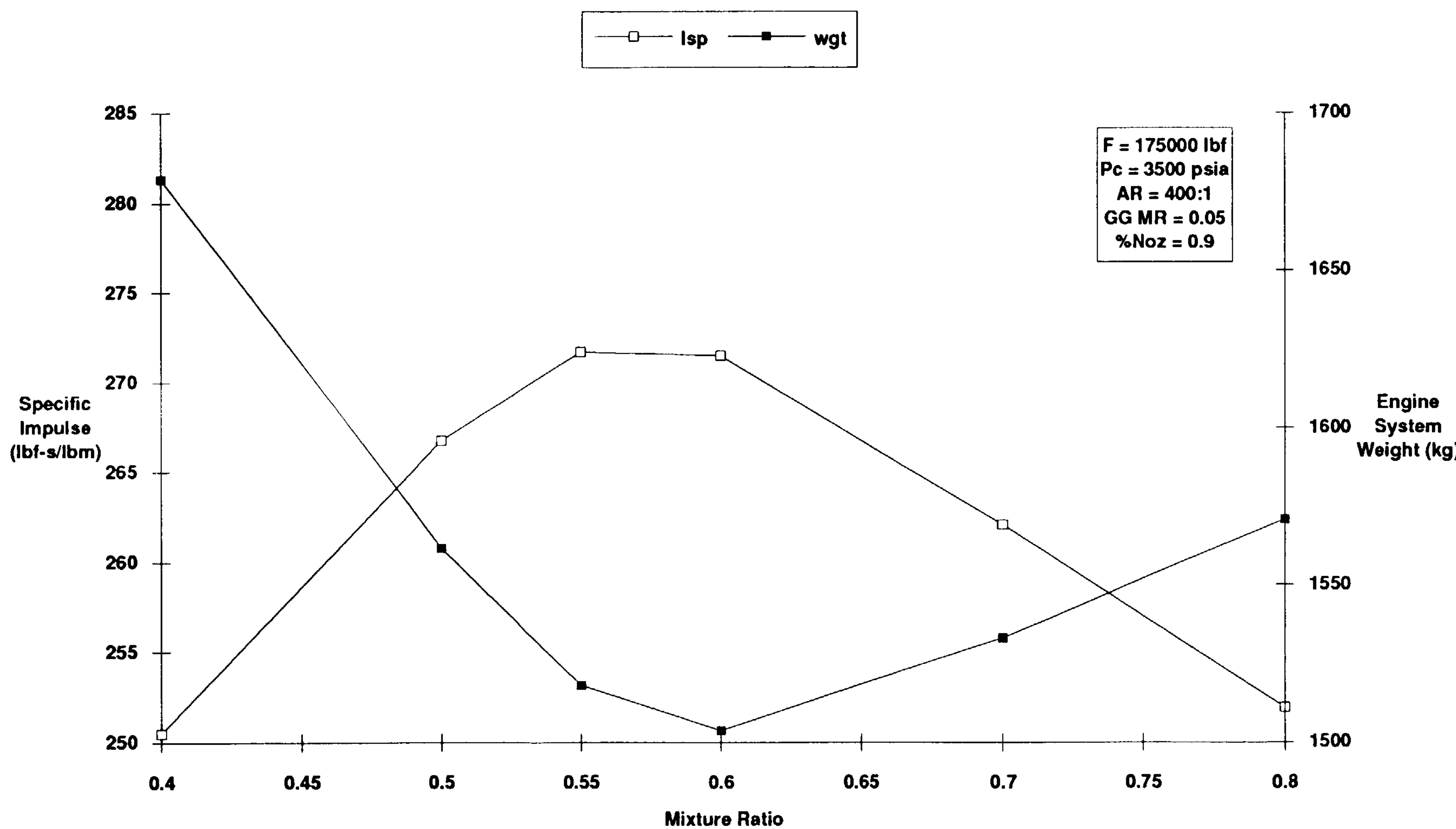
LO2/CO GAS GENERATOR CYCLE ENGINE
– Performance and Weight as a Function of Mixture Ratio –



PE01.13

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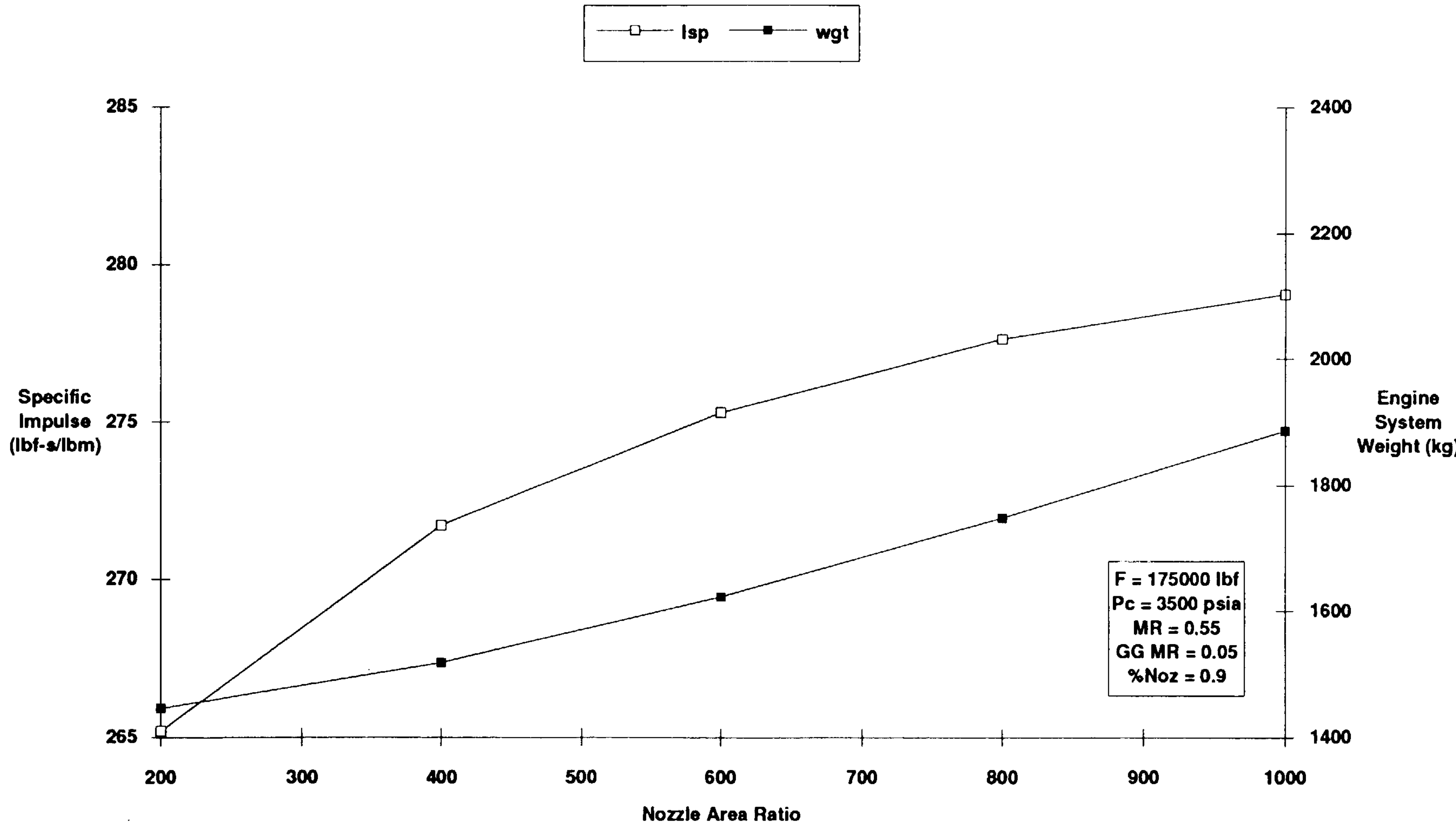
LO2/CO GAS GENERATOR CYCLE ENGINE
– Performance and Weight as a Function of Nozzle Area Ratio –



PE01.14

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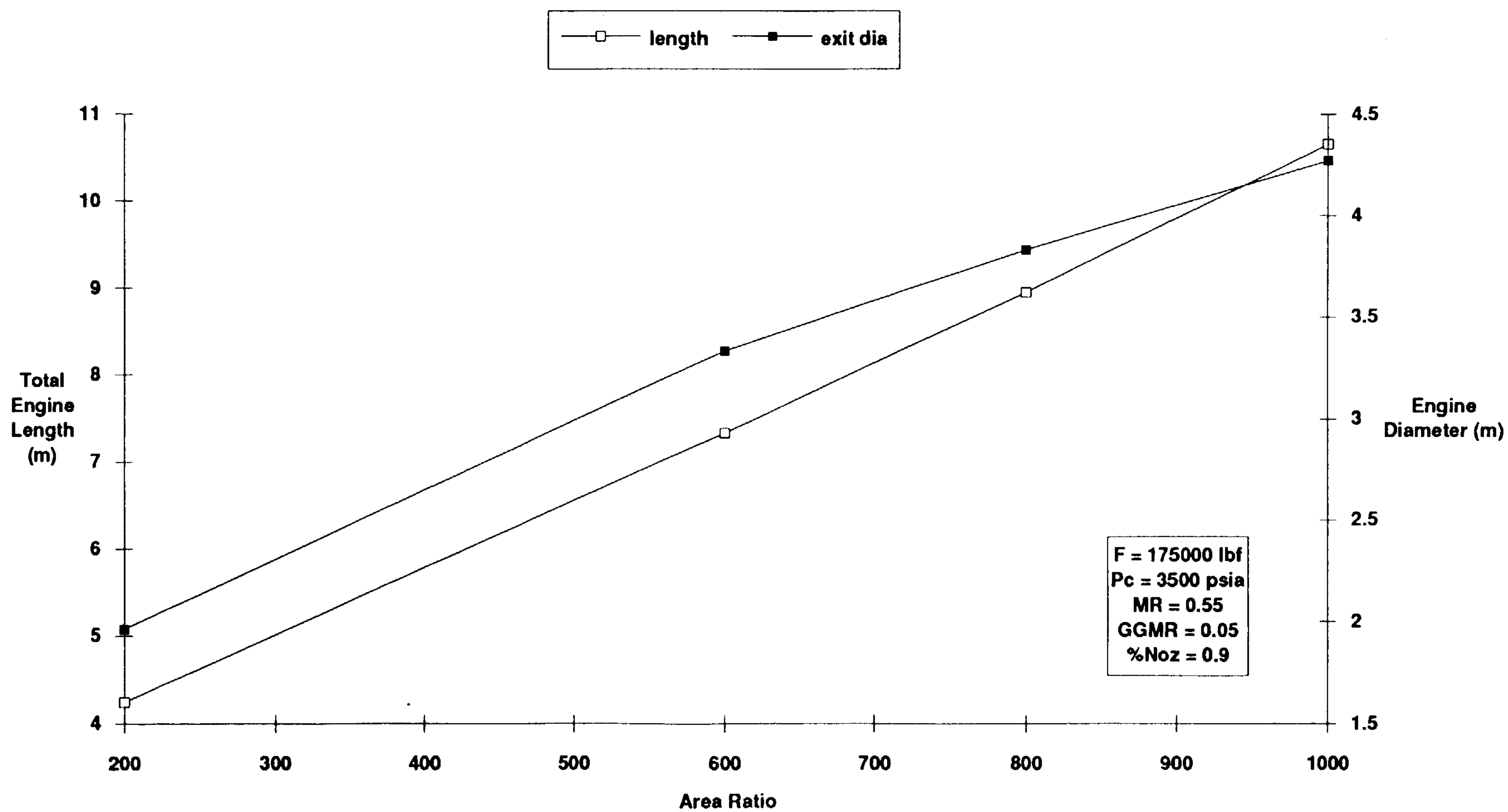
LO2/CO GAS GENERATOR CYCLE ENGINE
– Engine Length and Diameter as a Function of Nozzle of Area Ratio –



PE01.22

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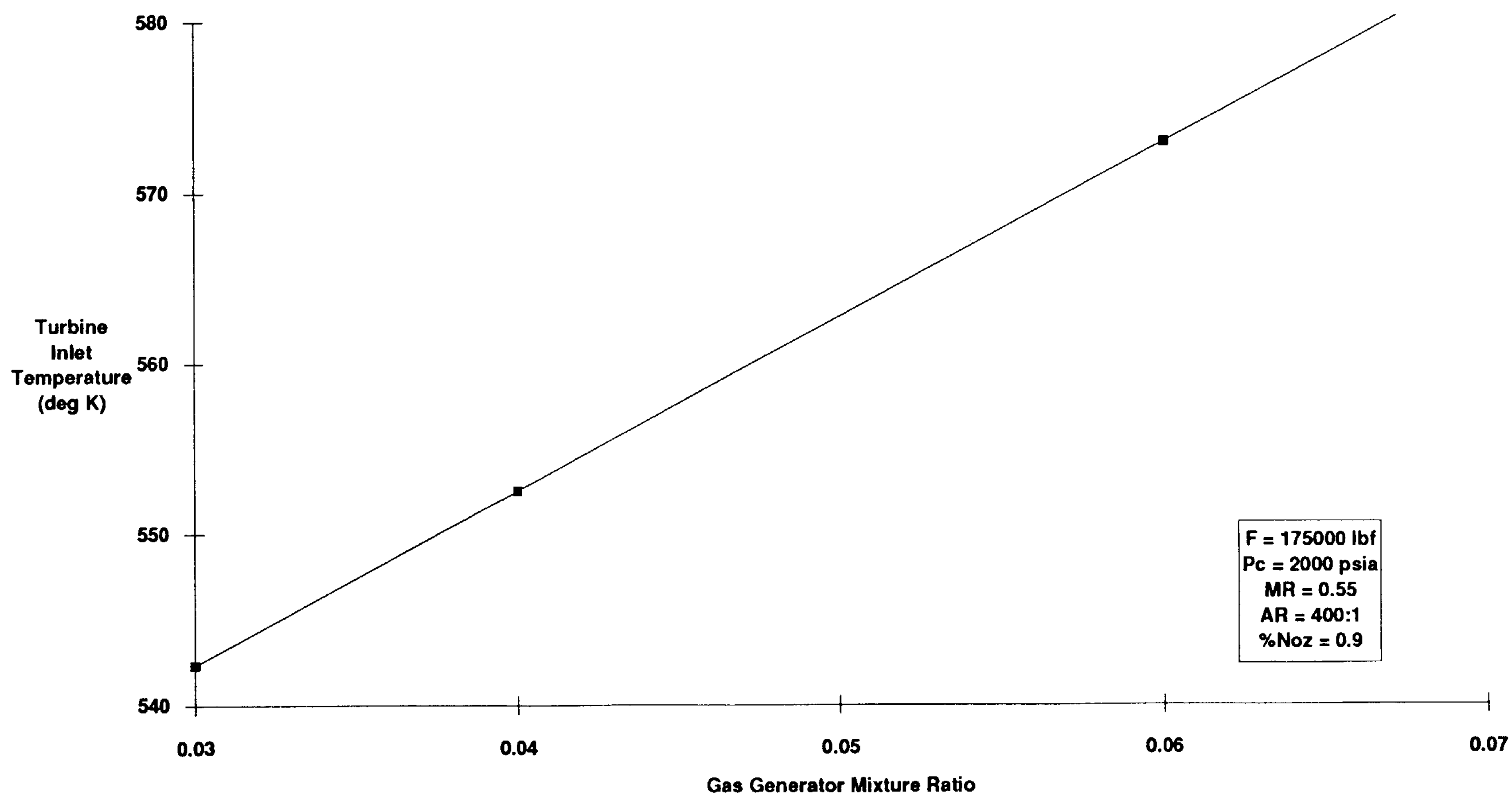
LO2/CO GAS GENERATOR CYCLE ENGINE
– Turbine Inlet Temperature as a Function of Gas Generator Mixture Ratio –



PE01.16

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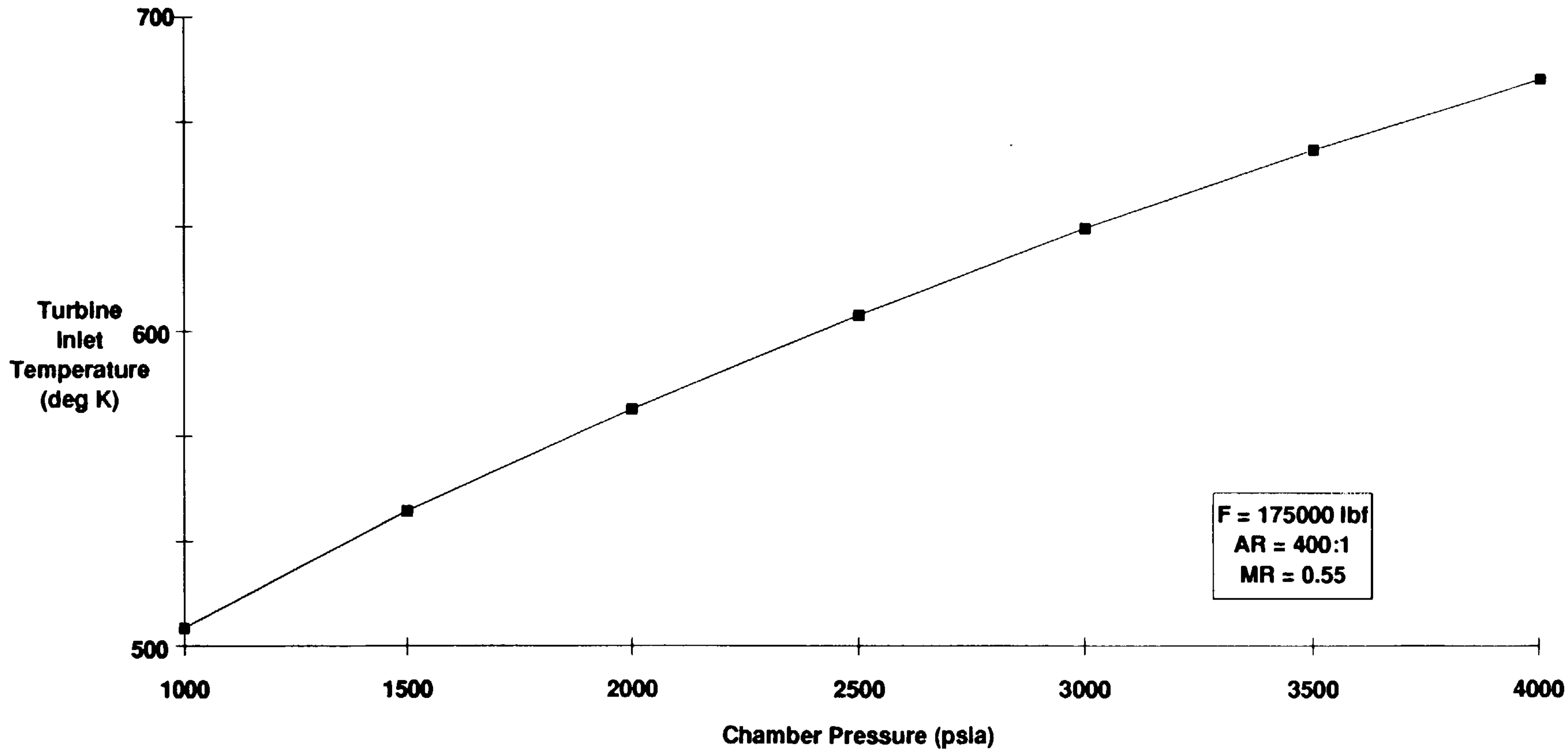
LO_2/CO GAS GENERATOR CYCLE ENGINE
- Turbine Inlet Temperature as a Function of Chamber Pressure -



TOR29J/27

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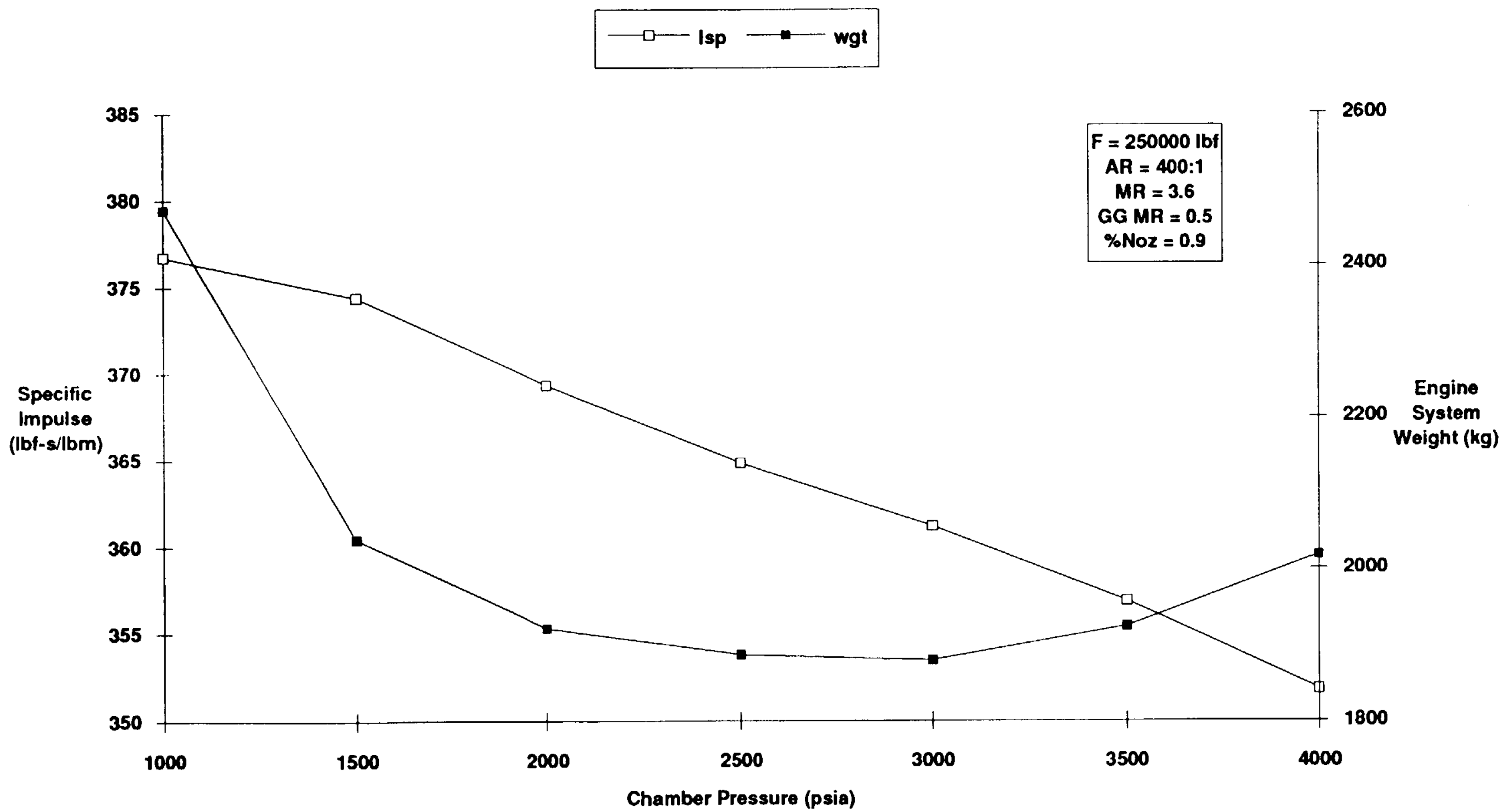
LO2/CH4 GAS GENERATOR CYCLE ENGINE - Performance and Weight as a Function of Chamber Pressure –



PE01.17

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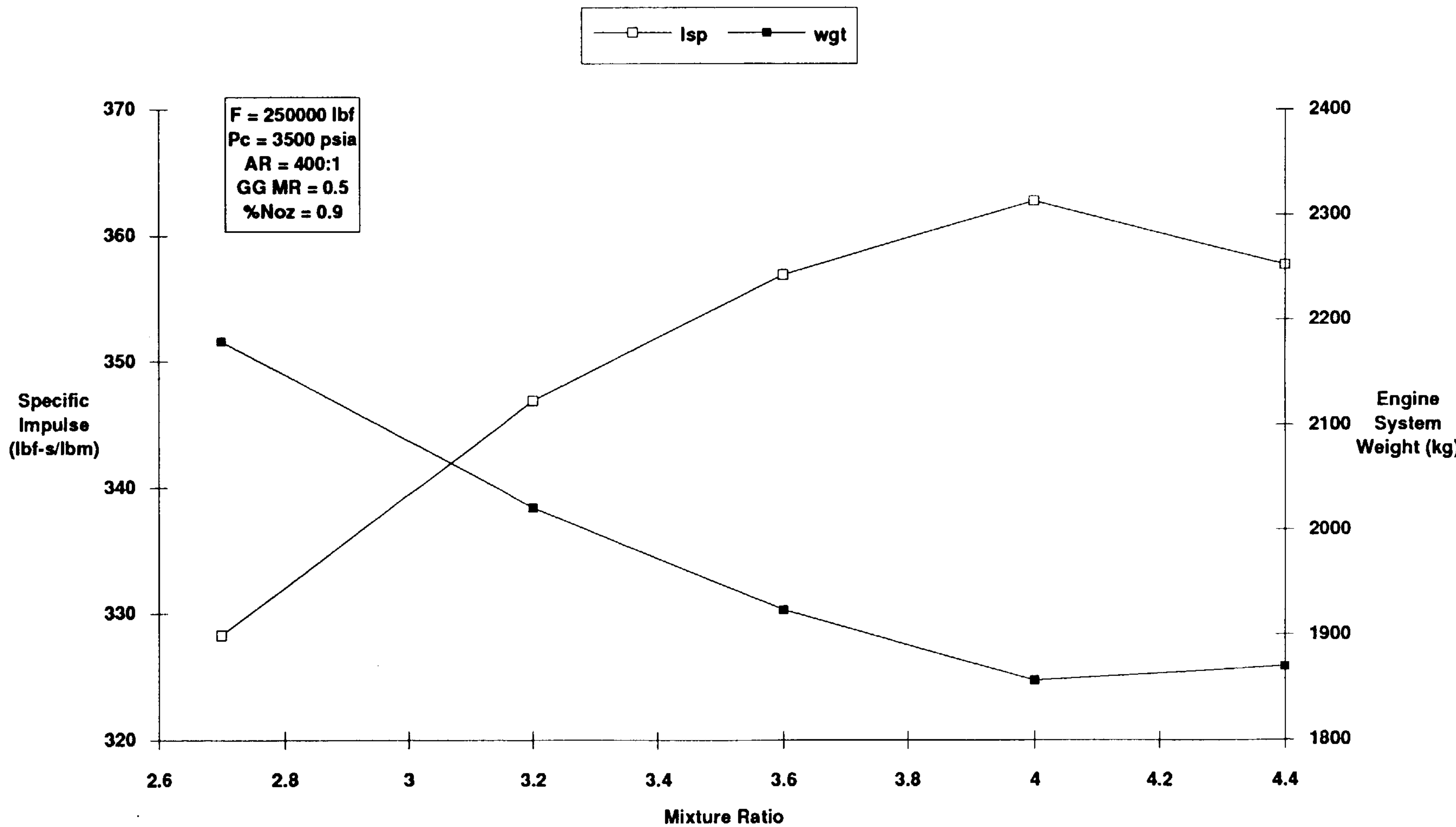
LO2/CH4 GAS GENERATOR CYCLE ENGINE
– Performance and Weight as a Function of Mixture Ratio –



PE01.19

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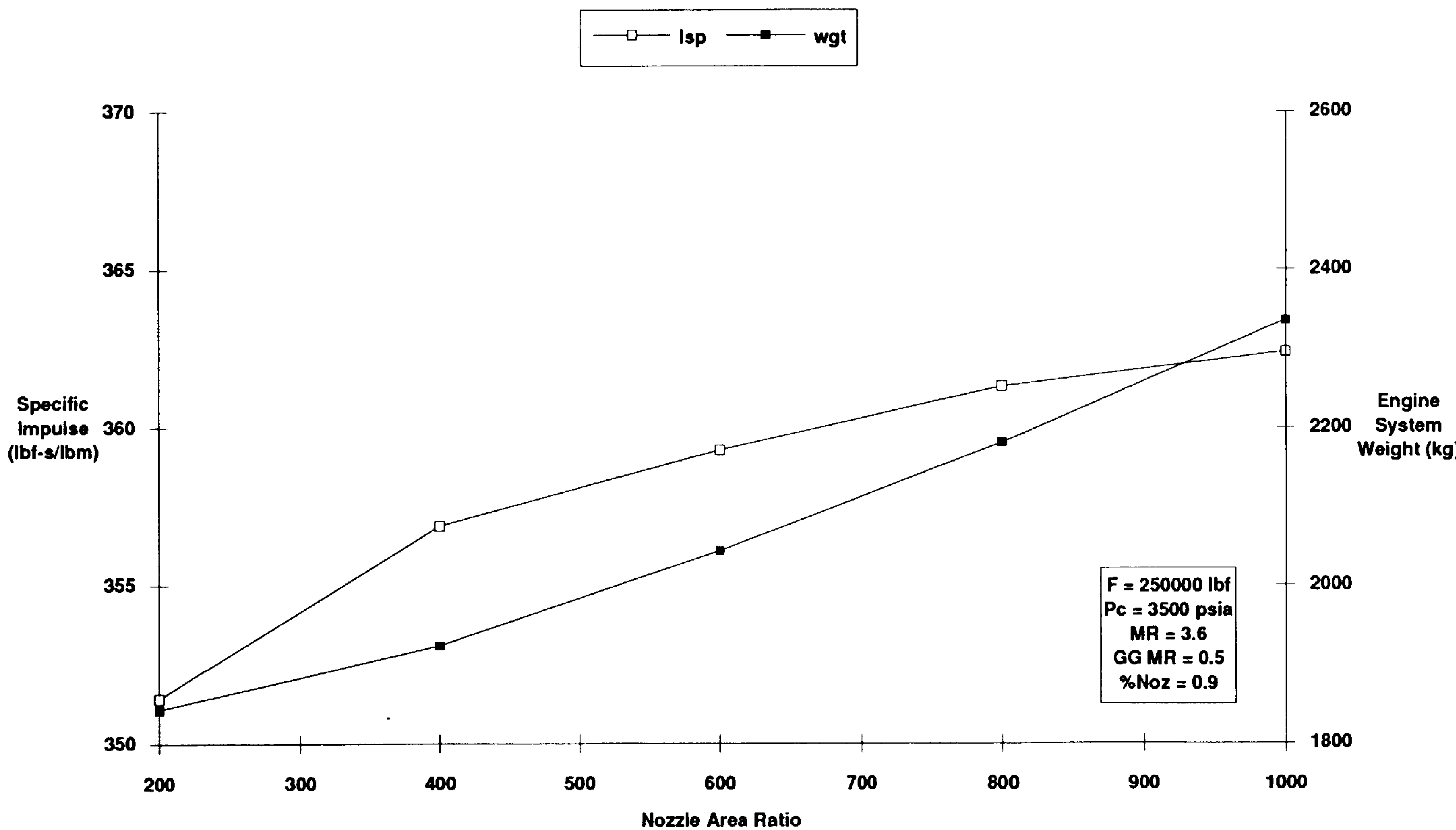
LO2/CH4 GAS GENERATOR CYCLE ENGINE
– Performance and Weight as a Function of Nozzle Area Ratio –



PE01.18

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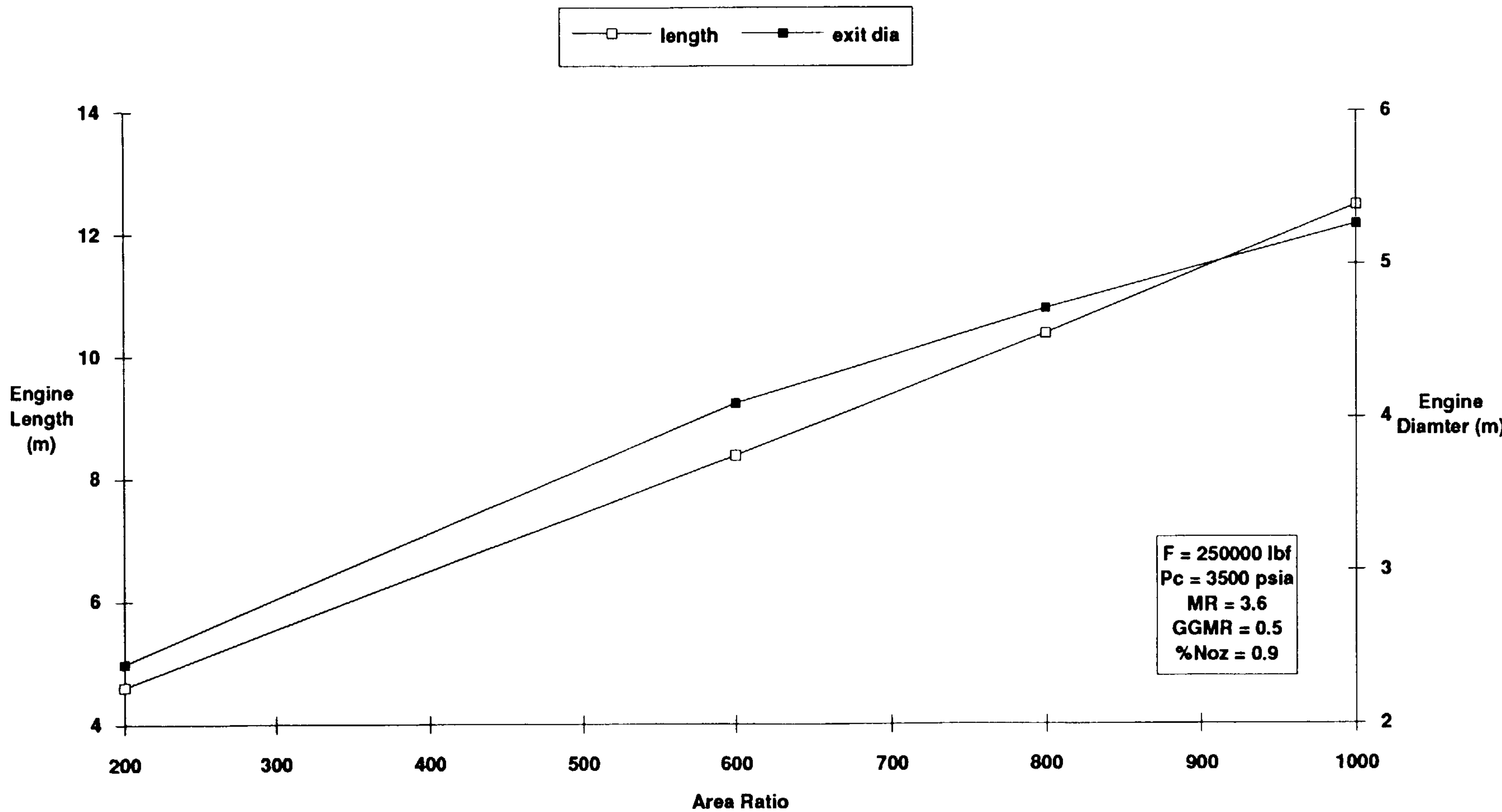
LO2/CH4 GAS GENERATOR CYCLE ENGINE – Engine Length and Diamter as a Function of Nozzle Area Ratio –



PE01.23

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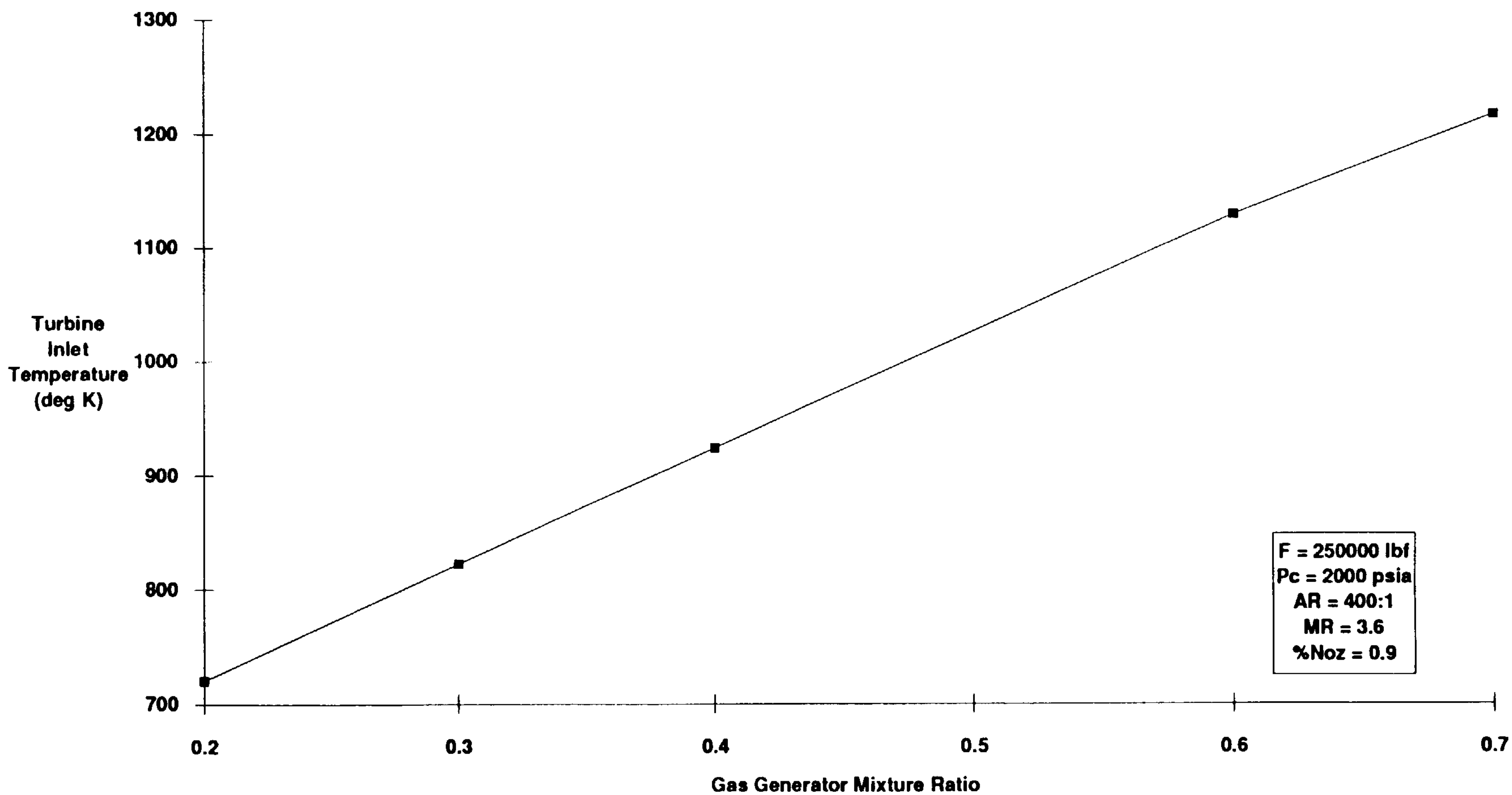
LO2/CH4 GAS GENERATOR CYCLE ENGINE
– Turbine Inlet Temperature as a Function of Gas Generator Mixture Ratio –



PE01.21

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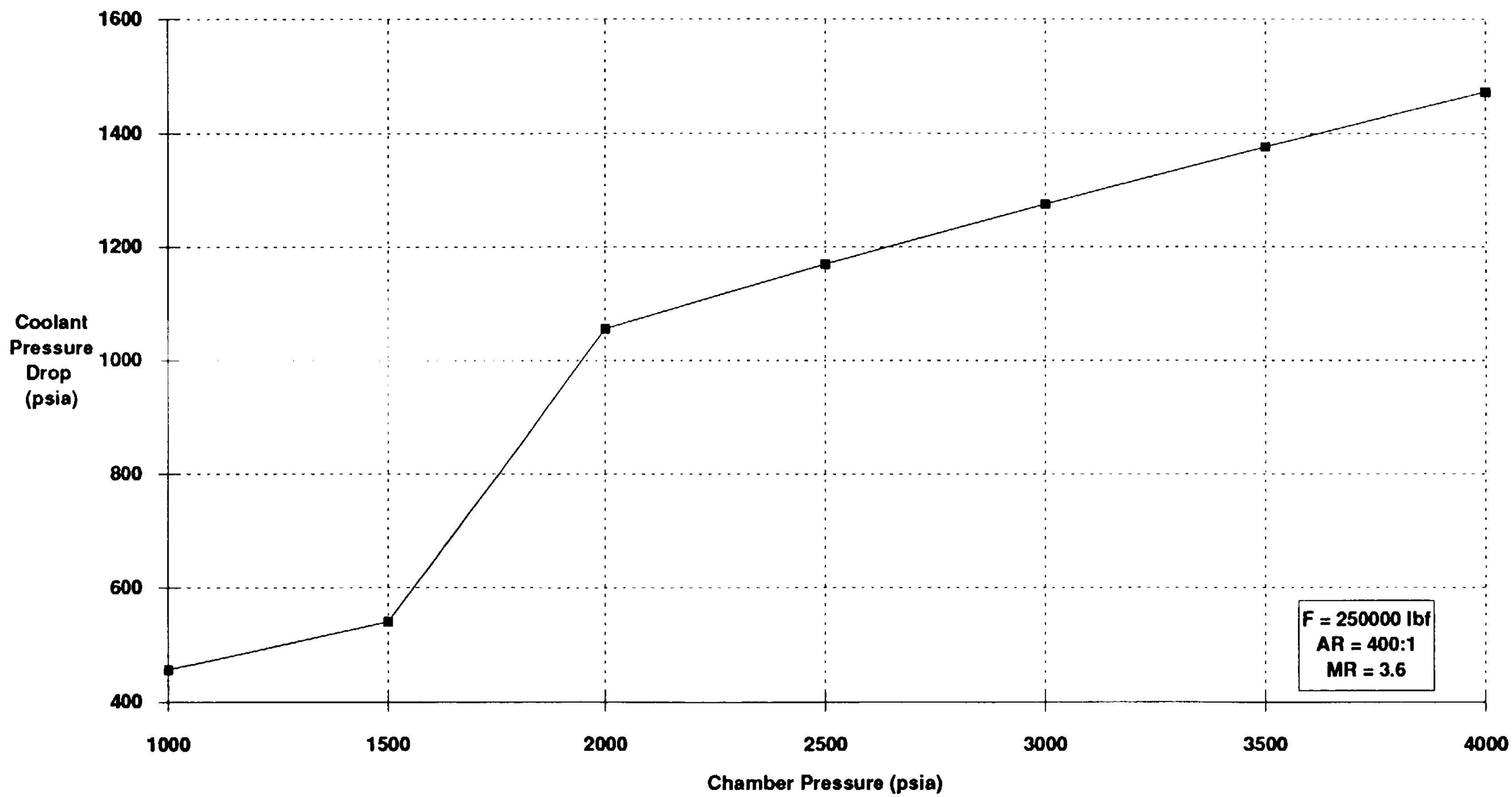
LO2/CH4 GAS GENERATOR CYCLE ENGINE
– Chamber Coolant Pressure Drop as a Function of Chamber Pressure –



PE01.20

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APPENDIX D

BASELINE ENGINE SYSTEM DESIGN DATA

APPENDIX D
BASELINE ENGINE SYSTEM DESIGN DATA

This appendix contains detailed engineering description data of the baseline engine systems discussed in Section 4.2.3. This database includes data pertaining to all these tripropellant engine systems baselined in this study for MEV applications and their bipropellant-based derivative designs for LEV and MEV applications. These engine systems are characterized for full rated power (100% thrust) and at reduced throttled (off-design) operating conditions. Typical engine system operational, thrust chamber/coolant, and chamber/injector design compatibility characteristics data are given in this appendix.

BASELINE ENGINES – FEATURES/DESCRIPTION AND OPERATIONAL CHARACTERISTICS

BASELINE ENGINES – FEATURES AND DESCRIPTION

Pe/02.58

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BASELINE ENGINES – FEATURES AND DESCRIPTION

- **Concept No. 1**
  - Tri-Propellant Engine - $LO_2/H_2/CO$
    - -- MTV Engine Candidate
  - Bi-Propellant Engines
    - -- LO_2/H_2
    - -- LO_2/CO

    LEV and/or MEV Engine Candidates

Pel/02.59

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	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		60%		60%		20%		20%	
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	699.2	kg	699.2	kg	699.2	kg	699.2	kg	699.2	kg	699.2	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CO		LO2/H2		LO2/CO		LO2/H2		LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6		0.55		6		0.55		6	
	Chamber Diameter	69.7	cm	69.7	cm	69.7	cm	69.7	cm	69.7	cm	69.7	cm
	Chamber Length	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3432	deg K	3309	deg K	3373	deg K	3189	deg K	3237	deg K
	Chamber Pressure	550	psia	585	psia	335	psia	350	psia	112	psia	115	psia
	Inconel Injector weight	313.3	kg	313.3	kg	313.3	kg	313.3	kg	313.3	kg	313.3	kg
	Propellant Mass Flow	270.7	kg/s	168.9	kg/s	163.9	kg/s	101.4	kg/s	56.3	kg/s	33.9	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg
	*Nozzle - Inconel, regen tubes	317.1	kg	317.1	kg	317.1	kg	317.1	kg	317.1	kg	317.1	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	34.8	cm	34.8	cm	34.8	cm	34.8	cm	34.8	cm	34.8	cm
	Exit Diameter	696.7	cm	696.7	cm	696.7	cm	696.7	cm	696.7	cm	696.7	cm
	Deployed Nozzle Length	971.3	cm	971.3	cm	971.3	cm	971.3	cm	971.3	cm	971.3	cm
	Delivered Vacuum Isp	293.22	sec	469.96	sec	290.56	sec	469.58	sec	282.09	sec	468.33	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	105000	lbf	105000	lbf	35000	lbf	35000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	14.7	kg	64.6	kg	14.7	kg	64.6	kg	14.7	kg	64.6	kg
	Material - Inconel												
	Number of Stages	1		2		1		2		1		2	
	Pressure Rise	873.0	psia	917.7	psia	522.4	psia	534.6	psia	160.1	psia	152.8	psia
	Pump Speed	14670	rpm	18699	rpm	11156	rpm	13987	rpm	5934	rpm	7188	rpm
	Pump Diameter	17.2	cm	30.8	cm	17.2	cm	30.8	cm	17.2	cm	30.8	cm
	Pump Horsepower	1958.5	HP	4546	HP	710.4	HP	1648.5	HP	78.7	HP	168.3	HP

	Pump Efficiency	0.857		0.733		0.855		0.707		0.812		0.661	
Main Oxidizer	Main Oxidizer Pump weight	4.1	kg	4.1	kg	4.1	kg	4.1	kg	4.1	kg	4.1	kg
Pump	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	2380.5	psia	1358.3	psia	2607.1	psia	731.9	psia	208.7	psia	453.5	psia
	Pump Speed	35554	rpm	28677	rpm	35745	rpm	20698	rpm	10281	rpm	15162	rpm
	Pump Diameter	9.4	cm	9.4	cm	9.4	cm	9.4	cm	9.4	cm	9.4	cm
	Pump Horsepower	1985.9	HP	1744.8	HP	1388.5	HP	564.6	HP	37.4	HP	120.3	HP
	Pump Efficiency	0.783		0.783		0.744		0.733		0.761		0.761	
Fuel Turbine	Fuel Turbine weight	6.8	kg	8.1	kg	6.8	kg	8.1	kg	6.8	kg	8.1	kg
	Material - Monel												
	Number of Stages	2		1		2		1		2		1	
	Pressure Ratio	2.7		1.18		2.45		1.11		1.15		1.04	
	Turbine Speed	14670	rpm	18699	rpm	11156	rpm	13987	rpm	5934	rpm	7188	rpm
	Turbine Efficiency	0.7		0.668		0.66		0.604		0.693		0.416	
	Turbine Diameter	11.4	cm	13.0	cm	11.4	cm	13.0	cm	11.4	cm	13.0	cm
Oxidizer	Oxidizer Turbine weight	2.0	kg	2.0	kg	2.0	kg	2.0	kg	2.0	kg	2.0	kg
Turbine	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.7		1.18		2.45		1.11		1.15		1.04	
	Turbine Speed	35554	rpm	28677	rpm	35745	rpm	20698	rpm	10281	rpm	15162	rpm
	Turbine Efficiency	0.7		0.585		0.639		0.496		0.566		0.449	
	Turbine Diameter	6.7	cm	6.7	cm	6.7	cm	6.7	cm	6.7	cm	6.7	cm
Fuel Boost Pump	Fuel Boost Pump weight	17.2	kg	7.6	kg	17.2	kg	7.6	kg	17.2	kg	7.6	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	132.5	psia	140.5	psia	60.1	psia	62.3	psia	11.8	psia	11.6	psia
	Pump Speed	6164	rpm	35392	rpm	5422	rpm	28655	rpm	3002	rpm	15321	rpm
	Pump Diameter	16.4	cm	11.1	cm	16.4	cm	11.1	cm	16.4	cm	11.1	cm
	Pump Horsepower	119.3	HP	264.5	HP	43.3	HP	90.3	HP	4.2	HP	7.9	HP
	Pump Efficiency	0.718		0.644		0.717		0.659		0.784		0.722	
Oxidizer Boost	Fuel Boost Pump weight	8.4	kg	8.4	kg	8.4	kg	8.4	kg	8.4	kg	8.4	kg
Pump	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	221.3	psia	307.9	psia	163.4	psia	129.4	psia	16.2	psia	33.4	psia
	Pump Speed	7723	rpm	8870	rpm	6229	rpm	5838	rpm	3914	rpm	5757	rpm

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	Pump Diameter	11.7	cm	11.7	cm	11.7	cm	11.7	cm	11.7	cm	11.7	cm
	Pump Horsepower	101.9	HP	103	HP	68.4	HP	30.8	HP	1.9	HP	6	HP
	Pump Efficiency	0.779		0.677		0.771		0.733		0.776		0.776	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	163.4	kg	163.4	kg	163.4	kg	163.4	kg	163.4	kg	163.4	kg
	Engine Lines	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg
	Main Valve	44.4	kg	44.4	kg	44.4	kg	44.4	kg	44.4	kg	44.4	kg
	Gimbal System	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	2994.78	kg		kg	2954.6	kg		kg	2954.6	kg		kg
	Throttling Factor Weight	1338.64	kg		kg	1338.6	kg		kg	1338.6	kg		kg
	Margin (2%)	86.6683	kg		kg	85.9	kg		kg	85.9	kg		kg
Total Engine	Weight	4420.08	kg		kg	4420.1	kg		kg	4420.1	kg		kg
System	Length	1164.3	cm		cm	1164.3	cm		cm	1164.3	cm		cm
	Diameter	696.7	cm		cm	696.7	cm		cm	696.7	cm		cm

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	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Expander		Expander	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	699.2	kg	699.2	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6	
	Chamber Diameter	69.7	cm	69.7	cm
	Chamber Length	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3432	deg K
	Chamber Pressure	550	psia	585	psia
	Inconel Injector weight	313.3	kg	313.3	kg
	Propellant Mass Flow	270.7	kg/s	168.9	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	1456.9	kg	1456.9	kg
	*Nozzle - Inconel, regen tubes	317.1	kg	317.1	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	1139.8	kg	1139.8	kg
	Area Ratio	400		400	
	Throat Diameter	34.8	cm	34.8	cm
	Exit Diameter	696.7	cm	696.7	cm
	Deployed Nozzle Length	971.3	cm	971.3	cm
	Delivered Vacuum Isp	293.2	sec	470.0	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

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ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	14.7	kg	64.6	kg
	Material - Inconel				
	Number of Stages	1		2	
	Pressure Rise	873.0	psia	917.7	psia
	Pump Speed	14670	rpm	18699	rpm
	Pump Diameter	17.2	cm	30.8	cm
	Pump Horsepower	1958.5	HP	4546	HP
	Pump Efficiency	0.857		0.733	
Main Oxidizer	Main Oxidizer Pump weight	4.1	kg	4.1	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	2380.5	psia	1358.3	psia
	Pump Speed	35554	rpm	28677	rpm
	Pump Diameter	9.4	cm	9.4	cm
	Pump Horsepower	1985.9	HP	1744.8	HP
	Pump Efficiency	0.783		0.783	
Fuel Turbine	Fuel Turbine weight	6.8	kg	8.1	kg
	Material - Monel				
	Number of Stages	2		1	
	Pressure Ratio	2.7		1.18	
	Turbine Speed	14670	rpm	18699	rpm
	Turbine Efficiency	0.7		0.668	
	Turbine Diameter	11.4	cm	13.0	cm
Oxidizer	Oxidizer Turbine weight	2.0	kg	2.0	kg
Turbine	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.7		1.18	
	Turbine Speed	35554	rpm	28677	rpm
	Turbine Efficiency	0.7		0.585	
	Turbine Diameter	6.7	cm	6.7	cm

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Fuel Boost Pump	Fuel Boost Pump weight	17.2	kg	7.6	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	132.5	psia	140.5	psia
	Pump Speed	6164	rpm	35392	rpm
	Pump Diameter	16.4	cm	11.1	cm
	Pump Horsepower	119.3	HP	264.5	HP
	Pump Efficiency	0.718		0.644	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.4	kg	8.4	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	221.3	psia	307.9	psia
	Pump Speed	7723	rpm	8870	rpm
	Pump Diameter	11.7	cm	11.7	cm
	Pump Horsepower	101.9	HP	103	HP
	Pump Efficiency	0.779		0.677	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	163.4	kg	163.4	kg
	Engine Lines	80.5	kg	80.5	kg
	Main Valve	44.4	kg	44.4	kg
	Gimbal System	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

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ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	2874.3	kg	2915.7	kg
	Throttling Factor Weight	1338.6	kg	1338.6	kg
	Margin (2%)	84.3	kg	85.1	kg
Total Engine	Weight	4297.2	kg	4339.5	kg
System	Length	1164.3	cm	1164.3	cm
	Diameter	696.7	cm	696.7	cm

	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		60%		60%		20%		20%	
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	165		165		165		165		165		165	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	720.5	kg	720.5	kg	720.5	kg	720.5	kg	720.5	kg	720.5	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CO		LO2/H2		LO2/CO		LO2/H2		LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6		0.55		6		0.55		6	
	Chamber Diameter	70.9	cm	70.9	cm	70.9	cm	70.9	cm	70.9	cm	70.9	cm
	Chamber Length	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3431	deg K	3309	deg K	3372	deg K	3189	deg K	3237	deg K
	Chamber Pressure	550	psia	580	psia	335	psia	345	psia	112	psia	115	psia
	Inconel Injector weight	328.9	kg	328.9	kg	328.9	kg	328.9	kg	328.9	kg	328.9	kg
	Propellant Mass Flow	280.2	kg/s	173.6	kg/s	169.6	kg/s	104.2	kg/s	58.1	kg/s	34.9	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	571.3	kg	571.3	kg	571.3	kg	571.3	kg	571.3	kg	571.3	kg
	*Nozzle - Inconel, regen tubes	328.4	kg	328.4	kg	328.4	kg	328.4	kg	328.4	kg	328.4	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg
	Area Ratio	165		165		165		165.0		165		165.0	
	Throat Diameter	35.4	cm	35.4	cm	35.4	cm	35.4	cm	35.4	cm	35.4	cm
	Exit Diameter	455.3	cm	455.3	cm	455.3	cm	455.3	cm	455.3	cm	455.3	cm
	Deployed Nozzle Length	513.6	cm	513.6	cm	513.6	cm	513.6	cm	513.6	cm	513.6	cm
	Delivered Vacuum Isp	283.23	sec	457.22	sec	280.73	sec	456.99	sec	273.10	sec	455.43	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	105000	lbf	105000	lbf	35000	lbf	35000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	15.3	kg	66.7	kg	15.3	kg	66.7	kg	15.3	kg	66.7	kg
	Material - Inconel												
	Number of Stages	1		2		1		2		1		2	
	Pressure Rise	873.0	psia	909.6	psia	522.5	psia	526.5	psia	160.2	psia	152.8	psia
	Pump Speed	14420	rpm	18328	rpm	10980	rpm	13668	rpm	5839	rpm	7075	rpm
	Pump Diameter	17.5	cm	31.2	cm	17.5	cm	31.2	cm	17.5	cm	31.2	cm
	Pump Horsepower	2021.8	HP	4625.5	HP	733.2	HP	1664.5	HP	81.1	HP	173.1	HP

	Pump Efficiency	0.859		0.733		0.857		0.707		0.814		0.66	
Main Oxidizer Pump	Main Oxidizer Pump weight	4.3	kg	4.3	kg	4.3	kg	4.3	kg	4.3	kg	4.3	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	2378.5	psia	1353.6	psia	2641.2	psia	724	psia	209.5	psia	464.1	psia
	Pump Speed	34911	rpm	28075	rpm	35281	rpm	20196	rpm	10102	rpm	15026	rpm
	Pump Diameter	9.6	cm	9.6	cm	9.6	cm	9.6	cm	9.6	cm	9.6	cm
	Pump Horsepower	2050.1	HP	1782.7	HP	1454.2	HP	572.5	HP	38.7	HP	126.6	HP
	Pump Efficiency	0.785		0.785		0.745		0.785		0.762		0.762	
Fuel Turbine	Fuel Turbine weight	7.1	kg	8.4	kg	7.1	kg	8.4	kg	7.1	kg	8.4	kg
	Material - Monel												
	Number of Stages	2		1		2		1		2		1	
	Pressure Ratio	2.69		1.18		2.46		1.1		1.16		1.04	
	Turbine Speed	14420	rpm	18328	rpm	10980	rpm	13668	rpm	5839	rpm	7075	rpm
	Turbine Efficiency	0.7		0.666		0.659		0.602		0.697		0.414	
	Turbine Diameter	11.6	cm	13.2	cm	11.6	cm	13.2	cm	11.6	cm	13.2	cm
Oxidizer Turbine	Oxidizer Turbine weight	2.1	kg	2.1	kg	2.1	kg	2.1	kg	2.1	kg	2.1	kg
	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.69		1.18		2.46		1.1		1.16		1.04	
	Turbine Speed	34911	rpm	28075	rpm	35281	rpm	20196	rpm	10102	rpm	15026	rpm
	Turbine Efficiency	0.7		0.584		0.638		0.493		0.558		0.449	
	Turbine Diameter	6.8	cm	6.8	cm	6.8	cm	6.8	cm	6.8	cm	6.8	cm
Fuel Boost Pump	Fuel Boost Pump weight	17.9	kg	7.8	kg	17.9	kg	7.8	kg	17.9	kg	7.8	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	132.5	psia	139.3	psia	60.1	psia	62.3	psia	11.8	psia	11.6	psia
	Pump Speed	6059	rpm	34986	rpm	5331	rpm	28655	rpm	2952	rpm	15148	rpm
	Pump Diameter	16.7	cm	11.2	cm	16.7	cm	11.2	cm	16.7	cm	11.2	cm
	Pump Horsepower	123.11	HP	269.5	HP	44.7	HP	90.3	HP	4.3	HP	8.1	HP
	Pump Efficiency	0.72		0.643		0.719		0.659		0.786		0.722	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.8	kg	8.8	kg	8.8	kg	8.8	kg	8.8	kg	8.8	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	221.3	psia	305.5	psia	164.9	psia	129.4	psia	16.3	psia	33.8	psia
	Pump Speed	7588	rpm	8641	rpm	6087	rpm	5838	rpm	3837	rpm	5712	rpm

ENGINE CONCEPT NO. 1 - MTV OPTIONS

	Pump Diameter	11.9	cm	11.9	cm	11.9	cm	11.9	cm	11.9	cm	11.9	cm
	Pump Horsepower	105.2	HP	104.8	HP	71.5	HP	30.8	HP	1.9	HP	6.3	HP
	Pump Efficiency	0.781		0.681		0.773		0.733		0.779		0.779	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	108.3	kg	108.3	kg	108.3	kg	108.3	kg	108.3	kg	108.3	kg
	Engine Lines	126.9	kg	126.9	kg	126.9	kg	126.9	kg	126.9	kg	126.9	kg
	Main Valve	46.0	kg	46.0	kg	46.0	kg	46.0	kg	46.0	kg	46.0	kg
	Gimbal System	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	2103.3	kg		kg	2581.7	kg		kg	2581.7	kg		kg
	Throttling Factor Weight	887.5	kg		kg	1147.9	kg		kg	1147.9	kg		kg
	Margin (2%)	59.8	kg		kg	74.6	kg		kg	74.6	kg		kg
Total Engine	Weight	3050.6	kg		kg	3050.6	kg		kg	3050.6	kg		kg
System	Length	706.6	cm		cm	706.6	cm		cm	706.6	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Expander		Expander	
	Area Ratio=	165		165	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	720.5	kg	720.5	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6	
	Chamber Diameter	70.9	cm	70.9	cm
	Chamber Length	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3431	deg K
	Chamber Pressure	550	psia	580	psia
	Inconel Injector weight	328.9	kg	328.9	kg
	Propellant Mass Flow	280.2	kg/s	173.6	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	571.3	kg	571.3	kg
	*Nozzle - Inconel, regen tubes	328.4	kg	328.4	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	242.9	kg	242.9	kg
	Area Ratio	400		400	
	Throat Diameter	35.4	cm	35.4	cm
	Exit Diameter	455.3	cm	455.3	cm
	Deployed Nozzle Length	513.6	cm	513.6	cm
	Delivered Vacuum Isp	283.2	sec	457.2	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	15.3	kg	66.7	kg
	Material - Inconel				
	Number of Stages	1		2	
	Pressure Rise	873.0	psia	909.6	psia
	Pump Speed	14420	rpm	18328	rpm
	Pump Diameter	17.5	cm	31.2	cm
	Pump Horsepower	2021.8	HP	4625.5	HP
	Pump Efficiency	0.859		0.733	
Main Oxidizer	Main Oxidizer Pump weight	4.3	kg	4.3	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	2378.5	psia	1353.6	psia
	Pump Speed	34911	rpm	28075	rpm
	Pump Diameter	9.6	cm	9.6	cm
	Pump Horsepower	2050.1	HP	1782.7	HP
	Pump Efficiency	0.785		0.785	
Fuel Turbine	Fuel Turbine weight	7.1	kg	8.4	kg
	Material - Monel				
	Number of Stages	2		1	
	Pressure Ratio	2.69		1.18	
	Turbine Speed	14420	rpm	18328	rpm
	Turbine Efficiency	0.7		0.666	
	Turbine Diameter	11.6	cm	13.2	cm
Oxidizer	Oxidizer Turbine weight	2.1	kg	2.1	kg
Turbine	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.69		1.18	
	Turbine Speed	34911	rpm	28075	rpm
	Turbine Efficiency	0.7		0.584	
	Turbine Diameter	6.8	cm	6.8	cm

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Fuel Boost Pump	Fuel Boost Pump weight	17.9	kg	7.8	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	132.5	psia	139.3	psia
	Pump Speed	6059	rpm	34986	rpm
	Pump Diameter	16.7	cm	11.2	cm
	Pump Horsepower	123.11	HP	269.5	HP
	Pump Efficiency	0.72		0.643	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.8	kg	8.8	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	221.3	psia	305.5	psia
	Pump Speed	7588	rpm	8641	rpm
	Pump Diameter	11.9	cm	11.9	cm
	Pump Horsepower	105.2	HP	104.8	HP
	Pump Efficiency	0.781		0.681	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	108.3	kg	108.3	kg
	Engine Lines	84.6	kg	84.6	kg
	Main Valve	46.0	kg	46.0	kg
	Gimbal System	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	1978.1	kg	2020.7	kg
	Throttling Factor Weight	887.5	kg	887.5	kg
	Margin (2%)	57.3	kg	58.2	kg
Total Engine	Weight	2922.9	kg	2966.4	kg
System	Length	706.6	cm	706.6	cm
	Diameter	457.0	cm	457.0	cm

	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		60%		60%		20%		20%	
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	395.6	kg	395.6	kg	395.6	kg	395.6	kg	395.6	kg	395.6	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CO		LO2/H2		LO2/CO		LO2/H2		LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6		0.55		6		0.55		6	
	Chamber Diameter	36.5	cm	36.5	cm	36.5	cm	36.5	cm	36.5	cm	36.5	cm
	Chamber Length	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm
	Chamber Temperature	3629	deg K	3637	deg K	3530	deg K	3540	deg K	3358	deg K	3399	deg K
	Chamber Pressure	2000	psia	2200	psia	1265	psia	1305	psia	425	psia	435	psia
	Inconel Injector weight	145.9	kg	145.9	kg	145.9	kg	145.9	kg	145.9	kg	145.9	kg
	Propellant Mass Flow	292.9	kg/s	171.0	kg/s	163.4	kg/s	102.7	kg/s	55.7	kg/s	34.4	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	287.0	kg	287.0	kg	287.0	kg	287.0	kg	287.0	kg	287.0	kg
	*Nozzle - Inconel, regen tubes	86.9	kg	86.9	kg	86.9	kg	86.9	kg	86.9	kg	86.9	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	200.1	kg	200.1	kg	200.1	kg	200.1	kg	200.1	kg	200.1	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm
	Exit Diameter	364.9	cm	364.9	cm	364.9	cm	364.9	cm	364.9	cm	364.9	cm
	Deployed Nozzle Length	508.8	cm	508.8	cm	508.8	cm	508.8	cm	508.8	cm	508.8	cm
	Delivered Vacuum Isp	289.66	sec	457.22	sec	286.97	sec	456.62	sec	280.72	sec	454.04	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	105000	lbf	105000	lbf	35000	lbf	35000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	9.3	kg	58.9	kg	9.3	kg	58.9	kg	9.3	kg	58.9	kg
	Material - Inconel												
	Number of Stages	1		3		1		3		1		3	
	Pressure Rise	3230.5	psia	3543.5	psia	2030.2	psia	2084.7	psia	667.1	psia	671.2	psia
	Pump Speed	34638	rpm	34468	rpm	26911	rpm	25895	rpm	14763	rpm	14064	rpm
	Pump Diameter	13.8	cm	26.6	cm	13.8	cm	26.6	cm	13.8	cm	26.6	cm
	Pump Horsepower	7709.1	HP	19672.7	HP	2976.7	HP	7199.3	HP	354.1	HP	847.6	HP

	Pump Efficiency	0.829	0.725	0.81	0.698	0.76	0.644
Main Oxidizer	Main Oxidizer Pump weight	3.5 kg	3.5 kg	3.5 kg	3.5 kg	3.5 kg	3.5 kg
Pump	Material - Inconel						
	Number of Stages	1	1	1	1	1	1
	Pressure Rise	3103.8 psia	3610.7 psia	1920.4 psia	2058.6 psia	619.5 psia	643.6 psia
	Pump Speed	43811 rpm	49015 rpm	33819 rpm	36307 rpm	18410 rpm	19445 rpm
	Pump Diameter	8.7 cm	8.7 cm	8.7 cm	8.7 cm	8.7 cm	8.7 cm
	Pump Horsepower	2500.3 HP	4664.9 HP	956.7 HP	1598.6 HP	112.3 HP	172.6 HP
	Pump Efficiency	0.774	0.774	0.757	0.774	0.713	0.751
Fuel Turbine	Fuel Turbine weight	14.6 kg	104.4 kg	14.6 kg	104.4 kg	14.6 kg	104.4 kg
	Material - Monel						
	Number of Stages	2	2	2	2	2	2
	Pressure Ratio	9.48	9.48	9.48	9.48	9.48	9.48
	Turbine Speed	34638 rpm	34468 rpm	26911 rpm	25895 rpm	14763 rpm	14064 rpm
	Turbine Efficiency	0.7	0.649	0.71	0.708	0.535	0.522
	Turbine Diameter	16.4 cm	42.2 cm	16.4 cm	42.2 cm	16.4 cm	42.2 cm
Oxidizer	Oxidizer Turbine weight	8.9 kg	8.9 kg	8.9 kg	8.9 kg	8.9 kg	8.9 kg
Turbine	Material - Monel						
	Number of Stages	2	2	2	2	2	2
	Pressure Ratio	9.48	9.48	9.48	9.48	9.48	9.48
	Turbine Speed	43811 rpm	49015 rpm	33819 rpm	36307 rpm	18410 rpm	19445 rpm
	Turbine Efficiency	0.7	0.549	0.709	0.428	0.528	0.215
	Turbine Diameter	13.0 cm	13.0 cm	13.0 cm	13.0 cm	13.0 cm	13.0 cm
Fuel Boost Pump	Fuel Boost Pump weight	24.5 kg	11.4 kg	24.5 kg	11.4 kg	24.5 kg	11.4 kg
	Material - Inconel						
	Centrifugal Pump						
	Pressure Rise	484.7 psia	532.7 psia	192.7 psia	236.4 psia	45.4 psia	48.1 psia
	Pump Speed	6075 rpm	34156 rpm	4073 rpm	22911 rpm	1757 rpm	26523 rpm
	Pump Diameter	19.3 cm	13.4 cm	19.3 cm	13.4 cm	19.3 cm	13.4 cm
	Pump Horsepower	397.9 HP	921.7 HP	150 HP	325.8 HP	18 HP	34.9 HP
	Pump Efficiency	0.82	0.789	0.82	0.788	0.762	0.798
Oxidizer Boost	Fuel Boost Pump weight	8.9 kg	8.9 kg	8.9 kg	8.9 kg	8.9 kg	8.9 kg
Pump	Material - Inconel						
	Centrifugal Pump						
	Pressure Rise	547 psia	1131.1 psia	249.2 psia	476.5 psia	49.4 psia	76.4 psia
	Pump Speed	7905 rpm	10472 rpm	5335 rpm	6984 rpm	6557 rpm	2955 rpm

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	Pump Diameter	12.0	cm	12.0	cm	12.0	cm	12.0	cm	12.0	cm	12.0	cm
	Pump Horsepower	127.6	HP	234.6	HP	47.8	HP	81	HP	5.2	HP	8.5	HP
	Pump Efficiency	0.774		0.786		0.774		0.78		0.783		0.774	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	55.5	kg	55.5	kg	55.5	kg	55.5	kg	55.5	kg	55.5	kg
	Engine Lines	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg
	Main Valve	25.5	kg	25.5	kg	25.5	kg	25.5	kg	25.5	kg	25.5	kg
	Gimbal System	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	73.7	kg	73.7	kg	73.7	kg	73.7	kg	73.7	kg	73.7	kg
	Gas Generator	18.4	kg	18.4	kg	18.4	kg	18.4	kg	18.4	kg	18.4	kg
	Gas Generator Features:												
	*Mixture Ratio	0.05		0.75		0.05		0.75		0.05		0.75	
	*Temperature	562.8	deg K	788.3	deg K	562.8	deg K	788.3	deg K	562.8	deg K	788.3	deg K
	*Pressure	948.2	psia	948.2	psia	948.2	psia	948.2	psia	948.2	psia	948.2	psia
	*Mass Flow Rate	23.1	kg/s	8.5	kg/s	8.8	kg/s	3.1	kg/s	1.0	kg/s	0.4	kg/s
Subtotal	Engine Weight	1429.9	kg		kg	1389.7	kg		kg	1389.7	kg		kg
	Throttling Factor Weight	454.8	kg		kg	454.8	kg		kg	454.8	kg		kg
	Margin (2%)	37.7	kg		kg	36.9	kg		kg	36.9	kg		kg
Total Engine	Weight	1922.4	kg		kg	1922.4	kg		kg	1922.4	kg		kg
System	Length	701.8	cm		cm	701.8	cm		cm	701.8	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CO/H2		LO2/CO/H2	
	Cycle Type	Gas Generator		Gas Generator	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	395.6	kg	395.6	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CO		LO2/H2	
	Mixture Ratio	0.55		6	
	Chamber Diameter	36.5	cm	36.5	cm
	Chamber Length	91.4	cm	91.4	cm
	Chamber Temperature	3629	deg K	3637	deg K
	Chamber Pressure	2000	psia	2200	psia
	Inconel Injector weight	145.9	kg	145.9	kg
	Propellant Mass Flow	292.9	kg/s	171.0	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	287.0	kg	287.0	kg
	*Nozzle - Inconel, regen tubes	86.9	kg	86.9	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	200.1	kg	200.1	kg
	Area Ratio	400		400	
	Throat Diameter	18.2	cm	18.2	cm
	Exit Diameter	364.9	cm	364.9	cm
	Deployed Nozzle Length	508.8	cm	508.8	cm
	Delivered Vacuum Isp	289.7	sec	457.2	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	9.3	kg	58.9	kg
	Material - Inconel				
	Number of Stages	1		3	
	Pressure Rise	3230.5	psia	3543.5	psia
	Pump Speed	34638	rpm	34468	rpm
	Pump Diameter	13.8	cm	26.6	cm
	Pump Horsepower	7709.1	HP	19672.7	HP
	Pump Efficiency	0.829		0.725	
Main Oxidizer Pump	Main Oxidizer Pump weight	3.5	kg	3.5	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	3103.8	psia	3610.7	psia
	Pump Speed	43811	rpm	49015	rpm
	Pump Diameter	8.7	cm	8.7	cm
	Pump Horsepower	2500.3	HP	4664.9	HP
	Pump Efficiency	0.774		0.774	
Fuel Turbine	Fuel Turbine weight	14.6	kg	104.4	kg
	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.48		9.48	
	Turbine Speed	34638	rpm	34468	rpm
	Turbine Efficiency	0.7		0.649	
	Turbine Diameter	16.4	cm	42.2	cm
Oxidizer Turbine	Oxidizer Turbine weight	8.9	kg	8.9	kg
	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.48		9.48	
	Turbine Speed	43811	rpm	49015	rpm
	Turbine Efficiency	0.7		0.549	
	Turbine Diameter	13.0	cm	13.0	cm

Fuel Boost Pump	Fuel Boost Pump weight	24.5	kg	11.4	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	484.7	psia	532.7	psia
	Pump Speed	6075	rpm	34156	rpm
	Pump Diameter	19.3	cm	13.4	cm
	Pump Horsepower	397.9	HP	921.7	HP
	Pump Efficiency	0.82		0.789	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.9	kg	8.9	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	547	psia	1131.1	psia
	Pump Speed	7905	rpm	10472	rpm
	Pump Diameter	12.0	cm	12.0	cm
	Pump Horsepower	127.6	HP	234.6	HP
	Pump Efficiency	0.774		0.786	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	55.5	kg	55.5	kg
	Engine Lines	80.5	kg	80.5	kg
	Main Valve	25.5	kg	25.5	kg
	Gimbal System	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	73.7	kg	73.7	kg
	Gas Generator	18.4	kg	18.4	kg
	Gas Generator Features:				
	*Mixture Ratio	0.05		0.75	
	*Temperature	562.8	deg K	788.3	deg K
	*Pressure	948.2	psia	948.2	psia
	*Mass Flow Rate	23.1	kg/s	8.5	kg/s

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ENGINE CONCEPT NO. 1 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	1215.0	kg	1341.3	kg
	Throttling Factor Weight	454.8	kg	454.8	kg
	Margin (2%)	33.4	kg	35.9	kg
Total Engine	Weight	1703.3	kg	1832.0	kg
System	Length	701.8	cm	701.8	cm
	Diameter	457.0	cm	457.0	cm

BASELINE ENGINES – FEATURES AND DESCRIPTION

- **Concept No. 2**
  - Tri-Propellant Engine - $LO_2/H_2/CH_4$
    - MTV Engine Candidate
  - Bi-Propellant Engines
    - LO_2/H_2
    - LO_2/CH_4

    LEV and/or MEV Engine Candidates

Pel/02.60

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	Rated Thrust (Vac)=	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf
	Percent Rated Thrust =	100%		100%		72.60%		72.60%		45.20%		45.20%	
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	672.2	kg	672.2	kg	672.2	kg	672.2	kg	672.2	kg	672.2	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CH4		LO2/H2		LO2/CH4		LO2/H2		LO2/CH4		LO2/H2	
	Mixture Ratio	3.6		6		0.55		6		0.55		6	
	Chamber Diameter	74.2	cm	74.2	cm	74.2	cm	74.2	cm	74.2	cm	74.2	cm
	Chamber Length	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm
	Chamber Temperature	3514	deg K	3457	deg K	3476	deg K	3422	deg K	3380	deg K	3457	deg K
	Chamber Pressure	700	psia	735	psia	508	psia	530	psia	320	psia	735	psia
	Inconel Injector weight	452.9	kg	452.9	kg	452.9	kg	452.9	kg	452.9	kg	452.9	kg
	Propellant Mass Flow	290.8	kg/s	240.0	kg/s	211.6	kg/s	174.2	kg/s	132.6	kg/s	240.0	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	1687.3	kg	1687.3	kg	1687.3	kg	1687.3	kg	1687.3	kg	1687.3	kg
	*Nozzle - Inconel, regen tubes	358.6	kg	358.6	kg	358.6	kg	358.6	kg	358.6	kg	358.6	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	1328.6	kg	1328.6	kg	1328.6	kg	1328.6	kg	1328.6	kg	1328.6	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	37.1	cm	37.1	cm	37.1	cm	37.1	cm	37.1	cm	37.1	cm
	Exit Diameter	741.9	cm	741.9	cm	741.9	cm	741.9	cm	741.9	cm	741.9	cm
	Deployed Nozzle Length	1034.5	cm	1034.5	cm	1034.5	cm	1034.5	cm	1034.5	cm	1034.5	cm
	Delivered Vacuum Isp	389.89	sec	472.29	sec	389.04	sec	472.41	sec	386.73	sec	472.29	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf	181500	lbf	181500	lbf	113000	lbf	113000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	6.9	kg	77.8	kg	6.9	kg	77.8	kg	6.9	kg	77.8	kg
	Material - Inconel												
	Number of Stages	1		2		1		2		1		2	
	Pressure Rise	1126.1	psia	1161.6	psia	813.0	psia	827.3	psia	506.9	psia	1161.6	psia
	Pump Speed	32829	rpm	19259	rpm	27631	rpm	16047	rpm	21392	rpm	19259	rpm
	Pump Diameter	12.1	cm	33.7	cm	12.1	cm	33.7	cm	12.1	cm	33.7	cm
	Pump Horsepower	1952.6	HP	7877.4	HP	1021.3	HP	4164.3	HP	406.5	HP	7877.4	HP

	Pump Efficiency	0.813		0.736		0.813		0.719		0.795		0.736	
Main Oxidizer	Main Oxidizer Pump weight	9.3	kg	9.3	kg	9.3	kg	9.3	kg	9.3	kg	9.3	kg
Pump	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3039.8	psia	1906.8	psia	1605.5	psia	1269.8	psia	760.8	psia	1906.8	psia
	Pump Speed	27134	rpm	21843	rpm	19811	rpm	17650	rpm	13531	rpm	21843	rpm
	Pump Diameter	13.9	cm	13.9	cm	13.9	cm	13.9	cm	13.9	cm	13.9	cm
	Pump Horsepower	5782.3	HP	3306.6	HP	2225	HP	1598.3	HP	663.8	HP	3306.6	HP
	Pump Efficiency	0.83		0.83		0.83		0.83		0.826		0.83	
Fuel Turbine	Fuel Turbine weight	1.9	kg	9.7	kg	1.9	kg	9.7	kg	1.9	kg	9.7	kg
	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.43		1.3		1.67		1.23		1.3		1.3	
	Turbine Speed	32829	rpm	19259	rpm	27631	rpm	16047	rpm	21392	rpm	19259	rpm
	Turbine Efficiency	0.7		0.659		0.709		0.625		0.707		0.659	
	Turbine Diameter	6.6	cm	14.1	cm	6.6	cm	14.1	cm	6.6	cm	14.1	cm
Oxidizer	Oxidizer Turbine weight	2.9	kg	2.9	kg	2.9	kg	2.9	kg	2.9	kg	2.9	kg
Turbine	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.43		1.3		1.67		1.23		1.3		1.3	
	Turbine Speed	27134	rpm	21843	rpm	19811	rpm	17650	rpm	13531	rpm	21843	rpm
	Turbine Efficiency	0.7		0.488		0.688		0.434		0.68		0.488	
	Turbine Diameter	7.9	cm	7.9	cm	7.9	cm	7.9	cm	7.9	cm	7.9	cm
Fuel Boost Pump	Fuel Boost Pump weight	9.3	kg	11.1	kg	9.3	kg	11.1	kg	9.3	kg	11.1	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	169.5	psia	176.9	psia	102	psia	105.4	psia	48.5	psia	176.9	psia
	Pump Speed	11762	rpm	30617	rpm	9216	rpm	23797	rpm	9935	rpm	30617	rpm
	Pump Diameter	12.2	cm	13.3	cm	12.2	cm	13.3	cm	12.2	cm	13.3	cm
	Pump Horsepower	109.6	HP	419.8	HP	54.4	HP	204.4	HP	22.6	HP	419.8	HP
	Pump Efficiency	0.739		0.704		0.778		0.747		0.73		0.704	
Oxidizer Boost	Fuel Boost Pump weight	23.5	kg	23.5	kg	23.5	kg	23.5	kg	23.5	kg	23.5	kg
Pump	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	282.1	psia	210.1	psia	153.6	psia	126	psia	66.5	psia	210.1	psia
	Pump Speed	4968	rpm	4224	rpm	3595	rpm	3234	rpm	2352	rpm	4224	rpm

	Pump Diameter	18.9	cm	18.9	cm	18.9	cm	18.9	cm	18.9	cm	18.9	cm
	Pump Horsepower	298.7	HP	169.6	HP	114.9	HP	82.6	HP	34.1	HP	169.6	HP
	Pump Efficiency	0.82		0.826		0.82		0.82		0.82		0.826	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg
	Thrust Support Hardware	185.8	kg	185.8	kg	185.8	kg	185.8	kg	185.8	kg	185.8	kg
	Engine Lines	263.5	kg	263.5	kg	263.5	kg	263.5	kg	263.5	kg	263.5	kg
	Main Valve	46.1	kg	46.1	kg	46.1	kg	46.1	kg	46.1	kg	46.1	kg
	Gimbal System	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	3533.8	kg		kg	3533.8	kg		kg	3446.0	kg		kg
	Throttling Factor Weight	304.4	kg		kg	304.4	kg		kg	304.4	kg	0.0	kg
	Margin (2%)	76.8	kg		kg	76.8	kg		kg	75.0	kg		kg
Total Engine	Weight	3915.0	kg		kg	3915.0	kg		kg	3915.0	kg		kg
System	Length	1214.9	cm		cm	1214.9	cm		cm	1214.9	cm		cm
	Diameter	741.9	cm		cm	741.9	cm		cm	741.9	cm		cm

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	Rated Thrust (Vac)=	250000	lbf	250000	lbf
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2	
	Cycle Type	Expander		Expander	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	672.2		672.2	
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CH4		LO2/H2	
	Mixture Ratio	3.6		6	
	Chamber Diameter	74.2	cm	74.2	cm
	Chamber Length	66.0	cm	66.0	cm
	Chamber Temperature	3514	deg K	3457	deg K
	Chamber Pressure	700	psia	735	psia
	Inconel Injector weight	452.9	kg	452.9	kg
	Propellant Mass Flow	290.8	kg/s	240.0	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	1687.3	kg	1687.3	kg
	*Nozzle - Inconel, regen tubes	358.6	kg	358.6	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	1328.6	kg	1328.6	kg
	Area Ratio	400		400	
	Throat Diameter	37.1	cm	37.1	cm
	Exit Diameter	741.9	cm	741.9	cm
	Deployed Nozzle Length	1034.5	cm	1034.5	cm
	Delivered Vacuum Isp	389.9	sec	472.3	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

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ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	6.9	kg	77.8	kg
	Material - Inconel				
	Number of Stages	1		2	
	Pressure Rise	1126.1	psia	1161.6	psia
	Pump Speed	32829	rpm	19259	rpm
	Pump Diameter	12.1	cm	33.7	cm
	Pump Horsepower	1952.6	HP	7877.4	HP
	Pump Efficiency	0.813		0.736	
Main Oxidizer	Main Oxidizer Pump weight	9.3	kg	9.3	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	3039.8	psia	1906.8	psia
	Pump Speed	27134	rpm	21843	rpm
	Pump Diameter	13.9	cm	13.9	cm
	Pump Horsepower	5782.3	HP	3306.6	HP
	Pump Efficiency	0.83		0.83	
Fuel Turbine	Fuel Turbine weight	1.9	kg	9.7	kg
	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.43		1.3	
	Turbine Speed	32829	rpm	19259	rpm
	Turbine Efficiency	0.7		0.659	
	Turbine Diameter	6.6	cm	14.1	cm
Oxidizer	Oxidizer Turbine weight	2.9	kg	2.9	kg
Turbine	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.43		1.3	
	Turbine Speed	27134	rpm	21843	rpm
	Turbine Efficiency	0.7		0.488	
	Turbine Diameter	7.9	cm	7.9	cm

ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Fuel Boost Pump	Fuel Boost Pump weight	9.3	kg	11.1	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	169.5	psia	176.9	psia
	Pump Speed	11762	rpm	30617	rpm
	Pump Diameter	12.2	cm	13.3	cm
	Pump Horsepower	109.6	HP	419.8	HP
	Pump Efficiency	0.739		0.704	
Oxidizer Boost Pump	Fuel Boost Pump weight	23.5	kg	23.5	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	282.1	psia	210.1	psia
	Pump Speed	4968	rpm	4224	rpm
	Pump Diameter	18.9	cm	18.9	cm
	Pump Horsepower	298.7	HP	169.6	HP
	Pump Efficiency	0.82		0.826	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg
	Thrust Support Hardware	185.8	kg	185.8	kg
	Engine Lines	175.6	kg	175.6	kg
	Main Valve	46.1	kg	46.1	kg
	Gimbal System	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

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Subtotal	Engine Weight	3347.5	kg	3427.9	kg
	Throttling Factor Weight	304.4	kg	304.4	kg
	Margin (2%)	73.0	kg	74.6	kg
Total Engine	Weight	3725.0	kg	3807.0	kg
System	Length	1214.9	cm	1214.9	cm
	Diameter	741.9	cm	741.9	cm

	Rated Thrust (Vac)=	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf
	Percent Rated Thrust =	100%		100%		72.60%		72.60%		45.20%		45.20%	
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	140		140		140		140		140		140	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	700.8	kg	700.8	kg	700.8	kg	700.8	kg	700.8	kg	700.8	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CH4		LO2/H2		LO2/CH4		LO2/H2		LO2/CH4		LO2/H2	
	Mixture Ratio	3.6		6		0.55		6		0.55		6	
	Chamber Diameter	75.8	cm	75.8	cm	75.8	cm	75.8	cm	75.8	cm	75.8	cm
	Chamber Length	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm
	Chamber Temperature	3514	deg K	3456	deg K	3476	deg K	3422	deg K	3380	deg K	3365	deg K
	Chamber Pressure	700	psia	730	psia	508	psia	525	psia	320	psia	325	psia
	Inconel Injector weight	480.8	kg	480.8	kg	480.8	kg	480.8	kg	480.8	kg	480.8	kg
	Propellant Mass Flow	303.3	kg/s	248.4	kg/s	220.7	kg/s	180.1	kg/s	138.0	kg/s	112.3	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	600.3	kg	600.3	kg	600.3	kg	600.3	kg	600.3	kg	600.3	kg
	*Nozzle - Inconel, regen tubes	374.1	kg	374.1	kg	374.1	kg	374.1	kg	374.1	kg	374.1	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	226.2	kg	226.2	kg	226.2	kg	226.2	kg	226.2	kg	226.2	kg
	Area Ratio	140		140		140		140.0		140		140.0	
	Throat Diameter	37.9	cm	37.9	cm	37.9	cm	37.9	cm	37.9	cm	37.9	cm
	Exit Diameter	448.3	cm	448.3	cm	448.3	cm	448.3	cm	448.3	cm	448.3	cm
	Deployed Nozzle Length	491.2	cm	491.2	cm	491.2	cm	491.2	cm	491.2	cm	491.2	cm
	Delivered Vacuum Isp	373.77	sec	456.46	sec	372.93	sec	457.02	sec	371.25	sec	456.39	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf	181500	lbf	181500	lbf	113000	lbf	113000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	7.3	kg	80.7	kg	7.3	kg	80.7	kg	7.3	kg	80.7	kg
	Material - Inconel												
	Number of Stages	1		2		1		2		1		2	
	Pressure Rise	1126.1	psia	1153.5	psia	813.0	psia	819.2	psia	507.0	psia	493.6	psia
	Pump Speed	32163	rpm	18851	rpm	27060	rpm	15686	rpm	20948	rpm	11954	rpm
	Pump Diameter	12.3	cm	34.3	cm	12.3	cm	34.3	cm	12.3	cm	34.3	cm
	Pump Horsepower	2028.1	HP	8085.4	HP	1061	HP	4257	HP	421.8	HP	1651.9	HP
	Pump Efficiency	0.815		0.736		0.815		0.719		0.797		0.696	

Main Oxidizer Pump	Main Oxidizer Pump weight	9.7	kg	9.7	kg	9.7	kg	9.7	kg	9.7	kg	9.7	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3076.7	psia	1910	psia	1622.6	psia	1266.5	psia	766	psia	709.9	psia
	Pump Speed	26790	rpm	21464	rpm	19560	rpm	17308	rpm	13335	rpm	12760	rpm
	Pump Diameter	14.1	cm	14.1	cm	14.1	cm	14.1	cm	14.1	cm	14.1	cm
	Pump Horsepower	6090.3	HP	3418.6	HP	2340.6	HP	1643.8	HP	694.5	HP	580.9	HP
	Pump Efficiency	0.832		0.832		0.832		0.832		0.829		0.823	
Fuel Turbine	Fuel Turbine weight	2.0	kg	10.0	kg	2.0	kg	10.0	kg	2.0	kg	10.0	kg
	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.44		1.3		1.67		1.23		1.3		1.15	
	Turbine Speed	32163	rpm	18851	rpm	27060	rpm	15686	rpm	20948	rpm	11954	rpm
	Turbine Efficiency	0.7		0.658		0.709		0.624		0.707		0.541	
	Turbine Diameter	6.7	cm	14.4	cm	6.7	cm	14.4	cm	6.7	cm	14.4	cm
Oxidizer Turbine	Oxidizer Turbine weight	2.9	kg	2.9	kg	2.9	kg	2.9	kg	2.9	kg	2.9	kg
	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.44		1.3		1.67		1.23		1.3		1.15	
	Turbine Speed	26790	rpm	21464	rpm	19560	rpm	17308	rpm	13335	rpm	12760	rpm
	Turbine Efficiency	0.7		0.486		0.687		0.432		0.68		0.338	
	Turbine Diameter	8.0	cm	8.0	cm	8.0	cm	8.0	cm	8.0	cm	8.0	cm
Fuel Boost Pump	Fuel Boost Pump weight	9.7	kg	11.4	kg	9.7	kg	11.4	kg	9.7	kg	11.4	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	169.5	psia	175.7	psia	102	psia	105.4	psia	48.4	psia	48.2	psia
	Pump Speed	11523	rpm	30034	rpm	9028	rpm	23797	rpm	9732	rpm	22659	rpm
	Pump Diameter	12.5	cm	13.5	cm	12.5	cm	13.5	cm	12.5	cm	13.5	cm
	Pump Horsepower	113.9	HP	430.3	HP	56.6	HP	204.4	HP	23.4	HP	81.8	HP
	Pump Efficiency	0.741		0.706		0.781		0.747		0.733		0.717	
Oxidizer Boost Pump	Fuel Boost Pump weight	24.7	kg	24.7	kg	24.7	kg	24.7	kg	24.7	kg	24.7	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	283.8	psia	209.4	psia	154.4	psia	126	psia	66.7	psia	58.2	psia
	Pump Speed	4864	rpm	4096	rpm	3509	rpm	3234	rpm	2292	rpm	2122	rpm
	Pump Diameter	19.4	cm	19.4	cm	19.4	cm	19.4	cm	19.4	cm	19.4	cm

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	Pump Horsepower	315.5	HP	175.9	HP	121.2	HP	82.6	HP	35.8	HP	29.8	HP
	Pump Efficiency	0.82		0.825		0.82		0.82		0.82		0.82	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg
	Thrust Support Hardware	118.9	kg	118.9	kg	118.9	kg	118.9	kg	118.9	kg	118.9	kg
	Engine Lines	283.3	kg	283.3	kg	283.3	kg	283.3	kg	283.3	kg	283.3	kg
	Main Valve	48.1	kg	48.1	kg	48.1	kg	48.1	kg	48.1	kg	48.1	kg
	Gimbal System	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	2464.4	kg		kg	2464.4	kg		kg	2370.0	kg		kg
	Throttling Factor Weight	194.9	kg		kg	194.9	kg		kg	194.9	kg	0.0	kg
	Margin (2%)	53.2	kg		kg	53.2	kg		kg	51.3	kg		kg
Total Engine	Weight	2712.5	kg		kg	2712.5	kg		kg	2712.5	kg		kg
System	Length	671.6	cm		cm	671.6	cm		cm	671.6	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

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	Rated Thrust (Vac)=	250000	lbf	250000	lbf
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2	
	Cycle Type	Expander		Expander	
	Area Ratio=	140		140	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	700.8		700.8	
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CH4		LO2/H2	
	Mixture Ratio	3.6		6	
	Chamber Diameter	75.8	cm	75.8	cm
	Chamber Length	66.0	cm	66.0	cm
	Chamber Temperature	3514	deg K	3456	deg K
	Chamber Pressure	700	psia	730	psia
	Inconel Injector weight	480.8	kg	480.8	kg
	Propellant Mass Flow	303.3	kg/s	248.4	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	600.3	kg	600.3	kg
	*Nozzle - Inconel, regen tubes	374.1	kg	374.1	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	226.2	kg	226.2	kg
	Area Ratio	140		140	
	Throat Diameter	37.9	cm	37.9	cm
	Exit Diameter	448.3	cm	448.3	cm
	Deployed Nozzle Length	491.2	cm	491.2	cm
	Delivered Vacuum Isp	373.8	sec	456.5	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

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Main Fuel Pump	Main Fuel Pump weight	7.3	kg	80.7	kg
	Material - Inconel				
	Number of Stages	1		2	
	Pressure Rise	1126.1	psia	1153.5	psia
	Pump Speed	32163	rpm	18851	rpm
	Pump Diameter	12.3	cm	34.3	cm
	Pump Horsepower	2028.1	HP	8085.4	HP
	Pump Efficiency	0.815		0.736	
Main Oxidizer Pump	Main Oxidizer Pump weight	9.7	kg	9.7	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	3076.7	psia	1910	psia
	Pump Speed	26790	rpm	21464	rpm
	Pump Diameter	14.1	cm	14.1	cm
	Pump Horsepower	6090.3	HP	3418.6	HP
	Pump Efficiency	0.832		0.832	
Fuel Turbine	Fuel Turbine weight	2.0	kg	10.0	kg
	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.44		1.3	
	Turbine Speed	32163	rpm	18851	rpm
	Turbine Efficiency	0.7		0.658	
	Turbine Diameter	6.7	cm	14.4	cm
Oxidizer Turbine	Oxidizer Turbine weight	2.9	kg	2.9	kg
	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.44		1.3	
	Turbine Speed	26790	rpm	21464	rpm
	Turbine Efficiency	0.7		0.486	
	Turbine Diameter	8.0	cm	8.0	cm

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Fuel Boost Pump	Fuel Boost Pump weight	9.7	kg	11.4	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	169.5	psia	175.7	psia
	Pump Speed	11523	rpm	30034	rpm
	Pump Diameter	12.5	cm	13.5	cm
	Pump Horsepower	113.9	HP	430.3	HP
	Pump Efficiency	0.741		0.706	
Oxidizer Boost	Fuel Boost Pump weight	24.7	kg	24.7	kg
Pump	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	283.8	psia	209.4	psia
	Pump Speed	4864	rpm	4096	rpm
	Pump Diameter	19.4	cm	19.4	cm
	Pump Horsepower	315.5	HP	175.9	HP
	Pump Efficiency	0.82		0.825	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg
	Thrust Support Hardware	118.9	kg	118.9	kg
	Engine Lines	188.9	kg	188.9	kg
	Main Valve	48.1	kg	48.1	kg
	Gimbal System	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

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ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	2267.8	kg	2351.1	kg
	Throttling Factor Weight	194.9	kg	194.9	kg
	Margin (2%)	49.3	kg	50.9	kg
Total Engine	Weight	2512.0	kg	2596.9	kg
System	Length	671.6	cm	671.6	cm
	Diameter	457.0	cm	457.0	cm

	Rated Thrust (Vac)=	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf	250000	lbf
	Percent Rated Thrust=	100%		100%		72.60%		72.60%		45.20%		45.20%	
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2		LO2/CH4/H2	
/	**Cycle Type**	Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CH4		LO2/H2		LO2/CH4		LO2/H2		LO2/CH4		LO2/H2	
	Mixture Ratio	4		6		4		6		4		6	
	Chamber Diameter	43.6	cm	43.6	cm	43.6	cm	43.6	cm	43.6	cm	43.6	cm
	Chamber Length	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm
	Chamber Temperature	3711	deg K	3636	deg K	3642	deg K	3575	deg K	3551	deg K	3496	deg K
	Chamber Pressure	2000	psia	2190	psia	1475	psia	1585	psia	920	psia	985	psia
	Inconel Injector weight	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg
	Propellant Mass Flow	317.6	kg/s	241.2	kg/s	211.5	kg/s	175.4	kg/s	133.5	kg/s	109.5	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	409.3	kg	409.3	kg	409.3	kg	409.3	kg	409.3	kg	409.3	kg
	*Nozzle - Inconel, regen tubes	123.3	kg	123.3	kg	123.3	kg	123.3	kg	123.3	kg	123.3	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	21.8	cm	21.8	cm	21.8	cm	21.8	cm	21.8	cm	21.8	cm
	Exit Diameter	435.6	cm	435.6	cm	435.6	cm	435.6	cm	435.6	cm	435.6	cm
	Deployed Nozzle Length	607.5	cm	607.5	cm	607.5	cm	607.5	cm	607.5	cm	607.5	cm
	Delivered Vacuum Isp	384.72	sec	463.03	sec	383.29	sec	462.31	sec	378.14	sec	460.82	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf	181500	lbf	181500	lbf	113000	lbf	113000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	18.7	kg	70.7	kg	18.7	kg	70.7	kg	18.7	kg	70.7	kg
	Material - Inconel												
	Number of Stages	1		3		1		3		1		3	
	Pressure Rise	3239.8	psia	3527.3	psia	2380.1	psia	2540.6	psia	1478.8	psia	1564.0	psia
	Pump Speed	30531	rpm	31722	rpm	26112	rpm	26574	rpm	20189	rpm	20453	rpm
	Pump Diameter	19.3	cm	29.1	cm	19.3	cm	29.1	cm	19.3	cm	29.1	cm
	Pump Horsepower	6303	HP	25257.1	HP	3393.8	HP	13536.7	HP	1366.1	HP	5383.6	HP

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	Pump Efficiency	0.725		0.735		0.708		0.718		0.684		0.694	
Main Oxidizer	Main Oxidizer Pump weight	11.7	kg	11.7	kg	11.7	kg	11.7	kg	11.7	kg	11.7	kg
Pump	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	4039.5	psia	4137.8	psia	2524.3	psia	2627.7	psia	1494.9	psia	1564.2	psia
	Pump Speed	26951	rpm	27458	rpm	21502	rpm	21713	rpm	16276	rpm	16458	rpm
	Pump Diameter	15.4	cm	15.4	cm	15.4	cm	15.4	cm	15.4	cm	15.4	cm
	Pump Horsepower	7826.2	HP	7349.4	HP	3598.1	HP	3430.4	HP	1378.9	HP	1310	HP
	Pump Efficiency	0.814		0.807		0.809		0.798		0.791		0.777	
Fuel Turbine	Fuel Turbine weight	22.0	kg	125.4	kg	22.0	kg	125.4	kg	22.0	kg	125.4	kg
	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	30531	rpm	31722	rpm	26112	rpm	26574	rpm	20189	rpm	20453	rpm
	Turbine Efficiency	0.7		0.648		0.705		0.706		0.687		0.684	
	Turbine Diameter	19.9	cm	46.3	cm	19.9	cm	46.3	cm	19.9	cm	46.3	cm
Oxidizer	Oxidizer Turbine weight	28.7	kg	28.7	kg	28.7	kg	28.7	kg	28.7	kg	28.7	kg
Turbine	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	26951	rpm	27458	rpm	21502	rpm	21713	rpm	16276	rpm	16458	rpm
	Turbine Efficiency	0.7		0.536		0.71		0.441		0.663		0.336	
	Turbine Diameter	22.6	cm	22.6	cm	22.6	cm	22.6	cm	22.6	cm	22.6	cm
Fuel Boost Pump	Fuel Boost Pump weight	12.2	kg	15.1	kg	12.2	kg	15.1	kg	12.2	kg	15.1	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	485.2	psia	530.3	psia	225.3	psia	318.4	psia	113.4	psia	149.3	psia
	Pump Speed	11755	rpm	29838	rpm	8979	rpm	23275	rpm	6246	rpm	16128	rpm
	Pump Diameter	13.9	cm	15.4	cm	13.9	cm	15.4	cm	13.9	cm	15.4	cm
	Pump Horsepower	294.85	HP	1176.45	HP	154.97	HP	617.4	HP	60.3	HP	237.3	HP
	Pump Efficiency	0.791		0.805		0.791		0.803		0.791		0.803	
Oxidizer Boost	Fuel Boost Pump weight	26.7	kg	26.7	kg	26.7	kg	26.7	kg	26.7	kg	26.7	kg
Pump	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	590.4	psia	511.7	psia	332.5	psia	291.6	psia	158.9	psia	139.7	psia
	Pump Speed	4971	rpm	4426	rpm	3515	rpm	3355	rpm	2431	rpm	2308	rpm

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ENGINE CONCEPT NO. 2 - MTV OPTIONS

	Pump Diameter	20.1	cm	20.1	cm	20.1	cm	20.1	cm	20.1	cm	20.1	cm
	Pump Horsepower	396.5	HP	368.96	HP	181.3	HP	170.4	HP	68	HP	65.3	HP
	Pump Efficiency	0.82		0.82		0.82		0.82		0.818		0.795	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg	35.4	kg
	Thrust Support Hardware	70.6	kg	70.6	kg	70.6	kg	70.6	kg	70.6	kg	70.6	kg
	Engine Lines	279.9	kg	279.9	kg	279.9	kg	279.9	kg	279.9	kg	279.9	kg
	Main Valve	30.6	kg	30.6	kg	30.6	kg	30.6	kg	30.6	kg	30.6	kg
	Gimbal System	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	221.4	kg	221.4	kg	221.4	kg	221.4	kg	221.4	kg	221.4	kg
	Gas Generator	26.5	kg	26.5	kg	26.5	kg	26.5	kg	26.5	kg	26.5	kg
	Gas Generator Features:												
	*Mixture Ratio	0.4		0.75		0.4		0.75		0.4		0.75	
	*Temperature	924.3	deg K	788.3	deg K	924.3	deg K	788.3	deg K	924.3	deg K	788.3	deg K
	*Pressure	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia
	*Mass Flow Rate	27.8	kg/s	11.2	kg/s	13.7	kg/s	5.9	kg/s	5.4	kg/s	2.3	kg/s
Subtotal	Engine Weight	2089.4	kg		kg	0.0	kg		kg	0.0	kg		kg
	Throttling Factor Weight	115.7	kg		kg	115.7	kg	0.0	kg	115.7	kg	0.0	kg
	Margin (2%)	44.1	kg		kg	0.0	kg		kg	0.0	kg		kg
Total Engine	Weight	2249.1	kg		kg	2249.1	kg		kg	2249.1	kg		kg
System	Length	787.8	cm		cm	787.8	cm		cm	787.8	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

	Rated Thrust (Vac)=	250000	lbf	250000	lbf
	Propellant Combination=	LO2/CH4/H2		LO2/CH4/H2	
	Cycle Type	Gas Generator		Gas Generator	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	403.2	kg	403.2	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CH4		LO2/H2	
	Mixture Ratio	4		6	
	Chamber Diameter	43.6	cm	43.6	cm
	Chamber Length	66.0	cm	66.0	cm
	Chamber Temperature	3711	deg K	3636	deg K
	Chamber Pressure	2000	psia	2190	psia
	Inconel Injector weight	242.9	kg	242.9	kg
	Propellant Mass Flow	317.6	kg/s	241.2	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	409.3	kg	409.3	kg
	*Nozzle - Inconel, regen tubes	123.3	kg	123.3	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	286.1	kg	286.1	kg
	Area Ratio	400		400	
	Throat Diameter	21.8	cm	21.8	cm
	Exit Diameter	435.6	cm	435.6	cm
	Deployed Nozzle Length	607.5	cm	607.5	cm
	Delivered Vacuum Isp	384.7	sec	463.0	sec
	Delivered Vacuum Thrust	250000	lbf	250000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	18.7	kg	70.7	kg
	Material - Inconel				
	Number of Stages	1		3	
	Pressure Rise	3239.8	psia	3527.3	psia
	Pump Speed	30531	rpm	31722	rpm
	Pump Diameter	19.3	cm	29.1	cm
	Pump Horsepower	6303	HP	25257.1	HP
	Pump Efficiency	0.725		0.735	
Main Oxidizer	Main Oxidizer Pump weight	11.7	kg	11.7	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	4039.5	psia	4137.8	psia
	Pump Speed	26951	rpm	27458	rpm
	Pump Diameter	15.4	cm	15.4	cm
	Pump Horsepower	7826.2	HP	7349.4	HP
	Pump Efficiency	0.814		0.807	
Fuel Turbine	Fuel Turbine weight	22.0	kg	125.4	kg
	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.877		9.877	
	Turbine Speed	30531	rpm	31722	rpm
	Turbine Efficiency	0.7		0.648	
	Turbine Diameter	19.9	cm	46.3	cm
Oxidizer	Oxidizer Turbine weight	28.7	kg	28.7	kg
Turbine	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.877		9.877	
	Turbine Speed	26951	rpm	27458	rpm
	Turbine Efficiency	0.7		0.536	
	Turbine Diameter	22.6	cm	22.6	cm

ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Fuel Boost Pump	Fuel Boost Pump weight	12.2	kg	15.1	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	485.2	psia	530.3	psia
	Pump Speed	11755	rpm	29838	rpm
	Pump Diameter	13.9	cm	15.4	cm
	Pump Horsepower	294.85	HP	1176.45	HP
	Pump Efficiency	0.791		0.805	
Oxidizer Boost Pump	Fuel Boost Pump weight	26.7	kg	26.7	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	590.4	psia	511.7	psia
	Pump Speed	4971	rpm	4426	rpm
	Pump Diameter	20.1	cm	20.1	cm
	Pump Horsepower	396.5	HP	368.96	HP
	Pump Efficiency	0.82		0.82	
Misc. Hardware	Thrust Mount	35.4	kg	35.4	kg
	Thrust Support Hardware	77.6	kg	77.6	kg
	Engine Lines	186.6	kg	186.6	kg
	Main Valve	30.6	kg	30.6	kg
	Gimbal System	32.8	kg	32.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	221.4	kg	221.4	kg
	Gas Generator	26.5	kg	26.5	kg
	Gas Generator Features:				
	*Mixture Ratio	0.4		0.75	
	*Temperature	924.3	deg K	788.3	deg K
	*Pressure	987.7	psia	987.7	psia
	*Mass Flow Rate	27.8	kg/s	11.2	kg/s

ENGINE CONCEPT NO. 2 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	1791.9	kg	1950.2	kg
	Throttling Factor Weight	116.4	kg	116.4	kg
	Margin (2%)	38.2	kg	41.3	kg
Total Engine	Weight	1946.5	kg	2107.9	kg
System	Length	787.8	cm	787.8	cm
	Diameter	457.0	cm	457.0	cm

BASELINE ENGINES – FEATURES AND DESCRIPTION

- **Concept No. 3**
  - Tri-Propellant Engine - $LO_2/CH_4/CO$
    - MTV Engine Candidate
  - Bi-Propellant Engines
    - LO_2/CH_4
    - LO_2/CO

    LEV and/or MEV Engine Candidates

Pel/02.61

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	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		58.30%		58.30%		16.60%		16.60%	
	Propellant Combination=	LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	699.2	kg	699.2	kg	699.2	kg	699.2	kg	699.2	kg	699.2	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CO		LO2/CH4		LO2/CO		LO2/CH4		LO2/CO		LO2/CH4	
	Mixture Ratio	0.55		3.6		0.55		3.6		0.55		3.6	
	Chamber Diameter	69.7	cm	69.7	cm	69.7	cm	69.7	cm	69.7	cm	69.7	cm
	Chamber Length	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3486	deg K	3303	deg K	3384	deg K	3168	deg K	3245	deg K
	Chamber Pressure	550	psia	555	psia	325	psia	327	psia	93	psia	93	psia
	Inconel Injector weight	313.3	kg	313.3	kg	313.3	kg	313.3	kg	313.3	kg	313.3	kg
	Propellant Mass Flow	270.7	kg/s	204.9	kg/s	159.3	kg/s	120.1	kg/s	46.9	kg/s	35.2	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg	1456.9	kg
	*Nozzle - Inconel, regen tubes	317.1	kg	317.1	kg	317.1	kg	317.1	kg	317.1	kg	317.1	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg	1139.8	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	34.8	cm	34.8	cm	34.8	cm	34.8	cm	34.8	cm	34.8	cm
	Exit Diameter	696.7	cm	696.7	cm	696.7	cm	696.7	cm	696.7	cm	696.7	cm
	Deployed Nozzle Length	971.3	cm	971.3	cm	971.3	cm	971.3	cm	971.3	cm	971.3	cm
	Delivered Vacuum Isp	293.22	sec	387.40	sec	290.39	sec	385.17	sec	280.47	sec	373.72	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	102000	lbf	102000	lbf	29000	lbf	29000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	14.7	kg	5.3	kg	14.7	kg	5.3	kg	14.7	kg	5.3	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	873.0	psia	890.4	psia	506.2	psia	518.6	psia	129.4	psia	138.7	psia
	Pump Speed	14670	rpm	33443	rpm	10970	rpm	24953	rpm	5299	rpm	12295	rpm
	Pump Diameter	17.2	cm	10.6	cm	17.2	cm	10.6	cm	17.2	cm	10.6	cm
	Pump Horsepower	1958.5	HP	1139.1	HP	669.8	HP	388.8	HP	53.5	HP	32.3	HP

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	Pump Efficiency	0.857		0.798		0.719		0.794		0.805		0.745	
Main Oxidizer	Main Oxidizer Pump weight	4.1	kg	4.1	kg	4.1	kg	4.1	kg	4.1	kg	4.1	kg
Pump	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	2380.5	psia	2001.5	psia	1826.3	psia	774.7	psia	217.7	psia	161.8	psia
	Pump Speed	35554	rpm	34514	rpm	30295	rpm	21369	rpm	10321	rpm	9408	rpm
	Pump Diameter	9.4	cm	9.4	cm	9.4	cm	9.4	cm	9.4	cm	9.4	cm
	Pump Horsepower	1985.9	HP	2818.8	HP	929.3	HP	640.7	HP	33.2	HP	39.3	HP
	Pump Efficiency	0.783		0.783		0.757		0.783		0.743		0.783	
Fuel Turbine	Fuel Turbine weight	6.8	kg	1.2	kg	6.8	kg	1.2	kg	6.8	kg	1.2	kg
	Material - Monel												
	Number of Stages	2		1		2		1		2		1	
	Pressure Ratio	2.7		1.9		2.09		1.31		1.13		1.06	
	Turbine Speed	14670	rpm	33443	rpm	10970	rpm	24953	rpm	5299	rpm	12295	rpm
	Turbine Efficiency	0.7		0.677		0.688		0.672		0.705		0.665	
	Turbine Diameter	11.4	cm	5.2	cm	11.4	cm	5.2	cm	11.4	cm	5.2	cm
Oxidizer	Oxidizer Turbine weight	2.0	kg	2.0	kg	2.0	kg	2.0	kg	2.0	kg	2.0	kg
Turbine	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.7		1.9		2.09		1.31		1.13		1.06	
	Turbine Speed	35554	rpm	34514	rpm	30295	rpm	21369	rpm	10321	rpm	9408	rpm
	Turbine Efficiency	0.7		0.708		0.623		0.709		0.589		0.658	
	Turbine Diameter	6.7	cm	6.7	cm	6.7	cm	6.7	cm	6.7	cm	6.7	cm
Fuel Boost Pump	Fuel Boost Pump weight	17.2	kg	6.3	kg	17.2	kg	6.3	kg	17.2	kg	6.3	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	132.5	psia	134.3	psia	58.3	psia	55.9	psia	9.2	psia	9.8	psia
	Pump Speed	6164	rpm	13895	rpm	5337	rpm	12113	rpm	2698	rpm	6263	rpm
	Pump Diameter	16.4	cm	10.1	cm	16.4	cm	10.1	cm	16.4	cm	10.1	cm
	Pump Horsepower	119.3	HP	69.1	HP	40.6	HP	23.5	HP	2.8	HP	1.6	HP
	Pump Efficiency	0.718		0.671		0.719		0.672		0.794		0.746	
Oxidizer Boost	Fuel Boost Pump weight	8.4	kg	8.4	kg	8.4	kg	8.4	kg	8.4	kg	8.4	kg
Pump	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	221.3	psia	390	psia	125.9	psia	147.3	psia	14.6	psia	18.4	psia
	Pump Speed	7723	rpm	10187	rpm	5656	rpm	6142	rpm	3997	rpm	3438	rpm

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	Pump Diameter	11.7	cm	11.7	cm	11.7	cm	11.7	cm	11.7	cm	11.7	cm
	Pump Horsepower	101.9	HP	159.7	HP	46.6	HP	35.5	HP	1.6	HP	2.2	HP
	Pump Efficiency	0.779		0.705		0.771		0.721		0.781		0.726	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	163.4	kg	163.4	kg	163.4	kg	163.4	kg	163.4	kg	163.4	kg
	Engine Lines	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg	120.7	kg
	Main Valve	44.4	kg	44.4	kg	44.4	kg	44.4	kg	44.4	kg	44.4	kg
	Gimbal System	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	2927.3	kg		kg	2887.0	kg		kg	2887.0	kg		kg
	Throttling Factor Weight	1499.3	kg		kg	1499.3	kg		kg	1499.3	kg		kg
	Margin (2%)	88.5	kg		kg	87.7	kg		kg	87.7	kg		kg
Total Engine	Weight	4515.1	kg		kg	4515.1	kg		kg	4515.1	kg		kg
System	Length	1164.3	cm		cm	1164.3	cm		cm	1164.3	cm		cm
	Diameter	696.7	cm		cm	696.7	cm		cm	696.7	cm		cm

ENGINE CONCEPT NO. 3 - LEV AND MEV OPTIONS

	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CO/CH4		LO2/CO/CH4	
	Cycle Type	Expander		Expander	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	699.2	kg	699.2	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CO		LO2/CH4	
	Mixture Ratio	0.55		3.6	
	Chamber Diameter	69.7	cm	69.7	cm
	Chamber Length	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3486	deg K
	Chamber Pressure	550	psia	555	psia
	Inconel Injector weight	313.3	kg	313.3	kg
	Propellant Mass Flow	270.7	kg/s	204.9	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	1456.9	kg	1456.9	kg
	*Nozzle - Inconel, regen tubes	317.1	kg	317.1	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	1139.8	kg	1139.8	kg
	Area Ratio	400		400	
	Throat Diameter	34.8	cm	34.8	cm
	Exit Diameter	696.7	cm	696.7	cm
	Deployed Nozzle Length	971.3	cm	971.3	cm
	Delivered Vacuum Isp	293.2	sec	387.4	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

ENGINE CONCEPT NO. 3 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	14.7	kg	5.3	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	873.0	psia	890.4	psia
	Pump Speed	14670	rpm	33443	rpm
	Pump Diameter	17.2	cm	10.6	cm
	Pump Horsepower	1958.5	HP	1139.1	HP
	Pump Efficiency	0.857		0.798	
Main Oxidizer Pump	Main Oxidizer Pump weight	4.1	kg	4.1	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	2380.5	psia	2001.5	psia
	Pump Speed	35554	rpm	34514	rpm
	Pump Diameter	9.4	cm	9.4	cm
	Pump Horsepower	1985.9	HP	2818.8	HP
	Pump Efficiency	0.783		0.783	
Fuel Turbine	Fuel Turbine weight	6.8	kg	1.2	kg
	Material - Monel				
	Number of Stages	2		1	
	Pressure Ratio	2.7		1.9	
	Turbine Speed	14670	rpm	33443	rpm
	Turbine Efficiency	0.7		0.677	
	Turbine Diameter	11.4	cm	5.2	cm
Oxidizer Turbine	Oxidizer Turbine weight	2.0	kg	2.0	kg
	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.7		1.9	
	Turbine Speed	35554	rpm	34514	rpm
	Turbine Efficiency	0.7		0.708	
	Turbine Diameter	6.7	cm	6.7	cm

Fuel Boost Pump	Fuel Boost Pump weight	17.2	kg	6.3	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	132.5	psia	134.3	psia
	Pump Speed	6164	rpm	13895	rpm
	Pump Diameter	16.4	cm	10.1	cm
	Pump Horsepower	119.3	HP	69.1	HP
	Pump Efficiency	0.718		0.671	
Oxidizer Boost	Fuel Boost Pump weight	8.4	kg	8.4	kg
Pump	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	221.3	psia	390	psia
	Pump Speed	7723	rpm	10187	rpm
	Pump Diameter	11.7	cm	11.7	cm
	Pump Horsepower	101.9	HP	159.7	HP
	Pump Efficiency	0.779		0.705	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	163.4	kg	163.4	kg
	Engine Lines	80.5	kg	80.5	kg
	Main Valve	44.4	kg	44.4	kg
	Gimbal System	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

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ENGINE CONCEPT NO. 3 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	2874.3	kg	2848.2	kg
	Throttling Factor Weight	1499.3	kg	1499.3	kg
	Margin (2%)	87.5	kg	86.9	kg
Total Engine	Weight	4461.0	kg	4434.4	kg
System	Length	1164.3	cm	1164.3	cm
	Diameter	696.7	cm	696.7	cm

	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		58.30%		58.30%		16.60%		16.60%	
	Propellant Combination=	LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4		LO2/CO/CH4	
	Cycle Type	Expander		Expander		Expander		Expander		Expander		Expander	
	Area Ratio=	165		165		165		165		165		165	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	720.5	kg	720.5	kg	720.5	kg	720.5	kg	720.5	kg	720.5	kg
	- includes Nozzle throat weight to area ratio 6; copper slotted regen construction												
	Propellant Type	LO2/CO		LO2/CH4		LO2/CO		LO2/CH4		LO2/CO		LO2/CH4	
	Mixture Ratio	0.55		3.6		0.55		3.6		0.55		3.6	
	Chamber Diameter	70.9	cm	70.9	cm	70.9	cm	70.9	cm	70.9	cm	70.9	cm
	Chamber Length	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3486	deg K	3303	deg K	3384	deg K	3168	deg K	3245	deg K
	Chamber Pressure	550	psia	555	psia	325	psia	327	psia	93	psia	93	psia
	Inconel Injector weight	328.9	kg	328.9	kg	328.9	kg	328.9	kg	328.9	kg	328.9	kg
	Propellant Mass Flow	280.2	kg/s	212.1	kg/s	164.9	kg/s	124.1	kg/s	48.4	kg/s	36.3	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	571.3	kg	571.3	kg	571.3	kg	571.3	kg	571.3	kg	571.3	kg
	*Nozzle - Inconel, regen tubes to area ratio 25	328.4	kg	328.4	kg	328.4	kg	328.4	kg	328.4	kg	328.4	kg
	*Nozzle Extension, Carbon-Carbo	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg	242.9	kg
	Area Ratio	165		165		165		165.0		165		165.0	
	Throat Diameter	35.4	cm	35.4	cm	35.4	cm	35.4	cm	35.4	cm	35.4	cm
	Exit Diameter	455.3	cm	455.3	cm	455.3	cm	455.3	cm	455.3	cm	455.3	cm
	Deployed Nozzle Length	513.6	cm	513.6	cm	513.6	cm	513.6	cm	513.6	cm	513.6	cm
	Delivered Vacuum Isp	283.23	sec	374.27	sec	280.58	sec	372.64	sec	271.66	sec	362.08	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	102000	lbf	102000	lbf	29000	lbf	29000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	15.3	kg	5.5	kg	15.3	kg	5.5	kg	15.3	kg	5.5	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	873.0	psia	890.4	psia	506.2	psia	518.6	psia	129.4	psia	138.7	psia
	Pump Speed	14420	rpm	32892	rpm	10797	rpm	24540	rpm	5214	rpm	12089	rpm
	Pump Diameter	17.5	cm	10.8	cm	17.5	cm	10.8	cm	17.5	cm	10.8	cm
	Pump Horsepower	2021.8	HP	1174.8	HP	691.3	HP	400.4	HP	55	HP	33.2	HP

	Pump Efficiency	0.859		0.8		0.856		0.796		0.807		0.747	
Main Oxidizer Pump	Main Oxidizer Pump weight	4.3	kg	4.3	kg	4.3	kg	4.3	kg	4.3	kg	4.3	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	2378.5	psia	2112.1	psia	1841.5	psia	778.4	psia	218	psia	162.12	psia
	Pump Speed	34911	rpm	33933	rpm	29838	rpm	21009	rpm	10132	rpm	9238	rpm
	Pump Diameter	9.6	cm	9.6	cm	9.6	cm	9.6	cm	9.6	cm	9.6	cm
	Pump Horsepower	2050.1	HP	2923.8	HP	968.4	HP	663.7	HP	34.3	HP	40.5	HP
	Pump Efficiency	0.785		0.785		0.759		0.785		0.745		0.785	
Fuel Turbine	Fuel Turbine weight	7.1	kg	1.2	kg	7.1	kg	1.2	kg	7.1	kg	1.2	kg
	Material - Monel												
	Number of Stages	2		1		2		1		2		1	
	Pressure Ratio	2.69		1.9		2.09		1.31		1.13		1.06	
	Turbine Speed	14420	rpm	32892	rpm	10797	rpm	24540	rpm	5214	rpm	12089	rpm
	Turbine Efficiency	0.7		0.677		0.687		0.708		0.707		0.664	
	Turbine Diameter	11.6	cm	5.3	cm	11.6	cm	5.3	cm	11.6	cm	5.3	cm
Oxidizer Turbine	Oxidizer Turbine weight	2.1	kg	2.1	kg	2.1	kg	2.1	kg	2.1	kg	2.1	kg
	Material - Monel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Ratio	2.69		1.9		2.09		1.31		1.13		1.06	
	Turbine Speed	34911	rpm	33933	rpm	29838	rpm	21009	rpm	10132	rpm	9238	rpm
	Turbine Efficiency	0.7		0.708		0.622		0.709		0.581		0.657	
	Turbine Diameter	6.8	cm	6.8	cm	6.8	cm	6.8	cm	6.8	cm	6.8	cm
Fuel Boost Pump	Fuel Boost Pump weight	17.9	kg	6.5	kg	17.9	kg	6.5	kg	17.9	kg	6.5	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	132.5	psia	134.3	psia	58.3	psia	55.9	psia	9.2	psia	9.8	psia
	Pump Speed	6059	rpm	13644	rpm	5248	rpm	11904	rpm	2653	rpm	6155	rpm
	Pump Diameter	16.7	cm	10.3	cm	16.7	cm	10.3	cm	16.7	cm	10.3	cm
	Pump Horsepower	123.11	HP	71.2	HP	42	HP	24.1	HP	2.9	HP	1.7	HP
	Pump Efficiency	0.72		0.674		0.72		0.674		0.796		0.749	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.8	kg	8.8	kg	8.8	kg	8.8	kg	8.8	kg	8.8	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	221.3	psia	390	psia	126.6	psia	147.3	psia	14.6	psia	18.4	psia
	Pump Speed	7588	rpm	9980	rpm	5220	rpm	6012	rpm	3915	rpm	3376	rpm

	Pump Diameter	11.9	cm	11.9	cm	11.9	cm	11.9	cm	11.9	cm	11.9	cm
	Pump Horsepower	105.2	HP	165.2	HP	48.5	HP	36.7	HP	1.7	HP	2.2	HP
	Pump Efficiency	0.781		0.709		0.773		0.726		0.784		0.729	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	108.3	kg	108.3	kg	108.3	kg	108.3	kg	108.3	kg	108.3	kg
	Engine Lines	126.9	kg	126.9	kg	126.9	kg	126.9	kg	126.9	kg	126.9	kg
	Main Valve	46.0	kg	46.0	kg	46.0	kg	46.0	kg	46.0	kg	46.0	kg
	Gimbal System	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg	0.0	kg
	Gas Generator Features:												
	*Mixture Ratio	0		0		0		0		0		0	
	*Temperature	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia	0	psia	0	psia	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s	0.0	kg/s
Subtotal	Engine Weight	2033.6	kg		kg	1991.3	kg		kg	1991.3	kg		kg
	Throttling Factor Weight	994.0	kg		kg	994.0	kg		kg	994.0	kg		kg
	Margin (2%)	60.6	kg		kg	59.7	kg		kg	59.7	kg		kg
Total Engine	Weight	3088.2	kg		kg	3088.2	kg		kg	3088.2	kg		kg
System	Length	706.6	cm		cm	706.6	cm		cm	706.6	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CO/CH4		LO2/CO/CH4	
	Cycle Type	Expander		Expander	
	Area Ratio=	165		165	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	720.5	kg	720.5	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CO		LO2/CH4	
	Mixture Ratio	0.55		3.6	
	Chamber Diameter	70.9	cm	70.9	cm
	Chamber Length	91.4	cm	91.4	cm
	Chamber Temperature	3403	deg K	3486	deg K
	Chamber Pressure	550	psia	555	psia
	Inconel Injector weight	328.9	kg	328.9	kg
	Propellant Mass Flow	280.2	kg/s	212.1	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	571.3	kg	571.3	kg
	*Nozzle - Inconel, regen tubes	328.4	kg	328.4	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbo	242.9	kg	242.9	kg
	Area Ratio	400		400	
	Throat Diameter	35.4	cm	35.4	cm
	Exit Diameter	455.3	cm	455.3	cm
	Deployed Nozzle Length	513.6	cm	513.6	cm
	Delivered Vacuum Isp	283.2	sec	374.3	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

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ENGINE CONCEPT NO. 3 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	15.3	kg	5.5	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	873.0	psia	890.4	psia
	Pump Speed	14420	rpm	32892	rpm
	Pump Diameter	17.5	cm	10.8	cm
	Pump Horsepower	2021.8	HP	1174.8	HP
	Pump Efficiency	0.859		0.8	
Main Oxidizer	Main Oxidizer Pump weight	4.3	kg	4.3	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	2378.5	psia	2112.1	psia
	Pump Speed	34911	rpm	33933	rpm
	Pump Diameter	9.6	cm	9.6	cm
	Pump Horsepower	2050.1	HP	2923.8	HP
	Pump Efficiency	0.785		0.785	
Fuel Turbine	Fuel Turbine weight	7.1	kg	1.2	kg
	Material - Monel				
	Number of Stages	2		1	
	Pressure Ratio	2.69		1.9	
	Turbine Speed	14420	rpm	32892	rpm
	Turbine Efficiency	0.7		0.677	
	Turbine Diameter	11.6	cm	5.3	cm
Oxidizer	Oxidizer Turbine weight	2.1	kg	2.1	kg
Turbine	Material - Monel				
	Number of Stages	1		1	
	Pressure Ratio	2.69		1.9	
	Turbine Speed	34911	rpm	33933	rpm
	Turbine Efficiency	0.7		0.708	
	Turbine Diameter	6.8	cm	6.8	cm

Fuel Boost Pump	Fuel Boost Pump weight	17.9	kg	6.5	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	132.5	psia	134.3	psia
	Pump Speed	6059	rpm	13644	rpm
	Pump Diameter	16.7	cm	10.3	cm
	Pump Horsepower	123.11	HP	71.2	HP
	Pump Efficiency	0.72		0.674	
Oxidizer Boost Pump	Fuel Boost Pump weight	8.8	kg	8.8	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	221.3	psia	390	psia
	Pump Speed	7588	rpm	9980	rpm
	Pump Diameter	11.9	cm	11.9	cm
	Pump Horsepower	105.2	HP	165.2	HP
	Pump Efficiency	0.781		0.709	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	108.3	kg	108.3	kg
	Engine Lines	84.6	kg	84.6	kg
	Main Valve	46.0	kg	46.0	kg
	Gimbal System	24.8	kg	24.8	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	0.0	kg	0.0	kg
	Gas Generator	0.0	kg	0.0	kg
	Gas Generator Features:				
	*Mixture Ratio	0		0	
	*Temperature	0.0	deg K	0.0	deg K
	*Pressure	0	psia	0	psia
	*Mass Flow Rate	0.0	kg/s	0.0	kg/s

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ENGINE CONCEPT NO. 3 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	1978.1	kg	1951.1	kg
	Throttling Factor Weight	994.0	kg	994.0	kg
	Margin (2%)	59.4	kg	58.9	kg
Total Engine	Weight	3031.5	kg	3004.0	kg
System	Length	706.6	cm	706.6	cm
	Diameter	457.0	cm	457.0	cm

	Rated Thrust (Vac)=	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf	175000	lbf
	Percent Rated Thrust =	100%		100%		58%		58%		16.60%		16.60%	
	Propellant Combination=	LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO		LO2/CH4/CO	
	Cycle Type	Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator		Gas Generator	
	Area Ratio=	400		400		400		400		400		400	
COMPONENT	**FEATURES**												
Chamber	Copper Chamber weight	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg	403.2	kg
	- includes Nozzle throat weight												
	to area ratio 6; copper slotted												
	regen construction												
	Propellant Type	LO2/CH4		LO2/CO		LO2/CH4		LO2/CO		LO2/CH4		LO2/CO	
	Mixture Ratio	4		0.55		4		0.55		4		0.55	
	Chamber Diameter	36.4	cm	36.4	cm	36.4	cm	36.4	cm	36.4	cm	36.4	cm
	Chamber Length	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm	66.0	cm
	Chamber Temperature	3711	deg K	3633	deg K	3597	deg K	3518	deg K	3391	deg K	3314	deg K
	Chamber Pressure	2000	psia	2035	psia	1185	psia	1190	psia	345	psia	345	psia
	Inconel Injector weight	145.5	kg	145.5	kg	145.5	kg	145.5	kg	145.5	kg	145.5	kg
	Propellant Mass Flow	221.8	kg/s	267.4	kg/s	120.0	kg/s	157.9	kg/s	35.1	kg/s	46.3	kg/s
	Coolant	LO2		LO2		LO2		LO2		LO2		LO2	
Nozzle	Nozzle Weight	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg	286.1	kg
	*Nozzle - Inconel, regen tubes	86.6	kg	86.6	kg	86.6	kg	86.6	kg	86.6	kg	86.6	kg
	to area ratio 25												
	*Nozzle Extension, Carbon-Carbo	199.6	kg	199.6	kg	199.6	kg	199.6	kg	199.6	kg	199.6	kg
	Area Ratio	400		400		400		400.0		400		400.0	
	Throat Diameter	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm	18.2	cm
	Exit Diameter	364.5	cm	364.5	cm	364.5	cm	364.5	cm	364.5	cm	364.5	cm
	Deployed Nozzle Length	508.3	cm	508.3	cm	508.3	cm	508.3	cm	508.3	cm	508.3	cm
	Delivered Vacuum Isp	383.34	sec	292.30	sec	379.78	sec	288.50	sec	368.78	sec	279.69	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf	102000	lbf	102000	lbf	29000	lbf	29000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2		LO2		LO2		LO2		LO2	
Main Fuel Pump	Main Fuel Pump weight	13.6	kg	9.1	kg	13.6	kg	9.1	kg	13.6	kg	9.1	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3240.0	psia	3287.5	psia	1909.2	psia	1909.7	psia	546.7	psia	537.5	psia
	Pump Speed	35484	rpm	35318	rpm	26928	rpm	26345	rpm	13676	rpm	13296	rpm
	Pump Diameter	16.6	cm	13.7	cm	16.6	cm	13.7	cm	16.6	cm	13.7	cm
	Pump Horsepower	4606	HP	7614.6	HP	1624.4	HP	2672.3	HP	148.3	HP	237.1	HP

	Pump Efficiency	0.716		0.828		0.688		0.809		0.621		0.751	
Main Oxidizer Pump	Main Oxidizer Pump weight	6.1	kg	6.1	kg	6.1	kg	6.1	kg	6.1	kg	6.1	kg
	Material - Inconel												
	Number of Stages	1		1		1		1		1		1	
	Pressure Rise	3341.3	psia	3147.3	psia	1885.7	psia	1802.9	psia	506.7	psia	497.9	psia
	Pump Speed	34760	rpm	32394	rpm	25678	rpm	24017	rpm	12693	rpm	12021	rpm
	Pump Diameter	11.4	cm	11.4	cm	11.4	cm	11.4	cm	11.4	cm	11.4	cm
	Pump Horsepower	4561	HP	2604.4	HP	1552.4	HP	911.1	HP	131.5	HP	80.7	HP
	Pump Efficiency	0.805		0.765		0.789		0.741		0.738		0.68	
Fuel Turbine	Fuel Turbine weight	16.1	kg	14.3	kg	16.1	kg	14.3	kg	16.1	kg	14.3	kg
	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	35484	rpm	35318	rpm	26928	rpm	26345	rpm	13676	rpm	13296	rpm
	Turbine Efficiency	0.7		0.655		0.708		0.707		0.493		0.485	
	Turbine Diameter	17.1	cm	16.2	cm	17.1	cm	16.2	cm	17.1	cm	16.2	cm
Oxidizer Turbine	Oxidizer Turbine weight	16.8	kg	16.8	kg	16.8	kg	16.8	kg	16.8	kg	16.8	kg
	Material - Monel												
	Number of Stages	2		2		2		2		2		2	
	Pressure Ratio	9.877		9.877		9.877		9.877		9.877		9.877	
	Turbine Speed	34760	rpm	32394	rpm	25678	rpm	24017	rpm	12693	rpm	12021	rpm
	Turbine Efficiency	0.7		0.661		0.706		0.705		0.471		0.474	
	Turbine Diameter	17.5	cm	17.5	cm	17.5	cm	17.5	cm	17.5	cm	17.5	cm
Fuel Boost Pump	Fuel Boost Pump weight	8.6	kg	24.4	kg	8.6	kg	24.4	kg	8.6	kg	24.4	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	485.2	psia	493.2	psia	169.3	psia	204.4	psia	33.8	psia	36.7	psia
	Pump Speed	13838	rpm	5919	rpm	8917	rpm	3924	rpm	10694	rpm	4667	rpm
	Pump Diameter	11.8	cm	19.3	cm	11.8	cm	19.3	cm	11.8	cm	19.3	cm
	Pump Horsepower	218	HP	392.6	HP	73.9	HP	134.5	HP	6	HP	11	HP
	Pump Efficiency	0.772		0.82		0.772		0.82		0.777		0.826	
Oxidizer Boost Pump	Fuel Boost Pump weight	16.7	kg	16.7	kg	16.7	kg	16.7	kg	16.7	kg	16.7	kg
	Material - Inconel												
	Centrifugal Pump												
	Pressure Rise	558	psia	284.7	psia	237.4	psia	135.8	psia	36.9	psia	28.1	psia
	Pump Speed	5953	rpm	4254	rpm	3840	rpm	2817	rpm	4406	rpm	4367	rpm

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	Pump Diameter	16.1	cm	16.1	cm	16.1	cm	16.1	cm	16.1	cm	16.1	cm
	Pump Horsepower	231.8	HP	127.6	HP	77.3	HP	45.8	HP	6.1	HP	3.5	HP
	Pump Efficiency	0.809		0.796		0.809		0.753		0.816		0.809	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg	32.8	kg
	Thrust Support Hardware	55.6	kg	55.6	kg	55.6	kg	55.6	kg	55.6	kg	55.6	kg
	Engine Lines	147.3	kg	147.3	kg	147.3	kg	147.3	kg	147.3	kg	147.3	kg
	Main Valve	21.4	kg	21.4	kg	21.4	kg	21.4	kg	21.4	kg	21.4	kg
	Gimbal System	30.5	kg	30.5	kg	30.5	kg	30.5	kg	30.5	kg	30.5	kg
	TPA Ignition	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg	5.6	kg
	Hot Gas Manifolding	121.9	kg	121.9	kg	121.9	kg	121.9	kg	121.9	kg	121.9	kg
	Gas Generator	15.7	kg	15.7	kg	15.7	kg	15.7	kg	15.7	kg	15.7	kg
	Gas Generator Features:												
	*Mixture Ratio	0.4		0.05		0.4		0.05		0.4		0.05	
	*Temperature	924.3	deg K	562.8	deg K	924.3	deg K	562.8	deg K	924.3	deg K	562.8	deg K
	*Pressure	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia	987.7	psia
	*Mass Flow Rate	18.0	kg/s	22.7	kg/s	6.2	kg/s	7.9	kg/s	0.5	kg/s	0.7	kg/s
Subtotal	Engine Weight	1391.3	kg		kg	1391.3	kg		kg	1391.3	kg		kg
	Throttling Factor Weight	510.6	kg		kg	510.6	kg		kg	510.6	kg		kg
	Margin (2%)	38.0	kg		kg	38.0	kg		kg	38.0	kg		kg
Total Engine	Weight	1940.0	kg		kg	1940.0	kg		kg	1940.0	kg		kg
System	Length	688.6	cm		cm	688.6	cm		cm	688.6	cm		cm
	Diameter	457.0	cm		cm	457.0	cm		cm	457.0	cm		cm

ENGINE CONCEPT NO.3 - LEV AND MEV OPTIONS

	Rated Thrust (Vac)=	175000	lbf	175000	lbf
	Propellant Combination=	LO2/CH4/CO		LO2/CH4/CO	
	Cycle Type	Gas Generator		Gas Generator	
	Area Ratio=	400		400	
COMPONENT	**FEATURES**				
Chamber	Copper Chamber weight	403.2	kg	403.2	kg
	- includes Nozzle throat weight				
	to area ratio 6; copper slotted				
	regen construction				
	Propellant Type	LO2/CH4		LO2/CO	
	Mixture Ratio	4		6	
	Chamber Diameter	36.4	cm	36.4	cm
	Chamber Length	66.0	cm	66.0	cm
	Chamber Temperature	3711	deg K	3633	deg K
	Chamber Pressure	2000	psia	2035	psia
	Inconel Injector weight	145.5	kg	145.5	kg
	Propellant Mass Flow	221.8	kg/s	267.4	kg/s
	Coolant	LO2		LO2	
Nozzle	Nozzle Weight	286.1	kg	286.1	kg
	*Nozzle - Inconel, regen tubes	86.6	kg	86.6	kg
	to area ratio 25				
	*Nozzle Extension, Carbon-Carbon	199.6	kg	199.6	kg
	Area Ratio	400		400	
	Throat Diameter	18.2	cm	18.2	cm
	Exit Diameter	364.5	cm	364.5	cm
	Deployed Nozzle Length	508.3	cm	508.3	cm
	Delivered Vacuum Isp	383.3	sec	292.3	sec
	Delivered Vacuum Thrust	175000	lbf	175000	lbf
	Coolant (area ratio = 6 to 25)	LO2		LO2	

ENGINE CONCEPT NO.3 - LEV AND MEV OPTIONS

Main Fuel Pump	Main Fuel Pump weight	13.6	kg	9.1	kg
	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	3240.0	psia	3287.5	psia
	Pump Speed	35484	rpm	35318	rpm
	Pump Diameter	16.6	cm	13.7	cm
	Pump Horsepower	4606	HP	7614.6	HP
	Pump Efficiency	0.716		0.828	
Main Oxidizer	Main Oxidizer Pump weight	6.1	kg	6.1	kg
Pump	Material - Inconel				
	Number of Stages	1		1	
	Pressure Rise	3341.3	psia	3147.3	psia
	Pump Speed	34760	rpm	32394	rpm
	Pump Diameter	11.4	cm	11.4	cm
	Pump Horsepower	4561	HP	2604.4	HP
	Pump Efficiency	0.805		0.765	
Fuel Turbine	Fuel Turbine weight	16.1	kg	14.3	kg
	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.877		9.877	
	Turbine Speed	35484	rpm	35318	rpm
	Turbine Efficiency	0.7		0.655	
	Turbine Diameter	17.1	cm	16.2	cm
Oxidizer	Oxidizer Turbine weight	16.8	kg	16.8	kg
Turbine	Material - Monel				
	Number of Stages	2		2	
	Pressure Ratio	9.877		9.877	
	Turbine Speed	34760	rpm	32394	rpm
	Turbine Efficiency	0.7		0.661	
	Turbine Diameter	17.5	cm	17.5	cm

ENGINE CONCEPT NO.3 - LEV AND MEV OPTIONS

Fuel Boost Pump	Fuel Boost Pump weight	8.6	kg	24.4	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	485.2	psia	493.2	psia
	Pump Speed	13838	rpm	5919	rpm
	Pump Diameter	11.8	cm	19.3	cm
	Pump Horsepower	218	HP	392.6	HP
	Pump Efficiency	0.772		0.82	
Oxidizer Boost Pump	Fuel Boost Pump weight	16.7	kg	16.7	kg
	Material - Inconel				
	Centrifugal Pump				
	Pressure Rise	558	psia	284.7	psia
	Pump Speed	5953	rpm	4254	rpm
	Pump Diameter	16.1	cm	16.1	cm
	Pump Horsepower	231.8	HP	127.6	HP
	Pump Efficiency	0.809		0.796	
Misc. Hardware	Thrust Mount	32.8	kg	32.8	kg
	Thrust Support Hardware	55.6	kg	55.6	kg
	Engine Lines	98.2	kg	98.2	kg
	Main Valve	21.4	kg	21.4	kg
	Gimbal System	30.5	kg	30.5	kg
	TPA Ignition	5.6	kg	5.6	kg
	Hot Gas Manifolding	121.9	kg	121.9	kg
	Gas Generator	15.7	kg	15.7	kg
	Gas Generator Features:				
	*Mixture Ratio	0.4		0.05	
	*Temperature	924.3	deg K	562.8	deg I
	*Pressure	987.7	psia	987.7	psia
	*Mass Flow Rate	18.0	kg/s	22.7	kg/s

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ENGINE CONCEPT NO.3 - LEV AND MEV OPTIONS

Subtotal	Engine Weight	1294.4	kg	1303.9	kg
	Throttling Factor Weight	510.6	kg	510.6	kg
	Margin (2%)	36.1	kg	36.3	kg
Total Engine	Weight	1841.1	kg	1850.8	kg
System	Length	688.6	cm	688.6	cm
	Diameter	457.0	cm	457.0	cm

BASELINE ENGINES – OPERATIONAL CHARACTERISTICS
– Comments –

- **Pump and/or Turbine Efficiencies < 0.45 Are Considered Marginal**
  - Condition Is Present for Some Engines at Their Low Thrust Operating Mode Condition

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BASELINE ENGINES – OPERATIONAL CHARACTERISTICS

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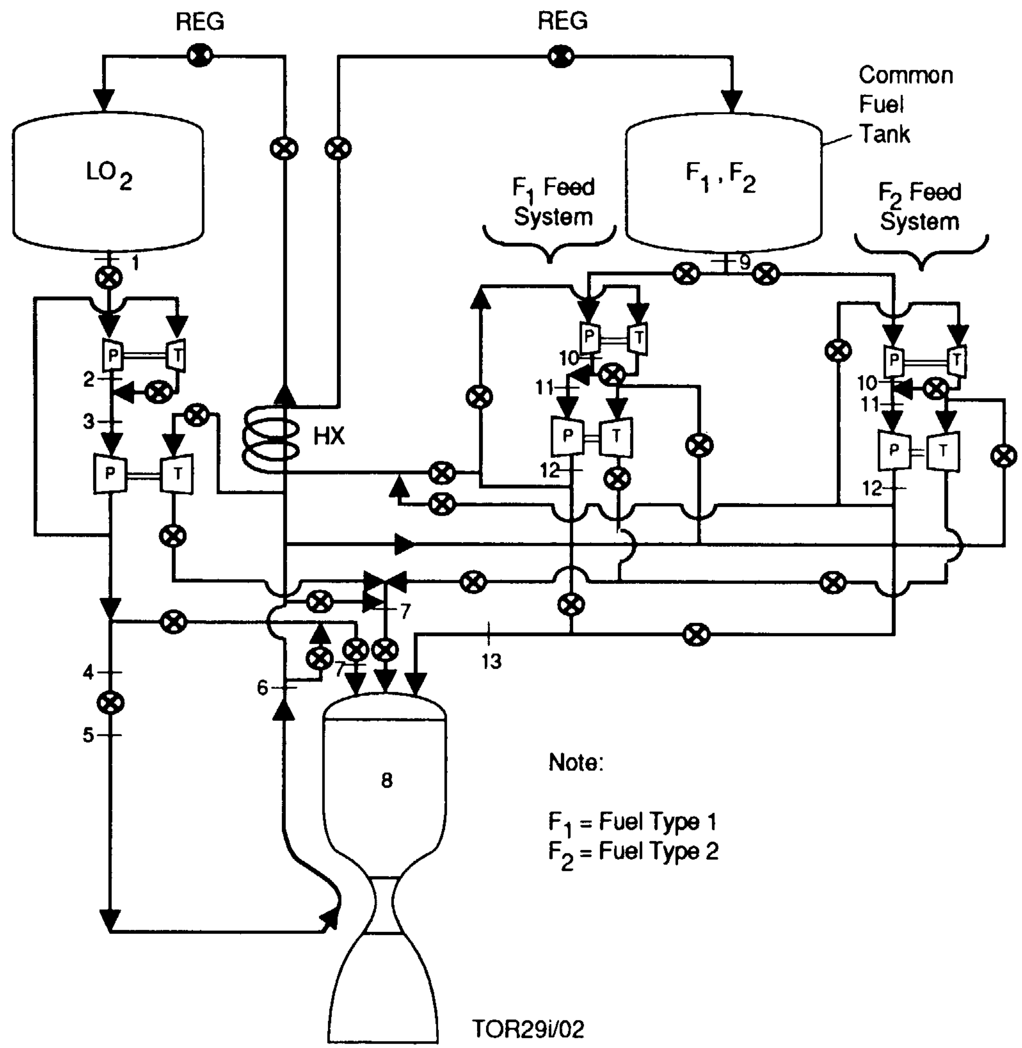
EXPANDER CYCLE ENGINE FLOW STATION LOCATIONS



Note:

F_1 = Fuel Type 1
F_2 = Fuel Type 2

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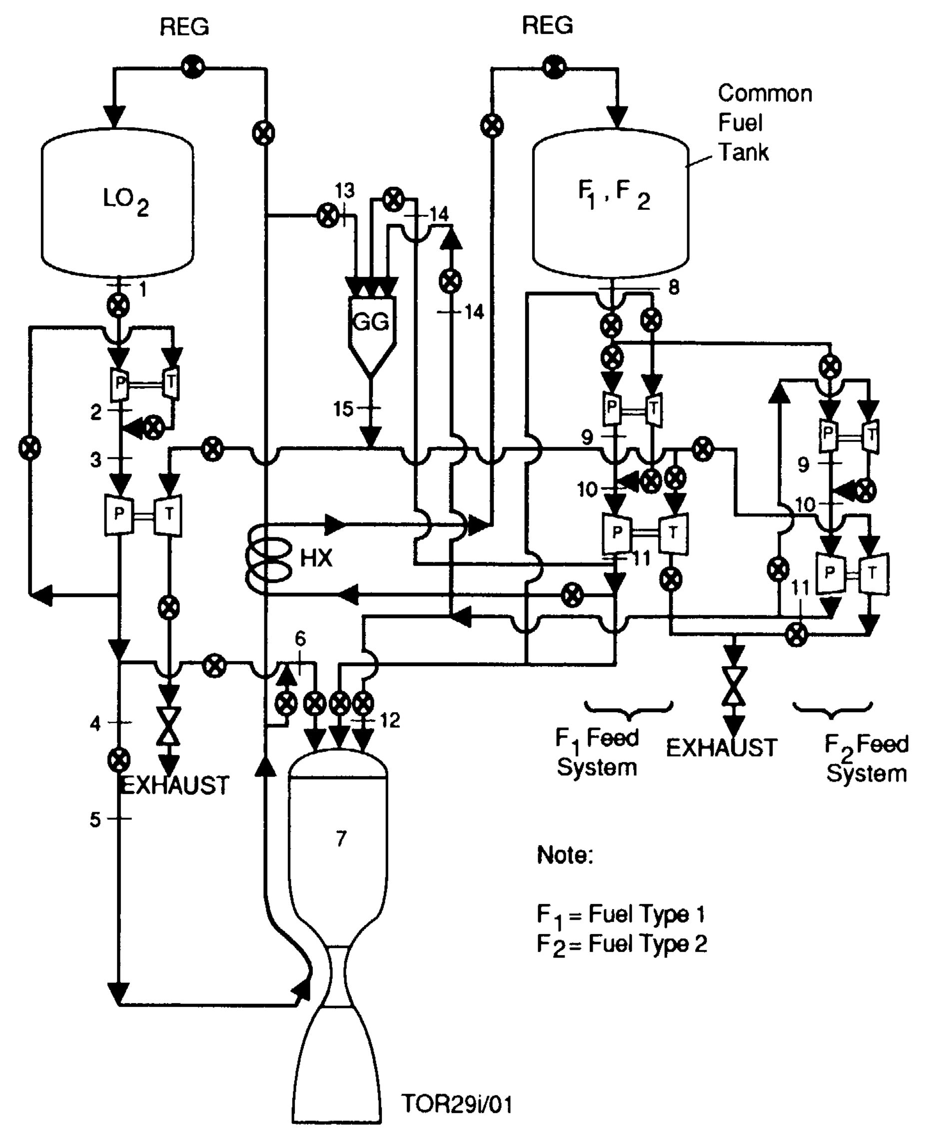
GAS GENERATOR CYCLE ENGINE FLOW STATION LOCATIONS



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TYPICAL ENGINE OPERATING CONDITIONS*

EXPANDER CYCLE**

STATION No.	No. 1 - LO2/CO/H2						No. 2 - LO2/CH4/H2						No. 3 - LO2/CO/CH4					
	LO2/CO			LO2/H2			LO2/CH4			LO2/H2			LO2/CO			LO2/CH4		
	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)
1	22.8	88.9	93.6	22.8	88.9	142.8	22.8	88.9	224.3	22.8	88.9	204.4	22.8	88.9	93.6	22.8	88.9	156.7
2	244.1	89.8	93.6	330.7	90.3	142.8	304.9	89.9	224.3	232.9	89.7	204.4	244.1	89.8	93.6	412.8	90.6	156.7
3	127.9	89.8	93.6	73.9	90.3	142.8	157.0	89.9	224.3	105.4	89.7	204.4	127.9	89.8	93.6	101.2	90.6	156.7
4	2508.4	99.2	93.6	1432.2	95.7	142.8	3196.8	101.4	224.3	2012.2	96.8	204.4	2508.4	99.2	93.6	2102.7	98.6	156.7
5	2351.2	99.2	93.6	1265.0	95.7	142.8	2996.7	101.4	224.3	1802.2	96.8	204.4	2351.2	99.2	93.6	1944.1	98.6	156.7
6	2270.8	126.3	93.6	1083.0	122.9	142.8	2599.2	115.7	224.3	1465.0	118.7	204.4	2270.8	126.3	93.6	1731.6	116.6	156.7
7	701.7	106.3	92.2	746.3	116.6	142.0	893.0	99.1	222.8	937.7	110.2	203.1	701.7	106.3	92.2	708.1	103.0	155.6
8	550.0	3402.7	270.7	585.0	3431.8	168.9	700.0	3514.2	290.8	735.0	3456.7	240.0	550.0	3402.7	270.7	555.0	3485.6	204.9
9	22.3	80.6	180.7	35.0	22.2	27.4	12.5	94.4	68.4	35.0	22.2	37.6	22.3	80.6	180.7	12.5	94.4	49.6
10	154.8	81.3	180.7	175.5	24.2	27.4	182.0	95.5	68.4	211.9	24.5	42.1	154.8	81.3	180.7	146.8	95.4	49.6
11	58.3	81.3	180.7	72.8	24.2	27.4	59.1	95.5	68.4	82.9	24.5	37.6	58.3	81.3	180.7	49.3	95.4	49.6
12	931.3	85.1	180.7	990.5	37.4	27.4	1185.2	101.9	68.4	1244.5	41.4	37.6	931.3	85.1	180.7	939.7	100.6	49.6
13	701.7	85.1	178.5	746.3	37.4	26.9	893.0	101.9	68.0	937.7	41.4	37.0	701.7	85.1	178.5	708.1	100.7	49.3

GAS GENERATOR CYCLE

STATION No.	No. 1 - LO2/CO/H2						No. 2 - LO2/CH4/H2						No. 3 - LO2/CO/CH4					
	LO2/CO			LO2/H2			LO2/CH4			LO2/H2			LO2/CO			LO2/CH4		
	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)	Pressure (psia)	Temperature (deg K)	Mass Flow (kg/s)
1	22.8	88.9	90.5	22.8	88.9	142.0	22.8	88.9	231.9	22.8	88.9	203.4	22.8	88.9	161.3	22.8	88.9	90.7
2	569.8	91.1	90.5	1153.9	93.4	142.0	613.2	91.1	231.9	534.5	90.8	203.4	580.8	91.1	161.3	307.5	90.0	90.7
3	147.2	91.1	90.5	143.5	93.4	142.0	190.7	91.1	231.9	199.3	90.8	203.4	158.3	91.1	161.3	162.1	90.0	90.7
4	3251.0	103.6	87.9	3754.2	108.1	141.2	4230.2	106.7	222.4	4337.1	106.9	202.2	3499.6	104.0	155.1	3309.4	102.8	89.3
5	2679.4	103.6	87.9	3125.5	108.1	141.2	3658.6	106.7	222.4	3711.3	106.9	202.2	2928.0	104.0	155.1	2727.8	102.8	89.3
6	2551.5	121.3	87.9	2806.7	124.1	141.2	2551.5	115.9	222.4	2793.9	119.2	202.2	2551.5	113.9	155.1	2596.2	117.1	89.3
7	2000.0	3629.3	269.9	2200.0	3637.2	171.0	2000.0	3711.1	290.0	2190.0	3636.4	241.2	2000.0	3711.1	203.8	2035.0	3633.3	267.4
8	22.3	80.6	204.0	35.0	22.2	30.3	12.5	94.4	87.3	35.0	22.2	39.6	12.5	94.4	61.6	22.3	80.6	180.4
9	507.0	82.8	204.0	567.7	28.4	30.3	497.7	97.3	87.3	565.3	28.3	39.6	497.7	97.4	61.6	515.5	82.8	180.4
10	155.9	82.8	204.0	181.5	28.4	30.3	146.6	97.3	87.3	180.8	28.3	39.6	146.6	97.4	61.6	158.2	82.8	180.4
11	3386.4	97.7	182.0	3725.0	54.9	29.8	3386.4	118.1	67.6	3708.1	54.6	39.0	3386.4	118.4	48.8	3445.7	98.0	178.1
12	2551.5	97.7	182.0	2806.7	54.9	29.8	2551.5	118.3	67.6	2793.9	54.6	39.0	2551.5	118.7	48.8	2596.2	98.0	178.1
13	1137.8	103.6	1.1	948.2	108.1	3.6	1480.6	106.7	7.9	987.7	106.9	4.8	1224.8	104.0	5.1	987.7	102.8	1.1
14	1185.2	97.7	22.0	948.2	54.9	4.9	1185.2	118.1	19.8	987.7	54.6	6.4	1185.2	118.4	12.9	987.7	98.0	21.6
15	948.2	562.8	23.1	948.2	788.3	8.5	987.7	924.3	27.8	987.7	788.3	11.2	987.7	924.3	18.0	987.7	562.8	22.7

*At 100% Rated Thrust Conditions

**Area ratio = 400:1

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BASELINE ENGINES – THRUST CHAMBER/ COOLANT CHARACTERISTICS

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TYPICAL EXPANDER CYCLE ENGINE THRUST CHAMBER/COOLANT CHARACTERISTICS
– At 100% Rated Thrust Conditions, LO_2 Cooled –

ENGINE CONCEPT	1 -LO2/CO/H2		2-LO2/CH4/H2		3-LO2/CO/CH4	
OPERATION MODE	LO2/CO	LO2/H2	LO2/CH4	LO2/H2	LO2/CO	LO2/CH4
Regen Inlet:						
Wall Temp. (deg K)	185.0	153.9	154.4	146.1	185.0	166.7
Coolant Temp. (deg K)	100.6	96.1	101.7	96.7	100.6	96.1
Throat:						
Wall Temp. (deg K)	208.9	236.1	162.2	202.8	208.9	181.1
Coolant Temp. (deg K)	116.1	111.7	111.1	110.6	116.1	106.7
Regen Outlet:						
Wall Temp. (deg K)	355.0	447.8	270.0	402.2	355.0	311.7
Coolant Temp. (deg K)	127.2	123.3	116.1	118.9	127.2	114.4
Total Regen Delta-T (deg K)	27.1	27.2	14.3	21.8	27.1	18.0
Total Regen Delta-P (psid)	-80.4	-182.0	-397.5	-337.2	-80.4	-212.5
Cooling Channels:						
Channel width (cm)	0.15	0.15	0.15	0.15	0.15	0.15
Land width (cm)	0.15	0.15	0.15	0.15	0.15	0.15
Channel height (cm)	0.76	0.76	0.76	0.76	0.76	0.76

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TYPICAL GAS GENERATOR CYCLE ENGINE THRUST CHAMBER/COOLANT CHARACTERISTICS
– At 100% Rated Thrust Conditions, LO_2 Cooled –

ENGINE CONCEPT	1-LO2/CO/H2		2-LO2/CH4/H2		3-LO2/CO/CH4	
OPERATON MODE	LO2/CO	LO2/H2	LO2/CH4	LO2/H2	LO2/CO	LO2/CH4
Regen Inlet:						
Wall Temp. (deg K)	214.4	170.0	173.9	161.7	185.6	208.9
Coolant Temp. (deg K)	103.9	107.8	104.4	105.0	103.9	102.8
Throat:						
Wall Temp. (deg K)	172.8	161.7	145.6	152.2	152.2	167.8
Coolant Temp. (deg K)	110.6	113.9	109.4	111.1	108.9	109.4
Regen Outlet:						
Wall Temp. (deg K)	350.6	353.9	270.0	327.2	295.6	340.0
Coolant Temp. (deg K)	121.1	123.9	113.9	117.2	113.9	117.2
Total Regen Delta-T (deg K)	17.7	16.0	9.3	12.4	9.9	14.3
Total Regen Delta-P (psid)	-127.9	-318.8	-1107.1	-917.3	-376.5	-131.6
Cooling Channels:						
Channel width (cm)	0.15	0.15	0.15	0.15	0.15	0.15
Land width (cm)	0.15	0.15	0.15	0.15	0.15	0.15
Channel height (cm)	0.76	0.76	0.76	0.76	0.76	0.76

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CHAMBER/INJECTOR DESIGN COMPATIBLITY
- Baseline Engines -

INJECTOR DESIGN CHARACTERISTICS

- Type: Co-Axial Design
- Density: 64.5 Elements/cm^2
- Material: Inconel
- Injector Orifice Dimensions:

CYCLE	ENGINE CONCEPT NO.	OXIDIZER ORIFICE DIAMETER (cm)	FUEL ORIFICE DIAMETER (cm)
Expander	1 - $LO_2/CO/H_2$	0.074	0.109
	2 - $LO_2/CH_4/H_2$	0.102	0.071
	3 - $LO_2/CH_4/CO$	0.074	0.109
Gas Generator	1 - $LO_2/CO/H_2$	0.102	0.152
	2 - $LO_2/CH_4/H_2$	0.135	0.091
	3 - $LO_2/CH_4/CO$	0.132	0.094

CHAMBER COOLING CHARACTERISTICS
- Film Cooling Effects -

CYCLE	ENGINE CONCEPT NO.	PRO-PELLANTS	BARRIER MIXTURE RATIO	BARRIER TEMP. (°K)	FUEL FILM COOLING FRACTION	CORE TEMP. (°K)
Expander	1 - $LO_2/CO/H_2$	LO_2/CO	0.03	696.1	0.06	3402.8
		LO_2/H_2	0.73	768.9	0.12	3431.7
	2 - $LO_2/CH_4/H_2$	LO_2/CH_4	0.28	773.3	0.09	3514.4
		LO_2/H_2	0.73	768.3	0.08	3456.7
	3 - $LO_2/CO/CH_4$	LO_2/CO	0.03	696.1	0.06	3402.8
		LO_2/CH_4	0.31	786.7	0.12	3485.6
Gas Generator	1 - $LO_2/CO/H_2$	LO_2/CO	0.02	703.9	0.12	3629.4
		LO_2/H_2	0.75	786.1	0.21	3637.2
	2 - $LO_2/CH_4/H_2$	LO_2/CH_4	0.17	779.4	0.17	3711.1
		LO_2/H_2	0.75	786.1	0.14	3636.1
	3 - $LO_2/CH_4/CO$	LO_2/CH_4	0.17	779.4	0.20	3711.1
		LO_2/CO	0.01	691.1	0.09	3633.3

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APPENDIX E

FINAL MISSION PROFILE/REQUIREMENTS DATA

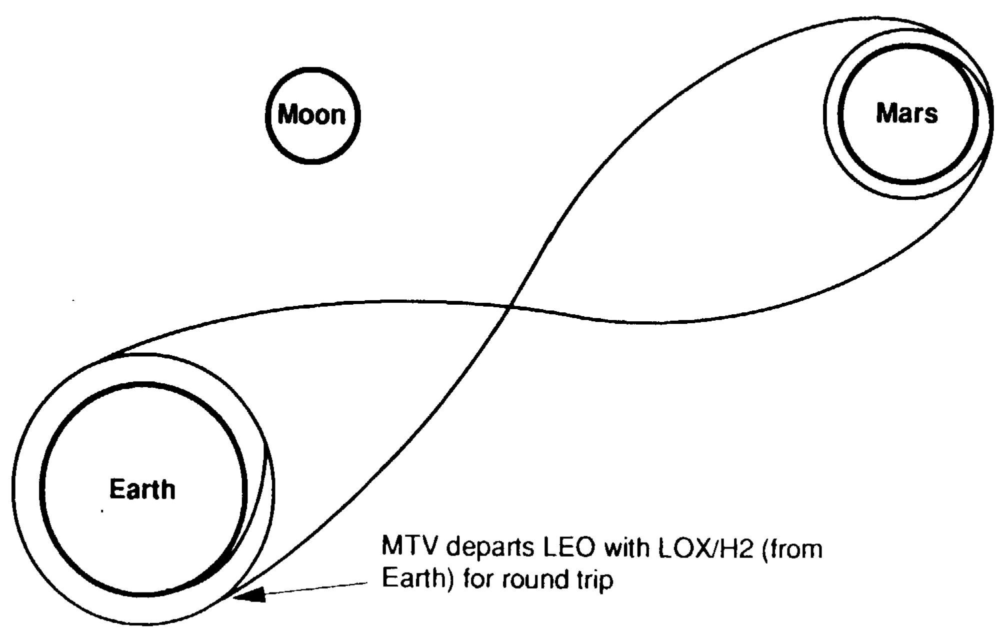
APPENDIX E
FINAL MISSION PROFILE/REQUIREMENTS DATA

Detailed mission profile and requirements data is presented in this appendix for the mission scenarios examined in Section 5.0. This data is based on engine systems engineering data which is presented and discussed in Section 4.0.

BASELINE CASE (NO LUNAR/MARS PROPELLANT): ALL EARTH LOX/H_2

Scenario 1A: 250 KLB EXPANDER CYCLE ENGINE

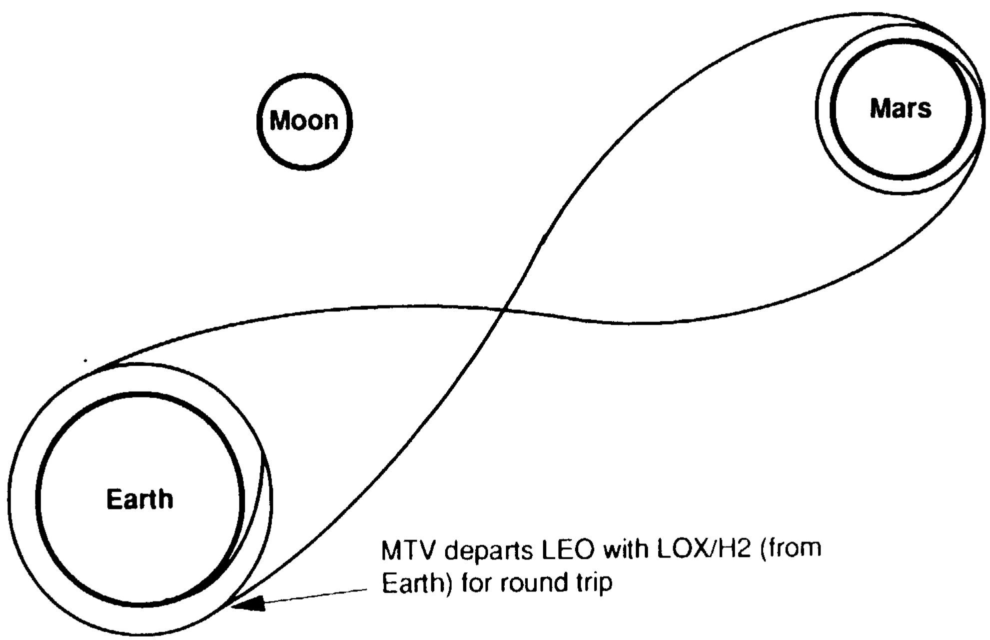
					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI											
LOI											
Lunar ascent											
Lunar descent											
TMI	Exp. Stg.	LEO	1682	1009	3982	127	4109	4x3915	472.3	1000	1030
MOI	MTV	LMO	611	277	2590	92	2682	3915	472.3	250	1119
Mars ascent	MEV	Mars Surf.	56	39	5300	5	5305	3807	472.3	250	159
Mars desent	MEV	LMO	99	19	930	0	930	3807	472.3	250	75
TEI	MTV	LMO	211	92	2521	10	2531	3915	472.3	250	370
EOI	MTV	LEO	113	69	4081	10	4091	3915	472.3	250	276



BASELINE CASE (NO LUNAR/MARS PROPELLANT): ALL EARTH LOX/H_2

Scenario 1B: 250 KLB GAS GENERATOR CYCLE ENGINE

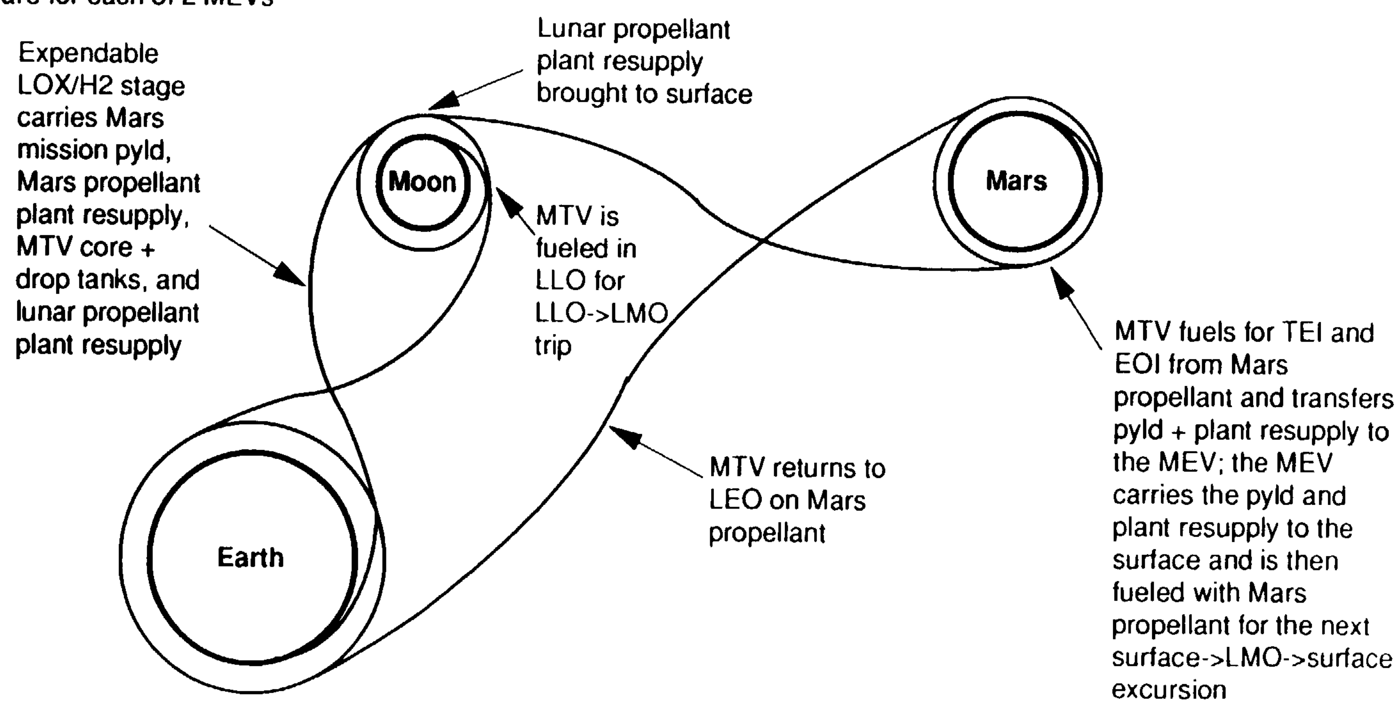
					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI											
LOI											
Lunar ascent											
Lunar descent											
TMI	Exp. Stg.	LEO	1676	1018	3982	125	4107	4x2249	463	1000	1019
MOI	MTV	LMO	603	277	2590	89	2679	2249	463	250	1098
Mars ascent	MEV	Mars Surf.	51	36	5300	4	5304	2108	463	250	142
Mars desent	MEV	LMO	93	18	930	0	930	2108	463	250	71
TEI	MTV	LMO	210	93	2521	10	2531	2249	463	250	366
EOI	MTV	LEO	111	69	4081	9	4090	2249	463	250	269



Scenario 2A: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CO FOR RETURN
175 KLB EXPANDER CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.	LEO	582	308	3300	72	3372	2x4420	470	350	894
LOI	Exp. stg.	LLO	259	57	1110	0.5	1111	2x4420	470	350	164
Lunar ascent	LEV	Lunar Surf.	410	144	1900	40	1940	2x4297	470	350	833
Lunar descent	LEV	LLO	110	40	2000	2	2002	2x4297	470	175	229
TMI	MTV	LLO	345	125	2005	21	2026	4420	470	175	727
MOI	MTV	LMO	210	94	2590	21	2611	4420	470	175	539
Mars ascent	MEV*	Mars Surf.	2750	2397	5300	151	5451	10x4340	293.2	1750	863
Mars desent	MEV*	LMO	122	35	930	0.3	930	10x4340	293.2	175	125
TEI	MTV	LMO	546	334	2521	104	2625	4420	293.2	175	1208
EOI	MTV	LEO	204	159	4081	41	4122	4420	293.2	175	574

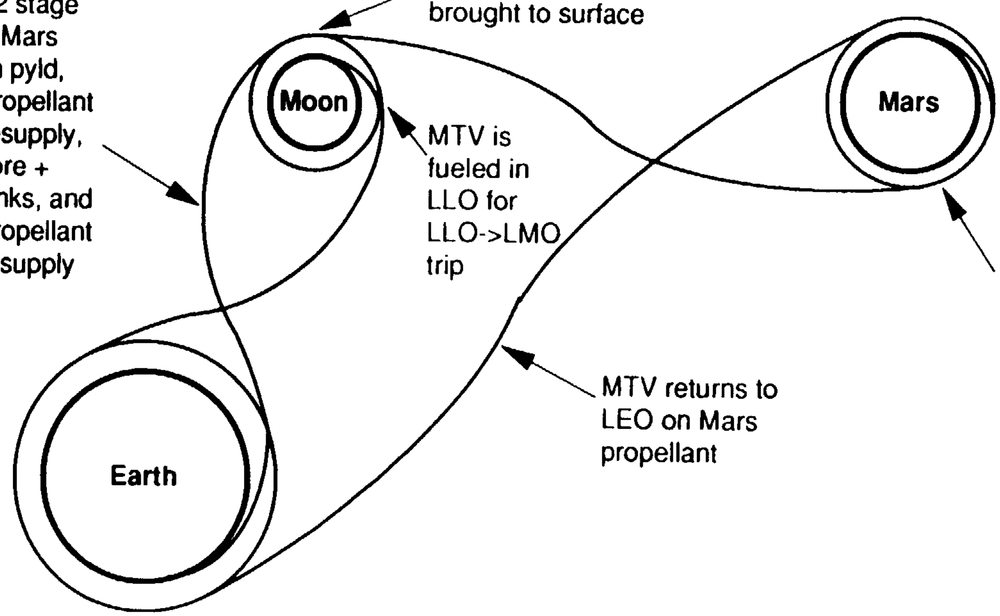
* These numbers are for each of 2 MEVs



LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CO FOR RETURN
Scenario 2B: 175 KLB GAS GENERATOR CYCLE ENGINE

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	531	286	3300	58	3358	2x1922	457.2	350	807
LOI	Exp. stg.	LLO	232	52	1110	0.4	1110	2x1922	457.2	350	146
Lunar ascent	LEV	Lunar Surf.	377	136	1900	33	1933	2x1832	457.2	350	762
Lunar descent	LEV	LLO	98	36	2000	1.6	2002	2x1832	457.2	175	202
TMI	MTV	LLO	317	117	2005	17	2022	1922	457.2	175	662
MOI	MTV	LMO	190	87	2590	17	2607	1922	457.2	175	483
Mars ascent	MEV*	Mars Surf.	2406	2100	5300	113	5413	10x1703	289.7	1750	747
Mars desent	MEV*	LMO	81	23	930	0.1	930	10x1703	289.7	175	83
TEI	MTV	LMO	529	325	2521	96	2617	1922	289.7	175	1161
EOI	MTV	LEO	196	153	4081	37	4118	1922	289.7	175	546

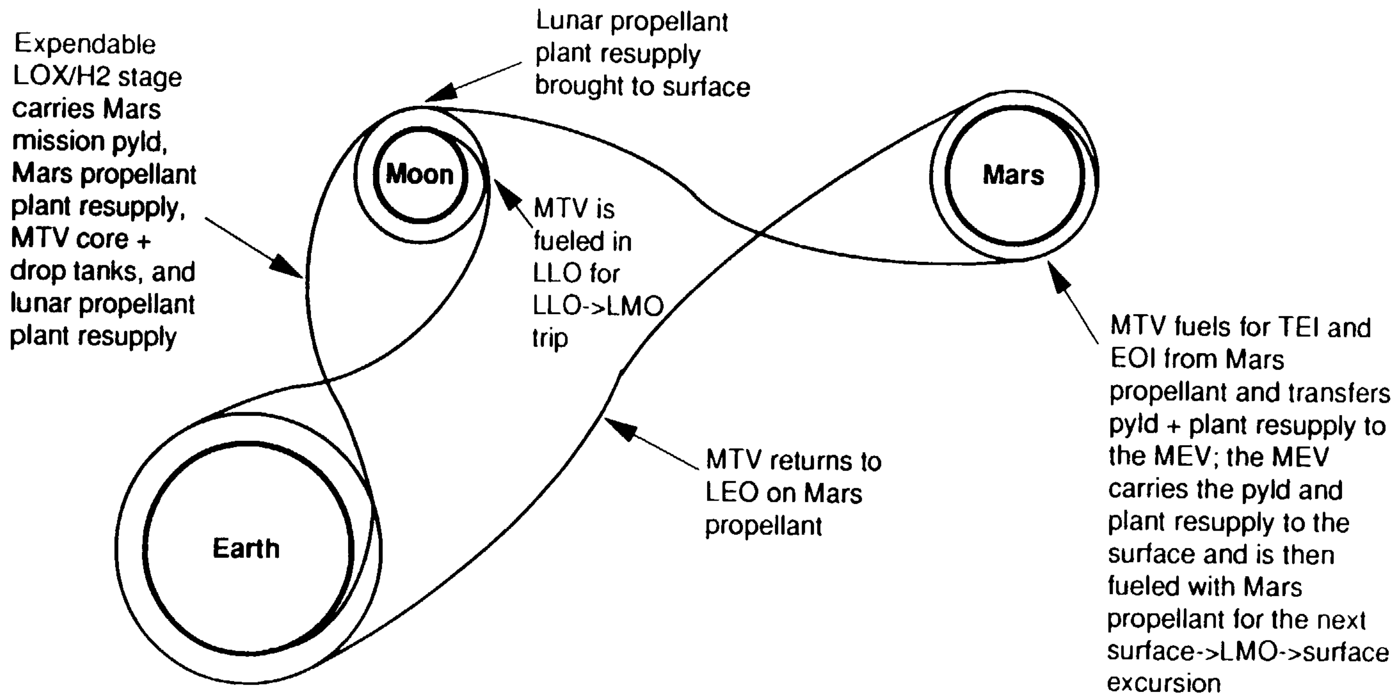
* These numbers are for each of 2 MEVs



Scenario 3A: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB EXPANDER CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Information Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	891	471	3300	83	3383	2X3915	472.3	500	962
LOI	Exp. stg.	LLO	399	87	1110	0.6	1111	2X3915	472.3	500	177
Lunar ascent	LEV	Lunar Surf.	611	215	1900	44	1944	3807	472.3	250	872
Lunar descent	LEV	LLO	145	52	2000	1.8	2002	3807	472.3	250	212
TMI	MTV	LLO	550	198	2005	6	2011	2X3915	472.3	500	404
MOI	MTV	LMO	338	150	2590	7	2597	2X3915	472.3	500	302
Mars ascent	MEV*	Mars Surf.	747	579	5300	85	5385	3X3725	389.9	750	647
Mars desent	MEV*	LMO	99	22	930	0.1	930	3X3725	389.9	250	73
TEI	MTV	LMO	297	146	2521	4	2525	2X3915	389.9	500	247
EOI	MTV	LEO	145	97	4081	3	4084	2X3915	389.9	500	164

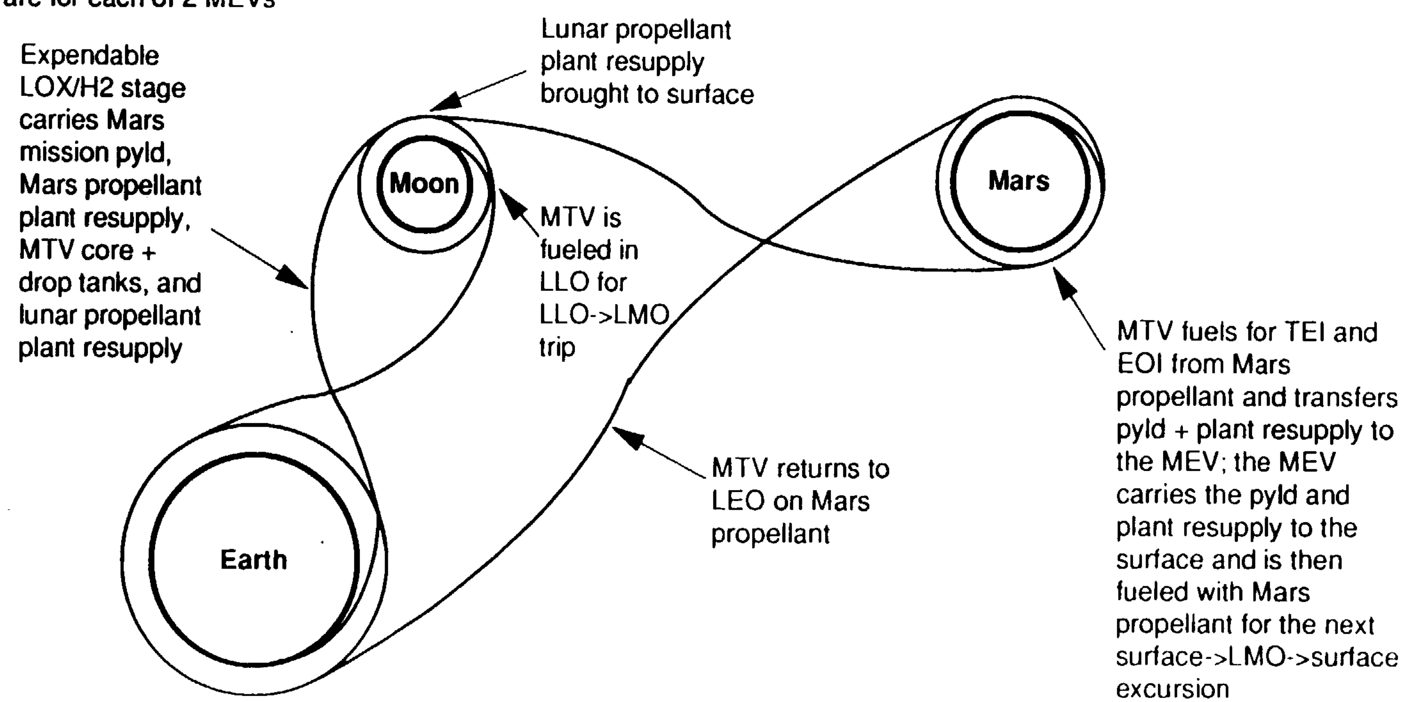
* These numbers are for each of 2 MEVs



Scenario 3B: LUNAR LOX (EARTH H2) FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB GAS GENERATOR CYCLE ENGINE

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.	LEO	863	462	3300	77	3377	2X2249	463	500	924
LOI	Exp. stg.	LLO	380	84	1110	0.5	1111	2X2249	463	500	169
Lunar ascent	LEV	Lunar Surf.	600	214	1900	42	1942	2108	463	250	852
Lunar descent	LEV	LLO	142	52	2000	1.7	2002	2108	463	250	207
TMI	MTV	LLO	530	194	2005	6	2011	2X2249	463	500	387
MOI	MTV	LMO	323	145	2590	6	2596	2X2249	463	500	287
Mars ascent	MEV*	Mars Surf.	700	545	5300	73	5373	3X1947	384.7	750	601
Mars desent	MEV*	LMO	89	20	930	0	930	3X1947	384.7	375	44
TEI	MTV	LMO	282	141	2521	4	2525	2X2249	384.7	500	234
EOI	MTV	LEO	137	92	4081	3	4084	2X2249	384.7	500	154

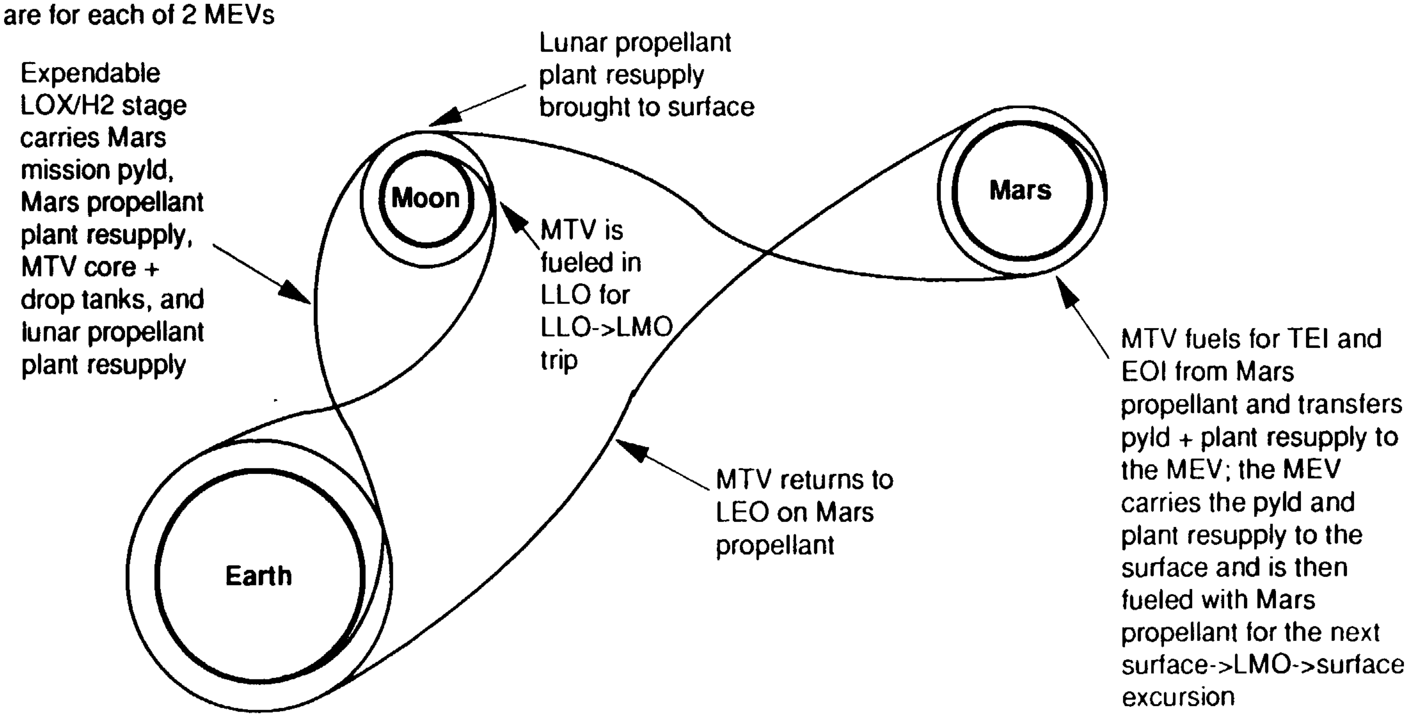
* These numbers are for each of 2 MEVs



Scenario 4A: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.	LEO	385	202	3300	31	3331	2X4420	470	350	586
LOI	Exp. stg.	LLO	172	38	1110	0.2	1110	2X4420	470	350	109
Lunar ascent	LEV	Lunar Surf.	560	228	1900	17	1917	2X4434	387.4	350	542
Lunar descent	LEV	LLO	38	16	2000	0.9	2001	2X4434	387.4	175	110
TMI	MTV	LLO	418	177	2005	28	2033	4516	387.4	175	845
MOI	MTV	LMO	235	119	2590	24	2614	4516	387.4	175	571
Mars ascent	MEV*	Mars Surf.	2774	2418	5300	153	5453	10X4461	293.2	1750	871
Mars desent	MEV*	LMO	124	35	930	0.3	930	10X4461	293.2	175	127
TEI	MTV	LMO	548	335	2521	105	2626	4516	293.2	175	1211
EOI	MTV	LEO	204	159	4081	41	4122	4516	293.2	175	575

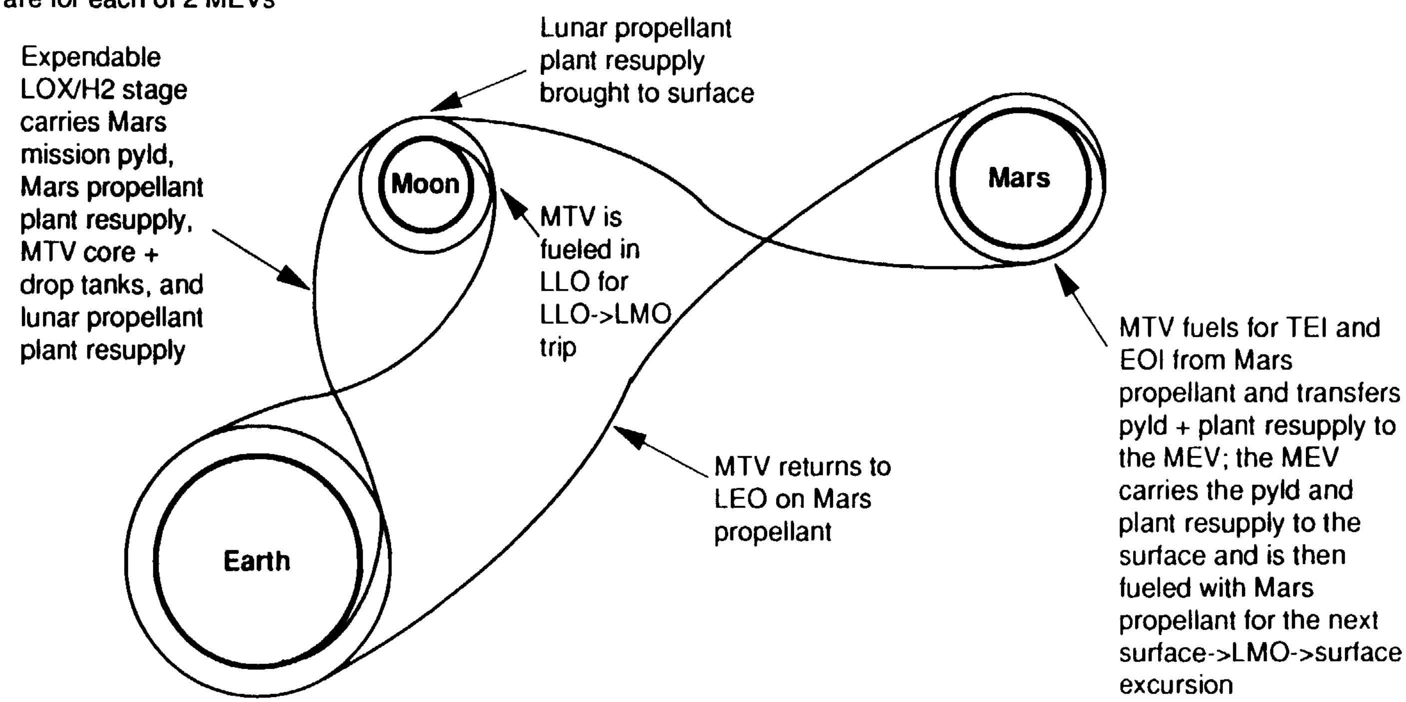
* These numbers are for each of 2 MEVs



Scenario 4B: LUNAR LOX/CH4 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB GAS GENERATOR CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.	LEO	339	181	3300	23	3323	2X1922	457.2	350	511
LOI	Exp. stg.	LLO	148	33	1110	0.1	1110	2X1922	457.2	350	94
Lunar ascent	LEV	Lunar Surf.	491	201	1900	13	1913	2X1841	383.3	350	473
Lunar descent	LEV	LLO	27	11	2000	0.4	2000	2X1841	383.3	175	107
TMI	MTV	LLO	372	158	2005	22	2027	1940	383.3	175	749
MOI	MTV	LMO	208	106	2590	19	2609	1940	383.3	175	503
Mars ascent	MEV*	Mars Surf.	2301	2002	5300	105	5405	10X1851	292.3	1750	719
Mars desent	MEV*	LMO	82	23	930	0.1	930	10X1851	292.3	175	83
TEI	MTV	LMO	515	314	2521	92	2613	1940	292.3	175	1134
EOI	MTV	LEO	193	150	4081	36	4117	1940	292.3	175	541

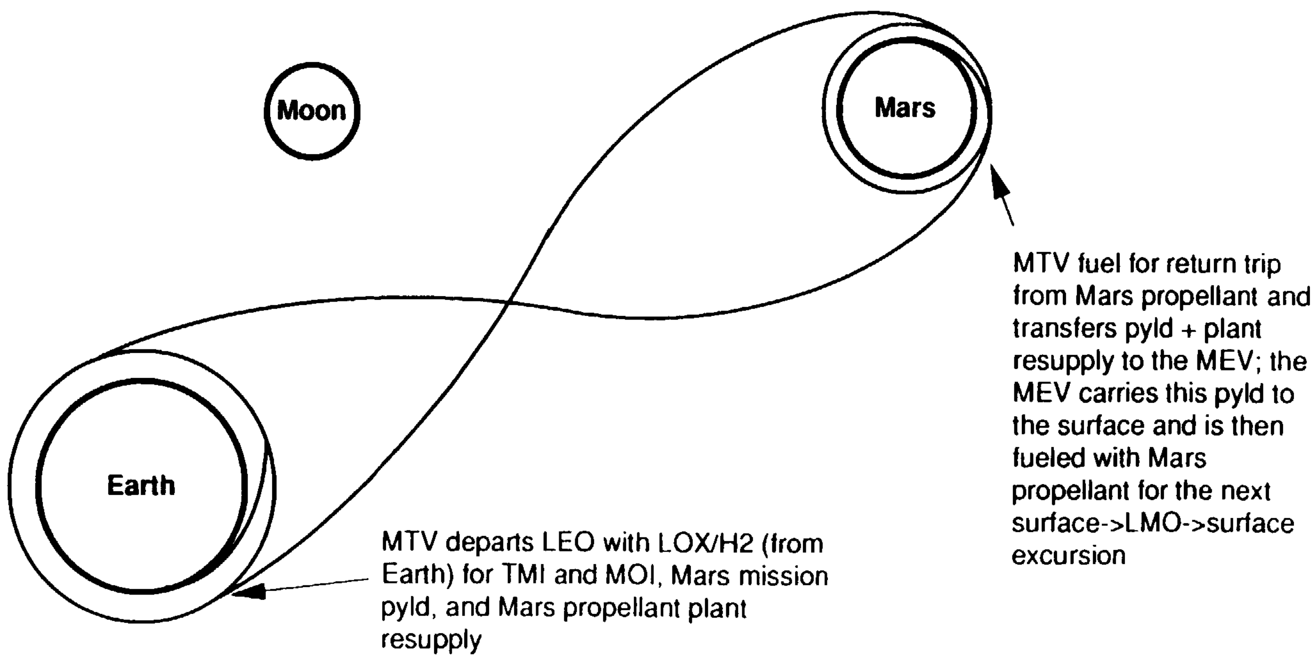
* These numbers are for each of 2 MEVs



Scenario 6A: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	584	352	3982	125	4107	2x4420	470	350	1020
MOI	MTV	LMO	208	93	2590	21	2611	4420	470	175	533
Mars ascent	MEV*	Mars Surf.	2724	2373	5300	148	5448	10x4340	293.2	1750	855
Mars desent	MEV*	LMO	121	34	930	0.3	930	10x4340	293.2	175	124
TEI	MTV	LMO	540	330	2521	102	2623	4420	293.2	175	1194
EOI	MTV	LEO	202	157	4081	40	4121	4420	293.2	175	569

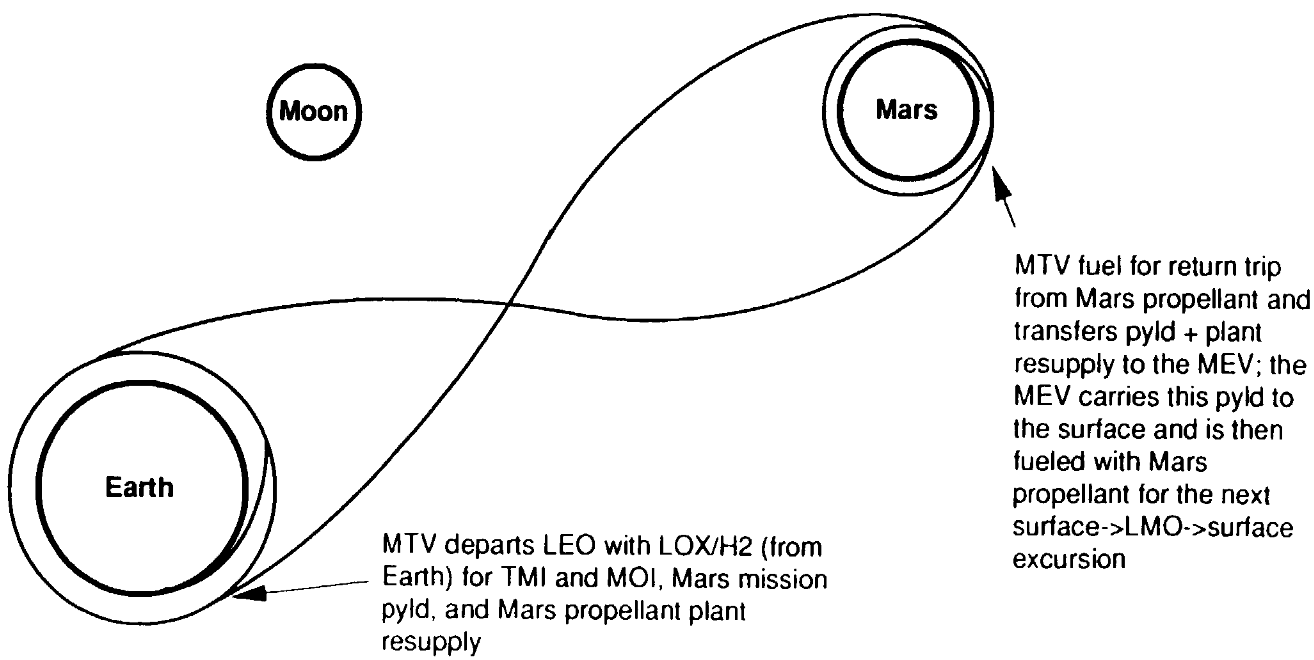
* These numbers are for each of 2 MEVs



Scenario 6B: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE W/ 165:1 AREA RATIO

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's			Engine Information			
					Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	588	360	3982	124	4106	2x3051	457.2	350	1016
MOI	MTV	LMO	206	94	2590	20	2610	3051	457.2	175	524
Mars ascent	MEV*	Mars Surf.	3038	2682	5300	176	5476	10x2966	283.2	1750	933
Mars desent	MEV*	LMO	108	32	930	0.3	930	10x2966	283.2	175	110
TEI	MTV	LMO	579	362	2521	114	2635	3051	283.2	175	1265
EOI	MTV	LEO	208	164	4081	41	4122	3051	283.2	175	574

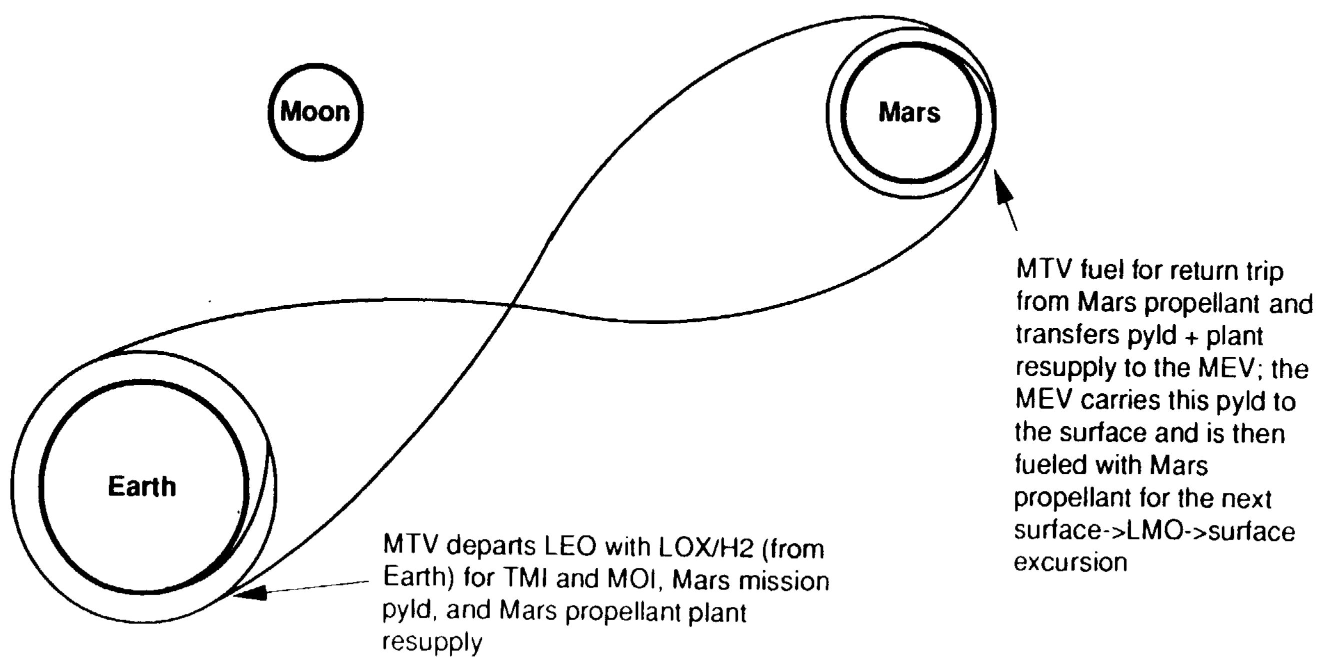
* These numbers are for each of 2 MEVs



Scenario 6C: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB GAS GENERATOR CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's Impulsive ΔV (m/sec)	ΔV's Gravity Loss ΔV (m/sec)	ΔV's Total ΔV (m/sec)	Engine Information Engine Mass (kg)	Engine Information Isp (sec)	Engine Information Thrust (klbs)	Engine Information Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	530	323	3982	100	4082	2x1922	457.2	350	912
MOI	MTV	LMO	188	86	2590	17	2607	1922	457.2	175	478
Mars ascent	MEV*	Mars Surf.	2381	2078	5300	111	5411	10x1703	289.7	1750	739
Mars desent	MEV*	LMO	81	23	930	0.1	930	10x1703	289.7	175	82
TEI	MTV	LMO	523	321	2521	94	2615	1922	289.7	175	1149
EOI	MTV	LEO	194	152	4081	37	4118	1922	289.7	175	542

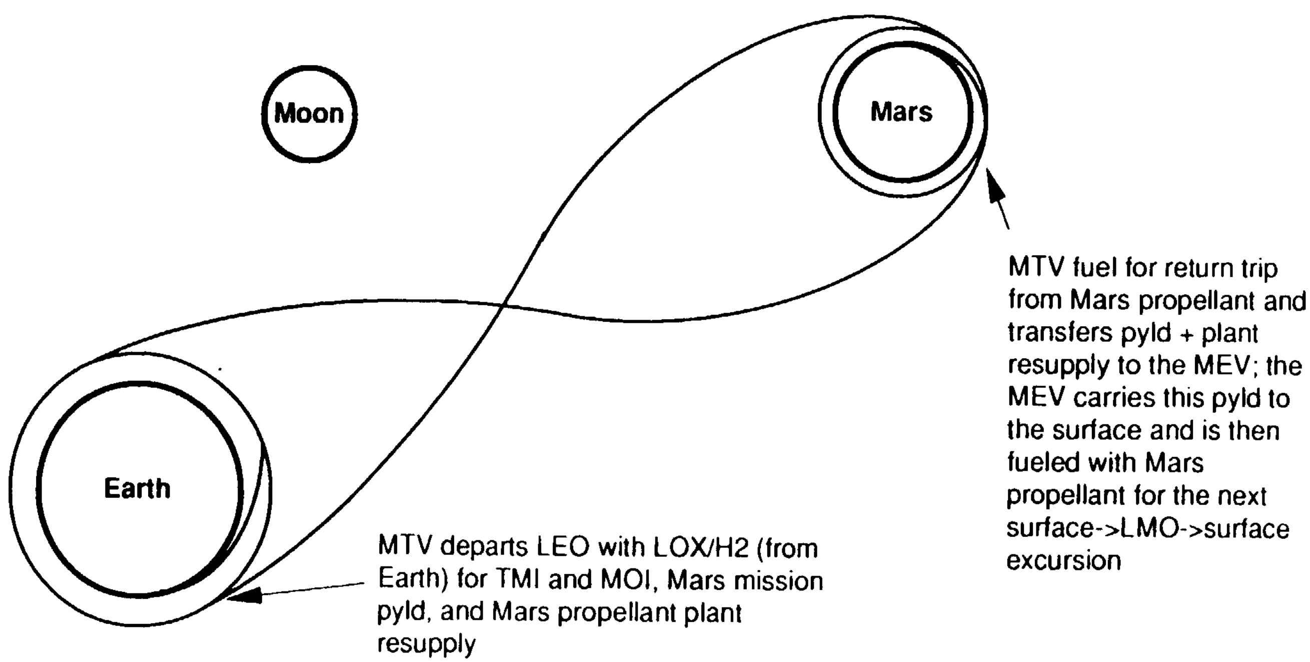
* These numbers are for each of 2 MEVs



EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN
Scenario 6D: 175 KLB EXPANDER CYCLE ENGINE, + 10% ENGINE MASS TRADE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	590	355	3982	127	4109	2x4862	470	350	1030
MOI	MTV	LMO	209	93	2590	21	2611	4862	470	175	536
Mars ascent	MEV*	Mars Surf.	2813	2453	5300	158	5458	10x4773	293.2	1750	884
Mars desent	MEV*	LMO	129	36	930	0.4	930	10x4773	293.2	175	131
TEI	MTV	LMO	547	334	2521	104	2625	4862	293.2	175	1209
EOI	MTV	LEO	205	159	4081	41	4122	4862	293.2	175	576

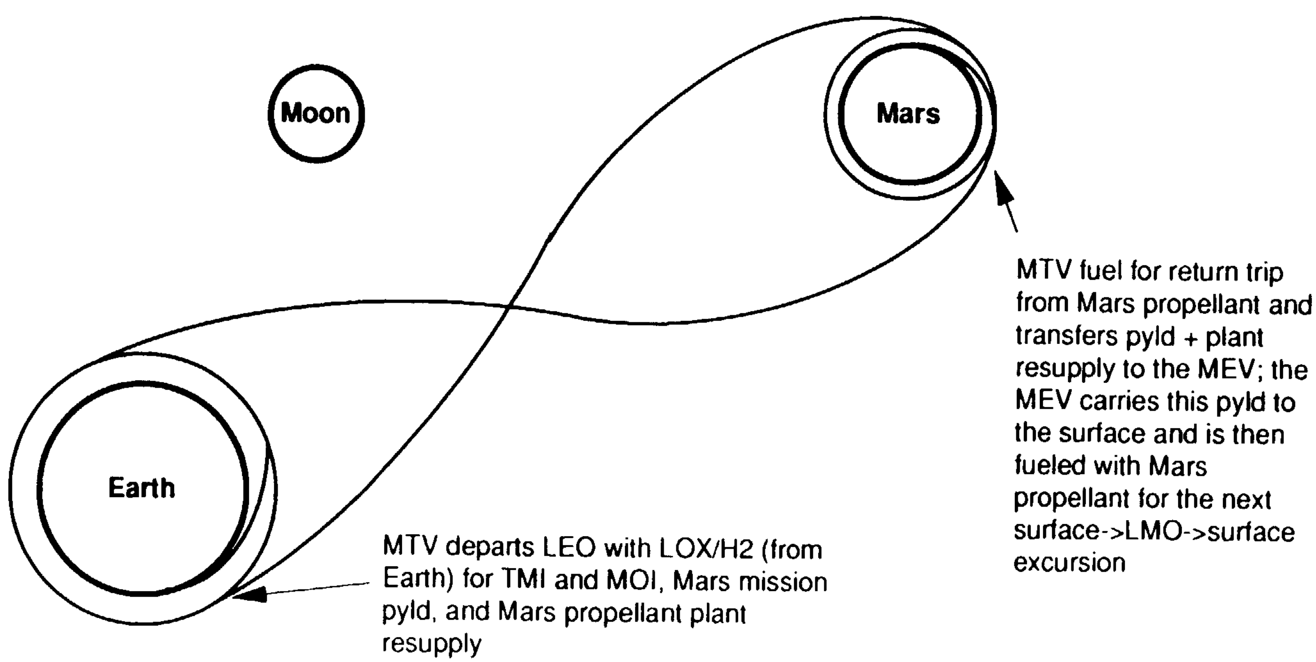
* These numbers are for each of 2 MEVs



Scenario 6E: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE, - 10% ENGINE MASS TRADE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	570	342	3982	118	4100	2x3978	470	350	994
MOI	MTV	LMO	204	204	2590	20	2610	3978	470	175	522
Mars ascent	MEV*	Mars Surf.	2635	2295	5300	138	5438	10x3906	293.2	1750	827
Mars desent	MEV*	LMO	114	32	930	0.3	930	10x3906	293.2	175	117
TEI	MTV	LMO	534	326	2521	99	2620	3978	293.2	175	1180
EOI	MTV	LEO	200	156	4081	40	4121	3978	293.2	175	563

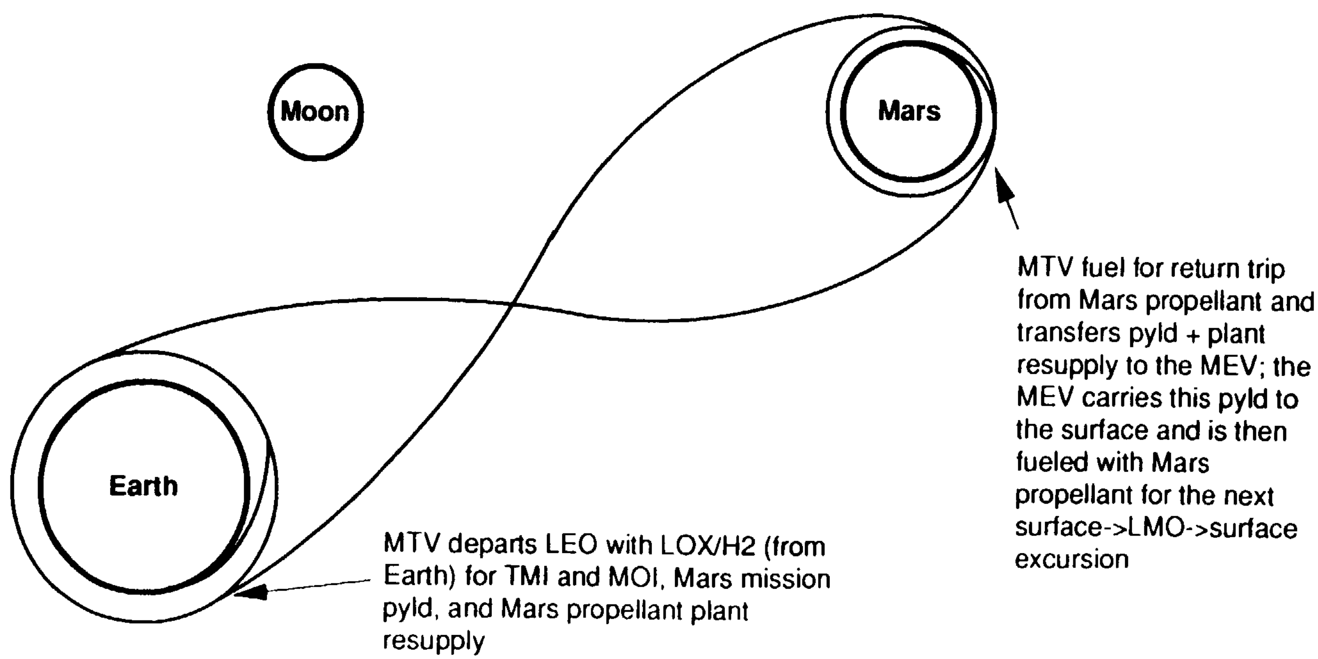
* These numbers are for each of 2 MEVs



Scenario 6F: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE, + 10% Isp TRADE

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	465	262	3982	84	4066	2x4420	517	350	835
MOI	MTV	LMO	182	75	2590	17	2607	4420	517	175	477
Mars ascent	MEV*	Mars Surf.	1681	1407	5300	63	5363	10x4340	322.5	1750	558
Mars desent	MEV*	LMO	104	27	930	0.2	930	10x4340	322.5	175	108
TEI	MTV	LMO	417	237	2521	64	2285	4420	322.5	175	945
EOI	MTV	LEO	173	128	4081	33	4114	4420	322.5	175	512

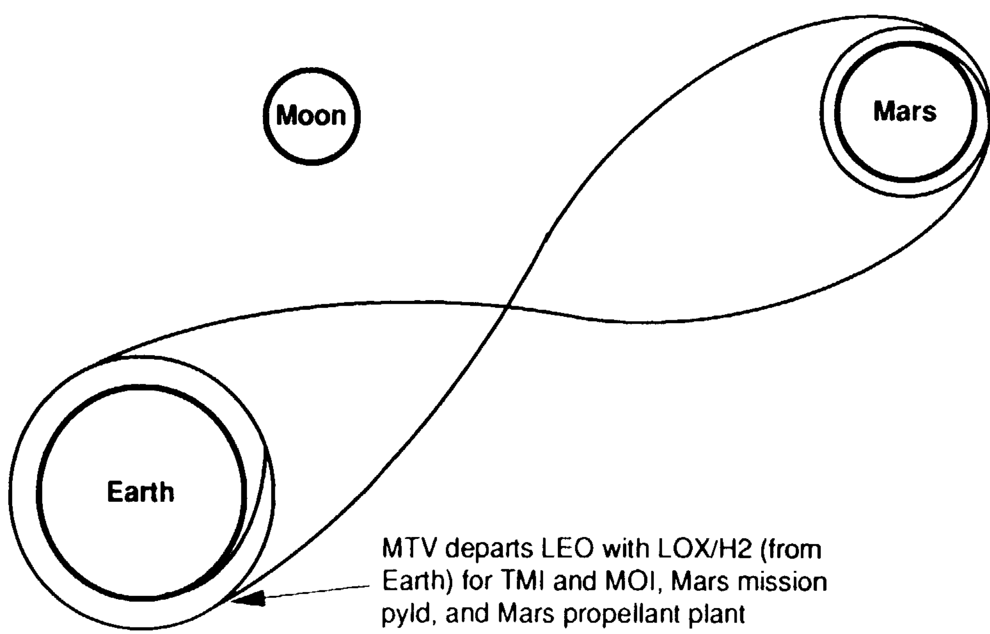
* These numbers are for each of 2 MEVs



Scenario 6G: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CO FOR RETURN 175 KLB EXPANDER CYCLE ENGINE, - 10% Isp TRADE

					ΔV's			Engine Information			
Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	935	615	3982	310	4292	2x4420	423	350	1606
MOI	MTV	LMO	286	138	2590	38	2628	4420	423	175	716
Mars ascent	MEV*	Mars Surf.	5643	5088	5300	170	5470	18x4340	263.9	3150	916
Mars desent	MEV*	LMO	216	67	930	0.3	930	18x4340	263.9	315	120
TEI	MTV	LMO	769	510	2521	197	2718	4420	263.9	175	1662
EOI	MTV	LEO	247	201	4081	54	4135	4420	263.9	175	656

* These numbers are for each of 2 MEVs



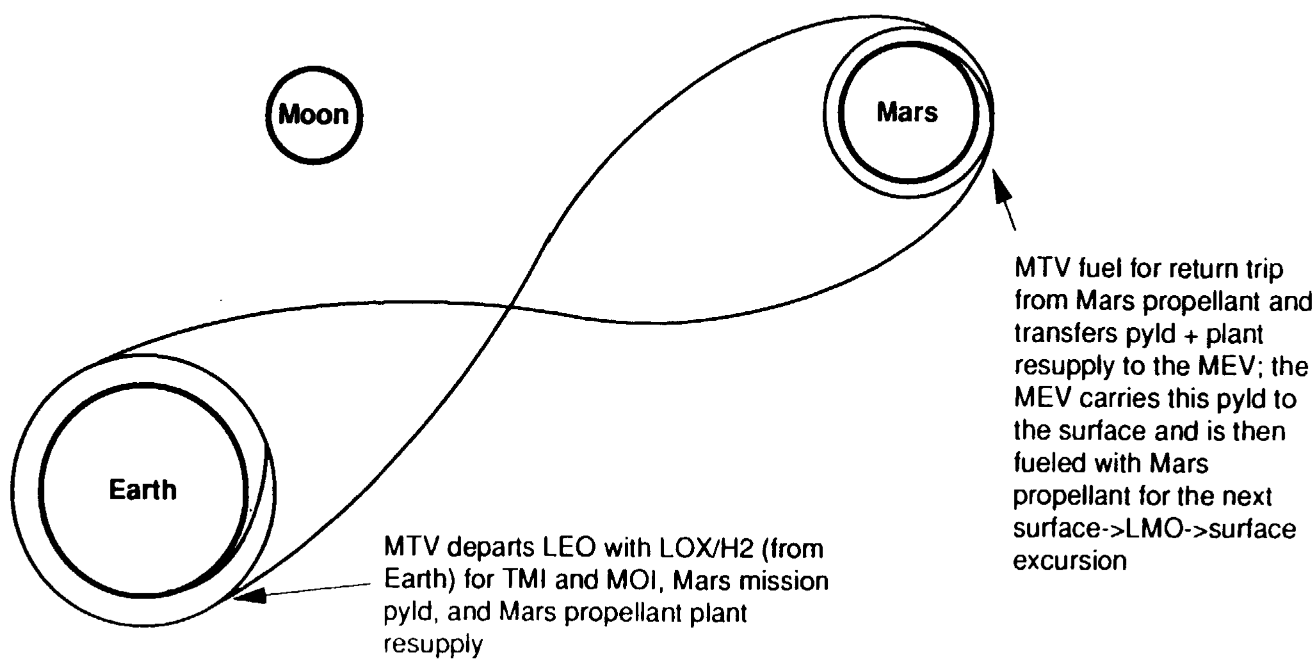
MTV departs LEO with LOX/H2 (from Earth) for TMI and MOI, Mars mission pyld, and Mars propellant plant resupply

MTV fuel for return trip from Mars propellant and transfers pyld + plant resupply to the MEV; the MEV carries this pyld to the surface and is then fueled with Mars propellant for the next surface->LMO->surface excursion

Scenario 7A: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB EXPANDER CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's Impulsive ΔV (m/sec)	ΔV's Gravity Loss ΔV (m/sec)	ΔV's Total ΔV (m/sec)	Engine Information Engine Mass (kg)	Engine Information Isp (sec)	Engine Information Thrust (klbs)	Engine Information Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	862	510	3982	32	4014	4x3915	472.3	1000	520
MOI	MTV	LMO	315	140	2590	24	2614	3915	472.3	250	566
Mars ascent	MEV*	Mars Surf.	688	533	5300	72	5372	3x3725	389.9	750	595
Mars desent	MEV*	LMO	94	21	930	0.1	930	3x3725	389.9	250	70
TEI	MTV	LMO	270	134	2521	14	2535	3915	389.9	250	450
EOI	MTV	LEO	132	88	4081	11	4092	3915	389.9	250	298

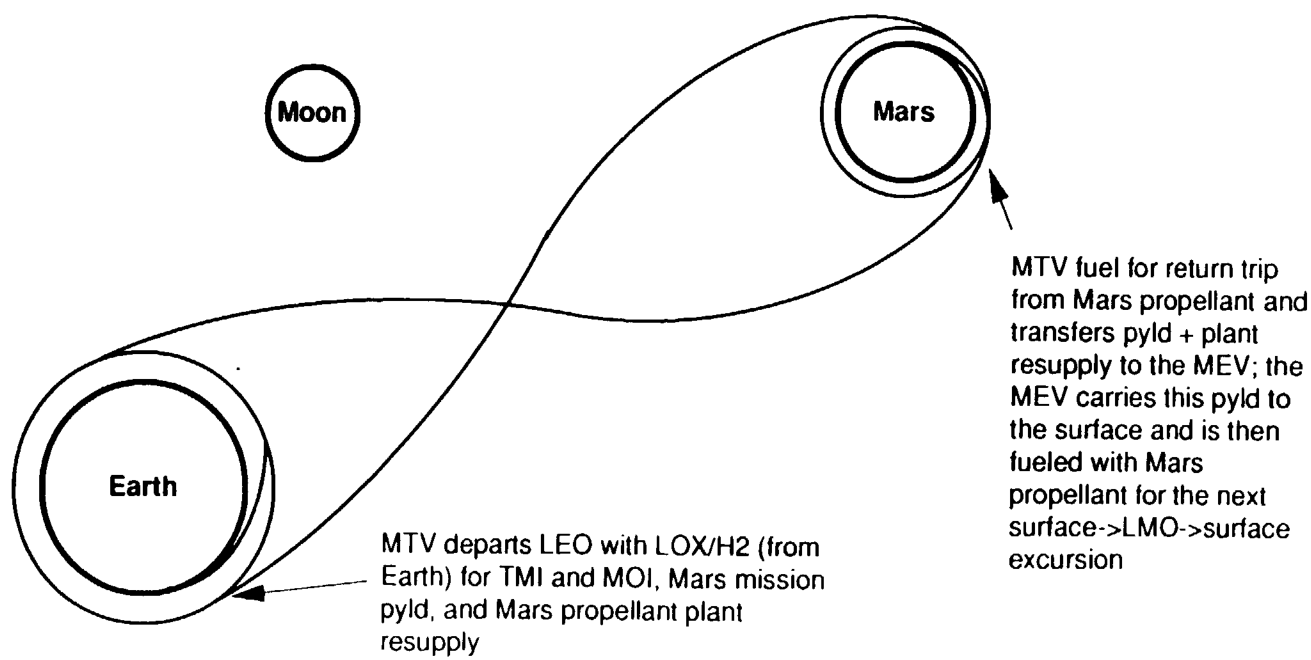
* These numbers are for each of 2 MEVs



Scenario 7B: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB EXPANDER CYCLE ENGINE, REUSE MTV MOC TANKS FOR TEI + EOC

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	968	573	3982	41	4023	4x3915	472.3	1000	585
MOI	MTV	LMO	355	158	2590	30	2620	3915	472.3	250	639
Mars ascent	MEV*	Mars Surf.	882	685	5300	119	5419	3x3725	389.9	750	766
Mars desent	MEV*	LMO	110	24	930	0.1	930	3x3725	389.9	250	82
TEI	MTV	LMO	327	162	2521	21	2542	3915	389.9	250	546
EOI	MTV	LEO	165	111	4081	17	4098	3915	389.9	250	373

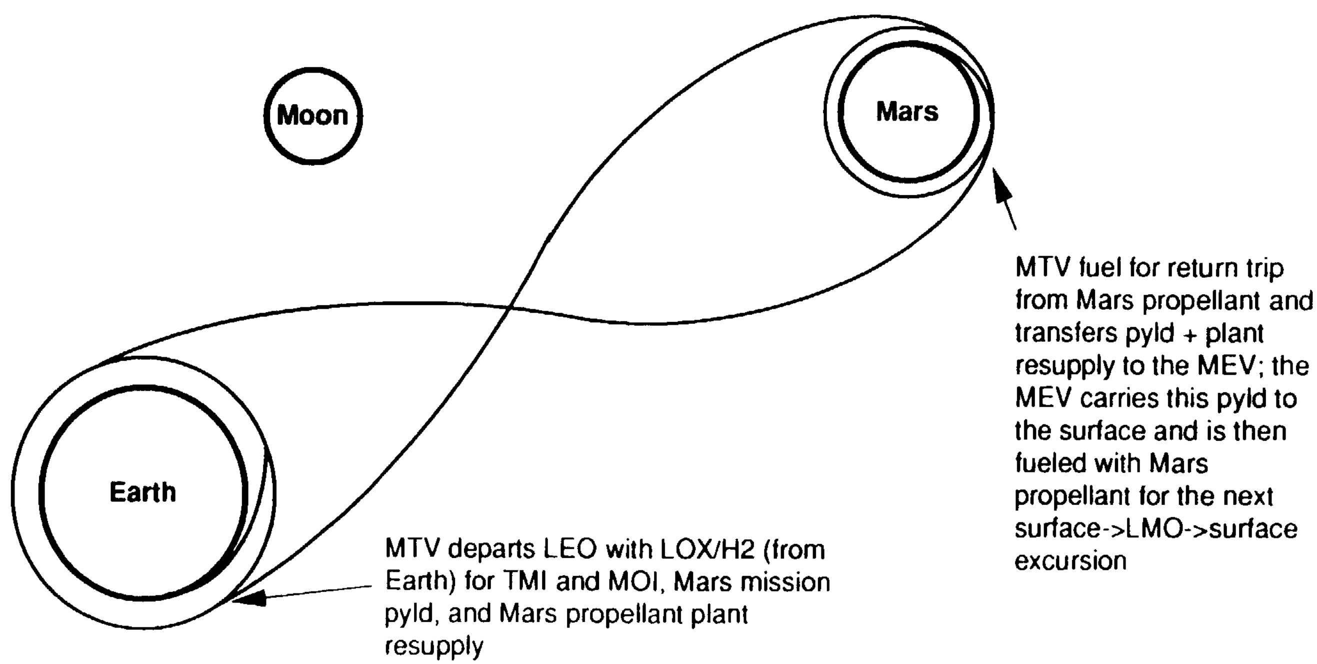
* These numbers are for each of 2 MEVs



Scenario 7C: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB EXPANDER CYCLE ENGINE, REUSE EVERYTHING (NO STAGING)

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's			Engine Information			
					Impulsive ΔV (m/sec)	Gravity Loss ΔV (m/sec)	Total ΔV (m/sec)	Engine Mass (kg)	Isp (sec)	Thrust (klbs)	Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	1280	762	3982	73	4055	4x3915	472.3	1000	777
MOI	MTV	LMO	516	228	2590	4	2594	4x3915	472.3	1000	230
Mars ascent	MEV*	Mars Surf.	1326	1044	5300	275	5575	3x3725	389.9	750	1167
Mars desent	MEV*	LMO	148	33	930	0.3	930	3x3725	389.9	250	110
TEI	MTV	LMO	530	261	2521	3	2524	4x3915	389.9	1000	220
EOI	MTV	LEO	269	180	4081	3	4084	4x3915	389.9	1000	152

* These numbers are for each of 2 MEVs

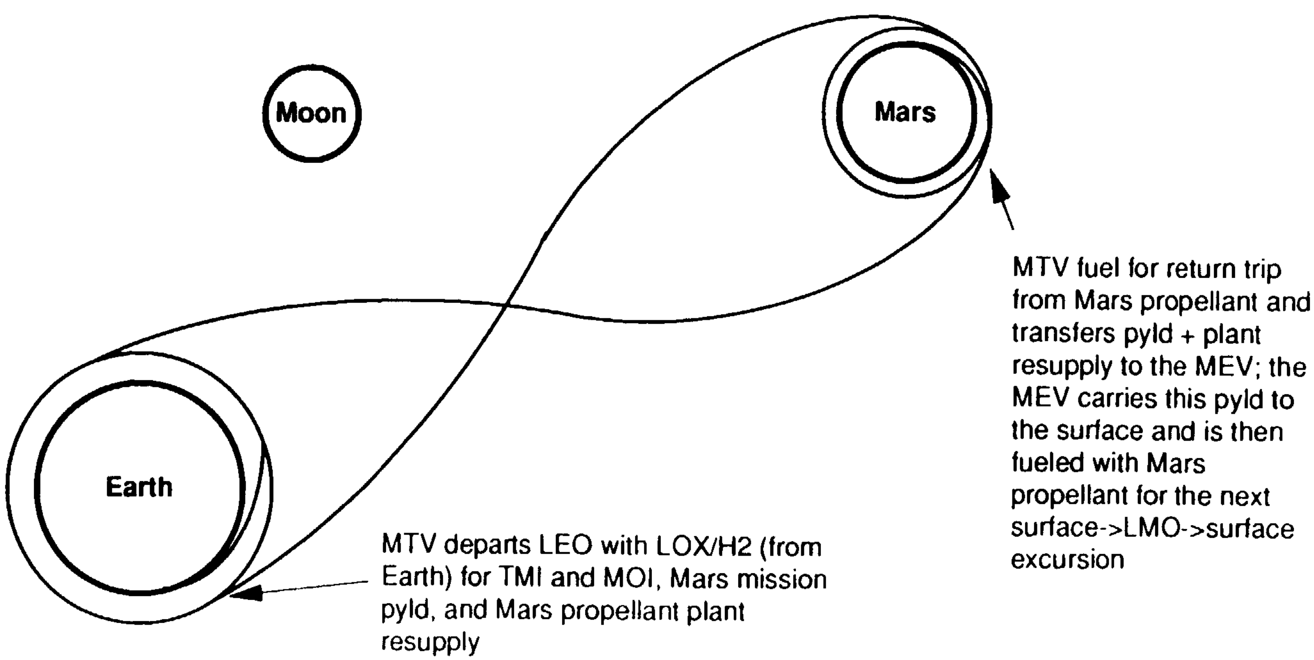


C-4.

Scenario 7D: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB GAS GENERATOR CYCLE ENGINE

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	843	504	3982	31	4013	4x2249	463	1000	505
MOI	MTV	LMO	308	139	2590	22	2612	2249	463	250	550
Mars ascent	MEV*	Mars Surf.	663	516	5300	65	5365	3x1947	384.7	750	569
Mars desent	MEV*	LMO	85	19	930	0	930	3x1947	384.7	250	63
TEI	MTV	LMO	266	133	2521	14	2535	2249	384.7	250	441
EOI	MTV	LEO	129	87	4081	10	4091	2249	384.7	250	289

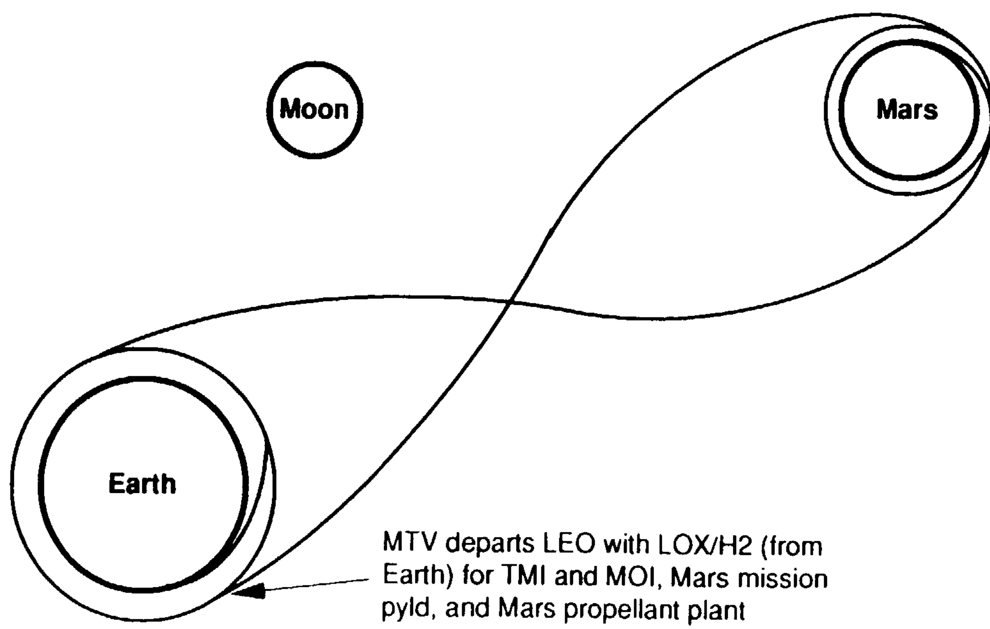
* These numbers are for each of 2 MEVs



Scenario 7E: EARTH LOX/H2 FOR OUTBOUND + MARS LOX/CH4 FOR RETURN 250 KLB EXPANDER CYCLE ENGINE, 2 MOV SETS-1 REUSED (&STAGED) FOR TEI, 1 REUSED FOR EOC

Burn	S/C	Location Of Burn	S/C Mass Prior To Burn (t)	Prop. used (t)	ΔV's: Impulsive ΔV (m/sec)	ΔV's: Gravity Loss ΔV (m/sec)	ΔV's: Total ΔV (m/sec)	Engine Information: Engine Mass (kg)	Engine Information: Isp (sec)	Engine Information: Thrust (klbs)	Engine Information: Burn Time (sec)
TLI	Exp. stg.										
LOI	Exp. stg.										
Lunar ascent	LEV										
Lunar descent	LEV										
TMI	Exp. stg.	LEO	837	495	3982	31	4013	4x3915	472.3	1000	505
MOI	MTV	LMO	305	136	2590	22	2612	3915	472.3	250	548
Mars ascent	MEV*	Mars Surf.	695	538	5300	73	5373	3x3725	389.9	750	602
Mars desent	MEV*	LMO	94	21	930	0	930	3x3725	389.9	250	70
TEI	MTV	LMO	275	136	2521	15	2536	3915	389.9	250	459
EOI	MTV	LEO	132	88	4081	11	4092	3915	389.9	250	298

* These numbers are for each of 2 MEVs



APPENDIX F

TANKAGE SYSTEM DESIGN DATA

APPENDIX F
TANKAGE SYSTEM DESIGN DATA

This appendix presents the detailed tankage system design analysis data for propellant tank systems evaluated in Section 4.3.

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 1
Mission Scenario No. : 1-Baseline Earth LOX/H_2
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 2-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 250,000
Number of Engines: 2
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 5
Total Exposure Time (days): 7
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 1,099,183
Propellants Burned (lbm): 1,090,409
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 843.3
Fuel Tank Inside Diameter (cm): 1,183.6 X 1,334.5
Oxidizer Tank Wall Thickness (cm): .109
Fuel Tank Wall Thickness (cm): .234
Oxidizer Tank Surface Area (m^2)**: 233.6
Fuel Tank Surface Area (m^2)**: 384.1

Oxidizer Tank Weight (lbm):
- -Tank Structure: 2629.4
- -Insulation: 930.6
- -Acquisition System: 13.7
- -Meteoroid Protection System: 1,030.2

Total: 4,603.9

Fuel Tank Weight (lbm):
- -Tank Structure: 12812.6
- -Insulation: 2112.8
- -Acquisition System: 15.6
- -Meteoroid Protection System: 1,693.8

Total: 16,634.8

Other Tankage System Weight (lbm):
- -Lines: 646.7
- -Tank Mounts: 4431.3
- -Pressurants Control System: 108.3

Total: 5,186.3

Total Tankage System Weight (lbm)*: 26,425.0

Total Tankage System Mass Fraction: .024

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 2
Mission Scenario No. : 1-Baseline Earth LOX/H_2
Mission Segment: MOC
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 2-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 250,000
Number of Engines: 2
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 300
Total Exposure Time (days): 300
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 605,699
Propellants Burned (lbm): 592,314
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 700.0
Fuel Tank Inside Diameter (cm): 1,088.1
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 162.7
Fuel Tank Surface Area (m^2)**: 366.4

Oxidizer Tank Weight (lbm):
- -Tank Structure: 1,512.7
- -Insulation: 619.6
- -Acquisition System: 12.8
- -Meteoroid Protection System: 717.5

Total: 2,862.6

Fuel Tank Weight (lbm):
- -Tank Structure: 3653.8
- -Insulation: 1431.9
- -Acquisition System: 14.8
- -Meteoroid Protection System: 1,615.8

Total: 6,716.3

Other Tankage System Weight (lbm):
- -Lines: 414.8
- -Tank Mounts: 2398.6
- -Pressurants Control System: 56.1

Total: 2,869.5

Total Tankage System Weight (lbm)*: 12,448.4

Total Tankage System Mass Fraction: .020

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 3
Mission Scenario No. : 1-Baseline Earth LOX/H_2
Mission Segment: TEI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 2-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 250,000
Number of Engines: 2
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 340
Total Exposure Time (days): 340
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 202,832
Propellants Burned (lbm): 195,850
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 484.6
Fuel Tank Inside Diameter (cm): 767.1
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 80.0
Fuel Tank Surface Area (m^2)**: 194.6

Oxidizer Tank Weight (lbm):
- -Tank Structure: 726.3
- -Insulation: 296.3
- -Acquisition System: 11.2
- -Meteoroid Protection System: 352.8

Total: 1,386.6

Fuel Tank Weight (lbm):
- -Tank Structure: 1817.2
- -Insulation: 685.1
- -Acquisition System: 13.3
- -Meteoroid Protection System: 858.2

Total: 3,373.8

Other Tankage System Weight (lbm):
- -Lines: 297.7
- -Tank Mounts: 795.0
- -Pressurants Control System: 54.7

Total: 1,147.4

Total Tankage System Weight (lbm)*: 5,907.8

Total Tankage System Mass Fraction: .028

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 4
Mission Scenario No. : 2-Lunar (Earth H_2) for Outbound and Mars LOX/CO for Return
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 1-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 14
Total Exposure Time (days): 14
Tank Material: 14
Insulation: Superfloc
Propellants Carried (lbm): 273,022
Propellants Burned (lbm): 270,691
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 536.9
Fuel Tank Inside Diameter (cm): 828.5
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 97.4
Fuel Tank Surface Area (m^2)**: 226.2

Oxidizer Tank Weight (lbm):
- -Tank Structure: 891.3
- -Insulation: 366.0
- -Acquisition System: 11.2
- -Meteoroid Protection System: 429.5

Total: 1,698.0

Fuel Tank Weight (lbm):
- -Tank Structure: 2118.5
- -Insulation: 868.8
- -Acquisition System: 13.3
- -Meteoroid Protection System: 997.5

Total: 3,998.1

Other Tankage System Weight (lbm):
- -Lines: 260.3
- -Tank Mounts: 1096.9
- -Pressurants Control System: 36.1

Total: 1,393.3

Total Tankage System Weight (lbm)*: 7,089.4

Total Tankage System Mass Fraction: .025

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 5
Mission Scenario No. : 2-Lunar LOX (Earth H_2) for Outbound and Mars LOX/CO for Return
Mission Segment: TEI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 1-B
Propellant Combination: LOX/CO
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 0.55
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 14
Total Exposure Time (days): 14
Tank Material: 14
Insulation: Superfloc
Propellants Carried (lbm): 325,607
Propellants Burned (lbm): 321,709
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 22.3
Oxidizer Tank Inside Diameter (cm): 421.1
Fuel Tank Inside Diameter (cm): 605.1
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 61.1
Fuel Tank Surface Area (m^2)**: 122.7

Oxidizer Tank Weight (lbm):
- -Tank Structure: 548.9
- -Insulation: 224.9
- -Acquisition System: 10.6
- -Meteoroid Protection System: 269.5

Total: 1,053.9

Fuel Tank Weight (lbm):
- -Tank Structure: 1130.7
- -Insulation: 464.5
- -Acquisition System: 12.2
- -Meteoroid Protection System: 541.1

Total: 2,148.5

Other Tankage System Weight (lbm):
- -Lines: 219.4
- -Tank Mounts: 1299.5
- -Pressurants Control System: 51.7

Total: 1,570.6

Total Tankage System Weight (lbm)*: 4,773.0

Total Tankage System Mass Fraction: .014

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 6
Mission Scenario No. : 4-LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-A
Propellant Combination: LOX/CH_4
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 3.6
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 14
Total Exposure Time (days): 14
Tank Material: 14
Insulation: Superfloc
Propellants Carried (lbm): 384,128
Propellants Burned (lbm): 381,711
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 12.5
Oxidizer Tank Inside Diameter (cm): 582.2
Fuel Tank Inside Diameter (cm): 567.9
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 113.9
Fuel Tank Surface Area (m^2)**: 108.6

Oxidizer Tank Weight (lbm):
- -Tank Structure: 1047.3
- -Insulation: 430.2
- -Acquisition System: 12.0
- -Meteoroid Protection System: 502.3

Total: 1991.8

Fuel Tank Weight (lbm):
- -Tank Structure: 996.7
- -Insulation: 409.3
- -Acquisition System: 11.9
- -Meteoroid Protection System: 478.9

Total: 1,896.8

Other Tankage System Weight (lbm):
- -Lines: 108.6
- -Tank Mounts: 1542.0
- -Pressurants Control System: 30.3

Total: 1,680.9

Total Tankage System Weight (lbm)*: 5,569.5

Total Tankage System Mass Fraction: .014

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 7
Mission Scenario No. : 4-Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: MOC
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-A
Propellant Combination: LOX/CH_4
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 3.6
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 300
Total Exposure Time (days): 300
Tank Material: 300
Insulation: Superfloc
Propellants Carried (lbm): 261,979
Propellants Burned (lbm): 257,938
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 12.5
Oxidizer Tank Inside Diameter (cm): 512.1
Fuel Tank Inside Diameter (cm): 500.9
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 88.9
Fuel Tank Surface Area (m^2)**: 85.2

Oxidizer Tank Weight (lbm):
- Tank Structure: 811.0
- Insulation: 331.3
- Acquisition System: 11.5
- Meteoroid Protection System: 392.2

Total: 1,546.0

Fuel Tank Weight (lbm):
- Tank Structure: 775.3
- Insulation: 315.2
- Acquisition System: 11.4
- Meteoroid Protection System: 375.7

Total: 1,477.6

Other Tankage System Weight (lbm):
- Lines: 157.5
- Tank Mounts: 1042.4
- Pressurants Control System: 30.0

Total: 1,229.9

Total Tankage System Weight (lbm)*: 4,253.5

Total Tankage System Mass Fraction: .016

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 8
Mission Scenario No. : 4-Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: TEI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-B
Propellant Combination: LOX/CO
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 0.55
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 14
Total Exposure Time (days): 14
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 731,017
Propellants Burned (lbm): 722,800
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 22.3
Oxidizer Tank Inside Diameter (cm): 551.2
Fuel Tank Inside Diameter (cm): 792.5
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 102.5
Fuel Tank Surface Area (m^2)**: 207.4

Oxidizer Tank Weight (lbm):
- -Tank Structure: 939.4
- -Insulation: 385.8
- -Acquisition System: 11.8
- -Meteoroid Protection System: 452.0

Total: 1,789.0

Fuel Tank Weight (lbm):
- -Tank Structure: 1937.6
- -Insulation: 385.8
- -Acquisition System: 13.4
- -Meteoroid Protection System: 914.6

Total: 3,251.4

Other Tankage System Weight (lbm):
- -Lines: 287.8
- -Tank Mounts: 2917.6
- -Pressurants Control System: 52.7

Total: 3,258.1

Total Tankage System Weight (lbm)*: 8,298.5

Total Tankage System Mass Fraction: .011

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 9
Mission Scenario No. : 1-Baseline Earth LOX/H_2
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 2-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 250,000
Number of Engines: 2
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 5
Total Exposure Time (days): 7
Tank Material: Al 2219-T87 Alloy
Insulation: Superfloc
Propellants Carried (lbm): 1,099,189
Propellants Burned (lbm): 1,099,409
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 856.0
Fuel Tank Inside Diameter (cm): 1,132.8X1,334.5
Oxidizer Tank Wall Thickness (cm): .183
Fuel Tank Wall Thickness (cm): .358
Oxidizer Tank Surface Area (m^2)**: 241.1
Fuel Tank Surface Area (m^2)**: 383.7

Oxidizer Tank Weight (lbm):
- -Tank Structure: 4574.6
- -Insulation: 930.6
- -Acquisition System: 13.7
- -Meteoroid Protection System: 1,063.3

Total: 6,582.2

Fuel Tank Weight (lbm):
- -Tank Structure: 20,738.3
- -Insulation: 2112.7
- -Acquisition System: 15.6
- -Meteoroid Protection System: 1,692.3

Total: 24,558.8

Other Tankage System Weight (lbm):
- -Lines: 646.7
- -Tank Mounts: 4,454.7
- -Pressurants Control System: 108.3

Total: 5,209.7

Total Tankage System Weight (lbm)*: 36,350.7

Total Tankage System Mass Fraction: .032

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 10
Mission Scenario No. : 4-Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: TEI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-B
Propellant Combination: LOX/CO
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 0.55
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 14
Tank Material: Al2219-T87 Alloy
Insulation: Superfloc
Propellants Carried (lbm): 731,012
Propellants Burned (lbm): 722,800
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 22.8
Oxidizer Tank Inside Diameter (cm): 551.2
Fuel Tank Inside Diameter (cm): 792.0
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .081
Oxidizer Tank Surface Area (m^2)**: 102.5
Fuel Tank Surface Area (m^2)**: 207.1

Oxidizer Tank Weight (lbm):
-Tank Structure: 982.6
-Insulation: 385.8
-Acquisition System: 11.8
-Meteoroid Protection System: 452.0
Total: 1,832.2

Fuel Tank Weight (lbm):
-Tank Structure: 2166.8
-Insulation: 796.7
-Acquisition System: 13.4
-Meteoroid Protection System: 913.3
Total: 3,890.2

Other Tankage System Weight (lbm):
-Lines: 287.8
-Tank Mounts: 2917.2
-Pressurants Control System: 52.7
Total: 3,257.7

Total Tankage System Weight (lbm)*: 8,980.1

Total Tankage System Mass Fraction: .012

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 11
Mission Scenario No. : 1-Baseline Earth LOX/H_2
Mission Segment: MOC
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 2-A
Propellant Combination: LOX/H_2
Thrust Level (lbf): 250,000
Number of Engines: 2
Mixture Ratio: 6.0
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 300
Total Exposure Time: 300
Tank Material: Weldalite
Insulation: MLI
Propellants Carried (lbm): 611,696
Propellants Burned (lbm): 592,314
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 35.0
Oxidizer Tank Inside Diameter (cm): 700.5
Fuel Tank Inside Diameter (cm): 1,105.4
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 158.6
Fuel Tank Surface Area (m^2)**: 390.9

Oxidizer Tank Weight (lbm):
- -Tank Structure: 1516.6
- -Insulation: 922.9
- -Acquisition System: 12.9
- -Meteoroid Protection System: 699.4

Total: 3,151.8

Fuel Tank Weight (lbm):
- -Tank Structure: 3770.0
- -Insulation: 2118.0
- -Acquisition System: 14.9
- -Meteoroid Protection System: 1,723.9

Total: 7,626.8

Other Tankage System Weight (lbm):
- -Lines: 412.7
- -Tank Mounts: 2398.9
- -Pressurants Control System: 56.1

Total: 2,867.7

Total Tankage System Weight (lbm)*: 13,646.3

Total Tankage System Mass Fraction: .022

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 12
Mission Scenario No. : 4-Lunar LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: MOC
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-A
Propellant Combination: LOX/CH_4
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 3.6
Average Orbit Distance from the Sun (A.U.): 1.3
Space Hold Time (days): 300
Total Exposure Time: 300
Tank Material: Weldalite
Insulation: MLI
Propellants Carried (lbm): 263,764
Propellants Burned (lbm): 257,938
Oxidizer Tank Pressure (psia): 22.8
Fuel Tank Pressure (psia): 12.5
Oxidizer Tank Inside Diameter (cm): 513.1
Fuel Tank Inside Diameter (cm): 502.9
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 86.0
Fuel Tank Surface Area (m^2)**: 82.7

Oxidizer Tank Weight (lbm):
-Tank Structure: 814.0
-Insulation: 489.8
-Acquisition System: 11.5
-Meteoroid Protection System: 379.2
Total: 1,694.5

Fuel Tank Weight (lbm):
-Tank Structure: 781.4
-Insulation: 466.0
-Acquisition System: 11.4
-Meteoroid Protection System: 364.8
Total: 1,623.6

Other Tankage System Weight (lbm):
-Lines: 155.5
-Tank Mounts: 1042.4
-Pressurants Control System: 30.0
Total: 1,227.9

Total Tankage System Weight (lbm)*: 4,546.0

Total Tankage System Mass Fraction: .017

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 13
Mission Scenario No. : 4-LOX/CH_4 for Outbound and Mars LOX/CH_4 Return
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-A
Propellant Combination: LOX/CH_4
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 3.6
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 14
Total Exposure Time: 14
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 386,289
Propellants Burned (lbm): 381,711
Oxidizer Tank Pressure (psia): 42.8
Fuel Tank Pressure (psia): 32.5
Oxidizer Tank Inside Diameter (cm): 582.2
Fuel Tank Inside Diameter (cm): 569.0
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 113.9
Fuel Tank Surface Area (m^2)**: 108.9

Oxidizer Tank Weight (lbm):
- Tank Structure: 1047.5
- Insulation: 430.3
- Acquisition System: 12.0
- Meteoroid Protection System: 502.3

Total: 1,992.1

Fuel Tank Weight (lbm):
- Tank Structure: 1001.0
- Insulation: 411.1
- Acquisition System: 11.9
- Meteoroid Protection System: 480.3

Total: 1,904.3

Other Tankage System Weight (lbm):
- Lines: 181.0
- Tank Mounts: 1542.0
- Pressurants Control System: 30.3

Total: 1,753.3

Total Tankage System Weight (lbm)*: 5,649.7

Total Tankage System Mass Fraction: .014

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

TANKAGE SYSTEM DATA SUMMARY*

Design No.: 14
Mission Scenario No. : 4-LOX/CH_4 for Outbound and Mars LOX/CO Return
Mission Segment: TMI
Vehicle Application: MTV
Engine Type (Cycle/No.): Expander/No. 3-A
Propellant Combination: LOX/CH_4
Thrust Level (lbf): 175,000
Number of Engines: 1
Mixture Ratio: 3.6
Average Orbit Distance from the Sun (A.U.): 1.0
Space Hold Time (days): 14
Total Exposure Time: 14
Tank Material: Weldalite
Insulation: Superfloc
Propellants Carried (lbm): 388,476
Propellants Burned (lbm): 381,711
Oxidizer Tank Pressure (psia): 62.8
Fuel Tank Pressure (psia): 52.5
Oxidizer Tank Inside Diameter (cm): 582.7
Fuel Tank Inside Diameter (cm): 571.0
Oxidizer Tank Wall Thickness (cm): .076
Fuel Tank Wall Thickness (cm): .076
Oxidizer Tank Surface Area (m^2)**: 114.1
Fuel Tank Surface Area (m^2)**: 109.7

Oxidizer Tank Weight (lbm):
- -Tank Structure: 1049.1
- -Insulation: 430.9
- -Acquisition System: 12.0
- -Meteoroid Protection System: 503.2

Total: 1,995.2

Fuel Tank Weight (lbm):
- -Tank Structure: 1008.2
- -Insulation: 414.0
- -Acquisition System: 12.0
- -Meteoroid Protection System: 483.8

Total: 1,918.0

Other Tankage System Weight (lbm):
- -Lines: 182.1
- -Tank Mounts: 1542.0
- -Pressurants Control System: 30.3

Total: 1,754.4

Total Tankage System Weight (lbm)*: 5,667.6

Total Tankage System Mass Fraction: .014

* Based on a single propellant tank set (fuel and oxidizer)
** Includes the thickness of insulation, but not the meteoroid protection system

APPENDIX G

TECHNOLOGY DEVELOPMENT PLAN PROGRAM ELEMENT PLAN DESCRIPTIONS

APPENDIX G
TECHNOLOGY DEVELOPMENT PLAN PROGRAM ELEMENT PLAN DESCRIPTIONS

Detailed descriptions of the program elements that make up the overall technology development plan associated with establishing the feasibility of Mars in situ-based propellant propulsion systems are presented in this appendix. Section 6.0 discussed in detail the rationale and interrelationship of these technology development plan program elements.

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 1

ISSUE: Materials Compatibility

DEVELOPMENT PHASE: Fundamental Research

TITLE: Mars In Situ Propellant Materials Compatibility Research

OBJECTIVE: Identify propulsion system material candidates that are compatible with potential Mars in situ propellants and/or propellant combinations. Propellants and/or propellant combinations for which material compatibility should be investigated include: CO, LOX, CO/H_2, H_2/CH_4, CO/CH_4.

MISSION IMPACT: Results will have a major impact on propulsion system weight, performance and vehicle tankage design approaches. These propulsion system parameters have a major impact on overall mission mass and ISPP requirements.

APPROACH:

1. Conduct screening task to identify candidate materials for the study.

2. Experimentally expose material specimens to propellant and/or propellant combinations to conditions typical of propellants tankage, thrust chamber, turbine drive, gas generator portions of an engine system (where appropriate) for corresponding exposure times.

3. Inspect specimens for chemical compatibility effects.

OUTPUTS/RESULTS: Listing of candidate propulsion materials that are compatible with potential propellants of interest.

SPECIAL FACILITIES/COMMENTS:

- Facility capabilities to expose material specimens to a variety of propellant(s) over a wide range of pressure and temperature conditions.

- Advanced material inspection instrumentation.

Title: 1. Mars In Situ Propellant Propulsion System
Materials Compatibility Research

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Material Screening Assessment					
2. Experimental Facility Design/ Development					
3. Compatibility Testing					
4. Speciman Evaluation					
5. Final Report					
ESTIMATED COST * ($K)	750	750			

* Total Estimated Cost in 1992 Dollars ($K) = 1,500

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TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 2

ISSUE: CO Cooling Data

DEVELOPMENT PHASE: Fundamental Research

TITLE: Fundamental CO Cooling Data Study

OBJECTIVE: Establish a fundamental database associated with CO cooling for conditions which are typical of thrust chambers and turbopumps.

MISSION IMPACT: Establishes operating limitations of LOX/CO engine options which greatly influences engine mass. This impacts overall mission mass and ISPP requirements.

APPROACH:

1. Define experimental facility requirements (heated tube and calorimetric thrust chamber).
2. Conduct tests at appropriate conditions.
3. Review results and establish CO cooling correlations and limitations.
4. Upgrade engine design analysis models with new data.

OUTPUTS/RESULTS: Accurate fundamental CO cooling database for the range of conditions to support the design of LOX/CO engines.

SPECIAL FACILITIES/COMMENTS:

- Heat tube and calorimetric thrust chamber facilities.

Title: 2. Fundamental CO Cooling Data Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Define Facilty Requirements					
2. Modify/Upgrade Facilities as Appropriate					
3. Design, Build the Test Article(s) and Conduct Tests					
4. Establish Database/ Upgrade Engineering Design Models					
5. Final Report					
ESTIMATED COST * ($K)	750	750			

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* Total Estimated Cost in 1992 Dollars ($K) = 1,500

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 3

ISSUE: LOX/CO Ignition/Combustion

DEVELOPMENT PHASE: Fundamental Research

TITLE: LOX/CO Ignition/Combustion Research

OBJECTIVE: Establish a fundamental database associated with LOX/CO ignition and combustion for conditions typical of an engine system.

MISSION IMPACT: Establishes LOX/CO engine performance and operating conditions that directly influence overall mission mass and ISPP requirements.

APPROACH: Experimentally measure LOX/CO ignition and combustion characteristics for conditions typical of engine systems; gas generator and main combustion chamber conditions. Establish ignition and stability limitations as well as measure performance for a host of injector/chamber design options. Results will then be included in an appropriate engineering design model.

OUTPUTS/RESULTS: Fundamental LOX/CO ignition and combustion database for the range of conditions of interest. Updated design correlation and models.

SPECIAL FACILITIES/COMMENTS:

- Breadboard combustor facility with advanced instrumentation capabilities.

Title: 3. LOX/CO Ignition/Combination Research

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Define Facilty Requirements					
2. Modify/Upgrade the Facility					
3. Design, Build the Test Article(s) and Conduct Tests					
4. Review Results and Establish Design Correlations					
5. Final Report					
ESTIMATED COST * ($K)	1000	1000			

* Total Estimated Cost in 1992 Dollars ($K) = 2,000

TOR29J/34b

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 4

ISSUE: CO Pumping

DEVELOPMENT PHASE: Fundamental Research

TITLE: Fundamental CO Pumping Database

OBJECTIVE: Establish CO pumping database for the range of conditions typical of a LOX/CO engine.

MISSION IMPACT: Support in establishing the design limitations of a LOX/CO engine, such as chamber pressure. This influences overall mission mass and ISPP requirements.

APPROACH:

1. Review CO pumping data from related areas such as the petrochemical industry.
2. Define an experiment and upgrade a facility to measure key parameters associated with the pumping of CO.
3. Review results and establish engineering correlations and limitations.
4. Upgrade engineering design models.

OUTPUTS/RESULTS: Fundamental CO pumping database for the range of conditions of interest. Updated design correlations and models.

SPECIAL FACILITIES/COMMENTS:

- Highly instrumented pumping facility which can operate over the conditions of interest.

Title: 4. Fundamental CO Pumping Database

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Literature Review					
2. Define Facility Requirements					
3. Modify/Upgrade the Facility					
4. Design, Build the Test Article(s) and Conduct Tests					
5. Review Results and Establish Design Correlations					
6. Final Report					
ESTIMATED COST * ($K)	300	300			

* Total Estimated Cost in 1992 Dollars ($K) = 600

TOR29J/34c

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 5

ISSUE: Injector Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: Common Multipropellant Injector Design Feasibility Study

OBJECTIVE: Establish feasibility and identify promising injector design(s) that can operate with more than one Mars in situ-based propellant combination over a wide operating range. Main injector and gas generator injector designs are to be investigated.

MISSION IMPACT: Addresses a critical Mars tripropellant engine design issue. This study can impact the Mars propellant options that can be used as well as the limits of operation conditions of such engines. Mission options, mass, and ISPP requirements can be greatly affected.

APPROACH: Design and experimental demonstration tasks that investigates the performance and limitation of injector designs for the conditions of interest.

1. Design concept screening study.

2. Select promising injector concepts for further study.

3. Modify/upgrade test facility.

4. Fabricate and test injector concepts.

5. Recommend most promising injector designs.

OUTPUTS/RESULTS: Recommendation of most promising common injector design(s) with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- Breadboard combustor facility with supporting instrumentation capability required.

Title: 5. Common Multipropellant Injection Design
Feasibility Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Design Screening Study					
2. Injector Concept Select Down					
3. Modify/Upgrade Test Facility					
4. Design, Build Concept(s) and Conduct Tests					
5. Establish Design Feasibility and Correlations					
6. Recommend Most Promising Injection Design(s)					
7. Final Report					
ESTIMATED COST * ($K)	1000	1750	1250		

* Total Estimated Cost in 1992 Dollars ($K) = 4,000

TOR29J/34d

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 6

ISSUE: Feed System/Turbopump Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: Common Multipropellant Feed System/Turbopump Design Feasibility Study

OBJECTIVE: Establish feasibility and identify promising feed system/turbopump design(s) that can operate efficiently with more than one Mars in situ-based fuels over a wide operating range.

MISSION IMPACT: Can influence the engine thrust-to-weight ratio that affects overall mission mass and ISPP requirements.

APPROACH: Design and experimental demonstration tasks which investigates feed system/ turbopumps designs that efficiently supply (pump) more than one fuel of interest over a wide operating range.

1. Design screening study.

2. Select promising feed system/turbopump design concepts for further study.

3. Modify/upgrade test facility.

4. Build and test candidate feed system design concept(s).

5. Establish feasibility of common feed system/turbopump design(s) and recommend most promising design concept(s), if possible.

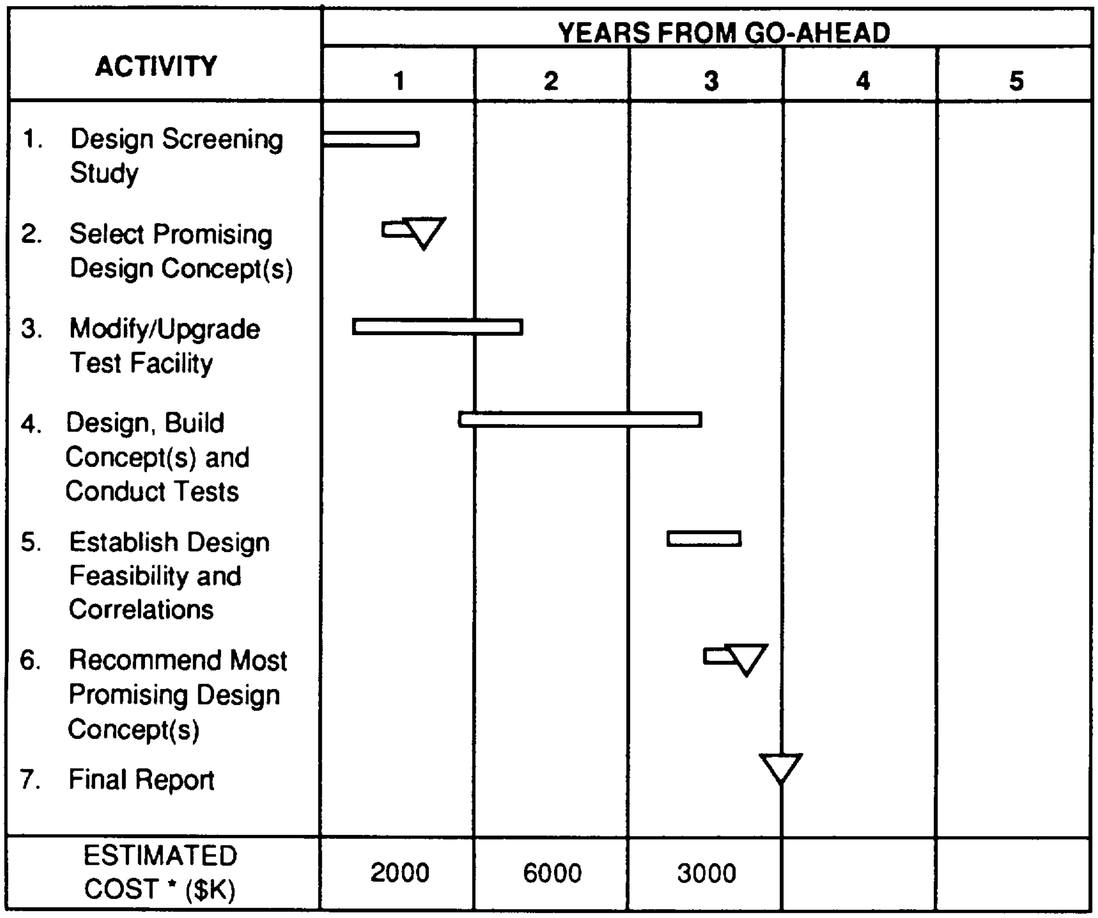
OUTPUTS/RESULTS: Establish the feasibility of common feed system/turbopump design options. Recommendations, if possible, of the most promising design with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- Highly flexible feed system/turbopump development test facility with extensive instrumentation required.

Title: 6. Common Multipropellant Feed System
Turbopump Design Feasibility Study

- SCHEDULE/COST -



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* Total Estimated Cost in 1992 Dollars ($K) = 11,000

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 7

ISSUE: Thrust Chamber Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: Common Thrust Chamber Design Feasibility Study

OBJECTIVE: Establish feasibility and identify promising thrust chamber design(s) that can operate with more than one Mars in situ-based propellant combination over a wide operating range.

MISSION IMPACT: Addresses a critical Mars tripropellant engine design issue. This study can impact the engine systems thrust-to-weight ratio and performance that affects overall mission mass and ISPP requirements.

APPROACH: Design and experimental demonstration tasks that investigate common thrust chamber design option(s) that can use more than one in situ propellant over a wide operating range.

1. Design screening study.

2. Select promising thrust chamber design concept(s) for further study.

3. Modify/upgrade test facility.

4. Build and test candidate thrust chamber design concept(s).

5. Establish feasibility of thrust chamber design(s) and recommend most promising concepts, if possible.

OUTPUTS/RESULTS: Establish the feasibility of common propellant cooled thrust chamber design option(s), if possible. Recommendations, if possible, of the most promising design concept(s) with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- Flexible breadboard subscale engine test facility with supporting instrumentation is required.

Title: 7. Common Thrust Chamber Design Feasibility Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Design Screening Study					
2. Select Promising Design Concept(s)					
3. Modify/Upgrade Test Facility					
4. Design, Build Concept(s) and Conduct Tests					
5. Establish Design Feasibility and Correlations					
6. Recommend Most Promising Design Concept(s)					
7. Final Report					
ESTIMATED COST * ($K)	1500	3000	2000		

* Total Estimated Cost in 1992 Dollars ($K) = 6,500

TOR29J/34f

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 8

ISSUE: Gas Generator Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: LOX/CO Gas Generator Design Feasibility Study

OBJECTIVE: Establish feasibility and identify LOX/CO gas generator design(s) that can operate over a wide range of operating conditions.

MISSION IMPACT: Addresses a critical LOX/CO gas generator (GG) cycle engine design. If feasible, such engine systems may be possible with high thrust-to-weight characteristics that impact overall mission mass and ISPP requirements.

APPROACH: Design and experimental investigation tasks that examine LOX/CO gas generator design concept, such as a - stoichmotric gas generator design, which can operate over a wide range.

1. Design concept screening study.

2. Select promising GG design concept(s).

3. Modify/upgrade test facility.

4. Build and test candidate GG design concept(s).

5. Establish feasibility of such design(s) and recommend most promising concept(s), if possible.

OUTPUTS/RESULTS: Establish the feasibility of LOX/CO GG design option(s), if possible. Recommendations, if possible, of the most promising design concept(s) with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- Burner/chamber test facility with support instrumentation is required.

Title: 8. LOX/CO Gas Generator Design Feasibility Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Design Screening Study					
2. Select Promising Design Concepts					
3. Modify/Upgrade Test Facility					
4. Design, Build Concept(s) and Conduct Tests					
5. Establish Design Feasibility and Correlations					
6. Recommend Most Promising Design Concept(s)					
7. Final Report					
ESTIMATED COST * ($K)	1000	1500	1000		

TOR29J/34g

* Total Estimated Cost in 1992 Dollars ($K) = 3,500

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 9

ISSUE: System Control/Health Monitoring

DEVELOPMENT PHASE: Exploratory Development

TITLE: Common Control/Health Monitoring System Design Feasibility Study

OBJECTIVE: Establish feasibility and identify promising common control/health monitoring system(s) that can operate with numerous in situ Mars propellant combinations for various engine system operating modes.

MISSION IMPACT: Addresses a critical Mars tripropellant engine design issue. Can impact engine propellant combination options and mission options.

APPROACH: Identify common control/health monitoring system design issues. Identify promising system architecture option(s) and candidate system design(s) through real-time simulation.

OUTPUTS/RESULTS: Establish the feasibility and identify promising design approaches, if possible. Provide support engineering data and development plans of promising design concept option(s).

SPECIAL FACILITIES/COMMENTS:

- Real time engine control simulation facility is required.

Title: 9. Common Control/Health Monitor System
Design Feasibility Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD 1	2	3	4	5
1. Define System Issues					
2. Investigate System Design Approach(es)					
3. Recommend Design Approach(es)					
4. Develop Simulation Facility and Test					
5. Analyze Results					
6. Recommend Most Promising Designs Concept(s)					
7. Final Report					
ESTIMATED COST * ($K)	300	500			

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* Total Estimated Cost in 1992 Dollars ($K) = 800

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 10

ISSUE: Propellant Tank Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: Common Propellant Tank Design and Supporting Operations Study

OBJECTIVE: Establish feasibility and identify common propellant tank design(s) and supporting operation requirements and design approaches, such as for resupply. Identification of high payoff alternative tank designs will also be investigated.

MISSION IMPACT: Can have a major impact on MTV designs, overall mission mass and ISPP requirements.

APPROACH: System analysis design and experimental study which:

1. Establishes in situ tank requirements and issues.

2. Screens design options and their supporting operations requirements.

3. Demonstrates subscale tank design options and supporting operations under simulated environmental conditions.

OUTPUTS/RESULTS: Recommendation of the most promising tank design(s) and supporting operational approach(s) with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- Propellant storage/handling and an adequate long-term space simulation facility is required.

Title: 10. Common Propellant Tank Design and Supporting Operations Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Define Tank Design and Supporting Operations Issues					
2. Screen Design Approaches					
3. Recommend Promising Design(s) for Further Study					
4. Modify/Upgrade Test Facility					
5. Design, Fabricate and Test the Promising Concept(s)					
6. Review Results					
7. Recommend Most Promising Concept(s)					
8. Final Report					
ESTIMATED COST * ($K)	1500	2000	1000		

* Total Estimated Cost in 1992 Dollars ($K) = 4,500

TOR29J/34i

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 11

ISSUE: Nozzle Design

DEVELOPMENT PHASE: Exploratory Development

TITLE: Lightweight, Compact High Area Ratio Nozzle Design Study

OBJECTIVE: Identify lightweight compact high area ratio nozzle designs for Mars in situ tripropellant engine systems employing LOX/CO as one of its two propellant combinations.

MISSION IMPACT: Addresses a critical design issue of Mars in situ tripropellant engine systems that employ LOX/CO. Such advanced nozzle designs are required to reduce engine system mass and stowed volume requirements. This impacts overall mission mass and LEO vehicle support options and ISPP requirements.

APPROACH: Systems analysis, design and experimental demonstration of promising lightweight, compact (while stowed) nozzle design(s) will be undertaken. High area ratio nozzle design option(s) for such engine systems including translated and alternate nozzle concepts will be examined. Promising design option(s) will be demonstrated by subscale high pressure gas and breadboard engine testing, respectively.

OUTPUTS/RESULTS: Identification of promising nozzle design concept(s) with supporting engineering data.

SPECIAL FACILITIES/COMMENTS:

- A hot high pressure gas facility as well as a subscale breadboard engine system/test facility are required.

Title: 11. Lightweight, Compact High Area Ratio Nozzle Design Study

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Define Nozzle Requirements					
2. Screen Design Approaches					
3. Recommend Promising Design(s) for Further Study					
4. Modify/Upgrade Test Facility					
5. Design, Fabricate and Test Design Option(s)					
6. Review Results					
7. Recommend Most Promising Concept(s)					
8. Final Report					
ESTIMATED COST * ($K)	750	2000	2500		

TOR29J/34j

* Total Estimated Cost in 1992 Dollars ($K) = 5,250

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 12

ISSUE: Engine System Demonstration

DEVELOPMENT PHASE: Prototype Demonstration

TITLE: Mars Tripropellant Subscale Engine System Demonstration Program

OBJECTIVE: Successfully demonstrate and establish feasibility of a subscale (15,000-60,000 lbf thrust) candidate Mars in situ propellant-based tripropellant engine system design concept.

MISSION IMPACT: Will verify feasibility and characterize a Mars tripropellant engine design concept. This will lead to more accurate assessment of Mars in situ propellant-based propulsion system and mission options.

APPROACH: Design, fabricate, and ground test a subscale candidate Mars in situ propellant-based tripropellant engine system design concept. Verify both design and off-design performance and reliability for such an engine system for its various operating modes.

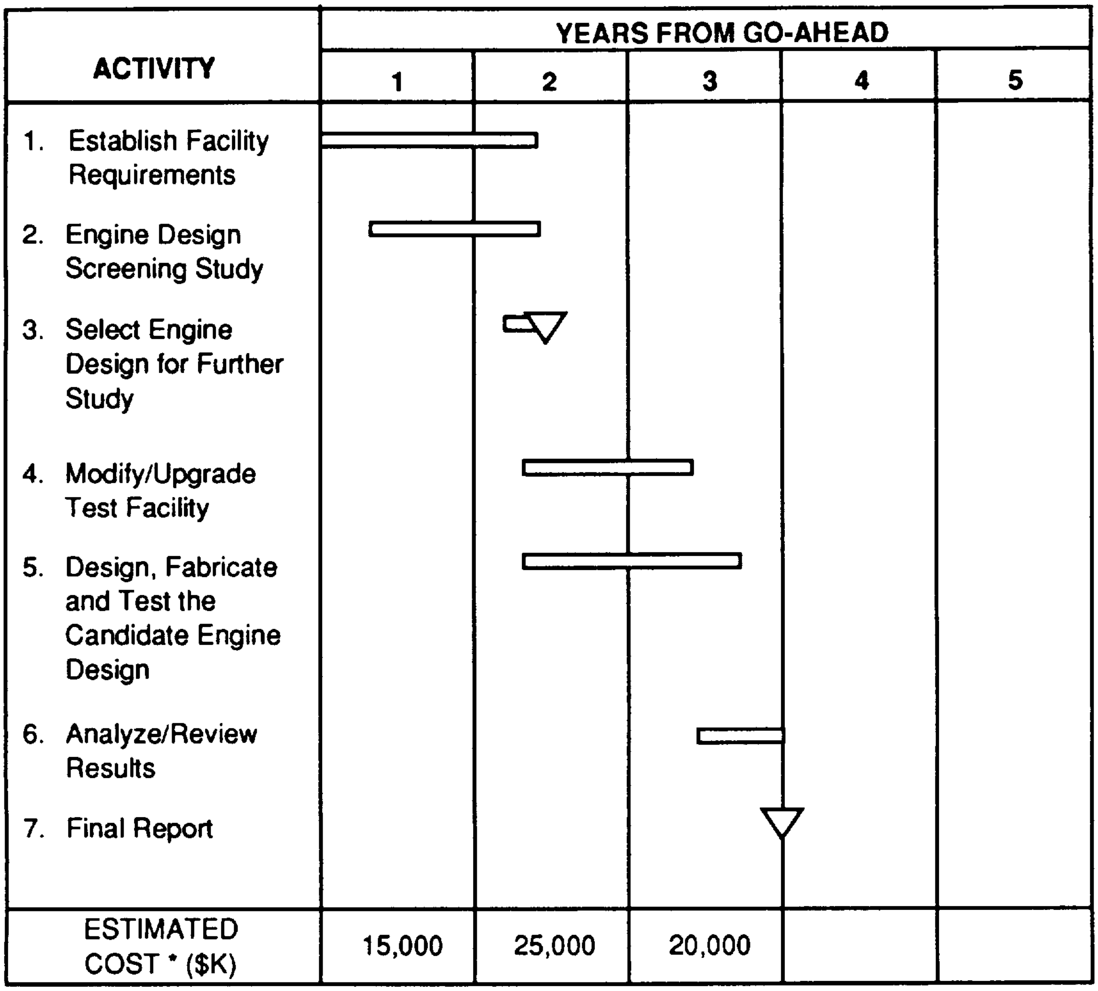
OUTPUTS/RESULTS: Engineering data characterizing the engine system that can support a flight system development decision.

SPECIAL FACILITIES/COMMENTS:

- Subscale engine test facility.

Title: 12. Mars Tripropellant Subscale Engine System
Demonstration Program

- SCHEDULE/COST -



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* Total Estimated Cost in 1992 Dollars ($K) = 60,000

TECHNOLOGY DEVELOPMENT PLAN ELEMENT

PROGRAM No.: 13

ISSUE: Preliminary Design/System Integration

DEVELOPMENT PHASE: System Engineering

TITLE: Preliminary Mars In situ Propellant Mission/Vehicle/Engine System Design Studies

OBJECTIVE: Assesses the impact of engine technology data as it becomes available, on evolving Mars in situ propellant-based mission, vehicle and engine system designs.

MISSION IMPACT: Will allow for more accurate assessment of Mars in situ propellant-based mission options as engine technology data becomes available.

APPROACH: An ongoing preliminary system design study, during the fundamental research and exploratory development engine development phases, which assesses mission options, vehicle and engine systems design concepts as engine technology data becomes available.

OUTPUTS/RESULTS: Mission, vehicle and engine system preliminary design (engineering and cost) data as Mars tripropellant engine technology matures.

SPECIAL FACILITIES/COMMENTS:

- None.

Title: 13. Preliminary Mars In Situ Propellant Mission/Vehicle/Engine System Design Studies

- SCHEDULE/COST -

ACTIVITY	YEARS FROM GO-AHEAD				
	1	2	3	4	5
1. Mission Studies					
2. Vehicle System Studies					
3. Engine System Studies					
ESTIMATED COST * ($K)	300	300	500	750	750

* Total Estimated Cost in 1992 Dollars ($K) = 2,600

TOR29J/341

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13. ABSTRACT *(Maximum 200 words)*

Recent studies have shown that there can be substantial advantages in using in situ propellants for fast transfers to, and explorations of, Mars when compared to chemical systems that use Earth-based propellants. Using vehicles that are powered by systems that use Martian resources has the potential to reduce low-earth-orbit mass requirements as well as increase mobility on the surface of Mars. A single propulsion system that can use two or more candidate propellant combinations, such as LOX/H_2, LOX/CH_4 and LOX/CO, could best leverage this exploration option. Design of such a propulsion system is challenging due to its requirements that it be inherently compatible with numerous candidate propellants, as well as operate efficiently over a large range of conditions.

A study was undertaken that identified and characterized promising chemical propulsion system designs that use two or more of the following propellant combinations: LOX/H_2, LOX/CH_4 and LOX/CO. Propulsion system requirements were established and expander and gas generator cycle tripropellant LOX-cooled engine systems were defined that used as much common hardware as possible. Overall mission impacts were quantified and Mars transfer vehicle propellant tank design strategies were evaluated. Critical propulsion system technologies and corresponding maturation plans are identified that are required to support development of such systems.

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