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**LUNAR SURFACE BASE
PROPULSION SYSTEM STUDY**

VOLUME 1: FINAL REPORT

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ABSTRACT

The efficiency, capability, and evolution of a lunar base will be largely dependent on the transportation system that supports it. Beyond Space Station in low Earth orbit (LEO), a Lunar-derived propellant supply could provide the most important resource for the transportation infrastructure. The key to an efficient Lunar base propulsion system is the degree of Lunar self-sufficiency (from Earth supply) and reasonable propulsion system performance. Lunar surface propellant production requirements must be accounted in the measurement of efficiency of the entire space transportation system. Of all chemical propellant/propulsion systems considered, hydrogen/oxygen (H/O) OTVs appear most desirable, while both H/O and aluminum/oxygen propulsion systems may be considered for the lander. Aluminized-hydrogen/oxygen and Silane/oxygen propulsion systems are also promising candidates. Lunar propellant availability and processing techniques, chemical propulsion/vehicle design characteristics, and the associated performance of the total transportation infrastructure are reviewed, conceptual propulsion system designs and vehicle/basing concepts, and technology requirements are assessed in context of a Lunar Base mission scenario.

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FOREWORD

This document represents Volume 1, the Final Report of the Lunar Surface Base Propulsion System Study, Contract No. NAS9-17468. Volume 1 details the study analyses and results. Volume 2 is the Lunar Base Propellant Manual which comprises the initial compilation of data for processing and use of Lunar-derived propellants. The contract effort was initiated on 15 January 1986 and continued through 15 February 1987.

The study was conducted by the Astronautics Corporation of America - Technology Center in Madison, Wisconsin. Aerojet TechSystems of Sacramento, California was a subcontractor contributing various propulsion and propellant analyses. Additional contributions were made by the Engineering Mechanics, Nuclear Engineering, and Chemistry Departments of the University of Wisconsin-Madison.

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1.0 INTRODUCTION AND CONCLUSIONS

The moon is relatively close on interstellar and even on solar system scales. However, on the scale of existing space transportation infrastructure, travel to the moon is very resource intensive and thus costly. With our present Earth-based transportation system, the transport of useful payload to the Moon requires that all equipment, consumables, and man-power be delivered from Earth. Use of Lunar resources to provide propellants provides two major assets to support this transportation burden: (1) a Lunar "gas station" to refuel vehicles returning to Earth, and (2) another source of propellant that can be delivered to Earth orbit. The first reduces the size, mass, and cost of Earth-Moon transportation vehicles. Because of the lower gravity on the Moon, the second asset may reduce the delivery cost of propellant requirements in Earth orbit. Much less energy is required to transport mass from the moon to low Earth orbit (LEO) than from Earth to LEO.

The Apollo missions enabled man to briefly visit the Moon, but such a transportation system must be enhanced if routine, long-term, Earth-Moon missions are to be affordable. The development of the Space Shuttle and the planned development of the Space Station are the first steps. Key goals of the Lunar-derived propellant production and associated Lunar propulsion systems in this study are to reduce the cost and increase the ease of transportation to the Lunar surface. In future decades, the 1980s may be considered the turning point in man's space exploration if Lunar-related analyses, studies and experiments are supported. The "extraterrestrial imperative", as worded by Krafft Ehricke, will have begun with the development of the low Earth orbit Space Station and a multi-national commitment to space.

A key stepping stone in development of the Earth-Moon transportation system is the Moon itself. Lunar resources and relatively low Lunar gravity provide an excellent environment for transportation base, propellant supply, science, industrial manufacturing/processing, and research nodes external to the Earth. However, any Lunar activity such as propellant processing to support the transportation system requires resources (e.g., equipment, machinery, facilities, and consumables). Therefore, self-sufficiency is the ultimate goal of any extraterrestrial activity. Key drivers for processing Lunar-derived propellants include minimal Earth-derived resources, Lunar availability of processing raw materials, simplistic recycling of any consumed materials, low power/thermal requirements, and a high acquisition efficiency of the propellant ingredients. Another key driver of Lunar propellant processing is the value of the biproducts. In many cases, a single product (such as oxygen) is desired; however, many process techniques exist that will yield not only oxygen but also metals, fuels, and metal/silicate oxides in addition to oxygen with only a minor additional resource burden. Prior to actual processing, beneficiation techniques could be coupled with solar wind gas extraction techniques to derive the existing volatiles such as hydrogen, helium, and nitrogen from the Lunar regolith.

This report addresses concepts of Earth-Moon transportation and the use of Lunar resources to produce propellants for those concepts. The overall objective of the study was to address a wide range of transportation options, and to identify economical alternatives for transportation between low Earth orbit and the Lunar surface. The approach to this study included five tasks, shown in Figure 1-1 with task interrelationships. Task 1 identified and analyzed the propellants, their sources, and the resource requirements involved in processing those propellants. Task 1 results are discussed in Section 2.0 of this report. Task 2 (Section 3.0) involved designing and analyzing vehicle systems, including the propulsion and vehicle subsystems that use these propellants. Task 3 (Section 4.0) involved assessing propellant processing techniques and vehicle families in an overall Lunar surface base mission model and scenario to develop life cycle costs and required mass flow from the Earth. Task 4 (Section 5.0) evaluated the technologies of propellant processing techniques, propulsion systems and vehicle systems. The scope of the effort is shown in Figure 1-2. Chemical vehicle propulsion and propellant supply techniques were emphasized in this study. Non-chemical vehicle propulsion and propellant supply techniques may have an equivalent or a supportive role in the total Earth-Moon transportation infrastructure and should be addressed in a future study.

The transportation system needed to support a Lunar surface base is not far beyond the current state-of-the-art. Lunar resources can be acquired to supplement and improve the efficiency of the transportation infrastructure in many ways. From a transportation standpoint, Lunar oxygen acquisition stands highest on the list of initial objectives for any Lunar surface base activity. Production of Lunar oxygen for any chemical rocket propulsion system could save

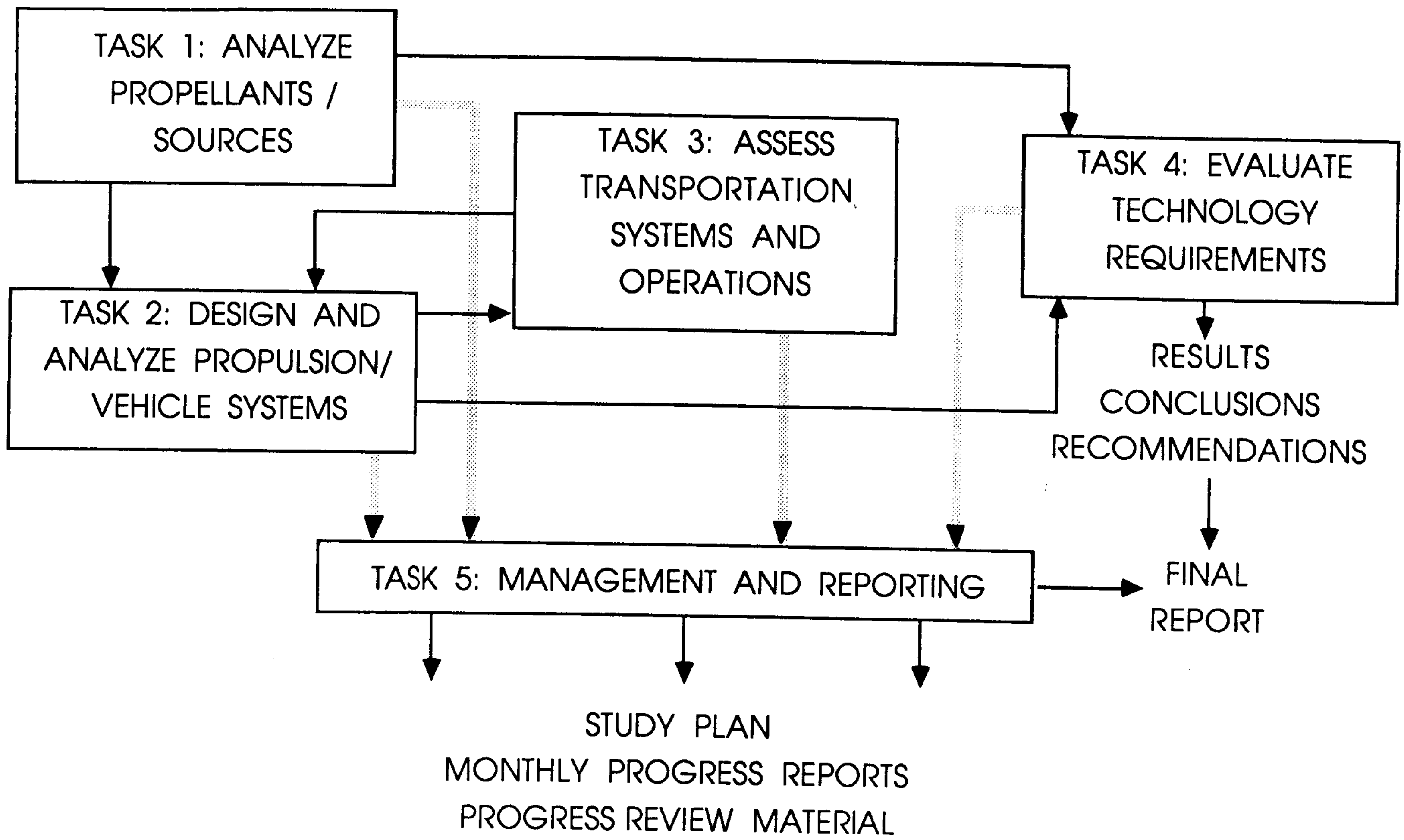

FIGURE 1-1. APPROACH

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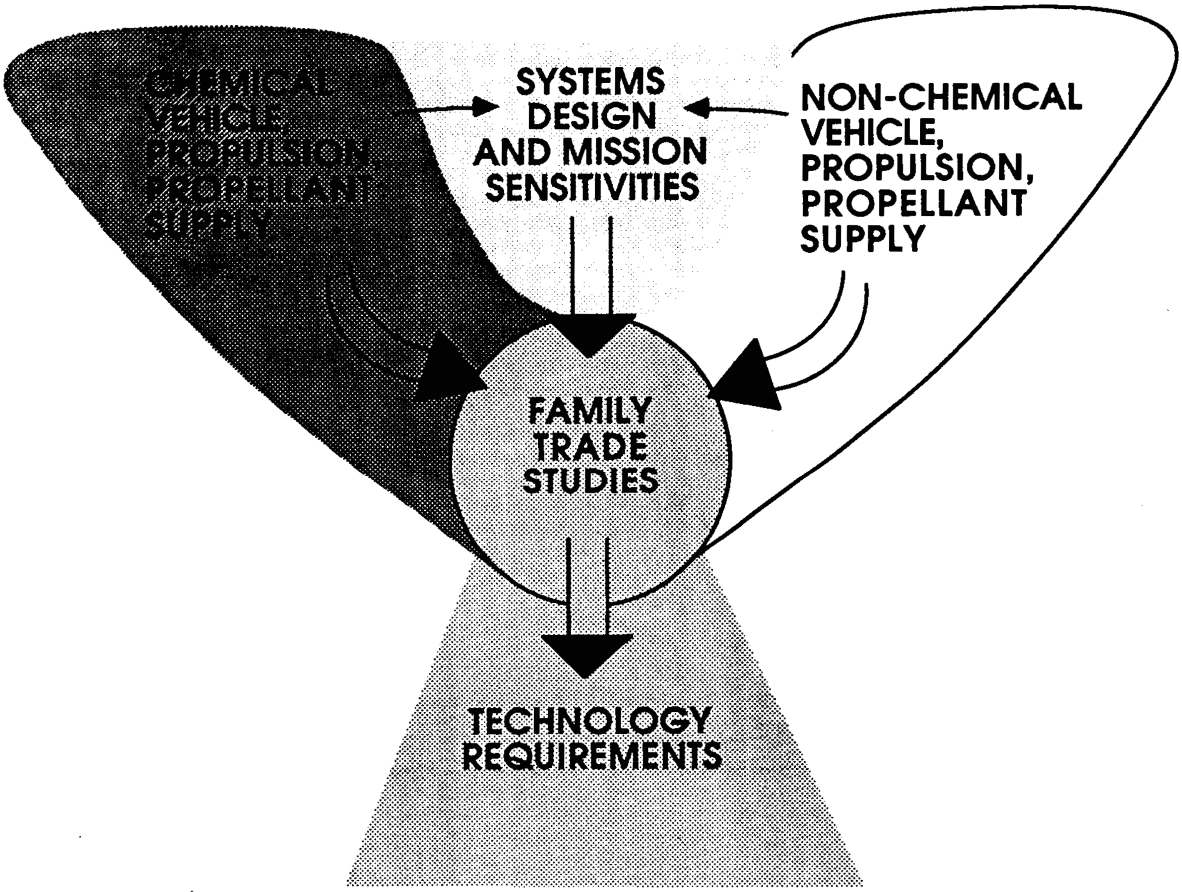


FIGURE 1-2. EARTH - MOON TRADES AND OPTIONS

about 50% of the required Earth launch mass. Rocket fuels also may be acquired on the Lunar surface. Hydrogen is available, but at very low levels of concentration. Other fuels are more plentiful, but they exhibit lower performance characteristics than hydrogen. The best Lunar fuels are hydrogen, aluminum, aluminumized-hydrogen, and silane in decreasing order of benefit to Lunar transportation. The use of Lunar hydrogen may save up to 30% of the Earth launch mass over the same system using Earth-supplied hydrogen. The combination of a hydrogen/oxygen OTV and a hydrogen/oxygen Lunar lander is the best propulsion option, if both hydrogen and oxygen can be acquired on the Moon. If Lunar hydrogen cannot be recovered in sufficient quantities then the use of other Lunar fuels (aluminum, silane) with Lunar oxygen in the lander appear attractive. Due to their performance limitations, those fuels do not appear attractive for the OTV. A hydrogen/oxygen OTV lander system with only Lunar oxygen available performs well but not as well as hydrogen/oxygen OTV coupled with a lower Isp aluminum/oxygen lander, if both Aluminum and oxygen are derived from the Launch surface.

Generally, the availability of Lunar oxygen reduces the burden of engine performance and engine/vehicle system technologies. Performance, cost, and Earth launch mass are less affected by variations in propulsion system specific impulse (Isp) and other design characteristics when Lunar oxygen is available.

The Lunar surface base mission scenario is possible with moderate extensions of existing technology. To achieve an economically attractive transportation system space basing is required. Also, either the OTV must be equipped with an efficient aerobrake, or Lunar oxygen must be utilized. Lunar base missions may be conducted without Lunar oxygen propellant delivery if a suitable aerobrake technology is developed. An aerobrake of specific mass much less than 40% of the re-entry weight is needed. Once Lunar oxygen becomes available the aerobrake is not as beneficial in reducing Earth launch mass, but remains economically attractive. To reduce Earth launch mass to a reasonable level, the aerobrake is a required technology without Lunar oxygen availability and Lunar oxygen is a required technology when aerobrakes are not present. If after Lunar propellant becomes available a market develops elsewhere, such as low Earth orbit, a very efficient aerobrake again may be very attractive.

Results of the propellant processing assessment indicate that hydrogen reduction by far is the best technique to supply Lunar oxygen, if that is the only Lunar material required. Lunar oxygen can reduce Earth launch mass by more than 50% of a given propulsion system over an entire mission model. However, a Lunar base will require a great deal more than just oxygen to be self-sufficient. Thus propellant processing techniques that are synergistic with respect to other needs should be considered. Processing techniques such as acid leach and vapor ion separation may be valuable to the entire Lunar surface base, not just the transportation system. For the processes considered, typical electrical energy consumed is on the order of 50 to 100 kilowatt hours for 10 MT of oxygen produced. System weights may range from 1 to 10 or more metric tons. Thermal energy requirements for 10 MT of oxygen produced can range from 10,000 kilowatt hours to hundreds of thousands of kilowatt hours, depending on the amount of soil being processed. Resources consumed for various processes may range from minute amounts for a process such as hydrogen reduction, to substantial amounts for acid leach and magma electrolysis processes.

Results from the vehicle propulsion analyses resulted in mass fractions ranging from .87 to .97. High mass fractions result from large vehicle concepts with requirements for extreme amounts of propellant. The baseline hydrogen/oxygen propulsion system was characterized by an Isp of 470 seconds and a mixture ratio of 5.5. Other propulsion systems addressed included aluminum/oxygen (hybrid and slurry), silane, and aluminumized-hydrogen/oxygen engine concepts.

Systems trades and sensitivities were addressed on the percent aerobrake mass, mixture ratios, specific impulses, payload masses, number of engines per vehicle, and the percent of Lunar propellant produced. Earth launch mass and total transportation costs were the figures of merit used in evaluating propellant production techniques and vehicle families. The lowest calculated total transportation cost, including Earth launch mass cost, DDT&E and production costs for the vehicles was approximately $6 billion.

Significant results were found in the trade-off analyses of aerobrakes, and high mixture ratio propulsion systems. High mixture ratios (greater than about 8) actually increase the amount of Earth-derived fuel and therefore are not beneficial. However, slight increases in mixture ratio (from 6 to less than 8) may still be beneficial in a Lunar-derived oxygen scenario. Aerobrakes can provide a significant amount of savings in Earth launch mass. Once developed, an aerobrake reduces the Earth launch mass by approximately 25% when no Lunar oxygen is available; and reduces Earth launch mass by 15% when Lunar oxygen is available.

Results indicate no new technologies are definitely required to produce Lunar oxygen on the Moon. However, for a variety of reasons, new technology options should be explored. Other than space basing, no new technologies are needed for hydrogen/oxygen propulsion systems. Current technology work on aerobrakes will support aerobrakes for Lunar return OTVs. Nothing beyond the current technology work of NASA OTV studies is required for near-term Lunar base operation. New technologies identified within the scope of this study that should be pursued are: (1) new and synergistic propellant processing techniques concentrating on consumable recycling; (2) space servicing/basing and propellant supply operations required for a refurbishable and reusable OTV and lander system; and (3) aluminum, silane fueled propulsion system feasibility studies and experiments.

2.0 LUNAR BASE PROPELLANT ALTERNATIVES

One of the major objectives of this study was to identify and characterize chemical propellants that could be practically produced on the Lunar surface from raw Lunar materials. This section of the report (which correlates to Task 1, "Analyze Propellants/Sources") identifies the full range of propellant sources from Lunar resources and Space Shuttle scavenging, and defines the processing techniques and requirements of producing such propellants. An initial list of propellants was developed and later refined based on considerations including: Lunar resource availability, production technique capabilities, storage and handling, and predicted performance when used in a propulsion system. Of the propellants investigated, oxygen was the most valuable oxidizer and Lunar-derived propellant. Hydrogen and aluminumized-hydrogen were the most valuable fuels identified. Silane and Aluminum were also addressed as promising Lunar-derived fuels but did not prove to be as beneficial as the first two fuels mentioned.

This task was scoped to include all aspects of propellant production, but largely concentrated on the actual chemical production techniques which are the drivers of raw material mining and preprocessing requirements. Very efficient, self-sufficient processes have been developed for a single element such as oxygen (e.g., Hydrogen Reduction process). However, production of a single product yield may not be as valuable in the context of an overall Lunar base scenario as a process that yields many useful products. "Synergistic processing" of multiple products may be used to produce both oxidizers and fuels. Synergistic processing is discussed in more detail in Section 2.5.

The identification of propellants was extensive in terms of liquid propellant candidates. Propellants for solid and hybrid systems were not given as much effort. Criteria for the evaluation of the propellants, and the production, storage and performance estimates were developed independently. This criteria was used to initially screen out candidates that are extremely incompatible with a Lunar-based propulsion system. The candidates that made it through the initial screen were then evaluated in terms of their processing techniques, storage requirements and predicted performance when used in a propulsion system. The candidates that passed this screening were then recommended for further analysis in propulsion systems. Processing requirements and total resource consumption for propellant production were estimated on these propellant candidates for input into the system tradeoff studies (as reported in Section 4.0). Technology requirements for processing, storage and use of the propellants were identified for input into the technology development plan developed in Task 4 which is described in Section 5.0.

The remainder of this Section describes work done specifically on Lunar propellant production. Section 2.1 provides a brief overview of the Lunar resources. The propellant alternatives are discussed in Section 2.2. Section 2.3 describes processing techniques found to be the most favored for Lunar resource processing. Section 2.4 describes the evaluation criteria used for evaluation of the propellant/processing techniques, and Section 2.5 discusses the potential for synergistic processing and associated benefits.

2.1 Lunar Resource Characterization

Many material needs of a Lunar base may be satisfied using processed Lunar resources. Elements such as oxygen, aluminum, iron and silicon exist in the Lunar regolith. Concentrations of these major elements can be seen in Table 2-1. In addition to the elements listed in Table 2-1, many elements, such as solar wind gases, are present in the Lunar regolith in much smaller concentrations. The concentration of these minor elements can be seen in Table F.5 of Appendix F.

Generally, Lunar resources may provide many useful applications for support of a Lunar base scenario. Criswell and Waldron have noted many uses of potential Lunar products in their chapter on Lunar Utilization in Volume II of Space Industrialization (Mandell, 1985). These applications in addition to a few developed during this study are shown in Table 2-2.

For this study, propellant processing approaches were analyzed for processing mare regolith. Mare regolith was chosen because of the quantity available and its easy accessability. Composition of each of the four major constituents of the mare and concentrations of each Lunar ore in the mare regolith are shown in Table 2-3.

2.2 Propellant Candidates

Many propellant candidates have been researched, tested, or used here on Earth. Our initial listing of propellant alternatives was developed exclusive of Lunar material availability or other criteria to enable a broad-based look at

TABLE 2-1. MAJOR ELEMENTAL COMPOSITION OF LUNAR REGOLITH

Element	Mare Reolith	Highlands	Basin Ejecta
Ca	7.9	10.7	7.70
Mg	5.8	4.6	6.1
Fe	13.2	4.9	8.7
Al	6.8	13.3	9.8
Ti	3.1	--	
Si	20.4	21.0	21.8
O	41.3	44.6	43.3
S	0.1	0.072	0.076
K	0.1	0.078	0.24
Na	0.3	0.48	0.38

TABLE 2-2. POTENTIALLY USEFUL LUNAR MATERIALS

Material	Uses
OXYGEN	PROPELLANT BIOSUPPORT THERMAL FLUID FUEL FOR POWER
ALUMINUM	PROPELLANT STRUCTURAL MATERIAL ELECTRICAL CONDUCTOR CEMENT ADDITIVE (Al2O3)
MAGNESIUM	STRUCTURAL MATERIAL REFRACTORY MATERIAL (MgO)
TITANIUM	STRUCTURAL MATERIAL THERMAL CONDUCTOR
IRON	STRUCTURAL MATERIAL ELECTRICAL CONDUCTOR MAGNETIC MATERIAL
CALCIUM	CEMENT ADDITIVE (CaO)
SILICON & SILICATES	STRUCTURAL MATERIAL GLASSES CERAMICS FIBROUS MATERIALS PLASTICS COMPOSITES CEMENT ADDITIVES
HELIUM	NUCLEAR FUSION FUEL (He-3) THERMAL FLUID
HYDROGEN	PROPELLANT BIOSUPPORT (H2O) THERMAL FLUID FUEL
NITROGEN	PURGE GAS BIOSUPPORT
CARBON	STRUCTURAL MATERIAL BIOSUPPORT

TABLE 2-3. LUNAR ORE COMPOSITION IN LUNAR MARE REGOLITH

PYROXENE - 50% of Mare Regolith

$CaO \cdot SiO_2$

$MgO \cdot SiO_2$

$FeO \cdot SiO_2$

$Al_2O_3 \cdot SiO_2$

$TiO_2 \cdot SiO_2$

OLIVINE - 15% of Mare Regolith

$2MgO \cdot SiO_2$

$2FeO \cdot SiO_2$

PLAGIOCLASE or **ANORTHITE** - 20% of Mare Regolith

$CaO \cdot Al_2O_3 \cdot 2SiO_2$

ILMENITE - 15% of Mare Regolith

$FeO \cdot TiO_2$

propellant usage on the Moon. Screening criteria were then used to narrow down this list of propellant characteristics to a workable number. The rationale behind the initial criteria is as follows: (1) easy access necessary raw materials from Lunar resources; (2) processability of the propellants; and (3) minimal equipment for the processing of propellants. Having an adequate source of raw materials is a must in any category; therefore, a plus was required in that initial screening criteria for the propellant to be retained for further study. The operational and initial setup requirements for the process techniques are of secondary importance and were considered in the evaluation only after adequate raw material availability was determined. Figure 2-1 shows the initial list of propellant candidates which were screened out to a secondary list and finally to a third list for which processing techniques were investigated and more detailed evaluation of these propellants through developed goals and criteria will be discussed in Sections 2.3 and 2.4, respectively.

The initial screen shown in Figure 2-1 is based on: (1) Lunar resource availability of the elements within the propellant; (2) the operational processing requirements based on earth based production techniques; and (3) the initial setup for Lunar base processing. The list of propellant candidates was narrowed a second time based on vehicle and engine design information and the system tradeoff analysis described in Sections 3.0 and 4.0. These will be discussed in Section 2.4 under propellant rankings.

2.3 Propellant Processing Techniques

This section includes results of analyses of potential Lunar base propellant processing scenarios.

Each propellant processing technique involves:

1. Mining/Beneficiation of raw material
2. Preprocessing
3. Processing desired propellants/products
4. Separating and collecting desired propellants/products

The processing schemes considered include:

- Solar Wind Gas Extraction from Lunar regolith
- Hydrogen Reduction of Ilmenite
- Magma Electrolysis of Ilmenite

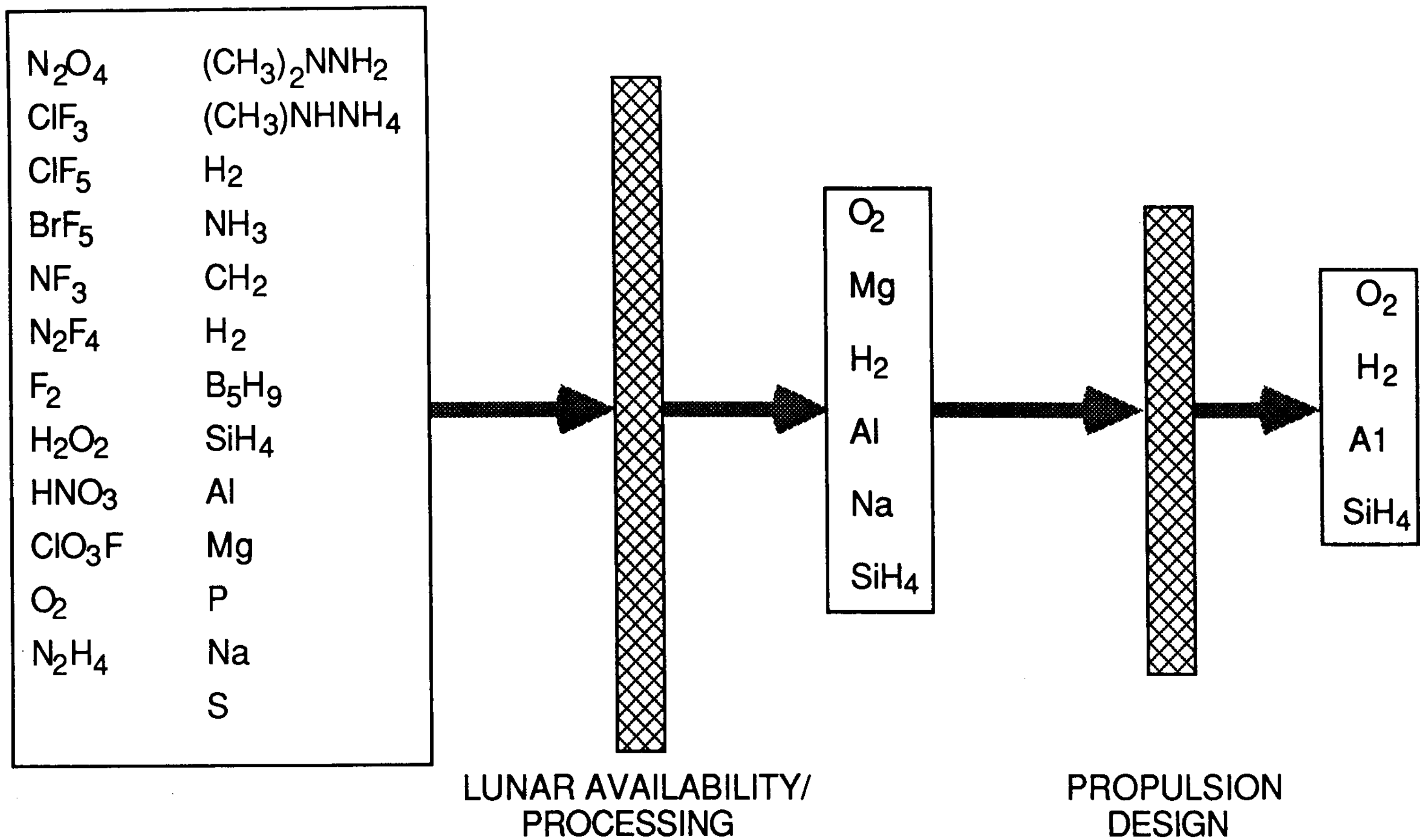


FIGURE 2-1. PROMISING CHEMICAL PROPELLANT CANDIDATES

- Carbochlorination of Anorthite
- HF Acid Leach
- Vapor-Ion Separation
- The Carbothermal Process

These processes were selected because of their potential for Lunar application and because a reasonable amount of data exists. Additional processes may be specifically designed for a given set of products. The initial analyses contained here will assist in development of future process design requirements. Table 2-4 summarizes the resource requirements for each of these processing candidates. Many of these processing scenarios involve similar mining and preprocessing requirements. Various mining and beneficiation techniques will be summarized in Section 2.3.1. Detailed propellant processing scenarios are discussed in Section 2.3.2.

2.3.1 Mining/Beneficiation and Preprocessing

Mining requirements vary from process to process depending on the regolith requirement for the processing scheme. Several mining scenarios for a Lunar base were analyzed in the 1977 Summer Study at NASA Ames Research Center (Williams, 1977). A mining system for small scale Lunar regolith collection may include a bladed scraper which transfers the regolith to a conveyor system. Trucks would then transport the regolith from stockpiles to a site where the regolith would be beneficiated. Another technique consisting of an automated system using a moderate-sized bucket-wheel excavator feeds the regolith to a shiftable conveyor system. Williams estimates that some components of a bucket-wheel excavator would require replacement every 150-200 hrs operating time. Typical mining operation would involve a shiftable conveyor running parallel to the strip to be mined. Strip sizes considered are 2m long and are mined to a depth of 2m (Williams, 1977). Mining techniques to date are not well defined but are similar to most processing techniques.

Once Lunar regolith is mined, it must be preprocessed, or beneficiated, to meet the requirements of the processing scheme. Some processes require the regolith to be sized to certain specifications. Other processes require separation of the Lunar ores in the mare; anorthite, ilmenite, olivine, pyroxene; or separation of individual metal silicates. Coarse sieves may be used to size Lunar regolith. Two other methods are used for more accurate sizing of regolith grains and separation of Lunar ores from the mare. These methods are

TABLE 2-4. PROPELLANT PROCESSING RESOURCE REQUIREMENTS

NOTE: All estimates based on production of10 MT O2 or 10 MT O2 per month

	HYDROGEN REDUCTION O2 ONLY	ELECTROLYSIS O2 ONLY	CARBOCHLOR-INATION O2 & Al	HF LEACH O2 & Al	VAPOR-ION SEPARATION O2 & Al	CARBO-THERMAL LOX & Si
ELECTRIC ENERGY	43,200 kWhr	46,500 kWhr	78,500 kWhr	119,800 kWhr	67,000 kWhr	43,200kWhr
THERMAL ENERGY	7,600 kWhr	62,700 kWhr	83,700 kWhr	41,600 kWhr	501,000 kWhr	168,500kWhr
SYSTEM WEIGHT EST.	1800 kg	980 kg	13,400 kg	11,700 kg	15,000 kg	12,000kg (c)
SYNERGISTIC POTENTIAL	POOR	POOR - FAIR	GOOD	GOOD	EXCELLENT	FAIR
(kg/10MT - O2) RESOURCES CONSUMED • MARE SOIL • ADDITIVES/CONSUMABLES	662,000 300	1,325,000 250 (a)	238,000 112,000 (b)	87,000 53,700 (a)	39,700 500 (b)	292,400,000 (d) 5,000
Al/Si PRODUCED	0	0	8.5 (Al)	5.9 (Al)	1.6 (Al)	5.0 (Si)
REACTANT RECOVERY POTENTIAL (%)	95	NA	75	70	NA	80
ACQUISITION EFFICIENCY [(O2 RECOVERED/ O2 AVAILABLE IN MARE) X 100]	3.7	1.7	10.2	27.8	61.0	0.016 (e)

(a) Caustic fluids may induce additional equipment replacement

(b) High temperatures may induce additional equipment replacement, also plasma sustenance required for selective ionization vapor-ion separation (e.g. Argon)

(c) Estimated using research conducted by Rosenberg, et. al.

(d) Calculation assumes beneficiation to Mg silicate, value would substantially decrease if other silicates are processed

(e) Acquisition efficiency for Si

electrostatic and magnetic separation. These separation techniques distinguish between mass-charge ratios, and other material properties.

2.3.2 Propellant Processing

Process methods include chemical reduction/oxidation, electrolysis, vaporization/ionization, pyrolisis, hydrolysis or a combination of these. Chemical reduction/oxidation methods will involve some consumption of Earth imports. Many of the needed Earth imports have the potential to be obtained from Lunar sources. Electrolysis methods have high energy requirements but often do not require chemical imports from Earth. Though terrestrial vaporization/ionization techniques have not been fully developed, processes using these techniques may prove to have the greatest potential for establishment of a Lunar base with minimal Earth support. Pyrolisis and hydrolysis are the simplest processing techniques and is utilized within most process scenarios.

2.3.2.1 Solar Wind Gas Extraction

A variety of useful Lunar resources are available as solar wind gases embedded in the top layer of the Lunar regolith. Some potentially available gases include hydrogen, helium, nitrogen, neon and argon. Neon and argon could be used in electric propulsion concepts. Helium-3 is a valuable commodity for fusion power cycles applicable to propulsion, however these propulsion concepts are not within the scope assessed in this report. These gases may be thermally released from Lunar regolith.

It is questionable how Lunar conditions will affect desorption rates and thermal release patterns or how the handling of Lunar samples will affect solar wind gas concentrations. The relative availability of solar wind gases in the mare regolith can be seen in the table below.

SOLAR WIND GAS EXTRACTION POTENTIAL

	g/gH_2
Hydrogen	1
Helium	0.5
Helium-3	0.0002
Nitrogen	1.7
Neon	0.05
Argon	0.01

Thermal release patterns for most solar wind related species show peaks at 600C and 1200C (Blanford, 1982).

Extraction of the solar wind gases may be done in-situ. Figure 2-2 shows the process schematic. Extraction hardware must be mobile and easy to maintain so that it may traverse large regolith areas. The processing system would be moved to a specific mining site and would then heat the regolith to about 600C. Partial pressures of all gaseous components of the regolith would create a laminar gas flow into a collection subsystem. After the gases are collected, the gaseous mixture is passed through a series of condensers to separate the various species. The gases will be collected in reverse order of their boiling points. Boiling points for several solar wind gases are listed in the table below.

	Boiling Point (K)	Boiling Point (K)
H_2	20.2	36.5
He	4.2	7.6
N	77.4	139.3
Ne	13.5	48.8
Ar	87.3	157.1

Note: Values at STP

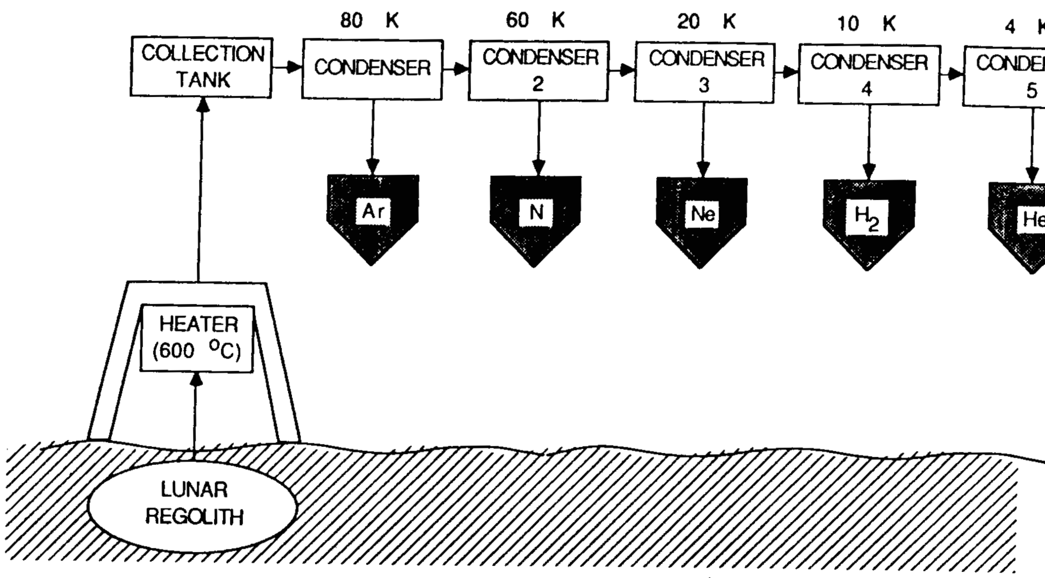
2.3.2.2 Hydrogen Reduction

Hydrogen Reduction is a very simple and efficient method of producing oxygen from Lunar ilmenite and is schematically shown in Figure 2-3. Lunar mare regolith is electrostatically sized and the larger grains are crushed. The fine powderized mare is magnetically separated to remove pyroxene and olivine, and then is electrostatically separated to remove anorthite and isolate ilmenite. The purity of the ilmenite produced can be improved by repeating the beneficiation process.

The ilmenite is then transferred to a reduction chamber where it is heated to process temperature in the range of 700 - 1000 C, the melting point of ilmenite is 1367 C. Hydrogen gas is then passed through the heated solid ilmenite inducting the following reduction reaction:

$$FeO \cdot TiO_2(s) + H_2(g) > Fe(s) + H_2O(g) + TiO_2(s)$$

If sufficient amounts of hydrogen gas are added, nearly 100% of all oxygen from FeO can be extracted. Excess hydrogen gas can be found in the water vapor and

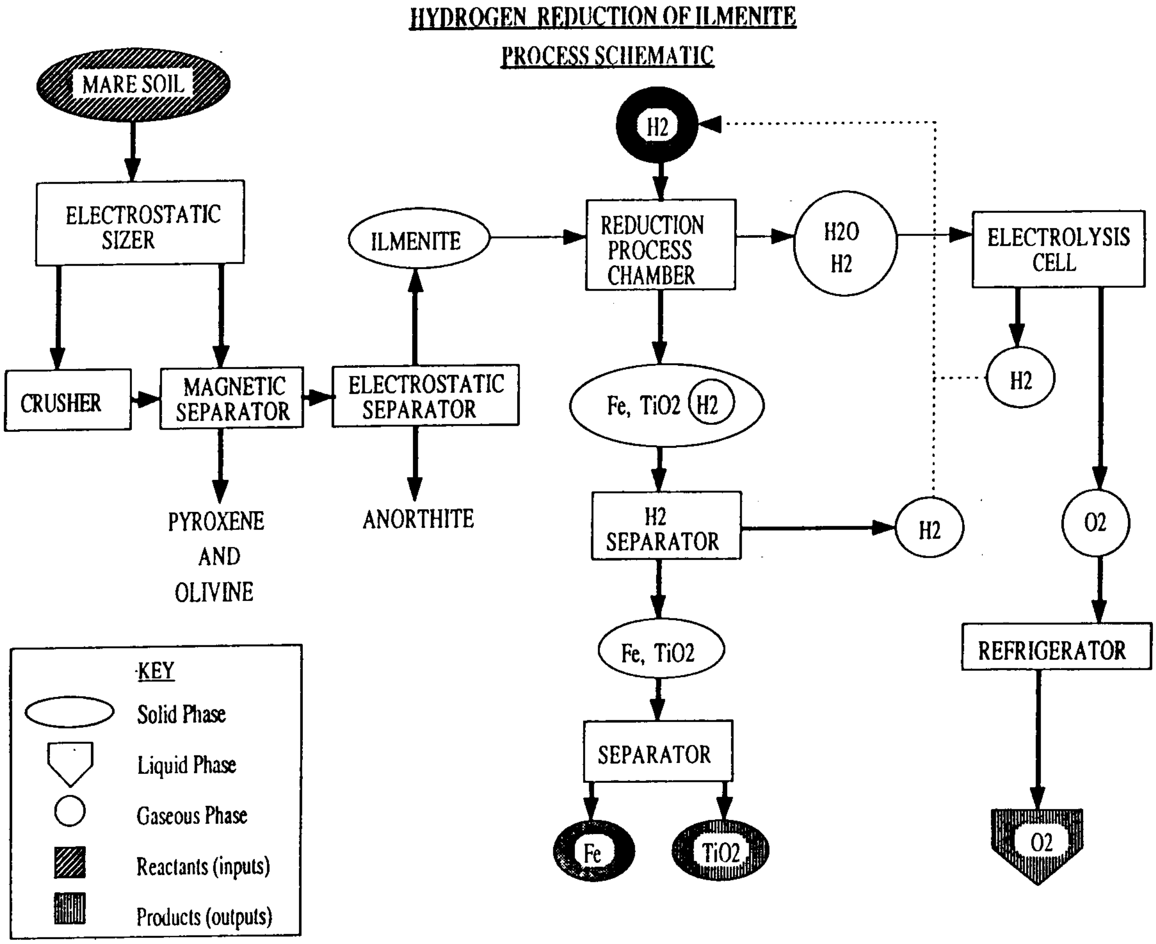


Lunar Mare Regolith Requirement
24.2 MT

Products
1325g Hydrogen
690g Helium
0.24g Helium-3
2310g Nitrogen
67g Neon
19g Argon

10 MT Oxygen

FIGURE 2-2. SOLAR WIND GAS EXTRACTION



Lunar Mare Regolith Requirement
662.2MT

Reactant Requirement
0.31MT Hydrogen with 95% recovery potential

Energy Requirement
1. Thermal
7,639 kWhrs
2. Electrical
43,200 kWhrs

Equipment Weight
1.83 MT

Reactant Recovery Potential
(based on assumptions stated in Appendix B)
95%

Aquisition Efficency
[(O2 recovered / O2 available in Mare) x 100]
3.7%

Products
10 MT Oxygen

FIGURE 2-3. HYDROGEN REDUCTION

interstitially trapped in the solid residue. Interstitially trapped hydrogen can be removed by heating the residue to melting temperature.

Water vapor with any excess hydrogen is electrolyzed to reclaim the hydrogen and isolate O_2. Approximately 95% of all hydrogen can be recovered by electrolysis and heating the residue. Oxygen gas is condensed and stored as liquid O_2. The solid residue may be further processed if Fe or TiO_2 are desired.

The advantages of this process are:

- Input requirements are low
- Hydrogen can be recycled without further chemical reactions
- The required beneficiation process is well understood
- Process has been proven in lab
- Relatively low process temperatures
- There is potential to obtain Lunar hydrogen which would eliminate need for Earth based consumed imports
- High process efficiency.

The disadvantages are:

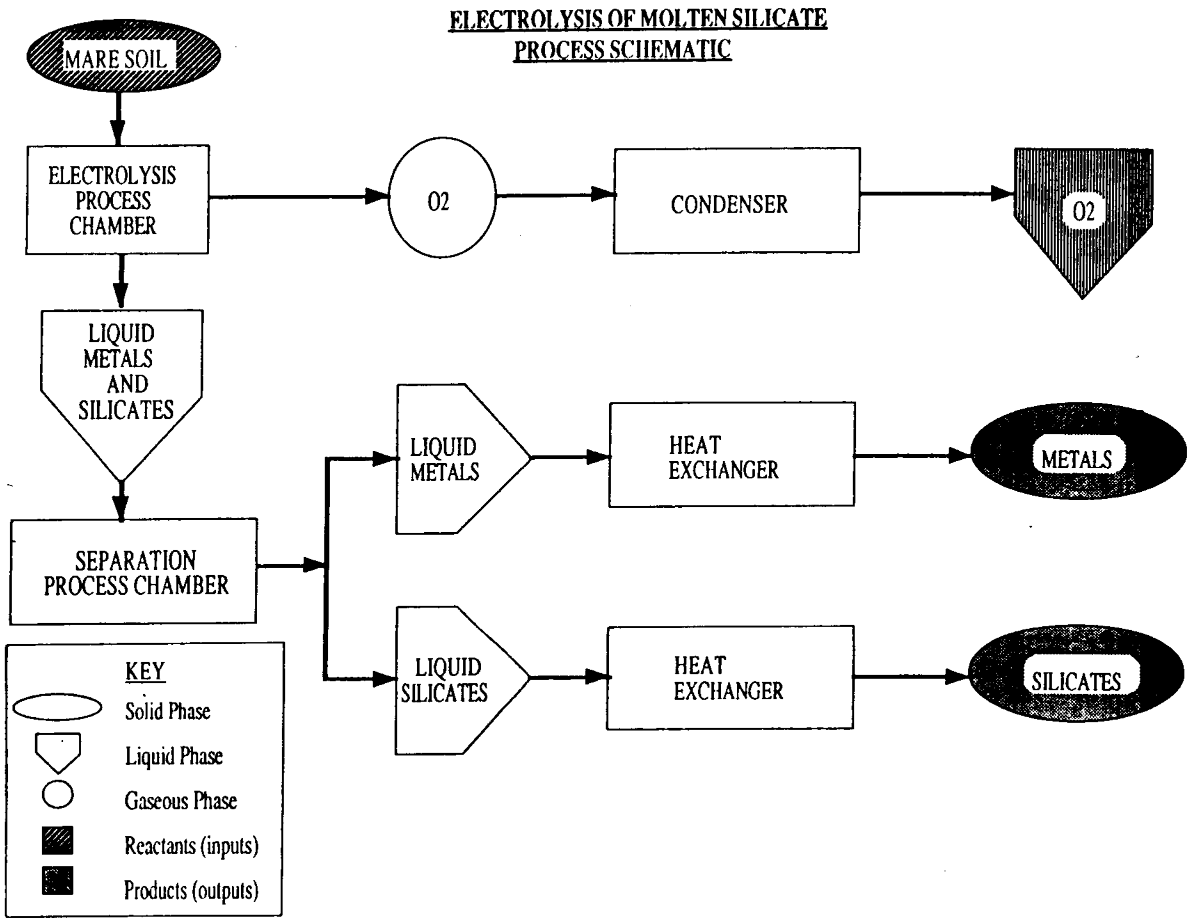
- Silicates not removed during beneficiation may cause extensive corrosion
- The kinetics of the reaction are slow
- Continuous processing has not yet been demonstrated in lab.

2.3.2.3 Magma Electrolysis

Magma Electrolysis is a second method which may be used to process oxygen from Lunar ilmenite and is shown schematically in Figure 2-4. Lunar mare regolith is magnetically separated to remove pyroxene and olivine, then electrostatically separated to isolate ilmenite. As with hydrogen reduction, repeating beneficiation will improve the purity of the ilmenite.

The ilmenite is then heated to just above its melting temperature, 1367 C. The liquid ilmenite is then transferred to an electrolysis cell where electrodes are placed across the melt and a potential difference is applied. The reactions can be represented by:

$$FeO\cdot TiO_2(s) + \text{heat} > FeO\cdot TiO_2(l)$$
$$FeO\cdot TiO_2(l) + \text{electrical energy} > Fe^{+2} + O^{-2} + FeO\cdot TiO_2$$



Lunar Mare Regolith Requirement
1325 MT

Energy Requirement
1. Thermal
62,705 kWhrs
2. Electrical
92,897 kWhrs

Equipment Weight
0.98 MT

Acquisition Efficiency
[(O2 recovered / O2 available in Mare) x 100]
1.7%

Products
10 MT Oxygen

FIGURE 2-4. MAGMA ELECTROLYSIS

Iron gathers at the cathode where it sinks to the bottom of the melt. Oxygen is released from the melt at the anode where it is condensed and stored as liquid O_2.

As the iron is removed from the melt, the conductivity drops and electrolysis stops. It is estimated that 50% of all oxygen from FeO can be extracted before electrolysis stops. The liquid ilmenite residue should then be removed and the process repeated. To improve yields using magma electrolysis, FeO in the already processed liquid ilmenite may be condensed out and slowly added to the electrolysis chamber to maintain more constant conductivity in the melt.

The advantages of this process are:

- o No Earth reactants are needed if fluxes are not used
- o Beneficiation required is well understood
- o Hardware requirement is very low.

The disadvantages are:

- o Poor process efficiency and yields
- o Any silicates not removed during beneficiation will cause extensive corrosion and electrode replacement will be required unless new
- o Mare regolith requirements are very high.

2.3.2.4 Carbochlorination

Carbochlorination is a fairly complicated process that may be used to produce oxygen and aluminum and is shown in Figure 2-5. Lunar mare regolith is magnetically separated to remove ilmenite, then electrostatically separated to isolate anorthite. The anorthite is then transferred to the carbochlorination unit where it is heated to process temperature in the range of 675 - 770 C. If process temperature exceeds 772 C, $CaCl_2$ melts and alters the thermodynamics of the carbochlorination reactions. The reactions within the carbochlorination unit are:

$$CaO(s) + C(s) + Cl_2(g) > CaCl_2(s) + CO(g)$$

$$SiO_2(s) + 2C(s) + 2Cl_2(g) > SiCl_4(g) + 2CO(g)$$

The total reaction can be represented by:

$$CaO \cdot Al_2O_3 \cdot 2SiO_2(s) + 8C(s) + 8Cl_2(g) > 2A_1Cl_3(g) + CaCl_2(g) + 2SiCl_4(g) + 8CO(g)$$

After the carbochlorination reactions are completed, the gaseous products, metal chlorides, salts and CO, are passed through a series of condensers. The first

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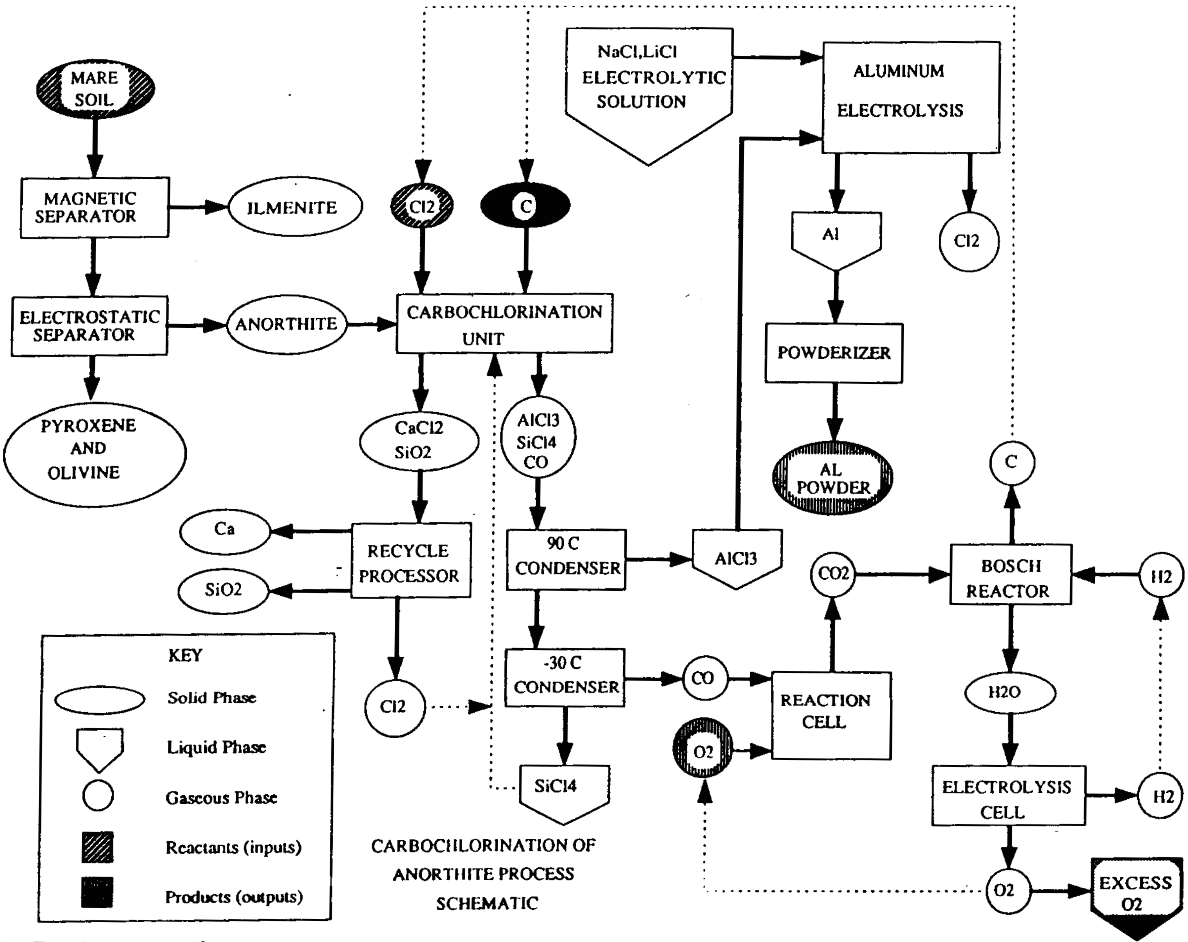

CARBOCHLORINATION OF ANORTHITE PROCESS SCHEMATIC

I. Production of O2
II. Production of O2 and Al

Lunar Mare Regolith Requirement
- I. 238 MT
- II. 238 MT

Reactant Requirement
- I. 95.8 MT Chlorine with 40% recovery potential
  16.3 MT Carbon with 100% recovery potential
- II. 95.8 MT Chlorine with 73% recovery potential
  16.3 MT Carbon with 100% recovery potential

Energy Requirement
1. Thermal
   - I. 82,245 kWhrs
   - II. 83,716 kWhrs
2. Electrical
   - I. 43,200 kWhrs
   - II. 78,561 kWhrs

Equipment Weight
- I. 12.45 MT
- II. 13.4 MT

Reactant Recovery Potential
(based on assumptions stated in Appendix B)
- I. 50%
- II. 75%

Acquisition Efficiency
[(O2 recovered / O2 available in Mare) x 100]
- I. 10.2%
- II. 10.2%

Products
- I. 10 MT Oxygen
- II. 10 MT Oxygen
  8.5 MT Aluminum

FIGURE 2-5. CARBOCHLORINATION

condenser removes enough heat to bring the temperature to 90 C, and $AlCl_3$ is removed as a liquid. The second condenser is cryogenic and brings the remaining gas to a temperature of -30 C. This second condenser removes all CO gas and the remaining liquid salt, $SiCl_4$, is cycled back into the carbochlorination unit. As the concentration of $SiCl_4$ builds up in the carbochlorination unit, it reacts w/CO and reverses the carbochlorination of SiO_2 reaction. This frees up chlorine atoms from $SiCl_4$ and C atoms from CO. The CO gas can then be converted to C and O by the Bosch reactor. It is estimated that approximately 50% of all chlorine from $SiCl_4$ could be recovered. To reclaim additional chlorine from $SiCl_4$ and the $CaCl_2$ precipitate, the residue in the carbochlorination unit must be further processed.

Advantages of this process are:

- o Low temperature requirement
- o Process is well understood for terrestrial applications
- o The Alcoa process can be used and is the most efficient way to extract Al,
- o Hardware for the Bosch Reactor may be used for many Lunar base applications,
- o There is potential to obtain Al in powderized form for use as propellant.

Disadvantages are:

- o Recycling of C and Cl_2 may require as much or more hardware than the hardware needed for extraction of desired resources,
- o For extraction of 1 kg Al, 10 - 20 kg of Cl_2 are required,
- o Many C-Cl-O combinations may be thermodynamically favored which necessitates monitoring the O_2 and Cl_2 fugacities during the reactions in the carbochlorination unit,
- o Hot chlorine gas is extremely corrosive and creates a maintenance problem and highly corrosion resistant materials must be used for the carbochlorination unit,
- o This process has little potential to run independent of Earth support for Lunar base applications.

2.3.2.5 Acid Leach

Acid Leach is a process that can be generally used to process metallic silicates for metals, silicon, and oxygen. Figure 2-6 shows resource require-

HF ACID LEACH

I. Production O2
II. Production of O2 and Al
III. Production of O2, Al and Mg

Lunar Mare Regolith Requirement
- I. 23.6 MT
- II. 87.3 MT
- III. 60.2 MT

Reactant Requirement
- I. 27.5 MT HF with 0% recovery potential
- II. 27.5 MT HF with 45% recovery potential
  26.2 MT NaOH with 100% recovery potential
- III. 30.6 MT HF with 53% recovery potential
  18.2 MT NaOH with 100% recovery potential
  5.42 MT CaO with 90% recovery potential
  1.35 MT Si with 90% recovery potential

Energy Requirement
1. Thermal
   - I. 26,663 kWhrs
   - II. 129,860 kWhrs
   - III. 106,770 kWhrs
2. Electrical
   - I. 43,200 kWhrs
   - II. 119,813 kWhrs
   - III. 96,419 kWhrs

Equipment Weight
- I. 2.85 MT
- II. 11.73 MT
- III. 15.93 MT

Reactant Recovery Potential
(based on assumptions stated in Appendix B)
- I. 0%
- II. 70%
- III. 73%

Acquisition Efficiency
[(O2 recovered / O2 available in Mare) x 100]
- I. 100%
- II. 28%
- III. 40%

Products
- I. 10 MT Oxygen
- II. 10 MT Oxygen
  5.9 MT Aluminum
- III. 10 MT Oxygen
  4.7 MT Aluminum
  2.35 MT Magnesium

FIGURE 2-6. HF ACID LEACH

ment sensitivities to the number of desired products. Figure 2-6a is a schematic of the main leach reaction. Figure 2-6b shows steps for reclaiming fluorine from TiF_4, CaF_2, and $FeSiF_6$. Figure 2.6c shows process steps for recovery of aluminum and magnesium from the respective metal fluorides.

Lunar mare regolith is leached with HF acid at 110 C. Leach reactions for processing bulk mare are:

1. $Al_2O_3 \cdot SiO_2 + 12\ HF > 2(AlF_3) \cdot H_2SiF_6 + 5H_2O$
2. $MgO \cdot SiO_2 + 6\ HF > MgF_2 \cdot SiF_4 + 3H_2O$
3. $FeO \cdot SiO_2 + 6\ HF > FeF_2 \cdot SiF_4 + 3H_2O$
4. $CaO \cdot SiO_2 + 6\ HF > CaF_2 \cdot SiF_4 + 3H_2O$
5. $TiO_2 \cdot SiO_2 + 8\ HF > TiF_4 \cdot SiF_4 + 4H_2O$

Reactions 1 and 2 above can represent leach reactions for processing beneficiated mare. The beneficiation required to separate the aluminum and magnesium silicates from the mare has not yet been determined.

The saturation point for dissolving of the Lunar ore is reached when the solution molar F:Si ratio reaches 5. During dissolution, precipitation of CaF_2 will lower the molar F:Si ratio while evaporation of SiF_4 will raise the ratio dissolution rate, acid should be added when the pH rises above 2. Leach reaction monitoring should ensure:

1. Solution Molar F:Si Ratio > 5
2. pH < 2

Processing from Al and Mg Silicates Al and Mg. Aluminum and magnesium silicates from the beneficiated mare are leached with HF. Leach reactions for the production of Al and Mg only is represented by Equations 1 and 2 above. Before each metal can be extracted, the metal fluorides must be separated from the fluorosilicates. Aluminum is obtained from reduction of AlF_3. Magnesium is obtained from processing MgF_2.

Aluminum may be obtained from AlF_3 by sodium reduction. Sodium is obtained from NaOH, which must be initially imported from Earth, but can potentially be Lunar derived, using Castner Electrolysis. In the Castner Electrolysis Cell, $Na^{+exponent}$ is discharged at the cathode and OH^{-1} is discharged at the anode. During the electrolysis, excess water must continuously be removed to prevent

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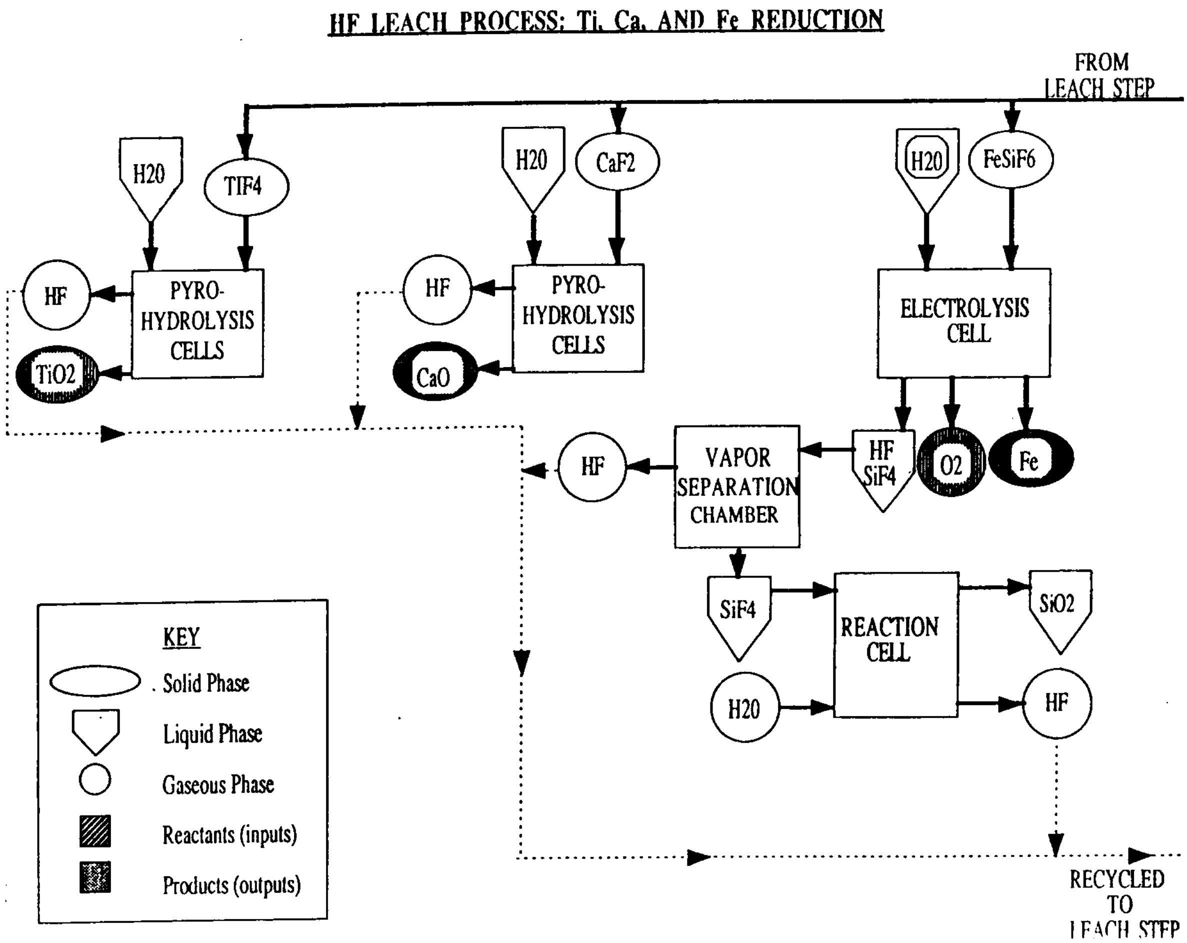

FIGURE 2-6a. HF ACID LEACH

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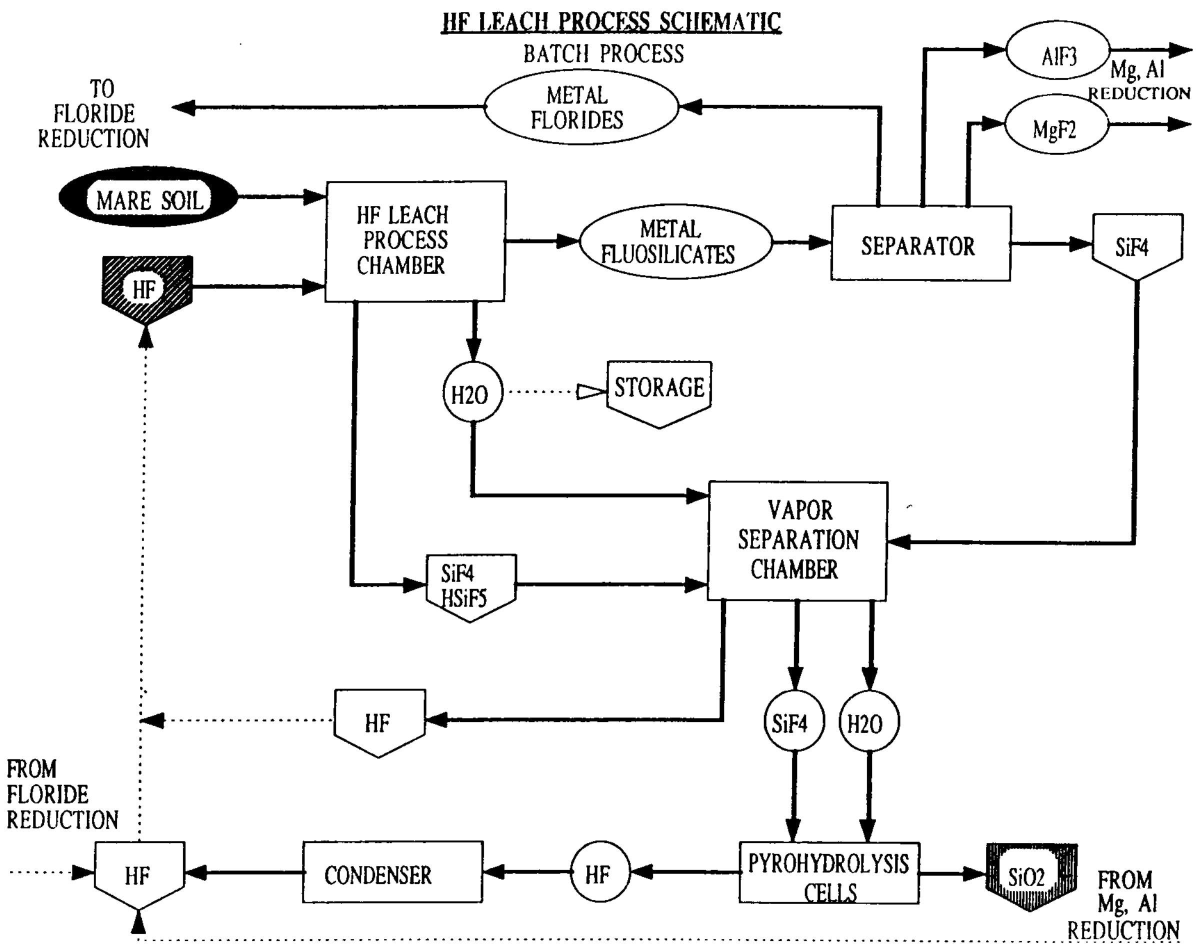

FIGURE 2-6b. HF ACID LEACH

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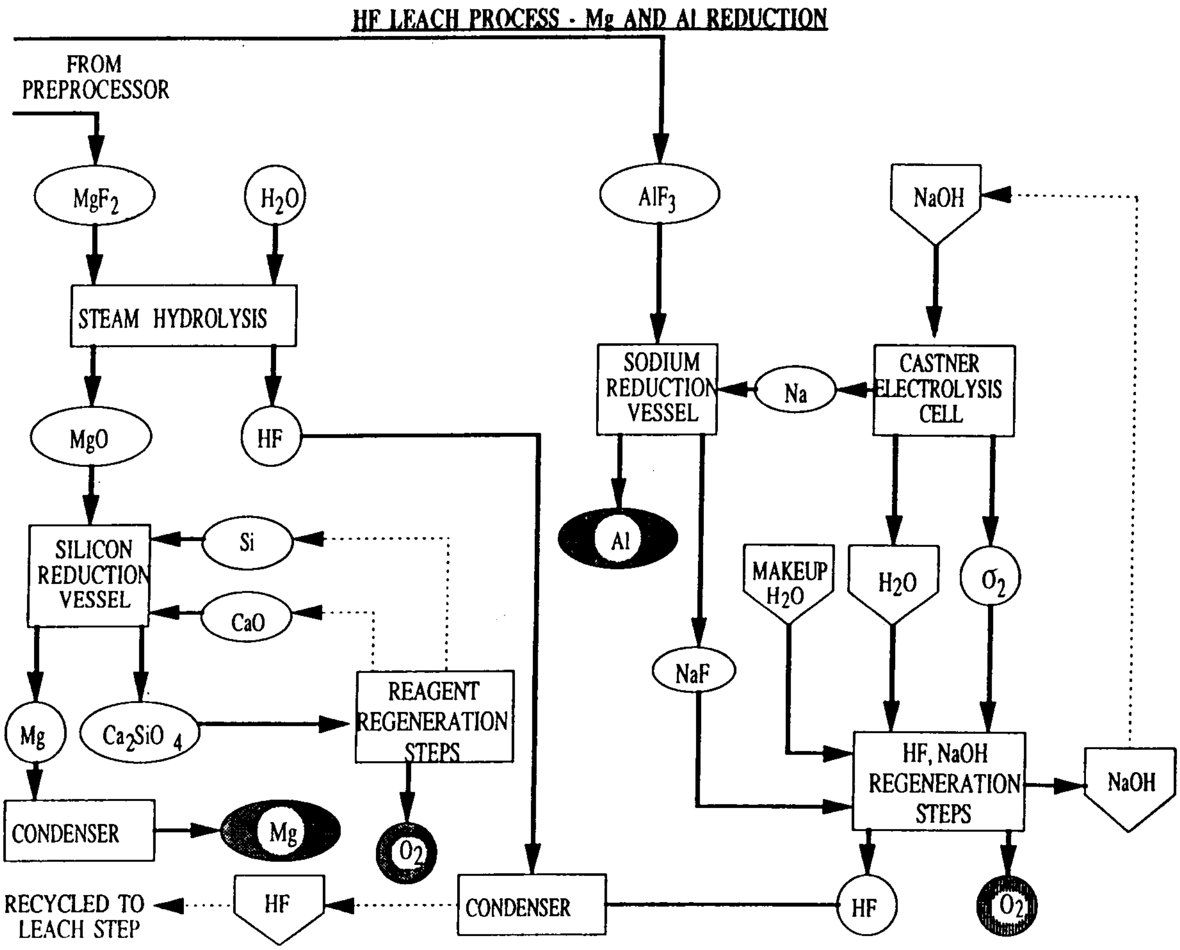


FIGURE 2-6c. HF ACID LEACH

unwanted electrolysis of water. Sodium is transferred to a reduction vessel where it reacts with AlF_3 forming Al and NaF. This reduction is carried out in steel reaction vessel at 900 C. The reaction can be represented by:

$$AlF_3(l) + 3Na(l) > Al(l) + 3NaF(l)$$

Aluminum is collected in liquid form.

Magnesium cannot be directly extracted from its metal fluoride. MgF_2 is first converted to MgO by steam hydrolysis. MgF_2 reacts with water at 1200 C forming MgO and HF. The reaction can be represented by:

$$HF_2(s) + H_2O(l) > MgO(s) + 2HF(l)$$

All required water was produced during the initial leach reactions. The HF formed is cycled back to the leach tank for further processing of lunar ores.

MgO is reduced by Si and CaO in an alloy steel reaction vessel at 1100 - 1200 C. The reaction can be represented by:

$$2MgO(s) + Si\ (s) + 2CaO(s) > 2Mg(l) + Ca_2SiO_4(s)$$

Mg distills into the cooler ends of the reaction vessel and is passed through a condenser to obtain solid Mg. Ca_2SiO_4, the precipitate of the reaction, can be recycled to reclaim Si and CaO.

Processing Unbeneficiated Mare Regolith. After the leach reactions, all metal fluorides must be separated from the fluorosilicates. AlF_3 and MgF_2 are processed in the same manner described above.

TiF_4 and CaF_2 react with water at elevated temperatures. These reactions can be represented by:

$$TiF_4(s) + 2H_2O(l) > TiO_2(s) + 4HF(l)$$
$$CaF_2(s) + H_2O(l) > CaO\ (s) + 2HF(l)$$

HF is cycled back to the leach tank for further processing of Lunar ores.

Iron is not separated from its fluorosilicate before processing. $FeSiF_6$ is electrolyzed with water. The reaction can be represented by:

$$FeSiF_6(s) + H_2O(l) + \text{electrical energy} = Fe(s) + 1/2O_2(g) + 2HF(l) + SiF_4(l)$$

Required amounts of water can be obtained from the initial leach reactions. Iron is obtained in solid form. Oxygen is collected as a gas and can be condensed to obtain liquid O_2. HF and SiF_4 exist together in a liquid solution. HF may be recovered by applying sufficient heat to release HF leach tank for further processing of Lunar ores. Water, obtained from the initial leach reactions, can be added to the liquid SiF_4 to reclaim fluorine. This reaction can be presented by:

$$SiF_4(l) + 2H_2O(l) > SiO_2(l) + 4HF(l)$$

The HF recovered here is cycled back to the leach tank.

The advantages of HF leach are:

- O_2, Al, Mg, Fe and, potentially, Ti and Ca can all be extracted
- Process is proven for terrestrial application
- Minimal new technology required for Lunar application.

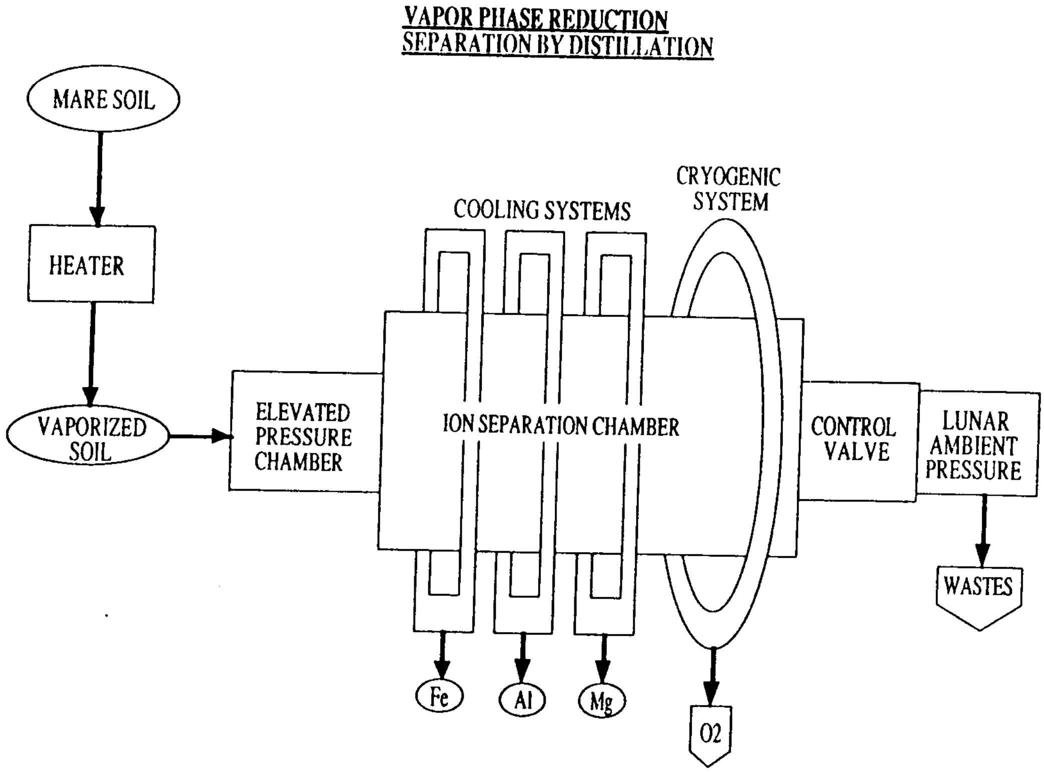
The disadvantages are:

- Requires extensive hardware for recycling of imported HF
- Electrical and thermal energy inputs are very high
- Desired beneficiation process has not been tested
- High maintenance requirement to maintain desired reactions in the leach tank,
- HF is highly reactive and may react to form hydrogen which is explosive under certain conditions. This hazard must be recognized in transporting and handling HF acid.

2.3.2.6 Vapor-Ion Separation

Vapor phase reduction is a relatively new concept in extractive metallurgy designed for space applications and represents, in this study, two major but similar methods of processing: (1) heating the soil to a vapor state and separating, and (2) injecting the soil into a plasma with subsequent separation. In this process, powdered mare regolith (< 100-micron particles) is vaporized and passed through a separation/collection chamber by a pressure gradient. Individual metals and oxygen can be separated with minimal to no raw material imports from Earth. Reducing agents can be used to prevent recombination of vapor constituents.

Three vapor-ion separation techniques have been suggested and are shown in Figure 2-7a through c. These are distillation, electrostatic and magnetic



I. Separation by Distillation
II. Separation Electrostatically
III. Separation Electromagnetically

Lunar Mare Regolith Requirement
24.2 MT

Energy Requirement
I. 34,000 kWhr/MT product
II. 44,000 kWhr/MT product
III. 96,000 kWhr/MT product

Products

I. Aluminum
Magnesium
Iron
Oxygen

II. Aluminum
Magnesium
Iron
Oxygen

III. Aluminum
Magnesium
Iron
Oxygen

FIGURE 2-7a. VAPOR-ION SEPARATION - RECOVERY BY DISTILLATION

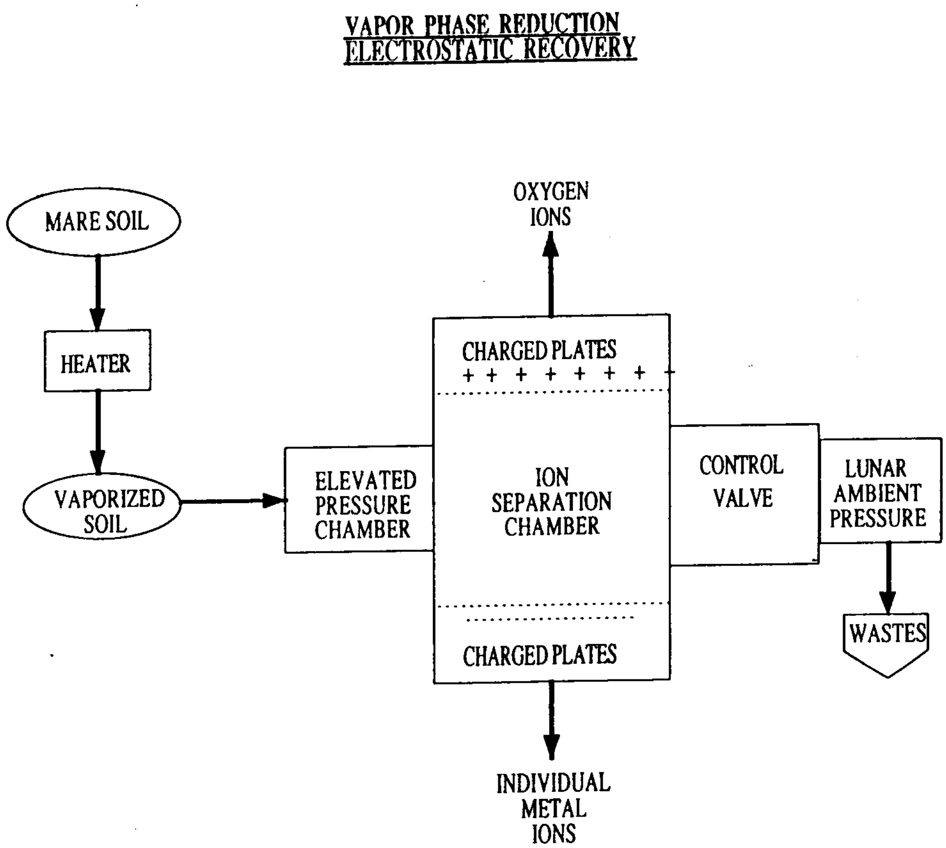


FIGURE 2-7b. VAPOR-ION SEPARATION - ELECTROSTATIC RECOVERY

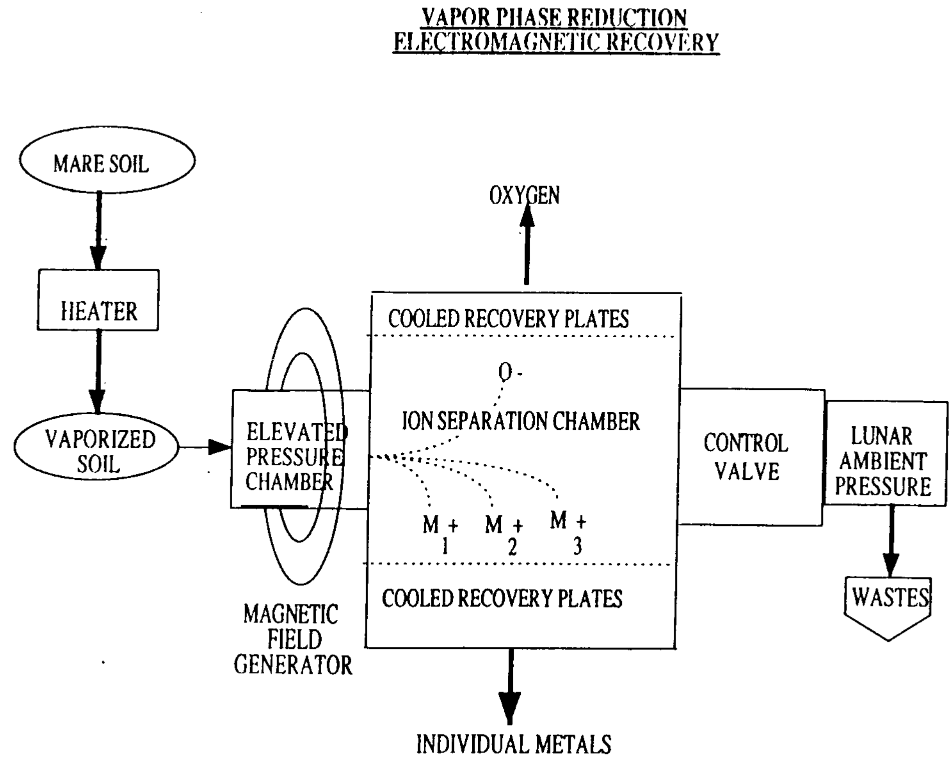


FIGURE 2-7c. VAPOR-ION SEPARATION - ELECTROMAGNETIC RECOVERY

separation.

Distillation. As the vapor is passed through the separation chamber, specific materials can be condensed and collected by selectively cooling. Melting points for some valuable resources in the mare regolith are:

O_2	-219 C	-362 F
Al	660 C	1220 F
Ti	1670 C	3038 F
Fe	1536 C	2797 F
Mg	649 C	1200 F

A separate cooling system is needed to extract each of these resources. To store oxygen, a cryogenic cooling system is required.

Electrostatic Separation. In this separation technique, charged plates are used to collect individual metals and oxygen. In the separation chamber, metal ions collect on the negative plate and oxygen is collected on the positive plate. Individual metals may be collected by altering the current to the charged plates. The current needed to collect a specific metal ion is based on the metal ion's mass/charge ratio.

Magnetic Separation. As the vapor enters the separation chamber, it is passed through a magnetic field. This field deflects the path of each vapor phase constituent according to its mass/charge ratio. Cooled recovery plates are positioned at distinct distances from the magnetic field to collect individual metals and oxygen.

General Discussion. Vapor phase reduction is more promising than chemical processing techniques in a Lunar base scenario because of its low consumable requirements. It uses the Moon's vacuum and high concentration of solar energy. Once all required hardware is transported to the Moon, further import requirements are negligible. Because Vapor-Ion Separation is an experimental technique, much additional information is needed before it can be selected for Lunar base application.

Vapor-Ion Separation faces two serious problems: process hardware required and the purity of the products obtained.

The purity of the products obtained can be affected by the recombination of vapor constituents before the collection process. Partial pressures of each constituent must be monitored or controlled to ensure a high-purity product. The problem of recombination is amplified when considering separation by distillation. Because the vapor is being cooled in stages, recombination occurs more readily and process efficiency will be less than process efficiencies of electrostatic or magnetic techniques.

One way to reduce the problem of recombination using electrostatic separation is to use plasma to vaporize the mare regolith. By passing the powderized regolith through a plasma at 8000 K, Al, Mg and Fe are ionized. These individual metals are separated and collected while oxygen and all unionized particles pass through the system. Plasma can be generated in a variety of ways:

Type of Generation	Power Req. (kW)	Working Fluid	Power Source Efficiency,%
D.C. Arc	150 - 200	Ar, N, H_2	90 - 95
3-Phase A.C. Arc	500 - 1000	Ar, N, H_2	95
High Freq. Arc	100 - 1000	No particular restrictions	30 - 80

(From Akashi's "Application of Thermal Plasma to Extraction Metallurgy and Related Fields")

Once the plasma is generated, it must be maintained using a magnetic field. An Anamar-type magnet, with a mass of about 400kg, requiring 10 MW, and producing a field strength of 2 Tesla, could be used.

The advantages of Vapor-Ion Separation Techniques are:

- The need for imports of raw materials from Earth is minimal
- There is no need for recycling
- The system uses the vacuum of the Lunar environment
- Has the potential to obtain more individual metals and oxygen with higher purity than chemical processing
- All products are extracted in one step which gives this processing techniques high potential for automation.

The disadvantages are:

- Temperatures to vaporize mare regolith are much greater than chemical processing techniques
- Hardware corrosion can be extensive if new materials are not found
- The process is extremely energy intensive
- Recombination of vaporized ions must be investigated
- More research is needed before extraterrestrial application.

2.3.2.7 Carbothermal Process

The Carbothermal Process has been developed by the Mobile Systems Department of Aerojet-General's Chemical Products Division for the manufacture of oxygen from Lunar materials and is schematically shown in Figure 2-8. In addition to oxygen, the Carbothermal Process isolates silicon when processing magnesium silicate. It is estimated that 10 MT oxygen and 10 MT Silane can be extracted from 300 MT mare regolith.

The Carbothermal Process involves four steps. The first step is the beneficiation of the mare regolith to isolate magnesium silicate ($MgO \cdot SiO_2$). To manufacture 10 MT O_2, approximately 50 MT $MgO \cdot SiO_2$ are required.

The second step involves the reduction of $MgO \cdot SiO_2$ with methane at a process temperature of 1625C.

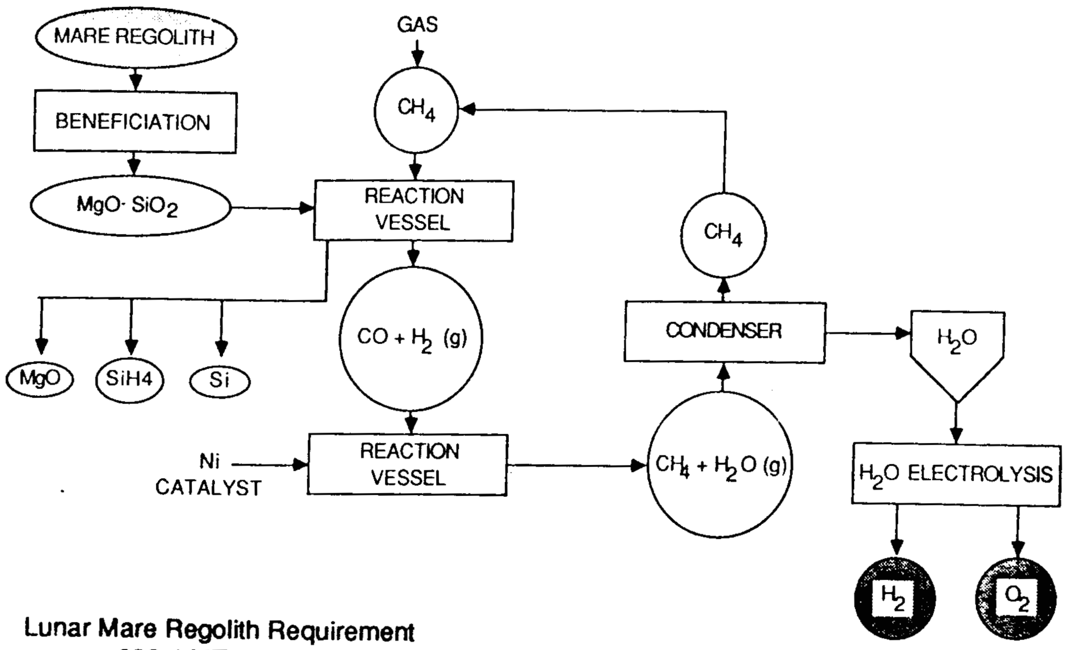
$$MgO \cdot SiO_2(l) + 2CH_4(g) > 2CO(g) + Si(l) + MgO(s) + 4H_2$$

The magnesium oxide precipitate is removed and can be stored for future magnesium or oxygen production. Product gases, CO and H_2, are collected and further processed to obtain oxygen. If 50 MT $MgO \cdot SiO_2$ is processed, 13-15 MT of silicon may be obtained as liquid Si.

The third step involves processing of product gases from the methane reduction. Carbon monoxide and hydrogen gases are combined at a process temperature of 250C.

$$2CO(g) + 6H_2(g) > 2CH_4(g) + 2H_2O(g)$$

Methane gas and water vapor are condensed to separate water from methane. Approximately 70% of the initial methane inputed is recovered here. In this step, hydrogen is the limiting reagent. The excess CO can be processed in a number of ways. First, CO can be processed with H_2 produced by electrolysis of



Lunar Mare Regolith Requirement
292.4 MT

Reactant Requirement
15.0 MT CH4

Energy Requirement
1. Thermal
168,453 kWhrs
2. Electrical
43,200 kWhrs

Reactant Recovery Potential
(based on assumptions stated in Appendix B)
66%

Acquisition Efficiency
[(O2 recovered / O2 available in Mare) x 100]
8%

Products
10 MT Oxygen
10 MT Silane
4.35 MT Silicon

FIGURE 2-8. CARBOTHERMAL PROCESS

water to recycle CH_4 and obtain additional O_2. Second, CO can be processed by itself to obtain carbon and oxygen. The advantage of the second alternative is that the hydrogen produced by electrolysis is free to be combined with elemental silicon to form silane, SiH_4. The decision of which way to process excess CO is made by weighing the value of producing silane to recycling methane.

The final step involves the extraction of oxygen from water produced in the third step. Water is electrolyzed to produce H_2 and O_2 gases at a process temperature of 75C.

$$2H_2O(l) = 2H_2(g) + O_2(g)$$

From processing 50 MT $MgO \cdot SiO_2$, 10 MT O_2 and 1.25 MT H_2 can be produced.

Silane, SiH_4, is obtained by combination of all hydrogen produced in electrolysis with silicon produced in the methane reduction step. From 50 MT $MgO \cdot SiO_2$, 10 MT SiH_4 may be produced and about 5 MT Si will be left over. This extra silicon may be stored for later use.

One use of the extra silicon is to reduce MgO to obtain magnesium. From the 50 MT $MgO \cdot SiO_2$, over 10 MT Mg are potentially obtainable. In addition to silicon, calcium oxide, CaO, is also needed to obtain Mg from MgO. Calcium oxide can be obtained from Lunar mare (over 10 wt% of mare regolith).

The Carbothermal Process can be potentially used to process other metal silicates besides $MgO \cdot SiO_2$. The metal silicates are reduced to metal oxides in the methane reduction step. Different processing steps must be taken to separate the metal from its metal oxide. To obtain aluminum, the aluminum oxide must first be transformed into AlF_3 or $AlCl_3$ before further reduction. The additional processing required to obtain elemental metals necessitates the combination of the carbothermal process with some part of another process if individual metals are desired.

The advantages of the Carbothermal Process are:

- o Process has been demonstrated for terrestrial application (by Aerojet)
- o Oxygen, hydrogen and silicon are produced synergistically
- o Little technology development needed for Lunar application.

The disadvantages are:

- o Large amounts of Mare (unless multiple silicates can be processed also) are processed
- o High temperature silicate environment creates corrosion problems
- o Less than 10% of all O_2 available in mare is obtainable
- o Methane gas or other hydrogen source must be continuously supply
- o Must be coupled to a subsystem of another processing scheme if individual metals are desired.

2.3.3 Separation and Collection

Separation and collection of the products may involve distillation or centrifuging. For a multi-phase mixture or a liquid mixture of components with varying densities, various specific constituents can be removed by centrifuge. For gaseous mixtures, specific constituents may be isolated in a distillation column or by condensers. Much research is currently being done on thermal management systems which will affect technologies involving liquefaction of gaseous species. Estimates of efficiencies of magnetic cooling versus fluid medium condensers show a weight savings of up to 3 times using magnetic cooling.

2.4 Propellant Evaluation

The assessment of Lunar-derived propellant supplies of this task was to identify, define and evaluate the supply of propellants derived from Lunar resources and making them available on the Lunar surface as well as in Lunar orbit. To be advantageous, a Lunar-based propellant supply should be low cost and highly self-sufficient, requiring minimal Earth-supplied resources. Four major criteria to be satisfied included: (1) adequate supply and availability of the Lunar surface raw materials; (2) low-cost processing and production of the propellant; (3) ease in storage and handling; and (4) effective use and ultimate performance of the propellant produced in the propulsion systems being considered. In addition, some general requirements for selection of the Lunar-derived propellant are: (1) the synergistic potential of the propellant production with the production of other materials needed to support the entire Lunar surface base scenario; and (2) the minimal technology risk. Manv of the ranking provided in this evaluation are subjective. Where possible, quantitative data has been developed to support the ranking.

2.4.1 Supply and Availability

The optimum propellant source is independent of external imports and has propellant available in large quantities sufficient to supply the Earth-Lunar transportation systems, and potentially, Mars and planetary transportation systems. The actual amount of propellant required depends on the performance of the propulsion systems that use propellants, the mission requirements/demands, and the support mass delivered from Earth needed to process the propellants on the Moon. Using the Baseline mission model (Section 4.1) and the Baseline hydrogen/oxygen-based transportation system (Section 3.2), the total amount of propellant required (for OTVs and landers) from 1995 to 2015 is about 7000 MT. For conventional bipropellant propulsion systems, the availability of the oxygen from Lunar resources is not a problem because of the large oxide content on the Moon. However, conventional fuel candidates for rocket applications (C, N, H) are scarce. Generally, for any propellant ingredient, the raw materials required for propellant production include the processing of Lunar regolith of some type, and in many cases, a consumable such as a reducing agent. These materials are dependent on the processing technique and therefore, the evaluation of supply and availability of the raw materials for propellant production was based on the recommended processing/production techniques. The following criteria were used to evaluate the supply of raw materials (not including the actual propellant processing). Asterisks denote criteria used for initial screening of propellants.

- Raw Material Availability*
- Consumable Raw Material Transportation
- Raw Material Acquisition
- Resources to Supply Raw Materials

The overall ranking for supply and availability candidates is shown in Table 2-11. The Solar Wind Gas Extraction, Hydrogen Reduction Techniques, and oxygen production, respectively, rank the best on raw material supply and availability. The rating system is discussed below for each of the subcriteria.

2.4.1.1 Raw Material Availability

These criteria allow assessment of the compatibility of propellant/processing techniques to Lunar resource availability. For many propellant candidates, only some of the raw materials are available on the Moon. Propellant candidates have been evaluated based on the availability of the raw materials on the Lunar surface without regard to the difficulty of acquiring, transporting, and/or processing of those raw materials. The ranking guidelines are as follows:

- 5 - 100% of all raw materials are available on the Lunar surface
- 4 - approximately 80% (by wgt) available on the Lunar surface
- 3 - approximately 50% (by wgt) available on the Lunar surface
- 2 - only 20% (by wgt) available on the Lunar surface
- 1 - raw materials only available on Earth

These criteria are considered very important and were used to evaluate propellant candidates in the early part of the study.

Table 2-5 shows the raw material availability for the various propellants and associated propellant techniques. Hydrogen Reduction, Ilmenite Electrolysis and Solar Wind Gas Extraction rank the highest while Acid Leach, Carbothermal, and Carbochlorination rank the worst.

2.4.1.2 Consumable Raw Material Transportation

The transportation of the raw materials will factor into the evaluation the actual location of the consumable raw materials source and the amount of consumable required. The baseline assumption is that the processing location is on the Lunar surface mare of highlands unless otherwise mentioned. Evaluation guidelines are provided below and are used for all of the consumable raw materials required in a specific propellant production process.

- 5 - No consumables or imports
- 4 - Consumables with source on Lunar surface
- 3 - Small mass transportation required from Earth orbit/surface
- 2 - Moderate mass transportation required from Earth orbit surface
- 1 - large mass transportation required from Earth orbit/surface

The data of concern in this evaluation are the process additive requirements shown in Table 2-5. Of these additives, only hydrogen can be obtained in sufficient quantity for the Hydrogen Reduction technique to be considered available on the Lunar surface. Approximately 31 g are required for 10 MT of oxygen of which 295 g may be recovered (calculation from 95% efficiency given by Knudsen, 1986). About 662 MT of mare regolith must be preprocessed to obtain enough ilmenite for 10 MT oxygen. Based on a hydrogen concentration of 54.8 ppm (by mass) in the mare soil, 36.2 g of hydrogen, more than twice of that needed, could be claimed from the mare which must be mined if 100% capture is assumed.

TABLE 2-5. RAW MATERIAL SUPPLY/AVAILABILITY

	LOX						Al				Mg			SiH4	H2	He3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
Availability	5	5	1	4	1	1	1	4	1	1	1	4	1	1	5	5
Transportation	4	5	2	4	2	1	2	4	2	1	2	4	2	2	5	5
Ease of Acquisition	4	3	4	5	1	2	4	5	1	2	4	5	1	1	5	5
Resources for Supply	3	2	4	2	1	3	4	5	1	3	4	5	1	1	1	1
TOTAL RANK	4.00	3.25	2.75	4.50	1.25	1.75	2.75	4.50	1.25	1.75	2.75	4.50	1.25	1.25	4.00	4.00

NOTE: 5 IS BEST; 1 IS WORST

Thus, enough hydrogen may be obtained from the Lunar soil to replenish the 15 g lost during processing.

Other process additives must be supplied from sources external to the Lunar environment, assumed here to be Earth. The ranking of the consumable transportation is shown in the second row of Table 2-5. Ilmenite Electrolysis and Solar Wind Gas Extraction ranks the best.

2.4.1.3 Lunar Raw Material Acquisition

Another important criteria of Lunar-based propellants is the ease at which the Lunar raw materials or the propellants themselves can be separated, collected and handled. No propellant or propellant raw material is available on the Lunar surface without some degree of collection and preparation. The most favorable to be expected would be one that requires selective collection without processing. The solar wind gases that have been entrained in the regolith matrix represent the easiest attainable and the most usable propellant candidates since they require only soil collection, heating, and gas collection separation systems; however, the available quantities of these gases is small and much regolith must be processed. The least favorable would include raw materials that are very difficult to locate, collect, separate and process specific samples to obtain the raw materials necessary for the propellant production process. An example of this is the magnesium silicate used in the carbothermal process.

Propellant candidates were evaluated using recommended processing techniques based on the amount of regolith to be mined/collected and the amount of preprocessing required to obtain the raw materials to produce an equivalent amount of propellant. Specific ranking guidelines

- 5 - Only Lunar regolith required as raw material without preference to regolith/Lunar rock characteristics; no preprocessing
- 4 - Simple separation of small to moderate amounts ($\leq$ 100 MT) of regolith
- 3 - Simple separation of large amounts (> 5000 MT) of regolith
- 2 - Complex separation of small to moderate amounts of regolith
- 1 - Complex separation of large amounts (> 5000 MT) of Lunar regolith

Simple separation is characterized as a one or two step of preprocessing (e.g. Ilmenite separation). Complex separation may be characterized by isolation of a single metal silicate (e.g. $Al_2O_3 \cdot SiO_2$, $MgO \cdot SiO_2$). The rankings of each propellant/process are given in the third row of Table 2-5. The Vapor Ion separation and Solar Wind Gas Extraction techniques ranked the highest.

2.4.1.4 Resources to Supply Raw Materials

The amount of equipment and power/thermal requirements required to obtain the raw materials is another consideration for Lunar-based propellant production. The equipment mass was estimated based on the amount of mare regolith which must be collected. The power/thermal requirement will be considered of lesser importance than the equipment mass. Specific evaluation guidelines are as follows:

- 5 - Low equipment mass requirements (< 5 MT), low power/thermal requirements (< 100 MW-hrs)
- 4 - Low equipment mass (< 5 MT) and moderate power/thermal needs (> 100 MW-hrs)
- 3 - Moderate equipment mass (10-20 MT), moderate power/thermal need (100-500 MW-hrs)
- 2 - Modest equipment mass (10 to 20 MT), high power/thermal needs (> 500 MW-hrs)
- 1 - High equipment mass (> 20 MT)

The evaluation of each propellant/process candidate is shown in the fourth row of Table 2-5. The Vapor Ion separation process ranks highest because of the ability to process any type of soil constituent and thus requires the least mare regolith and little/no separation and preprocessing.

2.4.2 Production and Processing

The attractiveness of propellant candidates is heavily dependent on the cost of Lunar processing relative to delivery from Earth. The Lunar surface provides a good base of operation; however, one kg delivered to the Moon requires to 3.5 to 7 kg of mass from Earth depending on the propellant/transportation system and the processing techniques Earth orbit assuming the transportation cost. Thus, the production of propellants must ultimately be self-sufficient on those resources that can be Lunar-derived. Some resources will be Lunar-derived such as chemical constituents of the soil, while other resources must first be "implanted", these will be considered Lunar-derived. Power and manpower are examples of the latter. The evaluation characteristics for propellant production and processing are listed below. Each of the criteria will have a ranking from 1 to 5, one being least favorable. The asterisks denote criteria used for preliminary screening of candidate propellant processes.

- Operational resource requirements*
- Initial set-up requirements*
- Technology risk
- Experience with technique (or similar technique)
- Process automation and control
- Output quality/purity for typical yields
- Value of bi-products
- Safety

2.4.2.1 Operational Resource Requirements

The demand and consumption rate of supplies is the driving factor of any Lunar-based activity and is a key determinant in the efficiency of the production technique. The operational resources considered include any consumable that must be periodically replenished. These resources include fluids and chemical additives, utilities such as power and thermal energy, equipment, facilities, tools, spares and manpower. Direct processing requirements of fluids/additives, utilities, and equipment were considered the most crucial and were given the most emphasis. Evaluations of these elements on a per mass product are shown in Tables 2-6a through c.

A key factor and determination whether a process will even theoretically be economical on the Moon is the ratio of consumables replenishment to product yield. If this ratio is greater than one, the process is not at all economical as it would be easier to simply deliver the products to the Moon. If this ratio is less than one, utility and equipment masses may be sufficiently large as to make delivery of the products cheaper. For these cases the break-even point is dependent on the initial equipment (including that for power and propellant) and the resupply requirements, and will be determined by the amount of propellant to be processed. Such a curve is presented for the recommended propellant scenarios in Section 2.6.

TABLE 2-6a. CONSUMABLE TO PRODUCT RATIO

	Hydrogen Reduction	Ilmenite Electrolysis	Carbo-Chlorination		Acid Leach			Vapor-ion Separation (b)	Carbo-Thermal
Consumed	.310	.25	112.1	112.1	27.5	53.7	55.57		15.0
Recycled	.300	.25	53.1	86.5	0	38.6	40.6		10.0
Products									
O2	10	10	10	10	10	10	10		10
Al				8.5		5.9	4.7		
Mg							2.4		
SiH4									10
Total Product	10	10	10	18.5	10	15.9	17.1		20
Import Consumable (a) / Total Product	0.001	0.025	5.90	1.38	2.57	0.95	0.87		0.25

(a) NOTE: SOLAR WIND GAS EXTRACTION ASSUMED NEAR ZERO

(b) CONSUMABLE REQUIREMENTS FOR VAPOR-ION SEPARATION TECHNIQUES ARE CURRENTLY UNAVAILABLE

TABLE 2-6b. INITIAL MASS TO PRODUCT RATIO

	Hydrogen Reduction	Ilmenite Electrolysis	Carbo-Chlorination		Acid Leach			Vapor-ion Separation	Carbo-Thermal
Equip. Mass	1.83	0.98	12.45	13.4	2.85	11.73	15.93	15.0	12.0
Power/Thermal System	12.35	30.12	35.37	44.03	18.24	67.79	55.28		62.01
Initial Materials	0.3	-0-	112.1	112.1	27.5	53.7	55.6	-0-	15.0
Total	14.48	31.10	159.92	165.53	48.59	133.22	126.81		89.01
Products									
O2	10	10	10	10	10	10	10	10 (a)	10
Al				8.5		5.9	4.7	1.6 (a)	
Mg							2.4	1.4 (a)	
SiH4									10
Total	10	10	10	18.5	10	15.9	17.1		20
Initial Mass to Product Ratio	1.45	3.11	15.99	9.16	4.86	8.38	7.42		4.45

(a) BASED ON 100% RECLAMATION OF THESE ELEMENTS FROM MARE

TABLE 2-6c. ENERGY / THERMAL (MW-HRS)

	Hydrogen Reduction	Ilmenite Electrolysis	Carbo-Chlorination		Acid Leach			Vapor-ion Separation (a)	Carbo-Thermal
Electrical Power	43.2	46.5	43.2	78.6	43.2	119.8	96.4		43.2
Thermal Energy	7.6	62.7	82.2	83.7	26.7	129.8	106.8		168.5
Total	50.8	109.2	125.4	162.3	69.9	249.6	203.2		211.7
Products									
O2	10	10	10	10	10	10	10		10
Al				8.5		5.9	4.7		
Mg							2.4		
SiH4									10
Total	10	10	10	18.5	10	15.9	17.1		20
Total Energy / Total Products	5.1	10.9	12.5	8.8	7.0	15.7	11.9		10.6

(a) ENERGY REQUIREMENT DATA FOR VAPOR-ION TECHNIQUES IS CURRENTLY UNAVAILABLE

The ratings of propellants/propellant candidates under this criteria are based on the consumable and power/thermal ratios presented in Tables 2-6a and c. Guidelines for the ratings are as follows:

- o 5 - little or no consumables
- o 4 - consumable to product mass ratio less than 0.3
- o 3 - consumable to product mass ratio 0.3-0.6
- o 2 - consumable to product mass ratio 0.6-1
- o 1 - consumable to total product mass ratio greater than unity

Half points were given based on the power/thermal requirements.

Table 2-7 shows the ratings for operational resource requirements in the first row. Hydrogen Reduction rates high as only 5% of all hydrogen used can not be reclaimed through the electrolysis and the residual $Fe-TiO_2$ fines. Vapor Ion separation uses no consumable except for the plasma working fluid but requires a significant power and thus, rates a 4.5 as does the Solar Wind Gas Extraction. For the HF Leach process to be valuable at all on the Lunar surface, at least Al and possibly Mg must be processed to recover enough fluoride (from the metallic fluorides Al_2F_6 and $MgFeF_6$). If the fluorides are not recovered, a significant amount of HF consumable is required to make the process viable. The carbochlorination process, as presented here, is not a logical solution to processing O_2 and/or Al because the consumable to product ratio is greater than unity.

2.4.2.2 Initial Set-Up Requirements

The initial set-up requirements for a particular propellant production technique is related to the equipment mass and initial resources required to start the processing; however, the cost of this "front-end" deliverv may be amortized over many years. The rating of this subcriterion will be determined by the equipment mass, the initial amount of consumables, and the power/thermal requirements as summarized in Table 2-6b. For this evaluation, the thermal power was considered supplied by electrical heaters at 75% efficiency. This is a conservative assumption as a significant amount of waste heat and solar-generated heat could be provided. The power system masses were calculated based on fission nuclear power system technology having an overall specific mass of 30 W/kg was used. Radiator heat rejection is also required with heat rejection requirements equal to power requirements. The specific mass of the thermal

TABLE 2-7 PROCESSING / PRODUCTION EVALUATION

	LOX						Al				Mg			Si	H2	He3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
Operational Resources	5.0	4.5	2.0	4.5	3.5	1.0	2.0	4.5	3.5	1.0	2.0	4.5	3.5	3.5	4.5	4.5
Initial Setup	5.0	4.0	3.0	2.0	3.0	2.0	3.0	2.0	3.0	2.0	3.0	2.0	3.0	3.0	4.0	4.0
Technology Risk/ Requirements	5.0	3.0	4.0	1.0	3.0	1.0	4.0	1.0	3.0	1.0	4.0	1.0	3.0	3.0	3.0	3.0
Terrestrial Experience	3.0	3.0	5.0	1.0	4.0	4.0	5.0	1.0	4.0	4.0	5.0	1.0	4.0	4.0	2.0	2.0
Automation and Control	5.0	5.0	3.0	4.0	3.0	3.0	3.0	4.0	3.0	3.0	3.0	4.0	3.0	3.0	5.0	5.0
Output Quality/Purity	5.0	4.0	4.0	3.0	4.0	3.0	4.0	3.0	4.0	3.0	4.0	3.0	4.0	4.0	3.0	3.0
Byproduct Value/Synergism	3.0	4.0	4.0	5.0	4.0	4.0	4.0	5.0	4.0	4.0	4.0	5.0	4.0	4.0	3.0	3.0
Safety	4.0	5.0	3.0	4.0	4.0	3.0	3.0	4.0	4.0	3.0	3.0	4.0	4.0	4.0	4.0	4.0
Average Rating	4.4	4.0	3.5	3.0	3.3	2.6	3.5	3.0	3.3	2.6	3.5	3.0	3.3	3.3	3.5	3.5

NOTE: 5 IS BEST, 1 IS WORST

rejection system was assumed as 0.05 g/W. A 50% duty cycle was considered for all power/thermal system estimates. The ratings for each propellant/process is given in the second line of Table 2-7.

2.4.2.3 Technology Risk

Useful technologies may either be enabling or enhancing. Processes with enabling technology needs will require extensive development programs before they become a feasible option and thus has a significant technology risk associated with it. Enhancing technologies would create a measurable difference in the performance of processes and therefore, have less technology risk associated with them. The processing technologies are addressed in Section 4.1. A process rating of "1" has many enabling technology development requirements; a "3" has no enabling technology requirements but would require extensive development of enhancing technologies before it could become a reasonably-efficient process; a "5" requires no technology development or the technology development would not appreciably benefit the process efficiency or other attributes. Ratings of 2 and 4 were also be given relative to 1, 3, and 5.

Table 2-7 shows the ratings for this subcriterion. The Hydrogen Reduction Process has been demonstrated on Earth. Work to date should be geared toward enhancing the hydrogen-ilmenite solid/gas mixing and the hydrogen gas recovery from the Fe-TiO_2 fines.

The Acid Leach process is rated 4.0 as it requires no enabling technologies and has been shown to work with platinum electrodes; however, new methods of recovering the HF need to be explored.

Carbothermal and Solar Wind Gas Extraction and Ilmenite Electrolysis are rated 3.0. The carbothermal process is fairly well developed for Earth but requires some technology development to expand its application to multi-species regolith and to reclaim the CH_4 consumable. The Solar Wind Gas Extraction process, while a fairly new development appears to have no enabling technologies and thus relatively low technology risk. Hydrogen extraction may be very similar to the H recovery from Fe-TiO_2 fines in Hydrogen Reduction. The major technology driver in the Ilmenite Electrolysis Process is reducing the degradation of electrical materials and increasing their durability against silicate material.

The Vapor Ion Separation and Carbochlorination processes are rated 1.0 because they show high risk. The Vapor Ion Separation process is still in the theoretical stage. The carbochlorination process requires substantial development in recycling the consumable carbon and chlorine before it is feasible for Lunar base applications.

2.4.2.4 Process Experience

This criterion will take into account the applicable experience with the processes on Earth or other space-related research and development. Although all applicable processes will be suitably demonstrated on Earth before use on the Moon, considerable expense and development risk may be saved if there exists prior experience. A rating of "5" has been given to those processes that are currently being used for commercial production on Earth; a "3" represents experience in a laboratory or small-scale production environment; a "1" represents no experience. Ratings are shown in Table 2-7.

The Acid Leach processes are used in aluminum and magnesium refining and have evolved over many years of commercial operation and thus ranks 5.0. The carbochlorination process has been used in special Earth-based processing applications, but requires more research before extraterrestrial application is made feasible. The carbothermal process has been demonstrated in the laboratory for processing of magnesium silicate. This process' ability to process other metal silicates has not been demonstrated in the lab and should be examined further to determine its potential for Lunar base application.

The Hydrogen Reduction and Ilmenite Electrolysis have not been specifically used for commercial application on Earth, but are in test stages with small scale plants demonstrated. No experience exists with the Solar Wind Gas Extraction as a system. However, experience with the components (including solid/gas mixing, electrolysis, solid/gas separation) does exist and is rated 2.0. No experience to date exists with Vapor Ion Separation although plasma separation and plasma processing is becoming a growing research area.

2.4.2.5 Process Automation and Control

The degree to which a process can be automated and controlled will be an important factor in the self-sustenance capability of any propellant production system established on the Lunar surface. The level of automation is largely dependent on the complexity of the process, but also on the tolerance of the process to environmental changes (e.g. temperature) and the stability of the process. This criteria is objectively determined based on the subsystem and component functions of the particular processes. A rating of "5" represents a continuous process that is stable, highly reliable tolerant of a varying environment, and relatively easy to automate; a ranking of "3" designates a system that would be potentially difficult to automate, hard to control because of high complexity or high tolerance levels, and would most probably experience a great deal of down-time; a "1" represents a process nearly impossible to control and/or automate.

The ratings for the subcriterion are also given in Table 2-7. No process was considered impossible to automate. The Hydrogen Reduction, Ilmenite Electrolysis, and Solar Wind Gas Extraction processes are relatively simple processes with few interactive components and subprocesses and are rated a 5.0. Maintaining temperature and pressure are the main consideration in addition to voltage for the Ilmenite Electrolysis. The Vapor Ion Separator process is a relatively confined process having few interacting components. However, plasma/vapor maintenance as well as plasma injection can be difficult to control and, in fact, must be automated to sustain plasma tolerances. Thus, Vapor Ion Separation is rated a 4.0.

The Acid Leach, Carbothermal, and Carbochlorination processes are rated 3.0 because of the many subprocessing branches and extensive recycling requirements to sustain processing over a long period of time.

2.4.2.6 Output Quality/Purity

The quality and purity of the product is important in the determination of additional requirements for product refinement or whether additional factors such as accommodation of impurity in the propulsion system should be considered. Factors influenced by an impure output may be reduced efficiency, reliability, maintainability, and life of the associated propulsion system. A rating of "5" means that the output is near 100% pure; a "4" means that impurities may be measurable, but not detrimental, a "3" represents output purities with potential harmful effects, a "2" means that harmful impurities exist, but can be filtered out with additional systems; a "1" means that degradant impurities will exist and are nearly impossible to avoid.

Table 2-7 provides a summary of the ratings for this subcriterion. Producing oxygen from water electrolysis is considered to yield very pure oxygen. Hydrogen Reduction is rated 5.0. Acid Leach and Carbothermal process also produces oxygen from water electrolysis, but have potential for impurity in the other products (aluminum and silane) and are thus rated 4.0. Ilmenite Electrolysis is also rated 4.0 because the oxygen is bubbled from the molten ilmenite and the impurities could arise from other gases trapped in the ilmenite matrix. Vapor Ion Separation and Solar Wind Gas Extraction may have impurities due to the lack of accuracy in ion and gaseous separation and are rated 3.0. Carbochlorination is also rated a 3.0 because of the aluminum separation from the chlorine.

2.4.2.7 Value of Bi-Products

All processes have some type of bi-product and some processes may accommodate more than one product through a specific process branch. We call this synergistic processing. A discussion of synergistic processing follows in Section 2.5. Potentially useful Lunar base materials were listed in Table 2-2. A ranking of "5" means that bi-products will be adequately available for a designated use; a "4" means that many generally useful bi-products exist; a "3" represents bi-products that are not extremely useful or are available in moderate quantities only; a "2" designates bi-products that are difficult to use without a great deal of additional processing; and a "1" means that no or unuseable bi-products are available.

The Vapor Ion Separation is by far the most synergistic process considered. Theoretically, any type of Lunar regolith can be crushed and vaporized for elemental separation of its constituents. Many useful materials could be formulated for use in the Lunar base scenario.

Acid Leach and Carbochlorination are rated 4.0 because they accommodate multiple process branches that yield various metals/gases such as aluminum, magnesium, silicon and oxygen. The carbothermal process has produced magnesium, silicon and oxygen in the laboratory. If processing of additional metal silicates is found possible, this process scenario can potentially yield aluminum, magnesium, iron, calcium, silicon and oxygen. Ilmenite Electrolysis yields a bi-product of molten $Fe \cdot TiO_2/FeO \cdot TiO_2$ which may be formed for various applications and is also rated 4.0.

Solar Wind Gas Extraction, rated 3.0, will yield volatile gases such as hydrogen, oxygen helium, neon, argon, and possibly water vapor which may be used as working fluids, as input to life support, or as input to various processing or scientific applications. However, the amount of these volatiles is extremely limited.

The Hydrogen Reduction process, rated 2.0, yields powdered $Fe \cdot TiO_2$ which would require extensive processing to be useful.

2.4.2.8 Safety

The safety criterion is a measure of potential hazard to man and the Lunar base systems during the processing phase. These hazards include explosions, degradation of equipment operation, risk of contamination, and risk of life. More potential of hazard will induce burdens on system design safety factors, operational constraints and reduce the relative integration potential of the process to the Lunar base infrastructure. A rating of "5" is given for processes with few safety concerns; a "3" means moderate hazards exist, but can be mitigated; a "1" means that a process is believed to be a hazard, is extremelv dangerous and may not be mitigated without very costly measures.

Ilmenite Electrolysis is considered a 5.0 with regard to safety because process rates are fairly slow and the melt-through hazard is not considered a significant or likely event. Hydrogen Reduction is fairly safe, and was rated 4.0. The potential for gaseous explosions exists for the Vapor Ion Separation, Solar Wind Gas Extraction and Carbothermal processes. Acid Leach and Carbochlorination are rated 3.0 because of the caustic materials and explosive gases present in the systems.

2.4.3 Storage and Handling

Propellant candidates exhibiting ease of storage and handling will simplify equipment and facility needs for both the propellant depot and the transportation tankage. Operationally, however, a Lunar or space storable propellant may not present the safest propellants to handle. Of course, technology plays a very important role in the long-term storage capability and the resources required to store. To address these characteristics, we have used three major subcriteria in which to evaluate the storage and handling aspects of propellant candidates:

- o Complexity and requirements of containment
- o Long-term storage capability
- o Safety.

Table 2-8 summarizes the rating among these subcriteria. Much of the evaluation is propellant dependent rather than process dependent.

2.4.3.1 Complexity and Requirements of Containment

The complexity and size of the storage vessel depends primarily on the temperature and pressure at which the propellant must be stored. Cryogenic tanks are much more complex than storables; solids are easier to store than storables. Below, we rank the relative difficulty of storage and the amount of equipment and facilities required to store and subsequently acquire from storage a specific propellant. Guidelines for this evaluation are as follows:

- 5 - Simple stock piling as a bulk material in large storage containers
- 4 - Lunar surface-storable or space-storable in sample containers
- 3 - Storage requires modest environmental control of the storage container
- 2 - Significant environmental control and support required to store (e.g. cryogens)
- 1 - Extremely complicated storage techniques required.

The ratings presented in Table 2-8 are relatively straightforward for individual propellants. The variance in ratings is due to the process-end temperature which may require additional cooling.

2.4.3.2 Long-Term Storage Capability

The lifetime of a propellant in its storage container is dependent on the technology used to store it. A ranking of "5" represents no leakage/boiloff; a "3" depicts some boiloff, but no shortening of storage time below 1 year; a "1" represents significant boiloff.

Silane was rated a stable, storable propellant. Aluminum and magnesium may be stored as powders with low-humidity vessels (easy on the Moon) and thus rated a 5.0. Cryogenics (He, H_2, O_2) were rated the worst at 3.0 because of boiloff considerations. However, technologies such as magnetic reliquefaction can eliminate boiloff by recirculating and recooling vented gases.

2.4.3.3 Safety

The safety criterion is a measure of potential hazard to man and the Lunar base systems due to storage and handling of the propellant candidate. These hazards include explosions, degradation of equipment operation, risk of con-

TABLE 2-8. HANDLING AND STORAGE

	LOX						Al				Mg			SiH4	H2	He3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
Complexity/ Requirements	3.0	3.0	3.0	2.0	3.0	3.0	4.0	3.0	4.0	4.0	4.0	3.0	4.0	4.0	3.0	3.0
Long-term Storage	3.0	3.0	3.0	3.0	3.0	3.0	4.0	4.0	4.0	4.0	4.0	4.0	4.0	5.0	3.0	3.0
Safety	4.0	4.0	4.0	4.0	4.0	4.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	4.0	3.0	4.0
TOTALS	3.3	3.3	3.3	3.0	3.3	3.3	4.3	4.0	4.3	4.3	4.3	4.0	4.3	4.3	3.0	3.3

NOTE: 5 IS BEST; 1 IS WORST

tamination, and risk of life. Significant hazards will induce burdens on system design safety factors, operational constraints and reduce the relative integration potential of the Lunar base systems and the manned environment. A ranking of "5" means a minimum hazard present; a "3" means moderate hazard, but can be mitigated; a "1" means that propellant storage is extremely hazardous and may not be mitigated without very costly measures.

Aluminum and magnesium were rated 5.0 because they pose little or no hazard when handled or stored in oxygen-free environment. Silane, oxygen and helium were considered slightly hazardous and rated 4.0. Hydrogen was considered the most hazardous and rated a 3.0.

2.4.4 Propellant Use and Performance

The actual use and performance of the propellant in the space transportation operational scenario is a key criterion. Much of this depends on the propulsion system used for the propellant; however, a theoretical maximum has been calculated for the propellant candidates and will be used to rank them (see Propellant Manual). Also an important criterion for a Lunar-based propellant system is the mixture ratio (oxidizer to fuel ratio). Also, the exhaust toxicity and hazard, ease of loading and controlling the propellant within the propulsion system, and any operational constraints that may be imposed by the use of a particular propellant candidate were evaluated. These evaluations are relative to the propulsion system recommended for the appropriate propellant. The list of criteria includes Isp exhaust hazard toxicity, and ease of control.

2.4.4.1 Specific Impulse

The measure of the performance potential of a propellant may be given by the specific impulse (Isp). The Isp is a measure of the relative quantity of the total propellant that will be needed to perform for a given mission. For chemical systems, the guideline for rank are as follows:

- 5 - over 500 sec
- 4 - 450 - 490 sec
- 3 - 300 - 450 sec
- 2 - 150 - 300 sec
- 1 - less than 150 sec.

These ratings, which are extremely propellant-dependent and not process-dependent, are shown in Table 2-9.

TABLE 2-9. PROPELLANT USE / PERFORMANCE

	LOX						Al				Mg			SiH_4	H_2	He3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
Theoretical Isp / MR	4.0	4.0	4.0	4.0	4.0	4.0	3.0	3.0	3.0	3.0	2.5	2.5	2.5	3.0	4.0	5.0
Exhaust Toxicity	5.0	5.0	5.0	5.0	5.0	5.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0	5.0	3.0
Ease of Control	5.0	5.0	5.0	5.0	5.0	5.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0	4.0	5.0	3.0
AVERAGE	4.7	4.7	4.7	4.7	4.7	4.7	3.0	3.0	3.0	3.0	2.8	2.8	2.8	3.3	4.7	3.6

NOTE: 5 IS BEST; 1 IS WORST

Oxygen and hydrogen systems were rated 4.0 because of relatively high Isp. Silane was rated 3.0 because of its 360 second Isp. Aluminum was rated 3.0 as it could provide fairly high Isp when mixed with hydrogen (400 sec). Magnesium rated lower than aluminum because of reduced Isp. For information purposes, helium was rated a 5.0 because of the extremely high Isp (10^6).

2.4.4.2 Exhaust Hazards

The danger to crew and Lunar facilities is of primary importance. The hazard of the exhaust gases from the propulsion systems is an important consideration. Plume impingement, particulate debris, regolith disturbance, exhaust effects, and general environmental impact are concerns which were evaluated under this criterion. A ranking of "5" represents low risk exhaust products such as water from the H/O system; a "3" designates only a slight hazard with effluents of low toxicity or reactivity; a "1" represents exhaust products that are not necessarily desirable to the Lunar surface base environment. Use of fluorine as the oxidizer would be an example of a "1".

Hydrogen and oxygen rated 5. Silane, Aluminum and magnesium rated 4 because of the exhaust particulates.

2.4.4.3 Ease of Control

The characteristic of a propellant to be circulated in a propulsion device was evaluated here. If a propellant can be controlled easily during the operation of a propulsion system, a ranking of five has been given; propellants that require extensive component hardware to control their combustion rank a "3"; propellants that are nearly impossible to control rank a "1". Hydrogen and oxygen rated 5 as their control is state-of-the-art. Silane ranked 4 because of possible problems of pumping (residual silicon sediment) the silicon-hydrogen mix. Likewise because of tripropellant injection or pumping of a hybrid slurry, aluminum and magnesium rated a 3.0.

2.4.5 System-Level Characteristics

Some propellants may appear more synergistic, and require less development risk. These are the major criteria used to evaluate the overall benefits of a particular propellant.

2.4.5.1 Integration to Lunar Surface Base

A propellant or by product may function in many ways within the LSB scenario. These propellants will rank high "5". Potential secondary use include:

power, production, storage, and support of LSB and/or orbiting facilities. Propellants that have potential application in multiple systems, or that have only marginal value in their secondary application are rated "3". Those propellants without potential application within the LSB infrastructure are ranked "1". The ratings are shown in Table 2-10.

The Vapor-Ion Separation technique was rated a 5 because of its potential percentages to processing of most elemental constituents of the Lunar regolith. Acid Leach and Solar Wind Extraction were rated 4.0 because of the value of their synergistic processing, acid leach producing aluminum, magnesium and any other metal from metal silicates, solar wind extraction producing hydrogen, helium, carbon, nitrogen, argon and neon. The Carbothermal and Carbochlorination processes were rated 3 because of their potential for producing aluminum in addition to oxygen. Magnesium is also produced, however these processes are severely limited in their recovery of consumables when additional metal silicates are added. Ilmenite Electrolysis and Hydrogen Reduction were both rated low because of their lack of synergistic potential. Ilmenite Electrolysis rated slightly above hydrogen reduction because of the potential of using the liquid iron and titanium oxide byproducts.

2.4.5.2 Technology Risk

Major technology risks exist in acquisition of resources, processing of propellants, storage and handling, and the use of the propellant. In addition, technologies that are enabling carry a much greater risk than those that are enhancing. Requirements for developing enabling technology in the four categories listed above will require extensive research subtended by high risks. These propellants rank "1". Those propellants that have many technology requirements but the technologies are enhancing rather than enabling rank "2". Propellants with modest technology requirements (enabling and enhancing) are ranked "3". Propellants with only enhancing technology development requirements rank "4" while those with minimal technology development requirements rank "5". The ratings are shown in Table 2-10.

Table 2-11 shows the overall ranking of propellant/processing candidates on these major criteria. Each of these will be described in the following sections, including evaluations of each propellant/processing techniques by subcriteria. Hydrogen reduction is rated a 5 because it has been experimented and demonstrated in the context of a Lunar processing technique. Acid Leach,

TABLE 2-10. OVERALL SYSTEM CHARACTERISTICS

	LOX						Al				Mg			SiH4	H2	He3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
LSB Integration	1.0	2.0	4.0	5.0	3.0	3.0	4.0	5.0	3.0	3.0	4.0	5.0	3.0	3.0	4.0	4.0
Technology Risk	5.0	3.0	4.0	2.0	4.0	4.0	4.0	2.0	4.0	4.0	4.0	1.0	4.0	4.0	2.0	2.0
Average Rating	3.0	2.5	4.5	3.5	3.5	3.5	4.0	3.5	3.5	3.5	4.0	3.0	3.5	3.5	3.0	3.0

NOTE: 5 IS BEST; 1 IS WORST

TABLE 2-11. PROPELLANT / PROCESS EVALUATION

	LOX						Al				Mg			SiH_4	H_2	He^3
	H Reduction	Ilmenite Electrolysis	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbochlor.	Acid Leach (HF)	Vapor Ion Separation	Carbothermal	Carbothermal	Solar Wind Extraction	Solar Wind Extraction
Supply and Availability	4.00	3.25	2.75	4.50	1.25	1.75	2.75	4.50	1.25	1.75	2.75	4.50	1.25	1.25	4.00	4.00
Processing/ Production	4.4	4.0	3.5	3.0	3.3	2.6	3.5	3.0	3.3	2.6	3.5	3.0	3.3	3.3	3.5	3.5
Storage/ Handling	3.30	3.30	3.30	3.00	3.30	3.30	4.30	4.00	4.30	4.30	4.30	4.00	4.30	4.30	3.00	3.30
Use/ Performance	4.50	4.50	4.50	4.50	4.50	4.50	2.50	2.50	2.50	2.50	2.50	2.50	2.50	2.75	4.50	3.50
Overall System Char.	4.50	4.50	4.50	4.50	4.50	4.50	3.00	3.00	3.00	3.00	3.00	3.00	3.00	3.00	4.50	2.50
AVERAGE	4.14	3.91	3.71	3.90	3.37	3.33	3.21	3.40	2.87	2.83	3.21	3.40	2.87	2.92	3.9	3.36

NOTE: 5 IS BEST; 1 IS WORST

Carbothermal, and Carbochlorination techniques are rated 4 as the technology for Earth base requirements is state-of-the-art; however, the Lunar based applications have not been investigated. Ilmenite Electrolysis is rated a 3 and it is not readily used in the commercial sector but has been preliminarily tested on a very small scale. Vapor Ion Separation and Solar Wind Gas Extraction rate a 2. The system components and subsystems have been demonstrated and tested, however the technologies have not been addressed at a systems level for Lunar base applications.

2.5 Potential for Synergistic Processing

A synergistic approach to analyzing propellant processing schemes would involve producing a broad range of products using minimal amounts of resources needed from Earth. The first step would involve defining the needs of a Lunar base and assigning a degree of importance to each potential Lunar resource (applications of potential Lunar resources are shown in Table 2-2 of Section 2.1). In the infant stage of the Lunar base, propellants will have the greatest importance. If propellants were needed only for transportation to the Moon, additional equipment needed to support the Lunar base may be brought up as payload. This would reduce the time needed for Lunar base development. The initial material processing scheme may only have to address the need for extracting propellant resources, but must be adaptable to meet future needs of a Lunar base scenario.

The potential for synergistic processing of each propellant processing scheme was determined using three criteria:

1. The number of potential Lunar resources that may be extracted,
2. The value of each obtainable potential Lunar resource, and
3. The additional resource, equipment and energy requirements needed to obtain additional Lunar resources.

Hydrogen reduction of ilmenite and magma electrolysis ranked lowest in potential for synergistic processing. Both of these process schemes require beneficiation of the mare regolith to obtain ilmenite, $FeO \cdot TiO_2$. Ilmenite contains only three potential Lunar resources: iron, titanium and oxygen. Due to the difficulty in separating titanium from its oxide, hydrogen reduction and magma electrolysis are limited to production of iron and oxygen. The availability of oxygen using either of these process schemes is limited to the oxygen from FeO only. Capabilities of hydrogen reduction allow for potential extraction of all oxygen from FeO but additional separation hardware is required for extraction of iron. Capabilities of magma electrolysis allow for potential extraction of only 50% of all oxygen and iron from FeO in ilmenite.

The Carbothermal process has fair synergistic potential. Terrestrial application of the Carbothermal process has successfully extracted oxygen and silicon from $MgO \cdot SiO_2$ (Aerojet). With additional hardware, MgO, produced during the methane reduction step, may be processed to obtain additional oxygen and magnesium. Further experimentation using the Carbothermal process will demonstrate the process potential to manufacture oxygen silicon and various metals using other metal silicates from the mare regolith.

Carbochlorination and Acid Leach both have good synergistic potential. The Carbochlorination requires beneficiation of the mare regolith to anorthite, $CaO \cdot Al_2O_3 \cdot 2SiO_2$. All oxygen in anorthite can be recovered. Carbochlorination process requirements for oxygen and aluminum are only slightly greater than requirements for oxygen alone. No additional mare or reactants are required to extract aluminum in addition to oxygen and the reactant recovery potential increases by 50%. Thermal power requirements increase only slightly, but electrical requirements almost double. Less than a 10% increase in equipment weight is required for oxygen and aluminum production. In addition to oxygen and aluminum, the Carbochlorination process has potential to extract calcium and silicon from the beneficiated Lunar mare regolith.

In the Acid Leach process, all oxygen is extracted by electrolysis of water produced in the leach reactions. If unbeneficiated mare regolith is leached, all leach reaction products besides water are ignored and no acid is recycled, process requirements are considerably less than for the carbochlorination process scheme for oxygen production only. For aluminum and oxygen production, using the Acid Leach process, mare regolith is beneficiated to $Al_2O_3 \cdot SiO_2$ to reduce reactant consumption. This causes an increase in the mare regolith requirement by more than a factor of 3, as well as large increases in many other process requirements. The reactant requirement doubles, but reactant recovery potential is 70%. Thermal power requirements increase more than 4 times, and electrical power requirements nearly triple. A large increase is seen by the hardware requirement. Equipment weight increases by a factor greater than 4. Analysis of Acid Leach for production of aluminum, oxygen and magnesium exemplifies this process schemes potential for synergism. Mare regolith and power requirements both drop when processing Al, O_2 and Mg compared to just Al and O_2. Although the reactant requirement increases, the reactant recovery potential increases. The rise in required hardware weight is much smaller than the rise from production of O_2 only to O_2 and Al. These trends suggest that the Acid Leach process works most efficiently when used to obtain several products. If unbeneficiated mare is processed using the Acid leach process, and all additional process requirements are met, oxygen, aluminum, silicon, magnesium, iron, calcium and titanium may potentially be produced.

The Vapor Ion Separation process has the greatest potential for synergism. The proposed collection methods for this conceptual process can be fine tuned to extract specific elements from the vapor stream. It is possible that aluminum, iron, titanium, magnesium and oxygen can be extracted from unbeneficiated mare regolith in one automated process step. This process scheme requires no reactants and minimal additional hardware to obtain many useful lunar resources. This process level of automation, level of self-sufficiency and potential for synergism make it ideal for Lunar base application. This process is unproven in the lab and much research and development is needed before an accurate quantitative analysis can be made.

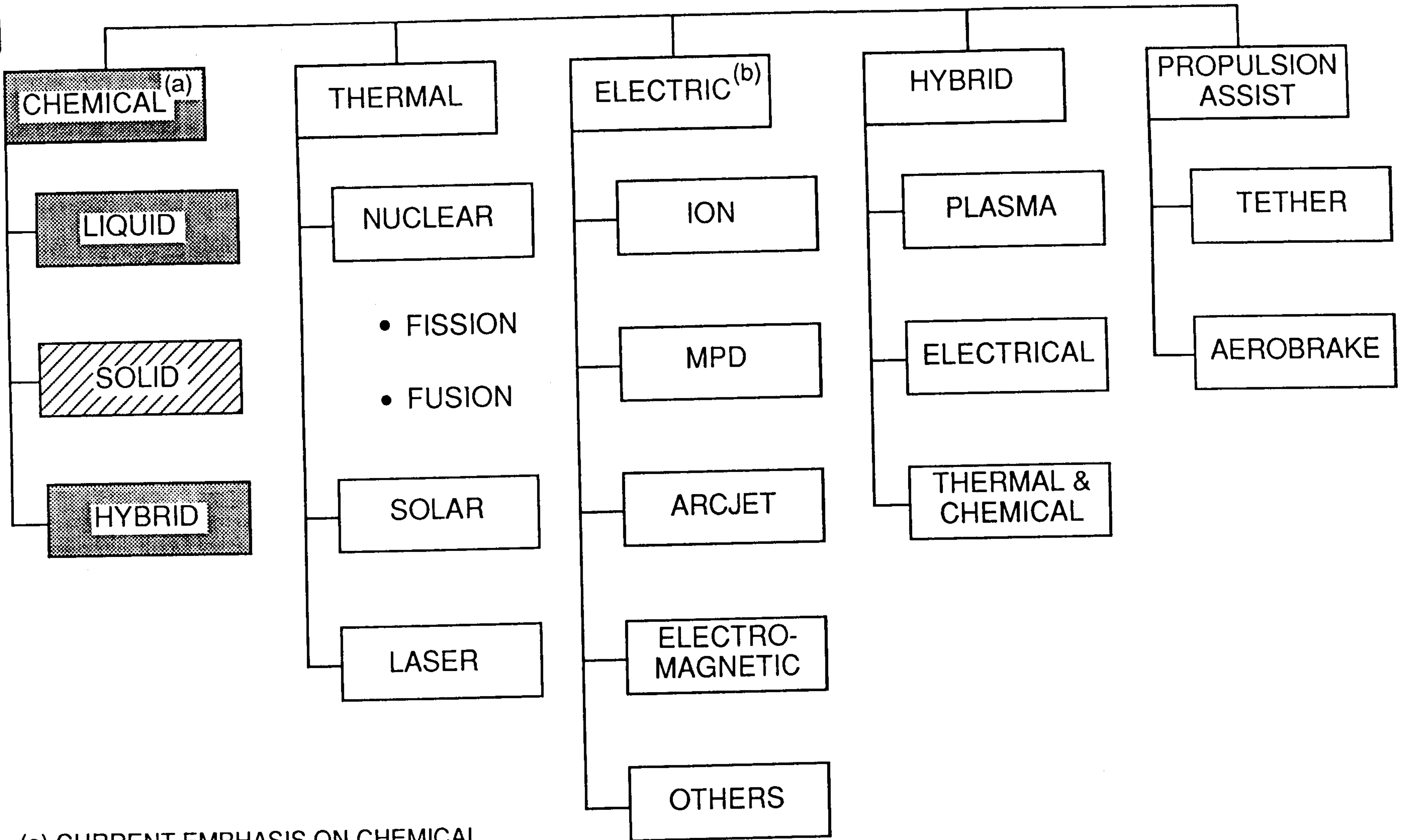
3.0 PROPULSION/VEHICLE SYSTEMS

There are many propulsion and vehicle systems and technology alternatives that can be applied to the Earth-Moon transportation infrastructure. The objective of Task 2, Design/Analyze Propulsion/Vehicle Systems, was to identify and select propulsion and vehicle systems capable of supporting the Lunar mission model. Vehicle families have been developed from propellant/vehicle alternatives based on mission model requirements. A vehicle family is a group of systems (e.g., an OTV and a lander) that are specifically designed for a given mission requirement. Figure 3-1 shows the scope of propulsion alternatives applicable to the Earth/Moon transportation scenario. The gray boxes indicate emphasis of scope within this study. Propulsion alternatives were chosen based on propellant availability identified in Task 1 and the hydrogen/oxygen, pump-fed rocket engine was considered the baseline propulsion system. The baseline vehicle system follows Orbital Transfer Vehicle (OTV) configurations currently being studied at NASA/MSFC. Vehicle options included aerobrakes, propellant tankage, support system mass and landing gear mass.

Characteristics of the baseline propulsion/vehicle designs were varied to address sensitivities and tradeoffs. Figure 3-2 presents the parameters for which sensitivities were examined. The parameters and their corresponding values at the ends of each line represent the data range explored for each tradeoff. In examining these tradeoffs, it must be remembered that the individual parameters are rarely independent. In many cases, changing one parameter changes another. For example, a change in the mixture ratio results in a change of the specific impulse, thrust, and other engine parameters.

To develop engine and stage parameters, mixture ratios and combustion environments were analyzed in the "ELES" computer code. ELES is a design tool to assist in preliminary systems analyses of liquid rocket engines/vehicles. The section of the code used here estimated size, weight and engine performance based on the standard Joint Army Navy NASA Air Force (JANNAF) thermochemical analysis methods. A description of the ELES code is provided in Appendix C. Outputs from the ELES code were obtained for 18 propulsion system design cases. These cases are presented in Sections 3.1 and 3.2

Weight and performance estimates from ELES were used by the computer code ASTROSIZE to size the aerobrake, propellant tankage and landing gear vehicle



(a) CURRENT EMPHASIS ON CHEMICAL

(b) POWER SYSTEM MAY BE NUCLEAR FISSION OR FUSION

FIGURE 3-1. LSB PROPULSION ALTERNATIVES

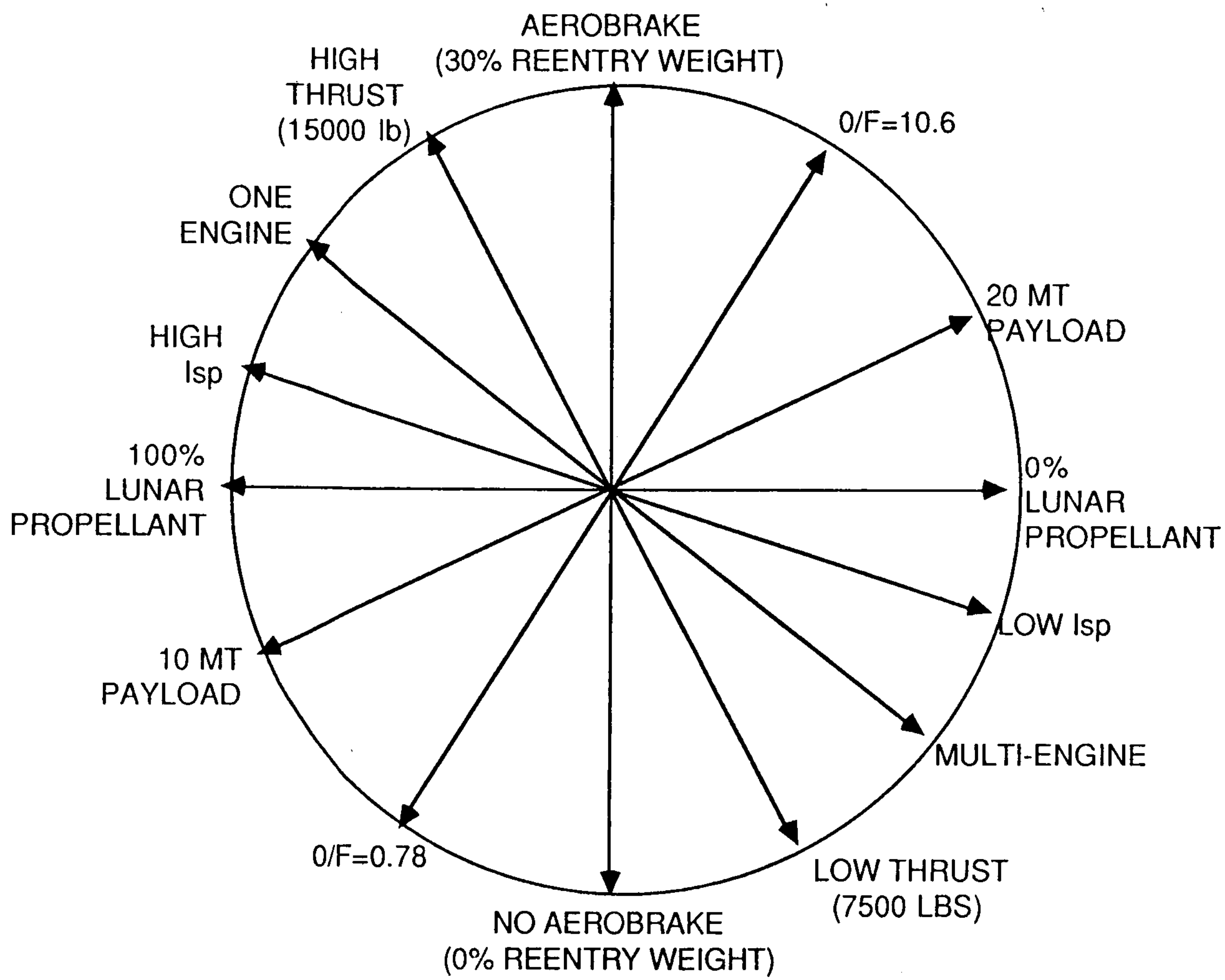


FIGURE 3-2. TRADEOFFS AND SENSITIVITIES

systems to meet mission requirements. The ASTROSIZE analysis includes consideration of specific propellant supply nodes, basing nodes, and vehicle design alternatives such as tank design parameters, aerobrake characteristics, landing gear parameters, number of engines, and payload capacity. A total vehicle mass and propellant capacity is output for a specific reference design mission. ASTROSIZE outputs then were fed to ASTROFEST which manifests mission payload requirements and propellant supply burdens to obtain total transportation requirements: the number of flights between each node and the propellant supply requirements at each supply node. From the outputs of ASTROFEST, total Earth Launch Mass (ELM) is calculated. The ELM is the total mass that must be launched from the surface of the Earth over the duration of the mission model. Brief descriptions of ASTROSIZE and ASTROFEST are provided in Appendix E. The ELM in addition to the number of OTVs and associated support mass required provides a relative measure of efficiency for a given vehicle family. Another measure associated with the ELM is mass payback ratio. Mass payback ratio is the total Earth launch mass divided by the delivered payload mass.

The mission scenario, propellant supply scenarios, and basing options are discussed in Section 4.0 with the sensitivity and trade studies. The propulsion systems, vehicle systems, and vehicle family options are discussed here in Section 3.0. The sections below describe the basic assumptions and the resulting characteristics and parameters of the propulsion and vehicle systems. Section 3.1 provides the basic mission requirements as dictated by the NASA/JSC mission model. Section 3.2 provides data on the baseline propulsion/vehicle concepts. Alternatives to the propulsion and vehicle systems are delineated in Sections 3.3 and 3.4, respectively. Section 3.5 integrates the propulsion and vehicle systems into sets of vehicles or families which are used in the tradeoff studies of Section 4. Section 3.6 provides an overview of the recommended propulsion/vehicle systems.

3.1 Mission Requirements Definition

NASA/JSC supplied the mission model, which is shown in Figure 3-3 and Table 3-1. The nominal mission model covers 20 years from 1995 to 2015. The darkened bars in Figure 3-3 represents the initial mass to the Lunar surface or to Lunar orbit. The open bars denote payload delivery to the Lunar Surface/orbit not directly associated with propellant production. The mission model shows gaps with no flights in 1997-1998, 2000-2002, and 2012 and uneven flight rates over the 20 year mission period. However, this does not greatly impact transpor-

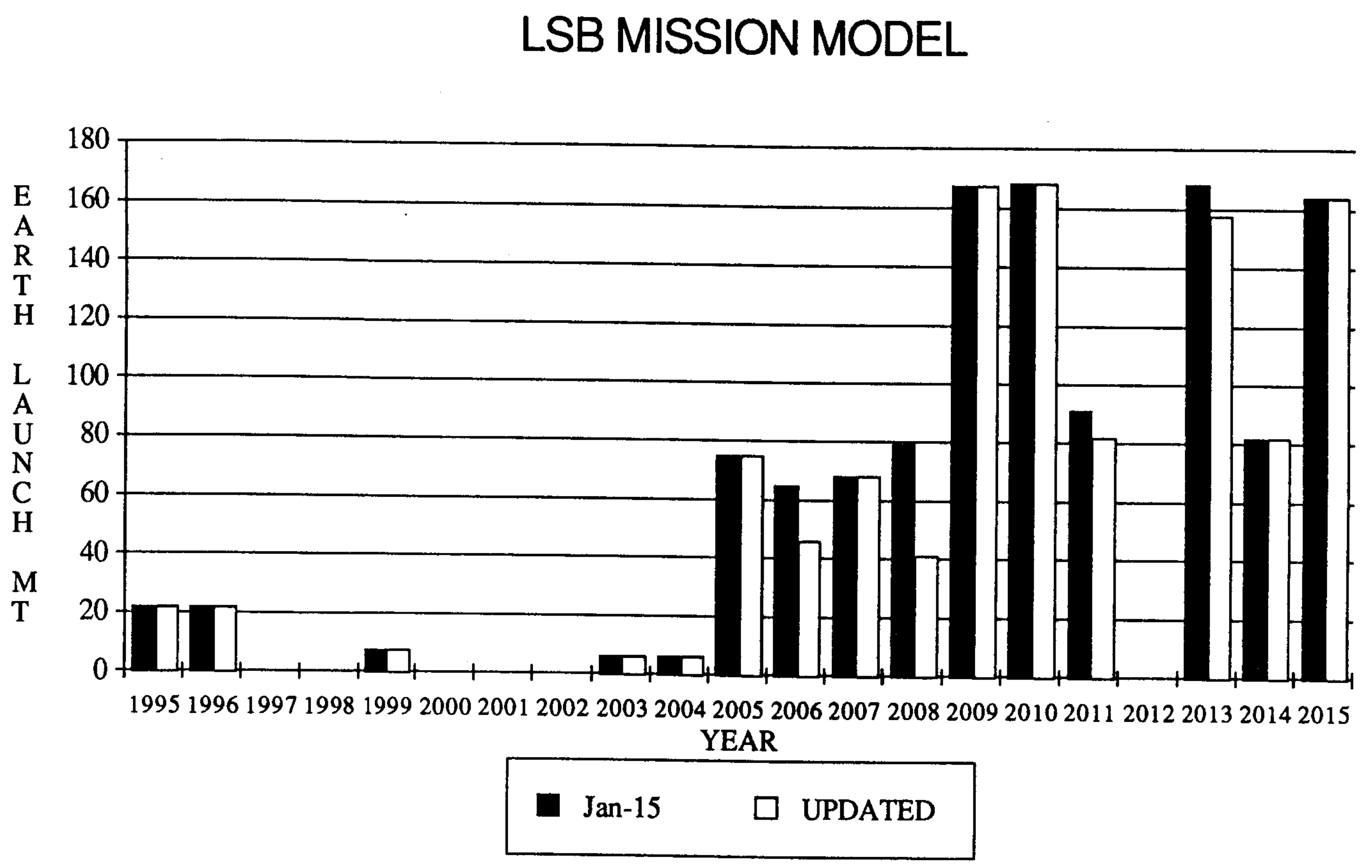


FIGURE 3-3. JSC MISSION MODEL, JANUARY 1986

TABLE 3-1. JSC LSB MISSION MODEL, JANUARY 1986

YEAR	DESTINATION LO	LS	RETURN	NO. OF PAYLOADS	MASS, MT (klbs)	MANNED
1995	X			1	22.7 (50)	
1996	X			1	22.7 (50)	
1999		X		1	8.2 (18)	
2003			X	1		X
2004			X	1		X
2005		X		2	2.3 (5)	X
		X		3	19.2 (42.4)	
2006		X		1	19.2 (42.4)	
				3	2.3 (5)	X
2007	X			1	22.7 (50)	
		X		3	2.3 (5)	X
		X		1	19.2 (42.4)	
2009		X		9	2.3 (5)	X
		X		2	19.2 (42.4)	
		X		3	15.9 (35)	
2010		X		4	15.9 (35)	X
		X		3	10.4 (23)	X
		X		3	3.6 (8)	
	X			1	22.7 (50)	
2011		X		1	15.9 (35)	X
		X		2	10.4 (23)	X
		X		4	3.6 (8)	
2013		X		5	15.9 (35)	X
		X		4	10.4 (23)	X
		X		1	3.6 (8)	
2014		X		3	10.4 (23)	X
		X		3	3.6 (8)	X
2015		X		1	3.6 (8)	X
		X		1	45.4 (100)	
		X		1	108.9 (240)	

tation vehicle comparisons since the analysis is primarily based on total flights over the 20-year mission period.

The key factors that influence the vehicle design are the delta-v requirements between basing and propellant supply nodes. Nodes were chosen in the most obvious locations, LEO and Low Lunar Orbit (LLO). The resulting velocity increments are shown in Figure 3-4. With the nominal Moon-to-Earth trajectory selected, inclusion of an aerobrake would save over 3 km/sec. (9840 ft/sec.).

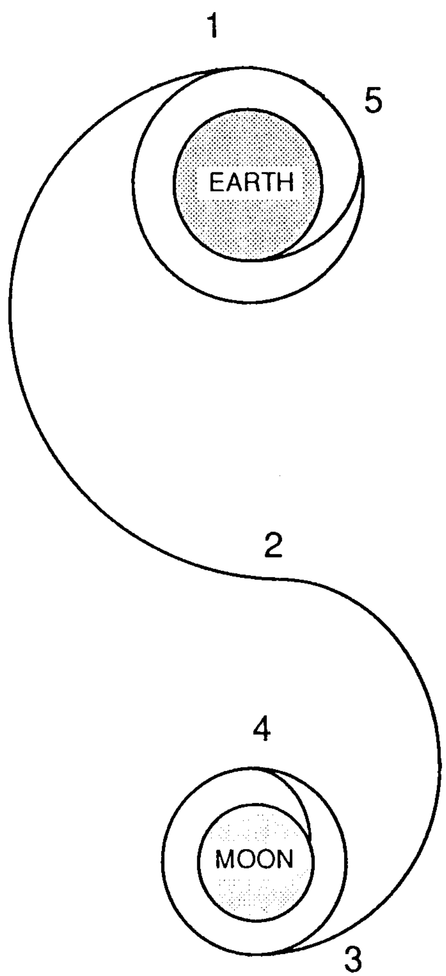
Other assumptions include typical payload and manned capsule mass. The manned capsule was assumed to have a mass of 6.9 metric tons (MT) (15 klbs) (NASA/JSC, 1986). Payloads fall into six major sizes; 17 payloads at 2.3 MT (5 klbs), 4 at 3.6 MT (8 klbs), 12 at 10.4 MT (23 klbs), 9 at 15.9 MT (35 klbs), 7 at 19.2 MT (42 klbs), and 4 payloads at 22.7 MT (50 klbs). The 22.7 MT payloads are Lunar orbiters. The 19.2 MT payloads include power stations, initial habitat/lab modules, and mobility/mining units. The 15.9 MT payloads are subsequent habitats, labs and scientific equipment. The 10.4 MT payloads are large payloads on a manned sortie mission while the 3.6 MT missions represent smaller payloads on a manned sortie and the 2.3 MT payloads are even smaller payloads accompanying manned sortie missions.

3.2 Baseline Propulsion/Vehicle Concept

The baseline propulsion/vehicle concept was modeled after current NASA OTV studies. The propulsion system used liquid hydrogen/oxygen propellants that were pump fed to a chamber pressure of 1380 N/cm^2 (2000 psi) and temperature of 3346 C (6055 F) at a O/F mixture ratio of 5.5. The resulting specific impulse was estimated at 470 seconds. The basic propulsion system design was modeled after the Aerojet dual propellant expander engine cycle. The total engine mass was estimated at 95 kg (210 lbs). Table 3-2 provides a summary weight statement for the baseline propulsion system derived from ELES.

Basic vehicle components were modeled after the Centaur D-1T. The Centaur D-1T masses are summarized in Table 3-3. The propellant tanks were assumed to be made from 301 CRES stainless steel at 0.36 mm (.014 in.) minimum guage size. The tanks were cylindrical with elliptical ends of ellipse ratio 1.38.

The baseline infrastructure included three basing/servicing nodes, reusable OTV, and a reusable lander. The nodes are at LEO, Low Lunar Orbit (LLO), and the Lunar Surface Base (LSB), all being nodes for servicing, propellant supply



1 3.15 km/sec LEO TO TRANSLUNAR

2 0.05 km/sec MIDCOURSE CORRECTION

3 0.97 km/sec TRANSLUNAR TO LLO

4 2.10 km/sec LLO TO LUNAR SURFACE

5 0.09 km/sec TRANSLUNAR TO LEO WITH AEROBRAKE

TOTAL TRIP FROM LUNAR SURFACE TO LEO

3.21 km/sec WITH AEROBRAKE

6.27 km/sec WITHOUT AEROBRAKE

FIGURE 3-4. ASSUMED EARTH-MOON VELOCITY REQUIREMENTS

(NASA/JSC, 1986)

TABLE 3-2. ELES CODE ENGINE DATA OUTPUT

EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN. COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

ENGINE DIMENSIONS (INCHES)	
THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.009
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTACH AREA RATIO	20.00
NOZZLE LENGTH / (MIN RAD LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00
ENGINE WEIGHTS (POUNDS)	
NOZZLE EXTENSION	23.01
CHAMBER	20.85
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	3.42
TCA CONSTRUCTION	2.51
SINGLE THRUST CHAMBER ASSY	56.05
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	4.11
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	112.10
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	8.22

PERFORMANCE	
DELIVERED ISP (VAC), SEC	469.61
IDEAL ISP (ODE), SEC	482.94
DELIVERED CSTAR, FT/SEC	7608
IDEAL CSTAR, FT/SEC	7647
CHAMBER PRESSURE, PSIA	2000
THRUST PER ENGINE (VAC), LBF	7500
TOTAL VAC THRUST, LBF	15000
BURN TIME, SEC	3587.93
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	27.02
FUEL FLOWRATE, LB/SEC	4.92
TOTAL FLOWRATE, LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE, DEG R	6515
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE, DEG R	3792
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY, ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.059
FUEL ORIFICE DIAMETER (IN)	0.078

TABLE 3-3. CENTAUR D1-T WEIGHT SUMMARY

UNMODELED COMPONENTS	MASS(kg)
STUB ADAPTER	114
EQUIPMENT MODULE	112
ACS	3
ULLAGE MOTORS	18
PROPELLANT UTILIZATION	21
AUXILIARY PROPELLANT SYSTEM	67
GUIDANCE SYSTEM	77
AUTOPILOT SYSTEM	66
ELECTRICAL SYSTEM	65
RANGE SAFETY SYSTEM	24
TRACKING SYSTEM	5
TLM SYSTEM	128
ADAPTER PAYLOAD	34
SEPARATION SYSTEM	24
HELIUM	4
ICE	5
UNMODELED COMP. MASS	767

MODELED SUBCOMPONENTS	MASS (kg)
BASIC STRUCTURE	371
SECONDARY STRUCTURE	124
MAIN ENGINE SYSTEM	274
FUEL SYSTEM	82
OX SYSTEM	52
PROPELLANT LOAD SYSTEM	8
HYDRAULIC SYSTEM	45
PRESSURIZATION SYSTEM	112
TOTAL	1068
UNMODELED COMP. MASS	767
CENTAUR DRY WEIGHT	1835
RESIDUAL PROPELLANT	77
GASEOUS PROPELLANT	115
AUXILIARY PROPELLANT	59
CENTAUR WEIGHT / RESIDUALS	2086

and payload changeout. The OTV has two engines and is designed for a fully loaded manned sortie of 15.9 MT (35 klbs) payload to LLO from LEO and returning with only the manned capsule of 6.9 MT (15 klbs). The payloads larger than 15.9 MT (35 klbs) are delivered in multiple trips and manifested to fill the vehicles. The baseline OTV includes an aerobrake with mass equal to 15% of the reentry mass for the reference design mission. The baseline lander is a reusable, two-engine vehicle with capability of delivering 15.9 MT (335 klbs) (manned capsule plus payload) to the Lunar surface from Lunar orbit and returning to Lunar orbit with 6.9 MT (the mass of the manned capsule alone). The ASTROSIZE output masses for the baseline OTV and lander concepts are provided in Appendix D, and are summarized below:

OTV Mass, kg		Lander Mass, kg	
Dry (less AB & tanks)	1030	Dry (less landing gear & tanks)	1030
Aerobrake	3411	Landing Gear	1841
Oxygen Tank	367	Oxygen Tank	83
Fuel Tank	1,618	Fuel Tank	366
Propellant Capacity	83,892	Propellant Capacity	24.553
Payload	15,873	Payload	15,873
Mass Fraction:	0.93	Mass Fraction	0.88

The high mass fraction of the OTV is largely due to the large velocity increments of the LEO-to-LLO trip and the associated large amount of propellant. Figure 3-5 shows the variation of mass fraction as a function of delta-v and dry stage mass for fixed Isp and payload mass. Current vehicle technology exhibits mass fractions of upper stages to around 0.86. If tankage on current vehicles (e.g. Centaur) is increased to provide larger propellant capacities, the mass fraction will be driven higher; and this is the case here. The initial estimates for the dry mass for the OTVs and landers were both derived from the ELES code and considered identical.

The design of the Baseline OTV also allows transport of 33,682 kg of propellant (from LEO to LLO) for the lander. No propellant is carried back from the Moon in the baseline case.

3.3 Alternative Chemical Propulsion Concepts

Alternatives to the baseline propulsion system were selected based on recommended propellant candidates from Section 2.0, and on the propulsion sen-

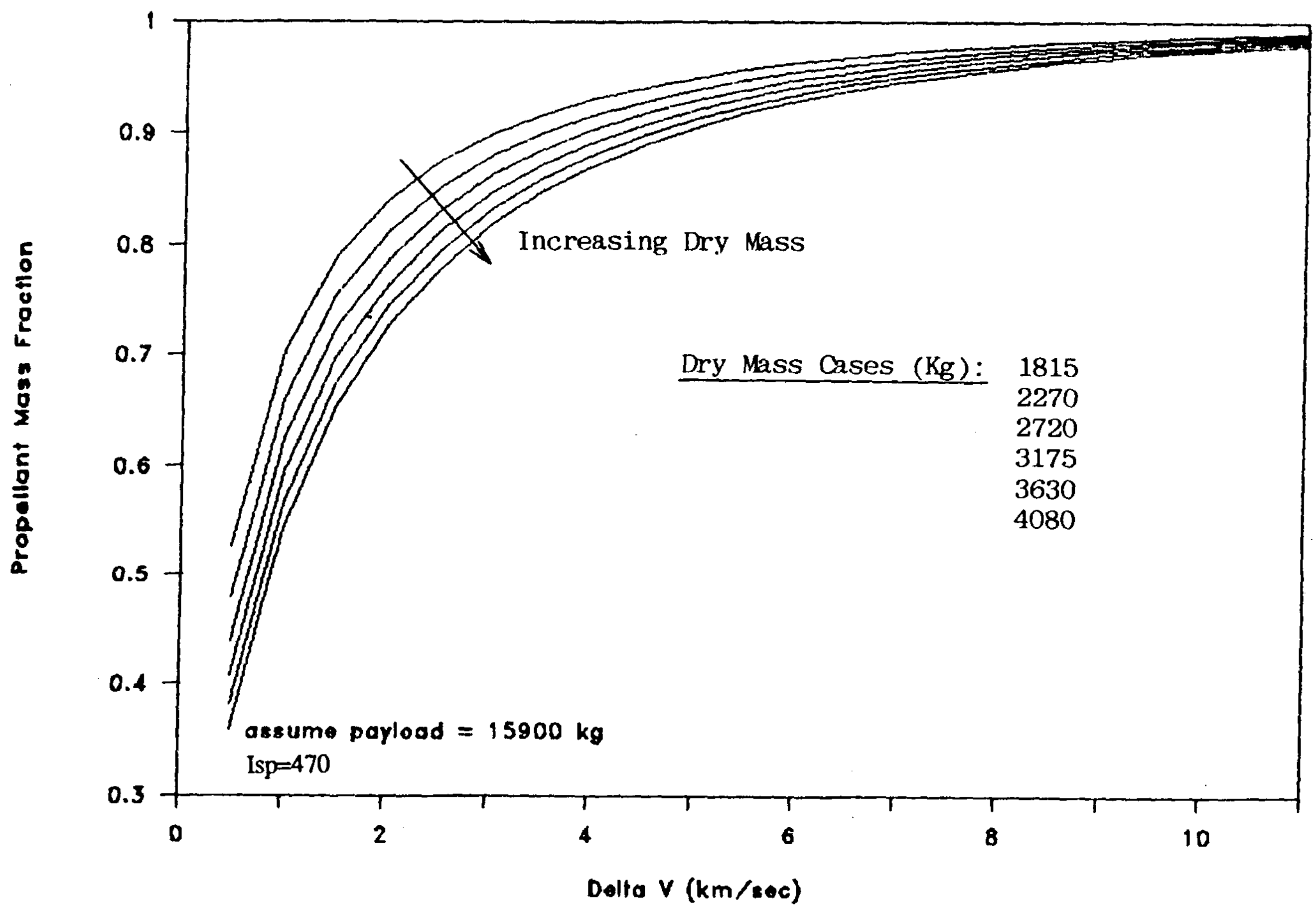


FIGURE 3-5. MASS FRACTION AS A FUNCTION OF DELTA-V AND DRY MASS

sitivities and trades identified in Figure 3-1. Section 5 will address technology requirements for each of the propulsion systems. Figure 3-6 presents a summary of candidate propulsion system performance characteristics.

3.3.1 Hydrogen/Oxygen Propulsion

Hydrogen/oxygen propulsion system options of interest can be obtained by varying mixture ratio, specific impulse, and thrust. Mixture ratios (mass ratio of oxidizer to fuel -- O/F ratio) of 5.5 (baseline), 8.7, and 10.6 were considered. High O/F ratios are of interest because oxygen is readily available on the Moon while Lunar fuels are problematic. Hydrogen/oxygen will theoretically maintain combustion potential through mixture ratios greater than 100 (Eagle Engineering, 1983). Mixture ratios of 35 and 50 have been studied (Waldron) with Isp's of 233 and 203 seconds and combustion temperatures of 2500 and 3000 C, respectively and will be discussed briefly in Section 4.3.4. Efforts in this study concentrated on mixture ratios within a factor of 2 of stoichiometric.

To account for uncertainty in baseline engine development, engine Isp of 460 and 490 were considered (baseline Isp = 470). Thrust variations were considered a reflection of the propulsion system design choices of the number and size of engines.

The goal of a Lunar-based propulsion system is to operate with propellants available on the Moon. For H/O engines, the higher mixture ratios will increase the percentage of oxygen used for any given mission. The hope was that by increasing utilization of Lunar oxygen, the propellant mass required from Earth, and, therefore, the Earth Launch Mass (ELM) would be reduced. However, as shown in Figure 3-7, the mixture ratio producing maximum specific impulse (Isp) is approximately 6, and variation of this mixture reduces the specific impulse. This reduction in performance increases the total propellant (fuel plus oxidizer) required for any given mission, which increases vehicle size and resources required for propellant production.

Therefore, there is a tradeoff. High mixture ratio engines can increase utilization of Lunar oxygen, but, due to performance (Isp) degradation, they will increase the total propellant required and the amount of fuel required from Earth. The question is, what is the net effect on ELM? Effects of the mixture ratio on the total ELM over the entire mission model is given in Section 4.3.4. The results indicate that high mixture ratios (>8.7) are not beneficial. ELM is increased.

SYSTEM TYPE	PROPELLANT	O/F	ISP	THRUST
BIPROPELLANT	H/O	5.5	470	7500
BIPROPELLANT	H/O	5.63	470	15000
BIPROPELLANT	H/O	8.7	421	7500
BIPROPELLANT	H/O	10.6	384	7500
BIPROPELLANT	H/O	12		7500
BIPROPELLANT	H/O	5.5	480	7500
BIPROPELLANT	MMH/O	1.43	371	7500
BIPROPELLANT	SiH4/O	0.78	366	7500
HYBRID SLURRY	Al/O	2.10	260	7500
TRIPROPELLANT	Al-H/O	3.1	---	7500

NOTE: ALL ENGINE CONCEPTS ARE PUMP FED

FIGURE 3-6. CHEMICAL PROPULSION CANDIDATES

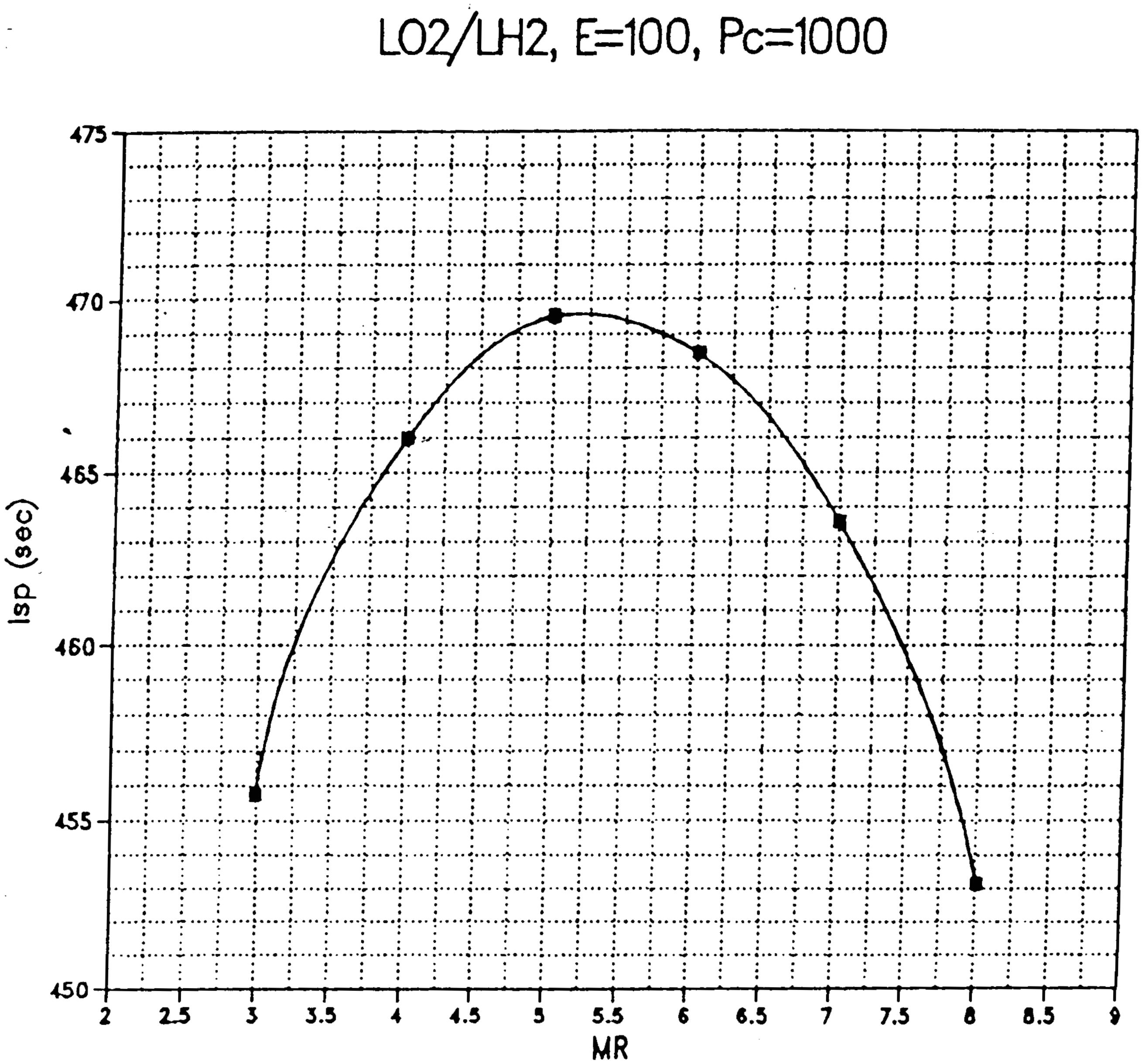


FIGURE 3-7. ISP AS A FUNCTION OF MIXTURE RATIO

Based on OTV studies, two different thrust levels, 33,363 N (7500 lb) and 66,726 (15,000 lb), were addressed. A tradeoff between number of engines and the overall propulsion reliability was made in those OTV studies with two engines being the preferred number. The propulsion concepts developed here nominally include two engines except where minimum accelerations and thrust to mass ratios cannot be met. This threshold is 1.8 for lander thrust to mass ratio and 0.1 for OTV thrust to mass ratio. The OTV thrust to mass ratio will determine the length of burn required to meet a given mission requirement. Thrust to mass ratios were monitored for lander configurations to detect when more than two engines were required.

The design characteristics of the propulsion systems are provided in Appendix C; overall vehicle mass estimates are provided in Appendix D.

3.3.2 Silane Propulsion

The silane/oxygen, $LSiH_4/LO_2$, bipropellant combination for use in Lunar base propulsion systems is a viable alternative to liquid hydrogen and oxygen as both the oxygen and the silane can be produced on the Moon. Silane is stable and storable in the space and Lunar environments with properties much like those of oxygen. The silane propulsion system developed here is very similar to a storable methane propulsion system with slightly higher Isp. The cycle uses a gas generator to pressurize and inject the propellants. The delivered Isp was estimated at 366 seconds, the nozzle exit area ratio was 300, the chamber pressure was 690 N/cm^2 (1000 psi), and thrust per engine was 33,363 N (7500 lbs) at an engine O/F mixture ratio of 0.78. Figure 3-8 shows the theoretical maximum Isp versus mixture ratio of 0.85 at a nozzle exit area ratio of 100. The basic engine design parallels the baseline hydrogen/oxygen propulsion systems. The silane engine will be fuel cooled as are hydrogen/oxygen systems. This presents a technology problem of providing a liquid silane circulation system. Fuel pump speeds are estimated at 59,000 rpms with a fuel boost pump to 19,700 rpms. The oxidizer pump speed also starts at 59,000 rpms with a boost pump to 31,000 rpms. The overall mass of each engine less fuel tankage is approximately 100 kg.

This conceptual design represents preliminary engine parameters. Chamber pressure of 345 (N/cm^2) (500 psi) with an expansion ratio of 20 was investigated resulting in a significantly lower Isp, 351.6 seconds, at an optimal mixture ratio (theoretical) of 0.82. Increasing the chamber pressure and expansion

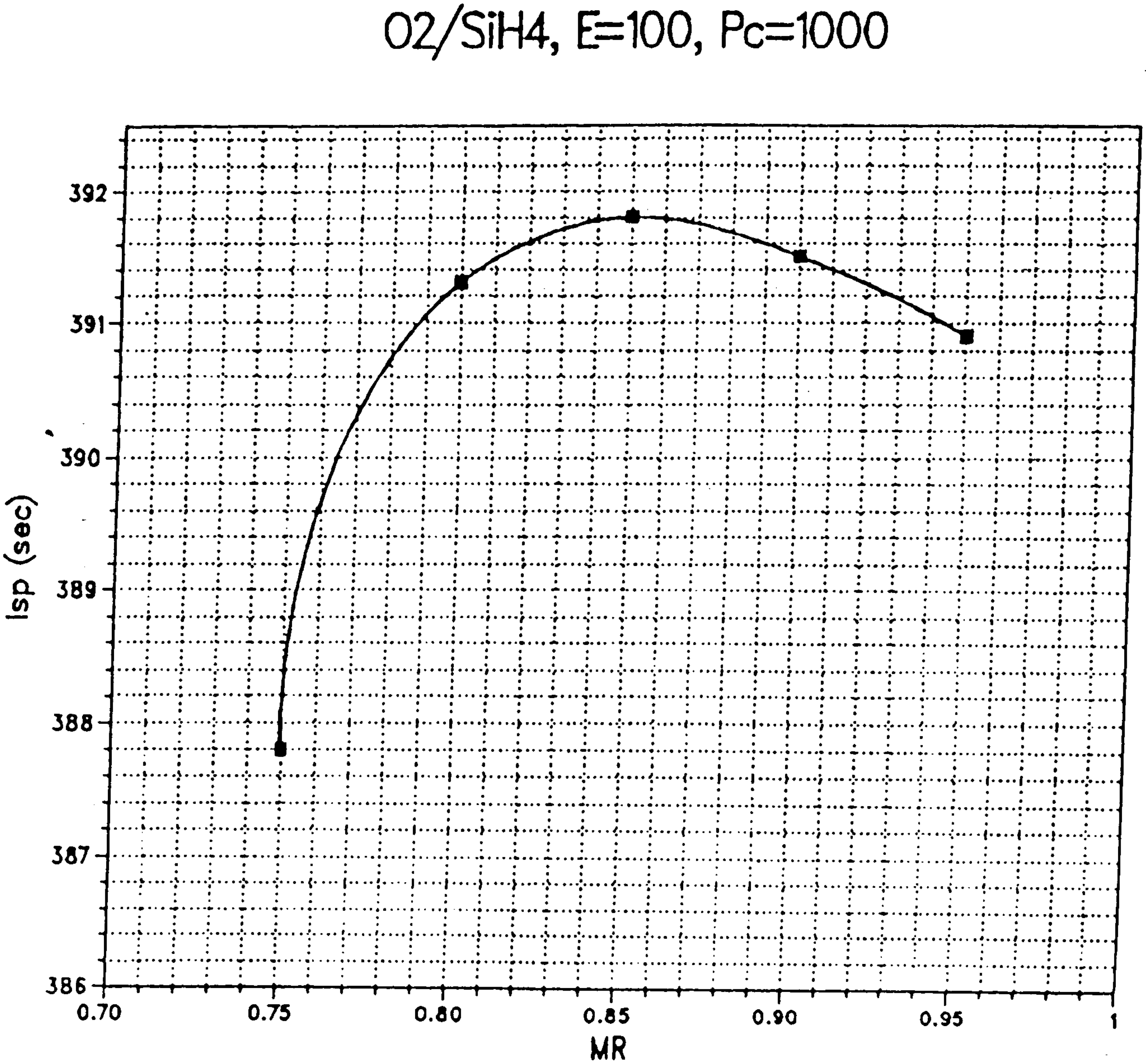


FIGURE 3-8. SiH4 / LOX ISP VERSES MIXTURE RATIO

ratio increased both the maximum Isp and the optimal mixture ratio. Additional trades among chamber pressure, expansion ratio, specific impulse, and mixture ratio should be analyzed.

3.3.3 Aluminum/Oxygen Hybrid Propulsion

An aluminum/oxygen hybrid propulsion system is another viable concept that could be used in an Earth-Moon transportation scenario. Both of these propellants may be produced using the same processing technique on the Lunar surface (see Section 2.0). Two different engine designs considered here were an aluminum/oxygen slurry, and a solid aluminum and liquid oxygen hybrid engine. Figure 3-9 provides a conceptual schematic of how solid aluminum rods may be fed into the oxygen injector nozzles to be burned in the combustion chamber. However, for modeling simplicity, the system analyzed to estimate the propellant system parameters was an aluminum/oxygen slurry. This system used a gas generator bleed cycle whose fuel (aluminum/oxygen slurry) and oxidizer (oxygen) tanks were both pressure fed with a cold gas (helium). The engine is cooled with the liquid oxygen, just prior to mixing with the aluminum and finally injected into the combustion chamber. The combustion temperature modeled was slightly above 4200 C with a combustion chamber pressure of 690 N/cm^2 (1000 psi). The resulting delivered Isp was 260 seconds. This concept also assumes a nozzle exit area ratio of 100, and a mixture ratio is 2.18. Figure 3-10 shows the theoretical Isp for pure aluminum and oxygen. The aluminum/oxygen slurry is stored at 3450 N/cm^2 (5000 psi), at 132 C in CRES 301 stainless steel tanks. The overall propellant flowrate is 51.7 kg per second.

The aluminum/oxygen slurry is a very dangerous propellant which may be very difficult to pump without inducing combustion. Thus, additional fluids may be introduced to substitute for the oxygen in the slurry. One of these fluids may be liquid hydrogen, which will be addressed in the next discussion of an aluminized-hydrogen/oxygen propulsion system.

3.3.4 Aluminized-Hydrogen/Oxygen Propulsion

To enhance the use of the Lunar derived aluminum and oxygen without the safety hazards associated with an aluminum/oxygen slurry, hydrogen may be used to fluidize the aluminum powder. This idea is not new, as aluminum has been added other liquid propellants and tested. Aerojet has tested the aluminum hydrazine mixture in the Titan vehicles. This propellant, called alumazine is really a gelled Aerozine-50 with aluminum suspended in it, with approximately

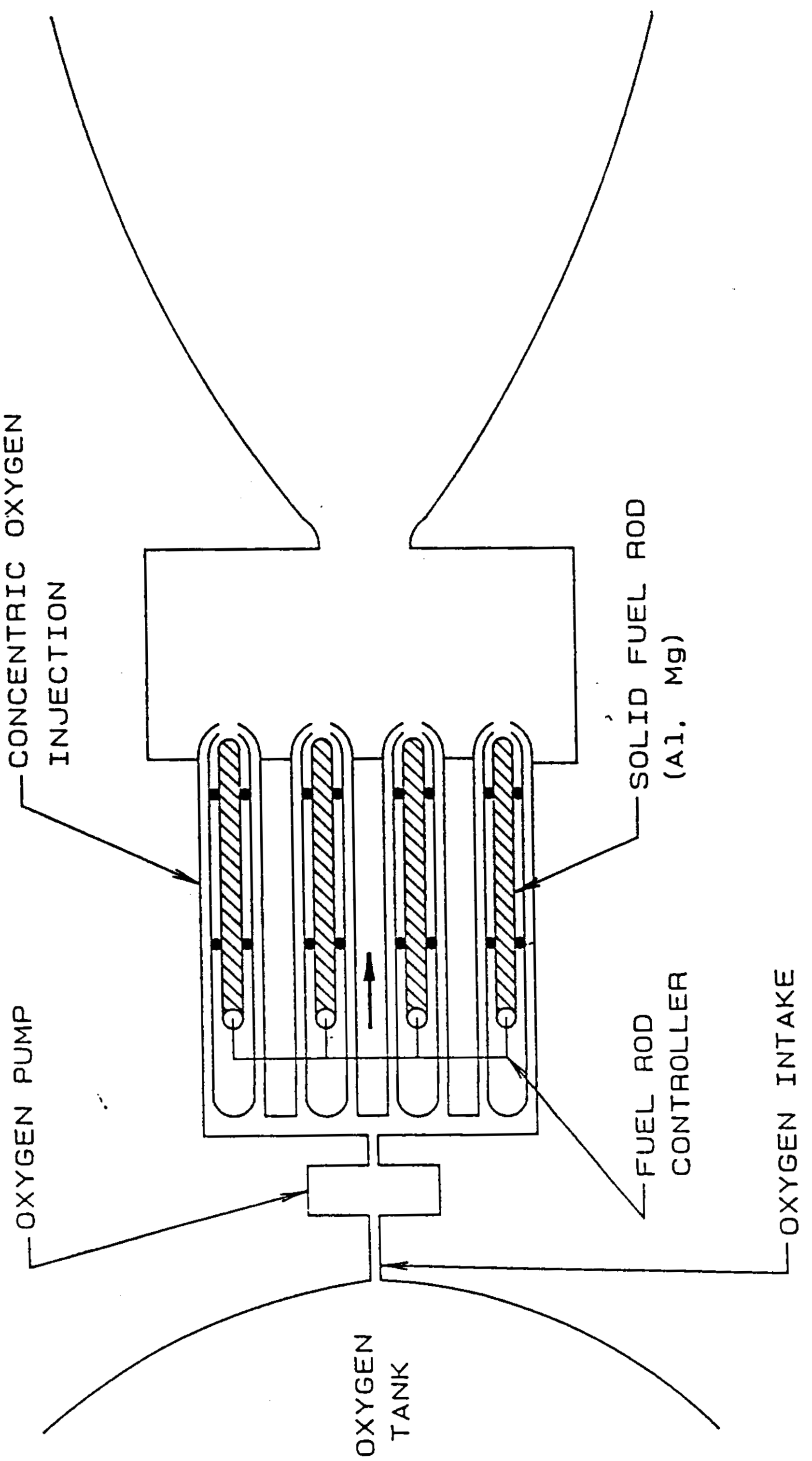


FIGURE 3-9. ASTRONAUTICS LOX / METAL HYBRID ENGINE CONCEPT

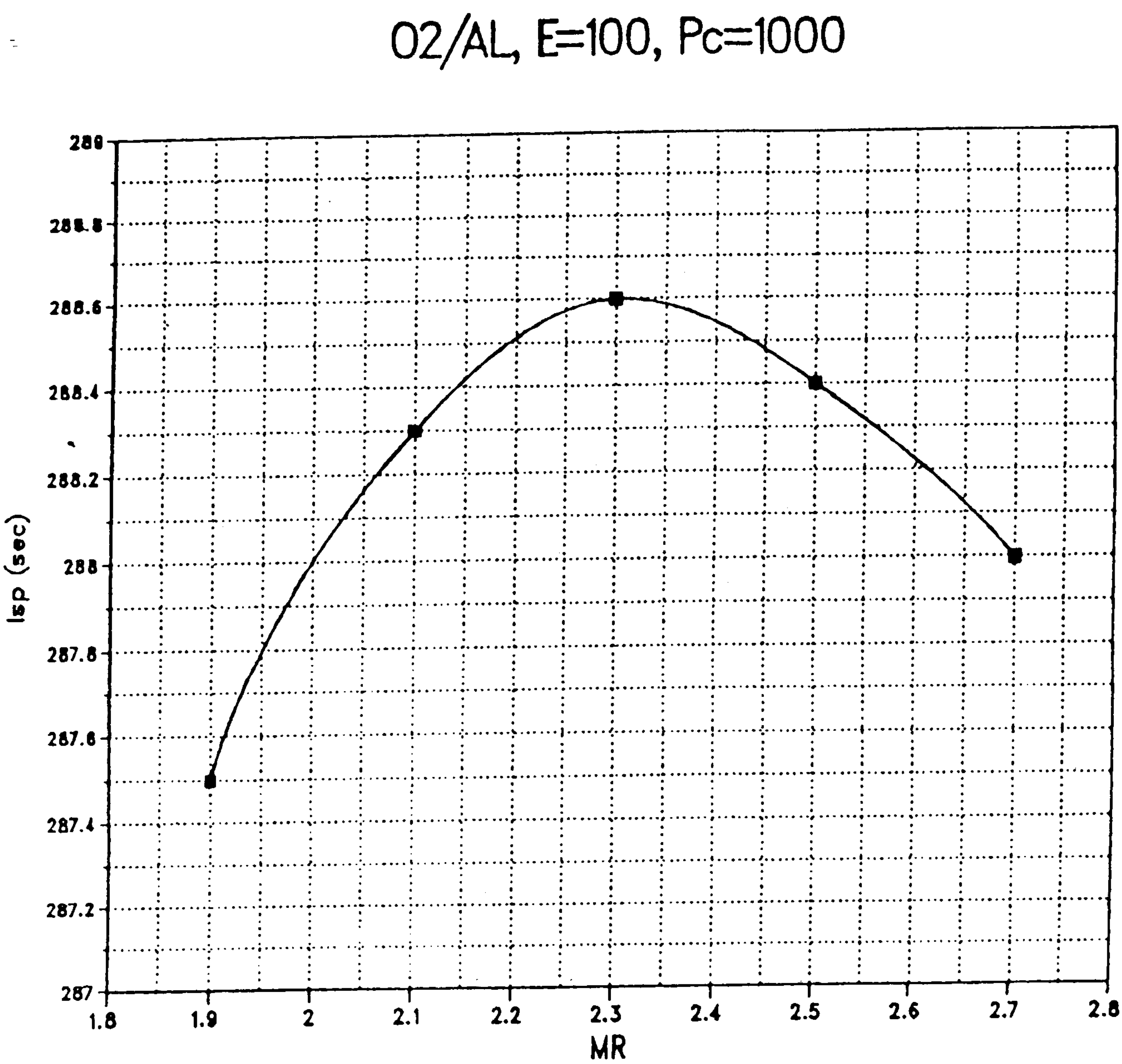


FIGURE 3-10. ISP VS. MIXTURE RATIO FOR ALUMINUM / OXYGEN

30% of the total propellant fuel weight being aluminum. The gel was achieved by addition of a substance called carbopol (about 1% by weight) which produces a thixotropic gel. The characteristics of this matrix is that it resides as a gel when it rests, but liquifies under sheer stress or vibration. When the stress or vibration is removed the matrix returns to a gel which accommodates a suspension of solid particles much more easily than pure liquid. The alumazine/N_2O4 was tested in a Titan II second stage in the early 1960s. Extremely good combustion efficiency was achieved, however the combustion zones were very hot and cooling was a definite problem as liquid hydrogen was not available for cooling. This concept was abandoned in the 1960s for hydrogen/oxygen and other propulsion technologies being developed at that time. However, the approach and the lessons learned from this technology development activity may be very useful in analyzing and developing new concepts for an aluminumized-hydrogen/oxygen system.

The purpose of this propulsion system is to mix hydrogen and aluminum into a fluid to be injected with oxygen into the combustion chamber. With the lack of large quantities of hydrogen on the Moon, the goal is to reduce the hydrogen to aluminum weight ratio, but still maintain the fluidized properties. An initial concept development has been pursued with 60% by weight hydrogen and 40% by weight aluminum. The combustion pressure would be 690 N/cm^2 (1000 psi) and nozzle expansion ratio at is 100. The mixture ratio of the aluminum and hydrogen with oxygen was varied from 1 to 5 with an optimum around 2.6, yielding a theoretical Isp of 472 seconds as shown in Figure 3-11. The delivered Isp was not modeled, but was estimated at about 400 sec by comparisons of theoretical and delivered Isp data. The overall propellant density at this mixture ratio is approximately 1250 kg/m^3 (0.0451 $lbsm/in^3$). The combustion temperature was estimated at 3270 K (5885 R). The engine mass without propellant tankage for the aluminumized hydrogen oxygen propulsion system was estimated at 140 kg per engine with engines that yield 33,363 N (7500 lbs) thrust each.

With its high performance, the aluminized-hydrogen/oxygen propulsion system has great potential and represents a bridge between H/O and Al/O propulsion; however, it is relatively unexplored technology. Fluidized aluminum engines have been addressed using helium as the fluid; however, no literature has been identified to date that addresses using hydrogen as the fluidizing mechanism. The 40/60 ratio of hydrogen to aluminum mass was chosen as an initial concept based on experience with aluminum in hydrazine.

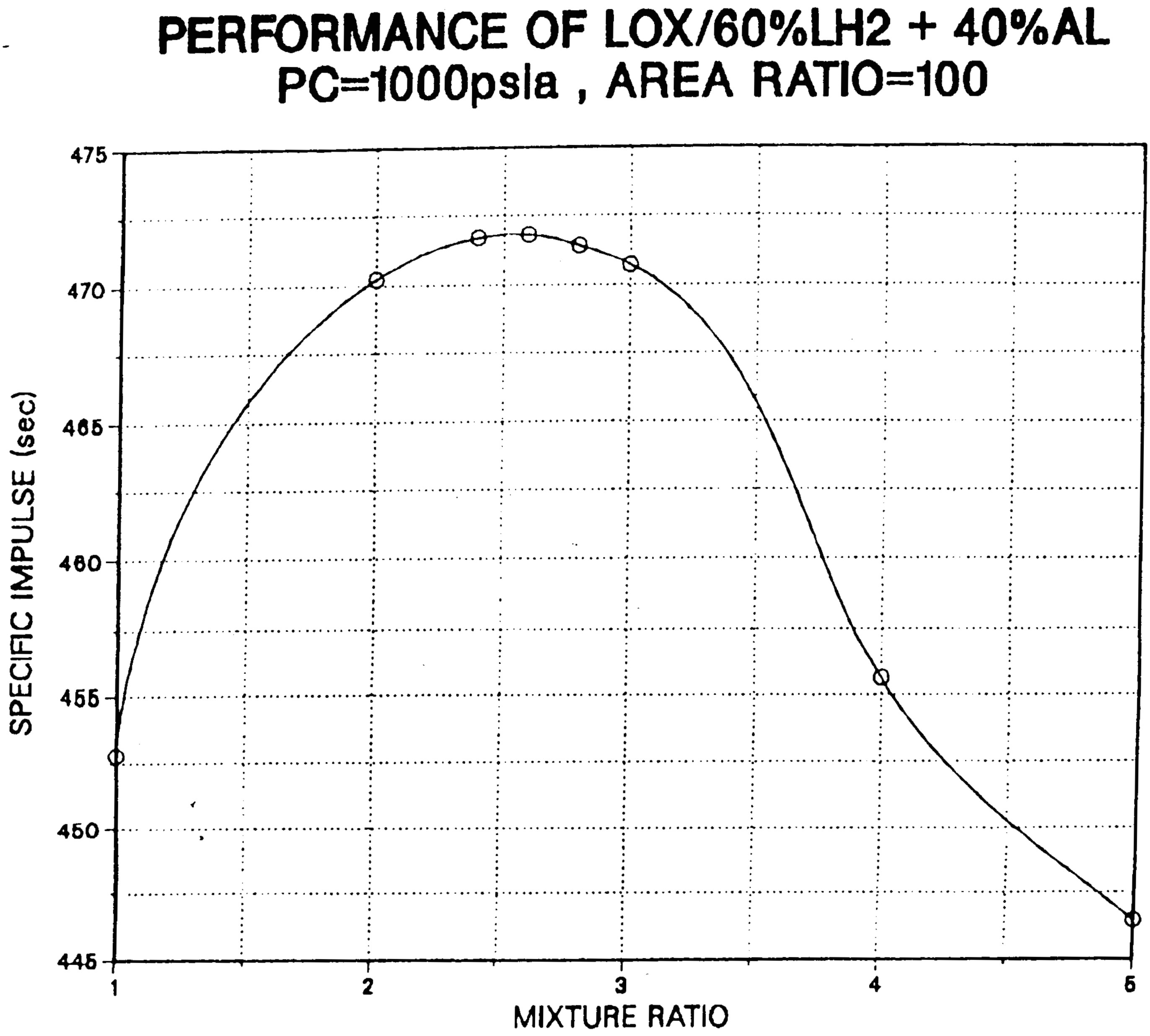


FIGURE 3-11. ISP VS. MIXTURE RATIO FOR ALUMINIZED-HYDROGEN / OXYGEN

A significant effort in modeling and analyses could contribute to the understanding and optimization of this propulsion system. The increase in the hydrogen/aluminum ratio may decrease the dependence on hydrogen, without significant reductions to performance. New concepts and technologies must be developed to handle the tripropellant system with various injection, mixing and propellant storage options.

3.4 Vehicle Systems Characterization

The major vehicle systems (other than propulsion) analyzed in this study included aerobrakes, landing gear, propellant tankage and support system mass. The support system mass included avionics structural support, thermal protection, fuel tank pressure weights, fuel boiloff, residual fuels, interstage mass, and miscellaneous masses all derived from the ELES code, which is based on Centaur data. The miscellaneous masses include tracking systems, range safety systems, auto-pilots, the electrical systems, the guidance systems, auxillary propellant systems, motors, attitude control systems, and any adaptors needed for the propulsion system itself. These mass estimates were given in Table 3-2. The aerobrake, landing gear, and tank masses were all calculated, for a given reference design mission.

Aerobrake masses were represented by a percentage of the reentry mass of the entire vehicle system. Typical Earth-produced aerobrakes range in masses from 15% to 50% of the reentry mass. We will see later in Section 4.0 that the aerobrake mass must not exceed about 35% of the reentry mass to be of use for Lunar base operations. The baseline aerobrake used in this study, given by JSC, was 15% of the reentry mass. Aerobrake sensitivity studies were run from 15% to 30% of Earth-produced aerobrakes. Another concept that may be quite valuable is that of a Lunar-produced aerobrake. JSC estimated that a Lunar-produced aerobrake mass would be approximately 18% of the reentry mass (Lunar Surface Return, 1984). Such an aerobrake would not have to be carried from LEO to Low Lunar Orbit and back again but simply from LLO to LEO. Also the Lunar derived aerobrake would not be part of the ELM. The aerobrakes in the vehicle families were calculated based on the reentry mass of design reference missions. The mission included: (1) return of OTV with full payload only; (2) return of OTV with full payload and a small portion of oxygen; and (3) return of OTV with full payload and enough oxygen for the next outbound leg from LEO to LLO. The third reference mission required an aerobrake of mass about 13 MT. Such a large aerobrake may not be viable.

Landing gear mass for a Lunar lander is a function of the landing mass of the Lunar Lander. Typical ratios of landing gear mass to total landing mass on the Moon are around 5%. Thus, a 5% lander mass was estimated for all Lunar Landers. No sensitivities were run on landing gear mass.

Propellant tankage is a major part of the total vehicle mass. However, advanced technologies in areas of tankage have allowed the specific mass of the tankage to be quite low, even for cryogenics. Tank estimates for vehicles developed here were derived from the Centaur data using CRES 301 stainless steel as the tank material, the tanks were configured in a cylindrical fashion with elliptical end using an elliptical ratio of 1.38. The thickness of the tank walls was estimated at 14 mils. Appropriate boiloff parameters were considered in the ELES code for various propellants with this tankage. Scaling of the tanks was done to accommodate specific reference design missions of the vehicle families and will be reported in the following section.

OTVs and landers were developed with reusability in mind. No expendable vehicles were addressed. Figure 3-12 shows life cycle cost of an expendable Centaur vehicle, a reusable ground based OTV and a reusable space based OTV. In all cases beyond the first eight years of the low mission model, the reusable OTVs were less costly. Even in the periods of the first nine years, the cost of the reusable OTVs over the expendable Centaur stage was not extreme. Expendable Lunar lander systems will burden the OTV greatly as unproductive mass delivered to the Lunar surface when compared to the reusable Lunar lander. This additional mass delivery would increase the total Earth launch mass by an order of magnitude and should only be considered for a short period of time in the initial stages of the Lunar base, if at all.

3.5 Propulsion/Vehicle System Family Descriptions

The propulsion and vehicle systems described in the last two sections were assembled in vehicle families (groups of vehicles that can satisfy the mission) to assist in providing data points for analysis. It is a goal that at least 2 points of each line on Figure 3-1 be addressed by a vehicle family system. The vehicle family systems were created by the ASTROSIZE computer code which utilized inputs from the ELES code and specific design reference missions to size the various vehicle subsystems; especially tankage, landing gear, and aerobrakes. All vehicle family systems were developed with the common basing scenario. The basing scenario includes the Space Station in LEO to provide servicing,

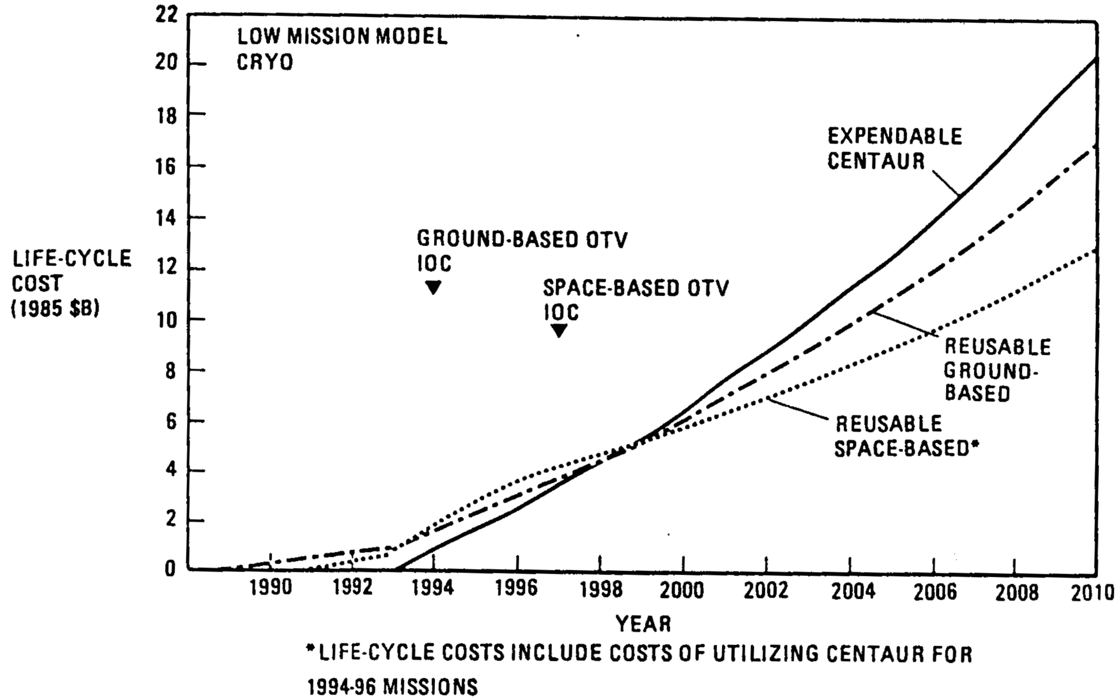


FIGURE 3-12. OTV REUSABLITY AND BASING TRADE STUDY

(GENERAL DYNAMICS, 1986)

payload accommodation and propellant supply. A similar basing node located at LLO will be needed ato operate as a propellant storage depot, a payload transfer from OTV to lander and for potential servicing of either the OTV or lander systems. The Lunar surface base was seen as a propellant supply source from which oxygen, aluminum, and silane could be supplied. In all, there are 30 different vehicle family systems, the ASTROSIZE vehicle mass outputs have been summarized and placed into Appendix D. A summary of these vehicle families is provided here.

Table 3-4 summarizes the mass and mass fraction data developed for all propulsion/vehicle family candidates. Each OTV in the table is identified by the letter "a"; each lander by the letter "b". OTVs were single stage vehicles providing transportation between LEO and LLO. The lander provided transportation between LLO and the Lunar surface. Single stage OTV/Lander concepts for transportation between LEO and the Lunar surface were analyzed in the first months of the study and ruled out because of high mass and propellant requirements.

Concepts with and without Lunar propellants were addressed. When a propellant source was not available on the Moon, the OTV was designed to accommodate enough propellant for its entire round trip and the lander trips required. When propellant was available on the Moon, the OTV did not have to carry all lander propellant or all propellant for its own return. The lander was responsible to deliver all payloads and OTV propellant required from the Lunar surface to LLO. However, the requirements for propellant delivery to LLO were not used to size the lander. The lander was sized based on the payload requirements. Multiple propellant delivery trips were provided where necessary. The parameters in parenthesis in Table 3.4 represent data for vehicle families designed without Lunar propellant being available.

The first family is comprised of an OTV and lander with a Baseline H/O propulsion system without Lunar propellants and with a nominal 15% mass aerobrake and 5% landing gear mass. The Baseline has only one reference mission: no Lunar propellant available. The second family represents the Baseline with Lunar oxygen being available and the third family represents the Baseline with both Lunar oxygen and Lunar hydrogen being available. Two reference missions exist for families two and three: (1) with Lunar propellant available, and (2) no Lunar propellant available. Note that the no Lunar propellant reference mission

TABLE 3-4. VEHICLE MASS SUMMARIES

VEHICLE		MASS, MT			OTHER		
		Engine	Propellant Tanks	A/B or Landing Gear	Dry	Total (less propellants)	Mass Fraction
1. BASELINE H/O	(a)	0.095	(1.99)*	(3.41)	0.84	(6.43)	(0.93)
	(b)	0.095	(0.45)	(1.84)	0.84	(3.32)	(0.88)
2. H/O with LLOX	(a)	0.095	1.33 (1.99)	2.89 (3.41)	0.84	5.25 (6.43)	0.89 (0.93)
	(b)	0.095	0.451 (0.45)	1.85 (1.84)	0.84	3.33 (3.32)	0.88 (0.88)
3. H/O with LLOX and LH2	(a)	0.095	0.545 (1.99)	2.80 (3.41)	0.84	4.37 (6.43)	0.87 (0.93)
	(b)	0.095	0.451 (0.45)	1.85 (1.84)	0.84	3.33 (3.32)	0.88 (0.88)
4. H/O OTV & Al/LOX Lander	(a)	0.095	0.597 (4.74)	2.08 (3.91)	0.84	3.70 (9.68)	0.87 (0.96)
	(b)	0.190	1.075 (1.070)	6.47 (6.45)	1.414	10.29 (10.26)	0.91 (0.91)
5. H/O OTV & Lander O/F = 8.7	(a)	0.095	0.982 (2.11)	2.40 (3.44)	0.84	4.41 (6.59)	0.91 (0.95)
	(b)	0.095	0.484 (0.483)	2.78 (2.77)	0.84	4.29 (4.39)	0.89 (0.89)
6. H/O OTV & Lander O/F = 10.6	(a)	0.110	0.932 (2.25)	2.75 (3.48)	0.84	4.74 (6.79)	0.93 (0.96)
	(b)	0.110	0.469 (0.466)	2.77 (2.77)	0.84	4.30 (4.41)	0.90 (0.90)
7. H/O OTV & Lander Isp = 460	(a)	0.095	1.36 (2.07)	2.95 (3.43)	0.84	5.34 (6.53)	0.89 (0.93)
	(b)	0.095	0.462 (0.461)	2.78 (2.78)	0.84	3.35 (3.35)	0.88 (0.88)
8. H/O OTV & Lander Isp = 490	(a)	0.095	0.997 (1.84)	1.74 (3.38)	0.84	3.76 (6.25)	0.90 (0.93)
	(b)	0.095	0.433 (0.433)	1.85 (1.85)	0.84	3.32 (3.32)	0.88 (0.88)
9. H/O OTV; No A/B H/O Lander	(a)	0.095	(2.97)	(0)	0.84	2.75(4.00)	0.94(0.97)
	(b)	0.095	(0.449)	(1.84)	0.84	3.33(3.51)	0.88(0.88)

(a) OTV (b) Lander * Number in brackets corresponds to reference mission without Lunar propellant availability

TABLE 3-4. VEHICLE MASS SUMMARIES, continued

VEHICLE		MASS, MT						
		Engine	Propellant Tanks	A/B or Landing Gear	Dry	Total (less Propellants)	Propellants	Mass Fraction
10. H/O OTV; 18% A/B	(a)	0.095	1.35 (2.02)*	3.57 (4.25)	0.84	6.15 (7.19)	42.0 (85.8)	0.88 (0.92)
H/O Lander	(b)	0.095	0.45 (0.45)	1.85 (1.84)	0.84	3.51 (3.52)	24.6 (24.6)	0.88 (0.88)
11. H/O OTV; 20% A/B	(a)	0.095	1.37 (2.04)	4.05 (4.85)	0.84	6.45 (7.93)	42.7 (87.1)	0.87 (0.92)
	(b)	0.095	0.45 (0.45)	1.85 (1.85)	0.84	3.33 (3.32)	24.6 (24.6)	0.88 (0.88)
12. H/O OTV; 25% A/B	(a)	0.095	1.41 (2.11)	5.34 (6.51)	0.84	7.76 (9.65)	44.8 (90.8)	0.85 (0.90)
	(b)	0.095	0.45 (0.45)	1.85 (1.84)	0.84	3.33 (3.32)	24.6 (24.6)	0.88 (0.88)
13. H/O OTV; 30% A/B	(a)	0.095	1.45 (2.11)	6.78 (8.41)	0.84	9.26 (11.63)	47.1 (95.1)	0.84 (0.87)
	(b)	0.095	0.45 (0.45)	1.85 (1.84)	0.84	3.33 (3.32)	24.6 (24.6)	0.88 (0.88)
14. SiH4/LOX OTV	(a)	0.100	0.392 (1.43)	1.90 (3.35)	0.84	3.33 (5.88)	49.3 (162.0)	0.94 (0.96)
& Lander	(b)	0.100	0.30 (0.30)	2.78 (2.78)	0.84	4.22 (4.22)	40.7 (40.6)	0.91 (0.91)
15. Al-H/LOX OTV	(a)	0.140	1.26 (3.11)	2.21 (3.65)	0.84	4.59 (7.88)	51.1 (138.7)	0.92 (0.95)
& Lander	(b)	0.140	0.67 (0.66)	2.78 (2.78)	0.84	4.71 (4.71)	37.7 (37.6)	0.89 (0.89)
16. H/O OTV & Lander	(a)	0.095	0.79 (1.45)	1.68 (2.25)	0.84	3.50 (4.74)	25.4 (60.0)	0.88 (0.93)
10 MT Payload	(b)	0.095	0.36 (0.35)	1.62 (1.62)	0.84	3.01 (3.00)	19.4 (19.3)	0.87 (0.87)
17. H/O OTV & Lander	(a)	0.095	1.43 (2.66)	2.70 (4.28)	0.84	5.15 (7.97)	46.6 (110.7)	0.90 (0.93)
20 MT Payload	(b)	0.095	0.63 (0.63)	2.78 (2.78)	0.84	4.54 (4.54)	34.6 (34.5)	0.88 (0.88)
18. H/O OTV & Lander	(a)	0.095	3.62	13.5	0.84	18.20	74.3	0.82
with LLOX Return **	(b)	0.095	0.45	1.85	0.84	3.34	24.7	0.88

(a) OTV (b) Lander * Number in brackets corresponds to reference mission without Lunar propellant availability

** 69.4 MT LLOX Returned to LEO

represents the Baseline case. As more propellant becomes available, the propellant tank mass for the OTV decreases by about 1.5 MT; oxygen availability reduces tankage mass by about 0.7 MT while hydrogen availability saves an additional 0.8 MT. The oxygen availability also yield a 0.5 MT savings to the aerobrake mass with Lunar hydrogen saving an additional 0.1 MT. The total propellant requirement for 15.9 MT payload delivery/return is reduced by over 50% with Lunar oxygen and by over 65% with both Lunar oxygen and hydrogen. The overall effect is a reduction of vehicle mass fraction requirements from 0.93 to 0.89 and 0.87 for Lunar oxygen and Lunar oxygen/hydrogen, respectively. Thus, availability of oxygen from the Moon reduces the overall technology of requirement of the single stage OTV to conduct a Lunar mission.

The fourth family shown is the baseline OTV with an aluminum/oxygen lander. The OTV mass parameters change because of the lander propellant requirements. The "no Lunar propellant" reference mission is not a logical mission because of the large amounts of propellants required from Earth for the lander. The size of the lander (when compared to the baseline lander) increases dramatically in every category. However, when both the aluminum and oxygen are available on the Moon, the lander becomes totally Moon-dependent and the OTV becomes partially Moon-dependent. The result is an OTV mass about 57% of the Baseline OTV with a mass fraction of 0.87. The total propellant requirement increases, but all lander propellant and about 10% of OTV propellant are derived from the Moon. This vehicle family is very beneficial for Lunar-basing and can be evolved from the Baseline by the addition of an oxygen/aluminum processing and propulsion systems.

The fifth and sixth vehicle families are very similar to the Baseline with the exception of higher mixture ratio propulsion systems. The reference missions of "no Lunar propellant available" are not logical options because they increase the propellant requirement from Earth. In both cases, the Lunar propellant available is oxygen; thus, comparison to Baseline with Lunar oxygen, family #2, is logical. A slight reduction in overall vehicle mass is gained by the higher mixture ratio systems but required mass fractions increase from 0.89 for mixture ratio of 5.5, 0.91 for mixture ratio of 8.7, and .93 for mixture ratio of 10.6.

The seventh and eighth vehicle families also are very similar to the Baseline, changing only Isp from 470 to 460 and 490 seconds, respectively. A reduction of Isp may increase the reliability of the engine by "unstressing" components. The reduction to 460 seconds increases both vehicle and propellant masses by less than 5% of the Baseline, but does not change the mass fraction. Increasing Isp to 490 reduces vehicle and propellant masses by 3% and 8%, respectively, without Lunar propellant. The same increase in Isp in the Lunar oxygen scenario reduces the vehicle and propellant masses by 18% and 28%, respectively. Thus, increasing Isp is much more effective in a Lunar propellant scenario. However, the increased Isp would significantly increase engine stress levels.

The ninth vehicle family shown is also a pertubation of the Baseline without an aerobrake. The tenth, eleventh, twelfth, and thirteenth vehicle families represent the Baseline with varied specific masses of the aerobrake of 18%, 20%, 25% and 30% as shown. The all-propulsive OTV is 50 to 60% less massive than the Baseline in the "no Lunar propellant" reference mission, Family #1, and Family #2 in the Lunar oxygen scenario. However, the propellant requirements are approximately 69% higher without Lunar oxygen and 10% higher with Lunar oxygen. Thus, Lunar oxygen availability makes the aerobrake and the all-propulsive vehicle designs much more competitive. The aerobrake tends to lower the mass fraction of the OTV as shown in Figure 3-13. Also shown in Figure 3-13 is the mass fraction reduction as the relative mass of the aerobrake increases. The mass of the vehicle increases from about 50% for aerobrakes mass increase from 15% to 30% of reentry mass while the total propellant requirement increases only about 14%. This result shows the benefit of the aerobrake, however, increasing aerobrake mass incurs diminishing returns of the benefit until the total propellant requirement from Earth is equal to the all-propulsive vehicle. This is discussed in the overall system analysis results of Section 4.3.2.

The fourteenth vehicle family is the Silane/oxygen OTV and lander using the 15% aerobrake, 5% landing gear weight. The Silane vehicle families show quite a reduction in vehicle mass because of the opportunity to utilize the lunar propellants for both the OTV and Lander. This vehicle family would not be used if Lunar propellants were not available. Because of the lower Isp, the mass fraction of the Silane OTV is 0.94, which is high when compared to the H/O OTV (with Lunar oxygen) of 0.89.

The fifteenth family shown is aluminized-hydrogen/oxygen OTV and lander with the Baseline aerobrake and landing gear. The aluminized-hydrogen/oxygen

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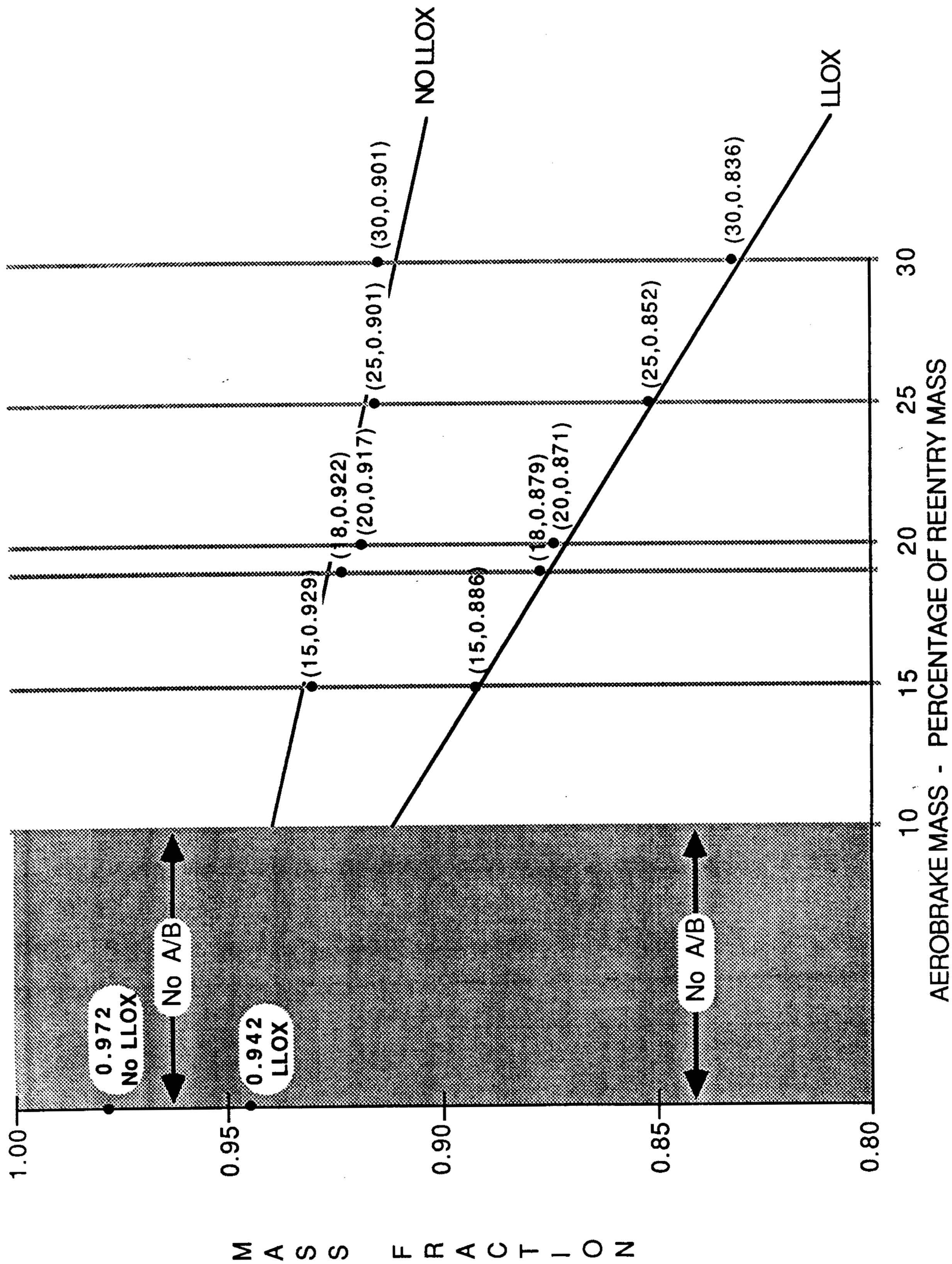

FIGURE 3-13. TOTAL SYSTEM MASS FRACTION VS. AEROBRAKE MASS

(Al-H/O) family also shows a reduction in mass from the H/O families, but not as great a reduction as the Silane vehicles. The Al-H/O system is more massive than the Silane system largely because of propellant tankage requirements. The mass fraction of the Al-H/O system falls between the H/O and Silane systems at 0.92. Without Lunar propellant availability, the Al-H/O system is not a logical transportation system.

Sixteenth and seventeenth vehicle families are perturbations of the Baseline with modified payload capabilities being 10 MT and 20 MT as shown. Little effect of payload was noticed when a constant total delivery mass to the moon is considered.

The eighteenth vehicle family is comprised of a baseline H/O OTV and lander with the nominal aerobrake and landing gear that provides the capability of returning enough Lunar oxygen to LEO to support its initial trip from LEO to Lunar orbit. Thus, this last case represents a vehicle family that has its fuel totally Earth-supplied, but its oxidizer totally Lunar-supplied. The amount of Lunar oxygen returned in this last vehicle family is approximately 69.4 MT. However, the aerobrake becomes extremely massive (13.5 MT) and the propellant requirements equally massive. Although the vehicle is huge, the mass fraction is relatively small at 0.82. In this case, the "cost" of Lunar propellant production becomes especially important.

The mass fractions of these vehicle families vary between 0.87 and 0.97. Higher mass fractions are found in the vehicles without Lunar base propellants. This is logical as more propellant must be carried to the Moon for the return trip of the OTV and the additional propellant tankage, which have relatively low specific mass, do not greatly increase the total vehicle dry mass. The mass fractions of the lander vehicles are relatively insensitive to the availability of Lunar propellant, due to the fact that propellant delivery to LLO from the Lunar surface was not used to size the lander. The propellant was accommodated through multiple lander trips.

In summary, looking at the different propellant alternatives, it appears that the hydrogen/oxygen systems have the lowest mass fraction, followed by the aluminized-hydrogen/oxygen OTV and the silane/oxygen OTV. These relationships also hold for the landers of these propellant options. The relationship between the vehicle mass fractions and the payload capabilities is relatively straightforward, heavier payloads require vehicles with higher mass fractions. When looking at aerobrake mass changes, heavier aerobrakes (those with higher percent mass) actually reduce the vehicle mass fraction. Also notice that for the case of no aerobrake, mass fractions required are extremely high. From these mass fraction estimates, one may deduce that some type of aerobrake may be necessary for efficient transportation to the Moon simply because we are not able to build an OTV with mass fractions as high as 0.972. However, note that the availability of Lunar oxygen significantly reduces the vehicle mass fraction.

A final note about the Baseline H/O OTV and lander designed to return enough oxygen to supply its next trip from LEO to LLO. The mass fractions of this vehicle are very low, around 0.82, but the dry mass of the vehicle itself is high, mainly due to the extremely heavy and large aerobrake required. Such aerobraking capability does not exist.

This vehicle family data was input to the program, ASTROFEST, to produce data on number of flights, propellant requirements, and total Earth launch mass for the entire mission model scenario given. Results are given in Section 4.0.

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4.0 LUNAR SURFACE BASE MISSION SENSITIVITIES AND TRADEOFFS

One of the goals of this study was to examine variations of propulsion system design characteristics and alternatives in lunar base mission operations and systems. Section 2 describes the propellant alternatives and their associated processing techniques. Section 3 describes the propulsion and vehicle systems alternatives and develops vehicle families (a "family" is a selected combination of vehicles (e.g. a hydrogen/oxygen (H/O), orbital transfer vehicle (OTV) and an aluminum/oxygen launch lander sized to satisfy all of the requirements of the mission model). This section applies those vehicle family alternatives to the Lunar base mission model to derive cost and Earth launch mass requirements for each alternative. This section describes the analyses and results of study Task 3 (Assessment of Transportation Systems and Operations) and determines which propellants and propulsion/vehicle systems are most efficient with respect to the Earth-Moon transportation system. Both transportation system cost and total Earth Launch Mass (ELM) will be considered. The analysis flow is depicted in Figure 4-1. To develop the data with which to compare alternatives, the mission model was manifested out to vehicles within a family to produce a traffic model which determines total propellant requirements at the various nodes. These propellant requirements then allowed estimation of propellant processing resource requirements required at the different nodes. The resource requirements then were integrated into the mission model as support requirements. The traffic model was updated as a result of additional propellant and propellant processing resource requirements. Iterations were performed to arrive at the total ELM for the given mission model scenario. Once the ELM was estimated and the final traffic model determined (with specific flight rates for the OTV and lander vehicles), the total transportation cost was determined by estimating the DDT&E, vehicle unit, and launch costs.

This section includes the cost estimates, total ELM estimates, and finally the results of sensitivity and tradeoff analyses. The sensitivity and tradeoff analyses will address the effects of Lunar propellant availability, specific aerobrake mass (with and without Lunar propellant availability), propellant processing consumable requirements, mixture ratio, specific impulse (Isp), payload mass, and Space Shuttle scavenging on ELM.

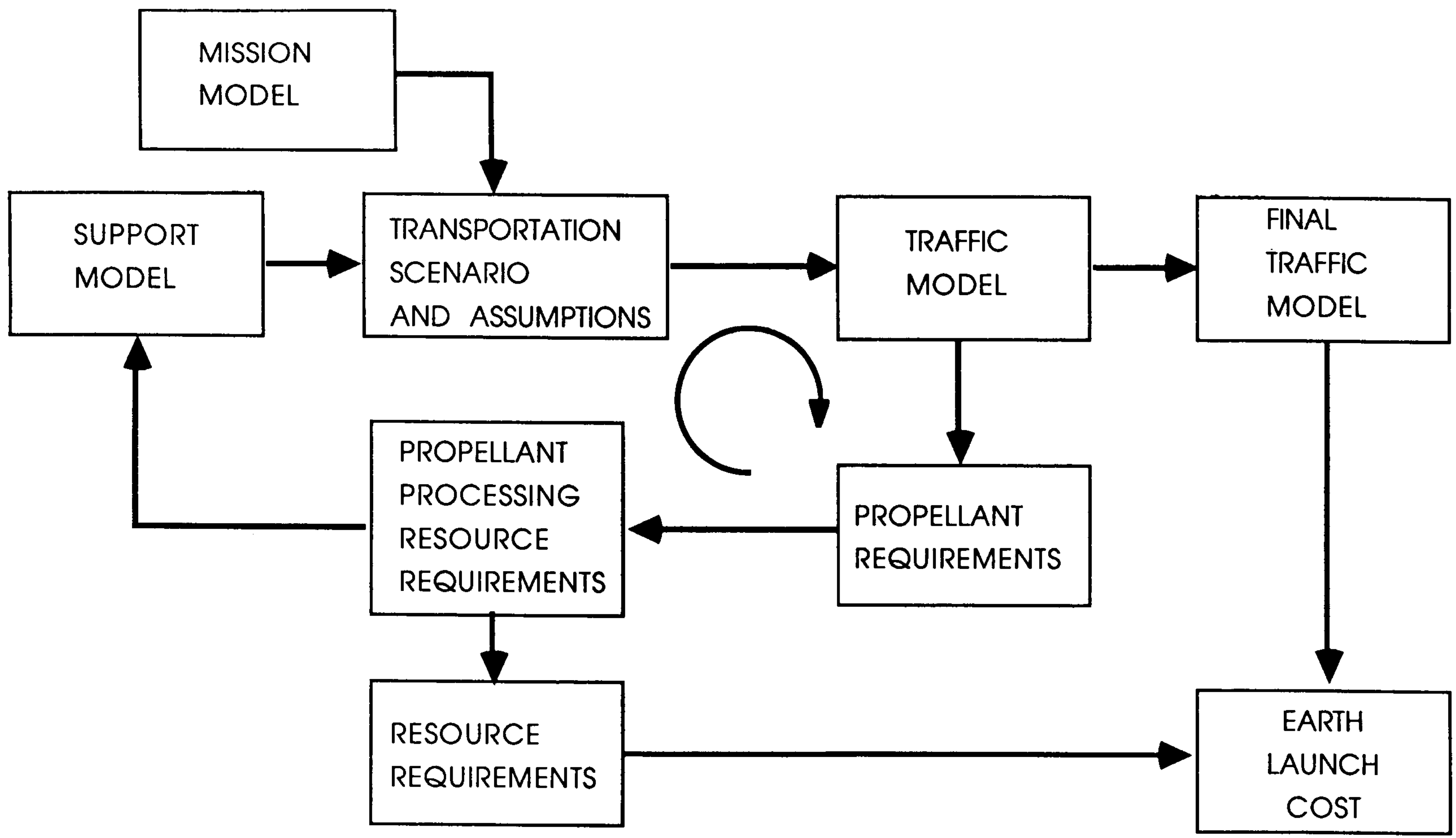

FIGURE 4-1. FAMILY STUDY FLOW

4.1 Relative Costs of the Lunar Base Mission Model Scenarios

The costs of the Lunar base mission scenarios are based on the total Earth launch mass for a given vehicle family, the procurement cost of the vehicles, and the DDT&E costs of those vehicle/propulsion systems. These cost figures are meant for relative comparison among vehicle families for the three major cost elements. Total ELM cost will include the cost of launching the mass of the missions themselves (the payloads), the propellants to deliver those payloads, the equipment and consumable resources required to produce propellants on the Moon, and the additional propellant required to deliver those resources. The vehicle procurement costs are relatively small because of the small vehicle flight rates (78-84 flights over 20 years) for landers and OTVs in most vehicle family scenarios. The DDT&E costs for the various propulsion vehicle systems are estimated based on available OTV estimates. Figure 4-2a and b shows the total transportation cost for each vehicle family, and ELM cost, vehicle DDT&E cost and production/acquisition costs. (Figure 4-2a is a legend that defines vehicle families identified by code number in Figure 4-2b; Figure 4-2b gives costs for vehicle concepts assuming Lunar propellant availability in 1995). The lowest cost vehicle family is the baseline hydrogen/oxygen OTV and lander with both Lunar oxygen and Lunar hydrogen available at a cost of about $6 billion dollars over the 20-year mission model. This does not include any of the Lunar base habitats or laboratories themselves, nor does it include the Low Earth Orbit (LEO) or Low Lunar Orbit (LLO) space station capabilities. DDT&E and production costs range from $2 to 4 billion dollars for all vehicle families. Note that in every case the ELM cost is still the major portion of the total transportation costs of the Lunar surface base mission scenario. Thus, reduction of Earth support becomes a major design driver of the vehicle/propulsion design. The following subsections discuss the cost estimates derived from the ELM and from the vehicle DDT&E and production estimates.

4.1.1 Earth Launch Mass Cost

Figure 4-3 shows the total ELM for the various vehicle families for cases of no Lunar propellant available, Lunar propellant available after 2005 and Lunar propellant available from 1995 through 2015. The numbers on the horizontal axis of this bar chart relate to the key in Figure 4-2a. Again, one can see that the baseline hydrogen/oxygen propulsion/vehicle system with both oxygen and hydrogen available on the Moon requires the least amount of resources from the Earth. Following as a close second is family number three which is the hydrogen/oxygen baseline OTV with an aluminum/oxygen lander.

(0) BASELINE, H/O OTV & LANDER (NO LUNAR PROPELLANT)

(1) H/O OTV & LANDER (LLOX AVAILABLE), DRY WEIGHT REDUCTION

(2) H/O OTV & LANDER (LLOX & LH_2 AVAILABLE), DRY WEIGHT REDUCTION

(3) H/O OTV & Al/ LOX LANDER; LANDER Isp = 260, O/F = 2.1

(4) H/O OTV & LANDER; OF = 8.7 (Isp = 421)

(5) H/O OTV & LANDER; OF = 10.6 (Isp = 384)

(6) H/O OTV & LANDER; I_{sp}= 460

(7) H/O OTV & LANDER; I_{sp}= 490

(8) H/O OTV & LANDER; PAYLOAD = 10MT

(9) H/O OTV & LANDER; PAYLOAD = 20MT

(10) H/O OTV & LANDER; NO AERO BRAKE

(11) H/O OTV & LANDER; AEROBRAKE MASS = 18% OF REENTRY MASS

(12) H/O OTV & LANDER; AEROBRAKE MASS = 20%

(13) H/O OTV & LANDER; AEROBRAKE MASS = 25%

(14) H/O OTV & LANDER; AEROBRAKE MASS = 30%

(15) SiH_4 OTV & LANDER

(16) Al-H/LOX OTV & LANDER

(17) H/O OTV & LANDER WITH LLOX RETURN TO LEO

FIGURE 4-2a. LEGEND FOR VEHICLE SUMMARY

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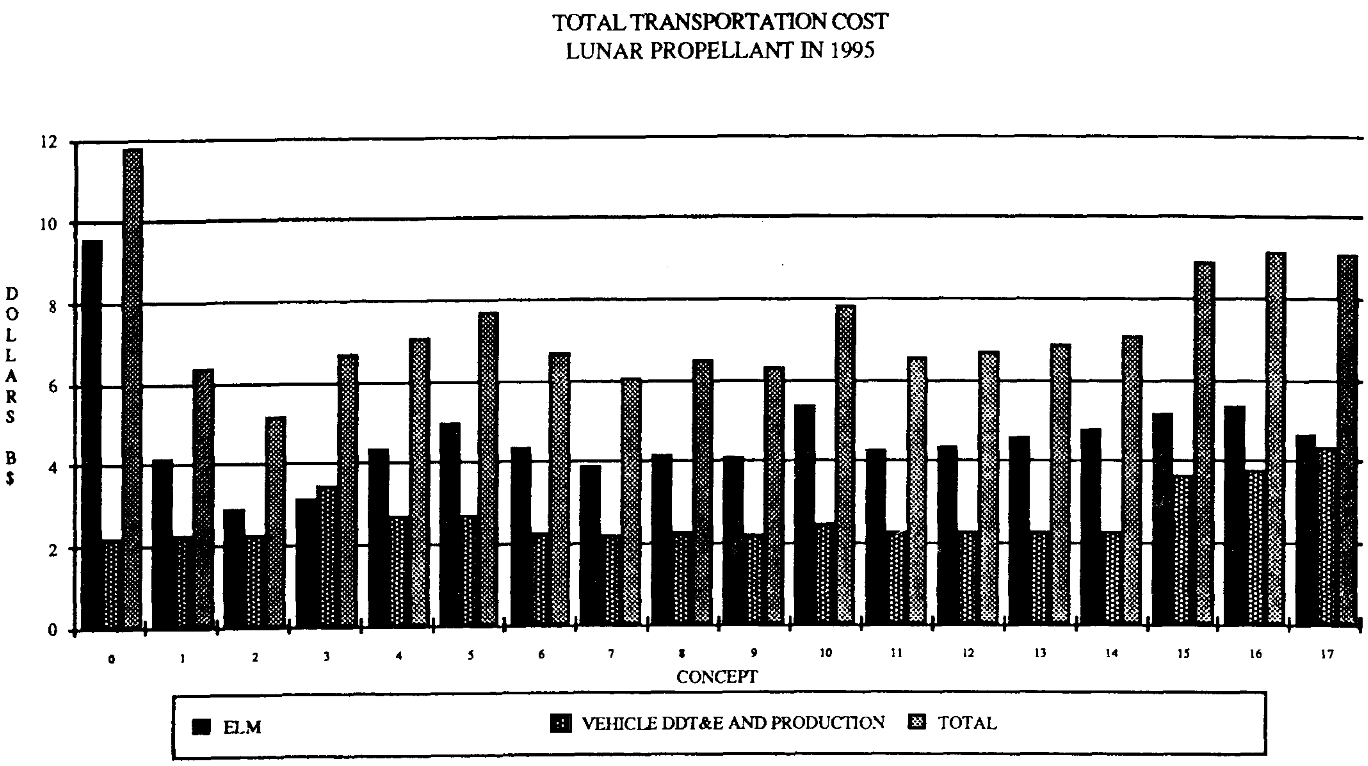

**FIGURE 4-2b. RELATIVE TRANSPORTATION COST
LUNAR PROPELLANT IN 1995**

VEHICLE FAMILY RESULTS SUMMARY

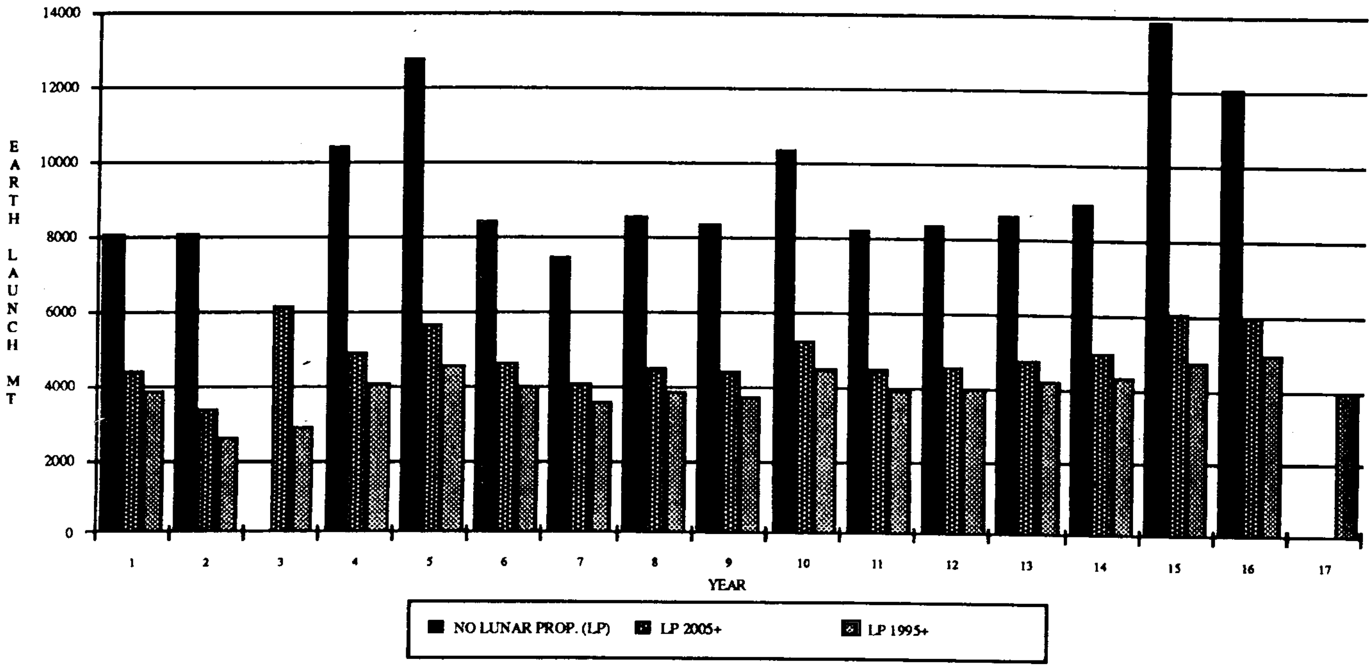

FIGURE 4-3. VEHICLE FAMILY RESULTS SUMMARY (BASED ON ENTIRE MISSION MODEL)

The data that was used to derive the ELM is provided in Appendix D from the ASTROFEST model. That data lists the propellant requirements at the Lunar surface and at LEO for each year of the mission model and sums up the total propellant and resource support requirements for the entire mission model. A comparison of some of the more promising vehicle families is provided in Figure 4-4 for each year of the mission model. The vehicle families included in this chart are Baseline hydrogen/oxygen family (0), the hydrogen/oxygen OTV with the aluminum oxygen lander (3), the hydrogen/oxygen OTV and lander concept with Lunar oxygen and Lunar hydrogen availability (2), the hydrogen/oxygen OTV and lander with Lunar oxygen availability (1), and, for comparison, the silane OTV and lander (15) and the aluminumized-hydrogen/oxygen OTV and lander (16) concepts. The number in parenthesis refer to Figure 4-2a and the number of the line on the graph.

The majority of ELM is propellant required to deliver both the payloads and the Lunar propellant processing resources. Figure 4-5 provides the Earth, Moon, and total propellant requirements for the various vehicle families. The vehicle family with the lowest total propellant requirement is not necessarily the family with the lowest Earth-supplied propellant requirement. The hydrogen/oxygen OTV with the aluminum oxygen lander which had the lowest total ELM requirement has one of the largest total propellant requirements; most of the propellant is Lunar-supplied. The hydrogen/oxygen OTV and lander with both oxygen and hydrogen available from the moon has by far the lowest Earth-derived propellant requirement and has the second lowest total propellant requirement. These variances are largely due to the differences in Isp and the differences of the propellant processing requirements. An assessment of the effects of Lunar propellant processing on the total Earth launch mass of these various vehicle families will be addressed later in Section 4.3.

The relationship between the ELM and cost is simply the delivery cost per pound to low Earth orbit. This ELM cost has been estimated at $2400 a pound (A.D. Little, 1986).

4.1.2 Vehicle Development and Production Cost

As was shown in the total cost figures, vehicle design, development, testing, and engineering (DDT&E) is not an insignificant portion of the total cost of the Lunar base mission scenario. This cost may be broken down into the cost of buying new vehicles as they are expended and the cost of design, deve-

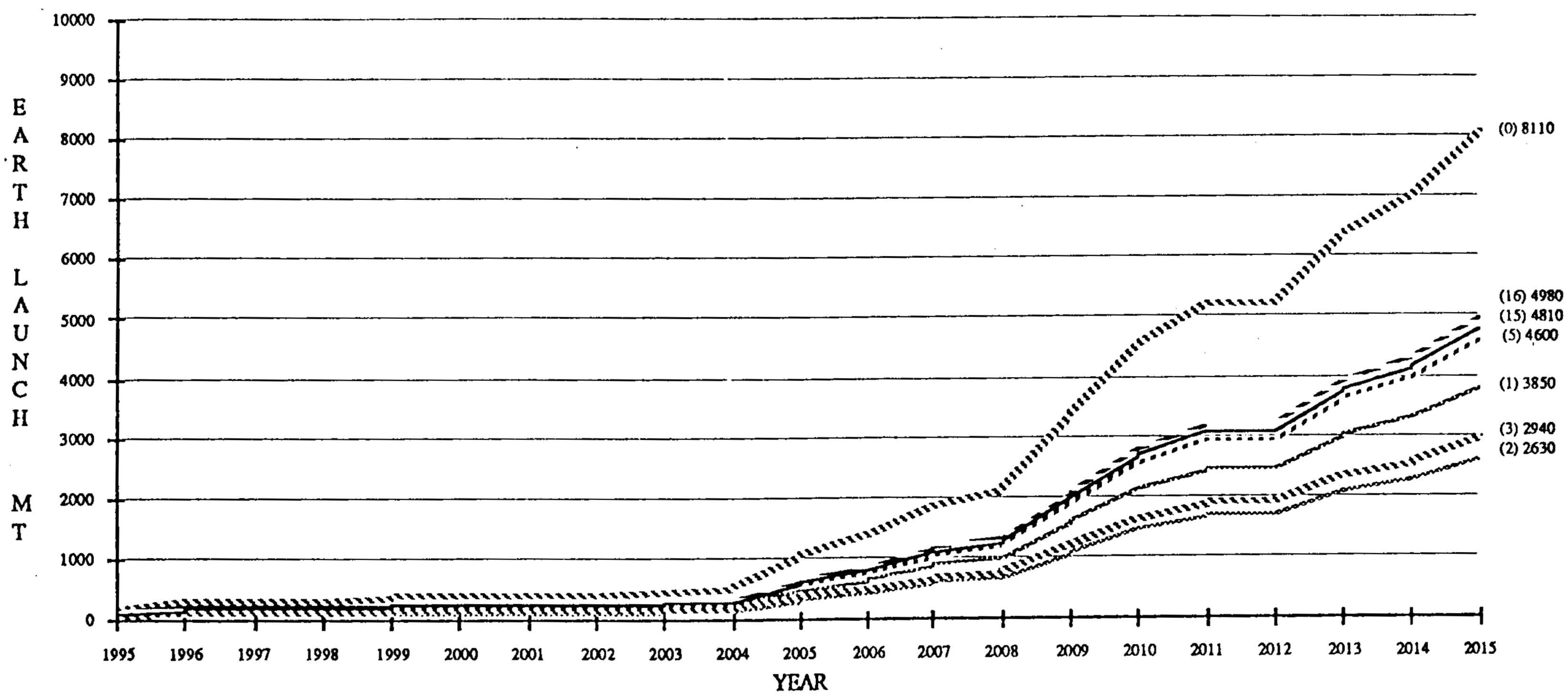

FIGURE 4-4. CONCEPT COMPARISON - LUNAR PROPELLANT IN 1995

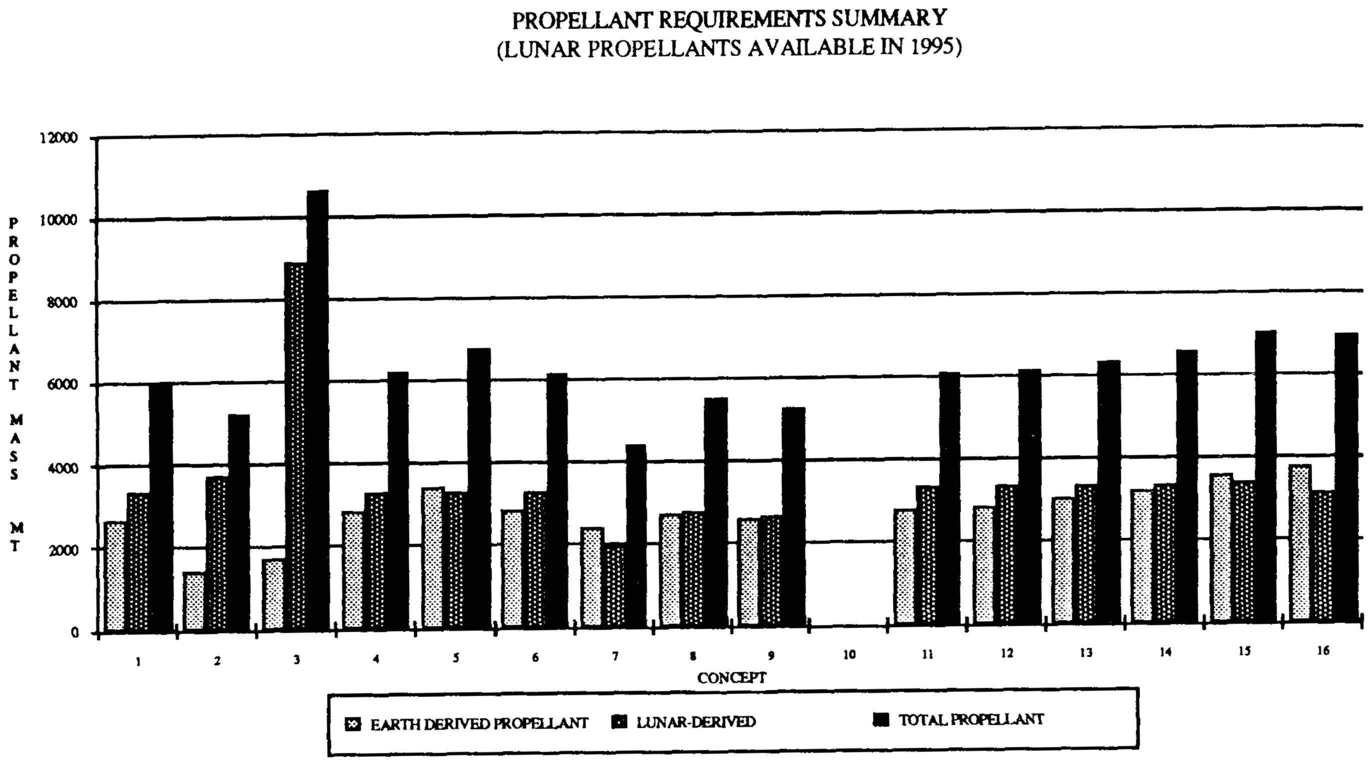

FIGURE 4-5. PROPELLANT REQUIREMENT SUMMARY
LUNAR PROPELLANT IN 1995

lopment, and testing, and engineering (DDT&E). Flight operations costs including aerobrake replacement and vehicle maintenance are not included in these estimates. Figure 4-6 shows the estimated DDT&E costs and unit production costs in millions of dollars for the various new vehicle propulsion concepts. All of these costs are derived from the current OTV studies being conducted by NASA. The DDT&E column has been adjusted for state-of-the-art technology levels, and technology development needs which will be discussed in Section 5.0. The unit costs also are based on the OTV studies. A lander is slightly less expensive than an OTV because it is assumed the OTV will be developed first and the lander will use many of the OTV systems and components; e.g., avionics, electronics, structures, and materials. The total vehicle production cost is based on the vehicle fleet size dictated by the mission model requirement. The flight rates are of fairly small magnitude, on the order of 10 to 15 per year, and thus only 2-4 OTV and landers are required. A more robust mission model could induce many more flights per vehicle and thus amortize the high design, development, and testing cost. The assumed vehicle lifetimes are as follows: (1) for all hydrogen/oxygen OTV and lander concepts with nominal mixture ratios of 5.5 and Isp's of around 470 seconds, the lifetime is estimated at 40 flights (NASA/MFSC, 1986); (2) for hydrogen/oxygen systems with higher oxidizer fuel (O/F) ratio, lifetimes are about 35 flights, largely due to cooling problems within the engine and the potential of having to go to an oxidizer-cooled engine; (3) the Silane/oxygen systems have an estimated lifetime of 30 flights because of technologies are in an early stage and the presence of abrasive silicon as a byproduct in the propellant fluids; (4) for the aluminumized-hydrogen/oxygen systems, a lifetime of 30 flights is assessed due to oxygen cooling and aluminum particulates in the propellant and effluent; and finally (5) for the aluminum/oxygen system, a 25 flight lifetime is estimated largely because of extremely high content of particulates, the absolute necessity of oxygen cooling, and difficulty with slurry injection and uniform combustion. These lifetimes were used with the estimated flight rates generated by ASTROFEST to yield estimated fleet size, and thus a total cost of the vehicle family fleet.

4.2 Earth Launch Mass Estimates

The total ELM estimates for any given vehicle family is comprised of the propellant requirements, the actual payloads that must be delivered to either low Lunar orbit or to the Lunar surface, and any support requirements including

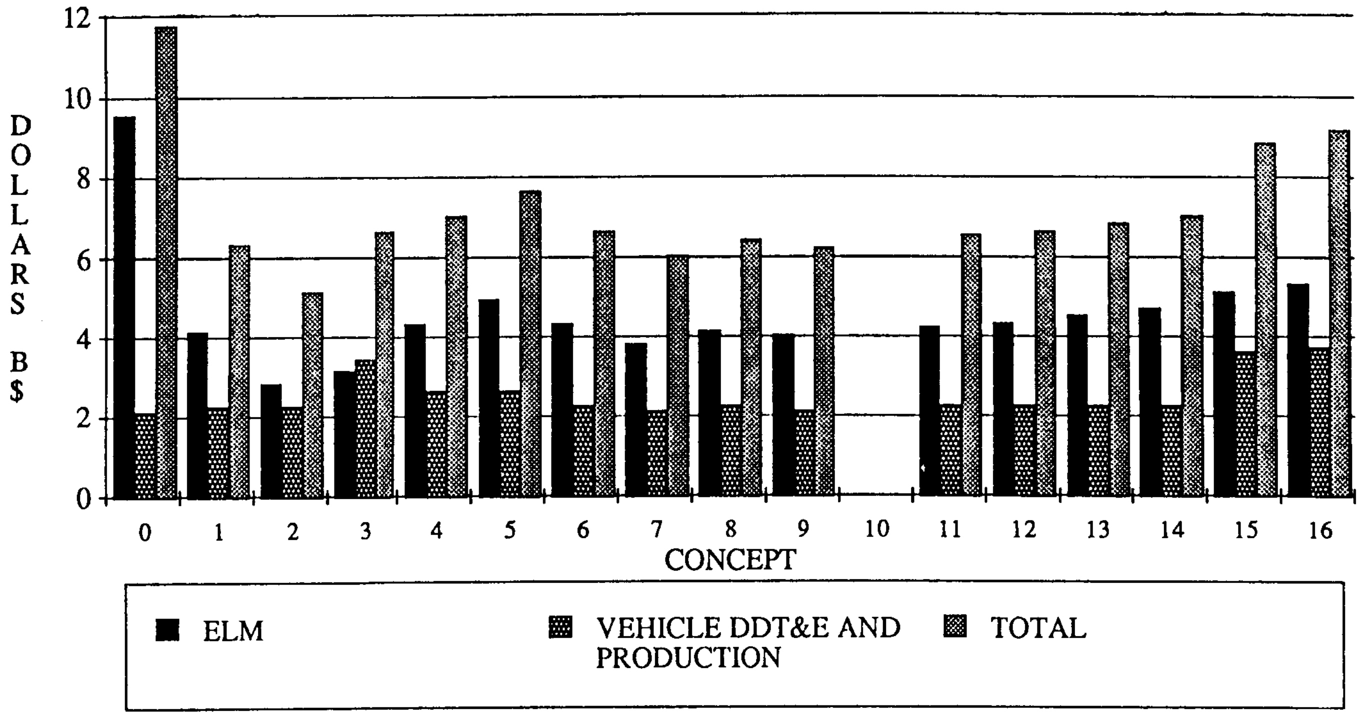

FIGURE 4-6. RELATIVE TRANSPORTATION COST LUNAR PROPELLANT IN 1995

equipment and consumables for propellant processing. Of these three, the propellant for delivery of the payloads and support requirements are by far the largest portion of the total ELM.

One key parameter related to ELM is a value known as the mass payback ratio. This ratio relates the mass required in LEO to the productive mass delivered to the Lunar surface. To obtain this mass payback ratio we have divided the total ELM by the total mission model mass. Thus, there is a different mass payback ratio for each vehicle family and these are listed in Table 4-1. These mass payback ratios range from 2.5 to 6.2 and can be used to estimate the total ELM of any given mission to the Moon on a rough order of magnitude basis. These mass payback ratios calculated here include only the propellant, and propellant processing support masses. They do not include the vehicle masses involved for the transportation system because these masses were not included in the ELM examined here. The mass of the transportation vehicles would change the mass payload ratios by about 0.5%.

4.3 Sensitivity and Tradeoff Analyses

The design characteristics of both the vehicle and propulsion systems as well as the propellant production scenarios will be compiled in this section and related to the total ELM of the Lunar based mission scenario. Key sensitivities and trades considered include:

- Effects of Lunar propellant production
- Effects of aerobrake specific mass
- Effects of consumable requirements for propellant processing
- Effects of hydrogen/oxygen mixture ratio
- Effects of hydrogen/oxygen specific impulse
- Effects of payload mass
- Influences of Shuttle scavenging.

The sensitivity analyses will address introduction of Lunar propellant processing at various periods of the mission model. Three basic scenarios exist: no Lunar propellant available at all, Lunar propellant available after the year 2005 through the year 2015, and Lunar propellant available over the entire mission scenario from 1995 to 2015.

4.3.1 Effects of Lunar Propellant Processing on Earth Launch Mass

When addressing effects of Lunar propellant processing on ELM one is actually measuring the value of the transportation system and associated Lunar

	VEHICLE FAMILY	MASS PAYBACK RATIO LUNAR PROP. - 1995
(0)	BASELINE OTV & LANDER (NO LUNAR PROPELLANT)	6.10
(1)	BASELINE OTV & LANDER (LUNAR OXYGEN AVAILABLE)	3.36
(2)	BASELINE OTV & LANDER (LUNAR OXYGEN AND HYDROGEN AVAILABLE	2.30
(3)	H/O OTV AND Al/O LANDER (LUNAR OXYGEN AND ALUMINUM)	2.56
(4)	H/O OTV & LANDER; MR = 8.7 (LUNAR OXYGEN AVAILABLE)	3.54
(5)	H/O OTV & LANDER; MR = 10.6 (LUNAR OXYGEN AVAILABLE)	4.01
(6)	H/O OTV & LANDER; Isp = 460 (LUNAR OXYGEN AVAILABLE)	3.50
(7)	H/O OTV & LANDER; Isp = 490; (LUNAR OXYGEN AVAILABLE)	3.13
(8)	H/O OTV & LANDER; PAYLOAD = 10 MT (LUNAR OXYGEN AVAILABLE)	3.38
(9)	H/O OTV & LANDER; PAYLOAD = 20 MT (LUNAR OXYGEN AVAILABLE)	3.29
(10)	H/O OTV & LANDER; NO AEROBRAKE (LUNAR OXYGEN AVAILABLE)	3.94
(11)	H/O OTV & LANDER; AEROBRAKE MASS = 18% OF REENTRY (LUNAR OXYGEN AVAILABLE)	3.44
(12)	H/O OTV & LANDER, AEROBRAKE MASS = 20% OF REENTRY (LUNAR OXYGEN AVAILABLE)	3.50
(13)	H/O OTV & LANDER, AEROBRAKE MASS = 25% OF REENTRY (LUNAR OXYGEN AVAILABLE)	3.66
(14)	H/O OTV & LANDER, AEROBRAKE MASS = 30% OF REENTRY (LUNAR OXYGEN AVAILABLE)	3.84
(15)	SiH4 / OXYGEN OTV & LANDER	4.20
(16)	ALUMINIZED - HYDROGEN OTV & LANDER (LUNAR OXYGEN AND ALUMINUM AVAILABLE)	4.34 (3.11)*
(17)	H/O OTV & LANDER WITH LUNAR OXYGEN RETURN TO LEO	3.49

* ALUMINUM, HYDROGEN AND OXYGEN AVAILABLE ON THE MOON

TABLE 4-1. MASS PAYBACK RATIOS FOR VARIOUS VEHICLE FAMILIES propellant processing system using ELM as the measurand. Generally, Lunar propellant availability reduces the total mass, the mass fraction, and the ELM of the OTV. In addition, the earlier establishment of the Lunar propellant supply within the mission scenario further lowers the ELM. Specifically, the effects of Lunar oxygen on Baseline hydrogen/oxygen OTV and lander, the effects of Lunar oxygen and Lunar hydrogen availability on the baseline systems, and the effects of Lunar oxygen and aluminum on the hydrogen/oxygen OTV and aluminum/oxygen lander system are addressed in this section.

The effects of the availability of Lunar oxygen are shown in Figure 4-7 for the Baseline system and the three major supply scenarios of: no Lunar propellant; Lunar propellant available after 2005 through the year 2015; and Lunar propellant available through the entire mission scenario 1995 through 2015. Lunar oxygen availability throughout the entire mission model will reduce the total ELM by 52.5% as compared to the Baseline where no Lunar propellant is available. Providing the Lunar oxygen at a later date, in this case 2005, still induces a substantial benefit of 45.5% over the Baseline because the major mission model activity resides in the years 2005 through 2015. These data show that an evolutionary system providing no Lunar propellant through the year 2005 with evolution to Lunar oxygen processing facility in 2015 does not substantially increase ELM as compared to providing Lunar oxygen at day one (1995) of the Lunar base. The potential effect of Lunar oxygen is sufficiently large if Lunar processing requirements are kept low to identify it as a priority item in the Lunar transportation system.

Similar results are found in Figure 4-8 when considering the availability of both Lunar oxygen and Lunar hydrogen. Supplying Lunar oxygen and hydrogen to the baseline system for the entire mission scenario, the ELM may be reduced by 67.5% over the Baseline. Introducing these two propellants from the Lunar surface at a later date, 2005, still reduces ELM by over 50%. Thus the overall affects of introducing Lunar propellant is seen as extremely valuable in reducing ELM by more than one half of that for a totally Earth LEO-based transportation system. Further evidence of this will be shown in future sensitivity results.

The combination of an aluminum/oxygen lander with a hydrogen/oxygen OTV is a potentially economical family. Providing both aluminum and oxygen from the Moon produces a greater benefit because both aluminum and oxygen can be produced

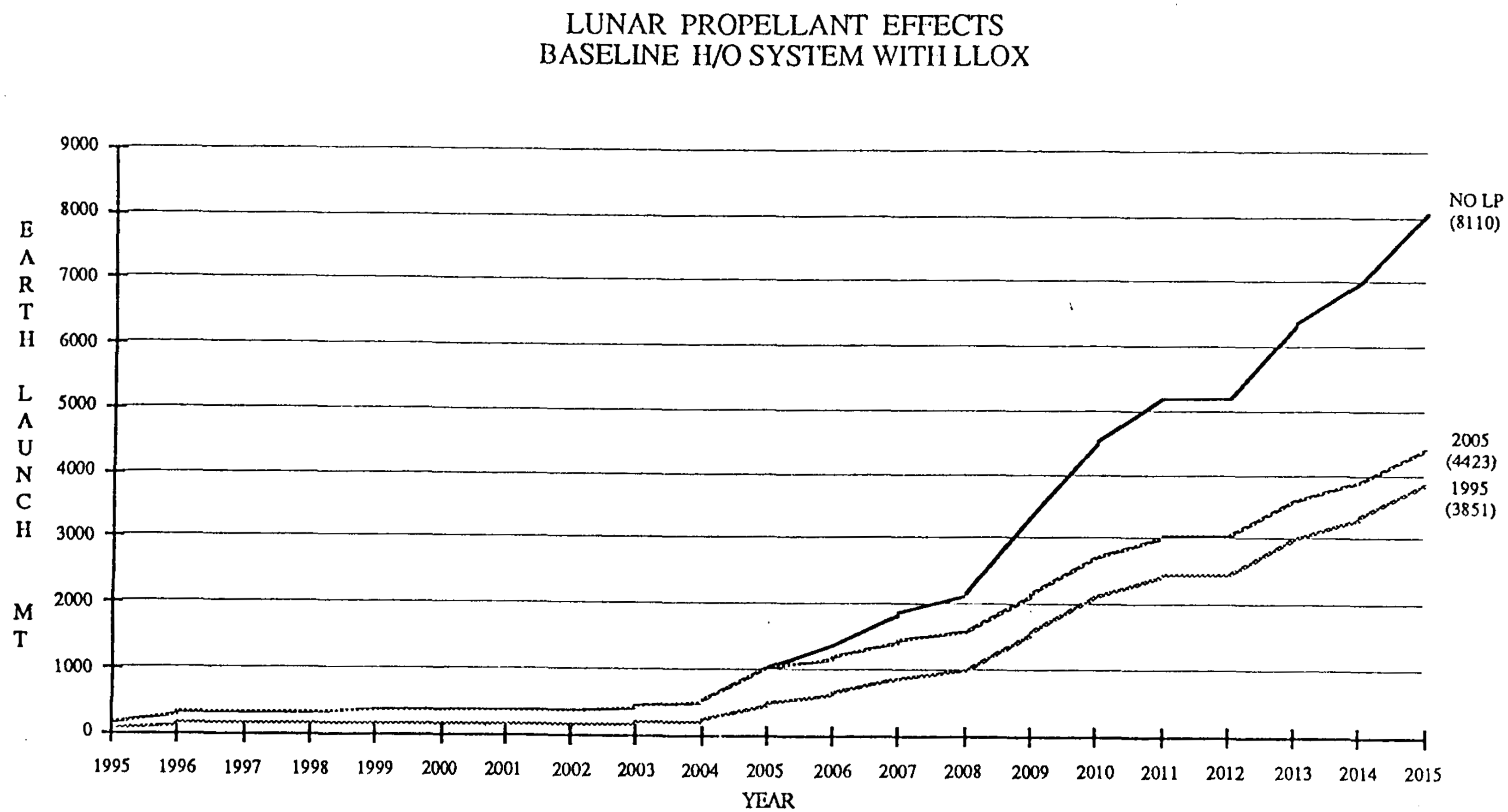

FIGURE 4-7. LUNAR PROPELLANT EFFECTS
BASELINE H / O SYSTEM WITH LLOX

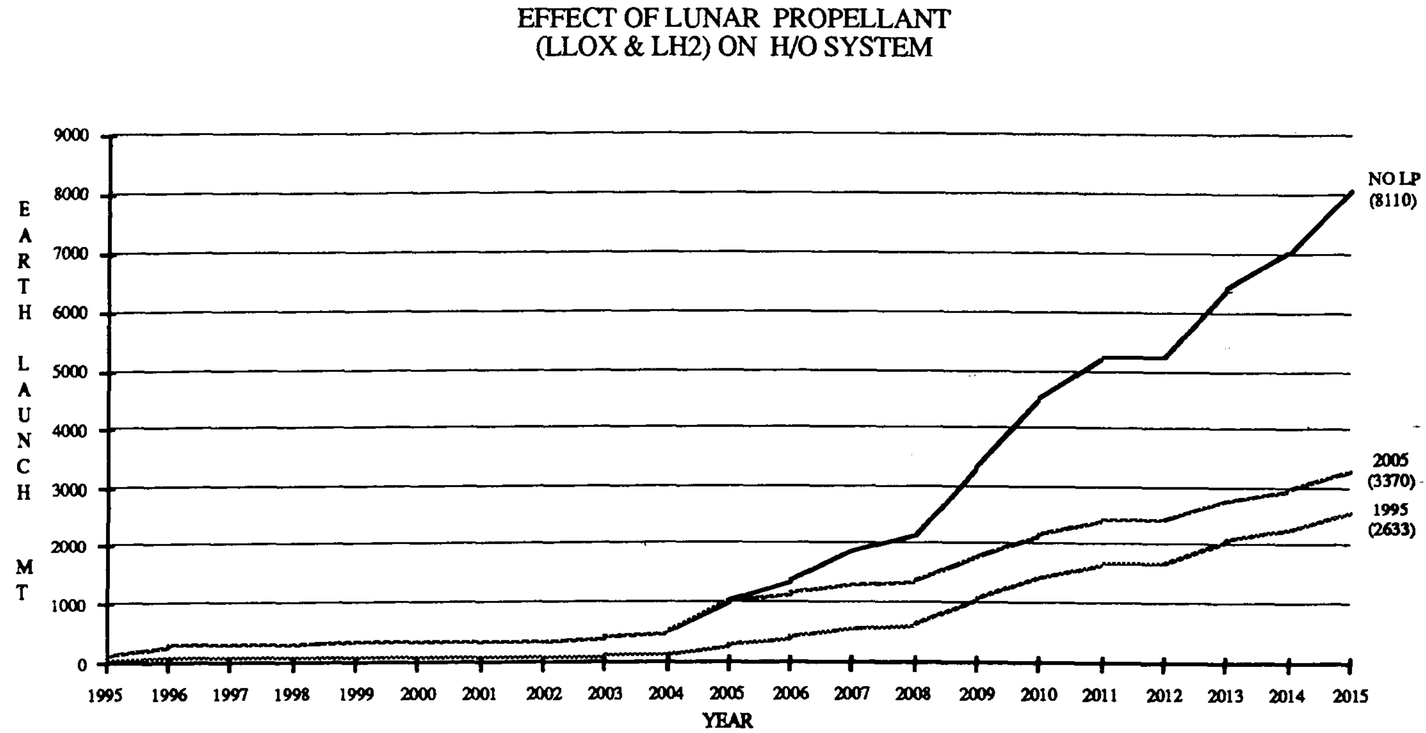

FIGURE 4-8. EFFECT OF LUNAR PROPELLANT (LLOX & LH2) ON H/O SYSTEM

in the same process. Because of the relatively low performance of the aluminum/oxygen lander, a great deal of propellant is required from Earth if none is available from the Moon. Therefore, the aluminum/oxygen lander is not logical to consider if no Lunar propellant is available. By introducing the Lunar propellant after 2005, the Earth launch mass can be reduced by more than 72%. By introducing both aluminum and oxygen as Lunar propellants in 1995, the ELM may be reduced by 88.7%, nearly an order of magnitude. The ELM for this family with Lunar propellant is comparable to the H/O systems with Lunar oxygen.

These results also apply to the Silane/oxygen and aluminized-hydrogen/oxygen families. The aluminized-hydrogen/oxygen (Al-H/O) family has two major propellant supply alternatives: (1) Lunar supply of aluminum and oxygen only, and (2) Lunar supply of hydrogen, aluminum, and oxygen. The ELM is 4974 metric tons (MT) for the first scenario and 3564 MT for the second scenario. This compares to 8083 MT for the Baseline case. Thus, supplying Lunar hydrogen in addition to aluminum and oxygen decreases the ELM requirements by 28%. The totally Lunar propellant-supplied Al-H/O vehicle family is attractive but less beneficial than the H/O system with Lunar hydrogen and oxygen available.

4.3.2 Effects of Aerobrake Specific Mass on Earth Launch Mass

Aerobrakes are a very effective means of reducing the propulsive velocity requirements (Delta-v), thus reducing propellant quantities required. However, the aerobrake represents an increase in dry mass which must be transported from Low Earth orbit (LEO) to low Lunar orbit (LLO) and back to LEO. Two major sources types of aerobrakes possible are Earth-produced aerobrakes and Lunar-produced aerobrakes. Results demonstrated here consider the nearer technology of Earth-produced aerobrakes and the sensitivities of those aerobrakes to the ELM. The mass of an aerobrake is typically stated as a percentage of the total reentry mass of the vehicle family. Figure 4-9 shows the effect of aerobrake masses on total ELM without any Lunar propellant supply available. The benefit of having a 15% specific mass aerobrake over no aerobrake is a savings of about 22% of the ELM. However, as the specific aerobrake mass increases from 15 to 30% the ELM increases from 8260 MT to approximately 9000 MT reducing the ELM benefit to 13.5%. Extrapolating these results as shown in Figure 4-10, the aerobrake specific mass must be less than 35% of the reentry mass.

When considering the same aerobrake masses for a Lunar oxygen supply scenario beginning at 1995 the 15% specific mass aerobrake saves only about 15% of

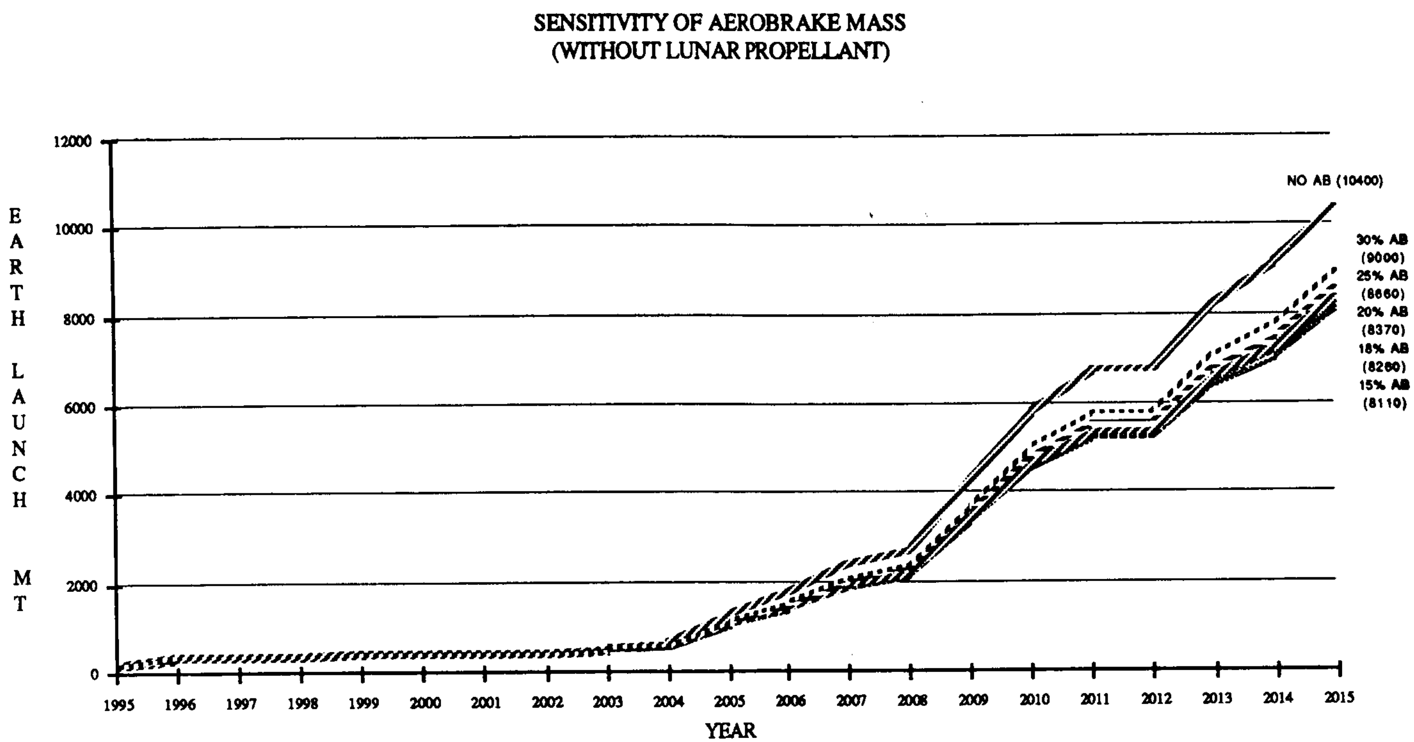

FIGURE 4-9. SENSITIVITY OF AEROBRAKE MASS (WITHOUT LUNAR PROPELLANT)

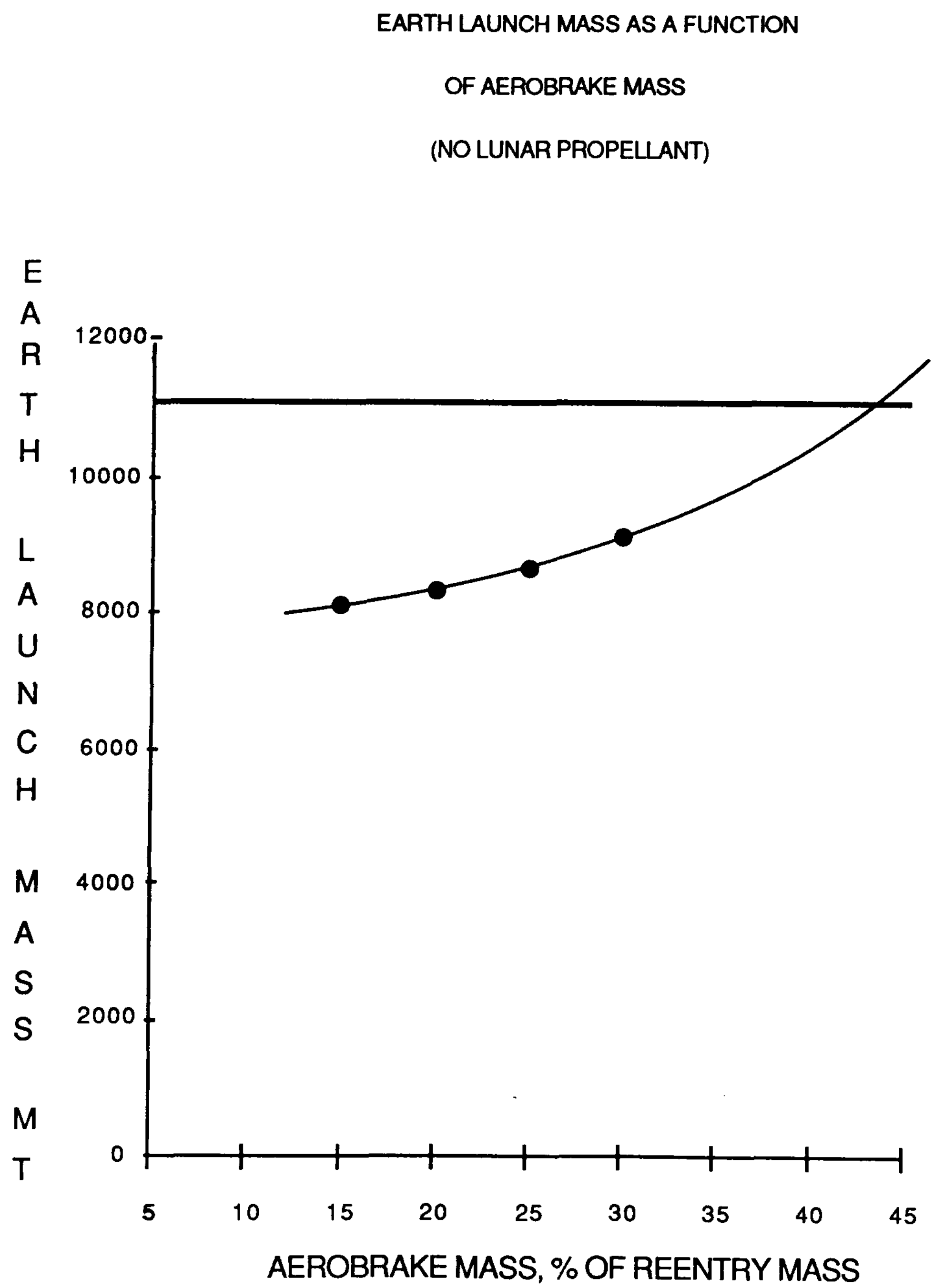

FIGURE 4-10. EARTH LAUNCH MASS AS A FUNCTION OF EARTH-DERIVED AEROBRAKE MASS (NO LUNAR PROPELLANT)

the ELM as shown in Figure 4-11. . The ELM then from 3850 MT to 4403 MT as the mass of the aerobrake increases from 15% to 30% of the reentry mass. The ELM is about 4520 MT without an aerobrake. Extrapolating the ELM as a function of aerobrake specific mass as shown in Figure 4-12, illustrates that the maximum mass of the aerobrake is about 32 MT to have a beneficial impact on the ELM.

In summary, the availability of Lunar oxygen is a much more significant, beneficial technology than the aerobrake. Figure 4-13 shows the relative benefits of any combination of single technology implementations with respect to aerobrakes and Lunar oxygen supply. The aerobrake masses effect ELM more significantly when a Lunar propellant supply of Lunar oxygen is not available. Earth launch mass by 260 MT when there is no Lunar oxygen supply.

4.3.3 Effects of Propellant Processing Consumable Requirements on Earth Launch Mass

The effects of propellant processing consumable requirements on ELM is a key consideration for two reasons. First, it may help determine when a specific propellant processing technique is no longer beneficial in terms of the amount of required consumable mass per unit mass of propellant. Second, it will show the sensitivity of our analysis results to consumable estimates and thus reveal where extreme care should be taken to ensure accurate assumptions and estimates. Consumable requirements will be addressed for three different systems: the baseline hydrogen/oxygen system with Lunar oxygen production on the Moon, the hydrogen/oxygen system with both Lunar oxygen and Lunar hydrogen on the Moon, and finally the hydrogen/oxygen OTV and aluminum oxygen lander with both aluminum and oxygen available on the Moon. The consumable rates, kilograms per unit mass of production, were increased by 4%, 8%, 12%, and, in some cases, 16%, and 20% to determine the additional resource requirements that must be delivered to the Moon from Earth to manufacture the same amount of propellant for the same mission scenario. Figure 4-14 shows the effects of changes in the consumable requirements for Lunar oxygen the baseline hydrogen/oxygen system. Notice that an increase of 20% in the Lunar consumable resource per unit of oxygen increases the total ELM by almost 100%. Figure 4-15 shows similar curves for the oxygen and hydrogen supply from the Moon for the same detailed system. The 20% increase in consumable requirements per pound of product when both oxygen and hydrogen are produced induces less than a 79% increase in Earth Launch Mass requirements. Finally, in Figure 4-16 the consumable requirements for production of aluminum and oxygen for the aluminum/oxygen lander in the H/O, OTV show

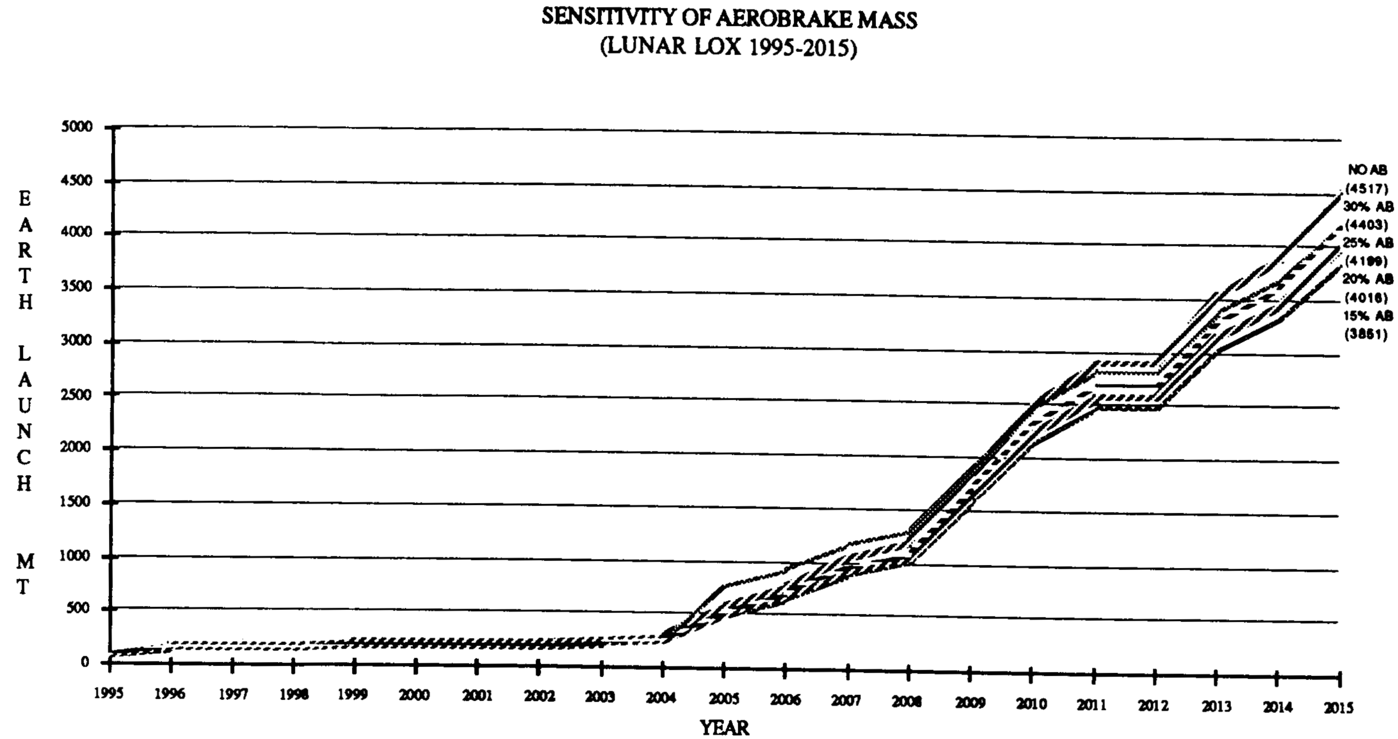

FIGURE 4-11. SENSITIVITY OF AEROBRAKE MASS - LLOX 1995 TO 2015

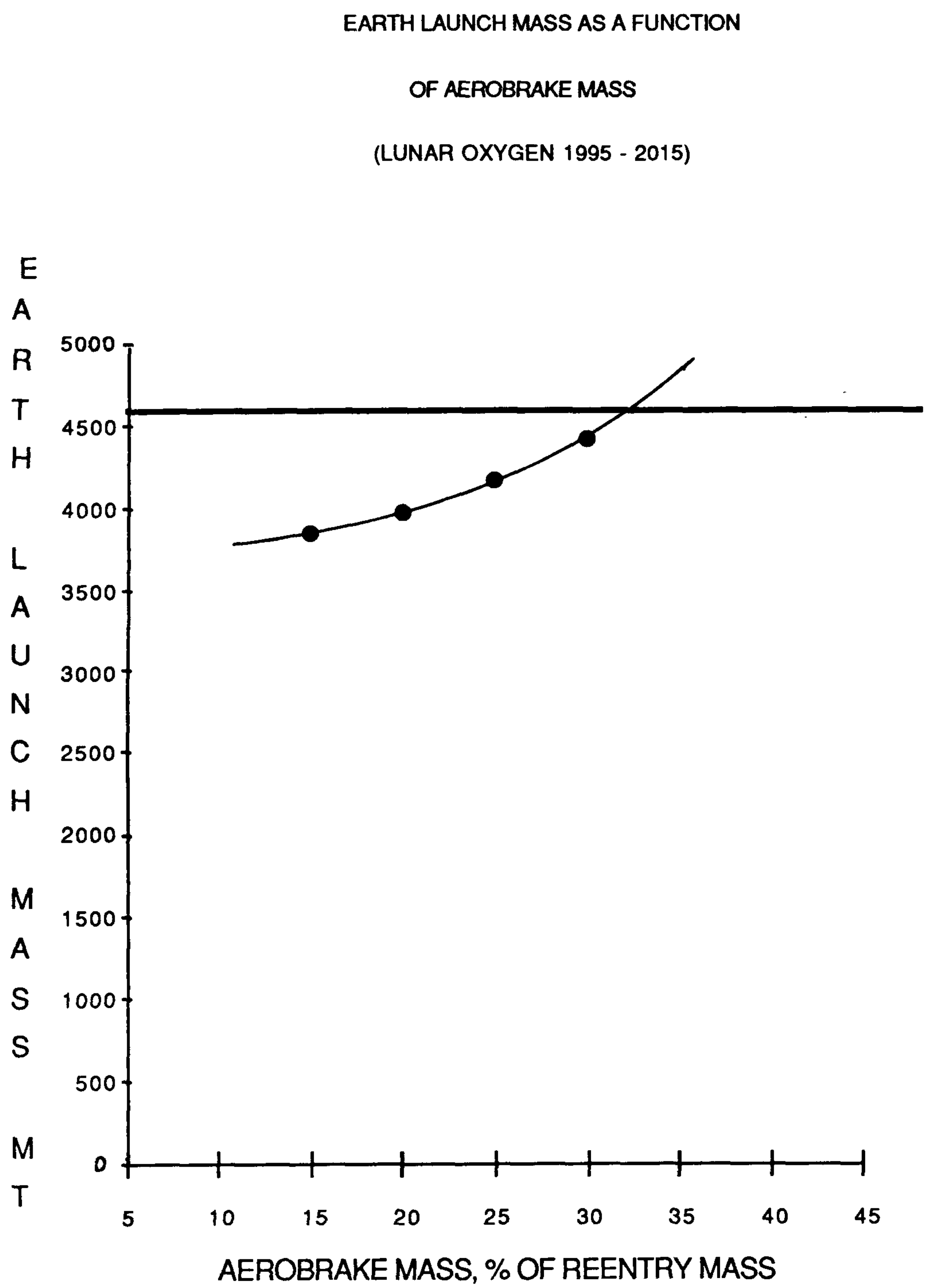

FIGURE 4-12. EARTH LAUNCH MASS AS A FUNCTION OF AEROBRAKE MASS (LUNAR OXYGEN 1995 TO 2015)

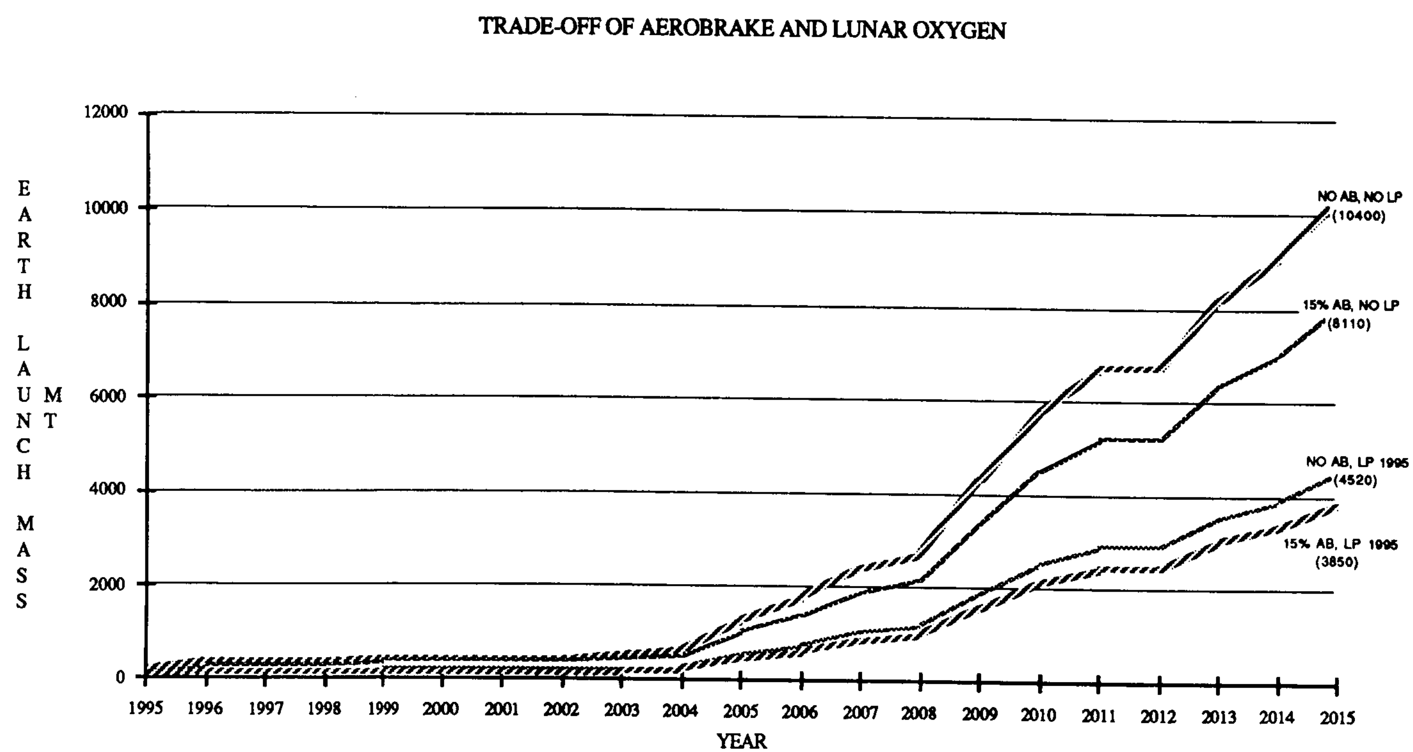

FIGURE 4-13. TRADE-OFF OF AEROBRAKE AND LUNAR OXYGEN

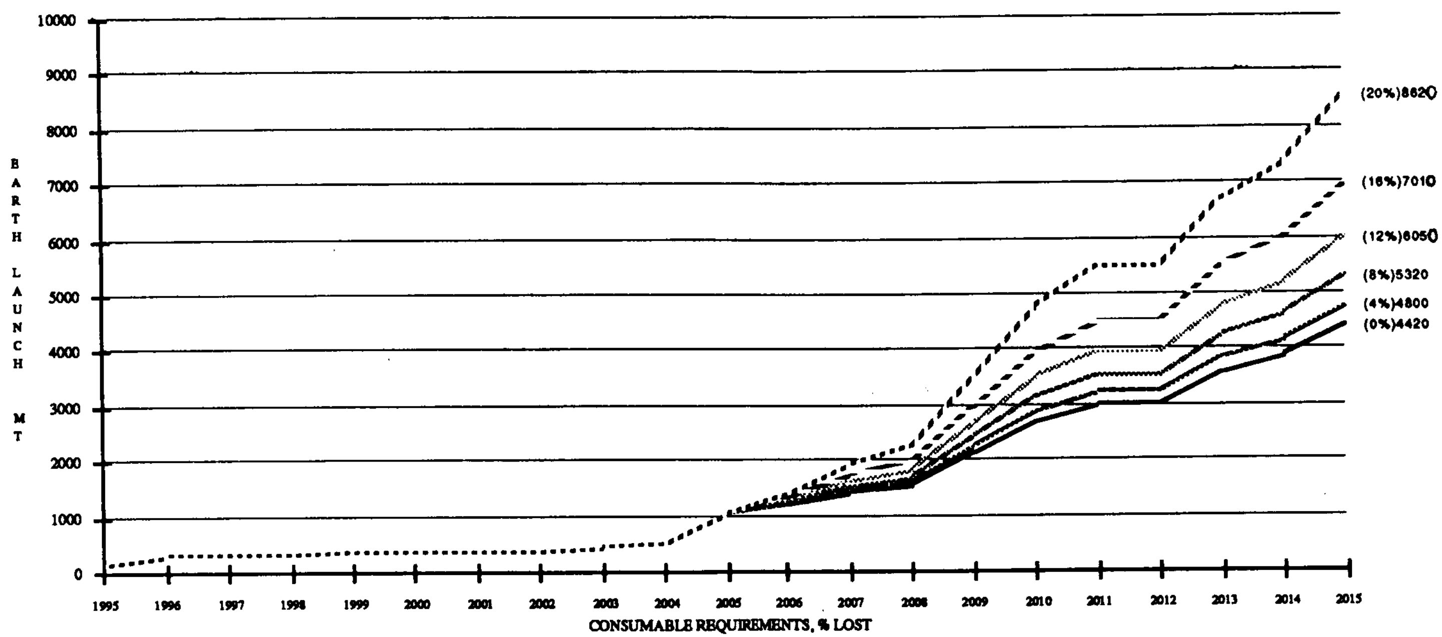

FIGURE 4-14. CONSUMABLE REQUIREMENTS EFFECT ON H / O SYSTEM WITH LLOX AFTER 2005

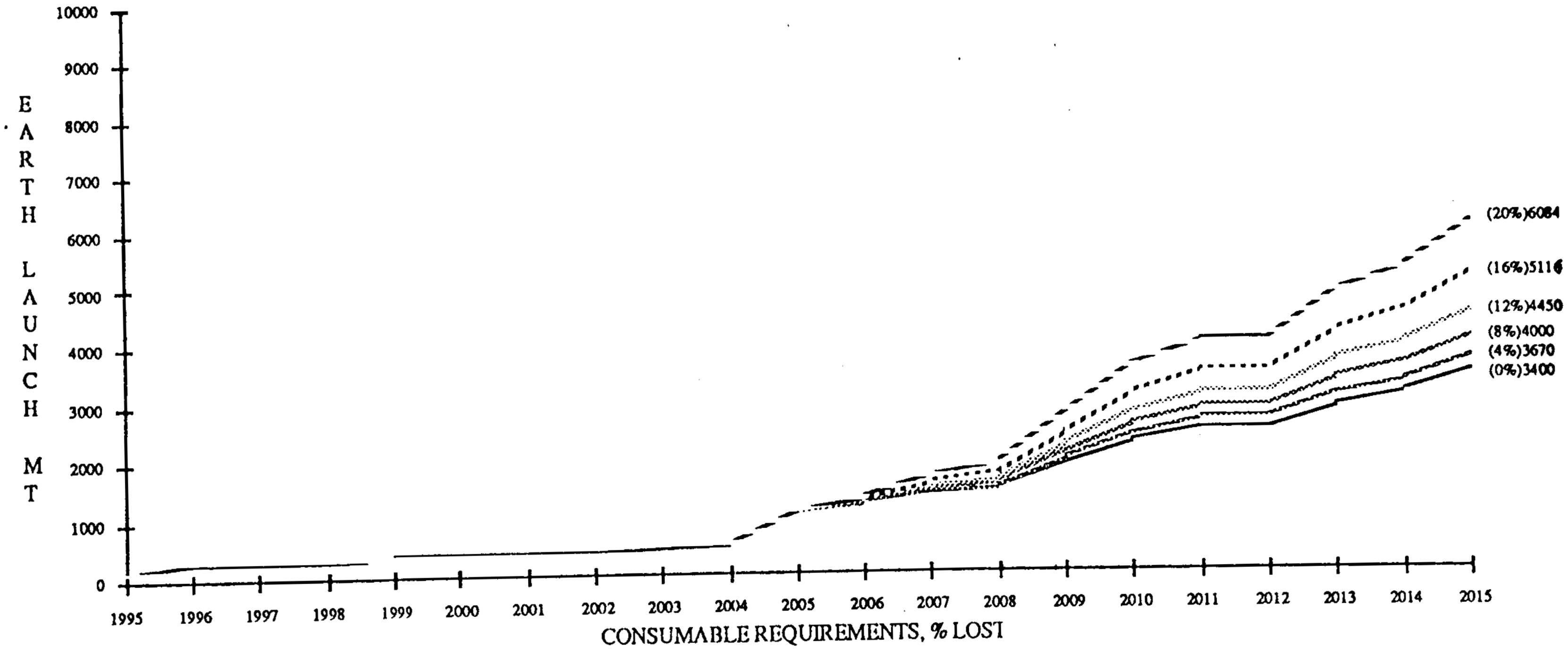

FIGURE 4-15. CONSUMABLE REQUIREMENTS EFFECT ON H / O SYSTEM WITH LLOX / LH2 AFTER 2005

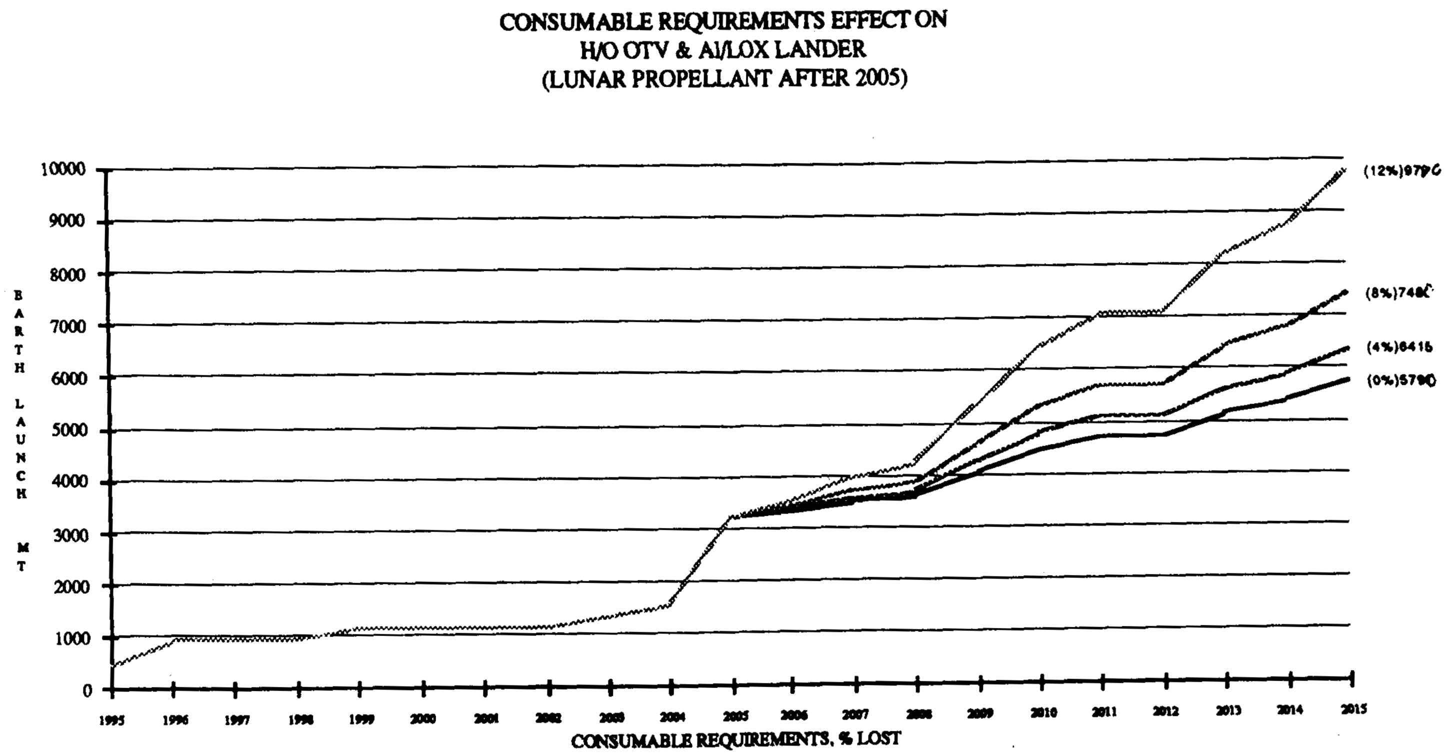

FIGURE 4-16. CONSUMABLE REQUIREMENTS EFFECT ON H/O OTV & Al/LOX LANDER - LUNAR PROPELLANT AFTER 2005

again a drastic increase in the ELM. For only a 12% increase in the consumable requirement per pound of oxygen a 69% increase in ELM is estimated.

To summarize, the ELM increases drastically with the percentage of consumable requirements increase, and the ELM is very sensitive to the consumable requirements per pound of products for any propellant production technique. The sensitivity naturally increases as more propellant is produced on the Lunar surface. Figure 4-17 shows the slow increases for the consumable requirements for Families 1, 2, and 3. The slope indicates the sensitivity of the ELM to changes in consumable requirements. The slopes are indeed greater for the aluminum/oxygen propellant production scheme. This shows that increased dependence on Lunar-derived propellant effects the maximum amount of consumable mass for any given processing technology.

4.3.4 Effects of Hydrogen/Oxygen Mixture Ratio on Earth Launch Mass

Increasing mixture ratio will reduce the percentage of fuel required as propellant for any given propulsion system. In the case of hydrogen/oxygen propulsion systems, because hydrogen is a relatively scarce product on the Moon, a high mixture ratio may be beneficial in meeting overall transportation requirements. However, as the mixture ratio diverges from the optimum the specific impulse decreases and increases the total propellant requirement for any given mission. Figure 4-18a shows the mass of hydrogen for the H/O OTV and lander systems for a single mission as a function of mixture ratio. When higher mixture ratios are used in the lander, specific impulse decreases and propellant requirements increase. This increase in overall propellant increased the requirement for hydrogen and hence the change in slope at a mixture ratio of about eight. This result can also be seen in Figure 4-19, the ELM increases as the mixture ratio increases from 5.5 to 10.6.

The question still arises: Is there an oxidizer to fuel ratio which takes advantage of the Lunar oxygen supply without being burdened by additional propellant requirements which would induce additional hydrogen delivery from Earth. When the vehicle families with hydrogen/oxygen propulsion system O/F ratios of 50 and 35 were run on ASTROFEST, very large numbers of lander flights were required to supply an OTV with enough oxygen for the return trip because of the reduced performance. This large number of lander flights required additional hydrogen to be imported from Earth using the OTV and thus the total ELM is much greater than for mixture ratios of 5.5 to 8. The only other question remaining

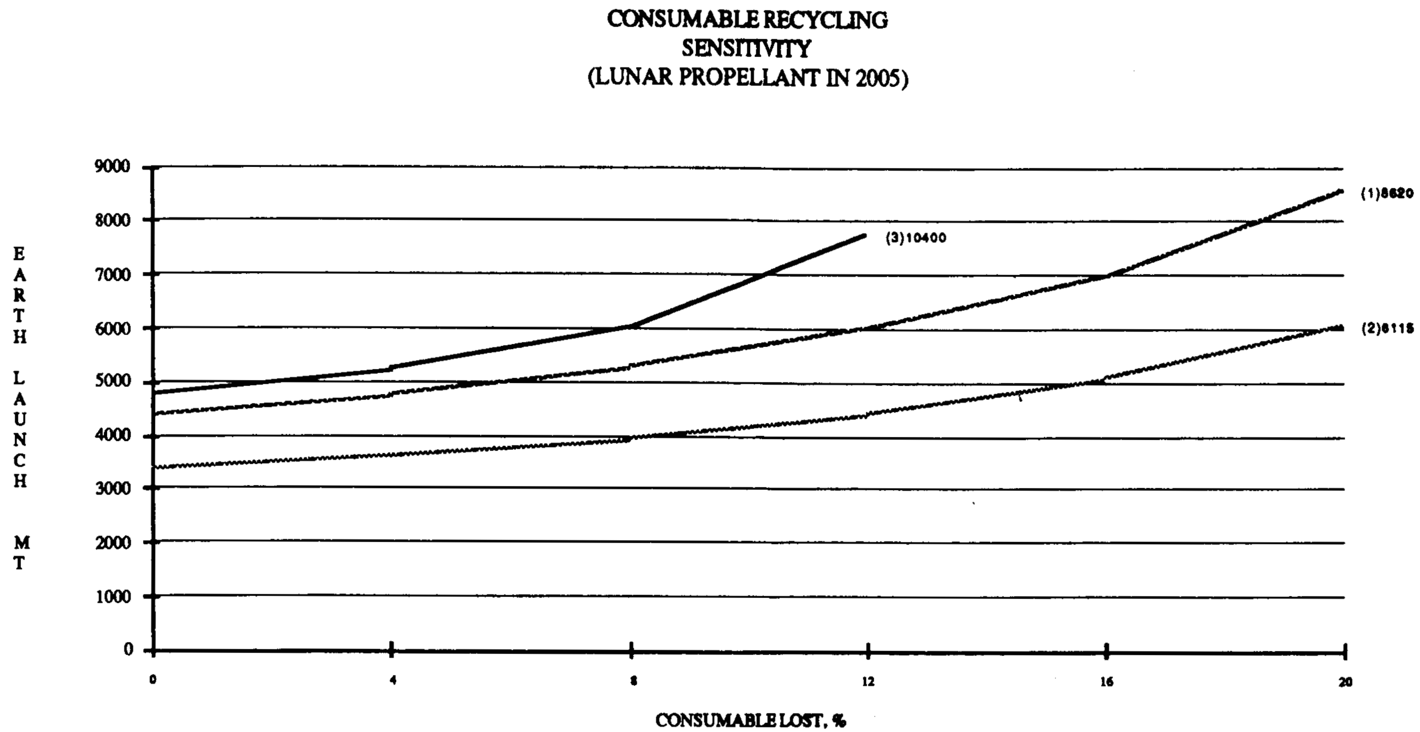

FIGURE 4-17. CONSUMABLE RECYCLING SENSITIVITY LUNAR PROPELLANT IN 2005

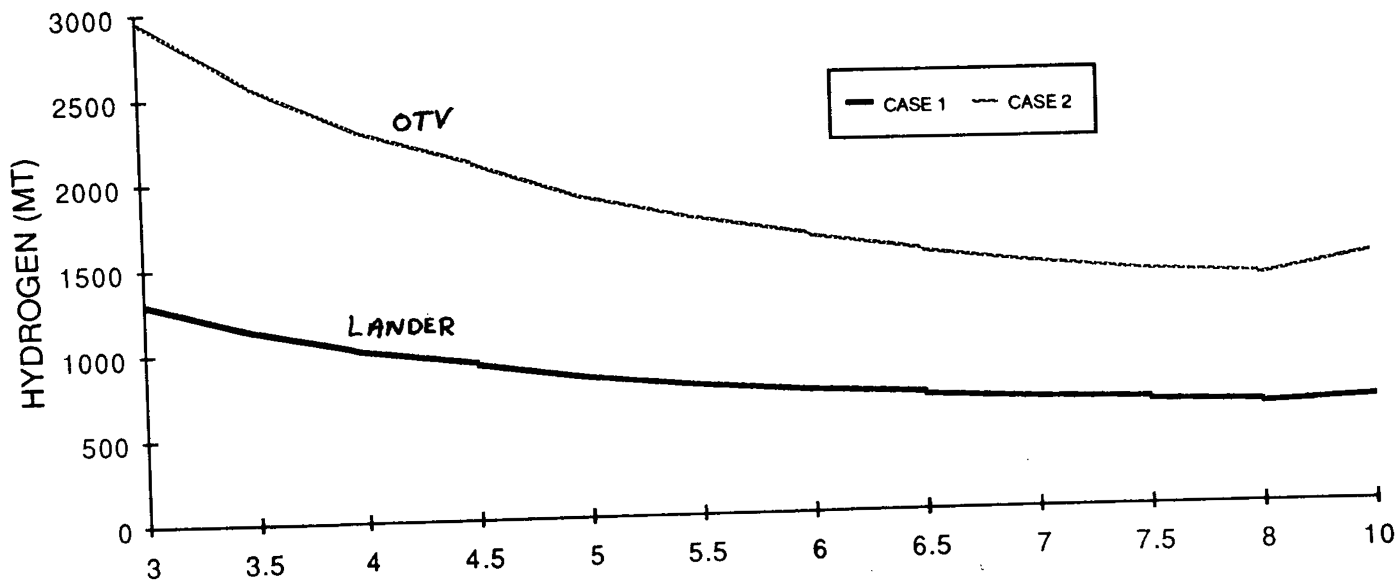

FIGURE 4-18. HYDROGEN REQUIREMENT AS A FUNCTION OF O/H MR

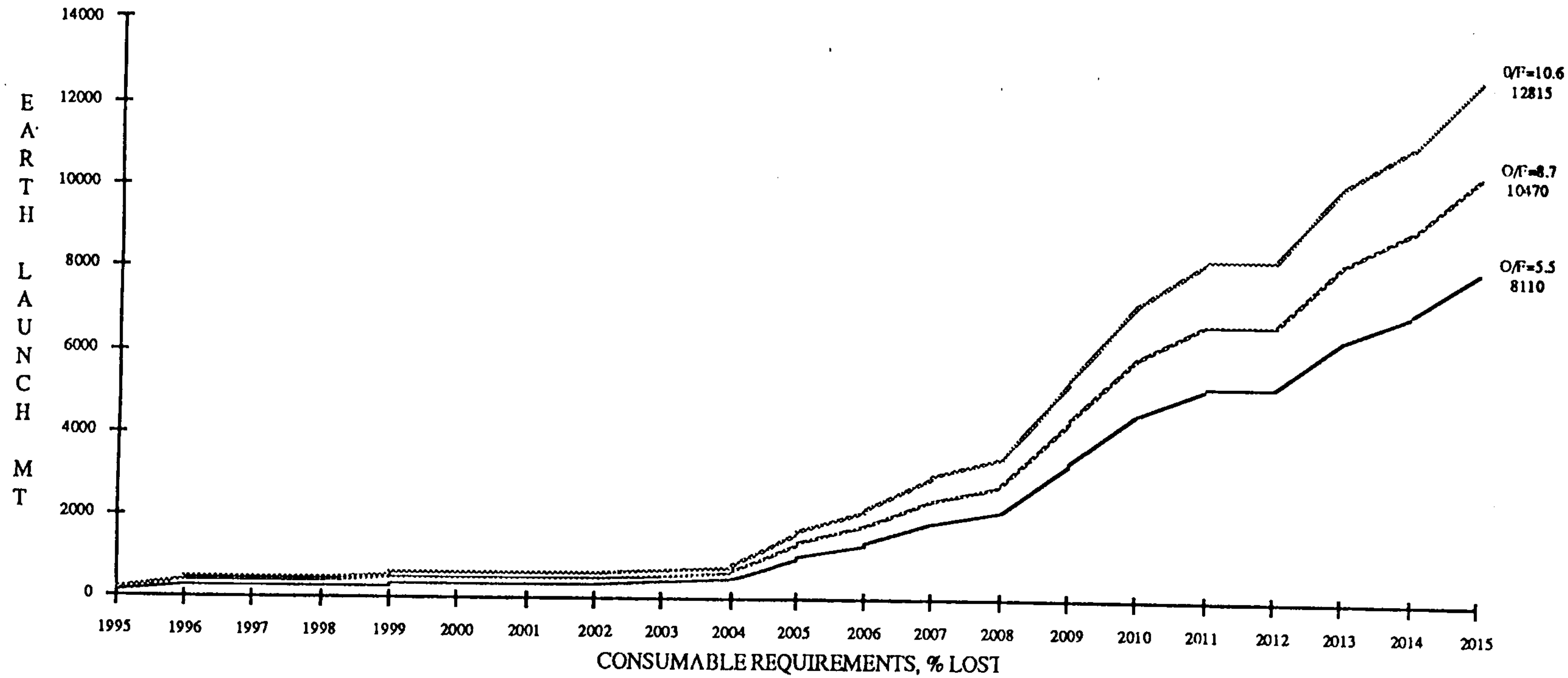

FIGURE 4-19. SENSITIVITY OF H/O O/F RATIO - NO LUNAR PROPELLANT

may be presented by Figure 4-20. Is there an oxidizer to fuel ratio between 5.5 and 8.7 for which the ELM is lower than that mixture ratio of 5.5? This question remains to be answered.

4.3.5 Effects of Specific Impulse on Earth Launch Mass

Three hydrogen and oxygen propellant systems of varying specific impulse were run through the entire mission scenario. The three Isp's were: 460 seconds, 470 seconds, and 490 seconds. The propellant supply scenarios of no Lunar propellant and Lunar propellant throughout the entire mission model were addressed and the results are presented in Figures 4-21 and 4-22. Without Lunar propellants available, a 30 second increase in Isp from 460 second to 490 seconds decreased the ELM by only 11%. When Lunar propellant was available the same increase in specific impulse yielded about the same decrease in ELM, approximately 10.6%. In summary, small increases in specific impulse on the order of 30 seconds reduce the ELM by approximately 10% with or without Lunar propellant production.

4.3.6 Effects of Payload Mass Capability on Earth Launch Mass

Sensitivities of the payload mass capability of any vehicle were investigated with samples taken at 10 MT, 15.9 MT and 20 MT of payload (15.9 MT being the baseline). As shown in Figure 4-23 little or no sensitivity was observed. This is largely due to the manifesting of the various payloads in the mission model which allowed little vacant space in any vehicle flight.

4.4 Space Shuttle Scavenging Impacts on Earth Launch Mass Requirements

Space Shuttle scavenging represents a potentially significant source of hydrogen and oxygen in LEO. This source will have the effect of reducing ELM requirements, specifically the propellant delivered from Earth to LEO. Current Shuttle scavenging studies indicate that up to 91 MT of hydrogen and oxygen propellant could be supplied per year. Over a 20 year period this could amount to up to 1800 MT (22% for baseline vehicle family). Thus, ELM of 1800 MT could be saved over the mission model scenario. This amount of propellant and corresponding reduction in Earth Launch Mass is equivalent to about 9.6 billion dollars using the guidelines of costing as stated in Section 4.1. With Space Shuttle scavenging and the use of Lunar oxygen propellant the transportation costs of the mission model could be less than $6 billion over the 20 year period given.

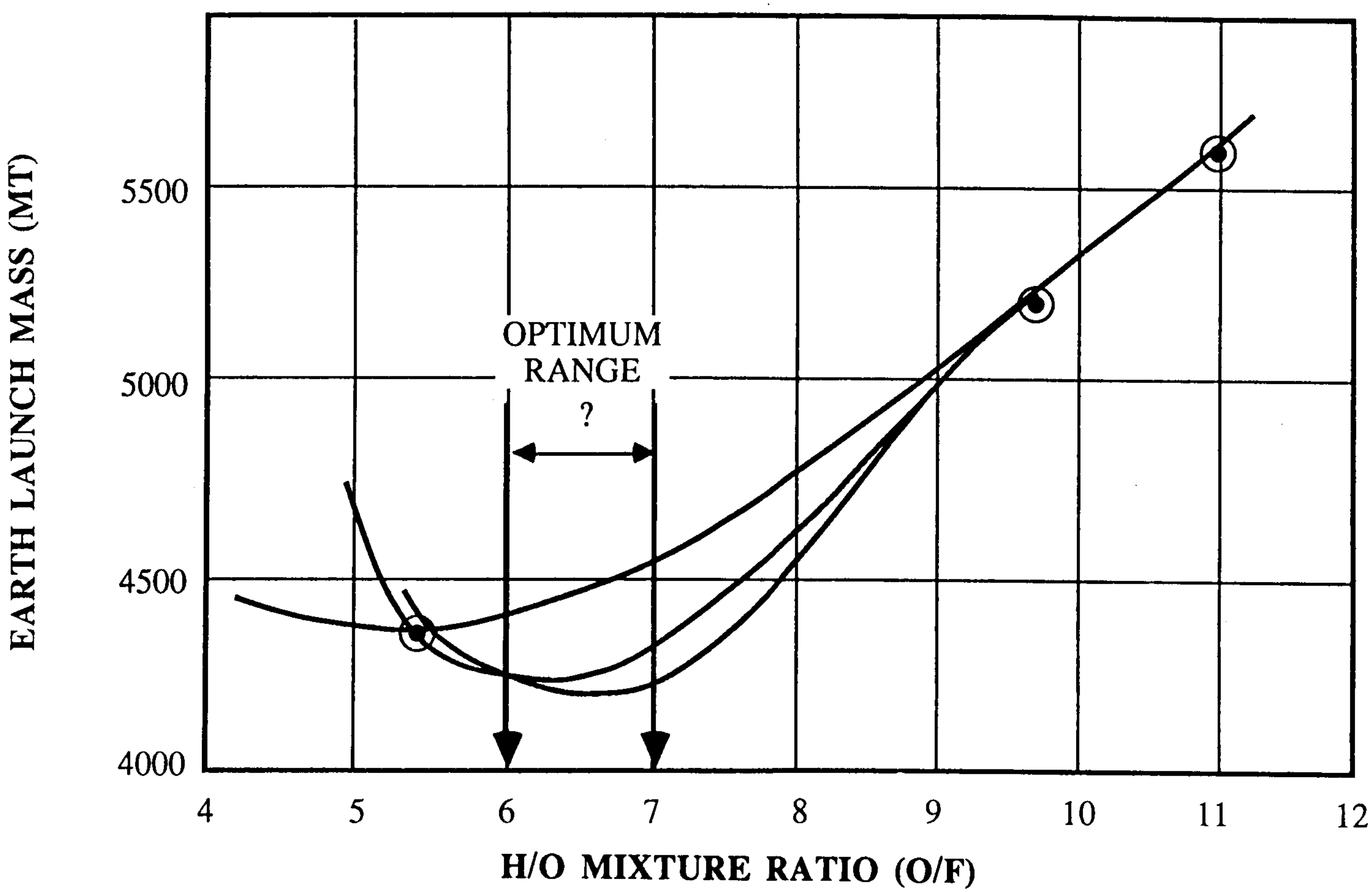

FIGURE 4-20. OPTIMUM H/O MIXTURE RATIO FOR LSB - LLOX

SENSITIVITY OF H/O SPECIFIC IMPULSE
(WITHOUT LUNAR PROPELLANT)

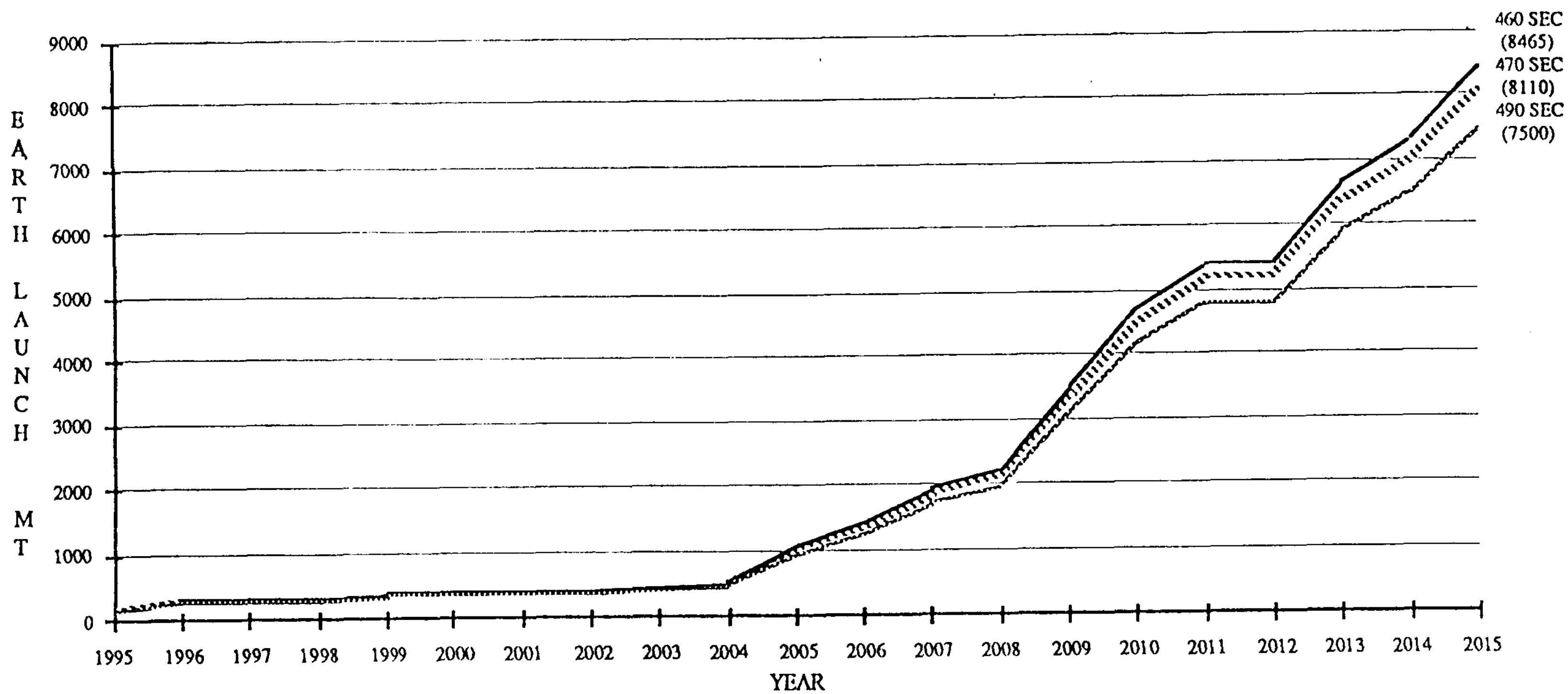

FIGURE 4-21. SENSITIVITY OF H/O SPECIFIC IMPULSE WITHOUT LUNAR PROPELLANT

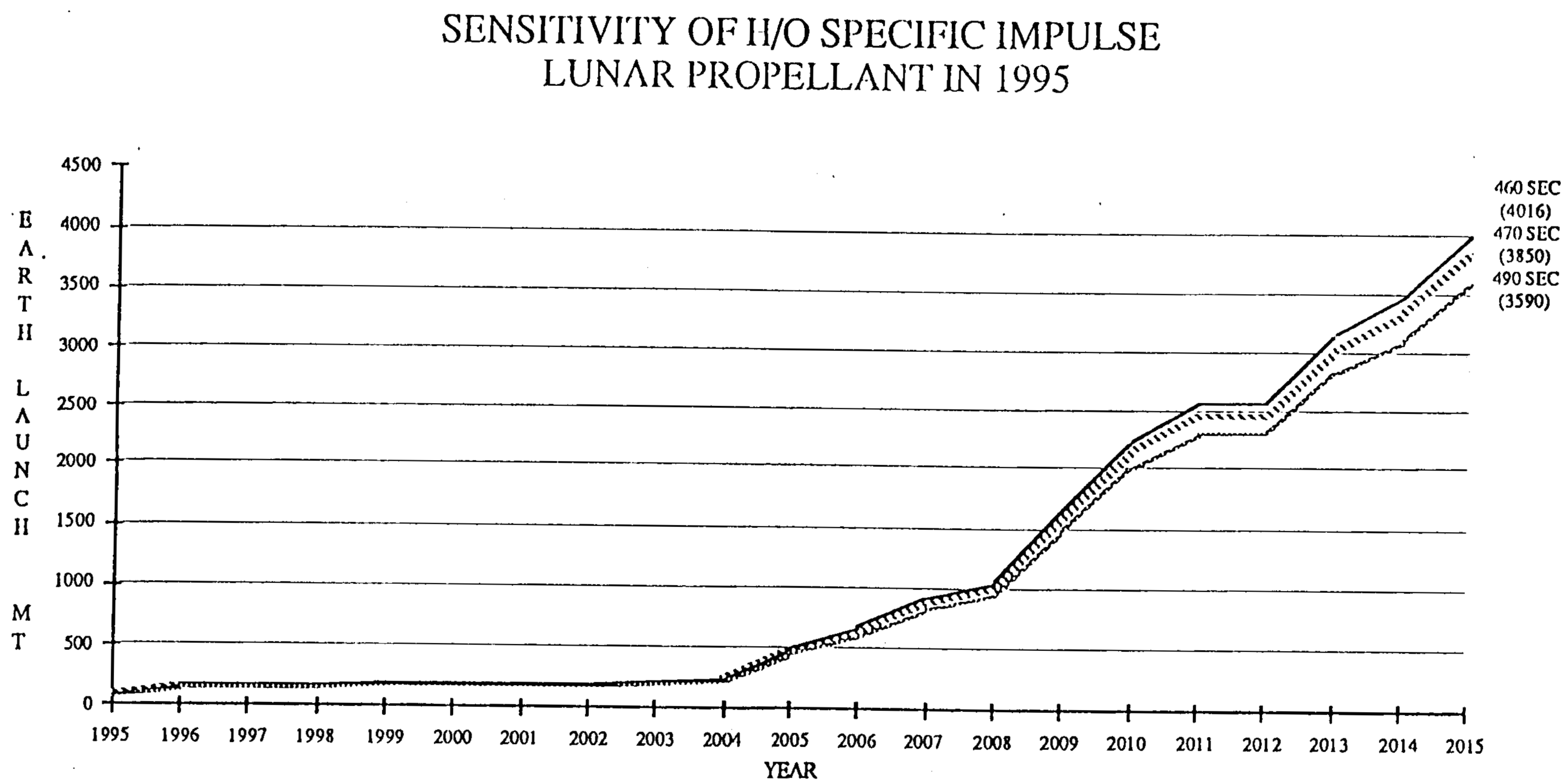

FIGURE 4-22. SENSITIVITY OF H/O SPECIFIC IMPULSE LUNAR PROPELLANT IN 1995

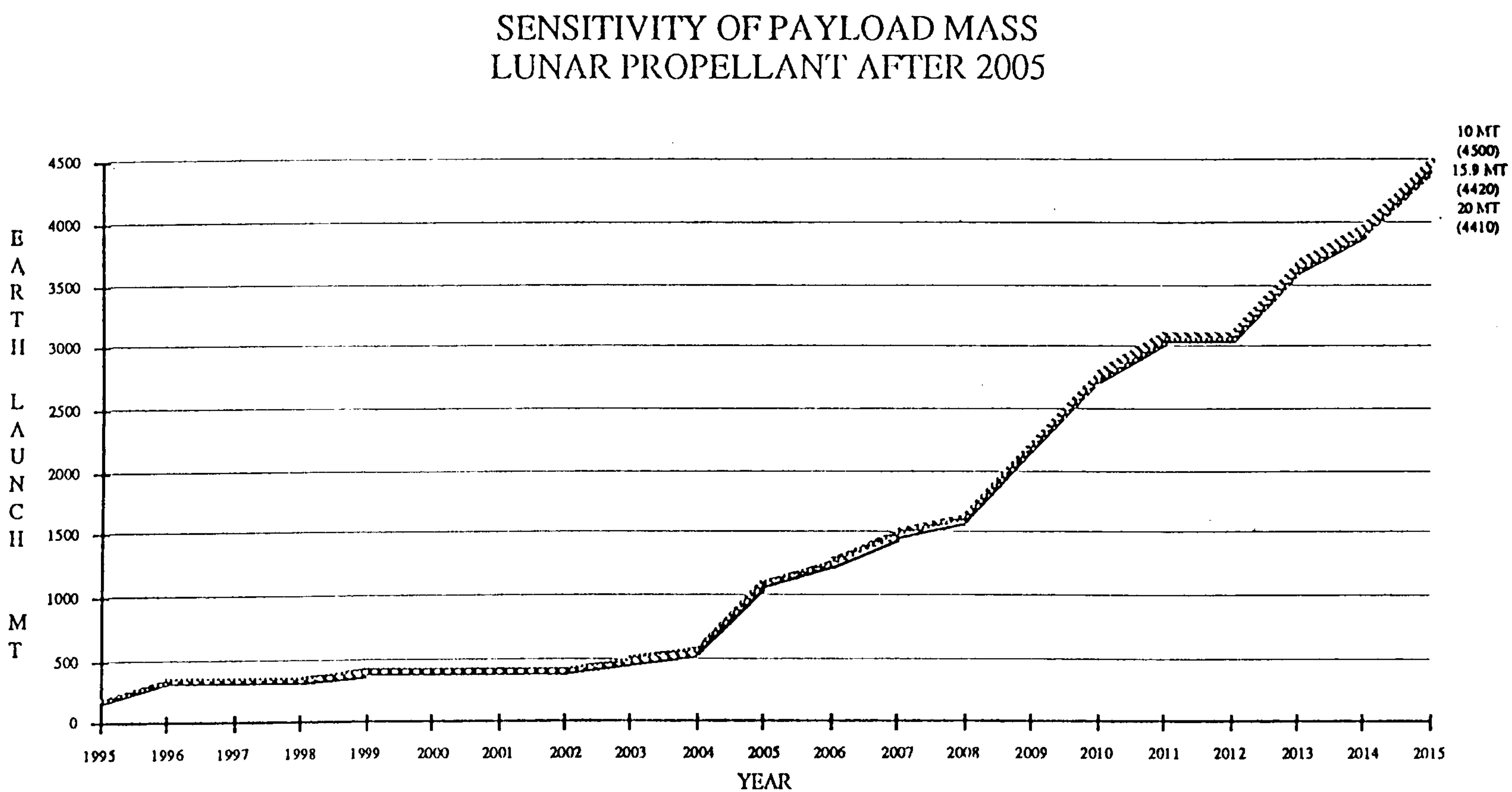

FIGURE 4-23. SENSITIVITY OF H/O SPECIFIC IMPULSE LUNAR PROPELLANT AFTER 2005

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5.0 TECHNOLOGY REQUIREMENTS

A program to develop appropriate propulsion system and propellant technology needs to begin now if we hope to establish a cost-effective base on the Moon by the year 2000. The technology plan for propellant/propulsion system must be integrated with the overall Lunar base scenarios and planning. This study recommends certain technology developments for the Earth-Moon transportation system including Lunar propellant production and usage.

The objective of the technology analyses within Task 4 of this study was to identify technology requirements and develop preliminary plans for satisfying those requirements. The scope included propellant supply, propulsion systems, and transportation vehicle systems. With respect to propellant supply, the effort focused heavily on the chemical processing of propellants from the Lunar regolith (see Section 2.0). The Lunar propellant selections (hydrogen, oxygen, silane, aluminum) coupled with mission and vehicle requirements largely determined propulsion system technology requirements. Vehicle system technology requirements include aerobrake systems and various operational considerations.

Tasks 1 through 3 of the study explored a large number of options for providing Earth-Moon transportation and are described in Sections 2 through 4 of this report. A screening process reduced the initial number of vehicle family options. A few were chosen as being most beneficial. Sufficient uncertainty exists in determining optimal system approaches at all levels (from overall space program goals and commitments down to specific system characteristics) and therefore, final selection of systems is premature. However, the concept recommendations made here will serve to identify and characterize technology needs and the associated benefits. A continuation of system studies and funding of technology development activities is needed to allow future selection of optimal systems. A detailed technology development plan is required with a timeline that is compatible with expected future decision points. A preliminary version of such a plan is one end item of this task and is presented later in this section.

This section of the report is organized into four subsections (5.1 to 5.4). Subsection 5.1 delineates the approach taken in developing requirements and plans. Subsection 5.2 presents a Lunar Surface Base development scenario used to establish timelines. Subsection 5.3 presents individual system technology requirements that are key to leading space transportation options. Subsection 5.4 develops the preliminary technology plan.

5.1 Technology Analysis Approach

Technology development activities for the leading or more promising systems must be carried forward in parallel with the goal of providing needed information and meeting expected decision schedules. The final selection of Lunar propulsion systems should follow key, well-planned technology developments and Phase A and B systems analyses.

To project dates when decisions would be needed, Astronautics constructed a Lunar base transportation system development scenario described in Section 5.2. The scenario was based on the mission model supplied by NASA JSC. The mission model first was used to establish Initial Operational Capability (IOC) dates and the evolutionary development of capabilities for major system elements. Once this information was established, lead times for system developments were estimated (based on historical precedence) to establish needed decision dates for "Authority to Proceed" (ATP) with those developments. This then dictated the timetable for system studies and technology development.

The most promising propulsion and transportation system options were determined in Tasks 2 and 3 which were used as guidelines in Task 4 to identify the technology development requirements of each of those recommended options (see Section 5.3). Each technology development item then was described, the work required to achieve it briefly defined, and evaluated in terms of its gross payback potential if implemented. Based on these results and other factors, such as operational considerations, an assessment of need was made to determine whether the needed technology development was "enabling" or "enhancing".

Given the technology need dates from the Lunar Surface Base Scenarios and the individual technology developments, a preliminary, three-year technology development plan was constructed and described in Section 5.4. The plan includes system level studies needed, focused technology analyses activities and technology experimentation and development activities. It was time-phased to provide logical sequencing of analytical and experimental activities.

5.2 Lunar Surface Base (LSB) Scenario

The entire Lunar Surface Base (LSB) Scenario is presented as an evolutionary development. In the pre-LSB IOC years, systems studies/trade-off analyses and experiments lead to technology development and typical phase A, B, and C/D of specific program definition, design, development and implementation. Post-LSB IOC represents technology development missions, pilot systems and continued growth of LSB systems/operations.

As in the current Space Station program, post-LSB IOC activities are as much a part of technology development as the pre-IOC developments. Figure 5-1a depicts the events that are estimated to occur after the IOC date which occurs at the start of year 1 and continues for twenty years. The scenario initially depends on an O/H transportation system using Earth-provided propellants. The Lunar-derived oxygen, hydrogen, aluminum, or silane propellant production and associated propulsion systems would evolve over time. Lunar Remote Sensing, site selection, mining beneficiation, oxygen production, propellant storage, and LSB and Lunar orbit activities will occur throughout the post-IOC era. Lunar remote sensing would be used to update maps of the Lunar surface, identify new resource locations, and continuously monitor the status of excavation and mining activities. Site selection will be a periodic activity throughout the LSB scenario dependent on location of new resource locations and new resource/production requirements. In-situ propellant production, acquisition, mining production, storage, and use proceed through a logical sequence of technology demonstration missions (TDMs), pilot facility construction and operation, and full capability system construction and operation. The Earth and Earth orbit testing of propulsion systems that would use Lunar propellants is assumed to occur simultaneously with, Lunar propellant facility construction/testing.

Pre-IOC technology developments are shown to accommodate the IOC implementation. Figure 5-1b presents the scenario for the ten years prior to IOC for development of major system elements. For example Phase A, Phase B, and Phase C/D program efforts are shown for Lunar oxygen (LLOX). These are preceeded by technology demonstration (2 years) and system level experimentation and technology development (2 years). A 10-year system development cycle is recommended for the Lunar oxygen production system. Subsystem and component level technology developments must preceed this system cycle. The Lunar remote sensing and initial site selection activities must preceed the IOC to establish the LSB site and specific resource locations.

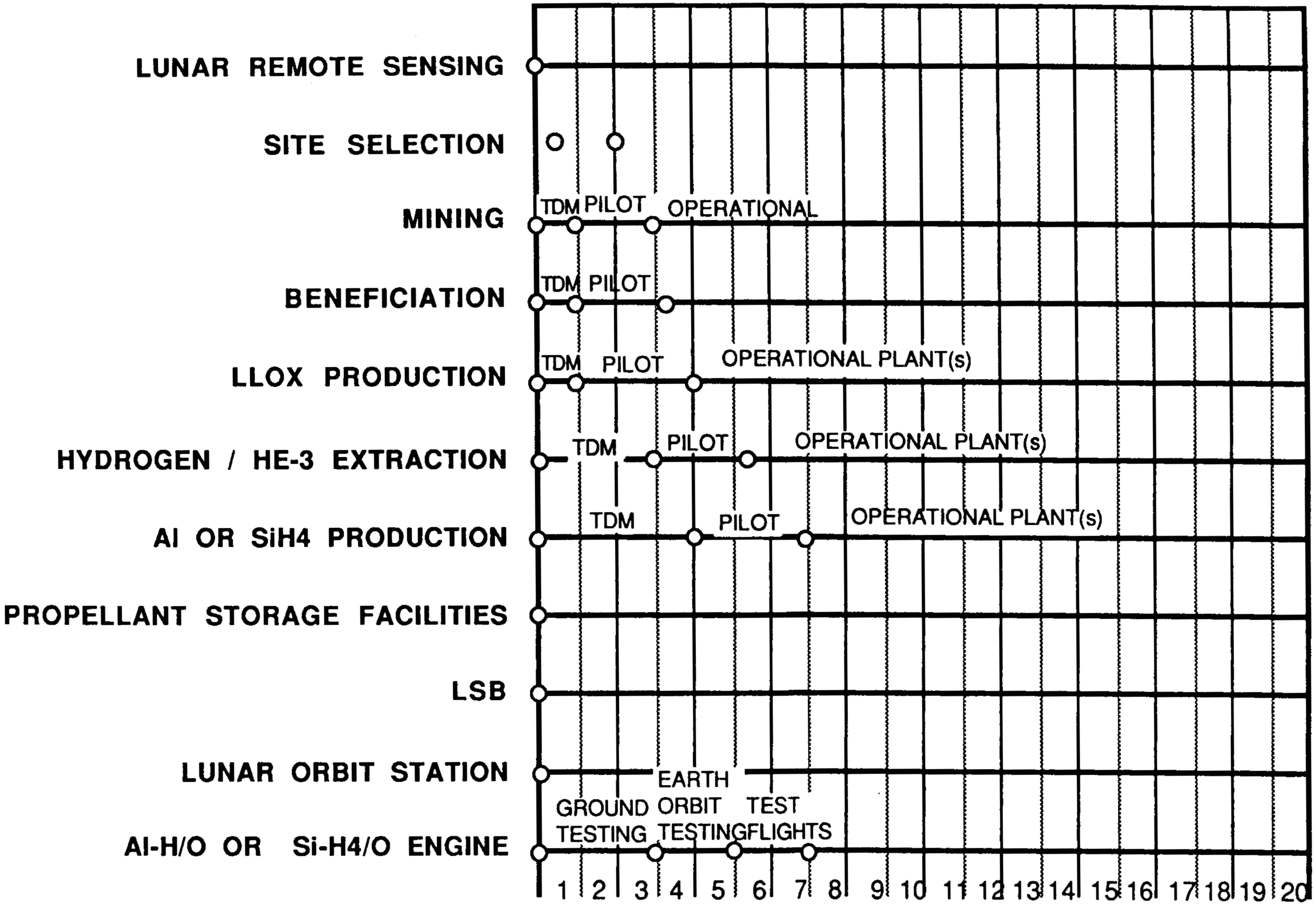

FIGURE 5-1a. LSB PROPULSION / PROPELLANT PRODUCTION SCENARIO (LSB IOC AND BEYOND)

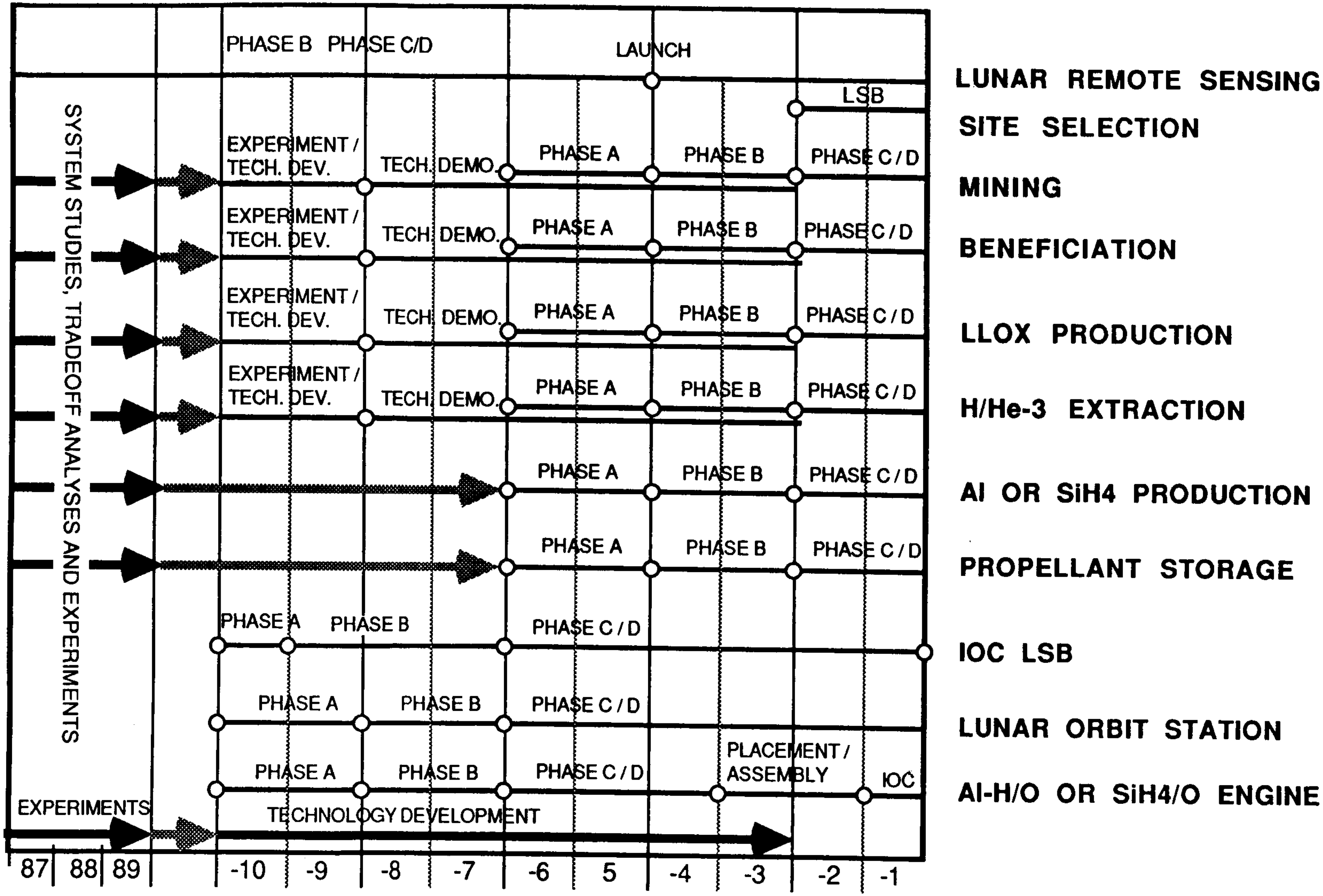

FIGURE 5-1b. LSB PROPULSION / PROPELLANT PRODUCTION SCENARIO (PRE-LSB IOC)

The LSB mission model projects an IOC of 1995. Astronautics LSB Scenario indicates the earliest IOC probably would be about the year 2000. This would allow a 10-year system development cycle preceeded by a 3-year subsystem/component technology development program (experiments). Astronautics also recommends that additional systems and tradeoff studies should be conducted in the next two to three years to steer initial technology developments and that detailed program plan for Lunar base development be prepared.

5.3 Key Technologies

This subsection describes the key technologies identified for leading transportation system options. For each technology needed, development activity is described, payback potential is estimated, and a determination is made as to whether the technology is enabling or enhancing. Enabling and enhancing technologies for Lunar base chemical propulsion systems are summarized in Figure 5-2. Those for propellant supply technologies are summarized in Figure 5-3, and those for vehicle technologies in Figure 5-4. Each technology is discussed below. The scope is limited to chemical propulsion system and propellant related technologies. Additional information on some technologies is included in Appendix G.

5.3.1 Key Propulsion/Vehicle System Technologies

This section provides discussion of technologies listed in Figures 5-2 and 5-4. Studies by current OTV efforts of propellant transfer/handling, extended life, light-weight structures and tankage and automation for propulsion/vehicle systems are directly applicable to Lunar transportation and are, therefore, not discussed here.

5.3.1.1 High Expansion Ratio Nozzle

Design studies and experiments necessary to establish the Isp gain versus weight penalty. Cooling requirements and solutions must be determined. Design concepts should be assessed against tradeoffs of engine sizing, thrust level, single versus multiple engines, and Lunar landing/takeoff operations. An Isp increase to 490 seconds would lead to reduction in propellant mass supplied from Earth by about 6.7% and reduce the mass payback ratio from 3.36 to 3.13 over the H/O systems with Lunar oxygen. This is an enhancing high-value technology -- a relatively low risk method of increasing Isp.

ENABLING	ENHANCING
SPACE BASING CAPABILITIES	HIGH EXPANSION RATIO NOZZLE
Al/O COOLING	HIGH SPEED, LONG LIFE TURBO- MACHINERY
Al-LH2/LO2 COOLING	Al/O COMBUSTION
ALUMINUM FEED IN Al/O ENGINE	Al-H2/O COMBUSTION
	SiH4/O COMBUSTION
	COMBUSTION / EXHAUST COMPOSITION
	ENGINE / SYSTEM HEALTH MONITORING
	ENHANCED O/F RATIO H/O ENGINE
	COOLING OF HIGH O/F RATIO ENGINES
	AUTOMATION / ROBOTICS

FIGURE 5-2. KEY PROPULSION SYSTEM TECHNOLOGIES (CHEMICAL)

ENABLING	ENHANCING
CRYOGENIC / STORAGE	SYNERGISTIC PROCESSING TECHNIQUES
CONSUMABLE RECYCLING IN PROCESSING OPERATION	HIGH TEMPERATURE MATERIALS PROCESSING
PLASMA PROCESSING	SEPARATION OF METAL OXIDES FROM SILICON OXIDES
ELECTRODE LIFETIME AND DURABILITY IMPROVEMENT	SEPARATION OF ILMENITE
SILICATE SEPARATION	RAW MATERIAL COLLECTION, TRANSPORTATION
HYDROGEN EXTRACTION	RESOURCE AND MINING SITE LOCATION

FIGURE 5-3. KEY PROPELLANT SUPPLY TECHNOLOGIES

ENABLING	**ENHANCING**
AEROBRAKING / CAPTURE	LIGHT WEIGHT STRUCTURES, TANKAGE
PROPELLANT TRANSFER / HANDLING	EXTENDED LIFE
SPACE BASING / SERVICING / REFURBISHMENT	AUTOMATION / ROBOTICS
	HEALTH MONITORING

FIGURE 5-4. KEY VEHICLE TECHNOLOGIES

5.3.1.2 High-Speed, Long-Life Turbomachinery

The use of high-speed turbomachinery is one of several approaches to liquid rocket engine performance improvement, but its use also introduces high stresses within the engine design. Typically, higher speed turbomachinery is easier to achieve for smaller engines needing smaller diameter pumps. This technology has cross-impacts with lifetime/durability considerations and the sizing/number of engines. Introducing high-speed turbomachinery will increase the Isp of a given engine and decrease the pump size/weight expense of reduced lifetime, reliability, etc. There is a need to assess the tradeoffs of higher performance versus lifetime and reliability issues which are also very important considerations in a Lunar base scenario. This is an enhancing, high-value technology to improve engine with low mass impacts in engine design.

5.3.1.3 Aluminum/Oxygen Combustion

Analytical studies to project/predict performance are needed. There is a need to develop means to optimize performance and resolve anticipated combustion problems. An AlO_2 film will form on Al particles that retards/prevents combustion. Also Al/O_2 usage leads to large, heavy combustion chambers and Isp lower than H/O. These problems and others need to be addressed for all of the potential Al feed mechanisms proposed. This technology could significantly reduce the mass that must be delivered from Earth to orbit. The work proposed is needed to produce stable, high-efficient combustion in compact combustors of acceptable mass. The effect of MR in performance is believed to be a key issue. This technology should enhance combustion efficiency and performance of the Al/LO_2 propellant combination.

5.3.1.4 Aluminized-Hydrogen/Oxygen Combustion

There is a requirement to conduct analytical studies to predict performance of Al-H/O rocket engines. The formulation and characterization of stable, premixed heterogeneous Al-H fuels will be difficult to accomplish because of the widely disparable physical properties of LH_2 and Al, e.g.,

	m.p.t.(K)	b.p.t. (K)	Density (g/cm^3)
H_2	114	20.2	0.07099
Al	387	2177	2.70

These heterogeneously fuels must be formulated and characterized as rocket propellants at one-g and zero-g. Means to optimize performance and deal with combustion problems must be developed. Also $Al\text{-}H_2/O$ use leads to large, heavy combustion chambers and Isp much lower than H/O. These issues and others need to be addressed. One approach that might help is a tripropellant injector (LO_2, LH_2, LH_2-Al). The work proposed is needed to produce stable, high-efficiency combustion in compact combustors of acceptable weight. This technology enhances combustion efficiency and performance of the Al-LH_2/LO_2 propellant combination and would lead to a significant reduction in mass that must be lifted from Earth to Earth orbit and a mass payback ratio of less than 4.3.

5.3.1.5 Silane/Oxygen Combustion

The SiH_4/O combustion process and products need to be analyzed to predict performance. There is a need to conduct experiments to develop engine performance parameters and deal with problems of engine durability/lifetime. This technology could significantly reduce the mass flow from the Earth's surface to Earth orbit. The work performed is needed to provide stable high-efficiency combustion in compact combustion chambers of acceptable mass.

5.3.1.6 Combustion Efficiency and Combustion Chamber and Nozzle Durability for Si and Al Based Fuels

There is a need to conduct spectroscopic analysis of combustion, exhaust to develop more complete understanding of combustion processes and efficiencies, and to investigate problems of slag formation and engine internal surface degradation and erosion. This technology development work can provide better performing, longer-lived engines. This technology enhances the performance, lifetime, reliability of engines that produce silicate and metallic oxide exhaust products.

5.3.1.7 Engine/System Health Monitoring

Advanced, long-lifed, space-based propulsion systems will need integrated condition monitoring and control system hardware with predictive capability. An on-board system is needed that will monitor the condition of components such as pumps, injectors, valves, etc. and predict their future condition. This requires development of sensor technology, systems integration, system soft/hardware, and prediction data base generation. This is one of the keys to successful space basing and represents an enhancing technology for economical Earth-Moon operation.

5.3.1.8 Enhanced O/F Ratio H/O Engines

There is a need to assess tradeoffs with respect to O/F ratio, Isp, weight, reliability and durability of high O/F ratio engines. A slight increase in O/F (below 8) may reduce Earth launch mass. Our analysis indicates that large increases in O/F ratio do not appear to reduce overall Earth launch mass and cost of the LSB mission scenario. Higher O/F ratios slightly higher than 6 may enhance the Lunar transportation system by reducing hydrogen supplied from Earth and increasing the dependence on Lunar-produced oxygen.

5.3.1.9 Cooling of High O/F Ratio H/O Engines

It is apparent that more analysis and tradeoffs of optimum MR for Lunar missions with Lunar oxygen availability are required. Experimental programs to develop and prove oxygen cooling techniques for these engines are needed. This cooling technology is required for very high MR engine operations and for Al/oxygen and Al-H/oxygen engines. Oxygen cooling is an enabling technology for an enhancing engine option and is considered in enhancing technology to Lunar base transportation.

5.3.1.10 Space Basing Capability

Propulsion systems must be compatible with space environments and operations on the Lunar surface and in cislunar space, in both operational and storage/maintenance modes. Issues that must be addressed include Lunar dust contamination, surface temperature fluctuations, design for semi-robotic maintenance and pre/post flight activities. Successful resolution of these issues is necessary for successful space basing. This technology area is <u>enabling</u> with respect to space basing.

5.3.1.11 Aluminum/Oxygen Engine Cooling

The conduct of preliminary design and analysis efforts to explore potential cooling solutions. Experimental and developmental efforts must be performed to verify the theory and implement solutions. Problems related to very high combustion temperatures and low efficiency cooling from Al/O propellants are anticipated. Al/O engines must be run at high MR (oxygen rich) to produce gas to expand out the nozzle but this produces a wall compatibility problem. A significant reduction in mass that must be launched from Earth to Earth orbit can be achieved. This proposed work must also consider that the engine will eventually be long-lifed, reuseable, and man-rated.

5.3.1.12 **Al-H/Oxygen Engine Cooling**

A preliminary design and analyses study on Al-H/O engine cooling concepts must be conducted to explore potential solutions followed by experimental and development efforts to verify and implement solutions. There is a need to address problems of very high combustion temperatures with two phase (fuel) coolant and low heat flux (oxygen) coolant. This technology could significantly reduce the mass that must be lifted from Earth to Earth orbit. The work proposed must also consider that this engine must also be long-lifed, reuseable, and man-rated. This technology is enabling to successful operation of Al-H/O engine.

5.3.1.13 **Aluminum Feed in Al/O Engine**

Analytical and experimental programs are needed for introducing/feeding aluminum into a combustion chamber to be burned with LO_2 for the Al/oxygen hybrid engine. Candidate methods include: coated Al in gelled LO_2; liquid Al injection (with associated back-up systems); solid Al alloy plus LO_2; and Al powder introduction is similar to diesel or coal combustion. A significant reduction in mass that must be lifted from Earth to orbit can be achieved and a reduction in mass payback ratio to 2.56. This technology is enabling to the Al/oxygen hybrid engine concept.

5.3.1.14 **Aerobrakeing/Capture**

The use of aerobrakes is either enabling or enhancing to the Lunar transportation system depending on the availability and use of Lunar oxygen. The development of a lightweight (15% of Earth entry mass) system is needed if aerobrakes are to be of value. Without Lunar oxygen production on the Moon aerobrake technology enables an efficient transportation system. With Lunar propellant available the aerobrake only enhances the transportation system. If Lunar oxygen is transported to LEO, the aerobrake again becomes an enabling technology. However, this Lunar propellant return scenario will require a very large transportation system with an aerobrake with mass of approximately 13 MT. Technologies in large, low mass aerobrake systems should be continued and tested.

5.3.2 **Propellant Acquisition Technology**

5.3.2.1 **Locate Desired Lunar Resources and Mining Sites with Minimal Multi-specimen Sampling**

Remote sensing and other techniques such as nuclear bombardment have been used successfully on Earth and should be employed on the Moon to detect mean particle size, surface roughness, and possibly chemical composition. The development and use of such techniques should reduce crew-hours (100's of hours) and transport equipment requirements. This is key to the site location of any Lunar resource processing plant. This technology would enhance the performance of preprocessing and processing techniques.

5.3.2.2 Collection and Transportation of Raw Materials

Terrestrial mining machinery has been conceptually adopted for the Moon. More innovative equipment and techniques should be investigated to better utilize Lunar resources. Specific functional needs may include scooping, crushing, dredging, plowing, and hauling. Also techniques of integrating the collection/transportation systems and the preprocessing systems. This technology would reduce the consumable requirements required to take the Lunar regolith from its natural state. This is enhancing technology to reduce basing and support requirements.

5.3.3 Propellant Preprocessing/Beneficiation Technology

5.3.3.1 Separation of Ilmenite

Some experimentation of terrestrial simulant and Apollo ilmenite separation has been achieved (Agosto, 1983) using magnetic and electrostatic separation techniques, collecting up to 51-55% ilmenite concentrate in one pass. However, an inverse relationship exists between the concentration and recovery percentage of ilmenite. Multiple passes on larger scale separations should be accomplished. The effects of the lack of Fe^{3+} in Lunar ilmenite separation should be investigated. This technology could reduce the specific consumption of oxygen production by 30 - 40%. This technology would enhance efficiency of Hydrogen Reduction oxygen processing by reducing the amount of initial mass to be processed. It is a required and enabling technology for oxygen production using Magma electrolysis. SOA electrodes cannot accommodate the presence of silicate material without extreme degradation.

5.3.3.2 Separation by Silicate

Separation experimentation is needed using magnetic/electrostatic techniques coupled with sifting and/or other regolith refinement techniques to separate out $Al_2O_3 \cdot SiO_2$ from Anorthosite. This technology reduces the consumables required to produce aluminum by approximately 60%. This technology would allow processing of specific metals such as aluminum and enhance the performance of the acid leach or vapor ion separation production techniques.

5.3.3.3 Separation of Metal Oxides from Silicon Oxides

Potential silicate separation techniques must be identified to enhance production efficiency (none uncovered to date). Experimentation of most valuable concepts should be pursued on Lunar simulant and/or Apollo samples. This technology could reduce the consumable requirements in the Acid Leach process by 45%. Removing silicates would also reduce hardware corrosion rates.

5.3.4 Propellant Pocessing/Production Technology

5.3.4.1 Hydrogen Extraction

Identification and analysis of potential hydrogen extraction techniques (such as RF/microwave, thermal release, and vibrational release) and experimentation in the laboratory with Lunar regolith samples impacted with a hydrogen plasma are required. This technology could reduce Earth launch mass from the Lunar oxygen scenario by 30%. (A 65% reduction from the baseline scenario.) This technology would enable a 100% Lunar-derived and Lunar-based H/O OTV and lander.

5.3.4.2 Consumable Recycling

The hydrogen reduction oxygen processing techniques recycle 90-95% of the hydrogen through hydrogen release from TiO_2/FeO power and electrolysis of H_2O. Recycling of HF in the acid leach process requires not only recovery of H through electrolysis, but also recovery of F from SiF_4 and AlF_3. Additional analysis should be given to these sub-processes to identify feasible improvements. This technology could reduce consumable requirements approximately 50% thereby making the Acid Leach process competitive. This technology would <u>enable</u> many processes including acid leaching as an economical aluminum and oxygen processing.

5.3.4.3 Electrode Lifetime and Durability

There is a requirement to identify and categorize existing electrode materials. Carbon graphite and platinum are seen as currently viable electrodes but suffer from immense degradation from silicates and high temperatures. Research should be conducted for longer-lasting electrodes more tolerable of the molten slag content. Electrodes material could also be derived from Lunar resources (although with degraded performance) to still potentially reduce the Earth-support requirements. This technology could reduce consumable requirements of the molten electrolysis oxygen production by 50%. This is near-enabling technology for the magma electrolysis for oxygen to reduce the replacement requirements of electrode resupply.

5.3.4.4 Plasma Processing

Not much is known to date on this process. Key technologies include plasma sustenance, plasma flow and ion separation. An initial analysis of the feasibility should be done as Phase 1 followed by a conceptual definition in Phase 2 and finally some experimental work in Phase 3. This technology could reduce consumables to near-negligible fractions (e.g. 5 g of production material per metric ton of O_2 produced).

5.3.4.5 High Temperature Materials Processing

Key development areas include the container of the molten material, valves and piping, and heater elements. Lunar materials may be produced into refactory linings such as MgO. Constant temperatures must be maintained to avoid burn-through or build-up in valves and piping. Heater concepts should avoid direct contact with magma to avoid degradation. All three areas require further analysis to determine a feasible Lunar technique. If this technology is not developed, then maintenance on high temperature processing facilities could double the equipment requirements estimated. This technology enhances high temperature processing to increase productivity of magma electrolysis and other high-temperature processing techniques, and reduce maintenance problems.

5.3.4.6 Synergistic Processing

Identification of total LSB materials and potential areas of multiple resource processing and processing techniques is required. There is a requirement to refine the processes with concentration on separation of specific raw materials from the output materials and actual performance efficiency estimates in most feasible processes. If synergistic processing is practical, the cost of producing propellant could be spread among the entire Lunar base and may, in some cases, be a nearly-free biproduct from another process. This would enhance the overall raw materials availability in support of the total Lunar base.

5.3.5 Propellant Storage/Delivery Technology

5.3.5.1 Long-Term Storage

Another important issue is the identification of effects of Lunar storage, permanent shadow storage, and recycling techniques to reduce boil-off and energy requirements for cryogens. Also storage requirements and concepts for mixed hybrids and powdered fuels need to be identified. This technology could save on the order of 10% of the propellant stored on the Moon. This technology would enhance storage efficiency by reducing boil-off of cryogens and hybrids.

5.3.5.2 Cryogenic Cool-Down

Assessing the potential of magnetic refrigerators, hybrid magnetic/gas-cycle refrigerators and other potential cryogenic cooling techniques for oxygen, LH_2 and possibly solid O_2 should be investigated. If economic feasibility is evident, experimentation should be pursued. A significant reduction of power consumption and equipment mass over current gas cycle refrigerators could be achieved. This technology is enhancing and would decrease energy and equipment consumable requirements.

5.4 Technology Plan

Consistent with the outcome of Subsection 5.2 and using the results of Subsection 5.3 a preliminary technology plan was defined: see Figure 5-5 (a-d). The plan covers a three-year time span and includes both studies and specific technology development activities. Bolded items refer to technology developments of the recommended propulsion and propellant supply systems.

The first item in the plan is a two-year effort to upgrade the mission model and conduct various analyses and tradeoffs to enable proper system selections. Vehicle/propulsion systems analyses should include: combustion analysis, propellant characterization, cooling analysis, sizing, lifetime/durability assessment, operations, technology, and costs. The scope of these analyses should include hydrogen-oxygen, silane-oxygen, and aluminum-hydrogen-oxygen chemical systems, as well as electric, nuclear, EML and other systems. In addition, propulsion assist mechanisms (e.g. aerobrake and tether systems) need to be further explored.

Propellant processing analyses are needed to determine feasibility and assess efficiencies for all propellants being considered. The analyses must address acquisition, preprocessing/beneficiation, processing/production and storage/transfer.

Finally, an extensive series of propulsion system experiments needs to be conducted to determine which systems can provide needed performance and operational characteristics. The combustion of solid aluminum in oxygen, liquid alu-

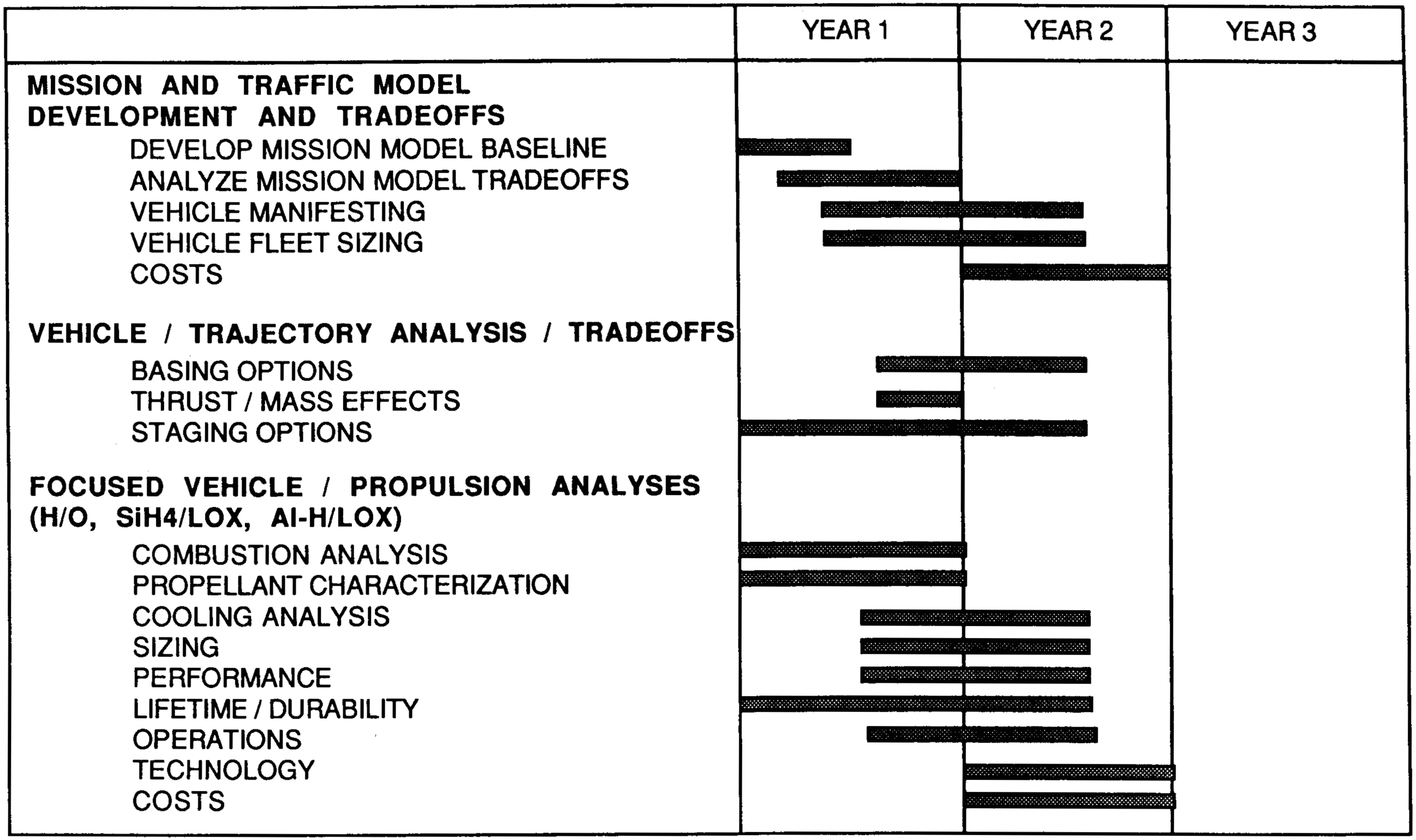

FIGURE 5-5a. TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

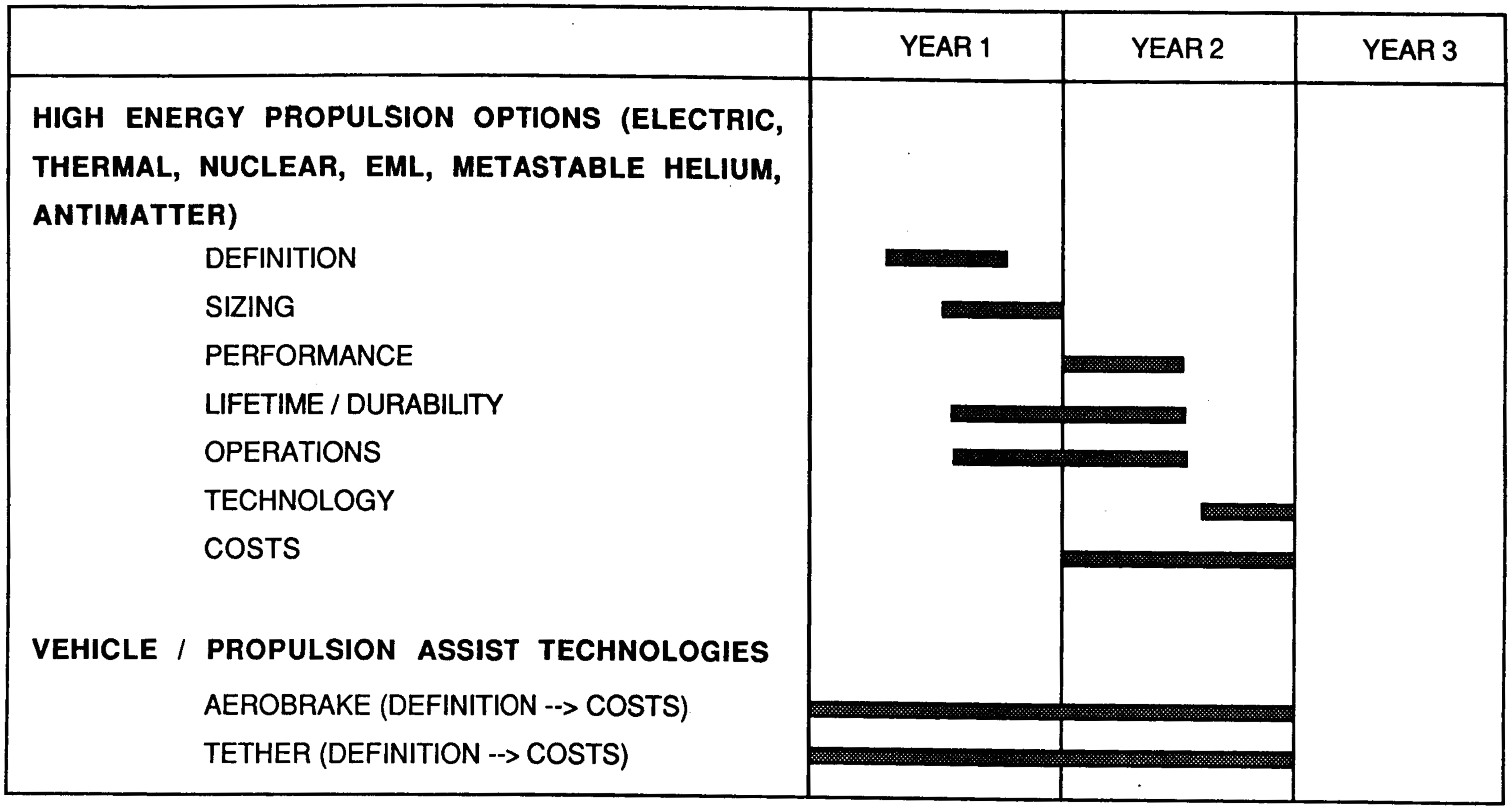

FIGURE 5-5b. TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

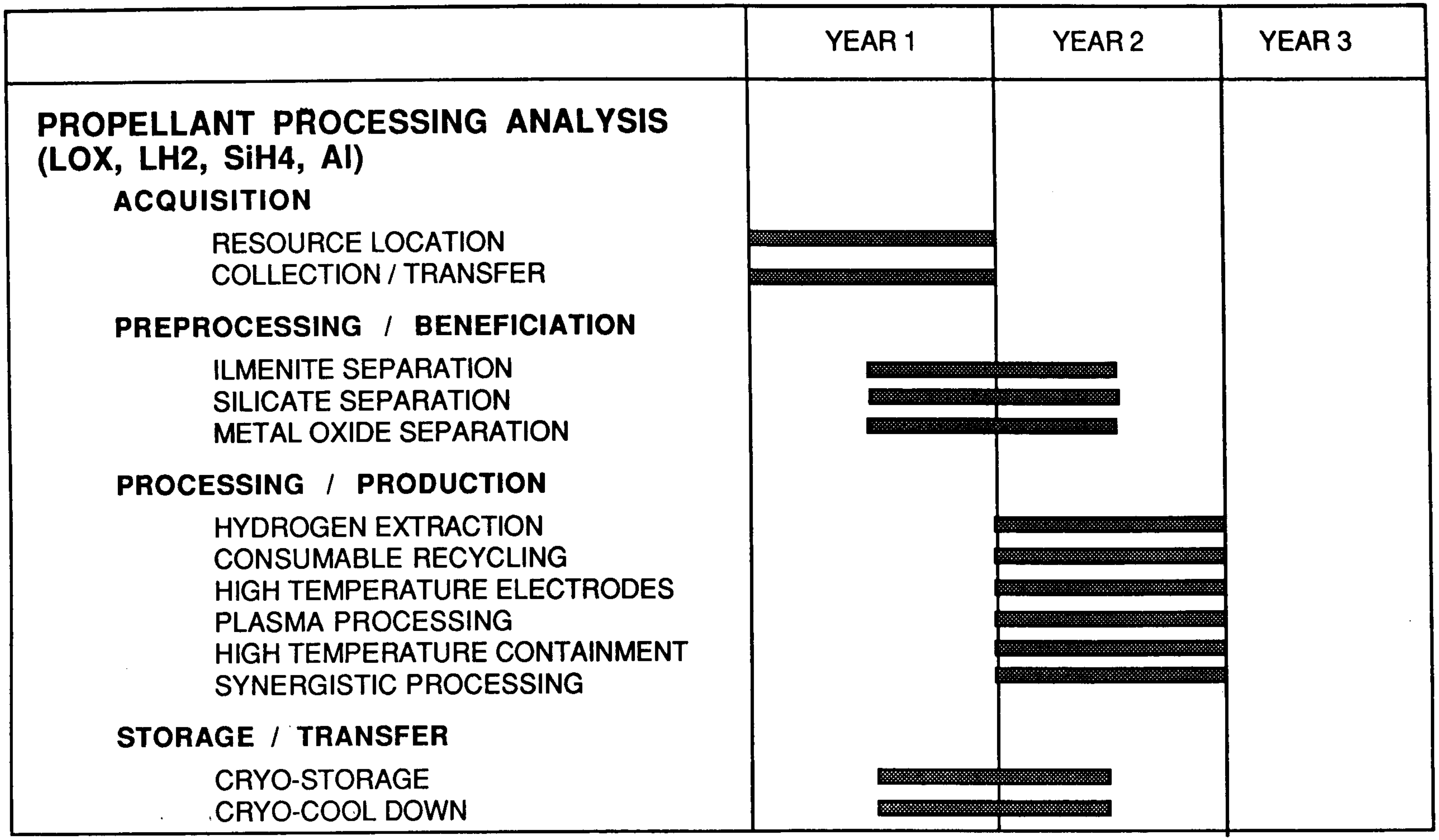

FIGURE 5-5c. TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

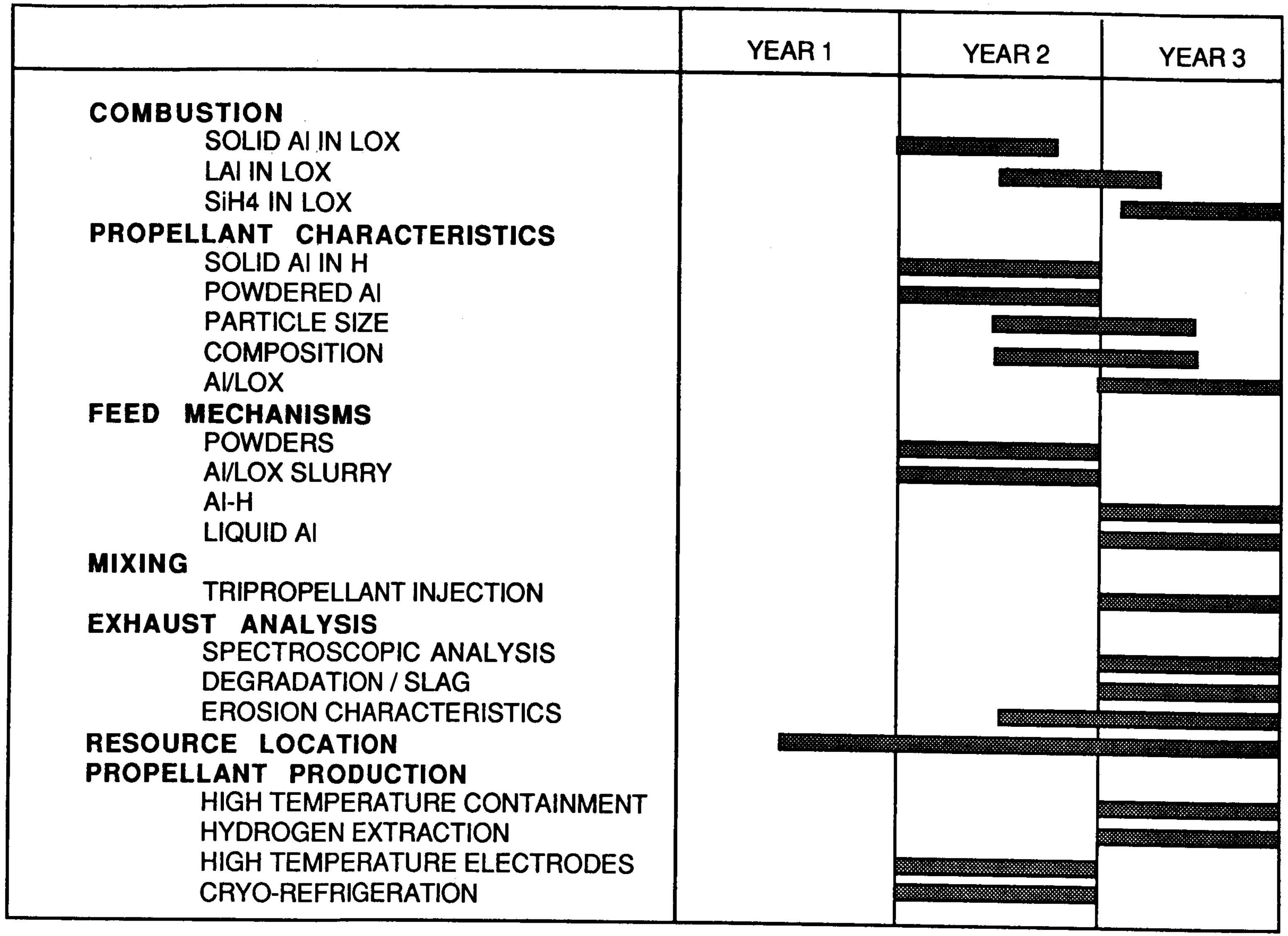

FIGURE 5-5d. TECHNOLOGY EXPERIMENT PLAN

minum in oxygen and silane in oxygen needs to be tested to establish performance and operational characteristics. Feed mechanisms are a significant concern and need to be tested and evaluated.

This plan is ambitious and will require a sizable investment of NASA resources. It is based on a three-year schedule that can support a transportation system IOC about the year 2000. Changes in the IOC date would allow corresponding changes in the technology plan.

APPENDIX A

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APPENDIX B

PROPELLANT PROCESSING
RESOURCE AND EQUIPMENT WEIGHT
ESTIMATES

B.0

The following calculations have been derived to obtain a fair comparison of the propellant processing candidates reviewed in this study. Values for energy and hardware requirements may not reflect the actual values determined by experimentation but can be used with other process values calculated in this study for resource requirement comparisons.

B.1 Solar Wind Gas Extraction

Lunar Regolith Requirement

Lunar regolith requirement is visually determined for a baseline production of 10 MT O_2 or 10 MT/month. Since O_2 is not a direct product of Solar Wind Gas Extraction, we will discuss its resource requirements using enough mare to supply 10 MT O_2 assuming 100% of all O_2 in the mare is processed by some other technique.

$$10 \text{ MT } O_2 \times \frac{1\text{g mare}}{0.413\text{g } O_2} = 24.2 \text{ MT Mare}$$

Reactant Requirement

No reactants are required.

Products

The following chart lists concentrations of solar wind gases in the mare.

TABLE B.1-1 Solar Wind Gas Concentration in Lunar Regolith

Solar Wind Gas	Concentration (PPM,Wt)		
	Mare	Highlands	Basin Ejecta
H	54.8	56.0	76.5
He	28.5	6.0	8.0
He-3	0.01	0.002	0.003
N	95.4	98.0	121.0
Ne	2.75	1.0	2.0
Ar	0.8	1.2	1.0

For extracting these gases from the mare, we will assume 100% efficiency though actual efficiency may be closer to 50%. The following table shows production potential for each solar wind gas. The values for processing 24.2 MT Mare (containing 10 MT O_2).

TABLE B.1-2 Solar Wind Gas Production

Solar Wind Gas	Potential Production (g) Potentials Mare
H	1325
He	690
He-3	0.24
N	2310
Ne	67
Ar	19

Resource and Equipment Weight Estimates

All resource and equipment weight estimates are based on production of 10 MT O_2 or 10 MT O_2/month. All estimates do not include beneficiation of power source hardware weights.

B.1 Solar Wind Gas Extraction

B.2 Hydrogen Reduction

Lunar Regolith Requirement

Assuming 100% of all O_2 from FeO can be extracted from ilmenite, from 1g ilmenite:

$$0.449\text{gFeO} \times \frac{1 \text{ mole FeO}}{71.8\text{g FeO}} \times \frac{1 \text{ mole } O_2}{2\text{molesFeO}} \times \frac{32.2\text{g}O_2}{1\text{mole}O_2}$$

$$= 0.10068\text{g } O_2$$

The ilmenite required for 10 MT O_2 production is·

$$\frac{1\text{g Ilmenite}}{0.10068\text{g}O_2} \times 10 \text{ MT } O_2 = 99{,}325 \text{ Kg Ilmenite}$$

$$= 99.3 \text{ MT Ilmenite}$$

The mare required to obtain 99.3 MT ilmenite, assuming ilmenite comprises 15wt% of the mare, is:

$$99.3 \text{ MT Ilmenite} \times \frac{1\text{g mare}}{0.15\text{g Ilmenite}} = 662.2 \text{ MT Mare}$$

Reactant Requirement

Hydrogen gas is the only required reactant. Though H_2 initially must be imported from Earth, there is good potential for extracting it from lunar regolith. The reduction reaction is:

$$FeO \cdot TiO_2 + H_2 = Fe + H_2O + TiO_2$$

The hydrogen required is determined by stoichiometry:

$$10 \text{ MT } O_2 \times \frac{1 \text{ mole } O_2}{32.2g O_2} \times \frac{2 \text{ moles } H_2O}{1 \text{ mole } O_2} \times \frac{1 \text{ mole } H_2}{1 \text{ mole} H_2O} \times \frac{2g H_2}{1 \text{mole} H_2}$$

$$= 311 Kg\ H_2 = 0.31 \text{ MT } H_2$$

Reactant Recovery Potential

It can be assumed that 95% of all H_2 used may be recovered. It is recovered from two different places. It is a product of the H_2O electrolysis which produces O_2. The H_2 interstitially trapped in the melt may be released during heat of microwaves. The amount of H_2 recoverable is:

$$311 \text{ Kg } H2 \times 0.95 = 295 \text{ Kg } H_2 = 0.30 \text{ MT } H_2$$

Energy Requirement

1. Thermal

The thermal power required is determined from the thermodynamics of the hydrogen reduction reaction:

$$\frac{42.0KJ}{1 \text{ mole Ilmenite}} \times \frac{1 \text{ mole Ilmenite}}{151.7g \text{ Ilmenite}} \times 99.3 \text{ MT Ilmenite}$$

$$= 7,639 \text{ KWhrs}$$

2. Electrical

The electrical power required is determined by the dissociation free energy of H_2O = 249 KJ/mole H_2O. Using stoichiometry:

$$\frac{249 \text{ KJ}}{1 \text{mole } H_2O} \rightarrow \frac{4.32 \text{ KWhrs}}{1Kg\ O_2} \times 10 \text{ MT } O_2 = 43,200 \text{ KWhrs}$$

Equipment Weights

Since water vapor can continuously be captured, we will assume a continuous automated process. We will also assume a chamber volume of $0.5m^3$, wall thickness of 4cm and material density of $7500Kg/m^3$.

$$4/3\ t4\ r^3 = 0.5m^3,\ r = 0.49n$$

$$\text{Reduction vessel weight} = \frac{7500Kg}{m^3} (4/3\ tr\ (0.53^3 - 0.49^3))$$

$$= 981 \text{ Kg} = 0.98 \text{ MT}$$

The weight of the elecrolysis hardware is determined using the NASA Technology Model Volume 1, Part B. The equation used to determine H_2O electrolysis hardware weight is:

$$\frac{150}{1000} x + 350, \text{ where } x = \text{man-days of oxygen consumption}$$

From the Technology Model, we know that 17.3 lbs O_2/day is consumed for 8 people. This number can also be expressed as 0.98Kg O_2/man-day. Production of 10 MT O_2 is equivalent to:

$$10 \text{ MT } O_2 \times \frac{1 \text{ man-day}}{0.98\text{Kg}O_2} = 10{,}204 \text{ man-days}$$

Using the equation for electrolysis hardware weight, we have:

$$\frac{150}{1000} (10{,}204 \text{ man-days}) + 350 = 1880\text{lbs} = 854\text{Kg} = 0.85\text{MT}$$

Total required equipment weight is:

$$0.98 \text{ MT} + 0.85 \text{ MT} = 1.83 \text{ MT}$$

B.3 Magma Electrolysis

Lunar Regolith Requirement

Assuming 50% of all O_2 from FeO can be extracted from ilmenite, from 1g ilmenite:

$$0.449\text{gFeO} \times \frac{1\text{moleFeO}}{71.8\text{gFeO}} \times \frac{1\text{mole}O_2}{2\text{molesFeO}} \times \frac{32.2\text{g}O_2}{1\text{mole}O_2} \times 0.50 = 0.0503\text{g}O_2$$

The ilmenite required for 10 MT O_2 production is:

$$\frac{1\text{g Ilmenite}}{0.0503\text{g}O_2} \times 10 \text{ MT } O_2 = 198{,}807\text{Kg Ilmenite} = 199 \text{ MT Ilmenite}$$

The mare required to obtain 199 MT Ilmenite, assuming ilmenite comprises 15wt% at the mare, is:

$$199 \text{ MT Ilmenite} \times \frac{1\text{g mare}}{0.15\text{g Ilmenite}} = 1325 \text{ MT Mare}$$

Reactant Requirement

If no fluxes are used, no reactants are required.

Energy Requirement

1. Thermal

The thermal energy required is determined using Mcp T and the thermodynamic properties of ilmenite.

Cp of Ilmenite	=	130 J/mole K
Process Temperature	=	1350°C = 1623 K
Initial Temperature	=	25°C = 298K
Moles of Ilmenite	=	$198{,}807\text{Kg} \times \frac{1\ \text{mole}}{0.151\ \text{Kg}}$
	=	1,310,527 moles
(1,310,527) (130) (1623-298)	=	62,705 KWhrs

2. Electrical

Using the dissociation energy of FeO and assuming 50% of FeO electrolyzed:

$$\frac{269\ \text{KJ}}{\text{mole FeO}} \times \frac{1\text{moleFeO}}{71.8\text{gFeO}} \times 89{,}264{,}343\text{gFeO} \times 0.5 = 46{,}449\ \text{KWhrs}$$

Equipment Weights

Assuming a chamber vol of 1.5m^3 and a continuous automated process, wall thickness of 2cm and material density of 7500Kg/m^3:

$$4/3 \quad r^3 = 1.5\text{m}^3,\ r = 0.71\text{m}$$

$$\text{Electrolysis Cell Weight} = \frac{7500\text{Kg}}{\text{m}^3} \left(4/3 \quad (0.73^3 - 0.71^3)\right)$$

$$= 981\text{Kg} = 0.98\ \text{MT}$$

Electrode corrosion rate using graphite electrodes is observed in terrestrial industry as 7Kg/day. This number reflects the electrode corrosion problem of this process but is very conservative. An optimum material may be found to reduce this rate substantially.

B.4 Carbochlorination

Carbochlorination resource and equipment weights have been determined in two ways:

I Production of O_2

II Production of Al and O_2

Calculations for the production of Al include the production of 10 MT O_2.

I. Production of O_2

<u>Lunar Regolith Requirement</u>

From stoichiometry and the composition of the mare regolith:

$$\frac{4.76\text{g Anorthite}}{1\text{g}O_2} \times 10\text{ MT }O_2 = 47.6\text{ MT Anorthite}$$

The mare regolith to obtain 47.6 MT anorthite, assuming anorthite comprises 20wt% of the mare, is:

$$47.6\text{ MT Anorthite} \times \frac{1\text{g mare}}{0.2\text{g Anorthite}} = 238\text{ MT Mare}$$

<u>Reactant Requirement</u>

The carbon required is determined by the stoichrometry of the carbochlorination reactions. To process 1g anorthite:

$$0.337\text{g}Al_2O_3 \times \frac{1\text{mole}Al_2O_3}{102.0\text{g}Al_2O_3} \times \frac{3\text{moleC}}{1\text{mole}Al_2O_3} \times \frac{12.0\text{gC}}{1\text{moleC}} = 0.1189\text{gC}$$

$$0.181\text{g}CaO \times \frac{1\text{mole}CaO}{56.0\text{g}CaO} \times \frac{1\text{moleC}}{1\text{mole}CaO} \times \frac{12.0\text{gC}}{1\text{moleC}} = 0.0388\text{gC}$$

$$0.461\text{g}SiO_2 \times \frac{1\text{mole}SiO_2}{60.0\text{g}SiO_2} \times \frac{2\text{molesC}}{1\text{mole}SiO_2} \times \frac{12.0\text{gC}}{1\text{moleC}} = 0.1844\text{gC}$$

$$0.1189 + 0.0388 + 0.1844 = \frac{0.342\text{gC}}{1\text{gAnorthite}}$$

Since 47.6 anorthite is required for production of 10 MT O_2:

$$47.6\text{ MT Anorthite} \times \frac{0.342\text{gC}}{1\text{gAnorthite}} = 16{,}279\text{KgC} = 16.3\text{ MT C}$$

The cholorine required is determined using stoichiometry and the composition of anorthite:

$$\frac{11.3gCl_2}{1gAl} \times \frac{1gAl}{1.18gO_2} \times 10\ MT\ O_2 = 95.8\ MTCl_2$$

Reactant Recovery Potential

Assume of al Cl_2 from $CaCl_2$ is recovered. From lg anorthite:

$$0.181gCaO \times \frac{1moleCaO}{56.0gCaO} \times \frac{1moleCaCl_2}{1moleCaO} \times \frac{1moleCl_2}{1moleCaCl_2} \times \frac{70.9gCl_2}{1moleCl_2} = 0.229gCl_2$$

Assuming 50% of all Cl_2 from $SiCl_4$ is recovered, from lg anorthite:

$$0.461gSiO_2 \times \frac{1moleSiO_2}{60.0gSiO_2} \times \frac{1moleSicl_4}{1moleSiO_2} \times \frac{2molesCl_2}{1moleSiCl_4} \times \frac{70.9gCl_2}{1moleCl_2}$$

$$= 1.090gCl_2$$

$$1.0900gCl_2 \times (0.50) = {}_x0.545gCl_2$$

Total Cl_2 recovered from processing lg anorthite:

$$0.229 + 0.545 = \frac{0.774gCl_2}{1gAnorthite}$$

To produce 10 MT O_2, 47.6 MT Anorthite are required:

$$\frac{0.774gCl_2}{1gAnorthite} \times 47.6\ MT\ Anorthite = 36.8\ MT\ Cl_2$$

Assume 100% of all C from CO produced in the carbochlorination unit is recovered. Total CO produced from processing lg anorthite is:

$$0.337gAl_2O_3 \times \frac{1moleAl_2O_3}{102.0gAl_2O_3} \times \frac{3molesCO}{1moleAl_2O_3} = 0.0099\ moles\ CO$$

$$0.181gCaO \times \frac{1moleCaO}{56.0gCaO} \times \frac{1moleCO}{1moleCaO} = 0.00323\ moles\ CO$$

$$0.461gSiO_2 \times \frac{1moleSiO_2}{60.0gSiO_2} \times \frac{2molesCO}{1moleSiO_2} = 0.0154\ moles\ CO$$

$$0.0099 + 0.00323 + 0.1054 = 0.0285\ \frac{moles\ CO}{1g\ Anorthite}$$

For 10 MT O_2 production, 47.6 MT anorthite are required:

$$0.0285 \ \frac{\text{molesCO}}{\text{1g Anorthite}} \times 47.6 \text{ MT Anorthite} = 625{,}000 \text{ moles CO}$$

$$1{,}356{,}600 \text{ moles CO} \times \frac{\text{1moleC}}{\text{1moleCO}} \times \frac{\text{12.0gC}}{\text{1moleC}} = 16{,}279\text{KgC} = 16.3 \text{ MTC}$$

Energy Requirement

1. Thermal

For carbochlorination reactions and production of 10 MT O_2 using stoichiometry and thermodynamics:

$$\frac{\text{50KJ}}{\text{1moleAnorthite}} \times \frac{\text{1moleAnorthite}}{\text{278gAnorthite}} \times 47.6 \text{ MT Anorthite} = 2{,}380\text{KWHrs}$$

Since O_2 is recovered from Al_2O_3 and CaO, only 0.0099 + 0.00323 moles of CO are processed.

$$0.0099 + 0.00323 = 0.01313 \text{ moles CO/1g Anorthite}$$

$$\frac{\text{0.01313 moles CO}}{\text{1g Anorthite}} \times 47.6 \text{ MT Anorthite} = 625{,}000 \text{ moles CO}$$

For heating of CO to convert to CO_2:

$$\frac{\text{280KJ}}{\text{1moleCO}} \times 625{,}000 \text{ moles CO} = 48{,}611 \text{ kWhrs}$$

For heating CO_2 in Bosch Reactor:

$$\frac{\text{180KJ}}{\text{moleCO}_2} \times \frac{\text{1moleCO}_2}{\text{1mole CO}} \times 625{,}000 \text{ mole CO} = 18{,}750 \text{ kWhrs}$$

Total thermal for processing:

$$2{,}380 + 48{,}611 + 18{,}750 = 69{,}741 \text{ kWhrs}$$

$$\text{Total Thermal} = 15{,}504 + 69{,}741 = 82{,}245 \text{ kWhrs}$$

For reclamation of Cl_2 from $CaCl_2$, $CaCl_2$ must be heated to 1000°C. Using stoichiometry and thermodynamics:

$$\frac{785KJ}{moleCaCl_2} \times \frac{1moleCaCl_2}{1moleCaO} \times \frac{1moleCaO}{56.0gCaO} \times \frac{56gCaO}{40gCA} \times 2,844,000gCa$$
$$= 15,504 \text{ KWhrs}$$

2. Electrical

Electrical power is directly used for electrolysis of H_2O. From dissociation energy of H_2O:

$$4.32 \frac{KWhs}{kgO_2} \times 10 \text{ MT } O_2 = 43,200 \text{ kWhrs}$$

Equipment Weights

Weight of the carbochlorination unit is based on a mass flow rate of 60kg/hr, 4cm wall thickness and material density of 7500kg/m^3.

600kg --> 0.6MT

Weights for the condensers was obtained from Strobridge's "Cryogenic Refrigerators - An Updated Study" and is based on a 90% duty cycle and 250 MT/yr O_2 production.

10 MT

Bosch reactor weight is obtained from Quattrone's "Extended Mission Life Support Systems".

1.0 MT

The H_2O electroylsis hardware weight is calculated using the equation from the NASA Technology model:

$$\frac{150}{1000} x + 350 \text{ where } x = \text{man-days of } O_2 \text{ consumption}$$

Production of 10 MT O_2 is equivalent to 10,204 man-days of O_2 consumption. Weight of the H_2O electrolysis cell is estimated as:

0.85 MT

Total Equipment weight = 12.45 MT

II. Production of Al and O_2

Lunar Regolith Requirement

The production of Al includes production of 10 MT O_2. The lunar regolith requirements are the same as in part I.

238 MT Mare

47.6 MT Anorthite

Products

From stoichiometry:

5.6 kg Anorthite -> 1kg Al

For production of 10 MT O_2, 47.6 MT Anorthite is required.

$$\frac{1 \text{ kg Al}}{5.6\text{kgAnorthite}} \times 47.6 \text{ MT Anorthite} = 8{,}500 \text{ KgAl} = 8.5 \text{ MT Al}$$

Total products obtained are:

10 MT O_2

8.5 MT Al

Reactant Requirements

Reactant requirements are the same as in part I.

95.8 MT Cl_2

16.3 MT C

Reactant Recovery Potential

Carbon is recovered in the same manner as in part I.

16.3 MT C

Chlorine recovered from $CaCl_2$ and $SiCl_4$ remains the same as in part I.

36.8 MT Cl_2

Additional Cl_2 may be recovered from $AlCl_3$:

$$\frac{0.337gAl_2O_3}{1gAnorthite} \times \frac{1moleAl_2O_3}{102gAl_2O_3} \times \frac{2molesAlCl_3}{1moleAl_2O_3} \times \frac{1.5molesCl_2}{1molesAlCl_3} \times \frac{70.9gCl_2}{1moleCl_2}$$

$$= 0.7027gCl_2/gAnorthite$$

$$\frac{0.7027gCl_2}{gAnorthite} \times 47.6 \text{ MT Anorthite} = 33.4 \text{ MT } Cl_2$$

Total Cl_2 recovered: 70.2 MT Cl_2

Energy Requirement

1. Thermal

Thermal energy to obtain 10 MT O_2 remains the same as in part I.

82,245 kWhrs

Additional thermal energy to obtain Al is determined using stoichiometry and thermodynamics:

$$\frac{36.4\text{KJ}}{\text{moleAl}} \times 3{,}929{,}000\text{gAl} \times \frac{1\text{moleAl}}{27.0\text{gAl}} = 1{,}471\text{kWhrs}$$

Total Thermal Energy = 82,245 + 1,471 = 83,716 kWhrs

2. Electrical

Electrical energy to obtain 10 MT O_2 remains the same as in part I.

43,200 kWhrs

Additional electrical energy to obtain Al is determined using 9kWhr/kgAl which is the electrical energy per kg Al used by Alcoa.

$$\frac{9\text{kWhr}}{\text{KgAl}} \times 3{,}929\ \text{kgAl} = 35{,}361\text{kWhrs}$$

Total Electrical Energy = 43,200 + 35,361 = 78,561kWhrs

Equipment Weights

From part I: 12.45 MT

Additional equipment weight for Al production comes from the Al electrolysis hardware. The estimate is based on cell volumes of 0.5m^3, 4cm wall thickness and material density of 7500Kg/m^3.

975Kg -> 0.975 MT

Total Equipment Weight = 13.4 MT

B.5 Acid Leach

Acid Leach resource and equipment weights have been determined two ways:

I. Production of O_2

II. Production of Al and O_2

III. Production of Al, Mg and O_2

Calculations for the production of Al include the production of 10 MT O_2 or 10 MT O_2/month.

I. Production of O_2

Lunar Regolith Requirement

To obtain only O_2, unbeneficiated mare is leached with HF. Calculations assume all O_2 comes from processing H_2O. The mare required for 10 MT O_2 production is:

$$\frac{1\text{g mare}}{0.424\text{g}O_2} \times 10 \text{ MT } O_2 = 23.6 \text{ MT Mare}$$

Reactant Requirement

HF is the only required reactant for O_2 production only. Using stoichiometry and the composition of mare regolith:

$$\frac{1.09\text{gHF}}{1\text{g Mare}} \times 23.6 \text{ MT Mare} = 27.5 \text{ MT HF}$$

Recovery Potential

After the leach reactions, only H_2O is processed to produce O_2 only. No HF is recovered without some processing of other leach reaction products.

Energy Requirement

1. Thermal

From stoichiometry and the thermodynamics of pre-heating mare regolith to leach process temperature:

0.00267 kWhrs required per g O_2

$$0.00267\frac{\text{kWhrs}}{\text{g } O_2} \times 10 \text{ MT } O_2 = 26{,}663 \text{ kWhrs}$$

2. Electrical

Using the dissociation of H_2O to determine electrical power for H_2O electrolysis:

$$4.32\frac{\text{kWhr}}{\text{Kg}O_2} \times 10 \text{ MT} = 43{,}200 \text{ kWhrs}$$

Equipment Weights

The estimate of all reagent containers required is scaled to process 4.2 MT lunar regolith per hour and is estimated from Criswell's "Extraterrestrial Materials Processing". For O_2 production only, it is estimated that only 10% of all reagent containers are required.

20 MT x 0.10 = 2.0 MT

The H_2O electrolysis hardware is estimated using the NASA Technology Model's equation:

$$\frac{150}{1000} \text{ x } +350 \text{ where X } = \text{man-days of } O_2 \text{ consumption}$$

It is stated that 8 people consume 17.3lbs O_2/day:

$$\frac{17.3\text{lbs}O_2}{\text{day/8men}} = 0.98 \text{ Kg/man-day}$$

Production of 10 MT O_2 is equivalent to:

$$10 \text{ MT } O_2 \times \frac{1\text{man-day}}{0.98\text{Kg}} = 10{,}204 \text{ man-days}$$

Using the H_2 electrolysis weight equation, for 10 MT O_2:

H_2O Electrolysis Weight = 0.85 MT

Total Equipment Weight = 2.0 + 0.85 = 2.85 MT

II. Production of Al and O_2

Lunar Regolith Requirement

Mare regolith is first beneficiated to isolate the aluminum silicate, $Al_2O_3 \cdot SiO_2$. From the stoichiometry of the leach reactions and composition of mare:

$$10 \text{ MT } O_2 \times \frac{1\text{mole}O_2}{32.0\text{g}O_2} \times \frac{1\text{mole}Al_2O_3 \cdot SiO_2}{2.5\text{mole}O_2} \times \frac{162\text{g}Al_2O_3 \cdot SiO_2}{1\text{mole}Al_2O_3 \cdot SiO_2}$$

$$= 20.3 \text{ MT } Al_2O_3 \cdot SiO_2$$

$$20.3 \text{ MT } Al_2O_3 \cdot SiO_2 \times \frac{1\text{gmare}}{0.232\text{g}Al_2O_3 \cdot SiO_2} = 87.3 \text{ MT Mare}$$

Products

From lunar mare composition: 87.3 MT Mare x 0.068 = 5.9 MT Al

Total Products are: 10 MT O_2
5.9 MT Al

Reactant Requirement

The amount of HF required is determined by stoichiometry:

$$10\ \text{MT}\ O_2 \times \frac{1\text{mole}O_2}{32.0\text{g}O_2} \times \frac{11\text{molesHF}}{2.5\text{moles}O_2} \times \frac{20\text{gHF}}{1\text{moleHF}} = 27.5\ \text{MT HF}$$

To determine the amount of NaOH required, stoichiometry is used. From stoichiometry:

69gNa is required per 27gAl

$$5.9\ \text{MT Al} \times \frac{69\text{gNa}}{27\text{gAl}} \times \frac{40\text{gNaOH}}{23\text{gNa}} = 26.2\ \text{MT NaOH}$$

Reactant Recovery Potential

All F ions from AlF_3 will be recycled to form HF:

$$5.6\ \text{MT Al} \times \frac{1\text{moleAl}}{27\text{gAl}} \times \frac{3\text{molesF}}{1\text{moleAl}} \times \frac{19\text{gF}}{1\text{moleF}} = 11.8\ \text{MT F}$$

$$11.8\ \text{MT F} \times \frac{1\text{moleF}}{19\text{gF}} \times \frac{1\text{moleH}}{1\text{moleF}} \times \frac{1\text{gH}}{1\text{moleH}} = 0.6\ \text{MT H}$$

Total HF recovered = 12.4 MT HF

All NaOH is recycled.

Energy Requirements

1. Thermal

From the leach reactions using stoichiometry and thermodynamics for beneficiated mare:

$$\frac{2855\text{KJ}}{\text{mole}Al_2O_3\cdot SiO_2} \times \frac{1\text{mole}Al_2O_3\cdot SiO_2}{162.0\text{g}Al_2O_3\cdot SiO_2} \times 20{,}300{,}000\text{g}\ Al_2O_3\cdot SiO_2$$
$$= 99{,}376\ \text{kWhrs}$$

From AlF_3 reduction using stoichiometry and thermodynamics:

$$\frac{18.6\text{KJ}}{\text{gAl}} \times 5.9\ \text{MT Al} = 30{,}483\ \text{kWhrs}$$

Total Thermal = 99,376 + 30,483 = 129,860 kWhrs

2. Electrical

From H_2O electrolysis using dissociation energy of H_2O:

$$\frac{4.32\,\text{kWhrs}}{\text{Kg } O_2} \times 10 \text{ MT } O_2 = 43{,}200 \text{ kWhrs}$$

From processing of NaOH using dissociation energy of NaOH:

$$\frac{100.64\,\text{Kcal}}{\text{moleNaOH}} \times \frac{0.00116222\,\text{kWhrs}}{\text{Kcal}} \times \frac{1\text{moleNaOH}}{40\text{gNaOH}} \times 26.2 \text{ MT NaOH}$$

$$= 76{,}613 \text{ kWhrs}$$

Total Electrical = 43,200 + 76,613 = 119,813 kWhrs

Equipment Weights

The estimate of all reagent containers required is scaled to produce 1110MT O_2/yr and is estimated from Criswell's "Extraterrestrial Materials Processing". Also, weights of the condensers required are included here. For Al and O_2 production, it is estimated that only 45% of all reagent containers are required.

20 MT x 0.45 = 9.0 MT

The weight of water electrolysis hardware is the same as in part I.

0.85 MT

To determine the weight of the Castner Electrolysis Cell an equivalence factor is used. The factor represents the ratio of atomic weights of NaOH and H_2O. The equivalence factor for Castner electrolysis is 2.22. The equivalence factor is used to derive an equation for Castner Cell weight from the H_2O electrolysis equation in the NASA Technology Model. The weight of the Castner Cell is estimated to be:

1.88 MT

Total Equipment Weight = 9.0 + 1.88 + 0.85 = 11.73 MT

III. Production of Al, Mg and O_2

Lunar Regolith Requirement

Mare regolith is beneficiated to isolate aluminum and magnesium silicates, $Al_2O_3 \cdot SiO_2$ and $MgO \cdot SiO_2$. From the stoichiometry of the leach reactions and composition of mare:

Assume 1g mare:

$0.232g\ Al_2O_3 \cdot SiO_2$

$0.161g\ MgO \cdot SiO_2$

$$0.232gAl_2O_3 \cdot SiO_2 \times \frac{1moleAl_2O_3 \cdot SiO_2}{162.0gAl_2O_3 \cdot SiO_2} \times \frac{2.5molesO_2}{1moleAl_2O_3 \cdot SiO_2} \times \frac{32.0gO_2}{1moleO_2} = 0.115gO_2$$

$$0.161gMgO \cdot SiO_2 \times \frac{1moleMgO \cdot SiO_2}{100.3gMgO \cdot SiO_2} \times \frac{1.5molesO_2}{1moleMgO \cdot SiO_2} \times \frac{32.0gO_2}{1moleO_2} = 0.077gO_2$$

After leaching, $MgO \cdot SiO_2$ appears as $MgF_2 \cdot SiF_4$. Once MgF_2 is removed from $MgF_2 \cdot SiF_4$, it must be converted back to MgO before further reduction. Oxygen from MgO is not recovered.

$$0.161gMgO \cdot SiO_2 \times \frac{1moleMgO \cdot SiO_2}{100.3gMgO \cdot SiO_2} \times \frac{1moleMgO}{1moleMgO \cdot SiO_2} = 0.001605moles\ MgO$$

$$0.001605molesMgO \times \frac{1moleO_2}{2molesMgO} \times \frac{32.0gO_2}{1moleO_2} = 0.0257g\ O_2$$

O_2 obtainable from $MgO \cdot SiO_2$ = 0.077 - 0.02257 = $0.051gO_2$

Total O_2 recovered from 1g mare = 0.115 + 0.051 = 0.166g O_2

For 10 MT O_2 production:

$$10\ MT\ O_2 \times \frac{1gmare}{0.166gO_2} = 60.2\ MT\ Mare$$

<u>Products</u>

From 60.2 MT Mare:

60.2 MT Mare ($0.232gAl_2O_3 \cdot SiO_2$/gmare) = 14.0 MT $Al_2O_3 \cdot SiO_2$

60.2 MT Mare ($0.161gMgO \cdot SiO_2$/gmare) = 9.7 MT $MgO \cdot SiO_2$

$$14.0\ MT\ Al_2O_3 \cdot SiO_2 \times \frac{1mole\ Al_2O_3 \cdot SiO_2}{162.0gAl_2O_3 \cdot SiO_2} \times \frac{2molesAl}{1moleAl_2O_3 \cdot SiO_2} \times \frac{27.0gAl}{1moleAl} = 4.7\ MT\ Al$$

$$9.7\ MT\ MgO \cdot SiO_2 \times \frac{1 mole MgO \cdot SiO_2}{100.3 g MgO \cdot SiO_2} \times \frac{1 mole Mg}{1 mole MgO \cdot SiO_2} \times \frac{24.3 g Mg}{1 mole Mg} = 2.35\ MT\ Mg$$

Total Products are: 10 MT O_2

4.7 MT Al

2.35 MT Mg

<u>Reactant Requirement</u>

From 60.2 MT Mare:

14.0 MT $Al_2O_3 \cdot SiO_2$

9.7 MT $MgO \cdot SiO_2$

$$14.0\ MT\ Al_2O_3 \cdot SiO_2 \times \frac{1 mole Al_2O_3 \cdot SiO_2}{162.0 g Al_2O_3 \cdot SiO_2} \times \frac{11 moles HF}{1 mole Al_2O_3 \cdot SiO_2} \times \frac{20.0 g HF}{1 mole HF} = 19.0\ MT\ HF$$

$$9.7\ MT\ MgO \cdot SiO_2 \times \frac{1 mole MgO \cdot SiO_2}{100.3 g MgO \cdot SiO_2} \times \frac{6 moles HF}{1 mole MgO \cdot SiO_2} \times \frac{20.0 g HF}{1 mole HF} = 11.6\ MT\ HF$$

Total HF required = 19.0 + 11.6 = 30.6 MT HF

Stoichiometry is used to determine the amount of NaOH needed:

69gNa is required per 27gAl

$$4.1\ MT\ Al \times \frac{69 g Na}{27 g Al} \times \frac{40 g NaOH}{23 g Na} = 18.2\ MT\ NaOH$$

Magnesium oxide is reduced by silicon and calcium oxide. Si and CaO requirements are as follows:

$$9.7\ MT\ MgO \cdot SiO_2 \times \frac{1 mole MgO \cdot SiO_2}{100.3 g MgO \cdot SiO_2} \times \frac{1 mole MgO}{1 mole MgO \cdot SiO_2} \times \frac{40.3 g MgO}{1 mole MgO} = 3.9\ MT\ MgO$$

$$3.9\ MT\ MgO \times \frac{1 mole MgO}{40.3 g MgO} \times \frac{1 mole Si}{2 moles MgO} \times \frac{28.0 g Si}{1 mole Si} = 1.35\ MT\ Si$$

$$3.9 \text{ MT MgO} \times \frac{1\text{moleMgO}}{40.3\text{gMgO}} \times \frac{2\text{molesCaO}}{2\text{molesMgO}} \times \frac{56.0\text{gCaO}}{1\text{moleCaO}} = 5.42 \text{ MT CaO}$$

Total Reactant Requirement:

30.6 MT HF

18.2 MT NaOH

5.42 MT CaO

1.35 MT Si

Reactant Recovery Potential

All NaOH is recycled.

Due to processing inefficiencies, it is assumed that 90% of all Si and CaO and recycled.

5.42 MT CaO x (0.90) = 4.88 MT CaO recoverable

1.35 MT Si x (0.90) = 1.22 MT Si recoverable

As in Part II, all F ions from AlF_3 will be recycled to form HF.

HF recoverd from AlF_3 = 12.4 MT HF

All F ions from MgF_2 will be recycled to form HF:

$$2.35 \text{ MT Mg} \times \frac{1\text{moleMg}}{24.3\text{gMg}} \times \frac{1\text{moleMgF}_2}{1\text{moleMg}} \times \frac{2\text{molesHF}}{1\text{moleMgF}_2} \times \frac{20.0\text{gHF}}{1\text{moleHF}} = 3.9 \text{ MT HF}$$

Total HF recoverable = 12.4 + 3.9 = 16.3 MT HF

Energy Requirements

1. Thermal

From the leach reactions using stoichiometry and thermodynamics for beneficiated mare:

$$\frac{2855\text{KJ}}{\text{moleAl}_2\text{O}_3\cdot\text{SiO}_2} \times \frac{1\text{moleAl}_2\text{O}_3\cdot\text{SiO}_2}{162.0\text{gAl}_2\text{O}_3\cdot\text{SiO}_2} \times 14{,}000{,}000\text{gAl}_2\text{O}_3\cdot\text{SiO}_2$$

$$= 68{,}535 \text{ KWhrs}$$

$$\frac{146\text{KJ}}{\text{moleMgO}\cdot\text{SiO}_2} \times \frac{1\text{moleMgO}\cdot\text{SiO}_2}{100.3\text{gMgO}\cdot\text{SiO}_2} \times 9{,}700{,}000\text{gMgO}\cdot\text{SiO}_2 = 3922\text{KHrs}$$

From AlF_3 reduction:

$$\frac{18.6\,KJ}{gAl} \times 4.7\ MT\ Al = 24{,}283\ KWhrs$$

From conversion of MgF_2 to MgO:

at 1200°C -> 2.6 KJ/gMgF_2 converted

$$9.7\ MT\ MgO \cdot SiO_2 \times \frac{1\,mole\,MgO \cdot SiO_2}{100.3\,g\,MgO \cdot SiO_2} \times \frac{1\,mole\,MgF_2}{1\,mole\,MgO \cdot SiO_2} \times \frac{62.3\,g\,MgF_2}{1\,mole\,MgF_2} = 6.025\ MT\ MgF_2$$

$$6{,}025{,}000\,g\,MgF_2 \times \frac{2.6\,KJ}{g\,MgF_2} = 4{,}351\ KWhrs$$

From silicon reduction of MgO:

at 1200°C -> 8.7 KJ/gMg produced

$$2.35\ MT\ Mg \times \frac{8.7\,KJ}{gmg} = 5679\ KWhrs$$

Total Thermal Energy required = 68,535 + 3922 + 24,283 + 4351 + 5679 = 106,770 KWhrs

2. Electrical

From H_2O electrolysis:

$$\frac{4.32\,KWhrs}{Kg\,O_2} \times 10\ MT\ O_2 = 43{,}200\ KWhrs$$

From processing at NaOH:

$$\frac{0.11697\,KWhrs}{mole\,NaOH} \times \frac{1\,mole\,NaOH}{40\,g\,NaOH} \times 18.2\ MT\ NaOH = 53{,}219\ KWhrs$$

Total Electrical = 43,200 + 53,219 = 96,419 KWhrs

<u>Equipment Weights</u>

Equipment weights are determined in the same manner as in Part II. Weights of water electrolysis and Castner Cell subsystems remains same.

Water electrolysis = 0.85 MT

Castner Cell = 1.88 MT

Scaling Criswell's results for HF leach in "Extraterrestrial Materials Processing" and assuming 66% of all hardware will be required for O_2, Al and Mg production:

$$20 \text{ MT} \times 0.66 = 13.2 \text{ MT}$$

Total Equipment Weight = 0.85 + 1.88 + 13.2 = 15.93

B.6 Vapor-Ion Separation

Lunar Regolith Requirement

Theoretically, 100% of any element in the lunar regolith can be extracted using Vapor-Ion techniques. Using composition of the mare regolith, to produce 10 MT O_2:

$$10 \text{ MT } O_2 \times \frac{1\text{g Mare}}{0.413\text{g}O_2} = 24.2 \text{ MT Mare}$$

Reactant Requirement

No consumed reactants are required. If vaporization is achieved using plasma, Argon or another inert gas will be required to maintain the plasma.

Energy Equipment

The following table summarizes energy requirements for the various separation methods used for vapor-ion separation:

Separation Method	Products	Energy Requirement (KWhr/MT Product)
Distillation	Al, Mg, Fe, O_2	34,000
Electrostatic	Al, Mg, Fe	62,000
	Al, Mg, Fe, O_2	44,000
Electromagnetic	Al, Mg, Fe	72,000
	Al, Mg, Fe, O_2	96,000

*Values are from Steur's Extraterrestrial Materials Processing Equipment Weights

Since this process is in the development stage, no accurate calculations of hardware required and hardware weights are available.

B.7 Carbothermal Process

Lunar Regolith Requirement

From 1g mare regolith:

$$0.161gMgO\cdot SiO_2 \times \frac{1moleMgO\cdot SiO_2}{100.3gMgO\cdot SiO_2} \times \frac{2moles\ CO}{1moleMgO\cdot SiO_2} = 0.0032/molesCO$$

$$0.161gMgO\cdot SiO_2 \times \frac{1moleMgO\cdot SiO_2}{100.3gMgO\cdot SiO_2} \times \frac{4molesH_2}{1moleMgO\cdot SiO_2} = 0.00642molesH_2$$

Since the desired molar ratio of H_2 to CO is 3, H_2 is the limiting reagent. Using stoichiometry (see Eqs. 24-27 of Tble F.9) and assuming 100% of all O from H_2O is obtained:

$$0.00642molesH_2 \times \frac{2molesH_2O}{6molesH_2} \times \frac{1moleO_2}{2molesH_2O} \times \frac{32.0gO_2}{1moleO_2} = 0.0342gO_2$$

For 10 MT O_2:

$$10{,}000{,}000gO_2 \times \frac{1g\ mare}{0.0342gO_2} = 292.4\ MT\ Mare$$

$$292.4\ MT\ Mare \times \frac{0.161gMgO\cdot SiO_2}{1gMare} = 47.1\ MT\ MgO\cdot SiO_2$$

<u>Products</u>

$$47.1\ MT\ MgO\cdot SiO_2 \times \frac{1moleMgO\cdot SiO_2}{100.3gMgO\cdot SiO_2} \times \frac{1moleSi}{1moleMgO\cdot SiO_2} \times \frac{28.0gSi}{1moleSi}$$

$$= 13.1\ MT\ Si$$

If 10 MT O_2 are obtained:

$$10\ MT\ O_2 \times \frac{1moleO_2}{32.0gO_2} \times \frac{2molesH_2}{1moleO_2} \times \frac{2gH_2}{1moleH_2} = 1.25\ MT\ H_2$$

$$1.25\ MT\ H_2 \times \frac{1moleH_2}{2.0gH_2} \times \frac{1moleSiH_4}{2molesH_2} \times \frac{32.0gSiH_4}{1moleSiH_4} = 10.0\ MT\ SiH_4$$

$$10.0\ MT\ SiH_4 \times \frac{1moleSiH_4}{32.0gSiH_4} \times \frac{1moleSi}{1moleSiH_4} \times \frac{28.0gSi}{1moleSi} = 8.75\ MT\ Si$$

13.1 MT Si - 8.75 MT Si = 4.35 MT Si

Total Products are:

10 MT O_2
10.0 MT SiH_4
4.35 MT Si

Reactant Requirement

Methane is the only required reactant. All required CO, H_2, and H_2O are obtained as a result of methane reduction of $MgO \cdot SiO_2$. The methane required to obtain 10 MT O_2 is:

$$47.1\text{ MT } MgO\cdot SiO_2 \times \frac{1 mole MgO\cdot SiO_2}{100.3 g MgO\cdot SiO_2} \times \frac{2 moles CH_4}{1 mole MgO\cdot SiO_2} \times \frac{16 g CH_4}{1 mole CH_4} = 15.0\text{ MT } CH_4$$

Reactant Recovery Potential

Assume $1gCH_4$ used to process $MgO \cdot SiO_2$:

$$1g\ CH_4 \times \frac{1 mole CH_4}{16 g CH_4} \times \frac{4 moles H_2}{2 moles CH_4} = 0.125\text{ moles } H_2$$

$$0.125\text{ moles } H_2 \times \frac{2 moles CH_4}{6 moles H_2} \times \frac{16 g CH_4}{1 mole CH_4} = 0.667g\ CH_4$$

$$\text{Recovery potential} = \frac{0.667g}{1.0g} = 66.7\%$$

The amount of methane recovered from manufacture of 10 MT O_2:

15.0 MT CH_4 (0.667) = 10 MT CH_4 recovered

Energy Requirement

1. Thermal

From methane reduction of $MgO \cdot SiO_2$:

$$\frac{742 KJ}{mole MgO\cdot SiO_2} \times 47.1\text{ MT } MgO\cdot SiO_2 \times \frac{1 mole MgO\cdot SiO_2}{100.3 g MgO\cdot SiO_2} = 96{,}738\text{ KWhrs}$$

From reaction of CO with H_2:

$$\#\text{ Moles CO} = \frac{0.00642 moles H_2}{1g\ mare} \times \frac{2 moles CO}{6 moles H_2} \times 292.4\text{ MT Mare} = 625{,}731\text{ moles CO}$$

$$\frac{412.6 KJ}{mole CO} \times 625{,}731\text{ moles CO} = 71{,}715\text{ KWhrs}$$

Total Thermal = 96,738 + 71,715 = 168,453 KWhrs

2. Electrical

Using the dissociation energy of H_2O for electrical requirement for H_2O electrolysis:

$$4.32 \frac{\text{KWhr}}{\text{Kg}O_2} \times 10 \text{ MT } O_2 = 43{,}200 \text{ KWhrs}$$

APPENDIX C

ELES CODE DESCRIPTION

C.1 ELES DESCRIPTION
C.2 ELES CODE OUPUT

APPENDIX C.1

ELES DESCRIPTION

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ELES-1984: A THIRD GENERATION PRELIMINARY DESIGN TOOL

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Abstract

The ELES-1984 computer code is a landmark development in the preliminary systems analysis of liquid rocket vehicles. It is capable of revealing subsystem interactions and design choice impacts on total vehicle performance. Its use enables very rapid determination of optimum vehicle designs.

Overview

The liquid propulsion system models in ELES have been developed by Aerojet TechSystems Company under the auspices of AFRPL during the past few years (1980-1984) under contracts F04611-79-C-0054 and F04611-82-C-0062. The main purpose of ELES is to find optimum vehicle designs for specified mission requirements. Toward that end it is capable of evaluating the size, weight, and performance of system components over a range of design configurations, materials of construction, and operating points.

There are three main sections of the ELES computer code (see Fig. 1): a stage design section, a trajectory model, and a multivariable optimizer. The stage design section calculates the size, weight and engine performance of liquid or solid stages (see Fig. 2). The trajectory model uses a 2D round non-rotating earth, 1962 standard atmospheric data, Adams-Moulton/Runge-Kutta integration, and Kepler orbital mechanics. The optimizer provides optima for both stage design and vehicle guidance with design and guidance parameter sensitivites included. Mixed solid and liquid stage vehicles of up to 4 stages can be modeled by ELES.

The liquid engine feed system power cycles modeled by ELES are illustrated in Fig. 3. The list includes pressure fed engines and pump fed engines with the following turbopump power cycles: gas generator bleed, single preburner staged combustion, staged reaction, and

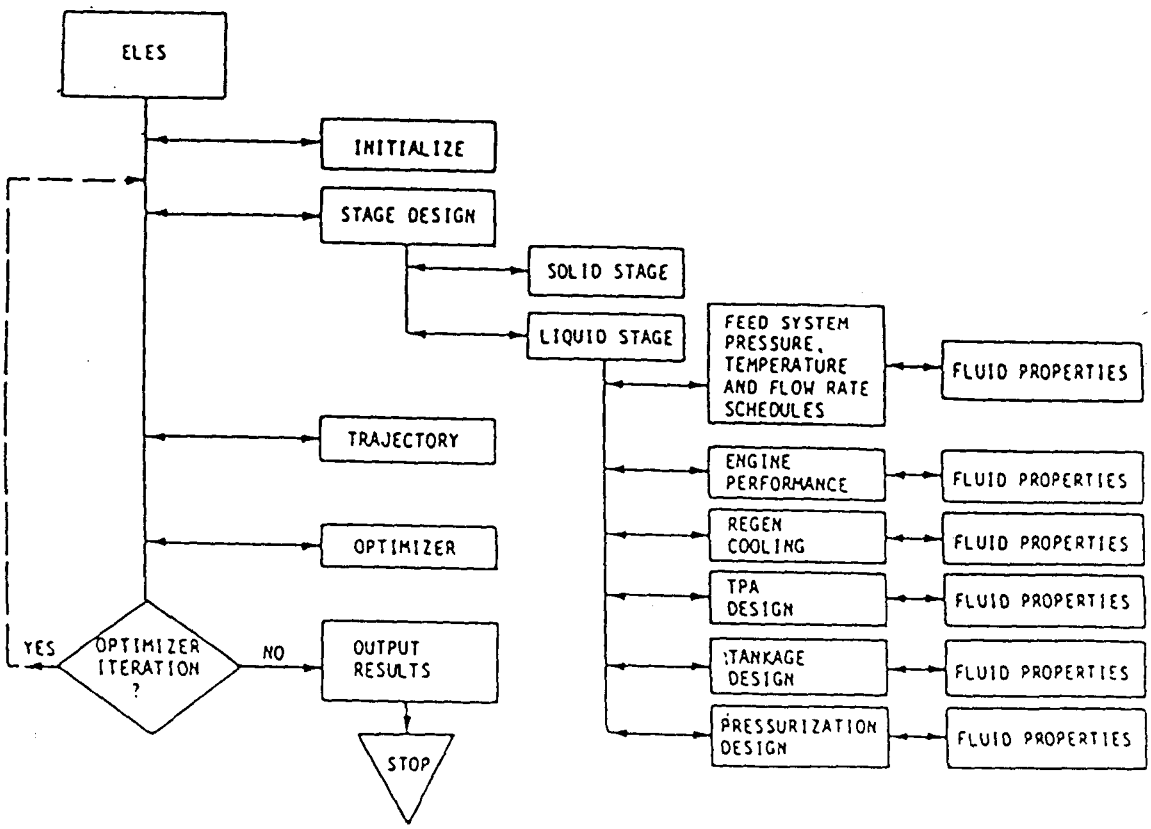

Fig. 1 ELES flow diagram

Propellant Tank Size/Weight

Pressurization Tank Size/Weight

Line Size/Weight

Positive Expulsion Size/Weight/Delta P

Engine Size/Weight/Performance (Nozzle, Valve, Injector, Chamber)

Thrust Mount Size/Weight

Gimbal System Size/Weight

Tank Residuals Weight

Tank Pressurization Requirements

Interstage Size/Weight

Delivered Specific Impulse (ideal one dimensional equilibrium performance degraded by kinetic, vaporization, boundary layer, mixing, two phase, divergence, and MR distribution losses)

Feed System Temperature/Pressure/Flowrate Schedules

Regenerative/Trans-Regen Cooling Requirements

Turbopump Assembly Size/Weight/Performance

Turbopump Design Parameter Breakdown

Regenerative Cooling Jacket Summary

Required Engine Barrier Mixture Ratio

Stage Tank Mixture Ratio

Fig. 2 Major Output Parameters of Liquid Stage Design Section

expander. The ELES engine analysis outputs engine size, weight, and performance, as well as turbopump assembly (TPA) size, weight and performance.

Engine performance is based on the standard JANNAF method. It begins with ideal one dimensional equilibrium (ODE) performance and degrades that ideal performance with loss multipliers. The calculation of these multipliers is performed by standard JANNAF procedures or by Aerojet derived methods. The analysis includes the effect of injector design, thrust chamber material, operating temperatures, propellant inlet temperatures, and thrust chamber geometry.

TPA design options are shown in Fig. 3 as gearbox, single shaft, and twin TPA. As required, the code will stage the pumps and turbines. The TPA is designed by considering system power requirements and drive fluid characteristics. Pump and turbine efficiencies are based on industry standards (Ref. NASA SP-8109, Fig. 6; AFRPL TR 72-45, Fig. 4).

The temperature and pressure drops across regenerative or trans-regenerative cooling jackets are calculated by creating a simplified thrust chamber geometry with slotted channels for coolant flow. Combustion gas and coolant heat transfer coefficients are calculated at discrete points along the chamber and are used to integrate the pressure drop necessary to maintain the chamber wall at nominal operating temperature. Transpiration cooled portions of the chamber are analyzed using techniques developed by Aerojet TechSystems for use with transpiration cooled re-entry vehicle nosetips.

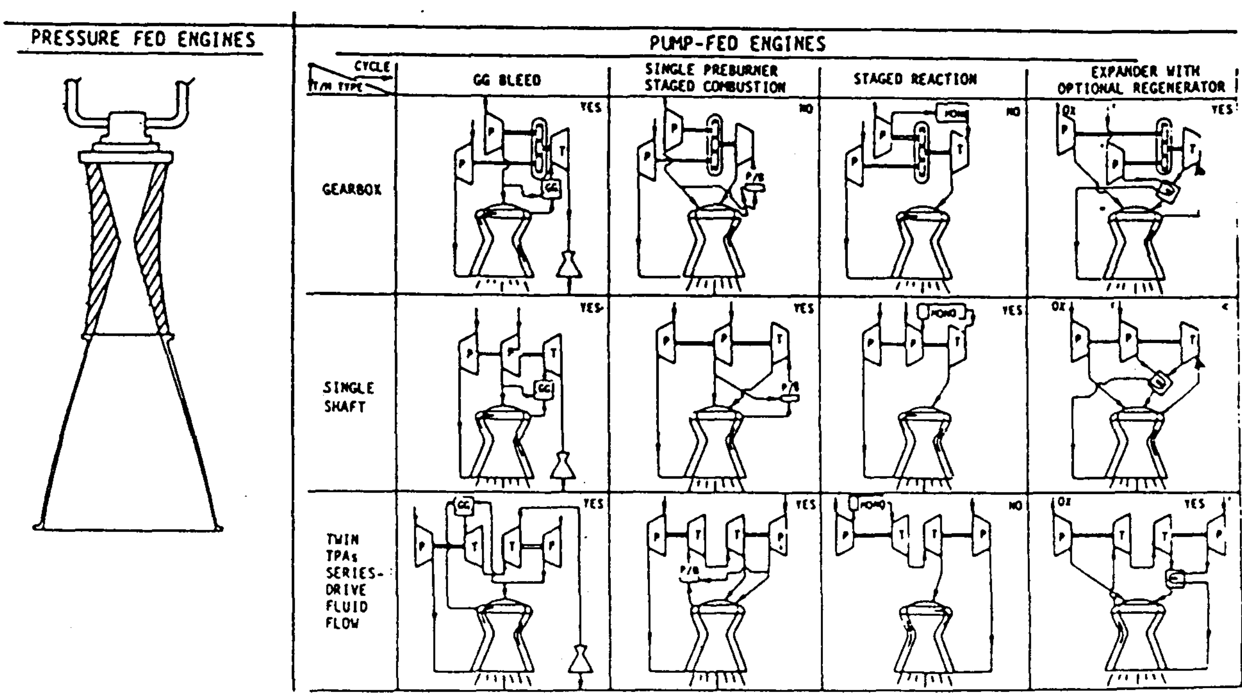

Fig. 3 Representative ELES engine cycles

A wide variety of tankage designs are available (see Fig. 4). Tandem tanks are designed by choosing tank head orientation, common or separate tank heads, suspended or monocoque construction, and pressurant tank location. The tanks may or may not contain a positive expulsion bladder or surface tension acquisition device. Non-conventional tankage is designed by choosing the number and type of propellant and pressurization tanks as well as propellant acquisition design. Each tank is individually specified to be toroidal, spherical, or cylindrical with elliptical heads. Tanks are located based on general location input and physical interference between the tanks and envelope.

Propellant tank pressurization options in ELES include cold gas, solid gas generator, and autogenous. With cryogenic propellants, the pressurant collapse is calculated with the Epstein* correlation. Pressurization requirements are affected by the vehicle operating temperature regime, and external heating loads.

Throughout the liquid stage design portion of the code there is a need for propellant properties data over an extremely wide range of temperature and pressure. This data is stored in tables for hydrogen and helium. The properties for all other propellants are calculated by the method of corresponding states. This allows analysis to occur in regimes where propellant data may not exist and for propellants which have very little experimental data.

Liquid Stage Design Procedure

The general procedure used for calculating the size/weight/performance of liquid stages is described in Fig. 5. It begins with the initialization of propellant feed circuit parameters (temperature, pressure, flowrate). The remainder of the procedure refines those initial estimates.

Refinements to the feed schedules include calculating the engine's barrier mixture ratio, engine performance, regenerative cooling jacket properties, turbopump assembly (TPA) design, propellant tank pressurization requirements, and tankage heat transfer. Iterative procedures are used for some of the parameters.

When the propellant feed schedules are finalized, the calculations of size, weight, and performance of the TPA, engine, and tankage can take place. A stage summary of those parameters and related parameters can then be made.

*Epstein, M. and Anderson, R, "An Equation for the Prediction of Cryogenic Pressurant Requirements for Axisymmetric Propellant Tanks," Advances in Cryogenic Engineering, Volume 13, New York (1968), Page 207.

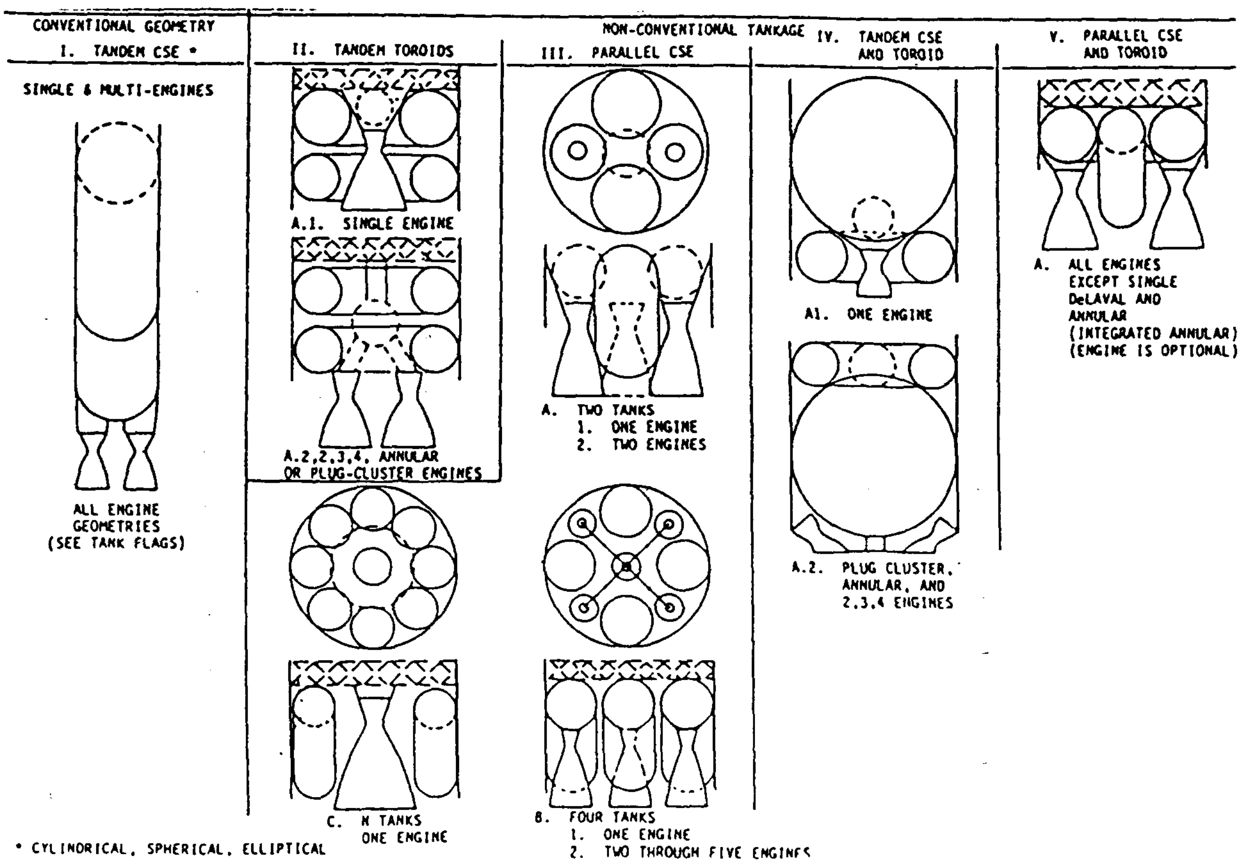

Fig. 4 Representative ELES tankage options.

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1) Initialize temperature schedule
2) Calculate engine barrier mixture ratio
3) Initialize flowrate schedule (using some rough estimates)
   - 3.1) Estimate tank sizes
     - 3.1.1) Estimate tank heat transfer
     - 3.1.2) Estimate pressurization requirements
4) Calculate feed system pressure schedule
   - 4.1) Calculate engine performance
   - 4.2) Perform regen cooling analysis (if required)
5) Perform non-conventional nozzle modifications
6) Calculate flowrate schedule (using improved estimates)
   - 6.1) Calculate tank sizes
     - 6.1.1) Calculate tank heat transfer
     - 6.1.2) Calculate pressurization requirements
7) Design TPA (if required) (iterate, if not power balanced)
8) Update propellant temperature schedule (iterate on temperature schedule, if required)
9) Calculate TPA size/weight (if required)
10) Calculate engine size/weight
11) Calculate tankage size/weight
12) Calculate stage summary size/weight/performance

Fig. 5 Liquid Stage Design Procedure

ELES Input

ELES-1984 operates in a "batch" type mode which means that during program execution there is no interaction between the user and the code. After normal program termination ELES will have created output files which can be examined by the user.

The main form of interaction between the user and ELES takes place prior to program execution when the user creates an input file. This input file is submitted to ELES at run time. The input file (named "ELESINP") contains up to 34 NAMELIST blocks which contain the input variables. Although all 34 blocks are not always read by ELES, it is recommended that all namelist blocks be included in ELESINP in their proper order. This precaution can prevent a whole class of termination errors.

Using the liquid stage models in ELES to their fullest potential involves the use of hundreds of inputs. In order to organize the input procedures for those variables, an input worksheet has been developed. The first portion of that worksheet is presented in the ELES New Users Guide, pages 29 through 41. The remainder is presented in the ELES Advanced Users Guide, Pages 4 through 52.

The new users worksheet is concerned with a general overview of basic ELES options; that worksheet is the best place to begin. There are two major types of input in the advanced users worksheet; 1) recurrent input which must always be considered and 2) contingent input which need only be considered if prior choices dictate.

The recurrent input includes general inputs, injector related inputs, thrust chamber inputs, and tankage inputs. These should be considered every time ELES is run.

The contingent input worksheet relates to tandem tanks, non-conventional tanks, cold gas pressurization, solid gas generator pressurization, turbopump assemblies, regen/trans-regen cooling, tankage heat transfer, positive expulsion bladders, user defined propellants, throttling trajectories, and short nozzle designs. Each category need only be considered if it is a part of the design in question.

It is highly recommended that the user photocopy all applicable worksheets and fill them out prior to program execution.

ELES Output

The output from ELES consists of detailed stage summary pages as well as an overall vehicle summary. For each liquid stage, there is an output page for warning messages, tankage summary, stage graphical schematic, engine summary, propellant summary, regenerative cooling jacket summary, turbopump assembly (TPA) summary, feed system temperature and pressure schedules, and an overall stage weight breakdown.

The purpose of the warning page is to alert the user to potential design flaws or program problems. Examples of warning messages include injector orifices diameters below a typical minimum, tank wall thicknesses design criteria (buckling, minimum gauge, hoop stress, etc.), or unusual termination of an iteration loop. It is the users responsibility to ignore or respond to warning messages.

The tankage summary gives a tank-by-tank description of the stage. Output includes tank contents, pressurization method, thicknesses, dimensions, materials of construction, safety factors, residual propellant weights, pressurant weight, line weights, propellant acquisition system weight, and tank weights.

The stage graphical schematic is drawn to scale on the line printer with actual tank head ellipse ratios. The size of the schematic is automatically adjusted to fill the page. Because some line printers do not use the standard number of characters per inch in the horizontal and vertical dimensions, that information may be input by the user. All graphics are performed by pseudo-Tektronix routines in ELES which mimic standard Tektronix commands. It is therefore relatively easy to convert ELES to create high resolution Tektronix schematics.

The engine summary begins with basic engine design information (power cycle, cooling method, propellant combination) and then proceeds to more

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detailed engine descriptions. The left side of the engine summary page is devoted to size and weight information. The right side is devoted to performance-related engine parameters including a breakdown of individual loss mechanisms to engine performance. References to "core" and "barrier" are due to the core and barrier stream tube model used in the performance calculations.

The propellant summary page applies over the operating temperature range of the on-board propellants. For storable propellants this corresponds to the operating temperature range of the vehicle. The first line of the propellant summary declares whether the propellant combination is a user defined propellant combination or a library propellant combination. ELES allows for easy simulation of non-library propellants using propellant property inputs. Using the method of corresponding states, ELES predicts propellant properties over a very wide range of temperature and pressure. These calculations are used to design tanks, pumps, regenerative cooling jackets, etc.

The propellant properties displayed are primarily tank design parameters. The density of each propellant at its maximum temperature is used to calculate the tank volume requirements. The vapor pressure is used in determining tank pressure requirements.

The regenerative cooling summary describes the heat transfer characteristics of the combustion chamber at various points along the gas side wall. The heat transfer coefficient and heat flux is indicated at each point as well as liquid coolant bulk temperature and pressure. Simplified one dimensional heat transfer and fluid hydraulics are used to estimate the overall temperature rise and pressure drop across the regen jacket.

The TPA summary gives detailed descriptions of the pumps and turbines in the power cycle. Speeds, dimensions, efficiencies, flowrates, number of stages, weights, horsepowers, and admission fractions are included for pumps, boost pumps, and turbines.

The pressure and temperature schedules show the pressure and temperature at various key points in the propellant feed system as well as pressure and temperature changes across key sections of the feed system. A flowrate schedule is also included which shows flowrates through the major components of the feed system.

The overall stage weight summary is a list of all items in the stage which contribute to its weight. Inert weights are presented separately from propellant or pressurant weights.

The final page of output is the vehicle summary which gives an overview of all vehicle stages. The stage masses, mass fractions, dimensions, and performances are overviewed.

Concluding Remarks

Since its initial configuration in 1980, ELES has been of great benefit to its creaters in analyzing propulsion system concepts in a timely, cost-effective manner. As its use spreads it is establishing itself as a standard in the field of preliminary propulsion system design. To the authors knowledge there is no comparable method by which propulsion system design parameters can be optimized with nearly the speed or accuracy which ELES offers.

APPENDIX C.2

ELES CODE OUPUT

OTV BASELINE WITH AEROBRAKE

H/O Engines
Isp = 470
O/F = 5.5
15.9 MT Payload
Delta V = 5.36 Km/Sec
15% Aerobrake
25% Added to Tanks and Engines

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

STAGE #1

..WEIGHT,LB..

PAYLOAD	35000.00
STAGE WEIGHT	110596.38
USABLE PROPELLANT	100100.00
FIXED INERT	
PROPULSION SYSTEM	9909.16
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.38
JETTISONED	0.00
GROSS IGNITION WEIGHT	145596.38
GROSS BURNOUT WEIGHT	45454.00
PROPELLANT MASS FRACTION	0.905

..DIMENSIONS,IN..

STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	26.86
NUMBER OF NOZZLES	2
STAGE LENGTH	626.17
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	626.17

..PERFORMANCE..

PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	15000.0
PC,PSIA	2000.0
USABLE PROPELLANT MR	5.49
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	3122.65
ISP,VACUUM DELIVERED,SEC	469.6
ISP EFFICIENCY	0.972
PROPELLANT FLOW RATE,LB/SEC	31.94

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ENGINE SIZE,WEIGHT,& PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.089
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	23.01
CHAMBER	20.85
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	4.24
TCA CONSTRUCTION	15.04

SINGLE THRUST CHAMBER ASSY	69.40
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	3.81
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	135.67
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	7.63

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	469.61
IDEAL ISP(ODE),SEC	482.94
DELIVERED CSTAR,FT/SEC	7608.
IDEAL CSTAR,FT/SEC	7647.
CHAMBER PRESSURE,PSIA	2000.
THRUST PER ENGINE(VAC),LBF	7500.
TOTAL VAC THRUST,LBF	15000.
BURN TIME,SEC	3122.65
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	27.02
FUEL FLOWRATE,LB/SEC	4.92
TOTAL FLOWRATE,LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE,DEG R	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE,DEG R	3792.
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.059
FUEL ORIFICE DIAMETER (IN)	0.078

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BASELINE

15.9 MT Payload
H/O Engines
Isp = 469
Delta V = 20567 Ft/Sec

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EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...	
THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.089
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAD LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00
... ENGINE WEIGHTS (POUNDS) ...	
NOZZLE EXTENSION	23.01
CHAMBER	20.85
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	3.42
TCA CONSTRUCTION	2.51
SINGLE THRUST CHAMBER ASSY	56.05
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	4.11
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	112.10
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	8.22

... PERFORMANCE ...	
DELIVERED ISP(VAC), SEC	$\boxed{469.61}$
IDEAL ISP(ODE), SEC	482.94
DELIVERED CSTAR, FT/SEC	7608.
IDEAL CSTAR, FT/SEC	7647.
CHAMBER PRESSURE, PSIA	2000.
THRUST PER ENGINE(VAC), LBF	7500.
TOTAL VAC THRUST, LBF	15000.
BURN TIME, SEC	3587.93
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	27.02
FUEL FLOWRATE, LB/SEC	4.92
TOTAL FLOWRATE, LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE, DEG R	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE, DEG R	3792.
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY, ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.059
FUEL ORIFICE DIAMETER (IN)	0.078

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT LB..	
PAYLOAD	35000.00
STAGE WEIGHT	119620.99
USABLE PROPELLANT	115000.00
FIXED INERT	
PROPULSION SYSTEM	3984.77
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.37
JETTISONED	0.00
GROSS IGNITION WEIGHT	154620.99
GROSS BURNOUT WEIGHT	39578.62
PROPELLANT MASS FRACTION	0.961
..DIMENSIONS, IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	26.86
NUMBER OF NOZZLES	2
STAGE LENGTH	696.60
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	696.60
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST, VACUUM DELIVERED, LBF	15000.0
PC, PSIA	2000.0
USABLE PROPELLANT MR	5.49
NOZZLE AREA RATIO	300.00
BURN TIME, SEC	3587.93
ISP, VACUUM DELIVERED, SEC	469.6
ISP EFFICIENCY	0.972
PROPELLANT FLOW RATE, LB/SEC	31.94

SINGLE STAGE TO MOON

H/O Engines: High Thrust (15000 lb/engine)
Isp = 471
15.9 MT Payload
Delta V = 20564 Ft/Sec
Lightweight Engines and Tanks

EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.069
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	23.01
CHAMBER	20.64
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	4.23
TCA CONSTRUCTION	15.43

SINGLE THRUST CHAMBER ASSY	69.36
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	6.15
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	138.77
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	12.30

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	469.61
IDEAL ISP(ODE),SEC	482.94
DELIVERED CSTAR,FT/SEC	7608.
IDEAL CSTAR,FT/SEC	7647.
CHAMBER PRESSURE,PSIA	2000.
THRUST PER ENGINE(VAC),LBF	7500.
TOTAL VAC THRUST,LBF	15000.
BURN TIME,SEC	6771.52
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	27.02
FUEL FLOWRATE,LB/SEC	4.92
TOTAL FLOWRATE,LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE,DEG R 4580	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE,DEG R	3792.
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.059
FUEL ORIFICE DIAMETER (IN)	0.078

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MOON BASE PROPULSION  6/9/86

.*** VEHICLE SUMMARY ****

                                   STAGE #1

   ..WEIGHT.LB..
                                   35000.00
PAYLOAD                           224373.89
STAGE WEIGHT                      217000.00
  USABLE PROPELLANT
  FIXED INERT
    PROPULSION SYSTEM               6358.49
    INTERSTAGE                         0.00
  EXPENDED INERT
    EXPELLED                          42.23
    JETTISONED                         0.00
GROSS IGNITION WEIGHT             259373.89
GROSS BURNOUT WEIGHT               42331.66
PROPELLANT MASS FRACTION              0.967

   ..DIMENSIONS.IN..

STAGE DIAMETER                       150.00
NOZZLE EXIT DIAMETER                  26.86
NUMBER OF NOZZLES                     2
STAGE LENGTH                        1178.67
PAYLOAD LENGTH                         0.00
TOTAL VEH LENGTH                    1178.67

   ..PERFORMANCE..

PROPELLANT                          LO2/LH2
THRUST,VACUUM DELIVERED,LBF         15000.0
PC,PSIA                              2000.0
USABLE PROPELLANT MR                   5.49
NOZZLE AREA RATIO                    300.00
BURN TIME,SEC                       6771.52
ISP,VACUUM DELIVERED,SEC              469.6
ISP EFFICIENCY                        0.972
PROPELLANT FLOW RATE,LB/SEC           31.94
```

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OTV BASELINE NO AEROBRAKE

H/O Engines
Isp = 470
O/F = 5.5
15.9 MT Payload
Delta V = 8.34 Km/Sec
25% Increase on Engines and Tanks
No Aerobrake

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ENGINE SIZE,WEIGHT,& PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	2.19
CHAMBER DIAMETER	4.38
NOZZLE EXIT DIAMETER	37.93
NOZZLE EXTENSION ATTACH DIAM	9.79
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.140
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.029
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	67.69
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	49.99
CHAMBER	49.93
BIPROPELLANT VALVE	3.43
INJECTOR	11.21
TCA SUPPORT HARDWARE	7.82
TCA CONSTRUCTION	5.73

SINGLE THRUST CHAMBER ASSY	128.11
THRUST MOUNT	33.41
GIMBAL SYSTEM	31.49
ENGINE BAY LINES	8.07
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	256.23
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	62.98
TOTAL ENGINE BAY LINES	16.15

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	$\boxed{471.08}$
IDEAL ISP(ODE),SEC	482.94
DELIVERED CSTAR,FT/SEC	7611.
IDEAL CSTAR,FT/SEC	7647.
CHAMBER PRESSURE,PSIA	2000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	1800.16
OVERALL EFFICIENCY	0.975
ENERGY RELEASE EFFICIENCY	0.991
NOZZLE EFFICIENCY	0.985
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.994
BOUNDARY LAYER EFFICIENCY	0.993
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	54.07
FUEL FLOWRATE,LB/SEC	9.61
TOTAL FLOWRATE,LB/SEC	63.68
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE,DEG R	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE,DEG R	3792.
ENGINE MIXTURE RATIO	5.63
FUEL FILM COOLING FRACTION	0.06
INJ ELEMENT DENSITY,ELEM/IN**2	10.29
OX ORIFICE DIAMETER (IN)	0.058
FUEL ORIFICE DIAMETER (IN)	0.066

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	119856.86
USABLE PROPELLANT	115000.00
FIXED INERT	
PROPULSION SYSTEM	4259.00
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.37
JETTISONED	0.00
GROSS IGNITION WEIGHT	154856.86
GROSS BURNOUT WEIGHT	39814.49
PROPELLANT MASS FRACTION	0.959
..DIMENSIONS,IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	37.93
NUMBER OF NOZZLES	2
STAGE LENGTH	710.31
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	710.31
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	2000.0
USABLE PROPELLANT MR	5.63
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	1800.16
ISP,VACUUM DELIVERED,SEC	471.1
ISP EFFICIENCY	0.975
PROPELLANT FLOW RATE,LB/SEC	63.68

ORIGINAL PAGE IS OF POOR QUALITY

LANDER (NO LANDING GEAR) O/F 8.73

H/O Engines
Isp = 421
O/F = 8.73
15.9 MT Payload
Delta V = 4.17 Km/s
No Landing Gear
25% Added to Tanks and Engines

ENGINE SIZE,WEIGHT,& PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...	
THROAT DIAMETER	1.77
CHAMBER DIAMETER	3.54
NOZZLE EXIT DIAMETER	30.61
NOZZLE EXTENSION ATTACH DIAM	7.90
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.071
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.026
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	54.64
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...	
NOZZLE EXTENSION	28.77
CHAMBER	19.38
BIPROPELLANT VALVE	2.04
INJECTOR	5.40
TCA SUPPORT HARDWARE	4.70
TCA CONSTRUCTION	16.67

SINGLE THRUST CHAMBER ASSY	76.95
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	2.52
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	153.89
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	5.03

... PERFORMANCE ...	
DELIVERED ISP(VAC),SEC	421.18
IDEAL ISP(ODE),SEC	434.07
DELIVERED CSTAR,FT/SEC	6650.
IDEAL CSTAR,FT/SEC	6622.
CHAMBER PRESSURE,PSIA	1500.
THRUST PER ENGINE(VAC),LBF	7500.
TOTAL VAC THRUST,LBF	15000.
BURN TIME,SEC	2155.30
OVERALL EFFICIENCY	0.970
ENERGY RELEASE EFFICIENCY	0.996
NOZZLE EFFICIENCY	0.974
KINETIC EFFICIENCY	0.990
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.999
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	31.96
FUEL FLOWRATE,LB/SEC	3.66
TOTAL FLOWRATE,LB/SEC	35.61
CORE MIXTURE RATIO	10.00
CORE TEMPERATURE,DEG R	6486.
BARRIER MIXTURE RATIO	2.42
BARRIER TEMPERATURE,DEG R	3773.
ENGINE MIXTURE RATIO	8.73
FUEL FILM COOLING FRACTION	0.13
INJ ELEMENT DENSITY,ELEM/IN**2	10.09
OX ORIFICE DIAMETER (IN)	0.060
FUEL ORIFICE DIAMETER (IN)	0.075

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	86330.32
USABLE PROPELLANT	77000.00
FIXED INERT	
PROPULSION SYSTEM	8877.96
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.42
JETTISONED	0.00
GROSS IGNITION WEIGHT	121330.32
GROSS BURNOUT WEIGHT	44287.90
PROPELLANT MASS FRACTION	0.892
..DIMENSIONS,IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	30.61
NUMBER OF NOZZLES	2
STAGE LENGTH	434.12
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	434.12
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	15000.0
PC,PSIA	1500.0
USABLE PROPELLANT MR	8.73
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	2155.30
ISP,VACUUM DELIVERED,SEC	421.2
ISP EFFICIENCY	0.973
PROPELLANT FLOW RATE,LB/SEC	35.61

LANDER (NO LANDING GEAR) O/F = 10.6

H/O Engines
Isp = 384
O/F = 10.6
15.9 MT Payload
Delta V = 4.17 Km/Sec
No Landing Gear
25% Added to Tanks and Engines

ENGINE SIZE,WEIGHT,& PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	2.15
CHAMBER DIAMETER	4.37
NOZZLE EXIT DIAMETER	37.82
NOZZLE EXTENSION ATTACH DIAM	9.77
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.068
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.025
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	67.50
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	41.84
CHAMBER	24.94
BIPROPELLANT VALVE	2.58
INJECTOR	7.45
TCA SUPPORT HARDWARE	6.49
TCA CONSTRUCTION	23.05

SINGLE THRUST CHAMBER ASSY	106.35
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	2.43
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	[illegible]
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	212.71
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	4.85

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	384.58
IDEAL ISP(ODE),SEC	396.90
DELIVERED CSTAR,FT/SEC	6179.
IDEAL CSTAR,FT/SEC	6140.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	7500.
TOTAL VAC THRUST,LBF	15000.
BURN TIME,SEC	2301.29
OVERALL EFFICIENCY	0.969
ENERGY RELEASE EFFICIENCY	0.997
NOZZLE EFFICIENCY	0.972
KINETIC EFFICIENCY	0.989
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	1.000
BOUNDARY LAYER EFFICIENCY	0.988
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	35.64
FUEL FLOWRATE,LB/SEC	3.36
TOTAL FLOWRATE,LB/SEC	39.00
CORE MIXTURE RATIO	12.00
CORE TEMPERATURE,DEG R	6190.
BARRIER MIXTURE RATIO	2.48
BARRIER TEMPERATURE,DEG R	3848.
ENGINE MIXTURE RATIO	10.60
FUEL FILM COOLING FRACTION	0.12
INJ ELEMENT DENSITY,ELEM/IN**2	10.15
OX ORIFICE DIAMETER (IN)	0.057
FUEL ORIFICE DIAMETER (IN)	0.072

ORIGINAL PAGE IS
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MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	99487.17
USABLE PROPELLANT	90000.00
FIXED INERT	
PROPULSION SYSTEM	9024.38
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.42
JETTISONED	0.00
GROSS IGNITION WEIGHT	134487.17
GROSS BURNOUT WEIGHT	44444.75
PROPELLANT MASS FRACTION	0.905
..DIMENSIONS,IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	37.82
NUMBER OF NOZZLES	2
STAGE LENGTH	459.67
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	459.67
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	15000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	10.60
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	2301.29
ISP,VACUUM DELIVERED,SEC	384.6
ISP EFFICIENCY	0.969
PROPELLANT FLOW RATE,LB/SEC	39.00

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LANDER (NO LANDING GEAR) CARRYING FUEL FOR OTV RETURN TRIP

H/O Engines
Isp = 470
O/F = 5.5
15.9 MT Payload
Delta V = 4.17 Km/Sec
No Landing Gear
Carries LH_2 for OTV Return Trip
25% Added to Engines and Tanks

ENGINE SIZE, WEIGHT, & PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.089
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	23.01
CHAMBER	20.92
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	4.24
TCA CONSTRUCTION	15.06

SINGLE THRUST CHAMBER ASSY	69.49
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	3.20
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	[illegible]
CLUSTER AREA RATIO	[illegible]
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	136.99
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	6.39

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	469.61
IDEAL ISP(ODE),SEC	482.94
DELIVERED CSTAR,FT/SEC	7608.
IDEAL CSTAR,FT/SEC	7647.
CHAMBER PRESSURE,PSIA	2000.
THRUST PER ENGINE(VAC),LBF	7500.
TOTAL VAC THRUST,LBF	15000.
BURN TIME,SEC	2150.74
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	27.02
FUEL FLOWRATE,LB/SEC	4.92
TOTAL FLOWRATE,LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE,DEG R	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE,DEG R	3792.
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.359
FUEL ORIFICE DIAMETER (IN)	0.078

ORIGINAL PAGE IS
OF POOR QUALITY

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	80872.60
USABLE PROPELLANT	69000.00
FIXED INERT	
PROPULSION SYSTEM	9393.72
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	42.39
JETTISONED	0.00
GROSS IGNITION WEIGHT	115872.60
GROSS BURNOUT WEIGHT	46830.21
PROPELLANT MASS FRACTION	0.853
..DIMENSIONS,IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	26.66
NUMBER OF NOZZLES	2
STAGE LENGTH	526.10
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	526.10
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	15000.0
PC,PSIA	2000.0
USABLE PROPELLANT MR	5.49
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	2150.74
ISP,VACUUM DELIVERED,SEC	469.6
ISP EFFICIENCY	0.972
PROPELLANT FLOW RATE,LB/SEC	31.94

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AL/LOX LANDER WITH LANDING GEAR

AL/LOX Engines
Isp = 260
O/F = 2.18
15.9 MT Payload
Delta V = 4.2 Km/Sec
5% Landing Gear
25% Added to Tanks and Engines

ENGINE SIZE+WEIGHT+& PERFORMANCE SUMMARY FOR STAGE #1
GAS GENERATOR BLEED CYCLE (USER DEFINED GG)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

... ENGINE DIMENSIONS (INCHES) ...	
THROAT DIAMETER	3.12
CHAMBER DIAMETER	5.40
NOZZLE EXIT DIAMETER	31.17
NOZZLE EXTENSION ATTACH DIAM	13.94
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.087
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	45.32
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00
... ENGINE WEIGHTS (POUNDS) ...	
NOZZLE EXTENSION	24.34
CHAMBER	45.32
BIPROPELLANT VALVE	4.98
INJECTOR	13.79
TCA SUPPORT HARDWARE	7.47
TCA CONSTRUCTION	26.52

SINGLE THRUST CHAMBER ASSY	122.40
THRUST MOUNT	33.41
GIMBAL SYSTEM	25.58
ENGINE BAY LINES	5.19
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	244.79
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	51.17
TOTAL ENGINE BAY LINES	10.39

... PERFORMANCE ...	
DELIVERED ISP(VAC),SEC	260.58
IDEAL ISP(ODE),SEC	282.96
DELIVERED CSTAR,FT/SEC	4288.
IDEAL CSTAR,FT/SEC	4343.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	1562.43
OVERALL EFFICIENCY	0.921
ENERGY RELEASE EFFICIENCY	0.987
NOZZLE EFFICIENCY	0.938
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.992
MR DISTRIBUTION EFFICIENCY	0.995
BOUNDARY LAYER EFFICIENCY	0.992
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.960
GG BLEED EFFICIENCY	0.995
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	78.56
FUEL FLOWRATE,LB/SEC	35.99
TOTAL FLOWRATE,LB/SEC	114.55
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE,DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE,DEG R	5518.
ENGINE MIXTURE RATIO	2.18
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	9.87
OX ORIFICE DIAMETER (IN)	0.121
FUEL ORIFICE DIAMETER (IN)	0.034

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	188432.10
USABLE PROPELLANT	181000.00
FIXED INERT	
PROPULSION SYSTEM	7054.79
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	223432.10
GROSS BURNOUT WEIGHT	41418.42
PROPELLANT MASS FRACTION	0.961
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	31.17
NUMBER OF NOZZLES	2
STAGE LENGTH	427.29
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	427.29
..PERFORMANCE..	
PROPELLANT	LIQUID
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	2.18
NOZZLE AREA RATIO	100.00
BURN TIME,SEC	1562.43
ISP,VACUUM DELIVERED,SEC	26[illegible].6
ISP EFFICIENCY	0.921
PROPELLANT FLOW RATE,LB/SEC	114.55

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AP/LOX LANDER WITH LANDING GEAR 1/2 PAYLOAD

AL/LOX Engines
Isp = 260
O/F = 2.18
7.95 MT Payload
Delta V = 4.2 Km/Sec
5% Landing Gear
25% Added to Tanks and Engines

GAS GENERATOR BLEED CYCLE (USER DEFINED GG)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	3.12
CHAMBER DIAMETER	5.40
NOZZLE EXIT DIAMETER	31.17
NOZZLE EXTENSION ATTACH DIAM	13.94
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.087
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	45.32
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	24.34
CHAMBER	45.30
BIPROPELLANT VALVE	4.98
INJECTOR	13.79
TCA SUPPORT HARDWARE	7.47
TCA CONSTRUCTION	26.52

SINGLE THRUST CHAMBER ASSY	122.40
THRUST MOUNT	33.41
GIMBAL SYSTEM	25.58
ENGINE BAY LINES	4.11
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	244.79
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	51.17
TOTAL ENGINE BAY LINES	8.22

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	260.58
IDEAL ISP(ODE),SEC	282.96
DELIVERED CSTAR,FT/SEC	4288.
IDEAL CSTAR,FT/SEC	4345.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	837.32
OVERALL EFFICIENCY	0.921
ENERGY RELEASE EFFICIENCY	0.987
NOZZLE EFFICIENCY	0.936
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.992
MR DISTRIBUTION EFFICIENCY	0.995
BOUNDARY LAYER EFFICIENCY	0.992
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.963
GG BLEED EFFICIENCY	0.995
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	78.56
FUEL FLOWRATE,LB/SEC	35.99
TOTAL FLOWRATE,LB/SEC	114.55
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE,DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE,DEG R	5518.
ENGINE MIXTURE RATIO	2.18
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	9.87
OX ORIFICE DIAMETER (IN)	0.121
FUEL ORIFICE DIAMETER (IN)	0.034

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MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	17500.00
STAGE WEIGHT	102291.38
USABLE PROPELLANT	97000.00
FIXED INERT	
PROPULSION SYSTEM	4993.16
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	119791.38
GROSS BURNOUT WEIGHT	22248.14
PROPELLANT MASS FRACTION	0.948
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	31.17
NUMBER OF NOZZLES	2
STAGE LENGTH	315.95
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	315.95
..PERFORMANCE..	
PROPELLANT	LIQUID
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	2.18
NOZZLE AREA RATIO	[illegible]
BURN TIME,SEC	837.39
ISP,VACUUM DELIVERED,SEC	[illegible]
ISP EFFICIENCY	0.921
PROPELLANT FLOW RATE,LB/SEC	114.55

ORIGINAL PAGE IS
OF POOR QUALITY

AL/LOX OTV (NO AEROBRAKE)

AL/LOX Engines
Isp = 260
O/F = 2.18
15.9 MT Payload
Delta V = 8.34 Km/Sec
No Aerobrake
25% Added to Tanks and Engines

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	6096761.31
USABLE PROPELLANT	6000000.00
FIXED INERT	
PROPULSION SYSTEM	92454.84
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	6131761.31
GROSS BURNOUT WEIGHT	98158.59
PROPELLANT MASS FRACTION	0.984
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	31.17
NUMBER OF NOZZLES	2
STAGE LENGTH	9794.71
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	9794.71
..PERFORMANCE..	
PROPELLANT	LIGUID
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	2.18
NOZZLE AREA RATIO	[illegible]
BURN TIME,SEC	51793.21
ISP,VACUUM DELIVERED,SEC	260.6
ISP EFFICIENCY	0.921
PROPELLANT FLOW RATE,LB/SEC	114.55

ORIGINAL PAGE IS
OF POOR QUALITY

GAS GENERATOR BLEED CYCLE (USER DEFINED GG)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	3.12
CHAMBER DIAMETER	5.40
NOZZLE EXIT DIAMETER	31.17
NOZZLE EXTENSION ATTACH DIAM	13.94
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.087
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	45.02
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	24.34
CHAMBER	45.30
BIPROPELLANT VALVE	4.98
INJECTOR	13.79
TCA SUPPORT HARDWARE	7.47
TCA CONSTRUCTION	26.52

SINGLE THRUST CHAMBER ASSY	122.40
THRUST MOUNT	33.41
GIMBAL SYSTEM	25.58
ENGINE BAY LINES	80.24
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	244.79
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	51.17
TOTAL ENGINE BAY LINES	160.48

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	260.58
IDEAL ISP(ODE),SEC	282.96
DELIVERED CSTAR,FT/SEC	4288.
IDEAL CSTAR,FT/SEC	4343.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	51793.21
OVERALL EFFICIENCY	0.921
ENERGY RELEASE EFFICIENCY	0.987
NOZZLE EFFICIENCY	0.938
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.992
MR DISTRIBUTION EFFICIENCY	0.995
BOUNDARY LAYER EFFICIENCY	0.992
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.960
GG BLEED EFFICIENCY	0.995
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	78.56
FUEL FLOWRATE,LB/SEC	35.99
TOTAL FLOWRATE,LB/SEC	114.55
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE,DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE,DEG R	5518.
ENGINE MIXTURE RATIO	2.18
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	9.87
OX ORIFICE DIAMETER (IN)	0.121
FUEL ORIFICE DIAMETER (IN)	0.034

AL/LOX OTV WITH AEROBRAKE

AL/LOX Engines
Isp = 260
O/F = 2.18
15.9 MT Payload
Delta V = 5.36 Km/Sec
15% Aerobrake
25% Added to Tanks and Engines

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	389645.62
USABLE PROPELLANT	375000.00
FIXED INERT	
PROPULSION SYSTEM	14106.26
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	424645.62
GROSS BURNOUT WEIGHT	47545.45
PROPELLANT MASS FRACTION	0.962
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	31.17
NUMBER OF NOZZLES	2
STAGE LENGTH	734.01
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	734.01
..PERFORMANCE..	
PROPELLANT	LIQUID
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	2.18
NOZZLE AREA RATIO	100.00
BURN TIME,SEC	3237.08
ISP,VACUUM DELIVERED,SEC	260.6
ISP EFFICIENCY	0.921
PROPELLANT FLOW RATE,LB/SEC	114.55

NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

ORIGINAL PAGE IS
OF POOR QUALITY

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	3.12
CHAMBER DIAMETER	5.40
NOZZLE EXIT DIAMETER	31.17
NOZZLE EXTENSION ATTACH DIAM	13.94
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.087
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	45.02
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	24.34
CHAMBER	45.30
BIPROPELLANT VALVE	4.98
INJECTOR	13.79
TCA SUPPORT HARDWARE	7.47
TCA CONSTRUCTION	26.52

SINGLE THRUST CHAMBER ASSY	122.40
THRUST MOUNT	33.41
GIMBAL SYSTEM	25.58
ENGINE BAY LINES	7.69
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	244.79
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	51.17
TOTAL ENGINE BAY LINES	15.39

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	260.58
IDEAL ISP(ODE),SEC	282.96
DELIVERED CSTAR,FT/SEC	4288.
IDEAL CSTAR,FT/SEC	4343.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	3237.08
OVERALL EFFICIENCY	0.921
ENERGY RELEASE EFFICIENCY	0.987
NOZZLE EFFICIENCY	0.938
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.992
MR DISTRIBUTION EFFICIENCY	0.995
BOUNDARY LAYER EFFICIENCY	0.992
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.960
GG BLEED EFFICIENCY	0.995
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	78.56
FUEL FLOWRATE,LB/SEC	35.99
TOTAL FLOWRATE,LB/SEC	114.55
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE,DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE,DEG R	5518.
ENGINE MIXTURE RATIO	2.18
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	9.87
OX ORIFICE DIAMETER (IN)	0.121
FUEL ORIFICE DIAMETER (IN)	0.034

SINGLE STAGE TO MOON

Alum/O Engines
Isp = 260
O/F = 2.10
15.9 MT Payload
Delta V = 20600 Ft/Sec
Lightweight Tanks and Engines

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	449036.50
USABLE PROPELLANT	440000.00
FIXED INERT	
PROPULSION SYSTEM	8454.68
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	484036.50
GROSS BURNOUT WEIGHT	41170.68
PROPELLANT MASS FRACTION	0.980
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	22.05
NUMBER OF NOZZLES	2
STAGE LENGTH	818.97
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	818.97
..PERFORMANCE..	
PROPELLANT	LIQUID
THRUST, VACUUM DELIVERED, LBF	15000.0
PC, PSIA	1000.0
USABLE PROPELLANT MR	2.10
NOZZLE AREA RATIO	100.00
BURN TIME, SEC	7564.16
ISP, VACUUM DELIVERED, SEC	259.7
ISP EFFICIENCY	0.918
PROPELLANT FLOW RATE, LB/SEC	57.41

ORIGINAL PAGE IS
OF POOR QUALITY

GAS GENERATOR BLEED CYCLE (USER DEFINED GG)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

.. ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	2.20
CHAMBER DIAMETER	3.82
NOZZLE EXIT DIAMETER	22.05
NOZZLE EXTENSION ATTACH DIAM	9.86
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.061
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.025
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	31.84
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

.. ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	11.21
CHAMBER	19.12
BIPROPELLANT VALVE	2.49
INJECTOR	5.72
TCA SUPPORT HARDWARE	2.63
TCA CONSTRUCTION	1.93
SINGLE THRUST CHAMBER ASSY	43.10
THRUST MOUNT	22.42
GIMBAL SYSTEM	17.41
ENGINE BAY LINES	5.20
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	86.21
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	34.82
TOTAL ENGINE BAY LINES	10.40

... PERFORMANCE ...

DELIVERED ISP(VAC), SEC	259.74
IDEAL ISP(ODE), SEC	282.96
DELIVERED CSTAR, FT/SEC	4280.
IDEAL CSTAR, FT/SEC	4343.
CHAMBER PRESSURE, PSIA	1000.
THRUST PER ENGINE(VAC), LBF	7500.
TOTAL VAC THRUST, LBF	15000.
BURN TIME, SEC	7564.16
OVERALL EFFICIENCY	0.918
ENERGY RELEASE EFFICIENCY	0.985
NOZZLE EFFICIENCY	0.937
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.993
MR DISTRIBUTION EFFICIENCY	0.992
BOUNDARY LAYER EFFICIENCY	0.991
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.960
GG BLEED EFFICIENCY	0.994
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	38.90
FUEL FLOWRATE, LB/SEC	18.51
TOTAL FLOWRATE, LB/SEC	57.41
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE, DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE, DEG R	5518.
ENGINE MIXTURE RATIO	2.10
FUEL FILM COOLING FRACTION	0.12
INJ ELEMENT DENSITY, ELEM/IN**2	9.86
OX ORIFICE DIAMETER (IN)	0.130
FUEL ORIFICE DIAMETER (IN)	0.034

ORIGINAL PAGE IS
OF POOR QUALITY

SINGLE STAGE TO MOON

AL/O Engines: High Thrust (1500 lbs/engine)
Isp = 260
O/F = 2.18
15.98 MT Payload
Delta V = 19440 Ft/Sec
Lightweight Tanks and Engines

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	448928.10
USABLE PROPELLANT	440000.00
FIXED INERT	
PROPULSION SYSTEM	8336.54
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	0.00
JETTISONED	0.00
GROSS IGNITION WEIGHT	483928.10
GROSS BURNOUT WEIGHT	41463.90
PROPELLANT MASS FRACTION	0.980
..DIMENSIONS,IN..	
STAGE DIAMETER	120.00
NOZZLE EXIT DIAMETER	31.17
NUMBER OF NOZZLES	2
STAGE LENGTH	838.73
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	838.73
..PERFORMANCE..	
PROPELLANT	LIQUID
THRUST,VACUUM DELIVERED,LBF	30000.0
PC,PSIA	1000.0
USABLE PROPELLANT MR	2.18
NOZZLE AREA RATIO	100.00
BURN TIME,SEC	3798.17
ISP,VACUUM DELIVERED,SEC	260.6
ISP EFFICIENCY	0.921
PROPELLANT FLOW RATE,LB/SEC	114.55

ORIGINAL PAGE IS
OF POOR QUALITY

ENGINE SIZE,WEIGHT,& PERFORMANCE SUMMARY FOR STAGE #1
GAS GENERATOR BLEED CYCLE (USER DEFINED GG)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS USER DEFINED

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	3.12
CHAMBER DIAMETER	5.40
NOZZLE EXIT DIAMETER	31.17
NOZZLE EXTENSION ATTACH DIAM	13.94
CONVERGENT CHAMBER LENGTH	4.00
CYLINDRICAL CHAMBER LENGTH	4.00
CHAMBER STRUCTURAL THICKNESS	0.087
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	100.00
CHAMBER CONTRACTION RATIO	3.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAD LENGTH)	1.250
NOZZLE LENGTH	45.02
CHAMBER LENGTH	8.00
INJECTOR FACE FORWARD LENGTH	14.78
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	24.34
CHAMBER	45.30
BIPROPELLANT VALVE	4.98
INJECTOR	13.79
TCA SUPPORT HARDWARE	6.03
TCA CONSTRUCTION	4.42
SINGLE THRUST CHAMBER ASSY	98.86
THRUST MOUNT	33.41
GIMBAL SYSTEM	25.58
ENGINE BAY LINES	8.53
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	197.72
TOTAL THRUST MOUNT	66.83
TOTAL GIMBAL SYSTEM	51.17
TOTAL ENGINE BAY LINES	17.07

... PERFORMANCE ...

DELIVERED ISP(VAC),SEC	260.58
IDEAL ISP(ODE),SEC	282.96
DELIVERED CSTAR,FT/SEC	4288.
IDEAL CSTAR,FT/SEC	4343.
CHAMBER PRESSURE,PSIA	1000.
THRUST PER ENGINE(VAC),LBF	15000.
TOTAL VAC THRUST,LBF	30000.
BURN TIME,SEC	3798.17
OVERALL EFFICIENCY	0.921
ENERGY RELEASE EFFICIENCY	0.987
NOZZLE EFFICIENCY	0.938
KINETIC EFFICIENCY	0.993
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.992
MR DISTRIBUTION EFFICIENCY	0.995
BOUNDARY LAYER EFFICIENCY	0.992
DIVERGENCE EFFICIENCY	0.992
TWO PHASE EFFICIENCY	0.960
GG BLEED EFFICIENCY	0.995
FOR 2 ENGINES	
OXIDIZER FLOWRATE,LB/SEC	78.56
FUEL FLOWRATE,LB/SEC	35.99
TOTAL FLOWRATE,LB/SEC	114.55
CORE MIXTURE RATIO	2.40
CORE TEMPERATURE,DEG R	8053.
BARRIER MIXTURE RATIO	0.00
BARRIER TEMPERATURE,DEG R	5518.
ENGINE MIXTURE RATIO	2.18
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY,ELEM/IN**2	9.87
OX ORIFICE DIAMETER (IN)	0.121
FUEL ORIFICE DIAMETER (IN)	0.034

SECOND STAGE HAD
DELTA V = 1.12 Km/Sec

1ST STAGE HAD NO GUIDELINES

H/O Engines
O/F = 5.5
Isp = 470

ORIGINAL PAGE IS
OF POOR QUALITY

ENGINE SIZE, WEIGHT, & PERFORMANCE SUMMARY FOR STAGE #1
EXPANDER CYCLE (FUEL SIDE)
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

ENGINE DIMENSIONS (INCHES) ...	
THROAT DIAMETER	1.55
CHAMBER DIAMETER	3.10
NOZZLE EXIT DIAMETER	26.86
NOZZLE EXTENSION ATTACH DIAM	6.93
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.089
GAS SIDE WALL THICKNESS	0.033
NOZZLE EXTENSION THICKNESS	0.027
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	47.93
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00
ENGINE WEIGHTS (POUNDS) ...	
NOZZLE EXTENSION	23.01
CHAMBER	20.85
BIPROPELLANT VALVE	1.73
INJECTOR	4.53
TCA SUPPORT HARDWARE	4.24
TCA CONSTRUCTION	15.04
SINGLE THRUST CHAMBER ASSY	69.39
THRUST MOUNT	22.42
GIMBAL SYSTEM	21.43
ENGINE BAY LINES	4.81
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	138.78
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	42.85
TOTAL ENGINE BAY LINES	9.62

... PERFORMANCE ...	
DELIVERED ISP(VAC), SEC	469.61
IDEAL ISP(ODE), SEC	482.94
DELIVERED CSTAR, FT/SEC	7608.
IDEAL CSTAR, FT/SEC	7647.
CHAMBER PRESSURE, PSIA	2000.
THRUST PER ENGINE(VAC), LBF	7500.
TOTAL VAC THRUST, LBF	15000.
BURN TIME, SEC	4680.53
OVERALL EFFICIENCY	0.972
ENERGY RELEASE EFFICIENCY	0.988
NOZZLE EFFICIENCY	0.984
KINETIC EFFICIENCY	1.000
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.997
MR DISTRIBUTION EFFICIENCY	0.991
BOUNDARY LAYER EFFICIENCY	0.989
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	27.02
FUEL FLOWRATE, LB/SEC	4.92
TOTAL FLOWRATE, LB/SEC	31.94
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE, DEG R	6515.
BARRIER MIXTURE RATIO	2.44
BARRIER TEMPERATURE, DEG R	3792.
ENGINE MIXTURE RATIO	5.49
FUEL FILM COOLING FRACTION	0.09
INJ ELEMENT DENSITY, ELEM/IN**2	10.19
OX ORIFICE DIAMETER (IN)	0.059
FUEL ORIFICE DIAMETER (IN)	0.078

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1	STAGE #2
..WEIGHT,LB..		
PAYLOAD	-	35000.00
STAGE WEIGHT	156053.42	20914.73
USABLE PROPELLANT	150000.00	12250.00
FIXED INERT		
PROPULSION SYSTEM	4994.25	8341.79
INTERSTAGE	298.12	0.00
EXPENDED INERT		
EXPELLED	42.34	42.44
JETTISONED	0.00	0.00
GROSS IGNITION WEIGHT	211968.15	55914.73
GROSS BURNOUT WEIGHT	61925.81	43622.29
PROPELLANT MASS FRACTION	0.961	0.586
..DIMENSIONS,IN..		
STAGE DIAMETER	150.00	150.00
NOZZLE EXIT DIAMETER	26.86	26.86
NUMBER OF NOZZLES	2	2
STAGE LENGTH	862.02	293.13
PAYLOAD LENGTH	-	0.00
TOTAL VEH LENGTH	-	1155.15
..PERFORMANCE..		
PROPELLANT	LO2/LH2	LO2/LH2
THRUST, VACUUM DELIVERED, LBF	15000.0	15000.0
PC, PSIA	2000.0	2000.0
USABLE PROPELLANT MR	5.49	5.49
NOZZLE AREA RATIO	300.00	300.00
BURN-TIME, SEC	4680.53	378.30
ISP, VACUUM DELIVERED, SEC	469.6	469.6
ISP EFFICIENCY	0.972	0.972
PROPELLANT FLOW RATE, LB/SEC	31.94	31.94

HIGH O/F OTV WITH AEROBRAKE (PRESSURE FED)

H/O Engines (Pressure Fed)
Isp = 381
O/F = 11.17
15.9 MT Payload
Delta V = 5.36 Km/sec
15% Aerobrake
25% Added to Tanks and Engines

NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

ORIGINAL PAGE IS
OF POOR QUALITY

ENGINE DIMENSIONS (INCHES) ...	
THROAT DIAMETER	3.45
CHAMBER DIAMETER	6.89
NOZZLE EXIT DIAMETER	59.71
NOZZLE EXTENSION ATTACH DIAM	15.42
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.072
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.022
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	106.55
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

ENGINE WEIGHTS (POUNDS) ...	
NOZZLE EXTENSION	93.08
CHAMBER	63.71
BIPROPELLANT VALVE	3.77
INJECTOR	15.19
TCA SUPPORT HARDWARE	14.85
TCA CONSTRUCTION	52.72
SINGLE THRUST CHAMBER ASSY	243.32
THRUST MOUNT	22.42
GIMBAL SYSTEM	26.78
ENGINE BAY LINES	1.00
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	486.63
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	53.56
TOTAL ENGINE BAY LINES	2.01

... PERFORMANCE ...	
DELIVERED ISP(VAC), SEC	381.26
IDEAL ISP(ODE), SEC	396.73
DELIVERED CSTAR, FT/SEC	6106.
IDEAL CSTAR, FT/SEC	6088.
CHAMBER PRESSURE, PSIA	400.
THRUST PER ENGINE(VAC), LBF	7500.
TOTAL VAC THRUST, LBF	15000.
BURN TIME, SEC	4448.04
OVERALL EFFICIENCY	0.961
ENERGY RELEASE EFFICIENCY	0.996
NOZZLE EFFICIENCY	0.965
KINETIC EFFICIENCY	0.982
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.996
MR DISTRIBUTION EFFICIENCY	1.000
BOUNDARY LAYER EFFICIENCY	0.987
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	36.11
FUEL FLOWRATE, LB/SEC	3.23
TOTAL FLOWRATE, LB/SEC	39.34
CORE MIXTURE RATIO	12.00
CORE TEMPERATURE, DEG R	6009.
BARRIER MIXTURE RATIO	2.77
BARRIER TEMPERATURE, DEG R	4165.
ENGINE MIXTURE RATIO	11.17
FUEL FILM COOLING FRACTION	0.07
INJ ELEMENT DENSITY, ELEM/IN**2	10.10
OX ORIFICE DIAMETER (IN)	0.045
FUEL ORIFICE DIAMETER (IN)	0.060

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT, LB..	
PAYLOAD	35000.00
STAGE WEIGHT	195523.50
USABLE PROPELLANT	175000.00
FIXED INERT	
PROPULSION SYSTEM	16332.22
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	1022.90
JETTISONED	0.00
GROSS IGNITION WEIGHT	230523.50
GROSS BURNOUT WEIGHT	54500.60
PROPELLANT MASS FRACTION	0.895
..DIMENSIONS, IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	59.71
NUMBER OF NOZZLES	2
STAGE LENGTH	718.37
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	718.37
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST, VACUUM DELIVERED, LBF	15000.0
PC, PSIA	400.0
USABLE PROPELLANT MR	11.17
NOZZLE AREA RATIO	300.00
BURN TIME, SEC	4448.04
ISP, VACUUM DELIVERED, SEC	381.3
ISP EFFICIENCY	0.961
PROPELLANT FLOW RATE, LB/SEC	39.34

OTV WITH AEROBRAKE (PRESSURE FED)
COMPOSITE TANKS

H/O Engines (Pressure Fed)
Isp = 466
O/F = 5.81
15.9 MT Payload
Delta V= 5.36 Km/Sec
15% Aerobrake
25% Added to Tanks and Engines

PRESSURE FED
CHAMBER IS REGEN COOLED (MILLED SLOT CONSTRUCTION)
NOZZLE IS RADIATION COOLED
PROPELLANT COMBINATION IS LO2/LH2

... ENGINE DIMENSIONS (INCHES) ...

THROAT DIAMETER	3.45
CHAMBER DIAMETER	6.91
NOZZLE EXIT DIAMETER	59.81
NOZZLE EXTENSION ATTACH DIAM	15.44
CONVERGENT CHAMBER LENGTH	5.00
CYLINDRICAL CHAMBER LENGTH	7.60
CHAMBER STRUCTURAL THICKNESS	0.072
GAS SIDE WALL THICKNESS	0.030
NOZZLE EXTENSION THICKNESS	0.022
NOZZLE EXIT AREA RATIO	300.00
CHAMBER CONTRACTION RATIO	4.00
NOZ EXTENSION ATTCH AREA RATIO	20.00
NOZZLE LENGTH/(MIN RAO LENGTH)	1.250
NOZZLE LENGTH	106.73
CHAMBER LENGTH	12.60
INJECTOR FACE FORWARD LENGTH	11.89
MOUNT LENGTH	2.00

... ENGINE WEIGHTS (POUNDS) ...

NOZZLE EXTENSION	93.43
CHAMBER	64.01
BIPROPELLANT VALVE	3.55
INJECTOR	15.25
TCA SUPPORT HARDWARE	14.89
TCA CONSTRUCTION	52.87

SINGLE THRUST CHAMBER ASSY	244.00
THRUST MOUNT	22.42
GIMBAL SYSTEM	26.78
ENGINE BAY LINES	0.97
TOTAL NUMBER OF ENGINES	2
CLUSTER EXIT RADIUS	0.00
CLUSTER AREA RATIO	0.00
MODULE TILT ANGLE (DEG)	0.00
TOTAL ENGINE	488.00
TOTAL THRUST MOUNT	44.84
TOTAL GIMBAL SYSTEM	53.56
TOTAL ENGINE BAY LINES	1.95

... PERFORMANCE ...

DELIVERED ISP(VAC), SEC	466.02
IDEAL ISP(ODE), SEC	481.82
DELIVERED CSTAR, FT/SEC	7488.
IDEAL CSTAR, FT/SEC	7515.
CHAMBER PRESSURE, PSIA	400.
THRUST PER ENGINE(VAC), LBF	7500.
TOTAL VAC THRUST, LBF	15000.
BURN TIME, SEC	3808.91
OVERALL EFFICIENCY	0.967
ENERGY RELEASE EFFICIENCY	0.993
NOZZLE EFFICIENCY	0.974
KINETIC EFFICIENCY	0.992
VAPORIZATION EFFICIENCY	1.000
MIXING EFFICIENCY	0.996
MR DISTRIBUTION EFFICIENCY	0.997
BOUNDARY LAYER EFFICIENCY	0.987
DIVERGENCE EFFICIENCY	0.995
TWO PHASE EFFICIENCY	1.000
FOR 2 ENGINES	
OXIDIZER FLOWRATE, LB/SEC	27.46
FUEL FLOWRATE, LB/SEC	4.73
TOTAL FLOWRATE, LB/SEC	32.19
CORE MIXTURE RATIO	6.00
CORE TEMPERATURE, DEG R	6100.
BARRIER MIXTURE RATIO	2.81
BARRIER TEMPERATURE, DEG R	4216.
ENGINE MIXTURE RATIO	5.81
FUEL FILM COOLING FRACTION	0.03
INJ ELEMENT DENSITY, ELEM/IN**2	10.23
OX ORIFICE DIAMETER (IN)	0.039
FUEL ORIFICE DIAMETER (IN)	0.065

MOON BASE PROPULSION 6/9/86

**** VEHICLE SUMMARY ****

	STAGE #1
..WEIGHT,LB..	
PAYLOAD	35000.00
STAGE WEIGHT	143661.30
USABLE PROPELLANT	122600.00
FIXED INERT	
PROPULSION SYSTEM	16714.15
INTERSTAGE	0.00
EXPENDED INERT	
EXPELLED	1036.82
JETTISONED	0.00
GROSS IGNITION WEIGHT	178661.30
GROSS BURNOUT WEIGHT	55024.48
PROPELLANT MASS FRACTION	0.853
..DIMENSIONS,IN..	
STAGE DIAMETER	150.00
NOZZLE EXIT DIAMETER	59.81
NUMBER OF NOZZLES	2
STAGE LENGTH	727.69
PAYLOAD LENGTH	0.00
TOTAL VEH LENGTH	727.69
..PERFORMANCE..	
PROPELLANT	LO2/LH2
THRUST,VACUUM DELIVERED,LBF	15000.0
PC,PSIA	400.0
USABLE PROPELLANT MR	5.81
NOZZLE AREA RATIO	300.00
BURN TIME,SEC	3808.91
ISP,VACUUM DELIVERED,SEC	466.0
ISP EFFICIENCY	0.967
PROPELLANT FLOW RATE,LB/SEC	32.19

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APPENDIX D

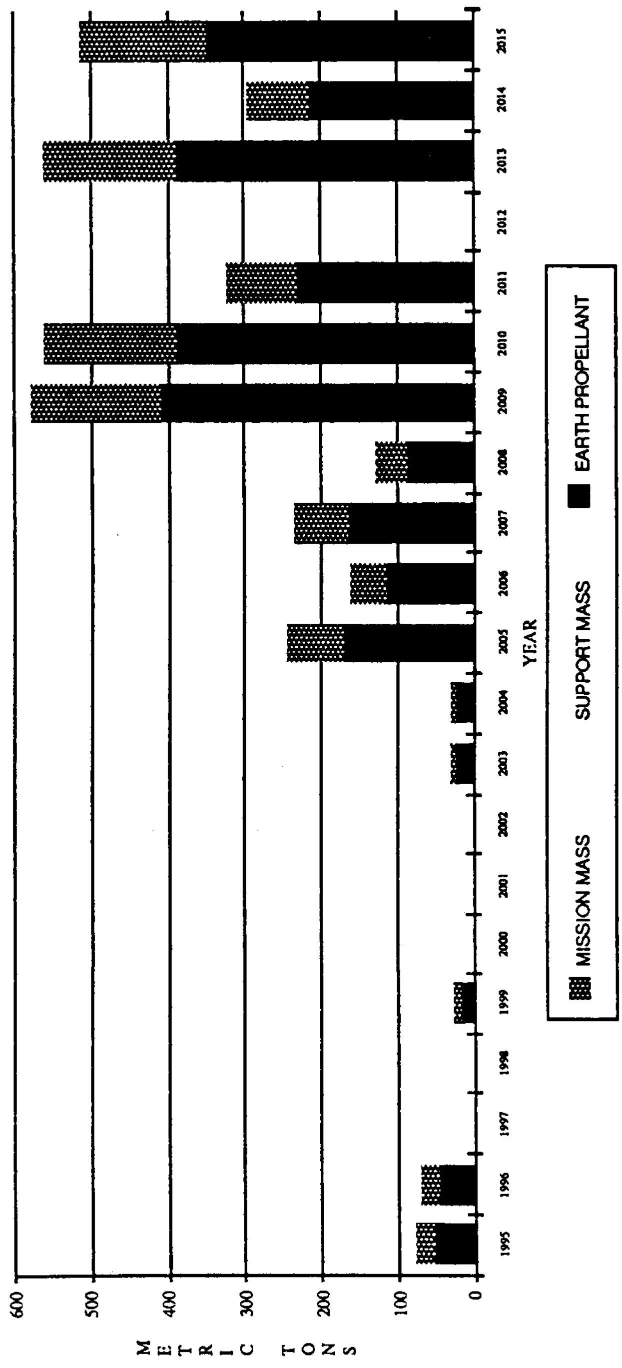
ENGINE/PERFORMANCE DATA

D.1 ASTROSIZE OUTPUT DATA
D.2 SAMPLE ASTROFEST OUTPUT DATA

APPENDIX D.1

ASTROSIZE OUTPUT DATA

H/O OTV & LANDER
(LLOX IN 1995)



```
RUN
Enter the name of the data file for the
vehicle you wish to use:ASA.DAT
Enter slope and Y-intercept of process function:.0015,2300
A) OTV & Lander: Lunar propellants available
B) OTV & Lander: Lunar propellants not available
C) Integrated OTV/Lander: Lunar propellants available
D) Integrated OTV/Lander: Lunar propellants not available
Choose which type of configuration you will use:A
Enter the year which you wish to deliver equipment:1995
Year:          1995

Lunar Propellant:                    89419.34
Lunar LOX:     89419.34
Lunar Fuel:    0

Earth Propellant:                    53553.43
Earth LOX:     31557.62
Earth Fuel:    21995.81

Number of Flights:                   2
Number of Manned Flights:            0
Mass Delivered (kg):                 25134.13
Additional Burdened Mass (kg):                      2434.129
Mass Required From Earth (kg):                      78687.56


Year:          1996

Lunar Propellant:                    87646.85
Lunar LOX:     87646.85
Lunar Fuel:    0

Earth Propellant:                    49371.22
Earth LOX:     28291.52
Earth Fuel:    21079.7

Number of Flights:                   2
Number of Manned Flights:            0
Mass Delivered (kg):                 22831.47
Additional Burdened Mass (kg):                      131.4707
Mass Required From Earth (kg):                      72202.69


Year:          1999

Lunar Propellant:                    41395.88
Lunar LOX:     41395.88
Lunar Fuel:    0

Earth Propellant:                    18957.8
Earth LOX:     9672.62
Earth Fuel:    9285.182

Number of Flights:                   1
Number of Manned Flights:            0
Mass Delivered (kg):                 8262.094
Additional Burdened Mass (kg):                      62.09375
Mass Required From Earth (kg):                      27219.9
```

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Year: 2003

Lunar Propellant: 33506.11
Lunar LOX: 33506.11
Lunar Fuel: 0

Earth Propellant: 23196.58
Earth LOX: 14473.09
Earth Fuel: 8723.488

Number of Flights: 1
Number of Manned Flights: 1
Mass Delivered (kg): 6850.26
Additional Burdened Mass (kg): 50.25928
Mass Required From Earth (kg): 30046.83

Year: 2004

Lunar Propellant: 33506.11
Lunar LOX: 33506.11
Lunar Fuel: 0

Earth Propellant: 23196.58
Earth LOX: 14473.09
Earth Fuel: 8723.488

Number of Flights: 1
Number of Manned Flights: 1
Mass Delivered (kg): 6850.26
Additional Burdened Mass (kg): 50.25928
Mass Required From Earth (kg): 30046.83

Year: 2005

Lunar Propellant: 219922.4
Lunar LOX: 219922.4
Lunar Fuel: 0

Earth Propellant: 171037.5
Earth LOX: 110889.8
Earth Fuel: 60147.67

Number of Flights: 5
Number of Manned Flights: 2
Mass Delivered (kg): 76129.88
Additional Burdened Mass (kg): 329.8828
Mass Required From Earth (kg): 247167.4

```
Do you wish to use another vehicle after 2005? (Y/N):N
Year:            2006

Lunar Propellant:               120493.2
Lunar LOX:       120493.2
Lunar Fuel:      0

Earth Propellant:               116720.4
Earth LOX:       80226.04
Earth Fuel:      36494.4

Number of Flights:              3
Number of Manned Flights:       3
Mass Delivered (kg):            46680.74
Additional Burdened Mass (kg):                 180.7383
Mass Required From Earth (kg):                 163401.2


Year:            2007

Lunar Propellant:               208038.9
Lunar LOX:       208038.9
Lunar Fuel:      0

Earth Propellant:               165853.2
Earth LOX:       108331.3
Earth Fuel:      57521.86

Number of Flights:              5
Number of Manned Flights:       3
Mass Delivered (kg):            69512.06
Additional Burdened Mass (kg):                 312.0547
Mass Required From Earth (kg):                 235365.2


Year:            2008

Lunar Propellant:               137207.4
Lunar LOX:       137207.4
Lunar Fuel:      0

Earth Propellant:               87593.42
Earth LOX:       53008.7
Earth Fuel:      34584.73

Number of Flights:              3
Number of Manned Flights:       0
Mass Delivered (kg):            41905.81
Additional Burdened Mass (kg):                 205.8125
Mass Required From Earth (kg):                 129499.2
```

Year: 2009

Lunar Propellant: 453490
Lunar LOX: 453490
Lunar Fuel: 0

Earth Propellant: 409828.3
Earth LOX: 277010.1
Earth Fuel: 132818.2

Number of Flights: 11
Number of Manned Flights: 9
Mass Delivered (kg): 168680.2
Additional Burdened Mass (kg): 680.2344
Mass Required From Earth (kg): 578508.6

Year: 2010

Lunar Propellant: 474745.7
Lunar LOX: 474745.7
Lunar Fuel: 0

Earth Propellant: 391417.2
Earth LOX: 258161.4
Earth Fuel: 133255.8

Number of Flights: 11
Number of Manned Flights: 6
Mass Delivered (kg): 169812.1
Additional Burdened Mass (kg): 712.125
Mass Required From Earth (kg): 561229.4

Year: 2011

Lunar Propellant: 240139.7
Lunar LOX: 240139.7
Lunar Fuel: 0

Earth Propellant: 231443
Earth LOX: 158891.8
Earth Fuel: 72551.16

Number of Flights: 6
Number of Manned Flights: 6
Mass Delivered (kg): 92260.21
Additional Burdened Mass (kg): 360.211
Mass Required From Earth (kg): 323703.2

Year: 2013

Lunar Propellant:		474745.7	
Lunar LOX:	474745.7		
Lunar Fuel:	0		
Earth Propellant:		391417.2	
Earth LOX:	258161.4		
Earth Fuel:	133255.8		
Number of Flights:		11	
Number of Manned Flights:		6	
Mass Delivered (kg):		169812.1	
Additional Burdened Mass (kg):			712.125
Mass Required From Earth (kg):			561229.4

Year: 2014

Lunar Propellant:		233134.8	
Lunar LOX:	233134.8		
Lunar Fuel:	0		
Earth Propellant:		214915.1	
Earth LOX:	145984.4		
Earth Fuel:	68930.75		
Number of Flights:		6	
Number of Manned Flights:		6	
Mass Delivered (kg):		83149.71	
Additional Burdened Mass (kg):			349.7031
Mass Required From Earth (kg):			298064.8

Year: 2015

Lunar Propellant:		505373.7	
Lunar LOX:	505373.7		
Lunar Fuel:	0		
Earth Propellant:		349410.7	
Earth LOX:	217905.4		
Earth Fuel:	131505.3		
Number of Flights:		11	
Number of Manned Flights:		1	
Mass Delivered (kg):		165458.1	
Additional Burdened Mass (kg):			738.0625
Mass Required From Earth (kg):			514868.8

=============TOTALS=============

Total Propellant from Moon:	3352766	
Total LOX from Moon:	3352766	
Total Fuel from Moon:	0	
Total Fuel for OTV from Moon:		0
Total Fuel for Lander from Moon:		0
Total Propellant from Earth:		2697912
Total LOX from Earth:	2915817	
Total Fuel from Earth:	930873.4	
Total OTV Fuel from Earth:	522047.8	
Total Lander Fuel from Earth:		408825.5
Total Propellant:	6050677	
Total Number of Flights:	79	
Total Number of Manned Flights:		44
Total Mass Delivered (kg):	1153329	
Total Additional Burdened Mass (kg):		7329.125
Total Mass Required From Earth (kg):		3851241
Vehicle Data File:	A5A.DAT	

0

APPENDIX D.2

SAMPLE ASTROFEST OUTPUT DATA

BASELINE

H / O OTV & LANDER

15% AEROBRAKE

470 sec Isp

15873 kg PAYLOAD (35000 lbs)

5.5 O / F

NO LUNAR PROPELLANT AVAILABLE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:BASE.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants. The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV. The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              470
Number of engines:              2
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):             95
LOX - Hydrogen engine with MR:                 5.5


OTV MASS  (kg):

Dry Mass:        1030
Aerobrake Mass:                 3410.811
LOX Tank Mass:                  367.044
Fuel Tank Mass:                 1618.33
Pressure Tank Mass:             0

Total Mass:      6426.185


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 91760.99
LOX Carried for OTV: 70985.13
LOX Carried for Lander: 20775.86
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 12906.39
Fuel Capacity Carried for Lander: 3777.43

Total Propellant Capacity: 83891.52

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9288491

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1841.305
LOX Tank Mass: 83.10345
Fuel Tank Mass: 366.4107
Pressure Tank Mass: 0

Total Mass: 3320.819

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20775.86
Fuel Capacity: 3777.43

Total Propellant Capacity: 24553.29

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 8973
Liftoff Payload Capability: 8973

Mass Fraction: .8808637

This data has been stored in a file called: BASE.DAT

H / O OTV & LANDER

LLOX AVAILABLE

Isp = 470

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A5A.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17766.54
LUNAR LOX USED BY OTV:          4308.442
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             13458.1
LEO-BASED LOX BURNED:           33926.18


 -----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:              5.5
```

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 2888.184
LOX Tank Mass: 135.7047
Fuel Tank Mass: 1194.38
Pressure Tank Mass: 0

Total Mass: 5248.27

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 33926.18
LOX Carried for OTV: 33926.18
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 6951.75
Fuel Capacity Carried for Lander: 5361.449

Total Propellant Capacity: 40877.93

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .8862193

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1846.201
LOX Tank Mass: 83.4128
Fuel Tank Mass: 367.7746
Pressure Tank Mass: 0

Total Mass: 3327.388

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20853.2
Fuel Capacity: 3791.49

Total Propellant Capacity: 24644.69

Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 15873
Liftoff Payload Capability: 8973
Tank Structure for Refueling OTV: 89.72999

Mass Fraction: .8810461

This data has been stored in a file called: A5A.DAT
Ok

H / O OTV & LANDER

LLOX AND LUNAR HYDROGEN AVAILABLE

Isp = 470

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A5.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
100,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
100,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17037.78
LUNAR LOX USED BY OTV:          4008.19
LUNAR FUEL USED BY OTV :        728.7618
LUNAR LOX RETURNED=             13029.59
LEO-BASED LOX BURNED:           25191.55


-----------------OTV DESIGN-----------------

OTV ENGINE DATA:
Isp:            470
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):        95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:             2798.1
```

```
LOX Tank Mass:                  100.7662
Fuel Tank Mass:                 444.2873
Pressure Tank Mass:             0

Total Mass:     4373.154


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:             25191.55
LOX Carried for OTV:            25191.55
LOX Carried for Lander:         0
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:          4580.281
Fuel Capacity Carried for Lander:            0

Total Propellant Capacity:   29771.83

Percent of return trip LOX from LSB:        100
Percent of return trip Fuel from LSB:       100

Payload Capability to LSB:   15873
Return Payload Capability:   15873

Mass Fraction:                  .8719241



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:            470
Number of engines:              2
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):         95
LOX - Hydrogen engine with MR:               5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:              1846.201
LOX Tank Mass:                  83.4128
Fuel Tank Mass:                 367.7746
Pressure Tank Mass:             0

Total Mass:     3327.388

LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                   20853.2
Fuel Capacity:                  3791.49

Total Propellant Capacity:   24644.69

Percent of Lander LOX supplied from LSB:    100
Percent of Lander Fuel supplied from LSB:              100

Maximum Payload Capability:              15873
Liftoff Payload Capability:              8973
Tank Structure for Refueling OTV:        89.72999

Mass Fraction:                  .8810461


This data has been stored in a file called:              A5.DAT
Ok
```

H / O OTV & Al / LOX LANDER

OTV Isp = 470 sec

LANDER Isp = 260 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:N
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,260
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
1414
Enter the maximum payload for OTV & for Lander:15873,12973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,2.18
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:7,33361
Enter mass for each engine & its thrust structure for Lander:190
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,B


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:7
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:          470
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:      1030
Aerobrake Mass:             3908.108
LOX Tank Mass:              1001.929
Tank for OTV Fuel:          3138.565
Tank for Lander Fuel:       99.80154
Pressure Tank Mass:         499.0077

Total Mass:    9677.412


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 250482.3
LOX Carried for OTV: 177959.9
LOX Carried for Lander: 72522.46
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 32356.34
Fuel Capacity Carried for Lander: 33267.18

Total Propellant Capacity: 210316.2

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9560105

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 260
Number of engines: 7
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 190
LOX - Aluminum engine with MR: 2.18

LANDER MASS (kg):

Dry Mass: 2744
Landing Gear Mass: 6447.542
LOX Tank Mass: 471.396
Fuel Tank Mass: 99.80154
Pressure Tank Mass: 499.0077

Total Mass: 10261.75

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 72522.46
Fuel Capacity: 33267.18

Total Propellant Capacity: 105789.6

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873
Liftoff Payload: 12973

Mass Fraction: .9115759

H / O OTV & Al / LOX LANDER

OTV Isp = 470 sec

LANDER Isp = 260 sec

LANDER O / F = 2.18

LLOX & LUNAR Al AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A122.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,260
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1794
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,2.18
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:190
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,B


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
100,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:7

The Lander liftoff payload capability is:  12973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
===============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          12843.27
LUNAR LOX USED BY OTV:         3097.587
LUNAR FUEL USED BY OTV :       0
LUNAR LOX RETURNED=            9745.682
LEO-BASED LOX BURNED:          25058.97


 -----------------OTV DESIGN----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:             2
Thrust per engine (N):         33361
Mass of each engine & its thrust structure (kg):            95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:        1030
Aerobrake Mass:                2076.588
LOX Tank Mass:                 100.2359
Tank for OTV Fuel:             496.5794
Tank for Lander Fuel:          0
Pressure Tank Mass:            0
```

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```
Total Mass:      3703.403        -

OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:             25058.97
LOX Carried for OTV:            25058.97
LOX Carried for Lander:         0
Additional LOX Storage Capability for Return Trip:          0
Fuel Capacity for OTV:          5119.375
Fuel Capacity Carried for Lander:           0

Total Propellant Capacity:      25058.97

Percent of return trip LOX from LSB:        100
Percent of return trip Fuel from LSB:       0

Payload Capability to LSB:      15873
Return Payload Capability:      15873

Mass Fraction:                  .8712414


--------------LANDER DESIGN---------------

LANDER ENGINE DATA:

Isp:            260
Number of engines:              7
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):            190
LOX - Aluminum engine with MR:              2.18

LANDER MASS  (kg):

Dry Mass:       2744
Landing Gear Mass:              6474.846
LOX Tank Mass:                  473.6859
Fuel Tank Mass:                 100.2863
Pressure Tank Mass:             501.4318

Total Mass:      10294.25

LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                   72874.75
Fuel Capacity:                  33428.78

Total Propellant Capacity:      106303.5

Percent of Lander LOX supplied from LSB:    100
Percent of Lander Fuel supplied from LSB:                   100

Maximum Payload Capability:                 15873
Liftoff Payload Capability:                 12973
Tank Structure for Refueling OTV:           129.73

Mass Fraction:                  .9117114


This data has been stored in a file called:                 A122.DAT
Ok
```

H / O OTV & LANDER

O / F = 10.6

Isp = 384 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:


Y
What do you wish to call the data file:C1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:384,384
Enter OTV mass kg (NOT including tanks):955,955
Enter Lander mass kg (NOT including tanks and landing gear):
955845
Enter Lander mass kg (NOT including tanks and landing gear):
845
Enter the maximum payload for OTV & for Lander:15873,12173
Enter the O/F mixture ratio for the OTV & for the Lander:10.6,10.6
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:110
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:110
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:3
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            384
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):        110
LOX - Hydrogen engine with MR:             10.6


OTV MASS  (kg):

Dry Mass:       1065
Aerobrake Mass:             3481.878
LOX Tank Mass:              682.729
Fuel Tank Mass:             1561.904
Pressure Tank Mass:         0
```

Total Mass: 6791.511

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity:	170682.3
LOX Carried for OTV:	135223.2
LOX Carried for Lander:	35459.06
Additional LOX Storage Capability for Return Trip:	0
Fuel Capacity for OTV:	12756.9
Fuel Capacity Carried for Lander:	3345.195

Total Propellant Capacity: 147980.1

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9561192

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp:	384
Number of engines:	3
Thrust per engine (N):	33361
Mass of each engine & its thrust structure (kg):	110
LOX - Hydrogen engine with MR:	10.6

LANDER MASS (kg):

Dry Mass:	1175
Landing Gear Mass:	2767.737
LOX Tank Mass:	141.8362
Fuel Tank Mass:	324.484
Pressure Tank Mass:	0

Total Mass: 4409.057

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 35459.06
Fuel Capacity: 3345.195

Total Propellant Capacity: 38804.25

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 12173
Liftoff Payload Capability: 12173

Mass Fraction: .8979699

This data has been stored in a file called: C1.DAT

H / O OTV & LANDER

O / F = 10.6

Isp = 384 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:C2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,260384,384
Enter OTV mass kg (NOT including tanks):1065
Enter Lander mass kg (NOT including tanks and landing gear):
1065
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:10.6,10.6
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:110
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:110
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:,A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:3

The Lander liftoff payload capability is:  12173
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          12051.27
LUNAR LOX USED BY OTV:          3798.745
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             8252.525
LEO-BASED LOX BURNED:           43435.9

-----------------OTV DESIGN----------------


OTV ENGINE DATA:
Isp:            384
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):          110
LOX - Hydrogen engine with MR:          10.6


OTV MASS  (kg):

Dry Mass:       1065
Aerobrake Mass:             1881.986
LOX Tank Mass:              173.7436
Fuel Tank Mass:             758.2567
Pressure Tank Mass:         0

Total Mass:     3878.986
```

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OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 43435.9
LOX Carried for OTV: 43435.9
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 4456.1
Fuel Capacity Carried for Lander: 3360.979

Total Propellant Capacity: 47892

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9250742

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 384
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 110
LOX - Hydrogen engine with MR: 10.6

LANDER MASS (kg):

Dry Mass: 1175
Landing Gear Mass: 2777.484
LOX Tank Mass: 142.5055
Fuel Tank Mass: 326.0151
Pressure Tank Mass: 0

Total Mass: 4421.005

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 35626.38
Fuel Capacity: 3360.981

Total Propellant Capacity: 38987.36

Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 15873
Liftoff Payload Capability: 12173
Tank Structure for Refueling OTV: 121.73

Mass Fraction: .8981532

This data has been stored in a file called: C2.DAT
Ok

H / O OTV & LANDER

O / F = 8.73

Isp = 421 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:D1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:421,421
Enter OTV mass kg (NOT including tanks):840,
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:8.73,8.73
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:3
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            421
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):        95
LOX - Hydrogen engine with MR:             8.729999


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:               3443.29
LOX Tank Mass:                559.6293
Fuel Tank Mass:               1554.526
Pressure Tank Mass:           0

Total Mass:     6587.445


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 139907.3
LOX Carried for OTV: 107958.1
LOX Carried for Lander: 31949.27
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 12366.33
Fuel Capacity Carried for Lander: 3659.709

Total Propellant Capacity: 120324.4

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9480943

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 421
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 8.729999

LANDER MASS (kg):

Dry Mass: 1125
Landing Gear Mass: 2780.816
LOX Tank Mass: 127.7971
Fuel Tank Mass: 354.9918
Pressure Tank Mass: 0

Total Mass: 4388.605

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 31949.27
Fuel Capacity: 3659.709

Total Propellant Capacity: 35608.98

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873
Liftoff Payload: 15650.45

Mass Fraction: .8902782

This data has been stored in a file called: D1.DAT

H / O OTV & LANDER

O / F = 8.73

Isp = 421 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

D-36

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:D2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:421,421
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:8.73,8.73
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y]
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:3

The Lander liftoff payload capability is:  15545.07
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
==============================================


This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          15389.62
LUNAR LOX USED BY OTV:         4291.386
LUNAR FUEL USED BY OTV :       0
LUNAR LOX RETURNED=            11098.24
LEO-BASED LOX BURNED:          38271.77


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            421
Number of engines:             2
Thrust per engine (N):         33361
```

Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 8.729999

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 2401.317
LOX Tank Mass: 153.0871
Fuel Tank Mass: 828.9523
Pressure Tank Mass: 0

Total Mass: 4413.356

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 38271.77
LOX Carried for OTV: 38271.77
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 4875.505
Fuel Capacity Carried for Lander: 3670.395

Total Propellant Capacity: 43147.27

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9072057

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 421
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 8.729999

LANDER MASS (kg):

Dry Mass: 1125
Landing Gear Mass: 2780.816
LOX Tank Mass: 128.1702
Fuel Tank Mass: 356.0283
Pressure Tank Mass: 0

Total Mass: 4390.015

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 32042.54
Fuel Capacity: 3670.395

Total Propellant Capacity: 35712.94

D-38 Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873
Liftoff Payload: 15545.07
Tank Structure for Refueling OTV: 155.4508

Mass Fraction: .8905314

This data has been stored in a file called: D2.DAT

H / O OTV & LANDER

Isp = 460 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A5LI1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:460,460
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:             460
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):         95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:      1030
Aerobrake Mass:               3428.453
LOX Tank Mass:                383.3489
Fuel Tank Mass:               1690.22
Pressure Tank Mass:           0

Total Mass:    6532.022


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:           95837.21
```

LOX Carried for OTV: 74520.29
LOX Carried for Lander: 21316.92
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 13549.14
Fuel Capacity Carried for Lander: 3875.803

Total Propellant Capacity: 88069.43

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9309522

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 460
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1853.883
LOX Tank Mass: 85.26769
Fuel Tank Mass: 375.9529
Pressure Tank Mass: 0

Total Mass: 3345.103

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 21316.92
Fuel Capacity: 3875.803

Total Propellant Capacity: 25192.72

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873
Liftoff Payload: 8560.916

Mass Fraction: .8827836

This data has been stored in a file called: A5LI1.DAT

H / O OTV & LANDER

Isp = 460 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
SAVE"RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A5LI2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:460,460
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8509.004
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          16847.83
LUNAR LOX USED BY OTV:          4197.74
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             12650.09
LEO-BASED LOX BURNED:           35022.09


----------------OTV DESIGN----------------
```

OTV ENGINE DATA:
Isp: 460
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 2747.697
LOX Tank Mass: 140.0884
Fuel Tank Mass: 1221.57
Pressure Tank Mass: 0

Total Mass: 5139.356

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 35022.09
LOX Carried for OTV: 35022.09
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 7130.879
Fuel Capacity Carried for Lander: 5462.627

Total Propellant Capacity: 42152.97

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .8913279

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 460
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1853.883
LOX Tank Mass: 85.44023
Fuel Tank Mass: 376.7137
Pressure Tank Mass: 0

Total Mass: 3346.037

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity:	21360.06
Fuel Capacity:	3883.647
Total Propellant Capacity:	25243.7
Percent of Lander LOX supplied from LSB:	100
Percent of Lander Fuel supplied from LSB:	0
Payload to LSB:	15873
Liftoff Payload:	8509.004
Tank Structure for Refueling OTV:	85.09004
Mass Fraction:	.8829637

This data has been stored in a file called: A5LI2.DAT

H / O OTV & LANDER

Isp = 490 sec

NO LUNAR PROPELLANT AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:A5HI1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:490,490
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:               490
Number of engines:              2
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:             5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:                 3381.473
LOX Tank Mass:                  340.266
Fuel Tank Mass:                 1500.263
Pressure Tank Mass:             0

Total Mass:    6252.002


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 85066.49
LOX Carried for OTV: 65059.87
LOX Carried for Lander: 20006.62
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 11829.07
Fuel Capacity Carried for Lander: 3637.567

Total Propellant Capacity: 76888.94

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9248023

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 490
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1853.878
LOX Tank Mass: 80.02649
Fuel Tank Mass: 352.8439
Pressure Tank Mass: 0

Total Mass: 3316.749

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20006.62
Fuel Capacity: 3637.567

Total Propellant Capacity: 23644.19

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 15873
Liftoff Payload: 10137.77

Mass Fraction: .8769795

This data has been stored in a file called: A5HI1.DAT

H / O OTV & LANDER

Isp = 490 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
        RUN
D-50    Do you wish to create a data file? [Y]:Y
        What do you wish to call the data file:A5HI2.DAT
        Do you wish to use an aerobrake? [Y]:Y
        Enter the aerobrake mass percent:15
        Do you wish to use 2 separate vehicles? [Y]:Y
        Enter Isp for OTV and for Lander:490,490
        Enter OTV mass kg (NOT including tanks):840
        Enter Lander mass kg (NOT including tanks and landing gear):
        840
        Enter the maximum payload for OTV & for Lander:15873,15873
        Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
        Enter the number of engines & engine thrust for the OTV:2,33361
        Enter mass for each engine & its thrust structure for OTV:95
        Enter the number of engines & engine thrust for the Lander:2,33361
        Enter mass for each engine & its thrust structure for Lander:95
        A) LOX - HYDROGEN
        B) LOX - ALUMINUM
        C) LOX - MMH
        D) LOX - SILANE
        E) LOX - ALUMINIZED HYDROGEN
        Choose the type of engine to be used for the OTV & Lander:A,A


        Do you wish to use lunar propellants? [Y]:Y
        Enter the percent of fuel & of oxidizer from Moon for Lander:
        0,100
        Enter the percent of fuel & of oxidizer from Moon for OTV:
        0,100



        Should the ammount of Lunar LOX returned be the driving factor
        for the vehicle design? (Y/N):N
        Enter the maximum number of engines allowed for the Lander:2

        The Lander liftoff payload capability is:  10080.95
        Do you wish to change the engine constraint to allow a
        larger payload? (Y/N):N
        ==============================================



        This is a two vehicle configuration which uses Lunar propellants.
        The OTV travels to LLO carrying a payload and propellant
        for the Lander.  The Lander makes 1 round-trip(s) from LSB to
        LLO.  It carries the OTV payload to LSB and delivers Lunar
        propellant to the OTV.  After 1 Lander trip(s), the OTV
        departs for LEO, loaded with Lunar propellants.

        LUNAR LOX LOADED ONTO OTV AT LSB:           9980.144
        LUNAR LOX USED BY OTV:        2476.671
        LUNAR FUEL USED BY OTV :      0
        LUNAR LOX RETURNED=           7503.474
        LEO-BASED LOX BURNED:         27733.43


        -----------------OTV DESIGN-----------------


        OTV ENGINE DATA:
        Isp:            490
        Number of engines:            2
        Thrust per engine (N):        33361
        Mass of each engine & its thrust structure (kg):          95
```

LOX - Hydrogen engine with MR: 5.5

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 1737.292
LOX Tank Mass: 110.9337
Fuel Tank Mass: 886.4729
Pressure Tank Mass: 0

Total Mass: 3764.698

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 27733.43
LOX Carried for OTV: 27733.43
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 5492.746
Fuel Capacity Carried for Lander: 3646.151

Total Propellant Capacity: 33226.18

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .8982262

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 490
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1853.879
LOX Tank Mass: 80.21533
Fuel Tank Mass: 353.6766
Pressure Tank Mass: 0

Total Mass: 3317.771

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20053.83
Fuel Capacity: 3646.151

Total Propellant Capacity: 23699.98

Percent of Lander LOX supplied from LSB:		100	
Percent of Lander Fuel supplied from LSB:			0
Payload to LSB:	15873		
Liftoff Payload:	10080.95		
Tank Structure for Refueling OTV:		100.8095	
Mass Fraction:	.8772003		

This data has been stored in a file called: A5HI2.DAT

H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

10 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:E1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:10000,10000
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:             5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:               2253.996
LOX Tank Mass:                268.4688
Fuel Tank Mass:               1183.703
Pressure Tank Mass:           0

Total Mass:     4736.168


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:           67117.19
LOX Carried for OTV:          50765.56
```

LOX Carried for Lander: 16351.63
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 9230.102
Fuel Capacity Carried for Lander: 2973.024

Total Propellant Capacity: 59995.66

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 10000
Return Payload Capability: 10000

Mass Fraction: .926834

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1615.264
LOX Tank Mass: 65.40652
Fuel Tank Mass: 288.3833
Pressure Tank Mass: 0

Total Mass: 2999.054

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 16351.63
Fuel Capacity: 2973.024

Total Propellant Capacity: 19324.65

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 10000
Liftoff Payload: 10000

Mass Fraction: .8656561

This data has been stored in a file called: E1.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

10 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:E2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:10000,10000
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  10000
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          9900
LUNAR LOX USED BY OTV:        2508.797
LUNAR FUEL USED BY OTV :      0
LUNAR LOX RETURNED=           7391.203
LEO-BASED LOX BURNED:         21085.68


-----------------OTV DESIGN----------------


OTV ENGINE DATA:
Isp:           470
Number of engines:            2
Thrust per engine (N):        33361
```

Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

D-58

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 1681.109
LOX Tank Mass: 84.34272
Fuel Tank Mass: 706.0241
Pressure Tank Mass: 0

Total Mass: 3501.476

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 21085.68
LOX Carried for OTV: 21085.68
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 4289.905
Fuel Capacity Carried for Lander: 2988.693

Total Propellant Capacity: 25375.59

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 10000
Return Payload Capability: 10000

Mass Fraction: .8787454

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1620.72
LOX Tank Mass: 65.75126
Fuel Tank Mass: 289.9033
Pressure Tank Mass: 0

Total Mass: 3006.374

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 16437.82
Fuel Capacity: 2988.693

Total Propellant Capacity: 19426.51

Percent of Lander LOX supplied from LSB:	100	
Percent of Lander Fuel supplied from LSB:		0

Payload to LSB:	10000
Liftoff Payload:	10000
Tank Structure for Refueling OTV:	100

Mass Fraction: .8659836

This data has been stored in a file called: E2.DAT

H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

20 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:F1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840,
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:20000,20000
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:3
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):        95
LOX - Hydrogen engine with MR:            5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:               4277.554
LOX Tank Mass:                491.4358
Fuel Tank Mass:               2166.785
Pressure Tank Mass:           0

Total Mass:     7965.775


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity. 122859
LOX Carried for OTV: 93654.44
LOX Carried for Lander: 29204.52
D-62 Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 17028.08
Fuel Capacity Carried for Lander: 5309.912

Total Propellant Capacity: 110682.5

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload to LSB: 20000
Return Payload Capability: 20000

Mass Fraction: .9328623

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1125
Landing Gear Mass: 2780.818
LOX Tank Mass: 116.8181
Fuel Tank Mass: 515.0615
Pressure Tank Mass: 0

Total Mass: 4537.698

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 29204.52
Fuel Capacity: 5309.912

Total Propellant Capacity: 34514.43

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 20000
Liftoff Payload: 16595.92

Mass Fraction: .8838041

This data has been stored in a file called: F1.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

20 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:F2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:20000,20000
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:3

The Lander liftoff payload capability is:  16497.96
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:            16332.98
LUNAR LOX USED BY OTV:          4022.366
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             12310.61
LEO-BASED LOX BURNED:           38833.12


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
```

Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 2696.027
LOX Tank Mass: 155.3325
Fuel Tank Mass: 1272.312
Pressure Tank Mass: 0

Total Mass: 5153.672

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 38833.12
LOX Carried for OTV: 38833.12
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 7791.907
Fuel Capacity Carried for Lander: 5324.711

Total Propellant Capacity: 46625.03

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 20000
Return Payload Capability: 20000

Mass Fraction: .9004673

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1125
Landing Gear Mass: 2780.818
LOX Tank Mass: 117.1437
Fuel Tank Mass: 516.497
Pressure Tank Mass: 0

Total Mass: 4539.459

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 29285.92
Fuel Capacity: 5324.711

Total Propellant Capacity: 34610.63

Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB: 20000
Liftoff Payload: 16497.96
Tank Structure for Refueling OTV: 164.9796

Mass Fraction: .8840498

This data has been stored in a file called: F2.DAT

H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

NO AEROBRAKE

```
C:\basprog>RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:NOBRK.DAT
Do you wish to use an aerobrake? [Y]:N
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:             470
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:             5.5


OTV MASS (kg):

Dry Mass:        1030
Aerobrake Mass:             0
LOX Tank Mass:              548.5174
Fuel Tank Mass:             2418.463
Pressure Tank Mass:         0

Total Mass:      3996.981
```

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 137129.4
LOX Carried for OTV: 116353.5
LOX Carried for Lander: 20775.86
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 21155.18
Fuel Capacity Carried for Lander: 3777.43

Total Propellant Capacity: 137508.7

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9717539

---------------LANDER DESIGN---------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1841.305
LOX Tank Mass: 83.10345
Fuel Tank Mass: 366.4107
Pressure Tank Mass: 0

Total Mass: 3320.819

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20775.86
Fuel Capacity: 3777.43

Total Propellant Capacity: 24553.29

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 8973
Liftoff Payload Capability: 8973

Mass Fraction: .8808637

This data has been stored in a file called: NOBRK.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

NO AEROBRAKE

```
LIST RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:NOBRK2.DAT
Do you wish to use an aerobrake? [Y]:N
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 3 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 3 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          26649.81
LUNAR LOX USED BY OTV:          16036.96
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             10612.85
LEO-BASED LOX BURNED:           35436.17


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              470
Number of engines:              2
Thrust per engine (N):     ,    33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:        1030
Aerobrake Mass:                 0
LOX Tank Mass:                  141.7447
Fuel Tank Mass:                 1580.146
Pressure Tank Mass:             0

Total Mass:     2751.89
```

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OTV PROPELLANT CAPACITY (kg):

```
Total LOX Capacity:          35436.17
LOX Carried for OTV:         35436.17
LOX Carried for Lander:      0
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:       9358.751
Fuel Capacity Carried for Lander:         6931.408

Total Propellant Capacity:   44794.92

Percent of return trip LOX from LSB:      100
Percent of return trip Fuel from LSB:     0

Payload to LSB:              15873
Return Payload Capability:   15873

Mass Fraction:               .9421224
```

---------------LANDER DESIGN---------------

LANDER ENGINE DATA:

```
Isp:          470
Number of engines:           2
Thrust per engine (N):       33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:            5.5
```

LANDER MASS (kg):

```
Dry Mass:      1030
Landing Gear Mass:           1846.201
LOX Tank Mass:               83.4128
Fuel Tank Mass:              367.7746
Pressure Tank Mass:          0

Total Mass:    3327.388
```

LANDER PROPELLANT CAPACITY (kg):

```
LOX Capacity:                20853.2
Fuel Capacity:               3791.49

Total Propellant Capacity:   24644.69

Percent of Lander LOX supplied from LSB:   100

Percent of Lander Fuel supplied from LSB:              0

Payload to LSB:              15873
Liftoff Payload:             8973
Tank Structure for Refueling OTV:         89.72999

Mass Fraction:               .8810461
```

This data has been stored in a file called: NOBRK2.DAT
Ok

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H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

18% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:18BRK.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:18
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840,
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):         95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:               4254.073
LOX Tank Mass:                373.4099
Fuel Tank Mass:               1646.398
Pressure Tank Mass:           0

Total Mass:     7303.881


OTV PROPELLANT CAPACITY  (kg):
```

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Total LOX Capacity: 93352.47
LOX Carried for OTV: 72576.61
LOX Carried for Lander: 20775.86
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 13195.75
Fuel Capacity Carried for Lander: 3777.43

Total Propellant Capacity: 85772.36

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9215281

---------------LANDER DESIGN---------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1841.305
LOX Tank Mass: 83.10345
Fuel Tank Mass: 366.4107
Pressure Tank Mass: 0

Total Mass: 3320.819

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20775.86
Fuel Capacity: 3777.43

Total Propellant Capacity: 24553.29

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 8973
Liftoff Payload Capability: 8973

Mass Fraction: .8808637

This data has been stored in a file called: 18BRK.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

18% AEROBRAKE

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```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:18BRK2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:18
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:1030470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17766.54
LUNAR LOX USED BY OTV:        4440.367
LUNAR FUEL USED BY OTV :      0
LUNAR LOX RETURNED=           13326.17
LEO-BASED LOX BURNED:         34832.88


----------------OTV DESIGN----------------


OTV ENGINE DATA:
Isp:           470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:                5.5


OTV MASS  (kg):

Dry Mass:      1030
Aerobrake Mass:               3570.955
```

```
LOX Tank Mass:                139.3315
Fuel Tank Mass:               1212.698
Pressure Tank Mass:           0

Total Mass:     5952.985


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:           34832.88
LOX Carried for OTV:          34832.88
LOX Carried for Lander:       0
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:        7140.589
Fuel Capacity Carried for Lander:         5361.449

Total Propellant Capacity:    41973.47

Percent of return trip LOX from LSB:      100
Percent of return trip Fuel from LSB:     0

Payload Capability to LSB:    15873
Return Payload Capability:    15873

Mass Fraction:                .8757891



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:            470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:            5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:            1846.201
LOX Tank Mass:                83.4128
Fuel Tank Mass:               367.7746
Pressure Tank Mass:           0

Total Mass:     3327.388


LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                 20853.2
Fuel Capacity:                3791.49

Total Propellant Capacity:    24644.69

Percent of Lander LOX supplied from LSB:   100
Percent of Lander Fuel supplied from LSB:             0

Maximum Payload Capability:                15873
Liftoff Payload Capability:                8973
Tank Structure for Refueling OTV:          89.72999

Mass Fraction:                .8810461


This data has been stored in a file called:            1SERK2.DAT
Ok
```

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H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

20% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:20BRK.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:20
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:92,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              470
Number of engines:           2
Thrust per engine (N):       33361
Mass of each engine & its thrust structure (kg):         95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:              4854.119
LOX Tank Mass:               377.9398
Fuel Tank Mass:              1666.371
Pressure Tank Mass:          0

Total Mass:     7928.43


OTV PROPELLANT CAPACITY  (kg):
```

```
Total LOX Capacity:             94484.96
LOX Carried for OTV:            73709.1
LOX Carried for Lander:         20775.86
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:          13401.65
  Fuel Capacity Carried for Lander:          37743

  Total Propellant Capacity:   87110.75

  Percent of return trip LOX from LSB:       0
  Percent of return trip Fuel from LSB:      0

  Payload Capability to LSB:   15873
  Return Payload Capability:   15873

  Mass Fraction:               .9165772


  ---------------LANDER DESIGN---------------

  LANDER ENGINE DATA:

  Isp:            470
  Number of engines:            2
  Thrust per engine (N):        33361
  Mass of each engine & its thrust structure (kg):        95
  LOX - Hydrogen engine with MR:             5.5

  LANDER MASS  (kg):

  Dry Mass:       1030
  Landing Gear Mass:            1841.305
  LOX Tank Mass:                83.10345
  Fuel Tank Mass:               366.4107
  Pressure Tank Mass:           0

  Total Mass:     3320.819

  LANDER PROPELLANT CAPACITY  (kg):

  LOX Capacity:                 20775.86
  Fuel Capacity:                3777.43

  Total Propellant Capacity:   24553.29

  Percent of Lander LOX supplied from LSB:   0
  Percent of Lander Fuel supplied from LSB:              0

  Maximum Payload Capability:               8973
  Liftoff Payload Capability:               8973

  Mass Fraction:               .8808637


  This data has been stored in a file called:            20BRK.DAT
```

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

20% AEROBRAKE

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```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:20BRK2.FDAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:20
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
=============================================


This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17766.54
LUNAR LOX USED BY OTV:          4532.855
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             13233.68
LEO-BASED LOX BURNED:           35468.53

----------------OTV DESIGN----------------

OTV ENGINE DATA:
Isp:          470
Number of engines:             2
Thrust per engine (N):         33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:      1030
Aerobrake Mass:                4049.625
```

```
LOX Tank Mass:                 141.8741
Fuel Tank Mass:                1225.54
Pressure Tank Mass:            0

Total Mass:     6447.038


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:            35468.53
LOX Carried for OTV:           35468.53
LOX Carried for Lander:        0
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:         7272.979
Fuel Capacity Carried for Lander:            5361.449

Total Propellant Capacity:     42741.51

Percent of return trip LOX from LSB:         100
Percent of return trip Fuel from LSB:        0

Payload Capability to LSB:     15873
Return Payload Capability:     15873

Mass Fraction:                 .8689321



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:            470
Number of engines:             2
Thrust per engine (N):         33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:              5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:             1846.201
LOX Tank Mass:                 83.4128
Fuel Tank Mass:                367.7746
Pressure Tank Mass:            0

Total Mass:     3327.388


LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                  20853.2
Fuel Capacity:                 3791.49

Total Propellant Capacity:     24644.69

Percent of Lander LOX supplied from LSB:    100
Percent of Lander Fuel supplied from LSB:            0

Maximum Payload Capability:                 15873
Liftoff Payload Capability:                 8973
Tank Structure for Refueling OTV:           89.72999

Mass Fraction:                 .8810461


This data has been stored in a file called:         20BRK2.DAT
Ok
```

ORIGINAL PAGE IS
OF POOR QUALITY

H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

25% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:25BRK.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:25
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT_including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:,AA,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            470
Number of engines:          2
Thrust per engine (N):      33361
Mass of each engine & its thrust structure (kg):        95
LOX - Hydrogen engine with MR:              5.5


OTV MASS  (kg):

Dry Mass:      1030
Aerobrake Mass:             6505.939
LOX Tank Mass:              390.4098
Fuel Tank Mass:             1721.352
Pressure Tank Mass:         0

Total Mass:    9647.701


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 97602.46
LOX Carried for OTV: 76826.6
LOX Carried for Lander: 20775.86
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 13968.47
Fuel Capacity Carried for Lander: 3777.43

Total Propellant Capacity: 90795.06

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9039482

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1841.305
LOX Tank Mass: 83.10345
Fuel Tank Mass: 366.4107
Pressure Tank Mass: 0

Total Mass: 3320.819

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20775.86
Fuel Capacity: 3777.43

Total Propellant Capacity: 24553.29

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 8973
Liftoff Payload Capability: 8973

Mass Fraction: .8808637

This data has been stored in a file called: 25BRK.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

25% AEROBRAKE

ORIGINAL PAGE IS
OF POOR QUALITY


```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:25BRK2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:25
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
=============================================



This is a two vehicle configuration which uses Lunar propellents.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17766.54
LUNAR LOX USED BY OTV:          4781.688
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             12984.85
LEO-BASED LOX BURNED:           37178.72

-----------------OTV DESIGN-----------------

OTV ENGINE DATA:
Isp:            470
Number of engines:              2
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:                  5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:                 5337.451
```

```
LOX Tank Mass:                148.7149
Fuel Tank Mass:               1260.09
Pressure Tank Mass:           0

Total Mass:     7776.255


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:           37178.72
LOX Carried for OTV:          37178.72
LOX Carried for Lander:       0
Additional LOX Storage Capability for Return Trip:        0
Fuel Capacity for OTV:        7629.165
Fuel Capacity Carried for Lander:           5361.449

Total Propellant Capacity:    44807.89

Percent of return trip LOX from LSB:        100
Percent of return trip Fuel from LSB:       0

Payload Capability to LSB:    15873
Return Payload Capability:    15873

Mass Fraction:                .8521179



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:            470
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):         95
LOX - Hydrogen engine with MR:              5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:            1846.201
LOX Tank Mass:                83.4128
Fuel Tank Mass:               367.7746
Pressure Tank Mass:           0

Total Mass:     3327.388


LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                 20853.2
Fuel Capacity:                3791.49

Total Propellant Capacity:    24644.69

Percent of Lander LOX supplied from LSB:    100
Percent of Lander Fuel supplied from LSB:              0

Maximum Payload Capability:              15873
Liftoff Payload Capability:              8973
Tank Structure for Refueling OTV:        89.72999

Mass Fraction:                .8810461


This data has been stored in a file called:            25BRK2.DAT
Ok
```

H / O OTV & LANDER

Isp = 470 sec

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

30% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:30BRK.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:30
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,8973
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:2
==============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              470
Number of engines:           2
Thrust per engine (N):       33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:             5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:              8414.974
LOX Tank Mass:               404.8216
Fuel Tank Mass:              1784.895
Pressure Tank Mass:          0

Total Mass:     11634.69


OTV PROPELLANT CAPACITY  (kg):
```

Total LOX Capacity: 101205.4
LOX Carried for OTV: 80429.53
LOX Carried for Lander: 20775.86
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 14623.55
Fuel Capacity Carried for Lander: 3777.43

Total Propellant Capacity: 95053.08

Percent of return trip LOX from LSB: 0
Percent of return trip Fuel from LSB: 0

Payload Capability to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .8909463

---------------LANDER DESIGN---------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030
Landing Gear Mass: 1841.305
LOX Tank Mass: 83.10345
Fuel Tank Mass: 366.4107
Pressure Tank Mass: 0

Total Mass: 3320.819

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 20775.86
Fuel Capacity: 3777.43

Total Propellant Capacity: 24553.29

Percent of Lander LOX supplied from LSB: 0
Percent of Lander Fuel supplied from LSB: 0

Maximum Payload Capability: 8973
Liftoff Payload Capability: 8973

Mass Fraction: .8808637

This data has been stored in a file called: 30BRK.DAT

H / O OTV & LANDER

Isp = 470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

30% AEROBRAKE

ORIGINAL PAGE IS OF POOR QUALITY


```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:30BRK2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:30
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):1030
Enter Lander mass kg (NOT including tanks and landing gear):
1030
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
Choose the type of engine to be used for the OTV and Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  8973
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
The Lander does not have the lift capability to return
a manned capsule and the propellant needed for the OTV.
Enter a 0 if you wish to increase the number of Lander
trips.  Enter a 1 if you wish to increase the number of
Lander engines.0
===========================================


This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 2 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 2 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:          17766.54
LUNAR LOX USED BY OTV:          5059.157
LUNAR FUEL USED BY OTV :        0
LUNAR LOX RETURNED=             12707.38
LEO-BASED LOX BURNED:           39085.72

-----------------OTV DESIGN-----------------

OTV ENGINE DATA:
Isp:          470
Number of engines:              2
Thrust per engine (N):          33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:             5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:                 6773.479
```

```
LOX Tank Mass:                 156.3429
Fuel Tank Mass:                1298.616
Pressure Tank Mass:            0

Total Mass:     9258.438


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:            39085.72
LOX Carried for OTV:           39085.72
LOX Carried for Lander:        0
Additional LOX Storage Capability for Return Trip:       0
Fuel Capacity for OTV:         8026.342
Fuel Capacity Carried for Lander:          5361.449

Total Propellant Capacity:   47112.07

Percent of return trip LOX from LSB:       100
Percent of return trip Fuel from LSB:      0

Payload Capability to LSB:   15873
Return Payload Capability:   15873

Mass Fraction:               .8357574



---------------LANDER DESIGN---------------


LANDER ENGINE DATA:

Isp:            470
Number of engines:           2
Thrust per engine (N):       33361
Mass of each engine & its thrust structure (kg):        95
LOX - Hydrogen engine with MR:           5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:           1846.201
LOX Tank Mass:               83.4128
Fuel Tank Mass:              367.7746
Pressure Tank Mass:          0

Total Mass:     3327.388

LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                20853.2
Fuel Capacity:               3791.49

Total Propellant Capacity:   24644.69

Percent of Lander LOX supplied from LSB:   100
Percent of Lander Fuel supplied from LSB:          0

Maximum Payload Capability:          15873
Liftoff Payload Capability:          8973
Tank Structure for Refueling OTV:    89.72999

Mass Fraction:               .8810461


This data has been stored in a file called:          30BRK2.DAT
Ok
```

SILANE / O OTV & LANDER

Isp =366 sec

O / F = 0.78

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:SILANE1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:366,366
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
8340
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:.78,.78
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:100
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:100
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:D,D


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:3
==============================================
```

This is a two vehicle configuration which does not use Lunar propellants. The OTV travels from LEO to LLO, carrying a payload and all of the propellant needed by the Lander. The Lander makes one round trip from LSB to LLO, carrying the OTV payload to LSB and delivering a payload from LSB to the OTV. The OTV then returns to LEO.

```
-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              366
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):          100
LOX - Silane engine with MR:                   .78


OTV MASS  (kg):

Dry Mass:         1040
Aerobrake Mass:               3345.651
LOX Tank Mass:                355.1737
Fuel Tank Mass:               1138.377
Pressure Tank Mass:           0

Total Mass:       5879.202


OTV PROPELLANT CAPACITY  (kg):
```

```
Total LOX Capacity:          88793.42
LOX Carried for OTV:         70998.24
LOX Carried for Lander:      17795.19
Additional LOX Storage Capability for Return Trip:       0
Fuel Capacity for OTV:       91023.39
Fuel Capacity Carried for Lander:          22814.35

Total Propellant Capacity:   162021.6

Percent of return trip LOX from LSB:       0
Percent of return trip Fuel from LSB:      0

Payload to LSB:              15873
Return Payload Capability:   15873

Mass Fraction:               .9649841



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:            366
Number of engines:           3
Thrust per engine (N):       33361
Mass of each engine & its thrust structure (kg):        100
LOX - Silane engine with MR:               .78


LANDER MASS  (kg):

Dry Mass:       1140
Landing Gear Mass:           2780.538
LOX Tank Mass:               71.18077
Fuel Tank Mass:              228.1435
Pressure Tank Mass:          0

Total Mass:     4219.862


LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                17795.19
Fuel Capacity:               22814.35

Total Propellant Capacity:   40609.54

Percent of Lander LOX supplied from LSB:   0
Percent of Lander Fuel supplied from LSB:               0

Payload to LSB:              15873
Liftoff Payload:             10815.26

Mass Fraction:               .9058684


This data has been stored in a file called:          SILANE1.DAT
```

SILANE / O OTV & LANDER

Isp =366 sec

O / F = 0.78

LLOX AND LUNAR Si AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

ORIGINAL PAGE IS
OF POOR QUALITY


```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:SILANE2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:366,366
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,215873
Enter the O/F mixture ratio for the OTV & for the Lander:.78,.78
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:9100
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:100
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:D,D


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
87.5,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
87.5,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:3

The Lander liftoff payload capability is:  10729.06
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
==============================================


This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:         8762.836
LUNAR LOX USED BY OTV:         1657.103
LUNAR FUEL USED BY OTV :       1858.93
LUNAR LOX RETURNED=            7105.734
LEO-BASED LOX BURNED:          21471.91


----------------OTV DESIGN----------------


OTV ENGINE DATA:
```

Isp:	366
Number of engines:	2
Thrust per engine (N):	33361
Mass of each engine & its thrust structure (kg):	100
LOX - Silane engine with MR:	.78

OTV MASS (kg):

Dry Mass:	1040
Aerobrake Mass:	1900.469
LOX Tank Mass:	85.88765
Fuel Tank Mass:	306.5141
Pressure Tank Mass:	0
Total Mass:	3332.871

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity:	21471.91
LOX Carried for OTV:	21471.91
LOX Carried for Lander:	0
Additional LOX Storage Capability for Return Trip:	0
Fuel Capacity for OTV:	27793.66
Fuel Capacity Carried for Lander:	2857.757
Total Propellant Capacity:	49265.57
Percent of return trip LOX from LSB:	100
Percent of return trip Fuel from LSB:	87.5
Payload to LSB:	15873
Return Payload Capability:	15873
Mass Fraction:	.9366356

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp:	366
Number of engines:	3
Thrust per engine (N):	33361
Mass of each engine & its thrust structure (kg):	100
LOX - Silane engine with MR:	.78

LANDER MASS (kg):

Dry Mass:	1140
Landing Gear Mass:	2780.453
LOX Tank Mass:	71.32962
Fuel Tank Mass:	228.6206
Pressure Tank Mass:	0
Total Mass:	4220.403

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity:	17832.4

Fuel Capacity: 22862.06

Total Propellant Capacity: 40694.46

Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 87.5

Payload to LSB: 15873
Liftoff Payload: 10729.06
Tank Structure for Refueling OTV: 107.2906

Mass Fraction: .9060355

This data has been stored in a file called: SILANE2.DAT

Al - H2 / O OTV & LANDER

Isp =400 sec

O / F = 3.1

NO LUNAR PROPELLANTS AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:ALUMH1.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:400,400
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:3.1,3.1
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:140
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:140
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:E,E


Do you wish to use lunar propellants? [Y]:N
Enter the maximum number of engines allowed for Lander:3
=============================================



This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:             400
Number of engines:            2
Thrust per engine (N):        33361
Mass of each engine & its thrust structure (kg):            140
LOX - Aluminized Hydrogen engine with MR:  3.1


OTV MASS  (kg):

Dry Mass:       1120
Aerobrake Mass:               3645.597
LOX Tank Mass:                533.3873
Fuel Tank Mass:               2580.907
Pressure Tank Mass:           0

Total Mass:     7879.891
```

OTV PROPELLANT CAPACITY (kg):

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Total LOX Capacity:	133346.8
LOX Carried for OTV:	104897.9
LOX Carried for Lander:	28448.92
Additional LOX Storage Capability for Return Trip:	0
Fuel Capacity for OTV:	33838.04
Fuel Capacity Carried for Lander:	9177.072
Total Propellant Capacity:	138735.9
Percent of return trip LOX from LSB:	0
Percent of return trip Fuel from LSB:	0
Payload to LSB:	15873
Return Payload Capability:	15873
Mass Fraction:	.9462549

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp:	400
Number of engines:	3
Thrust per engine (N):	33361
Mass of each engine & its thrust structure (kg):	140
LOX - Aluminized Hydrogen engine with MR:	3.1

LANDER MASS (kg):

Dry Mass:	1260
Landing Gear Mass:	2780.802
LOX Tank Mass:	113.7957
Fuel Tank Mass:	550.6243
Pressure Tank Mass:	0
Total Mass:	4705.222

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity:	28448.92
Fuel Capacity:	9177.072
Total Propellant Capacity:	37625.99
Percent of Lander LOX supplied from LSB:	0
Percent of Lander Fuel supplied from LSB:	0
Payload to LSB:	15873
Liftoff Payload:	13316.41
Mass Fraction:	.8888475

This data has been stored in a file called: ALUMH1.DAT

Al - H2 / O OTV & LANDER

Isp =400 sec

O / F = 3.1

LLOX AND LUNAR Al AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:ALUMH2.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:400,400
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:3.1,3.1
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:140
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:140
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:E,E


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
40,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
40,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):N
Enter the maximum number of engines allowed for the Lander:3

The Lander liftoff payload capability is:  13219.93
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
==============================================



This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 1 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 1 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.

LUNAR LOX LOADED ONTO OTV AT LSB:           12646.93
LUNAR LOX USED BY OTV:          3416.214
LUNAR FUEL USED BY OTV :        440.8018
LUNAR LOX RETURNED=             9230.715
LEO-BASED LOX BURNED:           38157.57


-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:            400
```

Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 140
LOX - Aluminized Hydrogen engine with MR: 3.1

OTV MASS (kg):

Dry Mass: 1120
Aerobrake Mass: 2207.01
LOX Tank Mass: 152.6303
Fuel Tank Mass: 1109.412
Pressure Tank Mass: 0

Total Mass: 4589.052

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 38157.57
LOX Carried for OTV: 38157.57
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 0
Fuel Capacity for OTV: 12970.1
Fuel Capacity Carried for Lander: 5520.098

Total Propellant Capacity: 51127.67

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 40

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .9176359

---------------LANDER DESIGN----------------

LANDER ENGINE DATA:

Isp: 400
Number of engines: 3
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 140
LOX - Aluminized Hydrogen engine with MR: 3.1

LANDER MASS (kg):

Dry Mass: 1260
Landing Gear Mass: 2780.794
LOX Tank Mass: 114.082
Fuel Tank Mass: 552.0097
Pressure Tank Mass: 0

Total Mass: 4706.886

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity: 28520.5

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Fuel Capacity:	9200.162
Total Propellant Capacity:	37720.67
Percent of Lander LOX supplied from LSB:	100
Percent of Lander Fuel supplied from LSB:	40
Payload to LSB:	15873
Liftoff Payload:	13219.93
Tank Structure for Refueling OTV:	132.1993
Mass Fraction:	.8890606

This data has been stored in a file called: ALUMH2.DAT

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H / O OTV & LANDER

Isp =470 sec

LLOX AVAILABLE

15.9 MT PAYLOAD

15% AEROBRAKE

```
RUN
Do you wish to create a data file? [Y]:Y
What do you wish to call the data file:]LLLOXR.DAT
Do you wish to use an aerobrake? [Y]:Y
Enter the aerobrake mass percent:15
Do you wish to use 2 separate vehicles? [Y]:Y
Enter Isp for OTV and for Lander:470,470
Enter OTV mass kg (NOT including tanks):840
Enter Lander mass kg (NOT including tanks and landing gear):
840
Enter the maximum payload for OTV & for Lander:15873,15873
Enter the O/F mixture ratio for the OTV & for the Lander:5.5,5.5
Enter the number of engines & engine thrust for the OTV:2,33361
Enter mass for each engine & its thrust structure for OTV:95
Enter the number of engines & engine thrust for the Lander:2,33361
Enter mass for each engine & its thrust structure for Lander:95
A) LOX - HYDROGEN
B) LOX - ALUMINUM
C) LOX - MMH
D) LOX - SILANE
E) LOX - ALUMINIZED HYDROGEN
Choose the type of engine to be used for the OTV & Lander:A,A


Do you wish to use lunar propellants? [Y]:Y
Enter the percent of fuel & of oxidizer from Moon for Lander:
0,100
Enter the percent of fuel & of oxidizer from Moon for OTV:
0,100



Should the ammount of Lunar LOX returned be the driving factor
for the vehicle design? (Y/N):Y
Enter the minimal ammount of Lunar LOX you wish to return as
a percent of the LEO-based LOX which is burned:100
Enter the maximum number of engines allowed for the Lander:2

The Lander liftoff payload capability is:  9056.048
Do you wish to change the engine constraint to allow a
larger payload? (Y/N):N
No. of Lander Trips:        2
No. of Lander Trips:        3
No. of Lander Trips:        4
No. of Lander Trips:        5
No. of Lander Trips:        6
No. of Lander Trips:        7
No. of Lander Trips:        8
No. of Lander Trips:        9
No. of Lander Trips:        10
==============================================


This is a two vehicle configuration which uses Lunar propellants.
The OTV travels to LLO carrying a payload and propellant
for the Lander.  The Lander makes 10 round-trip(s) from LSB to
LLO.  It carries the OTV payload to LSB and delivers Lunar
propellant to the OTV.  After 10 Lander trip(s), the OTV
departs for LEO, loaded with Lunar propellants.
```

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LUNAR LOX LOADED ONTO OTV AT LSB: 89654.88
LUNAR LOX USED BY OTV: 20190.58
LUNAR FUEL USED BY OTV : 0
LUNAR LOX RETURNED= 69464.3
LEO-BASED LOX BURNED: 69379.13

-----------------OTV DESIGN-----------------

OTV ENGINE DATA:
Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

OTV MASS (kg):

Dry Mass: 1030
Aerobrake Mass: 13544.54
LOX Tank Mass: 297.2478
Fuel Tank Mass: 3327.25
Pressure Tank Mass: 0

Total Mass: 18199.04

OTV PROPELLANT CAPACITY (kg):

Total LOX Capacity: 74311.93
LOX Carried for OTV: 69379.13
LOX Carried for Lander: 0
Additional LOX Storage Capability for Return Trip: 4932.797
Fuel Capacity for OTV: 16285.4
Fuel Capacity Carried for Lander: 18016.15

Total Propellant Capacity: 85664.53

Percent of return trip LOX from LSB: 100
Percent of return trip Fuel from LSB: 0

Payload to LSB: 15873
Return Payload Capability: 15873

Mass Fraction: .8247794

---------------LANDER DESIGN---------------

LANDER ENGINE DATA:

Isp: 470
Number of engines: 2
Thrust per engine (N): 33361
Mass of each engine & its thrust structure (kg): 95
LOX - Hydrogen engine with MR: 5.5

LANDER MASS (kg):

Dry Mass: 1030

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Landing Gear Mass:	1853.881
LOX Tank Mass:	83.62193
Fuel Tank Mass:	368.6966
Pressure Tank Mass:	0

Total Mass: 3336.2

LANDER PROPELLANT CAPACITY (kg):

LOX Capacity:	20905.48
Fuel Capacity:	3800.996

Total Propellant Capacity: 24706.48

Percent of Lander LOX supplied from LSB: 100
Percent of Lander Fuel supplied from LSB: 0

Payload to LSB:	15873
Liftoff Payload:	9056.048
Tank Structure for Refueling OTV:	90.56048

Mass Fraction: .8810313

This data has been stored in a file called: LLOXR.DAT

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APPENDIX E

VEHICLE PERFORMANCE AND TRANSPORTATION MODELING

E.1 ASTROSIZE

E.2 ASTROFEST

E.1 ASTROSIZE

A VEHICLE SIZING PROGRAM

DECEMBER 8, 1986

INTRODUCTION

The ASTROSIZE Program is an in-house computer analysis tool developed at Astronautics Corporation of America which allows the user to quickly size a space vehicle and estimate the propellant requirements for a round-trip mission to any destination. When planning a mission, it is often difficult to define the effects of transportation system parameters on the actual size of a vehicle when the vehicle is adapted to the mission. ASTROSIZE allows the user to see the effects of these parameters on the vehicle size and the preliminary propellant requirement estimate. Figure 1 is a flow diagram of the program.

The major inputs, a sample output, as well as future program development plans are discussed in the following pages.

Major Inputs

The major inputs for ASTROSIZE consist of vehicle parameters and engine parameters. The vehicle parameters define the vehicle design, excluding the engines. They include, vehicle subsystem mass, payload capability, tank mass parameters, aerobrake percent of reentry mass and landing gear percent of landing mass. The vehicle subsystems mass is the mass of the vehicle without tanks, engines, and engine thrust structures. The payload capability is simply, the maximum payload which the vehicle can carry. A tank parameter is the mass of the tank divided by the mass of the propellant which the tank holds. This ratio is currently assumed to be linear and the program contains a small data base of values for some commonly used propellants. The user simply specifies the type of propellant to be used and the tank parameter will be assigned accordingly. The aerobrake mass is determined as a percent of the vehicle mass upon reentering the atmosphere.

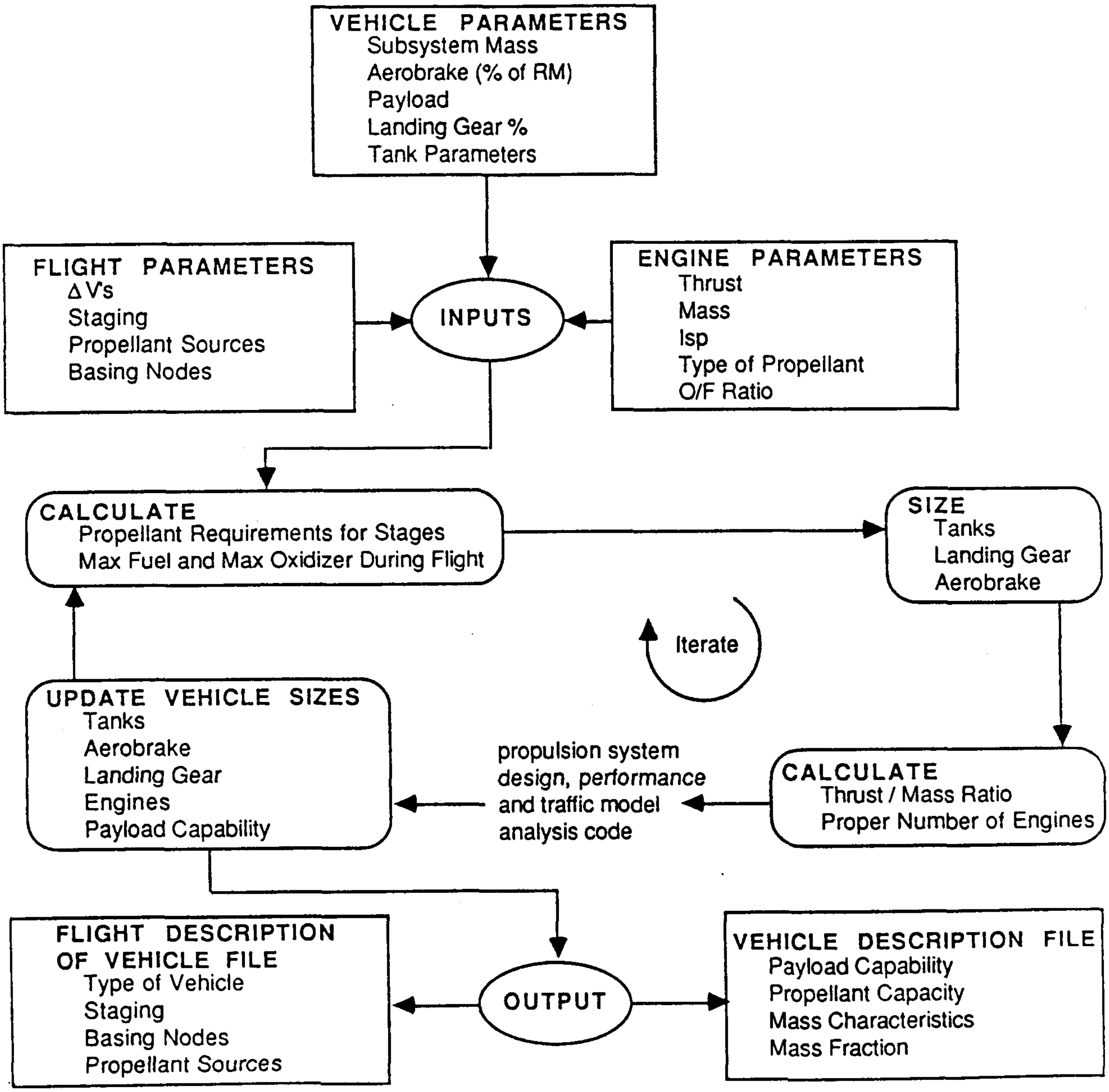

FIGURE 1. Astrosize Flow Diagram

The second major input, engine parameters, include engine thrust, mass, Isp, propellant type and O/F ratio. The engine thrust and mass contribute to determining how many engines will be needed for a given flight. From this information the size of the tanks can be calculated. The user can quickly determine the major design drivers for a vehicle, designed for a lunar mission, by simply varying the input parameter values.

Sample Output

Figure 2. contains a sample output for a round-trip mission to the moon. The mission uses two vehicles, an OTV and a lander. Both of the vehicles have LOX Hydrogen engines and use propellants from the moon. The output begins with a brief description of the mission, describing payload exchanges. This is followed by a list of data concerning lunar propellants which have been used or transported as well as the amount of oxygen needed from LEO.

The vehicle data follows under the headings, OTV Design and Lander Design. Each of these is broken down into 3 categories: engine data, vehicle mass data, and propellant capacity data. The remaining information for each vehicle consists of the percent of fuel and oxidizer taken from the propellant source, the payloads carried, and the mass fraction. This information is then stored in a data file which the user has specified.

Astrosize Operation

ASTROSIZE is run on an IBM PC/XT/AT or compatible. The program is currently written in advanced BASIC but will be converted to FORTRAN, in the near future.

FIGURE 2. Astrosize Sample Output

****

```
This is a two vehicle configuration which does not use
Lunar propellants.  The OTV travels from LEO to LLO, carrying
a payload and all of the propellant needed by the Lander.
The Lander makes one round trip from LSB to LLO, carrying
the OTV payload to LSB and delivering a payload from LSB to
the OTV.  The OTV then returns to LEO.

-----------------OTV DESIGN-----------------


OTV ENGINE DATA:
Isp:              470
Number of engines:               2
Thrust per engine (N):           33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:                 5.5


OTV MASS  (kg):

Dry Mass:       1030
Aerobrake Mass:                  3410.811
LOX Tank Mass:                   367.044
Fuel Tank Mass:                  1618.33
Pressure Tank Mass:              0

Total Mass:     6426.185


OTV PROPELLANT CAPACITY  (kg):

Total LOX Capacity:              91760.99
LOX Carried for OTV:             70985.13
LOX Carried for Lander:          20775.86
Additional LOX Storage Capability for Return Trip:       0
Fuel Capacity for OTV:           12906.39
Fuel Capacity Carried for Lander:              3777.43

Total Propellant Capacity:   83891.52

Percent of return trip LOX from LSB:       0
Percent of return trip Fuel from LSB:      0

Payload Capability to LSB:   15873
Return Payload Capability:   15873

Mass Fraction:               .9288491



---------------LANDER DESIGN----------------


LANDER ENGINE DATA:

Isp:              470
Number of engines:               2
Thrust per engine (N):           33361
Mass of each engine & its thrust structure (kg):          95
LOX - Hydrogen engine with MR:                 5.5


LANDER MASS  (kg):

Dry Mass:       1030
Landing Gear Mass:               1841.305
LOX Tank Mass:                   83.10343
Fuel Tank Mass:                  366.4107
Pressure Tank Mass:              0

Total Mass:     3320.819


LANDER PROPELLANT CAPACITY  (kg):

LOX Capacity:                    20775.86
Fuel Capacity:                   3777.43

Total Propellant Capacity:   24553.29

Percent of Lander LOX supplied from LSB:   0
Percent of Lander Fuel supplied from LSB:              0

Maximum Payload Capability:                8973
Liftoff Payload Capability:                8973

Mass Fraction:                   .8808637
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E.2 ASTROFEST PROGRAM

DECEMBER 2, 1986

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INTRODUCTION

The ASTROFEST Program is an in-house computer analysis tool developed at Astronautics Corporation of America to manifest the flight and resource requirements for long-term, multiple flight, space missions. ASTROFEST allows the user to choose the type of vehicle or vehicles, to be used for the mission, from a set of vehicle data files. These data files can be created by another program, developed at Astronautics, called ASTROSIZE. ASTROFEST considers the propellant requirements from the flights as well as equipment and consumable requirements for the production of lunar propellants, used during the mission. The output lists these requirements for each year as well as the total requirements for the complete mission. ASTROFEST allows the user to identify the most efficient vehicle, and the resource requirements for a multi-flight mission to the moon. Figure 1 is a flow diagram of the program. The major inputs and a sample output of the program are discussed in the following pages.

Major Inputs

The major inputs for ASTROFEST consist of, a list of the mass to be delivered to the moon each year, the name of the vehicle to be used in the mission and the slope and y intercept of a function describing lunar propellant processing. The first input, a data file created by the user, includes three columns. The first column lists the years in which flights will be made to the moon. The second column contains the bulk mass which must be delivered to the moon for the corresponding year. Finally, the third column is the number of manned flights which are planned for the corresponding year. (A manned capsule is assumed to have a mass of 6900 Kg.) The second input, the vehicle name entered by the user, must match with the name of a vehicle data file which was

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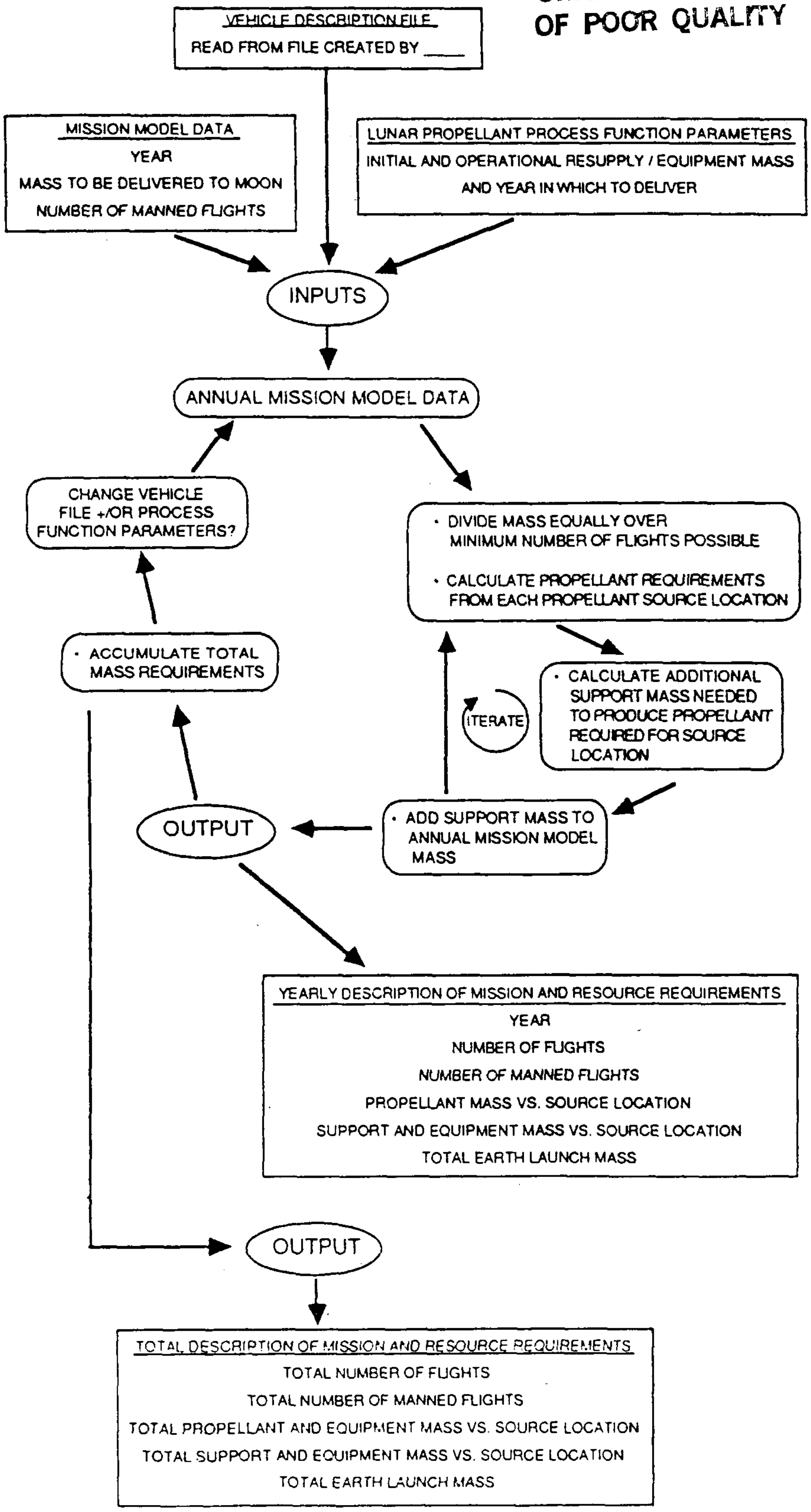

FIGURE 1. Astrofest Flow Diagram

created by another program called ASTROSIZE. ASTROFEST is designed to read the output, which contains vehicle size and performance data, from the program ASTROSIZE. If a vehicle which utilizes lunar propellants is chosen, the user must provide the slope and y intercept of the function which describes resource requirements for lunar propellant processing. The process function is assumed to be linear. The y intercept is the mass of the initial equipment and materials necessary to set up a lunar propellant processing base. The slope is the mass of consumables needed to be replaced per mass of propellant produced. For example, if a LOX Hydrogen OTV and Lander is chosen, and lunar LOX is to be used during the mission, the user must specify the slope and y intercept of a function describing the production of lunar LOX. By using different vehicle types with their corresponding process function, the user can identify the most efficient vehicle to use for the mission and the resource requirements.

Sample Output

Figure 2. contains a sample output for a LOX Hydrogen OTV and Lander which utilizes lunar LOX. The output lists the annual and total values of the lunar propellants used, the Earth propellants used, the number of flights and number of unmanned flights, the mass delivered to the moon, the additional burdened mas and the mass required from the Earth. The additional burdened mass is the mass of equipment and consumables required to produce the lunar propellants which were used. All values, except the number of flights and number of manned flights, are expressed in Kilograms.

ASTROFEST Operation

ASTROFEST is run on an IBM PC/XT/AT or compatible. The program is currently written in advanced BASIC but will be converted to FORTRAN in the near future.

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```
RUN
Enter the name of the data file for the
vehicle you wish to use:BASE.DAT
Enter slope and Y-intercept of process function:0,2300
A) OTV & Lander: Lunar propellants available
B) OTV & Lander: Lunar propellants not available
C) Integrated OTV/Lander: Lunar propellants available
D) Integrated OTV/Lander: Lunar propellants not available
Choose which type of configuration you will use:B
Enter the year which you wish to deliver equipment:2005
Year:           1995

Earth Propellant:                136704.5
Earth LOX:      115673.1
Earth Fuel:     21031.47

Earth Fuel for Lander:           4696.03
Earth Fuel for OTV:              16335.44


Lunar Propellant:0

Total Propellant:                136704.5

Number of Flights:               2
Number of Manned Flights:        0
Mass Delivered (kg):             22700
Additional Burdened Mass (kg):                        0
Mass Required From Earth (kg):                        159404.5


Year:           1996

Earth Propellant:                136704.5
Earth LOX:      115673.1
Earth Fuel:     21031.47

Earth Fuel for Lander:           4696.03
Earth Fuel for OTV:              16335.44


Lunar Propellant:0

Total Propellant:                136704.5

Number of Flights:               2
Number of Manned Flights:        0
Mass Delivered (kg):             22700
Additional Burdened Mass (kg):                        0
Mass Required From Earth (kg):                        159404.5


Year:           1999

Earth Propellant:                56640.75
Earth LOX:      47926.79
Earth Fuel:     8713.959

Earth Fuel for Lander:           1907.151
Earth Fuel for OTV:              6806.808


Lunar Propellant:0

Total Propellant:                56640.75

Number of Flights:               1
Number of Manned Flights:        0
Mass Delivered (kg):             8200
Additional Burdened Mass (kg):                        0
Mass Required From Earth (kg):                        64840.75


Year:           2003

Earth Propellant:                65903.4
Earth LOX:      55764.42
Earth Fuel:     10138.99

Earth Fuel for Lander:           2315.007
Earth Fuel for OTV:              7823.979


Lunar Propellant:0

Total Propellant:                65903.4

Number of Flights:               1
Number of Manned Flights:        1
Mass Delivered (kg):             6800
Additional Burdened Mass (kg):                        0
Mass Required From Earth (kg):                        72703.4


Year:           2004

Earth Propellant:                65903.4
Earth LOX:      55764.42
Earth Fuel:     10138.99
Earth Fuel for Lander:           2315.007
Earth Fuel for OTV:              7823.979


Lunar Propellant:0

Total Propellant:                65903.4

Number of Flights:               1
Number of Manned Flights:        1
Mass Delivered (kg):             6800
Additional Burdened Mass (kg):                        0
Mass Required From Earth (kg):                        72703.4


Year:           2005

Earth Propellant:                450075.1
Earth LOX:      380832.7
Earth Fuel:     69242.31

Earth Fuel for Lander:           15935.74
Earth Fuel for OTV:              53306.58


Lunar Propellant:0

Number of Flights:               5
Number of Manned Flights:        2
Mass Delivered (kg):             78100
Additional Burdened Mass (kg):                        2300
Mass Required From Earth (kg):                        528175.1


Do you wish to use another vehicle after 2005? (Y/N):Y
Enter the name of the data file for the
vehicle you wish to use:A5A.DAT
Enter slope and Y-intercept of process function:.0015,2
A) OTV & Lander: Lunar propellants available
B) OTV & Lander: Lunar propellants not available
C) Integrated OTV/Lander: Lunar propellants available
D) Integrated OTV/Lander: Lunar propellants not availab
Choose which type of configuration you will use:A
Year:           2006

Lunar Propellant:                120493.2
Lunar LOX:      120493.2
Lunar Fuel:     0

Earth Propellant:                116720.4
Earth LOX:      80226.04
Earth Fuel:     36494.4

Number of Flights:               3
Number of Manned Flights:        3
Mass Delivered (kg):             46680.74
Additional Burdened Mass (kg):                        180.7383
Mass Required From Earth (kg):                        163401.2


Year:           2007

Lunar Propellant:                208038.9
Lunar LOX:      208038.9
Lunar Fuel:     0

Earth Propellant:                165853.2
Earth LOX:      108331.3
Earth Fuel:     57521.86

Number of Flights:               5
Number of Manned Flights:        3
Mass Delivered (kg):             69512.06
Additional Burdened Mass (kg):                        [illegible]
Mass Required From Earth (kg):                        [illegible]


Year:           2008

Lunar Propellant:                137207.4
Lunar LOX:      137207.4
Lunar Fuel:     0

Earth Propellant:                [illegible]
Earth LOX:      53009.7
Earth Fuel:     34594.79

Number of Flights:               [illegible]
Number of Manned Flights:        [illegible]
Mass Delivered (kg):             41905.81
Additional Burdened Mass (kg):                        205.8125
```

FIGURE 2. Astrofest Sample Output

OF POOR QUALITY

Mass Required From Earth (kg):		129499.2
Year: 2009		
Lunar Propellant:	453490	
Lunar LOX: 453490		
Lunar Fuel: 0		
Earth Propellant:	409828.3	
Earth LOX: 277010.1		
Earth Fuel: 132818.2		
Number of Flights:	11	
Number of Manned Flights:	9	
Mass Delivered (kg):	168680.2	
Additional Burdened Mass (kg):		680.2344
Mass Required From Earth (kg):		578508.6
Year: 2010		
Lunar Propellant:	474745.7	
Lunar LOX: 474745.7		
Lunar Fuel: 0		
Earth Propellant:	391417.2	
Earth LOX: 258161.4		
Earth Fuel: 133255.8		
Number of Flights:	11	
Number of Manned Flights:	6	
Mass Delivered (kg):	169812.1	
Additional Burdened Mass (kg):		712.125
Mass Required From Earth (kg):		561229.4
Year: 2011		
Lunar Propellant:	240139.7	
Lunar LOX: 240139.7		
Lunar Fuel: 0		
Earth Propellant:	231443	
Earth LOX: 158891.8		
Earth Fuel: 72551.16		
Number of Flights:	6	
Number of Manned Flights:	6	
Mass Delivered (kg):	92260.21	
Additional Burdened Mass (kg):		360.211
Mass Required From Earth (kg):		323703.2
Year: 2013		
Lunar Propellant:	474745.7	
Lunar LOX: 474745.7		
Lunar Fuel: 0		
Earth Propellant:	391417.2	
Earth LOX: 258161.4		
Earth Fuel: 133255.8		
Number of Flights:	11	
Number of Manned Flights:	6	
Mass Delivered (kg):	169812.1	
Additional Burdened Mass (kg):		712.125
Mass Required From Earth (kg):		561229.4
Year: 2014		
Lunar Propellant:	233134.8	
Lunar LOX: 233134.8		
Lunar Fuel: 0		
Earth Propellant:	214915.1	
Earth LOX: 145984.4		
Earth Fuel: 68930.75		
Number of Flights:	6	
Number of Manned Flights:	6	
Mass Delivered (kg):	83149.71	
Additional Burdened Mass (kg):		349.7031
Mass Required From Earth (kg):		298664.5
Year: 2015		
Lunar Propellant:	505373.7	
Lunar LOX: 505373.7		
Lunar Fuel: 0		
Earth Propellant:	349410.7	
Earth LOX: 217905.4		
Earth Fuel: 131505.3		
Number of Flights:	11	
Number of Manned Flights:	1	
Mass Delivered (kg):	165458.1	
Additional Burdened Mass (kg):		758.0625
Mass Required From Earth (kg):		514868.8

==============TOTALS==============

Total Propellant from Moon:	2847369	
Total LOX from Moon:	2847369	
Total Fuel from Moon:	0	
Total Fuel for OTV from Moon:		0
Total Fuel for Lander from Moon:		0
Total Propellant from Earth:		3270530
Total LOX from Earth:	2329315	
Total Fuel from Earth:	941215.2	
Total OTV Fuel from Earth:	558599.3	
Total Lander Fuel from Earth:		382616
Total Propellant:	6117899	
Total Number of Flights:	79	
Total Number of Manned Flights:		44
Total Mass Delivered (kg):	1152571	
Total Additional Burdened Mass (kg):		6571
Total Mass Required From Earth (kg):		4423101

FIGURE 2. Astrofest Sample Output (Continued)

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APPENDIX F

LUNAR MATERIALS COMPOSITION

AND

RELATED THERMODYNAMIC DATA

TABLE F.1

CONSTITUENTS OF LUNAR ORES IN MARE REGOLITH

PYROXENE - 50wt% of Mare Regolith

$CaO \cdot SiO_2$

$MgO \cdot SiO_2$

$FeO \cdot SiO_2$

$Al_2O_3 \cdot SiO_2$

$TiO_2 \cdot SiO_2$

OLIVINE - 15wt% of Mare Regolith

$2MgO \cdot SiO_2$

$2FeO \cdot SiO_2$

PLAGIOCLASE or **ANORTHITE** - 20wt% of Mare Regolith

$CaO \cdot Al_2O_3 \cdot 2SiO_2$

ILMENITE - 15wt% of Mare Regolith

$FeO \cdot TiO_2$

TABLE F.2

OXIDE CONCENTRATIONS (BY WT%) IN MARE REGOLITH

Oxide	Pyroxene	Olivine	Anorthite	Ilmenite	Total
SiO_2	47.8	37.4	46.1	0	44
CaO	18.6	0.3	18.1	0	11
Al_2O_3	4.9	0	33.7	0.1	13
MgO	14.9	35.8	0.3	2.0	9
FeO	9.0	27.0	0.7	44.9	17
TiO_2	3.5	0.1	0.2	53.6	5

TABLE F.3

SILICATE CONCENTRATIONS (BY WT%) IN MARE REGOLITH

Silicate	Mare Regolith
$CaO \cdot SiO_2$	19.6
$MgO \cdot SiO_2$	16.1
$FeO \cdot SiO_2$	30.4
$Al_2O_3 \cdot SiO_2$	23.2
$TiO_2 \cdot SiO_2$	8.9

TABLE F.4

MAJOR ELEMENTAL COMPOSITION OF LUNAR REGOLITH (WT %)

Element	Mare Reolith	Highlands	Basin Ejecta
Ca	7.9	10.7	7.70
Mg	5.8	4.6	6.1
Fe	13.2	4.9	8.7
Al	6.8	13.3	9.8
Ti	3.1	--	
Si	20.4	21.0	21.8
O	41.3	44.6	43.3
S	0.1	0.072	0.076
K	0.1	0.078	0.24
Na	0.3	0.48	0.38
Total	99.0	99.7	98.1

TABLE F.5

MINOR ELEMENTAL COMPOSITION OF LUNAR REGOLITH (PPM,WT)

Element	Mare	Highlands	Basin Ejecta
Ar	0.8	1.2	1.0
B	4.78	22.45	19.0
Be	2.63	1.2	4.15
C	104.0	106.5	136.7
Cl	25.6	17.0	25.0
Cs	0.39	0.11	0.33
F	174.3	54.5	139.5
H	54.8	56.0	76.5
He	28.5	6.0	8.0
Hg	0.014	0.004	--
Li	12.9	6.6	52.3
N	95.4	98.0	121.0
Ne	2.75	1.0	2.0
Total	506.9	370.6	585.5

TABLE F.6

PROPERTIES OF LUNAR ORES IN MARE REGOLITH

Lunar Ore	g/mole	Melting Point (°C)	Heat of Vaporization (kWhr/Kg)
Pyroxene	120.5	1557	5.2
Olivine	167.4	1890	3.8
Anorthite	278.0	1557	5.4
Ilmenite	151.7	1367	3.7
Bulk Mare			4.8

TABLE F.7

THERMODYNAMICS OF TRANSFORMATIONS OF LUNAR ORES IN MARE REGOLITH

Lunar Ore	Transformation	H (Kcal/Mole)
ANORTHITE		
	$CaO \cdot Al_2O_3 \cdot 2SiO_2(s) \rightarrow CaO(s) + Al_2O_3(s) + 2SiO_2(s)$	25.3
	$Al_2O_3(s) \rightarrow 2Al_l + 3/2O_2(g)$	403.3
	$Al(l) \rightarrow Al(g)$	69.5
	$CaO(s) \rightarrow Ca(l) + 1/2O_2(g)$	153.0
	$Ca(l) \rightarrow Ca(g)$	36.7
	$2SiO_2(s) \rightarrow 2Si(s) + 2O_2(g)$	433.6
	$2Si(s) \rightarrow 2Si(g)$	187.8
PYROXENE		
	$CaO \cdot SiO_2(s) \rightarrow CaO(s) + SiO_2(s)$	22.1
	$SiO_2(s) \rightarrow Si(s) + O_2(g)$	216.8
	$Si(s) \rightarrow Si(g)$	93.9
	$CaO(s) \rightarrow Ca(l) + 1/2O_2(g)$	153.0
	$Ca(l) \rightarrow Ca(g)$	36.7
	$MgO \cdot SiO_2(s) \rightarrow MgO(s) + SiO_2(s)$	9.8
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
	$MgO(s) \rightarrow Mg(g) + 1/2O_2(g)$	174.4
	$FeO \cdot SiO_2(s) \rightarrow FeO(s) + SiO_2(s)$	4.3
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
	$FeO(s) \rightarrow Fe(l) + 1/2O_2(g)$	61.2
	$Fe(l) \rightarrow Fe(g)$	83.6
	$Al_2O_3 \cdot SiO_2(s) \rightarrow Al_2O_3(s) + SiO_2(s)$	2.1
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
	$Al_2O_3(s) \rightarrow 2Al(l) + 3/2\ O_2(g)$	403.3
	$2Al(l) \rightarrow 2Al(g)$	139.0
	$TiO_2 \cdot SiO_2(s) \rightarrow TiO_2(s) + SiO_2(s)$	
	$TiO_2(s) \rightarrow Ti(s) + O_2(g)$	224.9
	$Ti(s) \rightarrow Ti(l)$	3.7

Lunar Ore	Transformation	H (Kcal/Mole)
	$Ti(l) \rightarrow Ti(g)$	102.0
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
OLIVINE		
	$2MgO \cdot SiO_2(s) \rightarrow 2MgO(s) + SiO_2(s)$	16.1
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
	$1MgO(s) \rightarrow 2Mg(g) + O_2(g)$	174.4
	$2FeO \cdot SiO_2(s) \rightarrow 2FeO(s) + SiO_2(s)$	8.7
	$2FeO \rightarrow 2Fe(l) + O_2(g)$	122.4
	$1Fe(l) \rightarrow 2Fe(g)$	167.2
	$SiO_2(s) \rightarrow Si(l) + O_2(g)$	216.8
	$Si(l) \rightarrow Si(g)$	93.9
ILMENITE		
	$FeO \cdot TiO_2(s) \rightarrow FeO(s) + TiO_2(s)$	8.0
	$FeO \rightarrow Fe(l) + 1/2O_2(g)$	61.2
	$Fe(l) \rightarrow Fe(g)$	83.6
	$TiO_2(s) \rightarrow Ti(s) + O_2(g)$	224.9
	$Ti(s) \rightarrow Ti(l)$	3.7
	$Ti(l) \rightarrow Ti(g)$	102.0

TABLE F.8

CHEMICAL PROPERTIES OF CONSTITUENTS OF MARE REGOLITH

Constituent	Melting Point (°C)	Boiling Point (°C)	Dissociation Energy at 1000K (KJ/Mole)
CaO	2927	2850	634
SiO_2	1700	2230	902
MgO	2852	3600	609
FeO	1379	--	269
Al_2O_3	2018	2980	1670
TiO_2	1830	--	940
Ti	1670	3289	
Fe	1536	2862	
Al	660	2520	
Mg	649	1088	
O_2	-219	-183	

TABLE F.9

LUNAR BASE PROPELLANT PROCESSING REACTIONS

HYDROGEN REDUCTION

1. $FeO \cdot TiO_2(s) + H_2(g) = Fe(s) + H_2O(s) + TiO_2(s)$
2. $2\ H_2O(g) = 2H_2(g) + O_2(g)$

MAGMA ELECTROLYSIS

3. $FeO \cdot TiO_2(s) = FeO(s) + TiO_2(s)$
4. $FeO(s) = Fe(l) + 1/2O_2(g)$

CARBOCHLORINATION

5. $Al_2O_3(s) + 3C(s) + 3Cl_2(g) = 2AlCl_3(g) + 3CO(g)$
6. $CaO(s) + C(s) + Cl_2(g) = CaCl_2(s) + CO(g)$
7. $SiO_2(s) + 2C(s) + 2Cl_2(g) = SiCl_4(g) + 2CO(g)$
8. $FeTiO_3(s) + 3C(s) + 3/2Cl_2(g) = FeCl_3(g) + TiO_2(s) + 3CO(g)$
9. $MgO(s) + C(s) + Cl_2(g) = MgCl_2(l) + CO(g)$
10. $CaO \cdot Al_2O_3 \cdot 2SiO_2(s) + 8C(s) + 8Cl_2(g) = 2AlCl_3(g) + CaCl_2(s) + 2SiCl_4(g) + 8CO(g)$
11. $AlCl_3(l) = Al(l) + 3/2Cl_2(g)$
12. $CO(g) + 1/2O_2(g) = CO_2(g)$
13. $CO_2(g) + H_2(g) = C(s) + H_2O(l)$
14. $H_2O = 2H_2 + O_2$

ACID LEACH

15. $Al_2O_3 \cdot SiO_2(s) + 11HF(l) = 2AlF_3 \cdot HSiF_5(s) + 5H_2O(l)$
16. $MgO \cdot SiO_2(s) + 6HF(l) = MgF_2 \cdot SiF_4(s) + 3H_2O(l)$
17. $FeO \cdot SiO_2(s) + 6HF(l) = FeF_2 \cdot SiF_4(s) + 3H_2O(l)$
18. $CaO \cdot SiO_2(s) + 6HF(l) = CaF_2 \cdot SiF_4(s) + 3H_2O(l)$
19. $TiO_2 \cdot SiO_2(s) + 8HF(l) = TiF_4 \cdot SiF_4(s) + 4H_2O(l)$
20. $AlF_3(l) + 3Na(l) = Al(l) + 3NaF(l)$
21. $MgF_2(l) + H_2O(l) = MgO(s) + 2HF(l)$
22. $2MgO(s) + Si(s) + 2CaO(s) = 2Mg(l) + Ca_2SiO_4(s)$
23. $FeSiF_6(s) + H_2O(l) + \text{electrical energy} = Fe(s) + \tfrac{1}{2}O_2(g) + 2HF(l) + SiF_4(l)$

CARBOTHERMAL

F-12 24. $MgO \cdot SiO_2(l) + 2CH_4(g) = 2CO(g) + 4H_2(g) + Si(l) + MgO(s)$

25. $2CO(g) + 6H_2(g) = 2CH_4(g) + 2H_2O(g)$

26. $2H_2O(l) = 2H_2(g) + O_2(g)$

27. $MgO \cdot SiO_2(l) = O_2(g) + Si(l) + MgO(s)$

TABLE F.10

PROPELLANT PROCESSING HEATS OF REACTION

Reaction #	T (°C)	H (KJ/mole)
1	900	42.6
2		249
3	1370	-72.1
4	1370	267.5
5	725	-1204.5
6	725	-266.8
7	725	65.8
8	725	-370.6
9	725	-99.6
10	725	
11	700	1270
12	700	-282.6
13	625	147.2
14		249
15	110	-2855
16	110	-146
17	110	-131.2
18	110	-107.8
19	110	-416.8
20	900	-500.2
21	1200	103
22	1200	422.5
23		333.2
24	1625	742
25	250	-412.6
26		249

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APPENDIX G

ADDITIONAL LUNAR SURFACE BASE PROPULSION SYSTEM STUDY TECHNOLOGY INFORMATION

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

ENABLING VS. ENHANCING TECHNOLOGIES

SYNTHESIS AND FORMULATION OF PROPELLANTS FROM LUNAR RESOURCES ARE ENABLING TECHNOLOGIES.

TIME FOR DEVELOPMENT

SYNTHESIS OF LO_2
SYNTHESIS OF Al
FORMULATION AND CHARACTERIZATION OF LH_2/Al FUELS
- TERRESTRIAL RESEARCH - TWO YEARS
- SPACE STATION RESEARCH - ONE YEAR

(SYNTHESIS OF $LSiH_4$ -
- LABORATORY RESEARCH - TWO YEARS
- PILOT RESEARCH - TWO YEARS)

ROUGH ESTIMATE COST FOR DEVELOPMENT

SYNTHESIS OF LO_2

SYNTHESIS OF Al

FORMULATION AND CHARACTERIZATION OF LH_2/Al FUELS -

- TERRESTRIAL DEVELOPMENT - 15 MAN-YEARS EQUIVALENT
- SPACE STATION DEVELOPMENT -

SYNTHESIS OF $LSiH_4$ -

- LABORATORY DEVELOPMENT - 6 MAN-YEARS EQUIVALENT
- PILOT DEVELOPMENT - 20 MAN-YEARS EQUIVALENT

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

COMBUSTION AND COOLING

TECHNOLOGY GOALS: STABLE, HIGH-EFFICIENCY COMBUSTION IN COMPACT COMBUSTORS; COOLING COMPATIBLE WITH LONG LIFE, REUSABLE, MAN-RATED ENGINES.

PROPELLANT SYSTEMS	PROBLEM STATEMENTS
LO_2LH_2 AT LOW MR	NONE - SOA - REGENERATIVELY COOL THRUST CHAMBER ASSEMBLY (TCA) IGNITION, STABILIZATION, COMBUSTION OK
LO_2/LH_2 AT HIGH MR	TCA COOLING - REDUCED COOLANT HEAT CAPACITY (LESS H_2, MORE O_2). WALL COMPATIBILITY WITH HIGH-TEMPERATURE, OXYGEN-RICH PRODUCTS (MR = 8-12) (PROBABLY NEED THERMAL SHOCK-RESISTANT COOLING).

PROPELLANT SYSTEMS (CONT.)

LO_2/LH_2-Al	COMBUSTION - PURE ALUMINUM TENDS TO COLLECT AN OXIDE FILM WHICH RETARDS/PREVENTS Al COMBUSTION. LARGE COMBUSTORS/LOW Isp. COOLING - VERY HIGH COMBUSTION TEMPERATURES IN ADDITION TO TWO-PHASE COOLANT AND LOW HEAT FLUX (OXYGEN) COOLANT.
LO_2/Al	COMBUSTION - SAME AS LO_2/LH_2-Al, BUT WORSE, BECAUSE LACK H_2 TO PROMOTE COMBUSTION CHEMISTRY.

LO_2/Al (CONT.)

COOLING - MUST RUN OXIDIZER-RICH IN ORDER TO PRODUCE ANY GAS TO EXPAND IN THE ROCKET NOZZLE. VERY HIGH COMBUSTION TEMPERATURES IN ADDITION TO TWO-PHASE, LOW HEAT FLUX COOLANT (OXYGEN). SEE LO_2/LH_2 AT HIGH MR AND WALL COMPATIBILITY PROBLEM. FURTHER - ALUMINUM INTRODUCTION TECHNOLOGY NEEDS ADDRESSING:

- o COOLED Al IN GELLED LO_2
- o LIQUID Al INJECTION (WITH ASSOCIATED BACKUP SYSTEMS)
- o HYBRID SOLID Al ALLOY + LIQUID LO_2
- o Al DUST INTRODUCTION, A LA DIESEL OR COAL COMBINATION

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

ENABLING VS. ENHANCING TECHNOLOGIES

PROPELLANT SYSTEM	PROBLEM	ENABLING	ENHANCING
LO_2/LH_2 AT LOW MR	~NONE		X
LO_2/LH_2 AT HIGH MR	TCA COOLING (WALL COMPATIBILITY)	X	
LO_2/LH_2-Al	COMBUSTION		X
	TCA COOLING	X	
LO_2/Al	ALUMINUM INTRODUCTION	X	
	COMBUSTION		X
	COOLING	X	

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

TIME FOR DEVELOPMENT

PROPELLANT SYSTEM	PROBLEM	POTENTIAL SOLUTION	APPROXIMATE TOTAL DEVELOPMENT TIME
LO_2/LH_2 AT LOW MR	NONE	NORMAL DEVELOPMENT	5 YEARS
LO_2/LH_2 AT HIGH MR	TCA COOLING	COATING AND BASE MATERIAL	8 YEARS
LO_2/LH_2-Al	TCA COOLING	TRIPROPELLANT INJECTOR (LO_2, LH_2, LH_2-Al)	10 YEARS
	COMBUSTION		
LO_2/Al	ALUMINUM	MULTIPLE APPROACH PROGRAM	15 YEARS
	COOLING	TRANSPIRE WITH LO_2	
	COMBUSTION	RESEARCH/DEVELOPMENT	

DIAGNOSTICS AND CONTROL

TECHNOLOGY GOALS: INTEGRATED CONDITION MONITORING AND CONTROL SYSTEM HARDWARE, WITH PREDICTIVE CAPABILITY

PROPELLANT SYSTEMS	PROBLEM STATEMENT
LO_2/LH_2 AT LOW MR	SENSOR TECHNOLOGY, SYSTEMS INTEGRATION, SYSTEM SOFT/HARDWARE, PREDICTIVE DATA BASE GENERATION
LO_2/LH_2 AT HIGH MR	" "
LO_2/LH_2-Al	" "
LO_2/Al	" "

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

ENABLING VS. ENHANCING TECHNOLOGIES

PROBLEMS	REASON	ENABLING	ENHANCING
SENSOR TECHNOLOGY	IMPROVE SIZE, WEIGHT, COST, PERFORMANCE RELIABILITY		X
(SMART SENSORS ASSUMED, ELSE SUMMING EFFECT APPROACHES ENABLING)			
SYSTEM INTEGRATION	ALLOWS "SMARTER" CONTROL SYSTEM AND IMPROVES, AS ABOVE	X	X
SYSTEM SOFT/HARDWARE	ENGINE SPECIFIC, TO SOME DEGREE	X	
PREDICTIVE DATA BASE GENERATION	CANNOT PREDICT FAILURES W/O ALGORITHMS - DATA BASE GRADUALLY ALLOWS IMPROVED RESULTS	X	X

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

<u>TIME FOR DEVELOPMENT</u>

PROBLEMS	APPROXIMATE DEVELOPMENT TIME
SENSOR TECHNOLOGY	5-10 YEARS
SYSTEM INTEGRATION	5 YEARS (WITH ENGINE DEVELOPMENT)
SYSTEM SOFT/FIRMWARE	5-10 YEARS (WITH ENGINE DEVELOPMENT
PREDICTIVE DATA BASE	~10 YEARS (ASYMPTOTIC)

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

PAYBACK POTENTIAL

PROBLEMS/TECHNOLOGY	PAYBACK
SENSOR TECHNOLOGY	REDUCED WEIGHT, REDUCED FAILURES, ERRORS AND SIZE
SYSTEMS INTEGRATION	ENABLING - IMPROVED ENGINE CONTROL - AUTOMATIC AND THROUGH PILOT CRITICAL
SYSTEM SOFT/HARDWARE	ENABLING - MUST INDUCE AUTOMATION AND PREVENTIVE FAILURE DIAGNOSTICS
PREDICTIVE DATA BASE	ENABLING - DEEP SPACE FLIGHT FAILURE REDUCTION/ ELIMINATION - ENGINES

SPACE OPERATIONS

TECHNOLOGY GOALS:	ENGINE SYSTEM COMPATIBLE WITH SPACE ENVIRONMENTS AND OPERATIONS ON LUNAR SURFACE AND IN CISLUNAR SPACE, IN BOTH OPERATIONAL AND STORAGE/MAINTENANCE MODE

PROPELLANT SYSTEMS AND PROBLEMS

LO_2/LH_2 AT LOW MR	LUNAR "DUST" CONTAMINATION, SURFACE TEMPERATURE FLUCTUATIONS, DESIGN FOR SEMI-ROBOTIC MAINTENANCE AND PRE/POST FLIGHT ACTIVITIES
LO_2/LH_2 AT HIGH MR	" " " "
LO_2/LH_2-Al	" PLUS LUNAR GRAVITY FUEL PHASE SEPARATION
LO_2/Al	" PLUS TBD, BASED UPON Al INTRODUCTION SYSTEM

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

ENABLING VS. ENHANCING TECHNOLOGIES

PROBLEMS	REASON	ENABLING	ENHANCING
LUNAR "DUST" CONTAMINATION	FOUL ACTUATORS, INJECTORS, SENSORS DURING LIFT-OFF, RETURN, OR MAINTENANCE	X	
LUNAR DIURNAL TEMPERATURES	AVOID LAUNCHING FROM SUBSURFACE, OR ONLY AT NIGHTTIME		X

ENABLING VS. ENHANCING TECHNOLOGIES (CONT.)

PROBLEMS	REASON	ENABLING	ENHANCING
SEMI-ROBOTIC MAINTENANCE, ETC. (AIDED BY REMOTE HUMAN JUDGMENT)	AVOID EXCXESS MAN DANGER, EXPOSURE, EFFORT, TIME AT REMOTE, HOSTILE LOCATION WITH HUMAN ERRORS IMPLICIT	X	
H_2/Al SEPARATION	TO AVOID TRIPROPELLANT INJECTION DESIGN	X	

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

TIME FOR DEVELOPMENT

PROBLEMS	DEVELOPMENT TIME
LUNAR DUST CONTAMINATION	~5 YEARS (WITH ENGINE DEVELOPMENT)
LUNAR DIURNAL TEMPERATURES	REQUIRES SPECIFIC ANALYSES
SEMI-ROBOTIC MAINTENANCE	~10 YEARS (ROBOTIC EQUIPMENT IN SERIES WITH ENGINE DEVELOPMENT)
H_2/Al SEPARATION	SEE "PROPELLANTS"

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

ROUGH ESTIMATE COST FOR DEVELOPMENT

PROBLEM/TECHNOLOGY	ROUGH DEVELOPMENT COST ESTIMATE, $
LUNAR DUST CONTAMINATION	SMALL
LUNAR DIURNAL TEMPERATURES	REQUIRES SPECIFIC ANALYSES
SEMI-ROBOTIC MAINTENANCE	$\sim 1x10^8$
H_2/Al SEPARATION	SEE "PROPELLANTS"

LUNAR SURFACE BASE PROPULSION SYSTEM STUDY
TECHNOLOGY AREAS FOR FUTURE STUDY

PAYBACK POTENTIAL

PROBLEM/TECHNOLOGY	PAYBACK
LUNAR DUST CONTAMINATION	ENABLING
LUNAR DIURNAL TEMPERATURES	ESTIMATE COST OF LUNAR SILOS, OR LOSS OF DAYTIME LAUNCHES (29-DAY WINDOWS)
SEMI-ROBOTIC MAINTENANCE	ESTIMATE LOSS OF MEN OR LUNAR SILOS
H_2/Al SEPARATION	SEE "PROPELLANTS"

Technology Plan for $LO_2/LSiH_4$ Engine

Task No.

1. Demonstrate Low MR Gas Generator with LO_2 and SiH_4 Propellants.
   - MR range and turbine inlet temperature
   - Combustion (ignition, C*, and stability)
   - Longevity with SiO_2 in gases

2. Subsequently, Demonstrate TPA Turbine.
   - Performance
   - Longevity of nozzles, blades, rotor and casing
   - Seals
   - Bearings

3. In Parallel with Task 2 Demonstrate TCA Injector with LO_2/SiH_4 Propellants with a Workhorse Combustor.
   - Performance
   - Longevity (Thermal Compatibility)

4. Demonstrate TCA Cooling (Subsequent to 3).
   - Design engine - Determine if SiH_4 regenerative cooling is needed. If not: Demonstrate LO_2 cooling
     If so: Demonstrate SiH_4 cooling
   - Heated tube tests for "burnout heat flux"
   - SiH_4 decomposition conditions determination
   - Demonstrate TCA with injector - hot firing

5. Lastly - Demonstrate Engine with Testing.

6. Demonstrate Autogenous Pressurization with SiH_4, using Heat Exchanger in Turbine Exhaust after Task 1 is completed.
   - Heat exchanger performance and longevity with SiO_2 in gas
   - SiH_4 vapor generated (vs. Si solids)

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