NASA-CR-195527

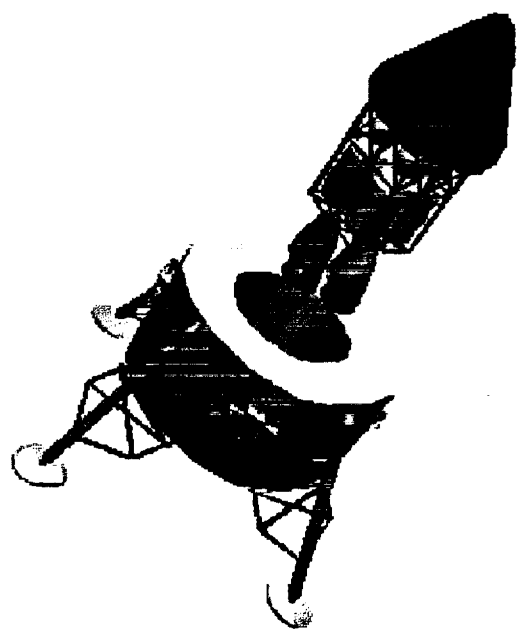
Extended Duration Lunar Lander

NASW-4435

Submitted to:
Dr. George W. Botbyl

IN-18-CR
204258
P. 116

Department of Aerospace Engineering
University of Texas at Austin



Selenium Technologies

May 15, 1993

(NASA-CR-195527) EXTENDED DURATION LUNAR LANDER (Texas Univ.) 116 p

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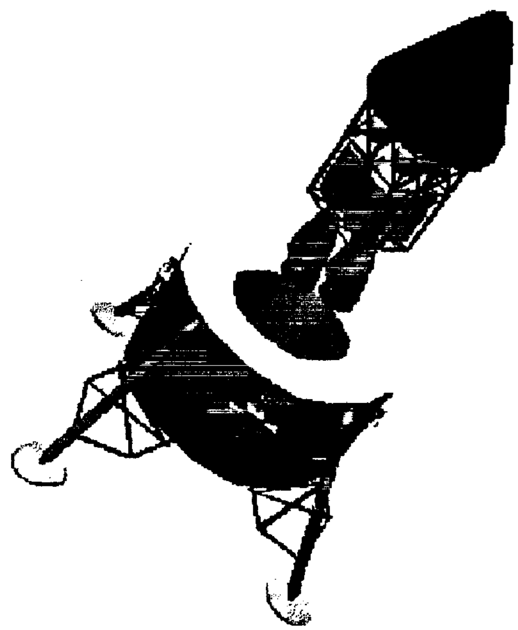

Selenium Technologies

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Matt Carter	Edward Hirst
Donna Cosper	Ray Li
David Garza	Martin Lindsey
Eloy Gonzalez	Tony Ng

Acknowledgments

Selenium Technologies would like to thank the following people for their support and guidance:

University of Texas at Austin

Department of Aerospace Engineering and Engineering Mechanics
Dr. George Botbyl, Dr. Wallace Fowler, Tony Economopoulos, Dr. John Lundberg, Scott Striepe, David Reed, Greg Cabe, everyone else who provided input during our design reviews, and Jaime Nualart

Advanced Graphics Laboratory
Mark Cartwright, and the technical staff

Johnson Space Center

Phil Deans, Charles Teixeira, Jim Masciarelli, and Eric Hurlbert

Others

Dr. Adam Bruckner
C. D. Limerick at Pratt and Whitney
Dr. Paul Robinson at the Jet Propulsion Laboratory

Executive Summary

Selenium Technologies has been conducting preliminary design work on a manned lunar lander for use in NASA's First Lunar Outpost (FLO) program. The resulting lander is designed to carry a crew of four astronauts to a prepositioned habitat on the lunar surface, remain on the lunar surface for up to 45 days while the crew is living in the habitat, then return the crew to Earth via direct reentry and land recovery. Should the need arise, the crew can manually guide the lander to a safe lunar landing site, and live in the lander for up to ten days on the surface. Also, an abort to Earth is available during any segment of the mission.

The main propulsion system consists of a cluster of four modified Pratt and Whitney RL10 rocket engines that use liquid methane (LCH_4) and liquid oxygen (LOX). Four engines are used to provide redundancy and a satisfactory engine out capability. Differences between the new propulsion system and the original system include slightly smaller engine size and lower thrust per engine, although specific impulse remains the same despite the smaller size. Concerns over nozzle ground clearance and engine reliability, as well as more information from Pratt and Whitney, brought about this change.

The power system consists of a combination of regenerative fuel cells and solar arrays. While the lander is in flight to or from the Moon, or during the lunar night, fuel cells provide all electrical power. During the lunar day, solar arrays are deployed to provide electrical power for the lander as well as electrolyzers, which separate some water back into hydrogen and oxygen for later use by the fuel cells. Total storage requirements for oxygen, hydrogen, and water are 61 kg, 551 kg, and 360 kg, respectively.

The lander is a stage-and-a-half design with descent propellant, cargo, and landing gear contained in the descent stage, and the main propulsion system, ascent propellant, and crew module contained in the ascent stage. The primary structure for both stages is a truss, to which all tanks and components are attached. The crew module is a conical shape similar to that of the Apollo Command Module, but significantly larger with a height and maximum diameter of 6 m.

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List of Acronyms

Ag-Zn	Silver-Zinc
C	Celsius
CAO	Chief Administrative Officer
CEO	Chief Engineering Officer
CO_2	Carbon Dioxide
DSCS	Defense Satellite Communications System
DSN	Deep Space Network
ECLSS	Environmental Control and Life Support System
EDC	Electrochemical Depolarized Cells
EVA	Extra-Vehicular Activity
FLO	First Lunar Outpost
GN&C	Guidance, Navigation and Control
HLLV	Heavy Lift Launch Vehicle
H_2O	Water
IMU	Inertial Measurement Units
kW	Kilowatt
kWh	Kilowatt-Hours
LCH_4	Liquid Methane
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LOI	Lunar Orbit Insertion
LOX	Liquid Oxygen
MMH	Monomethyl Hydrazine
NASA	National Aeronautics and Space Administration
NTO	Nitrogen Tetroxide
Pu-238	Plutonium-238
RCS	Reaction Control System
RTG	Radioisotope Generators
TEI	Trans-Earth Injection
TLI	Translunar Injection

1.0 Introduction

As mankind advances toward the permanent settlement of space, NASA finds it necessary to establish a lunar habitat capable of supporting life for extended periods of time. These extended duration missions may last anywhere from 14 to 45 days. Essential to the success of this habitat is a spacecraft capable of transporting a given set of crewmembers and cargo to and from the habitat. Selenium Technologies has been conducting preliminary design work on this lunar lander according to the requirements set by NASA in the Request for Proposal received in late January. The lander will provide the crew with the necessary transportation, life support, and cargo space necessary for each mission.

The system will rely on a Heavy Lift Launch Vehicle (HLLV) to reach low Earth orbit (LEO). Four crewmembers and limited cargo will be transferred to a lunar orbit. The spacecraft will be able to descend to any predetermined location on the surface of the Moon while providing the capability of redesignating the landing aim point during descent. The craft will provide the four crewmembers with life support for up to 10 days on the lunar surface while they prepare the lunar habitat for use. After as long as 45 days on the lunar surface, the craft will ascend from the lunar surface and return to Earth using a direct reentry and with a land recovery.

2.0 Orbits

2.1 Introduction

The orbits subsystem provides the mission trajectory (i.e., Δv burns required) which must be established before the sizing of the other subsystems can be completed. When Δv burns are known, the propulsion system can find the required fuel mass/volume and the structures subsystem can begin sizing. This section of the report will outline some important requirements and constraints on the trajectory of the spacecraft and detail a mission trajectory.

2.2 Requirements and Constraints

There are several major requirements for the orbits subsystem. Time of flight between the Earth and the Moon must be no longer than four days. The lunar lander must be capable of landing at any site on the Moon. The crew module must make a land touchdown at the end of the mission. The lunar lander must be able to abort at any phase of the mission.

Several constraints and considerations also exist. Nominal lunar landings will occur near the local lunar sunrise and nominal lunar liftoffs will occur near the local lunar sunset. These two conditions would provide a maximum period of lighting on the lunar surface for the mission.

Two assumptions are made in this mission scenario. First, the FLO is assumed to have already been placed on the Moon's surface. Second, it is assumed that the HLLV will be capable of lifting a 200 metric ton payload into a 185 km altitude orbit.

2.3 Mission Trajectory

The mission trajectory chosen for our lunar lander is the minimal-energy trajectory detailed in NASA's FLO Conceptual Flight Profile document[1]. The approximate Δv burns that are listed in the following sections come from that document.

2.3.1 Earth-to-Moon Trajectory

Figure 2.1 shows the trajectory from the Earth to the Moon for a mission to Mare Symthii. On December 5, 1999, a HLLV carrying the lunar lander will launch from Kennedy Space Center. The HLLV will boost the lunar lander and a lunar injection stage into a 185-km altitude parking orbit with a 33 degree inclination. At the first injection opportunity, the lunar injection stage will perform a 3140 m/s Δv burn to place the lunar lander on its four-day transfer trajectory. For midcourse corrections along the way, a Δv of 30 m/s is budgeted.

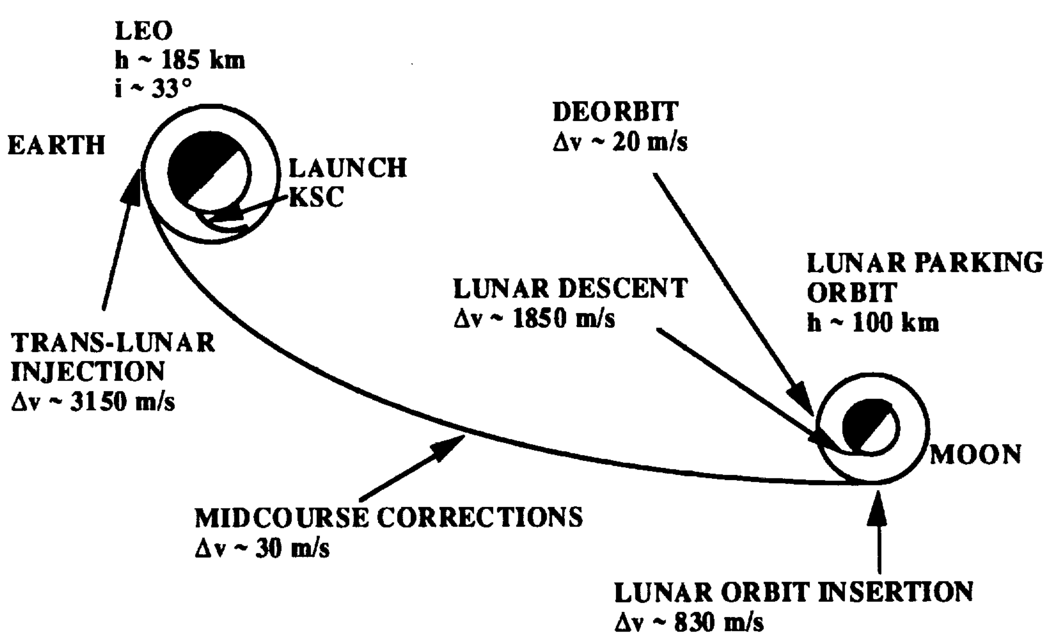

Figure 2.1 Earth-to-Moon Trajectory

At the end of the outbound transfer trajectory, the lunar lander will make an 830 m/s Δv burn to circularize around the Moon. The altitude of the temporary parking orbit around the Moon will be 100 km. When the appropriate phasing is reached, the lunar lander will make a 20 m/s Δv burn to deorbit. During the powered descent phase, the lunar lander will make a total of 1850 m/s Δv burn. Nominal touchdown will occur on December 9, 1999. The Δv burns for the major events on the outbound trajectory are summarized in Table 2.1.

Table 2.1 Δv Summary for Earth-to-Moon Trajectory

Event	Δv (m/s)
Translunar Injection (TLI)	3140
Lunar Orbit Insertion (LOI)	830
Deorbit	20
Landing	1850

2.3.2 Moon-to-Earth Trajectory

Figure 2.2 shows the trajectory from the Moon to the Earth. After a 42 day stay (45 days for contingencies) on the Moon, the crew will liftoff from the Moon's surface in the ascent stage, leaving the descent stage on the surface of the Moon. The powered ascent phase of the mission will require a total of 1830 m/s Δv. Once the ascent stage reaches an altitude of 100 km, it will make a 20 m/s Δv burn to circularize the orbit.

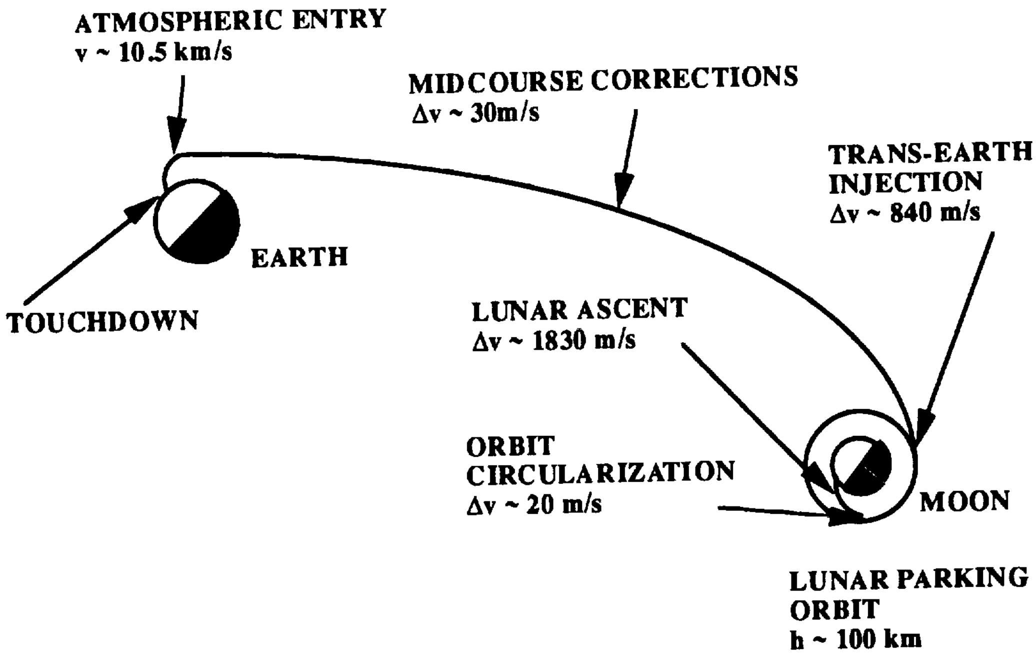

Figure 2.2 Moon-to-Earth Trajectory

When the phase conditions are met between the ascent stage and the Earth, the ascent stage will make an 840 m/s Δv burn to begin the transfer to the Earth. For midcourse corrections, a Δv of 30 m/s is budgeted. As the ascent stage nears the Earth, the crew module will separate from the rest of the ascent structure. The crew module will make a direct re-entry into the Earth's atmosphere. When atmospheric reentry begins, the crew module will be traveling at a relative velocity of approximately 10.5 km/s. The crew module will follow an Apollo-type reentry profile. After an approximately 15 minute reentry through the Earth's atmosphere, the crew module will deploy parachutes and fire retro-rockets (with approximately 20 m/s total Δv burn) to make a soft land touchdown. Nominal touchdown will occur on January 24, 2000. The Δv burns for the major propulsive events in the return trajectory are summarized in Table 2.2.

Table 2.2 Delta-v Summary for Moon-to-Earth Trajectory

Event	Δv (m/s)
Lift-off	1830
Lunar Orbit Circularization	20
Trans-Earth Injection	840

2.3.3 Δv Budget

The Δv burn numbers given in the previous two sections correspond to the particular Earth-Moon geometry at the time of launch. Table 2.3 shows the approximate Δv numbers corresponding to the worst-case geometry between the Earth and the Moon.

Table 2.3 Δv Budget for Worst-Case Earth-Moon Geometry

Event	Δv (m/s)
Translunar Injection	3200
Outbound Midcourse Corrections	30
Lunar Orbit Insertion (LOI)	890
Deorbit	20
Descent	1850
Ascent	1830
Lunar Orbit Circularization	20
Trans-Earth Injection	850
Inbound Midcourse Corrections	30

2.4 Free Return Trajectories

On a lunar free-return trajectory, if the spacecraft is unable to make the Δv burn to circularize its orbit around the Moon, only a minimal amount of Δv is required to place the spacecraft on a return trajectory to the Earth. Out of concern for the criticality of engine failure in our preliminary single-engine lunar lander design, we studied free-return trajectories as an alternative to the minimal energy trajectory proposed in the FLO Conceptual Flight Profile document.

Several reasons prompted us to choose the minimal energy trajectory over the free-return trajectory. The redesign of our lunar lander from one engine to four engines decreased the criticality of single-engine failure. Examination of typical Apollo Δv burns and preliminary free-return trajectory analysis based on a computer algorithm described in Battin[2] showed an increase in total mission Δv budget of 200-300 m/s over the total mission Δv budget for the minimal energy trajectory, creating unacceptable increases in mission mass estimates. NASA trade studies show that any mass savings gained by the faster transfer time of the lunar free-return trajectory are lost in the increase in propulsive mass. In addition, the minimal energy trajectory has larger launch and TLI windows than the free-return trajectory, allowing for more flexibility in mission scheduling and execution[3].

2.5 References

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2. Battin, Richard H. An Introduction to the Mathematics and Methods of Astrodynamics. AIAA Education Series: 1987, pp. 437-446.

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3.0 Power Subsystem

3.1 Introduction

The power subsystem provides electrical power to the spacecraft subsystems during all phases of the mission. The mission scenario calls for a system that can operate at different power levels and handle emergencies and aborts.

3.2 Requirements

Input from the subsystems is required to form an accurate picture of the power requirements during the different phases of the mission. The power requirements available from the subsystems are used for a comparative analysis of different power options. During this analysis, it is assumed that the landing site does not have any nearby terrain features such as cliffs or mountains that would obscure the lander from sunlight during the lunar day. Appendix A shows the worst case breakdown of the mission, the power allotted to each subsystem, and the percentage of such power that is used during each mission phase. The total time for the mission is assumed to be 1291 hours. During this time the total amount of energy required is 2789 kilowatt-hours (kWh) at an average rate of 2.16 kilowatts (kW). Peak power (4.71 kW) occurs during the landing, takeoff and transfer burns, while minimum power (1.21 kW) occurs while the crew is in the FLO habitat.

Figure 3.1 summarizes the power requirements using a power requirements timeline. The different shadings describe the different loading conditions for the power system.

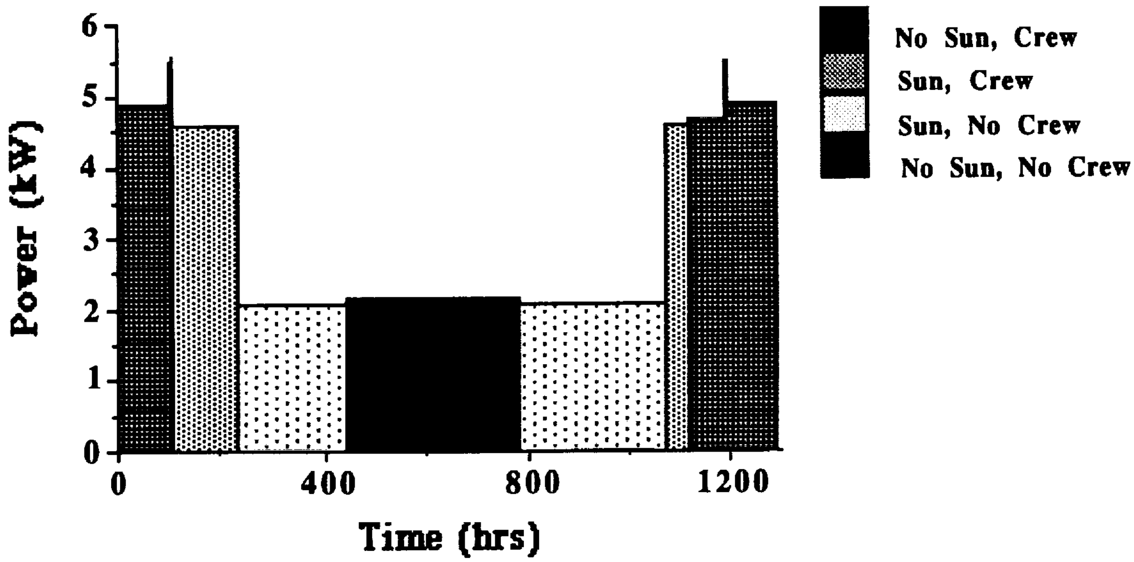

Figure 3.1 Power Requirements Timeline

3.3 Criteria and Concerns

The preliminary criteria that were considered in the initial selection of power system options were: mass, reliability, space qualification, complexity, and safety. These criteria were only used in an initial qualitative analysis of the power systems. Once a preliminary selection of the power system options was achieved, the mass of the system became the main driver for the determination of the most adequate power system.

3.4 Power System Options

The initial group of power system options was qualitatively analyzed according to the criteria. Solar cells and rechargeable batteries, solar cells and regenerative fuel cells, and fuel cells were chosen for further analysis. A summary of the characteristics of the chosen systems is presented in this section, as well as a justification for the exclusion of other systems.

3.4.1 Solar Cells and Rechargeable Batteries

A schematic of a solar cell and rechargeable battery system is shown in Figure 3.2. This system employs secondary batteries to provide power during transit times from Earth to Moon, and during lunar night. During the lunar day, solar panels are deployed, taking over the power loads, and providing the energy required to recharge the batteries.

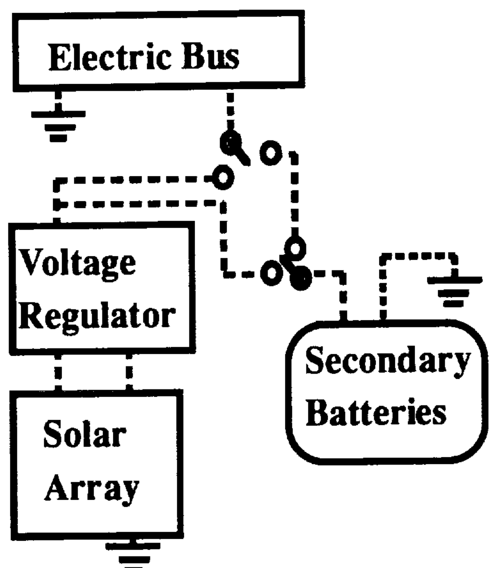

Figure 3.2 Solar Cells and Rechargeable Batteries System Schematic (based on regenerative fuel cell schematic)

3.4.2 Solar Cells and Regenerative Fuel Cells

Figure 3.3 shows a schematic of a typical solar cell and fuel cell configuration. This system works similar to the previous option in that fuel cells carry the load requirements during transits and the lunar night. Solar cells carry the load during the day and also provide power to electrolyzers for fuel regeneration. The fuel cell-electrolyzer combination requires fuel, tanks, piping, and control valves to control the passage of fluids throughout the system. Hydrogen and oxygen are fed to the fuel cells, which produce power and water. Some water is stored in a tank and electrolyzed into hydrogen and oxygen for use during latter portions of the mission.

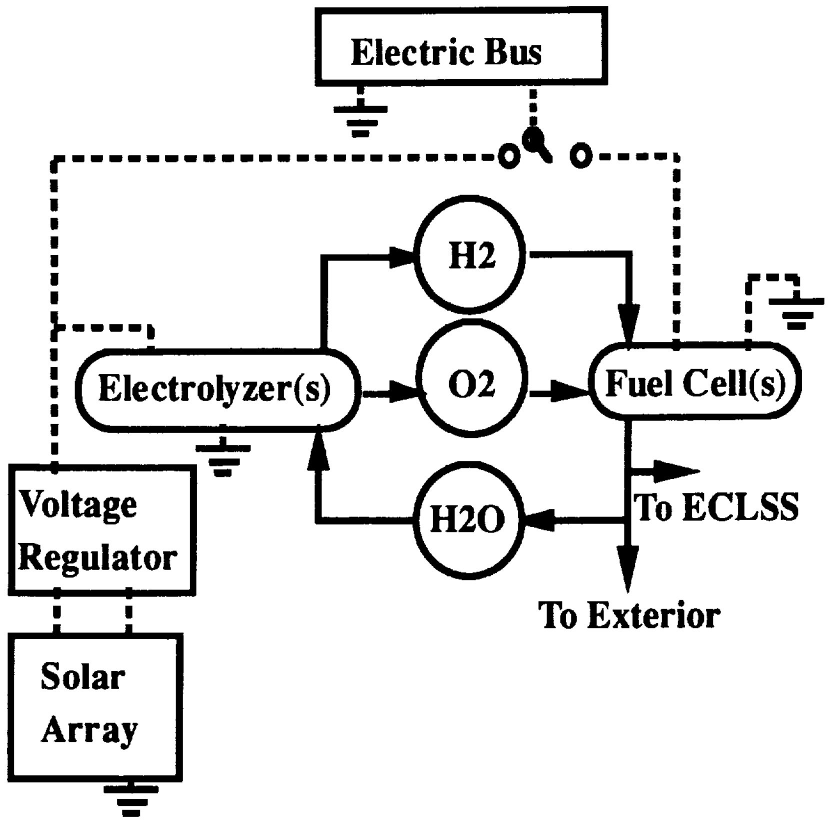

Figure 3.3 Solar Cells and Regenerative Fuel Cells System Schematic[1] (Note: solid lines = piping, dashed lines = electrical connections)

3.4.3 Fuel Cells

The fuel cell system depends on the transformation of hydrogen and oxygen for power production. The water that is produced during the mission is given to the life support system. Excess water is expelled from the lander. The nature of the system makes it completely independent of the sun. Figure 3.4 shows a schematic of the fuel cell system.

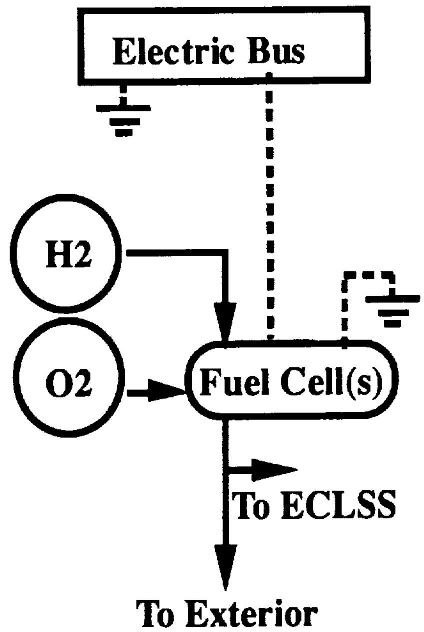

Figure 3.4 Fuel Cell System Schematic (based on regenerative fuel cell schematic)

3.4.4 Other Options

The other preliminary options that were considered as possible power systems for the lunar lander included radioisotope generators (RTGs), solar dynamic converters, and large-scale nuclear reactors. These options were subjected to the initial criteria and were discarded.

RTGs use radioactive materials as heat sources. This heat is converted to electric power by means of thermoelectrics or a working fluid passing through a generator. The major drawback with RTG use is fuel availability. Current RTG designs use plutonium-238 (Pu-238) as the heat source and the entire stockpile of Pu-238 has been committed to interplanetary missions. The radioactive nature of RTGs could place the crew in a life-threatening situation should the power system malfunction, and would also draw large resistance from the political, public, and scientific communities.[2]

Solar dynamic converters use a solar collector to concentrate solar energy and heat in a fluid. The fluid is then placed through a dynamic cycle, producing electricity. Solar dynamic converters are a relatively new, unproven concept in space power generation; power levels are predicted to range from 10 to 40 kW. The main deterrent of this option is its relatively unproven technology and the complexity of its energy conversion system.[3,4]

Finally, large-scale nuclear power reactors were quickly discarded. Their primary disadvantage is the inefficiency of the heat-to-electricity conversion. Most space-tested large-scale nuclear power systems have a thermodynamic efficiency of about five percent, resulting in tremendous amounts of waste heat. Additionally, the crew would require extensive shielding from the reactor's radioactive nature, thus significantly increasing the mass of the system. Add to this the political and public perception of nuclear power plants and it is easy to see the difficulty in using a system like this.[5]

3.5 Final Selection Based on a Mass Analysis

The main driver for the final selection of an adequate power system is the mass of the system. The mission timeline and subsystem power requirements were used as the basis for this analysis. The first step was to determine the amount of energy required by the subsystems during each phase of the mission. The amount of energy required by the power system itself was then added to the energy requirements. Each power system was then sized using these requirements. The final results of this mass analysis are found in Table 3.2; the spreadsheets containing the specific analysis and intermediate results are found in Appendix A.

Table 3.2 Power System Mass Comparison

Power System	Main System Mass	Solar Array Mass	Total Mass
Solar, Batteries	9624.74 kg	77.18 kg	9701.92 kg
Solar,Fuel Cell	1653.2 kg	174.07 kg	1827.3 kg
Fuel Cells	4134.26 kg	0 kg	4134.26 kg

From Table 3.2, the choice of the solar cell and fuel cell power system as an adequate power system is evident. The solar cell and battery system is about five times more massive and the fuel cell system is double the mass. Although the solar cell and fuel cell system is complex and has not been space-qualified, the mass savings are too great to disregard.

After the selection of the solar cell and regenerative fuel cells, a mission power timeline was developed. This timeline is different from the previously presented power requirements timeline in that it includes the power required for the power system itself during the different mission phases. These requirements include power for the pumps and valves in the fuel cell and electrolyzer sections of the subsystem and also the solar cell power used for electrolysis. The overall energy consumed during the mission now becomes 6342 kWh, used at an average rate of 4.91 kW. The maximum power output of the subsystem occurs during the times when solar power is available to perform the regeneration of the fuel, while the minimum power still occurs when the crew is in the habitat. The final mission timeline is found in Figure 3.5, while the mission breakdown is found in Appendix A.

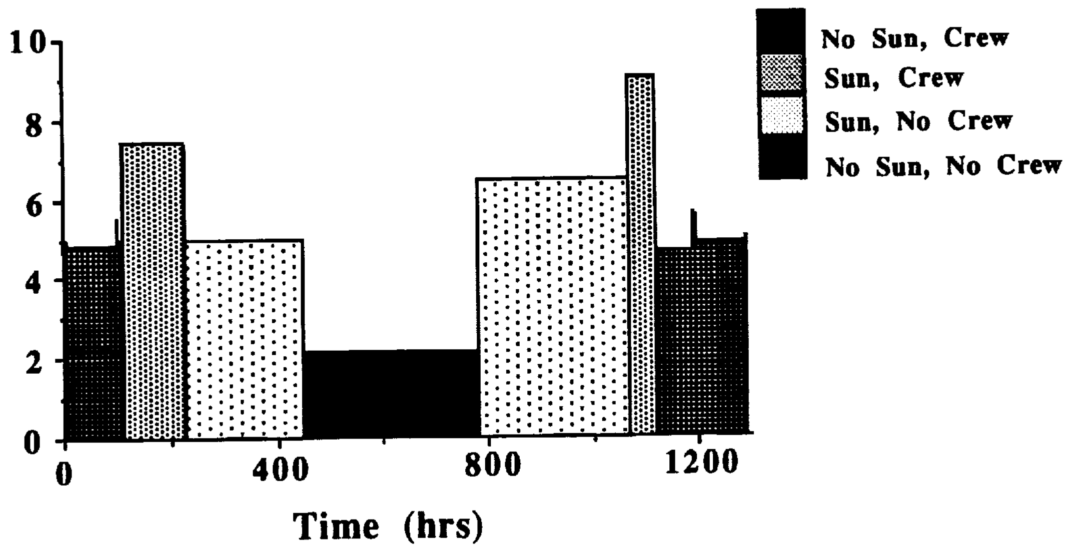

Figure 3.5 Final Mission Power Timeline

Further along during the development of the power system, it was decided that the fuel cell tanks were to be placed on the ascent truss, thus not permitting fuel cell power during the Earth reentry phase. The time allotted for reentry was small enough as to not affect the choice of the power option. However, a secondary battery system was chosen to provide power during this reentry phase. The specific batteries that were considered for this system are described in Appendix A. Four silver-zinc (Ag-Zn) rechargeable battery modules[6] were chosen to provide power during the reentry phase. These have a relatively good shelf life and can be trickle charged to full capacity to replace any leakage. It is also important to notice that the fuel requirements were sized so as to provide enough energy for Earth return at any time during the mission.

The final subsystem sizing is found in Appendix A, a breakdown of the masses of the solar cell and regenerative fuel cell system is found in Table 3.3, and a final system schematic is found in Figure 3.6.

Table 3.3 Power System Mass Breakdown

Component	Total Mass	Total Size
Three Fuel Cells	280 kg	1.50 m^3
Three Electrolyzers	280 kg	1.50 m^3
Hydrogen	79 kg	1.11 m^3
Oxygen	663 kg	0.58 m^3
Water (max. stored)	360 kg	0.36 m^3
Pumps, Pipes, Struc.	552 kg	N/A
Solar Arrays (GaAs)	264 kg	80 m^2
Array Support	132 kg	N/A
Four Ag-Zn batteries	27 kg	0.05 m^3

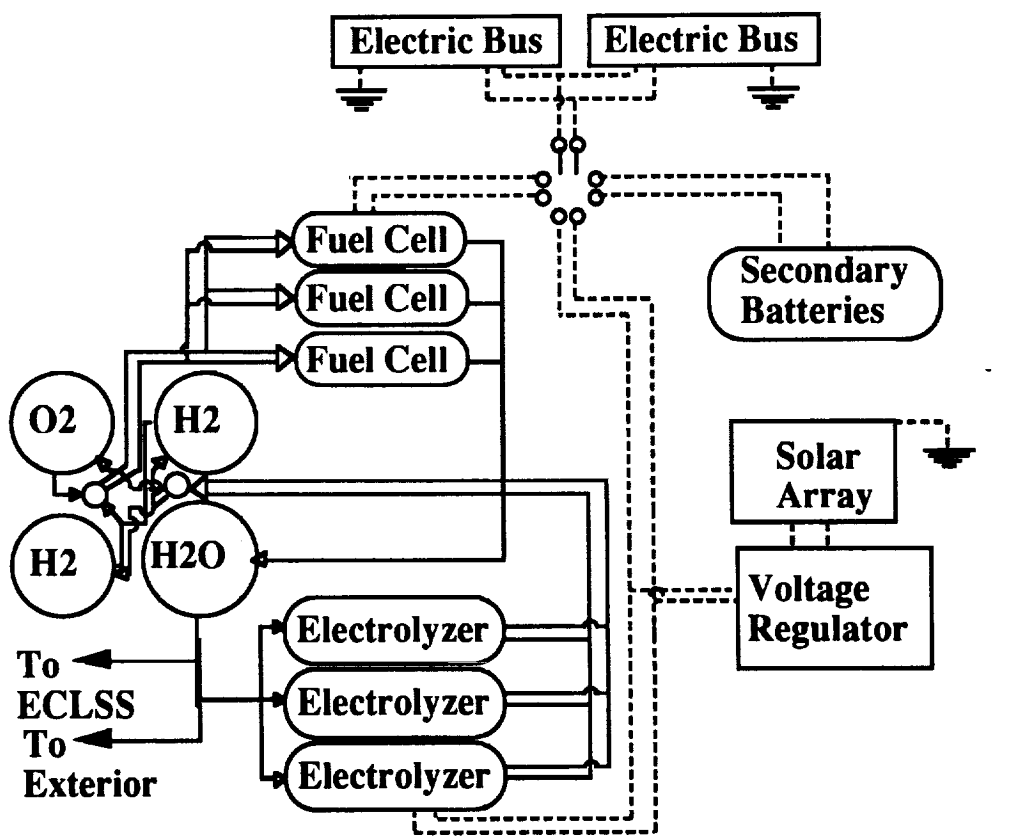

Figure 3.6 Final Power System Schematic

3.6 References

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3. Wertz, James and Wiley Larson eds., Space Mission Anaylsis and Design. Kluwer Academic Publishers and Microcosm, Inc.: Torrance, 1992, p. 393

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6. Lunar Lander Design for the First Lunar Outpost, Engineering Directorate, Systems Engineering Division, NASA, JSC - 25896, p. 9-1.

4.0 Propulsion

4.1 Introduction and Requirements

The propulsion subsystem performs all spacecraft propulsive activity. Table 4.1 gives the activities, primary and secondary, which are required in the mission scenario. The Δv estimates given are from a worst-case Earth-Moon geometry trajectory analysis performed by NASA.[1]

Table 4.1 Propulsive Requirements for the Mission

	MANEUVER	ESTIMATED Δv
Major Propulsive Maneuvers	Trans-Lunar TCM	30 m/s
	Lunar Orbit Insertion	882 m/s
	Lunar Deorbit Burn	20 m/s
	Powered Descent	1878 m/s
	Ascent to Lunar Orbit	1826 m/s
	Trans-Earth Injection	846 m/s
	Trans-Earth TCM	30 m/s
Secondary Propulsive Maneuvers	Attitude Control Burns	TBD
	Abort Maneuvers	TBD

To perform all the requirements, the propulsion system must have certain capabilities, such as:

- high thrust main engine(s) with throttle capability for landing
- restart capability for main engine(s)
- reliable reaction control system thrusters with proven multi-start capability and long lifetime
- the RCS propulsion system must be able to start in zero gravity conditions
- propellants which can be effectively stored for up to two months

4.2 Options

In considering options for the propulsion system, the propulsion system is subdivided into two parts: the main propulsion system and the secondary propulsion system, or Reaction Control System (RCS). The main propulsion system must perform all of the major maneuvers and some abort maneuvers. The RCS must perform all attitude control maneuvers and some abort maneuvers.

4.2.1 Main Propulsion System Options

Only propulsion systems which use liquid propellants were considered for this study, due to the need for restarting and throttling of engines. Hybrid solid propellant rockets with restart and throttle capability have been studied, but none have been flown into space, whereas numerous flight proven, restartable liquid propellant engines exist.[2] Low thrust systems such as ion thrusters, arcjets and resistojets were also ruled out early since the landing and takeoff phases of the mission require a higher maximum thrust than these systems can provide.

Nuclear thermal rocket engines were ruled out primarily because of safety concerns. The required crew protection radiation shielding would add mass to the spacecraft structure. Also, current designs using nuclear thermal engines are most mass efficient for thrust ranges well above what is required for this project.[3] Finally, given the current resistance from the public, scientific, and political communities to nuclear devices in space, any option requiring large quantities of radioactive material was ruled out for this project.

Monopropellant liquid propellant systems were not considered in depth because of their low performance compared to bipropellant options. For bipropellant systems, the two types studied were cryogenic and space storable. Cryogenic systems considered included liquid hydrogen/liquid oxygen(LH_2/LOX) and liquid methane/liquid oxygen(LCH_4/LOX). Storable propellant options included nitrogen tetroxide/monomethyl hydrazine(NTO/MMH). Other more exotic storable bipropellant combinations exist, but none have been extensively tested or flown.

The LH_2/LOX option was ruled out because of the requirement that the propellants be storable for up to two months. The cryogenic characteristics of the liquid hydrogen made propellant boiloff losses over two months significant. Considering the two remaining options, the LCH4/LOX combination has significantly higher performance than the NTO/MMH combination. Analyses showed that this performance increase resulted in significantly more mass delivered to the Moon's surface and to Earth reentry. Table 4.2 summarizes the results of some of the performance analyses performed using the Δv estimates from Table 4.1.

Table 4.2 Performance Estimates for Various Propulsion System Options

Oxidizer	Fuel	Engine	Isp (s)	Mass to Moon (kg)	Mass to Reentry (kg)
NTO	MMH	Transtage	315	39794	10702
NTO	MMH	OME/UR	340	42520	12622
LOX	LCH4	RL10A4-mod	376	46059	15340

Another factor considered which strongly influenced the design of the propulsion system was Mars Mission commonality. The project policy is to strive for commonality in systems in our project and the Mars Mission project. The Mars Mission is considering the use LCH4/LOX rocket engines as part of its propulsion system.

Based on the criteria discussed and the analyses performed, it has been decided that the best option for this mission is a pump-fed LCH4/LOX propulsion system. The decision matrix which was used to come to this conclusion appears in Table 4.3. This system, although much more complicated than the NTO/MMH system, has better performance and Mars Mission commonality.

4.2.2 Reaction Control System Options

Due to the need for restart capability, solid propellant systems were not considered for the RCS. Also, only propulsion systems which rely on chemical decomposition or reaction were analyzed, because other systems, such as arcjet, resistojet, and ion types, have thrusts that are too low for this mission. Nuclear thermal options were not considered for the same reasons discussed in section 4.2.1.

Only pressure-fed bipropellant and monopropellant systems were deemed feasible for this mission. Pump-fed systems were not considered because the added complexity would have reduced reliability dramatically, due to the large number of start/stop cycles the RCS must perform. Cryogenic propellants were not considered because these are used mostly in pump-fed systems and because no small cryogenic rocket engines are currently available.

A desire to use proven technology eliminates all space storable bipropellant combinations except for NTO/MMH. The monopropellant options considered include hydrazine and peroxide. Both hydrazine and peroxide have lower specific impulse ranges than NTO/MMH[4], although they are simpler and have lower mass than the bipropellant system. All the propellants considered are toxic, with the NTO/MMH being a hypergolic combination as well. The toxicity and hypergolic character complicate ground handling, but this propellant combination is so common that this complication is not viewed as a significant obstacle. The hypergolic character, however, was viewed as advantageous, since it negates the need for an ignition system, thereby enhancing reliability. The specifi impulse performance of the NTO/MMH systems is up to 100 seconds higher than either of the monopropellant systems. Based on this, and the extensive flight experience of NTO/MMH systems (including Apollo), a pressure fed, NTO/MMH propulsion system was selected for the RCS of both the ascent and descent stages.

Table 4.3: Decision Matrix Used to Select a Main Engine

Engine	Characteristics: Performance	Mass	Complexity	Mars mission commonality	Storability	Flight Experience	Cost	Totals
	7	*7*	*6*	*8*	*6*	*8*	*4*	
RL10 (LOX/LCH4)	7	3	5	10	6	3	7	268
RL10 (LOX/LH2)	9	5	5	0	1	10	8	195
Bell 6096	5	5	7	4	9	7	2	200
Lunar Module Descent	5	5	7	4	9	7	2	249
Advanced Space Engine	10	5	2	0	1	0	2	102

4.3 Rocket Engines

Based on the propellant choice, a modified RL10 has been chosen for the main propulsion system. The modified RL10 has a maximum thrust of 69.4 kN and a mass of 272 kg, and has both gimbaling and throttling capability. The maximum thrust level of the modified RL10 is such that at least two are required for landing. For redundancy, four engines have been baselined. If an engine fails, its opposite can be shut down, and the remaining two engines throttled up. In this case, the thrust vector would still pass through the center of gravity of the spacecraft. This is the only engine that the project knows of which has been tested with the LCH4/LOX propellant combination. The RL10 has the advantage of a long heritage, with many successful space firings on Centaur upper stages. A schematic diagram of the RL10 appears in Figure 4.1.[5,6] Miscellaneous engine performance parameters are outlined in Appendix B.

The choice for the RCS engines was based on availability, flight experience, mass, and thrust. For the RCS system, the tentative choice is the Marquardt R42 engine for the descent stage RCS, and the Rockwell SE-8 ablative thruster for the ascent stage RCS. Table 4.4 gives some of the characteristics of these engines. The choice of the R42 for the descent stage is preliminary, since the 890 N thrust is only an educated guess for what the RCS thrust requirement needs to be. The SE-8 is the same thruster used on the Apollo command module. The SE-8 was chosen primarily because it uses an ablative cooling method. This cooling method is important, since the thruster will be buried in the structure of the crew module and thus cannot be cooled radiatively.

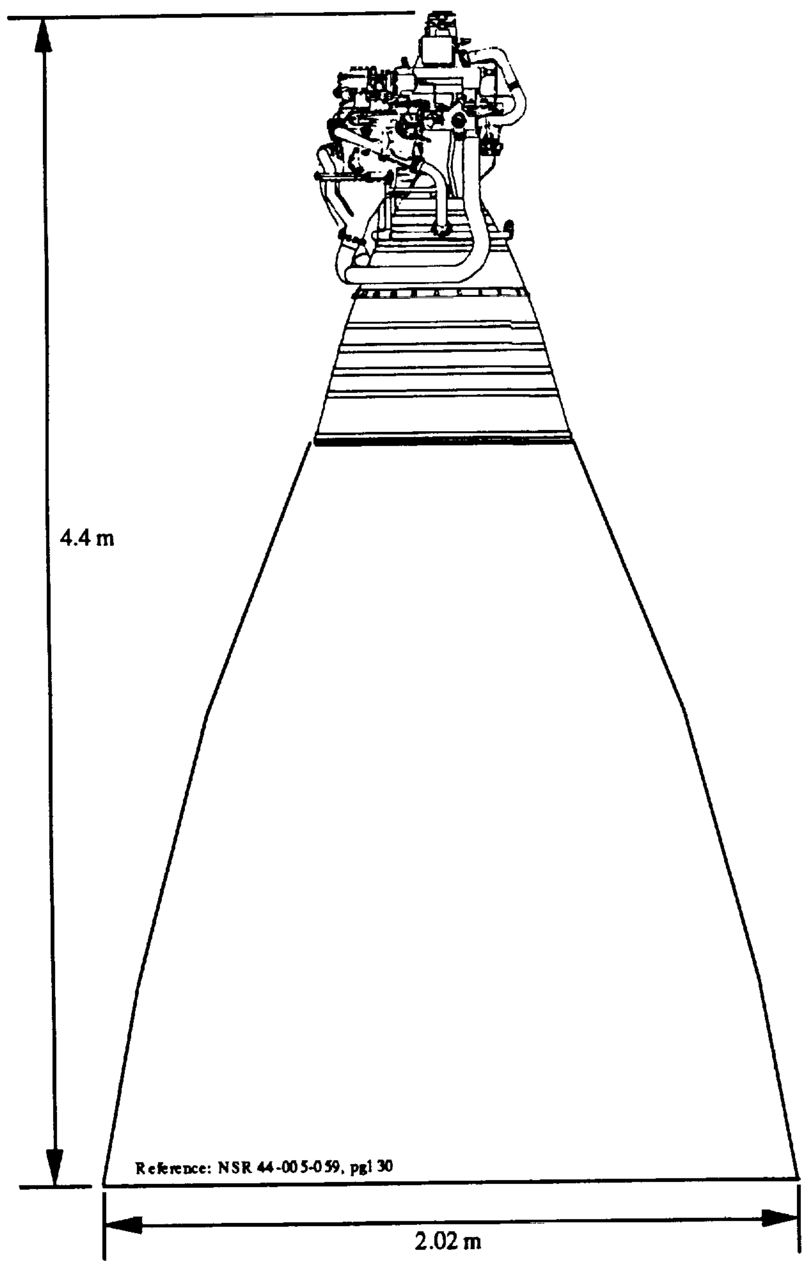


Figure 4.1 The RL10A4 Engine Modified for LOX/LCH$_4$ (Note the large nozzle extension)

Table 4.4 Performance Estimates for Various Propulsion System Options.[7,8]

Engine	Thrust (N)	Isp (s)	Mass (kg)	Manufacturer	O/F ratio (by mass)
R42	890	305	4.54	Marquardt	1.6
SE-8	414	273	3.69	Rockwell	2.1

4.4 Propellant Feed System Design

The propellant feed system must provide propellant to the rocket engines at a specified pressure and flow rate during all propulsive maneuvers. The feed system includes the propellant tanks, pressurant tanks, propellant lines, valves, filters, pressure regulators, turbines, pumps, and ground support equipment hookups.

4.4.1 Main Engine Propellant Feed System

The main engine propellant feed system must provide LOX and LCH4 to the engines during all major propulsive maneuvers. The system must accommodate at least eight start/stop cycles, and must be able to store the propellants for up to two months at a time.

The main engine propellant feed system uses turbine-driven pumps to provide high pressure propellants to the combustion chamber. The turbine is driven by methane used in the regenerative cooling of the engine. The propellant tanks are pressurized to 50 psia using a helium tank and regulator system. Helium-operated valves are used to isolate the propellant tanks from the engines between firings, and Pyro valves are used to isolate the propellant from the rocket engines during ground operations and launch, after which the

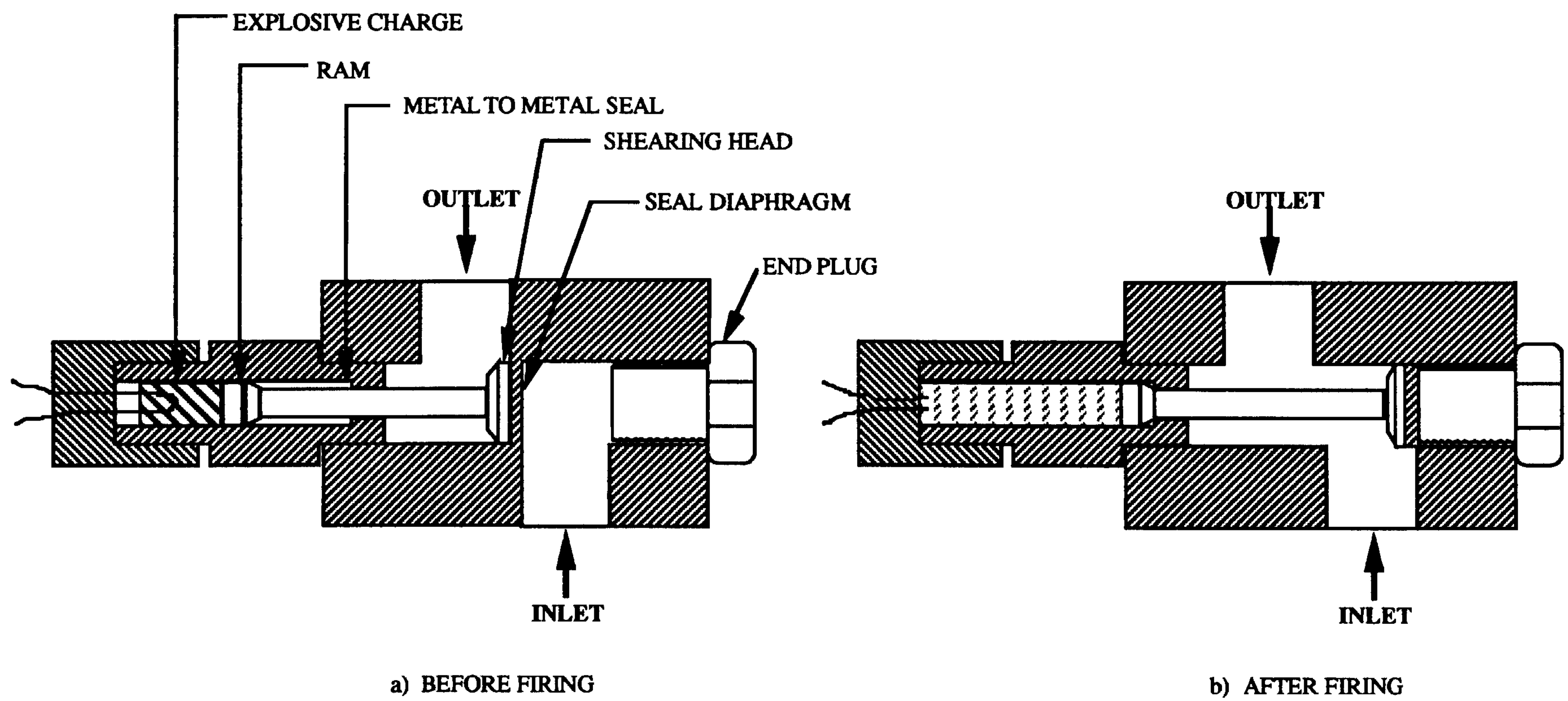
Pyro valves are fired to open the lines. Relief valves are manifolded to each tank to prevent tank rupture due to overpressure. A diagram of a Pyro valve appears in Figure 4.2.

The descent stage main engine propellant feed system also includes a helium pressurization system. The four oxidizer tanks and the four fuel tanks are linked manifolds. Each manifold is connected to a relief valve to prevent tank rupture due to overpressure. These valves are also used to vent boiloff of propellants. Schematics for the ascent stage and descent stage main propulsion systems appear in Figures 4.3 and 4.4. For clarity, the helium tank, lines, and valves necessary to operate the engine valves are not shown on the schematic.

4.4.2 RCS Propellant Feed System

The RCS propellant feed system must provide propellant to the RCS thrusters for all attitude control burns and for some abort maneuvers. The RCS must accommodate numerous start/stop cycles (100+) and must be able to start and function in both microgravity and high acceleration situations (i.e. the surface of the Moon), and must utilize storable propellants.

The RCS propellant feed system uses a pressure-fed design. High pressure gaseous helium tank(s) is (are) used to store the pressurant gas, which flows through a regulator into the propellant tanks. Since the engines must operate during various attitude maneuvers, including roll, yaw and pitch, the propellant tanks employ a positive expulsion system, thereby insuring that the propellant is always at the tank outlet. The tanks have burst discs/relief valves to prevent overpressure, and latch valves are used to isolate the thrusters from the tanks and the tanks from the pressurant tanks in between maneuvers. Pyro valves are used during ground handling and launch to isolate the



Reference: Huzel, pg 261

Figure 4.2 Diagrams of a Normally Closed PyroValve Before and After Firing. Note how the cutter head has trapped the seal diaphragm against the end plug.

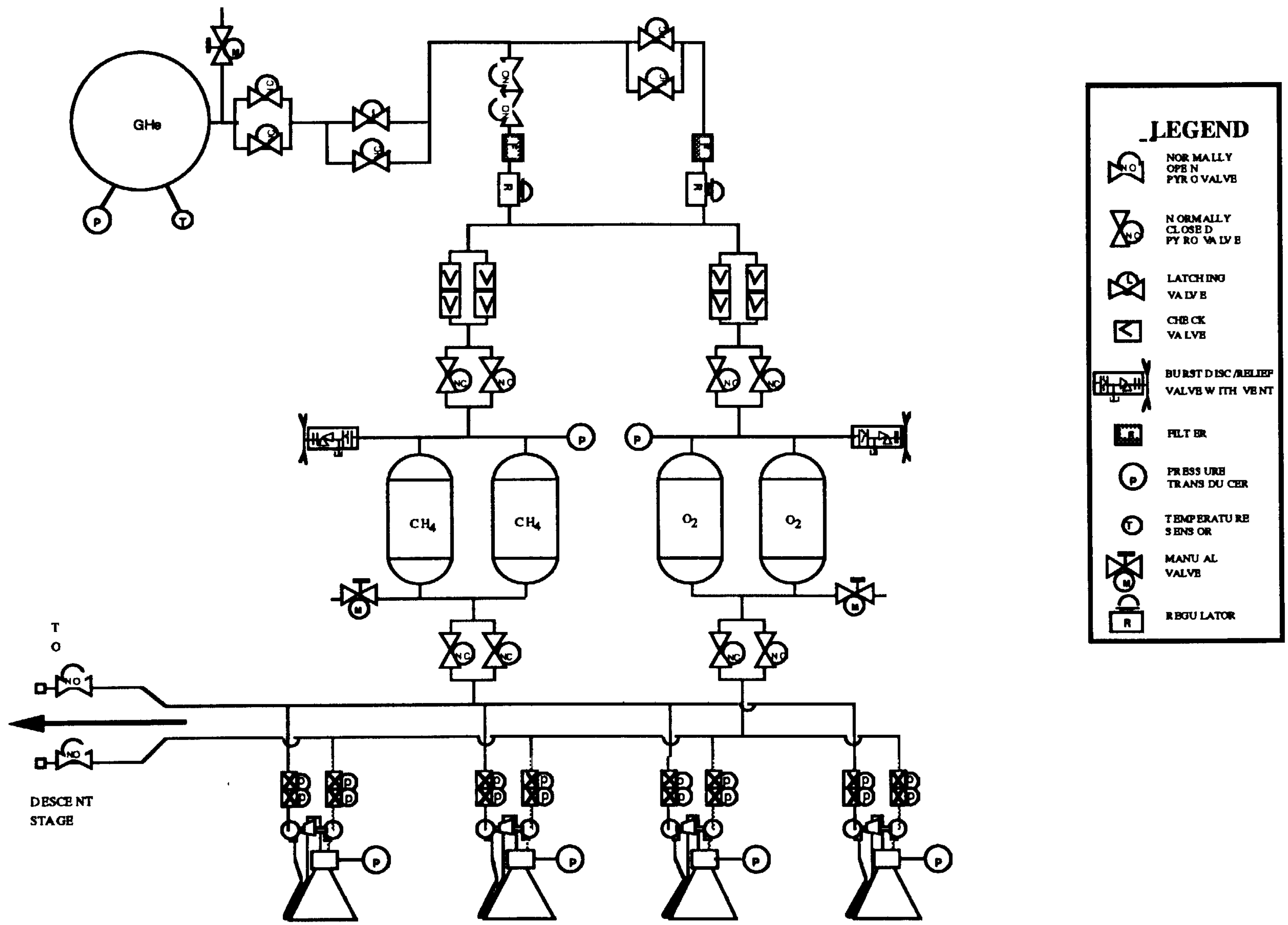


Figure 4.3 Ascent Stage Main Propulsion System Schematic

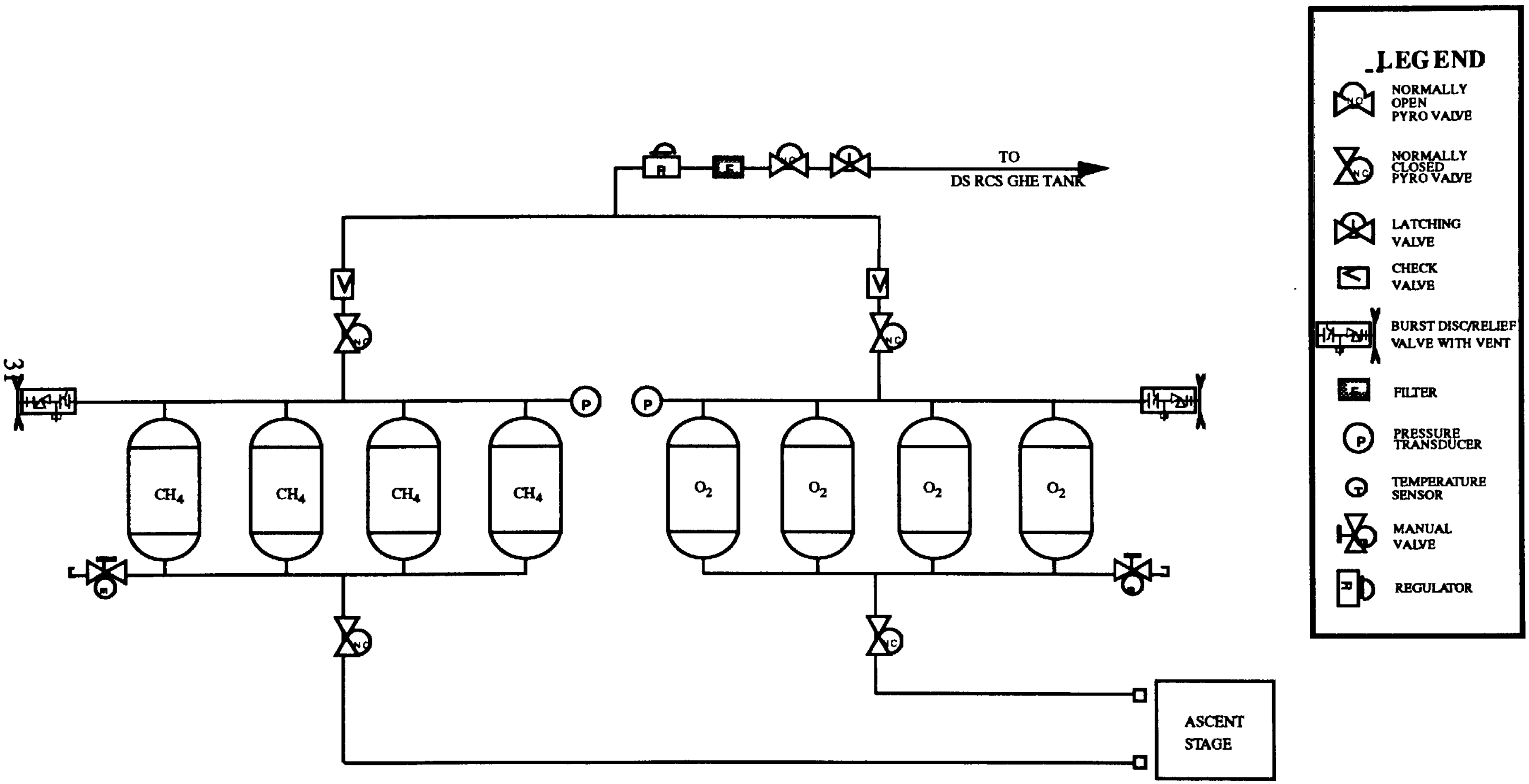


Figure 4.4 Descent Stage Main Propulsion System Schematic

pressurant gas from the propellant tanks and the propellant from the thrusters. Pyro valves are also provided in some places to circumvent a stuck-closed latch valve.

The spacecraft requires two RCS propellant feed systems, one for the descent stage and one for the ascent stage. The feed system for the ascent stage provides dual redundancy in many places to minimize single point failures. The descent stage feed system does not have as much redundancy, since the ascent stage must be used for abort maneuvers anyway.

A schematic diagram of the RCS for both the ascent stage and descent stage appears in Figures 4.5 and 4.6

4.5 Propellant Budget

Using the rocket equation[9], the ΔV estimates from NASA's FLO report, and the baseline propulsion system design, an approximate propellant budget was produced. Table 4.5 presents the budget; Appendix B shows the tk! Solver model used to calculate this budget.

4.6 Future Work

For future work, the attitude control requirements must be analyzed to more precisely determine the amount of RCS propellant required.

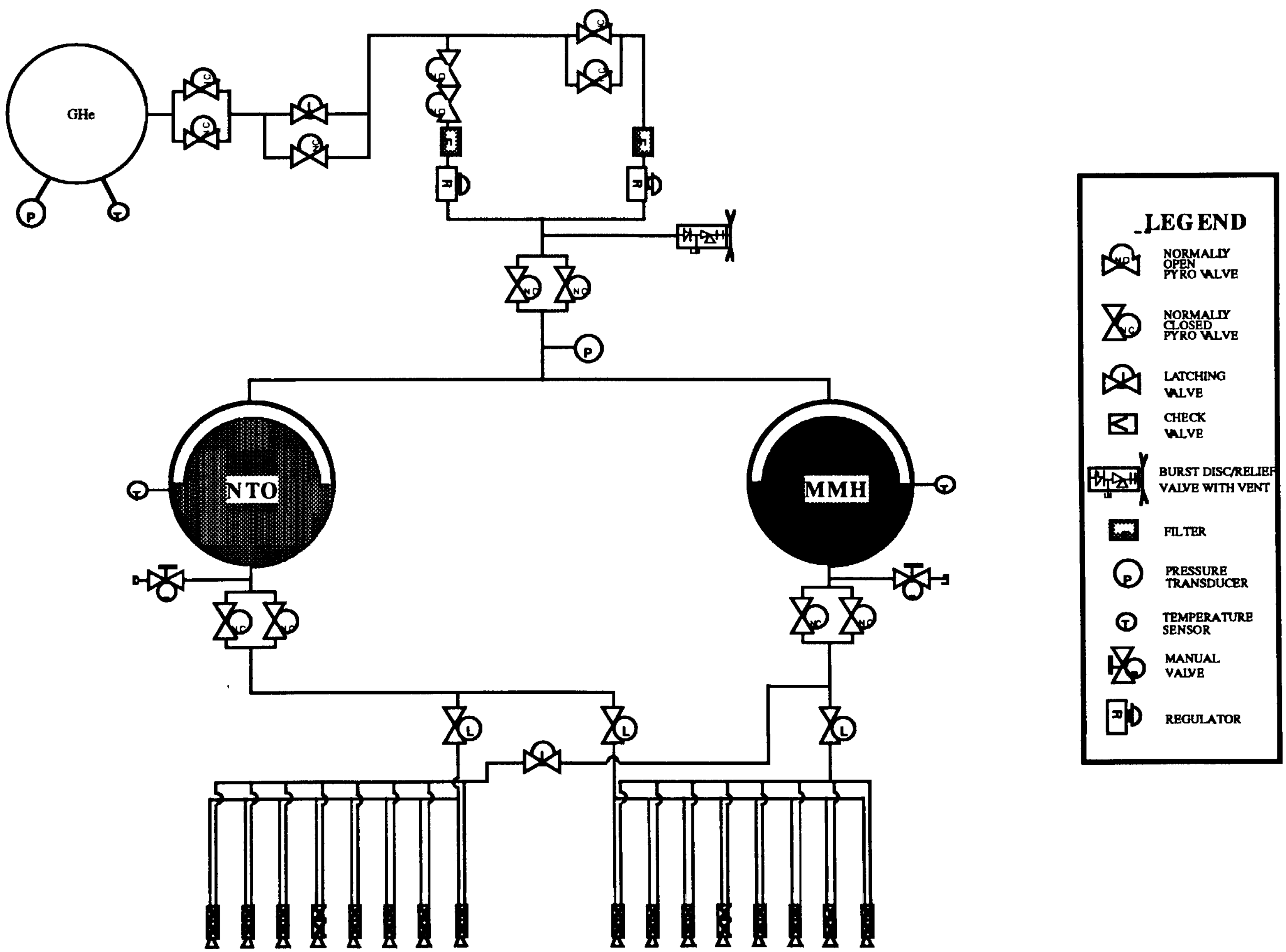


Figure 4.5 Schematic of the Ascent Stage RCS

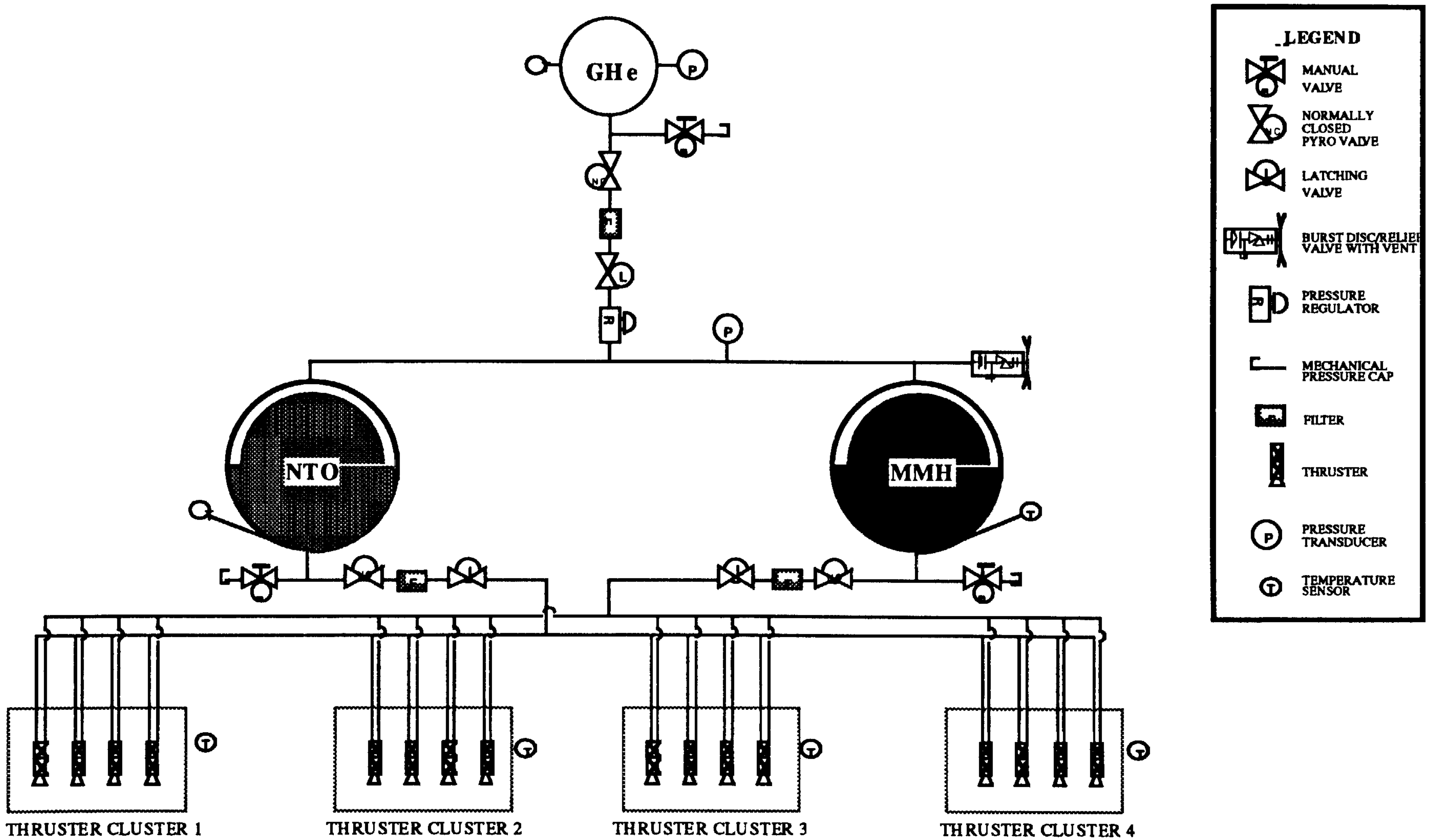


Figure 4.6 Schematic of the Descent Stage RCS

Table 4.5 Preliminary Propellant Budget for the Mission

Initial S/C Wet Mass (post TLI)	98000	kg
Descent Stage Dry Mass	6157	kg

Event	ΔV (m/s)	Propellant Expended MPS (kg)	Propellant Expended RCS (kg)	S/C Mass (kg)
				98000
TLI separation			0	98000
Mid Course Correction	30	793.826442		97206.1736
ACS transfer			0	97206.1736
				97206.1736
Lunar Orbit Insertion	882	20673.5946		76532.579
ACS LOI			0	76532.579
				76532.579
Lunar Deorbit burn	20	413.849671		76118.7293
Powered Descent	1878	30370.5237		45748.2056
ACS Lunar Descent			0	45748.2056
				45748.2056
Offload of Cargo				40748.2056
Descent Stage Separation				39591.2056
Propellant Boiloff		258		39333.2056
Loading of Cargo				34533.2056
				34533.2056
Lunar Ascent Burn	1826	13483.6972		21049.5084
ACS Ascent			0	21049.5084
				21049.5084
Trans Earth Injection	845	4309.68499		16739.8234
ACS Injection			0	16739.8234
Mid Course Correction	30	135.597087		16604.2263

g	9.81 m/s
Isp	376 s
Mdescent stage	6157 kg
Mboiloff	0 kg
Mcargo_to_moon	5000 kg
Mcargo_from_moon	200 kg

4.7 References

1. Langan, Michael P., et al., Mission Analysis Section. First Lunar Outpost (FLO) Conceptual Flight Profile. Engineering Directorate, Systems Engineering Division; NASA JSC, June 1992, p. A-1.

2. Wertz, James and Wiley Larson eds., Space Mission Analysis and Design. Kluwer Academic Publishers and Microcosm, Inc.: Torrance, 1992, p. 23

3. "Propulsion Subsystem Report and Database," Spacecraft Subsystems (Student Spacecraft Subsystems Descriptions). Department of Aerospace Engineering and Engineering Mechanics, The University of Texas at Austin: January 1993, p. 23

4. Sutton, George Paul, Rocket Propulsion Elements. Wiley: New York, 1992, p. 567

5. Wertz & Larson, p. 673

6. Personal Communication with C.D, Limerick, Pratt & Whitney.

7. Wertz & Larson, p. 646

8. "Propulsion Subsystem Report and Database," p. 81

9. Huzel, Dieter K. and David H. Huang, Modern Engineering for Design of Liquid-Propellant Rocket Engines. Washington, D.C.: AIAA, 1992, p. 12

5.0 Guidance, Navigation, & Control

5.1 Introduction

The guidance, navigation and control (GN&C) subsystem provides the spacecraft with the ability to determine the attitude and position of the spacecraft, calculate a path from one attitude and position to another, and control the spacecraft into that new attitude and position. This section of the report describes the preliminary GN&C system.

5.2 System Description

Inertial measurement units (IMUs) are used for attitude determination and, where applicable, acceleration measurements. For redundancy, the lunar lander has two IMUs, with the computer determining which IMU data to process. Four star trackers, stationed 90 degrees apart from one another along the circumference of the capsule, are used to realign the IMUs when they drift. Two radar altimeters are used to sense altitude above the lunar surface during powered ascent and descent.

The lunar lander uses 32 hypergolic thrusters for attitude control as well as translational control. More information about these thrusters is given in the propulsion section of this report.

The lunar lander uses autonomous and ground-based guidance. Autonomous guidance is used during time-critical events in the mission, such as the lower stages of powered descent and ascent, while ground-based guidance is used for other events, such as midcourse corrections. Powered descent is by automatic guidance, with provisions for manned guidance override.

GN&C software is processed by on-board computers. The lunar lander has multiple, independent computers for redundancy.

6.0 Environmental Control and Life Support System

6.1 Introduction

The Environmental Control and Life Support System (ECLSS) provides an atmosphere of tolerable pressure, temperature, humidity, and composition, food, water management for both sustenance and hygiene,waste management, and fire prevention and suppression. With such an important function, the ECLSS system is found in many different forms with varying capabilities and penalties.

The ECLSS is made from a combination of many components; however, systems in use today are usually classified as one of three types depending on the amount of recycling used. The "open" (or "open loop") system stores everything needed to maintain a compatible environment, and after use all of the resulting waste is either stored on board or jettisoned from the spacecraft. The "partially closed" system is similar to the "closed" system. In this case, though, Electrochemical Depolarized Cells (EDC) filter out atmospheric carbon dioxide (CO_2) and concentrate it for either removal from or storage aboard the spacecraft. The EDC produces electricity and heat as by-products of this CO_2 reduction process. All atmospheric moisture and most hygienic and potable (drinking) water is either stored directly or produced from fuel cells. Multifiltration is usually used to reclaim some of the waste water for hygienic (i.e., non-consumption) use. The "closed" system attempts to continually recycle all of the atmosphere and water necessary for life support. Commonly used elements include an EDC combined with a Sabatier reactor which converts CO_2 to potable water with methane and heat as waste products. Atmospheric oxygen is replenished by electrolyzing recovered wash water which also provides hydrogen necessary to run both the EDC and the Sabatier reactor. The remainder of the wash and waste water (including urine) is typically recovered by vapor compression distillation and used for non-consumptive purposes.

All of the systems briefly described above share two concepts. First, they all rely on the storage or ejection of solid waste produced by the crewmembers. Solid waste is stored in many ways, two of the most common being vacuum or freeze drying. Second, every system presented relies on the storage of foodstuffs rather than on board production. When food generation is self-contained inside the system, this is known as a "bio-regenerative" system. However, a bio-regenerative is not feasible for such a short-duration trip as this mission.

6.2 Requirements

Life Support requirements are based upon two criteria, the number of people that must be supported and the length of time they will have to be supported. NASA guidelines have clearly established the former criterion for this mission. The system must support four crewmembers. The latter criterion is not so well defined. The NASA requirements state that outside of a maximum lunar transit time of four days one way, the crew must be able to remain in the lander for a minimum of 48 hours on the lunar surface, and total lunar surface time is to be 45 days (one lunar day-night-day cycle). However, since the spacecraft is to be designed for an extended stay several factors come into play. First, to provide the crew with a habitation while preparing the FLO for occupation, life support capabilities in the lander have been extended to ten days. This leaves a maximum of 35 days in which the crewmembers will reside in the FLO, leaving the lander unoccupied. During this time, the lander will have to remain at some minimum state of readiness in case of emergency. Therefore, it is estimated that the ECLSS will have to run at approximately 25% capacity during this time in order to maintain a habitable atmosphere and allow for a rapid restart to full capacity if needed. Finally, when calculating mass, volume, power, and waste heat generated requirements, a 20% safety factor is added to ensure an adequate amount of life support capability.

6.3 Comparison of Systems

Preliminary sizing figures for the three different types of ECLSS systems described were obtained from a tk! Solver program.[1] The inputs were the criteria mentioned above, namely four crewmembers and a support capability of 38 days arrived at from the requirements discussed above. Both the program used and the input and output variables with their values are included in Appendix C. A note about the program worth mentioning is that one of the output variables is the mass of spares and consumables. Spares in this instance refers to both disposable parts like filters and redundant components in case of equipment failure. Consumables refers to food, oxygen, and other similar stores besides system hardware and spares.

The results of the sizing routine are clear. Figure 6.1 compares the total mass of each ECLSS option. The open system is the most massive with a total mass of 5544 kg, consisting mostly of consumables. The open system is 4.1 times as massive as the partially closed, and 8.6 times as massive as the closed. Of all the sizing results, this is the most significant. Mass in spacecraft terms translates directly to launch cost. In other words the open system is roughly 8.6 times more expensive to actually launch from the earth than the closed.

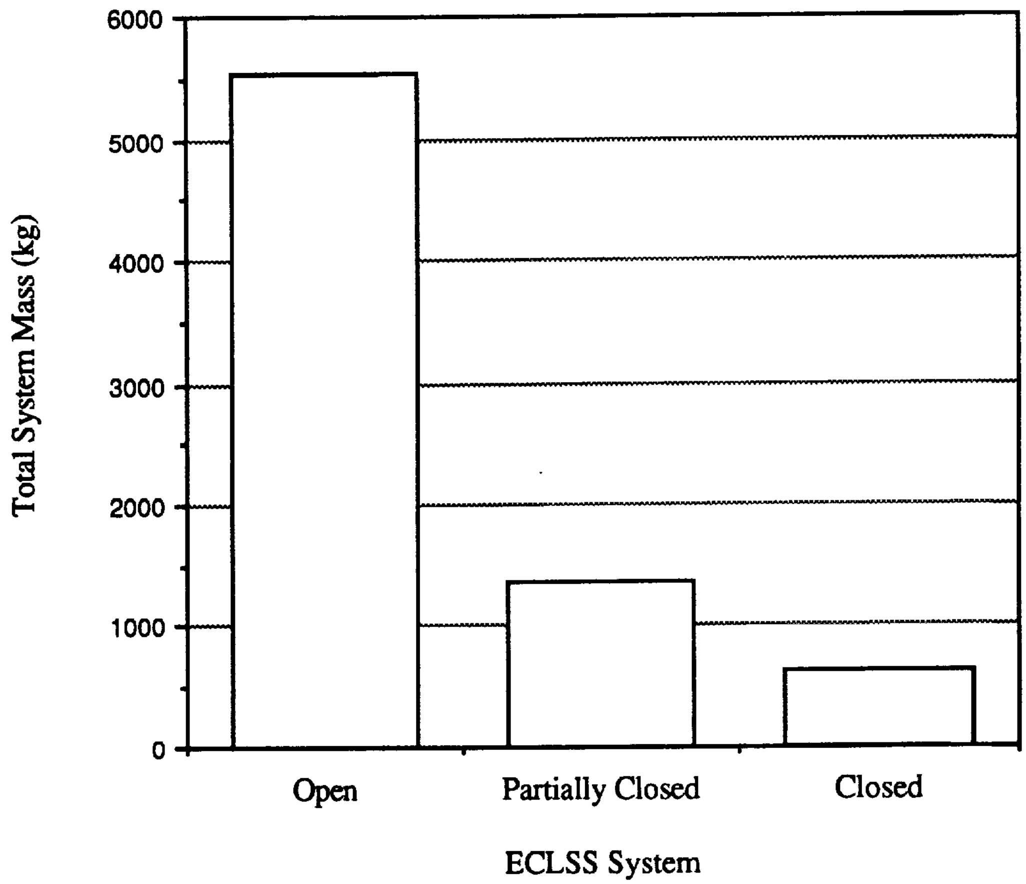


Figure 6.1 Comparison of ECLSS Total System Mass (4 Crewmembers, 38 Day Capability)

Figure 6.2 compares system volumes. Again, the need to store large amounts of consumables (primarily oxygen and hydrogen for air and water) with the open system results in a larger required volume. Preliminary sizing revealed that the open system required a volume of about 35.14 m^3, or 4.5 times that of the partially closed and 9.6 times that of the closed system. These results are important from a structural standpoint, because the more volume the system occupies, the greater the support structure needed to accommodate it. The increase in support structure means an increase in structural mass, which translates again into an increase in cost and possibly structural complexity.

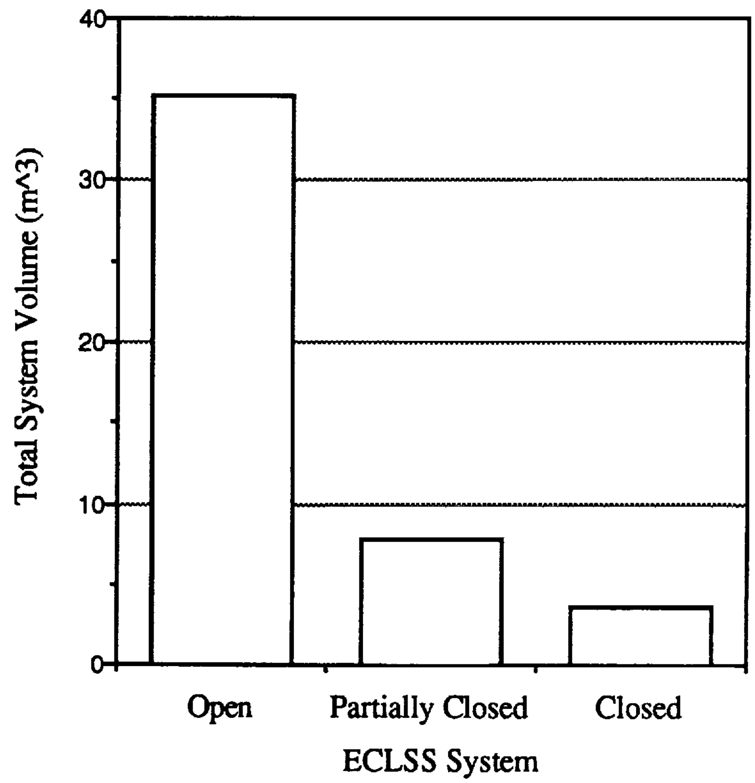


Figure 6.2 Comparison of ECLSS Total System Volume (4 Crewmembers, 38 Day Capability)

Figures 6.3 and 6.4 show the down side to systems that recycle part or nearly all of their wastes. Figure 6.3 is the comparison of system power requirements. This is calculated solely upon the number of crewmembers in the spacecraft and the resulting values represent peak power required (i.e., all systems being run simultaneously at full capability). In this case, the open system, being simpler in concept and requiring less hardware, requires less power, about 0.78 kW, or 79% of the partially closed and 44% of the closed system. These numbers are important from the perspective that power systems are limited in the energy they can produce. However, most power systems should have little problem handling the 1.79 kW required by the closed system. Waste heat generated, again dependent only on number of crewmembers, had the same trend as power consumption. As Figure 6.4 shows, the open system generated only 0.84 kW of heat at peak use. This is only 65% of the partially closed system's requirement and 54% of the closed system's requirement. Note that these numbers pertain only to the heat generated by the ECLSS system hardware directly and do not include the heat generated by computers and internal controls including lighting, which add significant amounts of heat that must be dissipated.

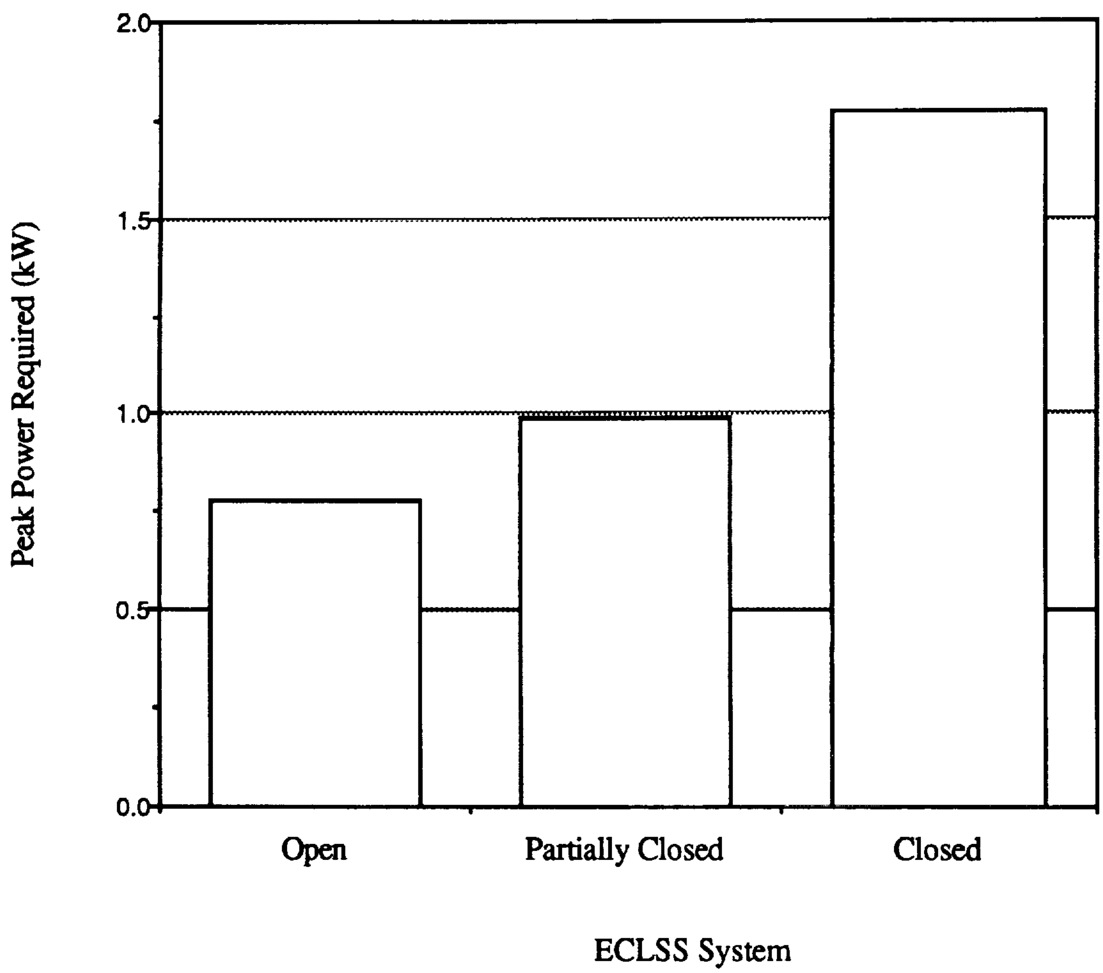


Figure 6.3 Comparison of ECLSS Peak Power Required (4 Crewmembers, 38 Day Capability)

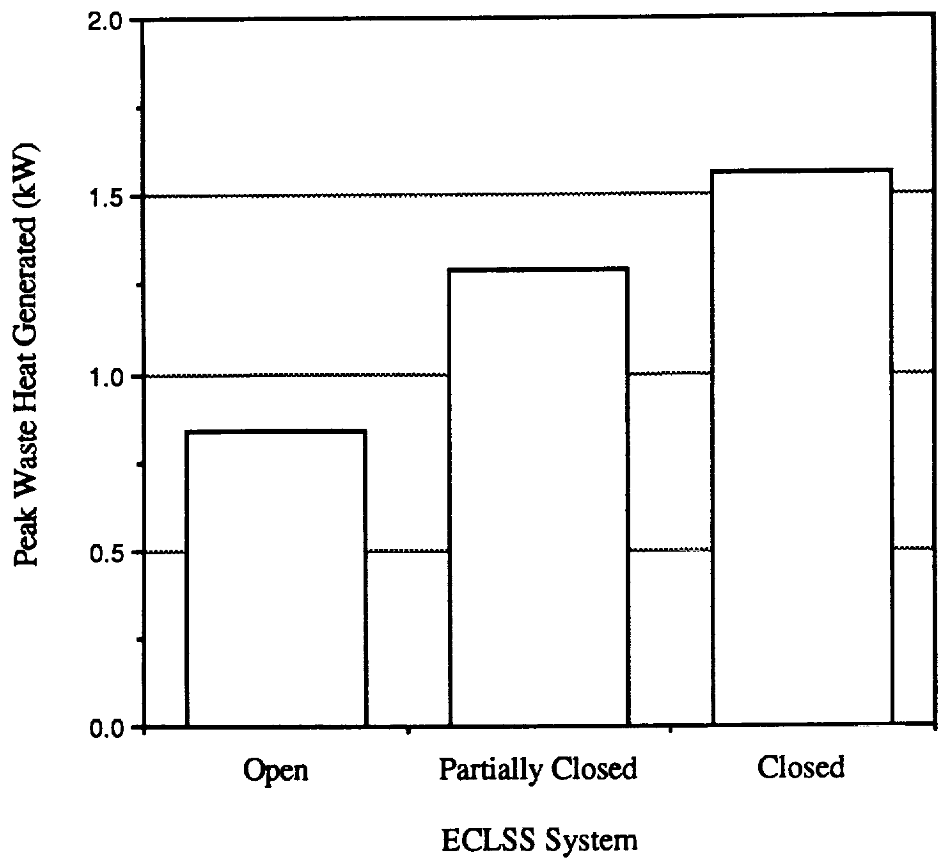


Figure 6.4. Comparison of ECLSS Peak System Waste Heat Generated (4 Crewmembers, 38 Day Capability)

6.4 Conclusions

Based on these numbers, a conclusion as to the type of system best-suited for this mission can be easily reached. Mass is the most important factor in overall spacecraft design. Even with the capabilities of the assumed HLLV, every kilogram sent into orbit and to the Moon is costly. The mass penalty combined with the proven ECLSS technologies used in the closed system make it hard to justify using an open system with 8.6 times the mass of a closed one. The same reasoning leads to the conclusion that the partially closed system is likewise less suited for this mission than the closed. Therefore, Selenium Technologies recommends the use of a closed ECLSS like the one described in the introduction to provide the life support functions for the crew.

Furthermore, Selenium Technologies recommends the use of a reduced pressure atmosphere of 34.5 kPa total pressure and partial pressures of 6.61-7.86 kPa O_2, 0.14 kPa CO_2, and 26.5-27.75 kPa N_2. The reason for this is to minimize or eliminatethe amount of pre-breathing time crewmembers will have to spend prior to leaving or entering the crew module. Other atmospheric conditions that have been established to maintain crew comfort are a temperature range of 18.3-26.7 °C, a dew point range of 4.4-15.6 °C, and a cabin ventilation range of 0.27-0.73 km/hr.

Two other recommendations are also made. First, the airlock on the crew module must be equipped with a vacuum disposal system to prevent lunar dust from EVA suits and equipment from contaminating the crew module and fouling the air revitalization system. One final recommendation is the inclusion of adequate fire detection and suppression in the module. Although fire does not tend to spread in the zero-g environment that will be encountered during transit, there is a potential fire hazard (especially electrical) during the lunar stay. Therefore, both smoke detectors and hand-held CO_2 fire extinguishers should be included in the crew module.

6.5 Reference

1. "Environmental Control and Life Support System," Spacecraft Subsystems (Student Spacecraft Subsystems Descriptions). Department of Aerospace Engineering and Engineering Mechanics, The University of Texas at Austin: January 1993

7.0 Active Thermal Control

7.1 Introduction

The active thermal control subsystem maintains all the components of the spacecraft within their temperature limits by either ridding them of excess heat or providing them with additional heat. This process is divided into three components: the acquisition component, the transport component, and the rejection component.[1] The first two components of the system are common for any environmentally controlled structure, but since the lunar day's environment is so hostile, a special system is required for the rejection phase on the Moon.

7.2 Requirements

This active thermal control system is designed to rid the spacecraft of a total of 7.5 kW of thermal energy. The ECLSS and power subsystems produce 1.5 kW and 3.0 kW of excess heat, respectively. The excess heat of the other subsystems used to size the radiators is based on previous studies of similar spacecraft.

7.3 Acquisition and Transport

The acquisition (the first function of the thermal bus) and the transport (the second function of the thermal bus) will use a separate loop system. The first loop of this system is the acquisition component. It consists of a one-phase water loop, which acquires the heat from (or gives heat to) all the components of the spacecraft whose temperatures need to be controlled.[2] The advantages of using water as the working fluid are that it is non-toxic, has high specific heat and extremely large heat of vaporization. Feasibility studies have shown that, even though the single-phase liquid system is more massive and requires more power than a two-phase system, it is more suitable for our system. Its simplicity and reliability have been proven numerous times in the past.

The second loop of the separate loop system is the transport phase. It consists of a two-phase ammonia loop. The advantages of using ammonia as the working fluid are that it requires less pumping power, has a small total weight, and requires smaller line sizes. This loop transports the thermal energy to the rejection system. The ammonia acquires the heat from the water in the acquisition system. Within the transport phase, the ammonia changes from a liquid to a gas. It is carried in this form to the rejection phase.

7.4 Rejection

The most difficult task for the active thermal control system is to reject the excess heat. Since the lunar surface phase is a more hostile atmosphere than the transit phase, a different system is required for heat rejection during the lunar surface stay.

7.4.1 Transit

During the transit period to and from the Moon, inflatable composite radiators will be used to reject excess thermal energy. Six radiators of two different sizes to maximize mass and volume efficiency will be located on retractable booms. During transit, the booms will be extended to their maximum length so that the radiators will get a maximum exposure to space and, hence, approach their maximum efficiency. The total mass of these radiators and booms is 102.5 kg. Their dimensions, amounts of thermal energy rejected, and ttemperatures of operation are as follows:

• 2 radiators with 1.5 m diameter, operating at 294 degrees Kelvin, rejecting 4.4 kW

• 3 radiators with 1.25 m diameter, operating at 275 degrees Kelvin, rejecting 3.3 kW

• 1 radiator with 1.5 m diameter, operating at either temperature, for redundancy.

The arrangement of the radiators on the spacecraft is shown in Figure 7.1.

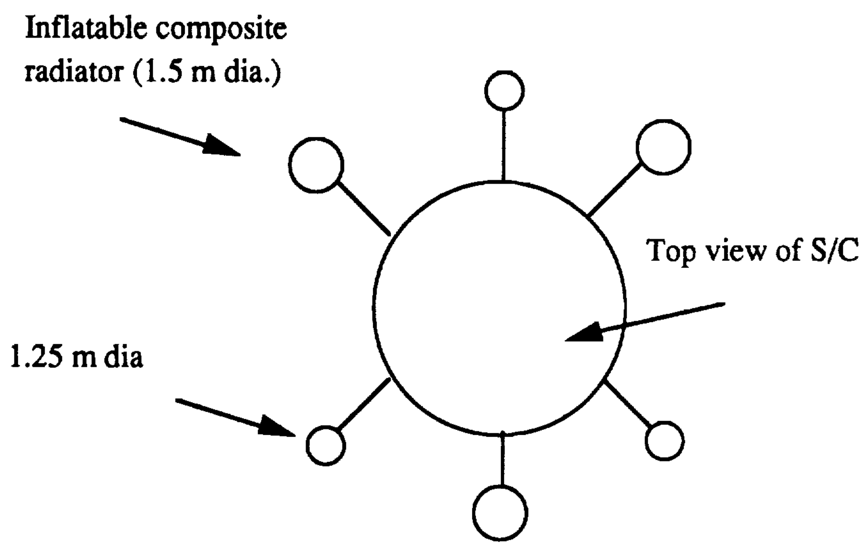


Figure 7.1 Arrangement of Radiators Around Lunar Lander[3]

7.4.2 Lunar Surface

During the lunar surface period, one vertical radiator along with a parabolic shading device will be deployed to the lunar surface to reject excess thermal energy. This system is shown in Figure 7.2. The system's total mass, including deploying equipment, is 207.55 kg. This mass is greater than for an unshaded system, but the system's efficiency is much higher than any unshaded system because its operating temperature of 286° K is much lower than for the 360° K operating temperature of the unshaded system. The total area of the radiator is only 15 m^2. Therefore, the efficiency of the shaded vertical radiator is much better than for the shaded horizontal system because it is able radiate from both sides of the radiator and take advantage of every bit of the system's area. The parabolic shading device blocks planetary infrared and reflects and focuses incident solar radiation above the vertical radiator located in the trough to insulate the radiator. The trough and radiator are positioned on the lunar surface so that the radiator is parallel to the plane of the Sun's path. This system is capable of rejecting 7.5 kW of thermal energy. The system will be stored in the cargo area during transit to and from the Moon. However, if so desired the system can be left on the lunar surface to allow for more cargo room on the return trip.

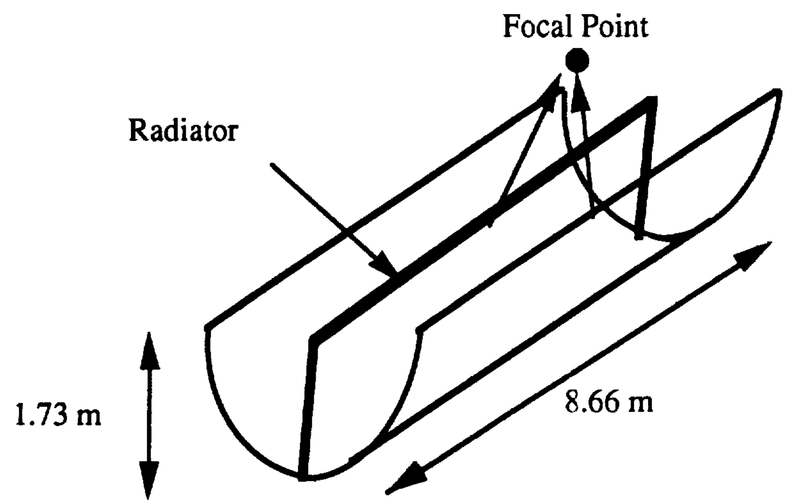


Figure 7.2 Vertical Radiator with Parabolic Shading Device.[3]

Both rejection systems are compatible with the acquisition and transport systems. The transfer from using the transit phase rejection system and the lunar stay rejection system should require little effort, even with the bulky EVA suits. After analysis of many available systems, the previously outlined active thermal control system was shown to be the most suitable system for this mission. Therefore, Selenium Technologies recommends that it be used on the Extended Duration Lunar Lander.

7.5 References

1. Guerra, Master's Thesis, "Active Thermal Control", University of Texas at Austin.

2. NASA, Engineering and Configurations of Space Stations and Platforms. Noyes Publications: Park Ridge, New Jersey, USA, 1985.

3. Ewert, Petete, Dzenitis, Advanced Environmental/Thermal Control and Life Support Systems. Active Thermal Control Systems for Space Exploration for a Lunar Base. Society of Automotive Engineers, Inc., 1990.

8.0 Communications Subsystem

8.1 Introduction

The communications subsystem provides communication links between all the components of the mission (i.e., spacecraft, earth). The communications subsystem needs to accommodate high and low data rates required for transmission of video, voice, science and telemetry, and command signals between the spacecraft and the ground stations on Earth. It is estimated that a high data rate of approximately 10 megabits/sec for Earth-Moon links is needed, mainly for transmission of compressed high-rate video signals.[1]

8.2 Communications Alternatives

A decision matrix used to rank the six alternatives for the communication subsystem is shown in Table 8.1. Alternative A uses frequencies in the S-band and a low-gain, wide-beam antenna for communications service during all phases of the mission. Alternative B uses the S-band for communication service while the spacecraft is in LEO, during descent to and ascent from the Moon, and during Earth reentry. During transfer to and from the Moon and during the lunar stay, the X-band is used for the communication link. Alternative C is similar to alternative B except that C uses frequencies in the Ka-band for the Earth-Moon link. Alternative D is also similar to alternative B except that D employs the Ka-band frequencies instead of the X-band frequencies during transfer to and from the Moon and during lunar stay. During transfer to and from the Moon and during lunar stay, alternative E generates optical links for communications service. It employs frequencies in the S-band while the spacecraft is in LEO, during reentry, and during

Table 8.1 **Communications Subsystem Decision Matrix**

ALTERNATIVES	A	B	C	D	E	F
LEO/REENTRY	**S-band**	**S-band**	**S-band**	**S-band**	**S-band**	**X-band**
TRANSFER TO/FROM	**S**	**X**	**X**	**Ka**	**optical**	**X**
MOON	**S**	**S**	**S**	**S**	**S-band**	**X**
DESCENT/ASCENT	**S**	**X**	**Ka**	**Ka**	**optical**	**X**
EARTH-MOON						
Reliability (5)	(5) 25	(5) 25	(3) 15	(2) 10	(1) 5	(5) 25
High data rates (5)	(2) 10	(4) 20	(5) 25	(5) 25	(5) 25	(4) 20
Continuous coverage (5)	(5) 25	(4) 20	(3) 15	(2) 10	(2) 10	(4) 20
Compatibility w/ ground station (4)	(5) 20	(5) 20	(3) 12	(3) 12	(1) 4	(5) 20
Low rain and cloud attenuation (5)	(4) 20	(4) 20	(3) 15	(3) 15	(1) 5	(4) 20
Mature technology (4)	(5) 20	(5) 20	(4) 16	(4) 16	(2) 8	(5) 20
Transponder power requirement (3)	(2) 6	(3) 9	(3) 9	(4) 12	(5) 15	(4) 12
Antenna (telescope) size and complexity (3)	(4) 12	(3) 9	(2) 6	(2) 6	(4) 12	(3) 9
System Mass (2)	(1) 2	(2) 4	(2) 4	(3) 6	(5) 10	(2) 4
Total # of Points	**140**	**147**	**117**	**112**	**94**	**150**

descent to and ascent from the Moon. Finally, alternative F uses the X-band during all phases of the mission.

The alternatives were ranked based on the requirements listed in the Table's left column. The requirements include: subsystem reliability, provision of high data rates, continuous ground station coverage by the antenna beam, compatibility with currently used communication networks, low rain and cloud attenuation of the communication links, mature technological development, transponder power requirement, antenna size and complexity, and subsystem mass.

These requirements are satisfied best by alternative F. Since this alternative uses frequencies in the X-band, the antenna beam width needs to be narrow in order to support high data rates with low power supply. However, a narrow beam may not provide enough coverage for a real-time, continuous link. Nevertheless, this problem can be solved either by employing a waveguide lens antenna that produces a single beam with multiple lobes, or a reflector with an offset switched feed array. A reflector with an offset switched feed array generates multiple beams or a single beam that is hopped or scanned over the Earth's surface.[2]

By using frequencies in the X-band, the communications subsystem avoids the overcrowded S-band for communications service. However, if the X-band link fails during any phase of the mission, the fail-soft design permits autonomous switching to an omnidirectional antenna and the S-band in order to reestablish the continuous link during the remaining part of the mission, or until the X-band link can be recovered.[3]

8.3 Frequencies

Table 8.2 shows the frequency ranges that are used by the communications subsystem. In the X-band, frequencies between 7.145 GHz and 7.190 GHz are used for the uplink transmission, and frequencies between 8.4 GHz and 8.5 GHz are used for the downlink transmission. The corresponding downlink-to-uplink carrier frequency ratio is 880/749. The S-band frequencies would be used only if the X-band link fails during the mission.[4] The above carrier frequency ratios and frequency ranges are different from those used by commercial broadcasting services.

Table 8.2 **Frequency Ranges Used by the Communications Subsystem**

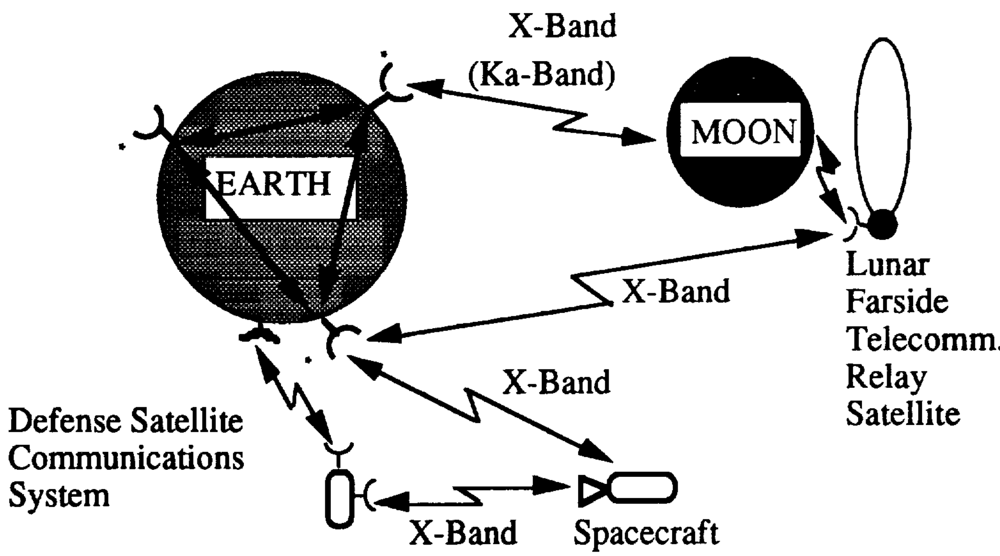
Frequency Band	**Uplink (GHz)**	**Downlink (GHz)**	**DL/UL Carrier Frequency Ratio**
X-Band	7.145 - 7.190	8.400 - 8.500	880/749
* **S-Band**	2.025 - 2.120	2.200 - 2.300	240/221

* Frequencies used in case X-band link fails during mission

8.4 Communications Architecture

The above frequencies and carrier frequency ratios are compatible with those used by the Deep Space Network (DSN)[4] and the Defense Satellite Communications System (DSCS)[5]. The DSN supports the communication link with the spacecraft during all phases of the mission as shown in Figure 8.1. If, in case of an emergency, the DSN is not available, the DSCS can be used to provide the link.

In case the mission requires that the spacecraft land on the far side of the Moon or in case of an emergency descent on the Moon, a lunar farside telecommunications relay satellite may be needed to provide the link between the spacecraft and the ground stations on Earth. This relay satellite would be placed in a halo orbit at the far side of the Moon, as shown in Figure 8.1. The relay satellite is a component of the complex communications network planned for the lunar and Mars missions of the Human Exploration Initiative[6].



* Deep Space Network

Figure 8.1 Communications Architecture

8.5 Recommendations for Future Work

The preliminary design of the communications subsystem should allow for further modifications. For example, if higher data rates are needed for the Moon-Earth link, it may be necessary to use higher frequencies in the Ka-band.

Also, communication networks other than the DSCS should be considered for backup support of the communication link in case the DSN is temporarily not available.

Furthermore, optical systems should be considered for establishing optical communication links during some parts of the mission. In general, optical links provide higher data rates than microwave links. Also, optical communication systems are usually lighter and smaller, and require less power supply than their microwave counterparts.[7] However, optical links are seriously attenuated by rain and clouds, which may result in discontinuities in the communication link[8]. Furthermore, the technological development of optical systems for space applications is still in the early stage. More work needs to be done before optical systems will be able to provide reliable communications service during manned space missions. Nevertheless, this mission should be considered further at least for experimental establishment of innovative communication links, such as those generated by optical systems.

8.6 References

1. "Report of the 90-Day Study on Human Exploration of the Moon and Mars," NASA News. National Aeronautics and Space Administration: Washington, D.C., November, 1989, p. 5-20.

2. Wertz, James and Wiley Larson eds., Space Mission Analysis and Design. Kluwer Academic Publishers and Microcosm, Inc.: Torrance, 1992, p. 542.

3. "Report of the 90-Day Study on Human Exploration of the Moon and Mars," p. 5-19.

4. Wertz and Larson, p. 516.

5. Fthenakis, E., Manual of Satellite Communications. McGraw Hill: New York City, New York, 1984, p. 332.

6. "Report of the 90-Day Study on Human Exploration of the Moon and Mars," p. 5-24.

7. Hauptman, R., "High Data Rate Atmospheric and Space Communications", Proceedings of SPIE-The International Society for Optical Engineers, Vol. 996. SPIE-The International Society for Optical Engineering: Bellingham, Washington, September, 1988, p. 4.

8. Wertz and Larson, p. 550.

9.0 Structures

9.1 Introduction

The structures subsystem provides a mechanical support for the other subsystems on the spacecraft. The structure of the spacecraft is formed by the primary structure, which carries the major loads, and the secondary structure, which provides support for different spacecraft components.

The structures subsystem group is responsible for all the areas of the design that are related to the structure of the spacecraft. These areas include ascent and descent stage configuration, crew cabin configuration, propellant tanks design, materials selection, and mass estimates.

9.2 Lunar Lander Configuration

The overall lunar lander configuration for the mission, shown in Figure 9.1, is formed by the ascent and the descent stages. The ascent stage is embedded in the descent stage, which is left on the Moon after the mission has been completed. The descent stage consists of the descent truss structure, propellant tanks for the descent, and the cargo. The ascent stage is formed by the RL10 engines, ascent truss, ascent propellant tanks, fuel cell tanks, and the crew module.

The configuration of the lunar lander is driven by the desire to have two separate stages which share one propulsion system. The overall dimensions of the lunar lander are driven by the constraints on the payload area of the HLLV. The height of the lunar lander is 16.2 m and the diameter is approximately 11 m.[1]

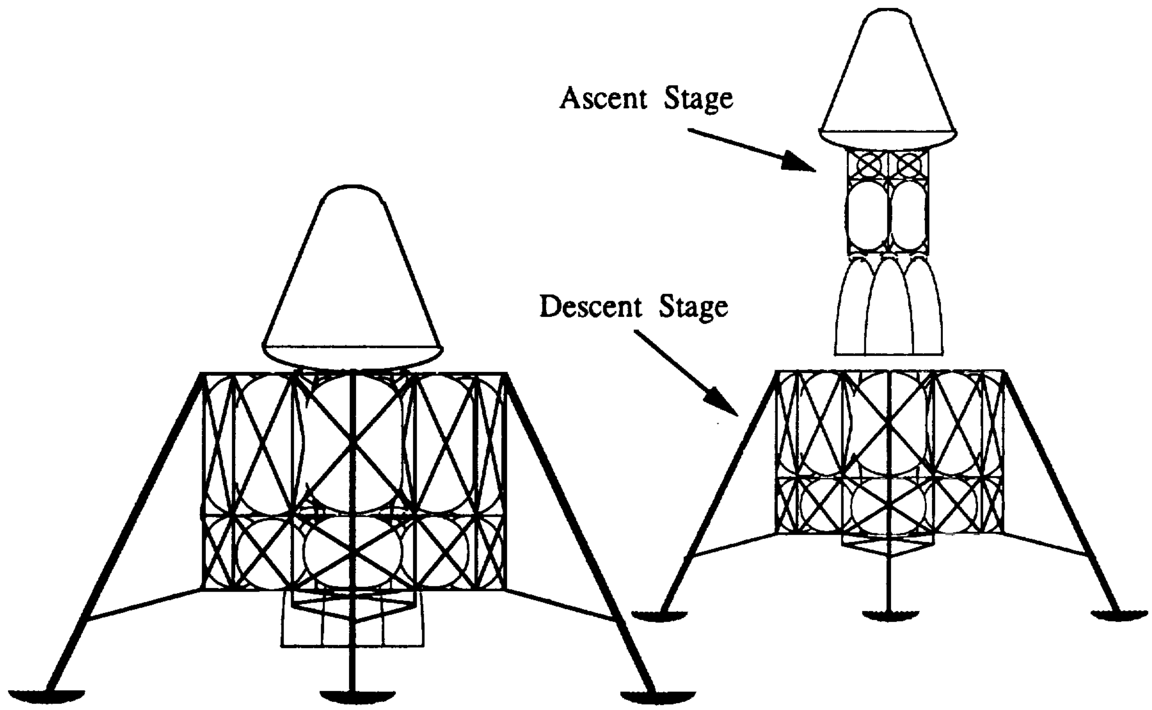


Figure 9.1 Lunar Lander Configuration

9.2 Descent Structure

The descent truss is composed of cylindrical aluminum members, with titanium end fittings for additional support. The structure holds all the fuel and oxidizer necessary for descent. This fuel will be linked to the ascent stage, connecting to the RL10 engines. The truss also houses the mission cargo and extra area for any life support or power equipment that cannot be contained within the crew module or in the ascent truss.

Figure 9.2 shows a top view of the descent stage with the legs deployed. As shown in the figure, the descent stage has a platform on top of it so that the astronauts can walk around the module when they are on the surface of the Moon. Mounted on this platform are two solar arrays that will be deployed during the lunar suface stay to provide power. These solar arrays are shown retracted in the figure, but they will be extended outwards to form a square with 6.2 m sides. There is also an elevator mechanism shown in this figure. The mechanism allows the astronauts to descend to ground level and go back up, without using the ladder that is mounted on one of the legs. The mechanism uses a small platform to go up and down that fits within the descent truss structure. The platform is lowered or raised with cables that are connected to winches mounted on top of the descent stage. The whole mechanism is powered electrically, but there are handles in the winches as a mechanical backup system in the case of power failure.

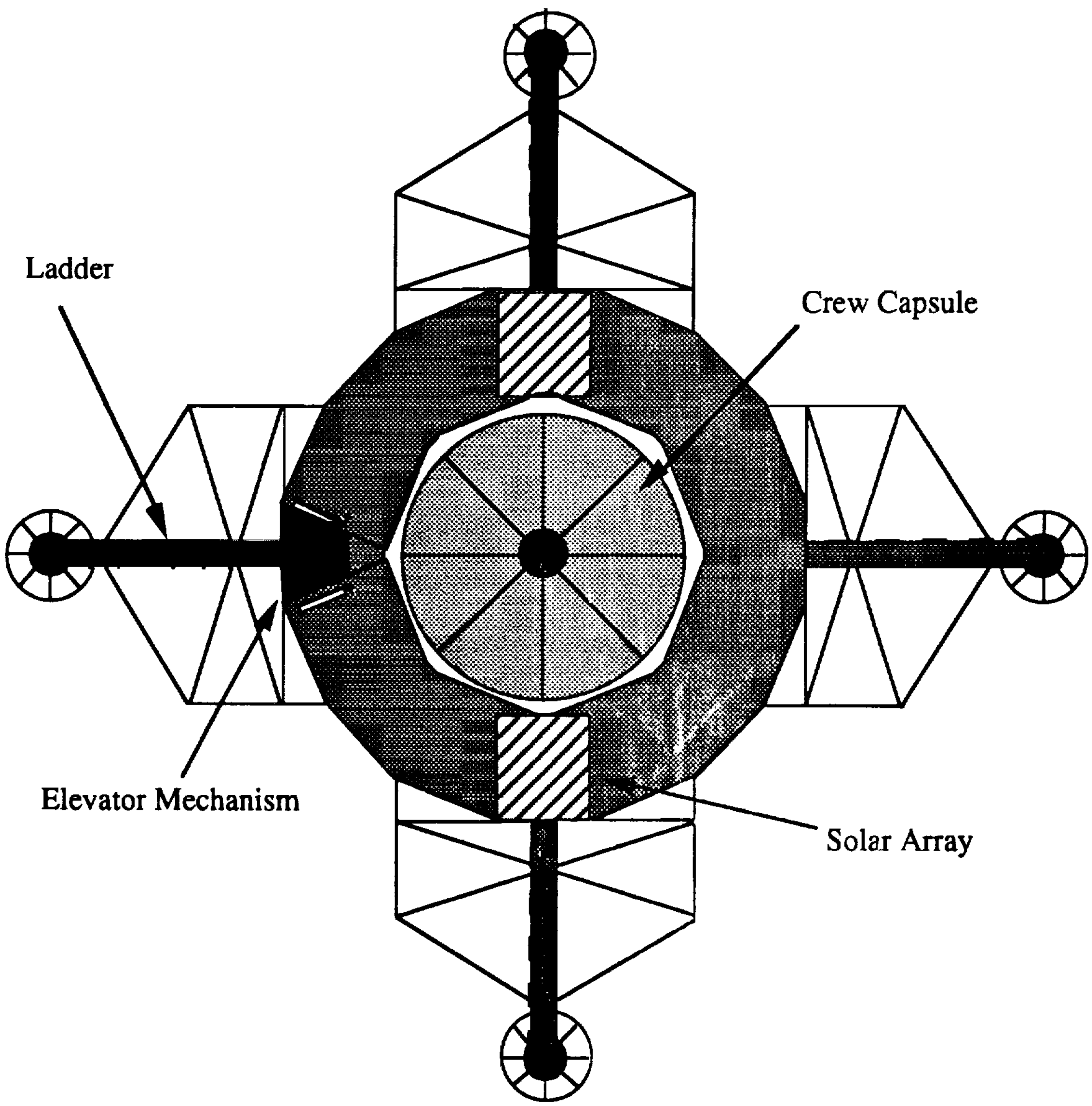


Figure 9.2 Descent Stage Top View

Figure 9.3 gives a top view of the outer truss. The structure is octagonal, with the fuel and oxidizer tanks arranged in a ring surrounding the inner diameter of the truss. The inner diameter of the truss is 6.3 m, which gives the ascent stage 0.6 m clearance. The outer diameter of the structure is 10.8 m in order to fit within the HLLV cargo constraints. Attached to the truss are four retractable lander legs, which give the descent structure a diameter of 20.8 m when the legs are fully extended.

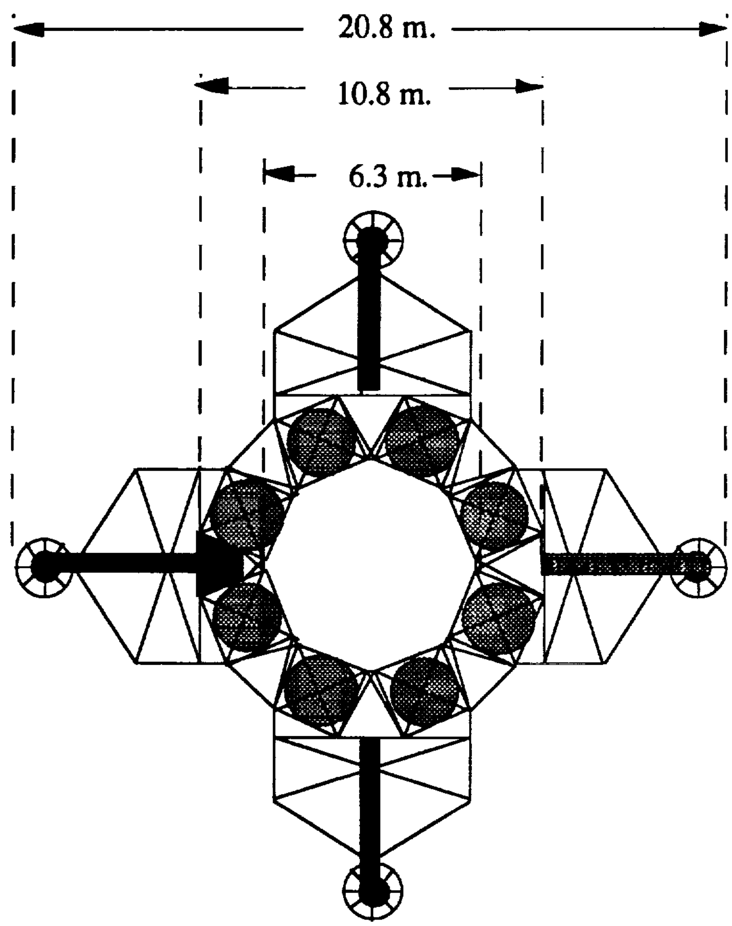


Figure 9.3 Descent Truss Top View

Figure 9.4 shows a side view of the descent structure. The fuel and oxidizer tanks are arranged in a ring around the inner diameter of the truss. The oxidizer and fuel tanks are placed contiguously at the top of the structure and occupy a height of 4.2 m. The bottom 2.5 m of the truss is allocated for cargo and the deployable radiators during lunar transit. The cargo bay employs a pulley system which lowers all the cargo to the lunar surface.

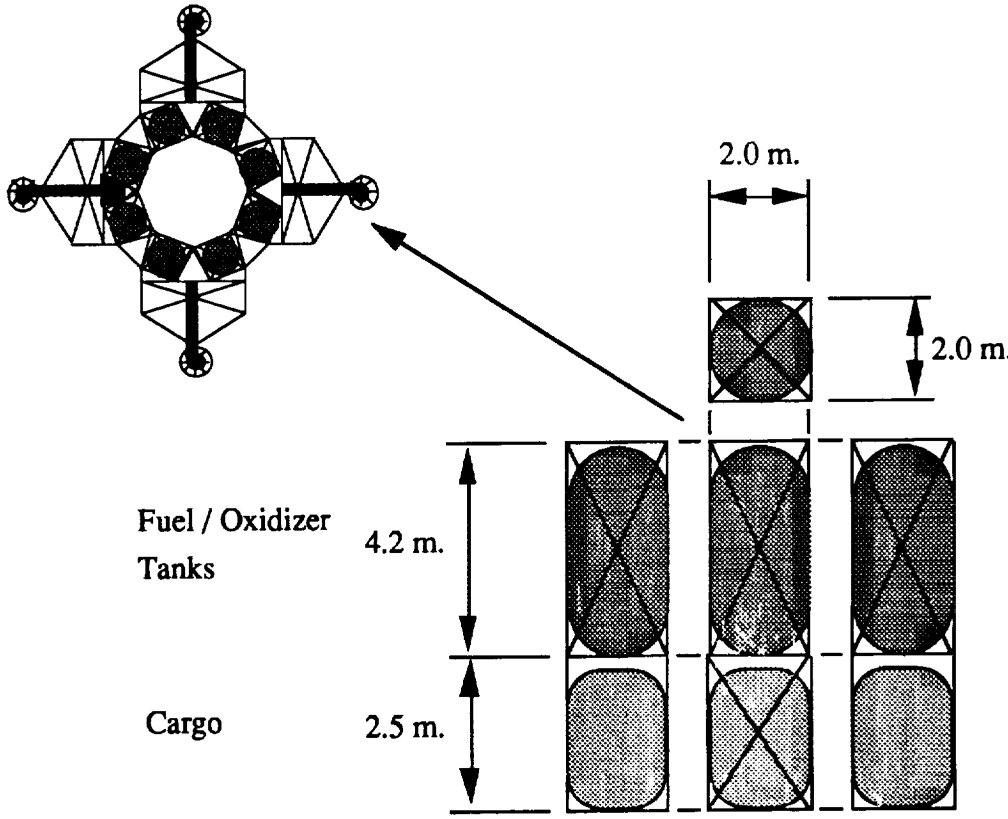


Figure 9.4 Descent Truss Side View

Figure 9.4 also shows the width and length of each section on the descent truss. Both of these dimensions are 2 m long, providing enough space to house the large oxidizer tanks.

9.3 Ascent Stage

The ascent stage can be divided into two major components: the ascent truss structure with the engines and the propellant tanks, and the crew module.

9.3.1 Ascent Truss Structure

The ascent truss structure is formed by the ascent truss, propellant tanks, fuel cell tanks, and the RL10 engines. As shown in Figure 9.5, the total height of the ascent truss structure is 10.2 m. This distance is given by the fuel cell and propellant volume requirements, the height of the RL10 engines, and the necessary ground clearance for the exhaust nozzle. With this design, the total height of the truss is 4.3 m, divided in two sections of 1.2 and 3.1 meters. The top section contains the fuel cell tanks, while the bottom section houses the propellant tanks. Finally, the RL10 engines have a total height of 4.4 m, which leaves a nozzle ground clearance of approximately 1.5 m.

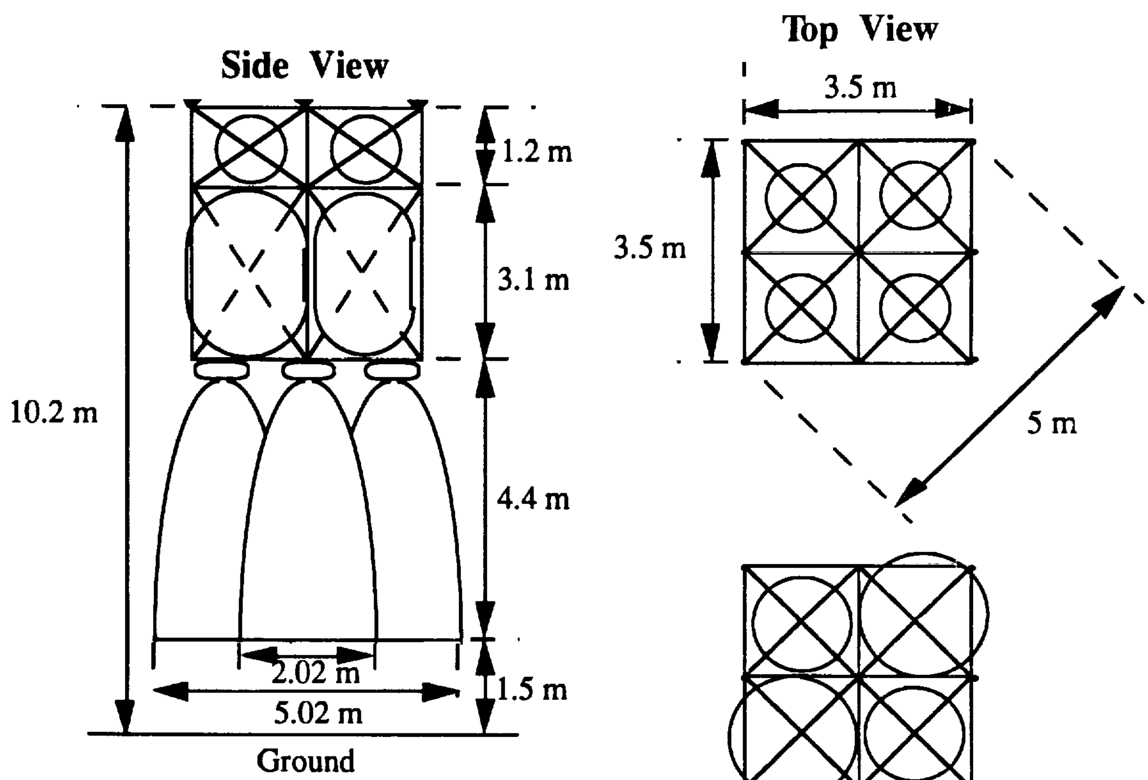


Figure 9.5 Ascent Truss Structure

The top view of the ascent truss shown in Fig 9.4 has two different cross-sections. The cross section on top shows the fuel cell tanks, with two hydrogen tanks, one oxygen tank, and one water tank. The cross-section on the bottom shows the propellant tanks, with two big tanks for the oxidizer and two smaller tanks for the fuel.

The ascent truss structure is a square with 4 m sides, as shown in Figure 9.4. This figure also shows that the maximum width of the ascent structure is 5.02 m, which gives approximately 0.6 m of clearance between the ascent and the descent truss structures at the closest point between the two. The members that form the ascent truss have been designed to carry major loads experienced during the mission, including bending, torsion, and compressive loads that produce buckling. Finally, the materials used in the ascent truss are aluminum for the truss members and titanium for the fittings that connect these members.

9.3.2 Crew Module Configuration

The crew module for the baseline lunar lander, as shown in Figure 9.6, is a capsule similar in shape to the proven Apollo Command Module. The crew capsule measures 6 m in height and 6 m in diameter at the base. The interior walls of the module will be mainly composed of an aluminum honeycomb material with additional aluminum support beams. The exterior of the cone is covered by HTP-6 tiles, an advanced form of the Space Shuttle's protective tiles. The base of the module is covered by an ablator, which is the primary thermal protection during the reentry into the Earth's atmosphere. As shown in Figure 9.7, the thickness of the cone protection is 7.5 cm, while the ablator has a thickness of 15 cm since the base of the crew capsule will reenter the atmosphere first and absorb most of the extreme heating that occurs during the reentry.

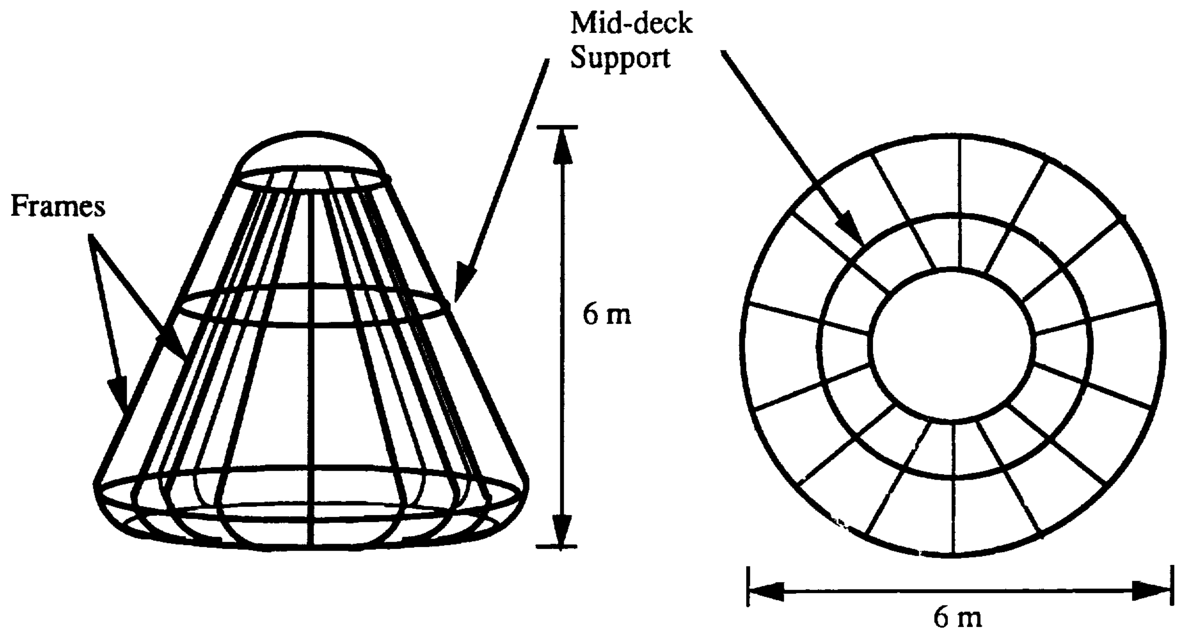


Figure 9.6 Crew Module Frame

The crew cabin interior is divided into two decks, with the command deck on top and the habitat deck on the bottom. The command deck measures 2.2 m in height and 1.6 m in radius at the base. This deck has a hatch used by the astronauts while on Earth to get in and out of the crew cabin, and it can also be used as an escape hatch in case there is an emergency landing in water. The habitat deck measures 2.8 m in height and 2.9 m in radius at the base. This deck is where the astronauts stay during lunar transit and before transferring to the FLO habitat. This deck has an airlock 2.1 m high and 1.6 m in diameter, which eliminates the need for depressurizing the entire crew module at the beginning and end of each EVA.

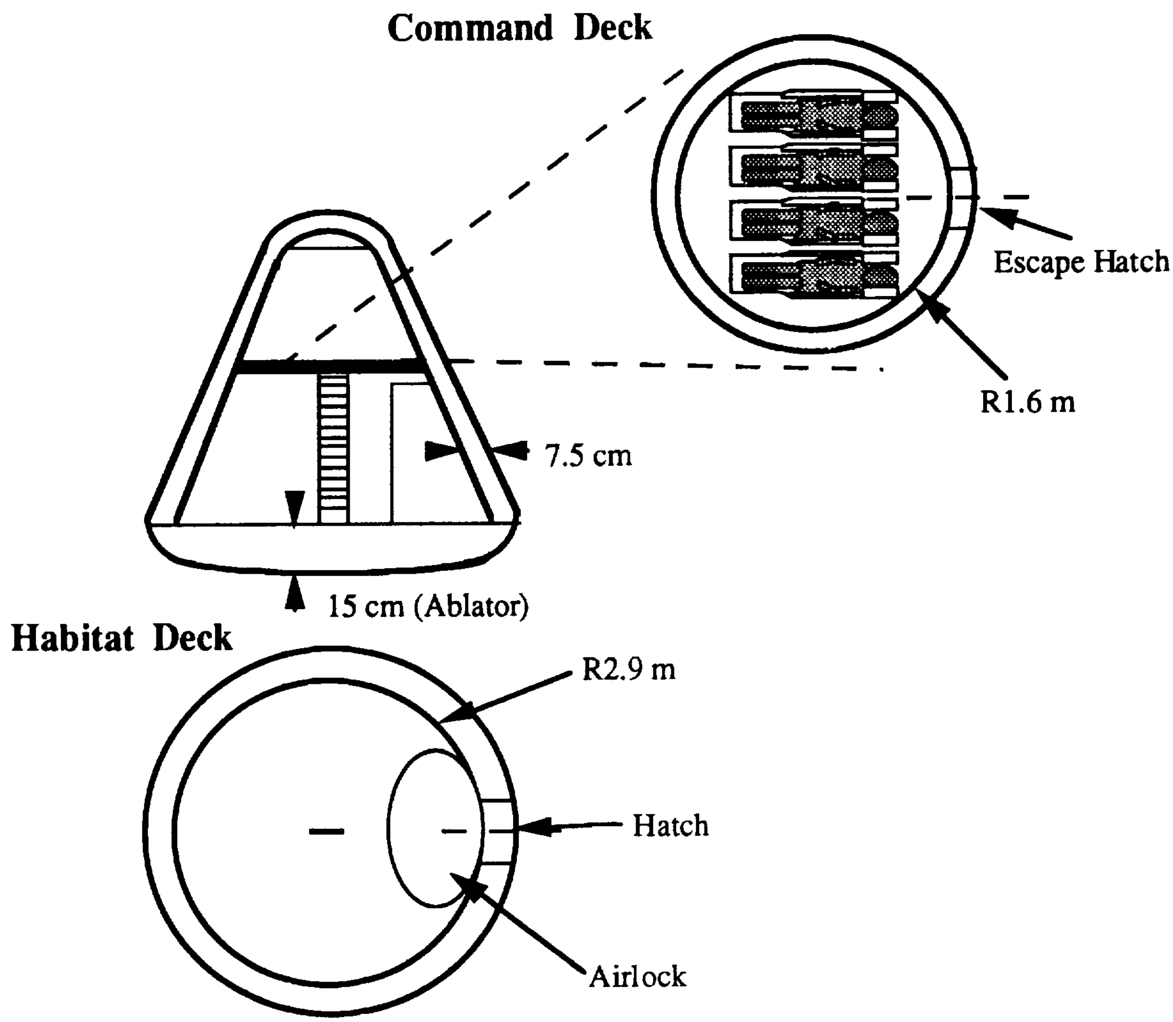


Figure 9.7 Crew Module Interior

9.4 Ascent-Descent Connections

Figure 9.5 is a schematic of the connections between the ascent and the descent stage. As can be seen in the top view of the figure, the total number of connections between the ascent and the descent in each connecting area is 16. These connections are simply aluminum rods that connect the descent stage with each grid point in the ascent stage, except for the center grid point.

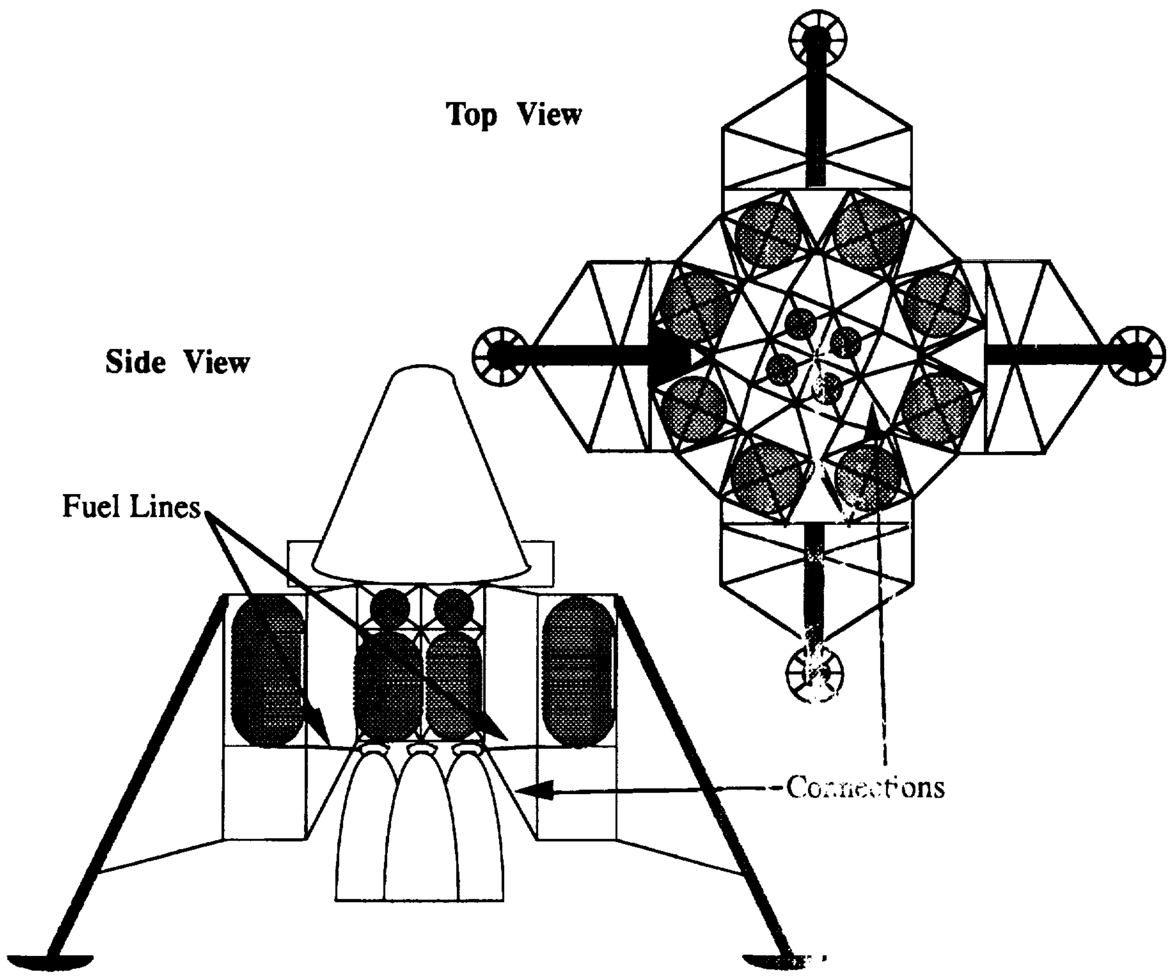


Figure 9.5 Ascent-Descent Connections

The side view of Figure 9.5 shows the two connecting areas between the ascent stage and the descent stage, one at the top and the other one at the bottom of the structures. Since there are two connecting areas between the ascent stage and the descent stage, the total number of connections is 32. These connections are all tilted, as shown in the figure, in order to decrease the bending loads transmitted by the connecting rods to the descent structure.

Finally, the figure also shows the fuel lines connecting the propellant tanks in the descent stage with the RL10 engines. These lines are placed at the bottom of the structure to reduce the length of piping, and they are tilted to provide gravity feed in case the pressure feed system fails.

9.5 Materials

One of the major tasks of the structures subsystem is to analyze the different materials suitable for space applications and select the ones that offer the best results. Some of the material properties that need to be considered in the selection of the materials are[2]:

- strength to density ratio
- stiffness
- stress corrosion resistance
- fracture and fatigue resistance
- thermal characteristics
- sublimation
- electrical and magnetic properties
- ease of manufacture.

In our design, the materials considered for the primary structure were aluminum, aluminum-lithium alloys, steel, titanium, intermetallic titanium alumides, magnesium, beryllium, and composites. The material that was finally chosen for the primary structure was aluminum due to its many advantages. Some of these qualities are: high stiffness to density ratio, high ductility, excellent workability, high corrosion, non-magnetism, moderate cost, and availability in numerous forms. The primary disadvantage of using aluminum is its low yield strength. Since aluminum lacks the strength to act as fittings between structural members, titanium was chosen for use in these areas.[3]

For the secondary structure, several materials were chosen according to their suitability for particular applications. The material selected for the inner wall of the propellant tanks was titanium, since the lunar lander uses cryogenic propellants and titanium exhibits good characteristics at low temperatures. For debris protection, we decided to use several layers of aluminized mylar insulation, which could also serve as thermal protection. Foam insulation and Schjeldahl coating, which has a low absorptivity to emittance ratio, also provide thermal protection.[3]

Aluminum was selected to provide radiation protection for the crew module. The 7.5 cm thick aluminum used in the primary structure provides the radiation protection so no extra material is required. The astronauts can be exposed to the amount of radiation this aluminum allows for up to six months with no ill effects.[4]

Finally, the materials chosen for reentry protection are AVCO-5026 ablator and HTP-6 tiles. The ablator material will be placed at the base of the crew module since the module re-enters the Earth atmosphere bottom first. The HTP-6 tiles will be placed in the other areas of the crew module, which do not experience the high temperatures of the base during reentry. These tiles are a new generation of Shuttle tiles and the ablator material is basically the same that was used in Apollo.[5]

9.5 Propellant Tanks

Figure 9.7 shows the sizes of the propellant tanks used for the ascent and the descent stages. As can be seen, the size of the tanks varies for the oxidizer and the fuel, and also for the ascent and the descent stages. The wall thickness of 8 cm for all the tanks is necessary to reduce boil-off.[6]

The spreadsheet used to size the tanks for this mission is found in Appendix D. The propellant tanks are sized according to the volume that is needed for each segment of the mission. A cylindrical design with hemispherical caps was chosen because of its advantages over a spherical design. The cylindrical design with hemispherical caps minimizes the transfer of energy to the propellants by reducing the overall area to volume ratio. Note that even though the area to volume ratio for a single cylindrical tank is higher than that of a spherical tank, the cylindrical design allows the number of tanks necessary to be minimized, which translates to a lower overall energy transfer.

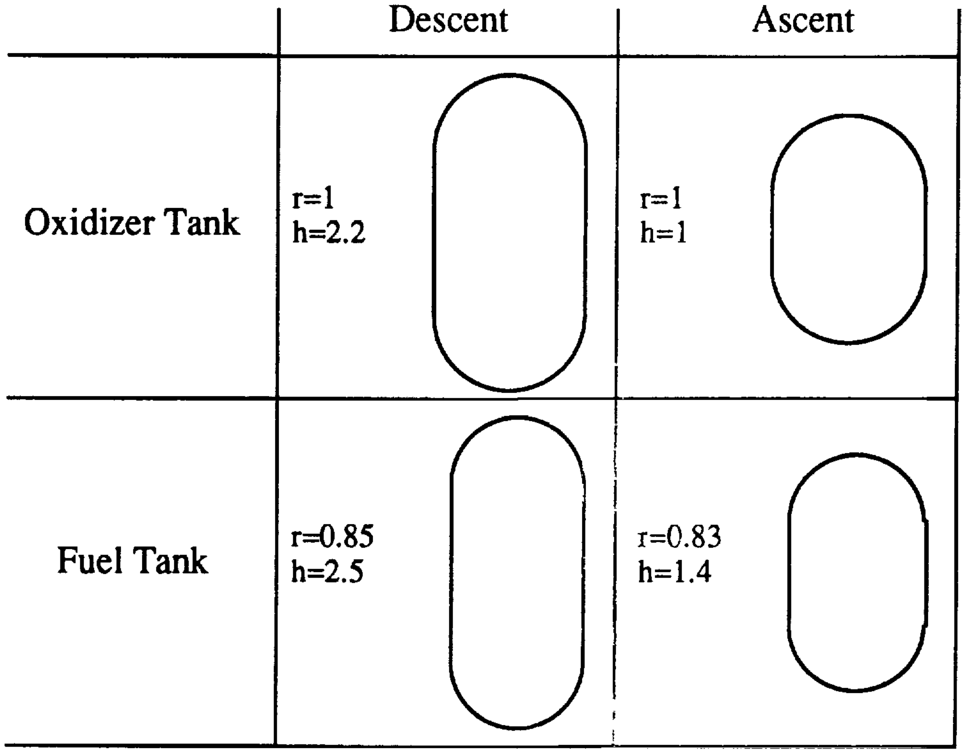


Figure 9.7 Propellant Tanks

9.6 Mass Estimate

The overall mass estimate for the mission is shown in Table 9.1. This overall mass estimate is based on the masses of the different subsystems, and also on the masses of Apollo and the FLO Mission done by NASA.[7]

Table 9.1 shows only the total masses of the mission. The detailed mass breakdown can be seen in Appendix D, which contains the spreadsheet used to calculate the mass of the lunar lander.

Table 9.1 Mass Estimate

	Crew Module	Ascent Stage	Descent Stage
Structure	3920	650	3500
Propulsion	252	1927	0
Power	841	120	2058
Other	4591	400	100
Dry Mass	9480	3097	5658
Non Cargo	609	300	0
Cargo	200	0	5000
Inert Mass	10289	3397	10658
Consumables	36	767	500
Propellant	210	17842	55000
Gross Mass	10535	21646	66158
TOTAL MASS	**(Post TLI)**	**98339 kg**	

9.7 References

1. Langan, Michael P., et al., Mission Analysis Section. First Lunar Outpost (FLO) Conceptual Flight Profile. Engineering Directorate, Systems Engineering Division; NASA JSC, June 1992. p. 1-4.

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5. Langan, et al., p. 7-2.

6. Langan, et al., Sec. 8.

7. Langan, et al. Sec. 3.

10.0 Project Management and Cost

10.1 Management

The management structure for the project is shown in Figure 10.1. The management team consists of a Project Manager (David Garza), a Chief Administrative Officer (Matt Carter), and a Chief Engineering Officer (Tony Ng). All design work is divided among four primary divisions: Orbital Mechanics/Guidance Navigation and Control, Structures, Propulsion / Power, and Life Support / Active Thermal Control/Communications.

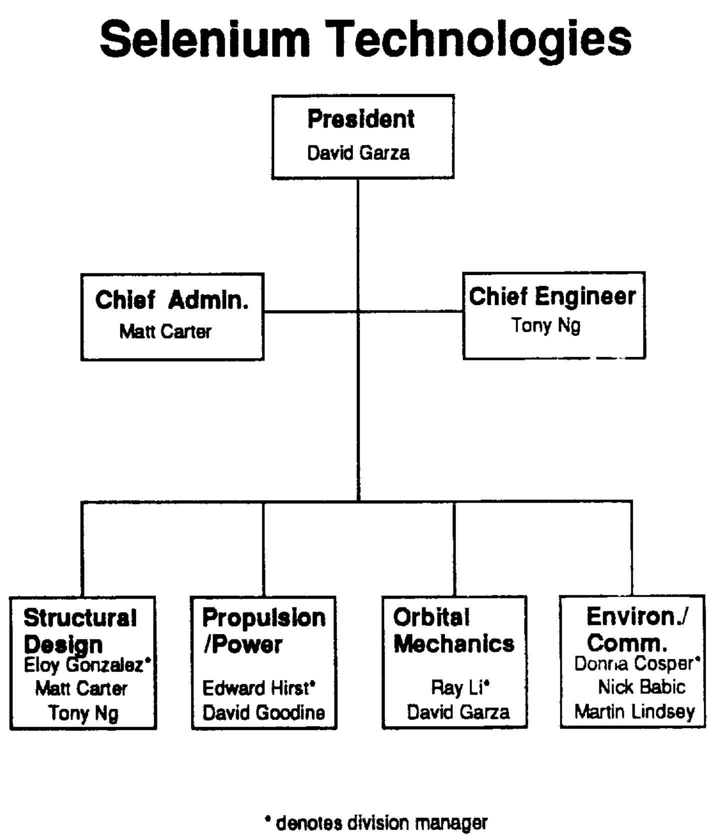


Figure 10.1 Selenium Technologies Company Structure

A Division Manager leads each Division, and reports to the Chief Administrative Officer (CAO) and the Chief Engineering Officer (CEO). The CAO and CEO in turn report to the Project Manager. A Division Manager may contact the Project Manager directly, but most work should filter through the CAO or CEO so that all management responsibilities are evenly dispersed. All design decisions must ultimately meet with the approval of the Project Manager. The responsibilities of the top management and the Division Managers, along with all project tracking information, are listed below.

10.1.1 Project Manager

The project manager oversees the entire project, and acts as the primary contact with the contracting organization. Overall program tracking and scheduling are handled by the Project Manager and administrative duties are handled jointly with the CAO. In the event of any major design obstacles, it is the duty of the Project Manager to make the necessary decisions needed to keep the project on track.

10.1.2 Chief Administrative Officer

The CAO handles the overall project management and shares all administrative duties with the Project Manager. Some of the management duties of the CAO include scheduling design meetings, maintaining a project notebook, tracking project costs, and acting as a link between the Division Managers and the Project Manager. During the absence of the Project Manager, the CAO acts as the presiding Manager.

10.1.3 Chief Engineering Officer

The CEO has the overall responsibility of resolving any technical dilemmas which may arise, including the integration of different subsystems, transfer of necessary data between divisions, and acquisition of technical data from outside sources. If the CEO cannot resolve an important issue, it is reported to the Project Manager and the issue is handled jointly. The CEO also supervises the technical progress of each division and acts as a technical consultant to each Division Manager.

10.1.4 Division Manager

The Division Managers have the responsibility of overseeing each division and insuring the completion of the tasks assigned to their divisions. Division Managers must also resolve any technical issues involving their divisions, as well as schedule division work assignments. If the issue remains unresolved, the CEO is contacted and the problem is analyzed jointly.

10.1.5 Project Tracking

A Gantt chart for the project is shown is Figure 10.2. This chart gives a sequential listing of the proposed project schedule.

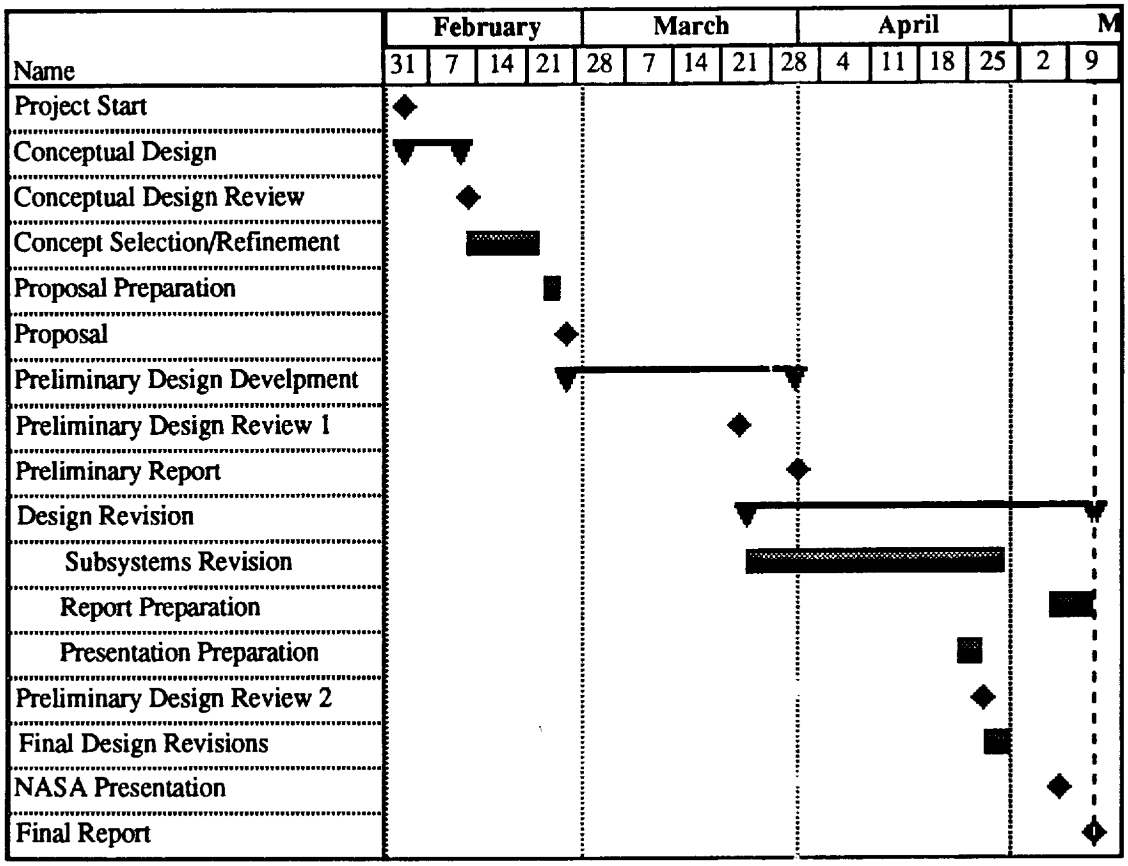


Figure 10.2 Project Gantt Chart

10.1.6 Changes After the Preliminary Report

The only major change in the team organization and scheduling after the Preliminary Report was the shifting of one member of the Life Support/Communications Group to the Structures Group.

10.2 Project Cost

The cost considerations for this project include personnel, computer, and supply costs. The cost analysis is drawn from twelve weeks of work. Figure 10.3 shows the current cost analysis for the personnel costs. These personnel costs are based upon the salaries provided by the Request for Proposal. The straight line in the graph depicts the estimated personnel cost which was initially laid out in the proposal. As can be seen, Selenium Technologies is well below this initial personnel cost estimate. This is primarily due to an over-estimation of the personnel costs.

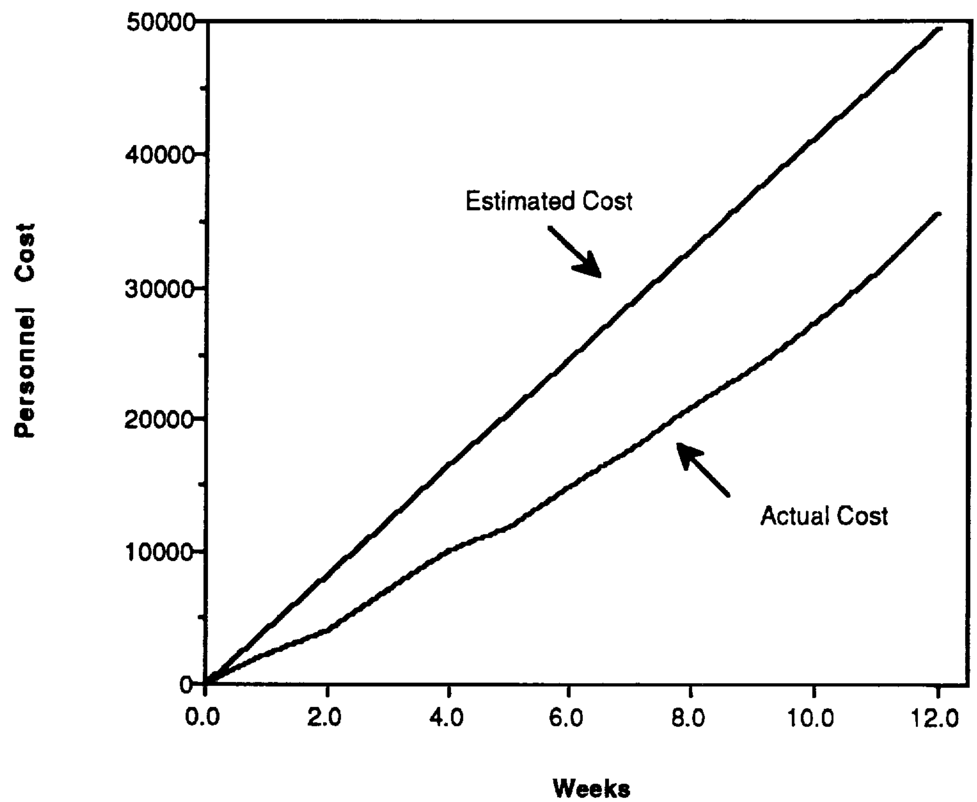


Figure 10.3 Personnel Costs

The project's computer costs were based upon the use of Macintosh hardware and software. The hardware costs are based on rental costs, while software costs were estimated as an initial lump sum. These software costs were "paid" within the first week and account for the large initial jump in cost. After the first week, the computer costs began to level out. Since all software costs were paid initially (which accounts for the over-budgeting), the computer costs were very near the proposed costs by the end of the project.

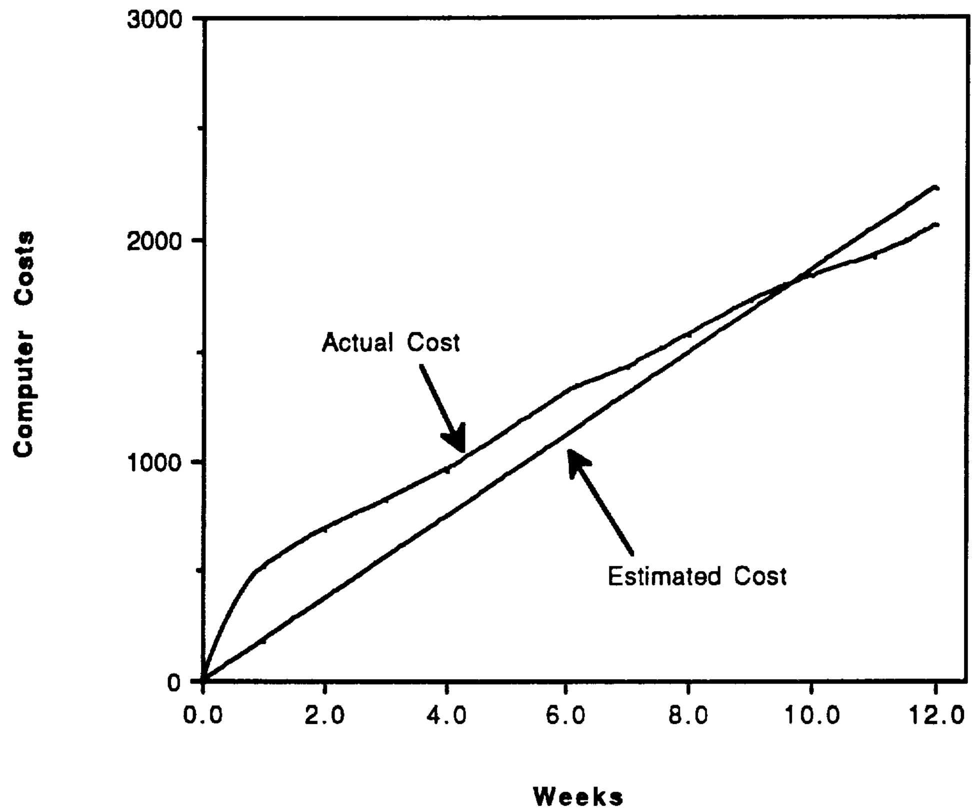


Figure 10.4 Computer Costs

The supply costs cover all the materials necessary for presentations and company communication. These materials include photocopies, transparencies, model, poster, and miscellaneous materials. Figure 10.5 shows the actual supply costs versus the estimated supply costs. Because of the added expense of the model and the poster, Selenium Technologies' supply costs are slightly over budget.

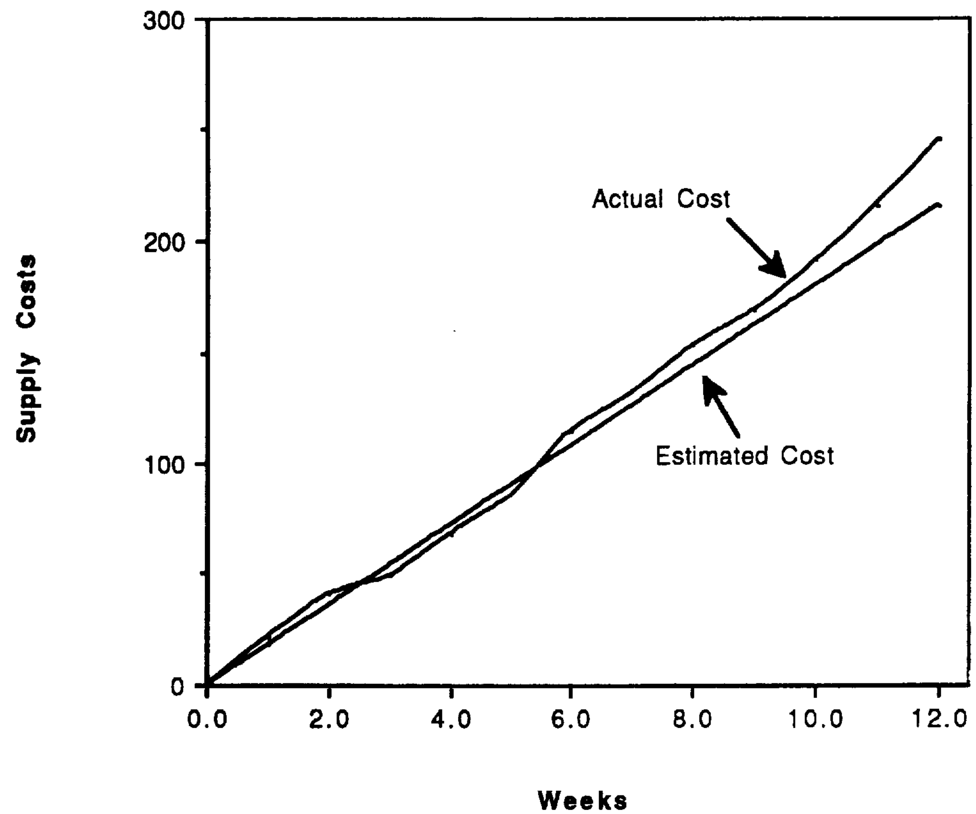


Figure 10.5 Supply Costs

The total project cost is shown below in Figure 10.6. Although supply costs have slightly exceeded Selenium Technologies' expectations, the low personnel costs have kept the total project costs under budget. Since the personnel costs are the largest project costs, they had the most effect on the overall project cost.

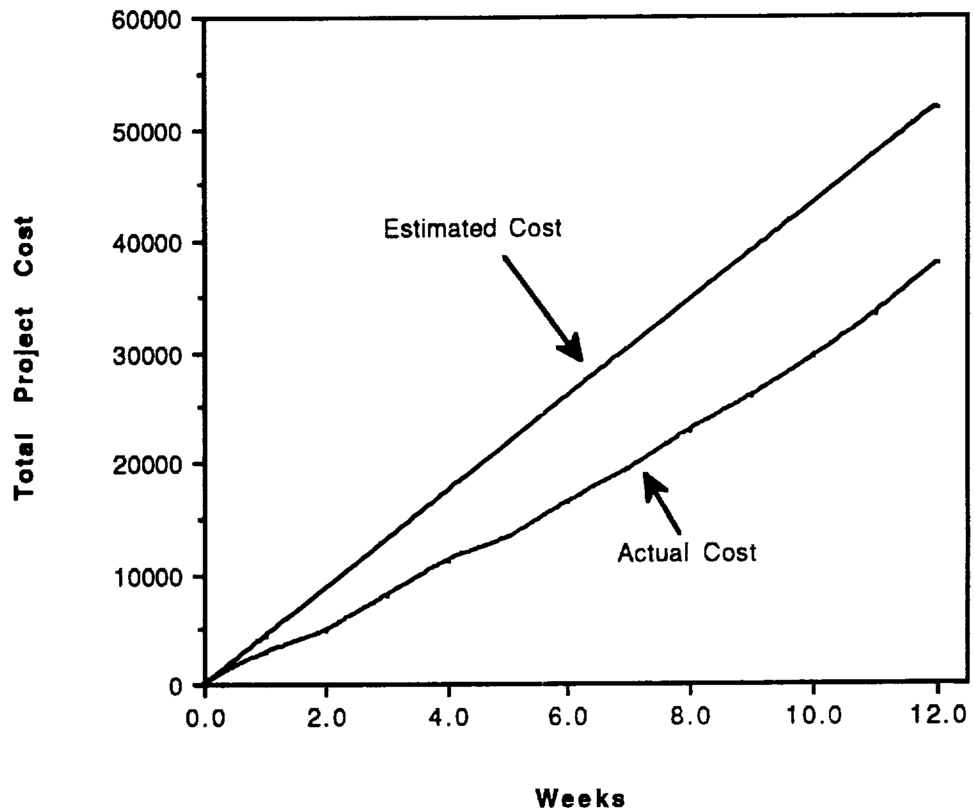


Figure 10.6 Total Project Cost

Appendix A. Power Subsystem Supplementary Information

Table A.1 Spacecraft Power and Energy Requirements

	Allotted Power (kW)	Attitude Control	Commu-nications	Compu-ters	GNC	RCS	Main Prop.	Lights & VDT's	ECLSS	Thermal Control	Power Consumption (kW)	Energy Consumption (kWh)
		0.200	0.145	1.000	0.100	0.336	0.616	0.800	1.800	0.150		
S/C Ops	time (hr)	Ratio of power used to power alloted										
Pad	0.167	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	4.015	0.669
Launch	0.167	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	4.015	0.669
E. orbit	4.500	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	4.015	18.069
Transit	96.000	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	4.019	385.801
LOI burn	0.083	1.000	0.312	1.000	1.000	0.000	1.000	0.900	1.000	1.000	4.631	0.386
M. orbit	6.000	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	4.015	24.092
M. land	1.000	1.000	0.312	1.000	1.000	0.000	1.000	1.000	1.000	1.000	4.711	4.711
Cap. day	120.000	0.000	0.312	1.000	0.000	0.000	0.000	0.900	1.000	1.000	3.715	445.848
FLO day	216.000	0.000	0.312	0.333	0.000	0.000	0.000	0.400	0.200	1.000	1.209	261.086
FLO nig.	336.000	0.000	0.312	0.333	0.000	0.000	0.000	0.500	0.200	1.000	1.289	433.014
FLO day	288.000	0.000	0.312	0.333	0.000	0.000	0.000	0.400	0.200	1.000	1.209	348.115
Cap. day	48.000	0.000	0.312	1.000	0.000	0.000	0.000	0.900	1.000	1.000	3.715	178.339
Cap. nig.	72.000	0.000	0.312	1.000	0.000	0.000	0.000	1.000	1.000	1.000	3.795	273.269
Launch	0.500	1.000	0.312	1.000	1.000	0.010	1.000	0.900	1.000	1.000	4.635	2.317
M. Orbit	6.000	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	4.015	24.092
TEI burn	0.083	1.000	0.312	1.000	1.000	0.000	1.000	0.900	1.000	1.000	4.631	0.386
Transit	96.000	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	4.019	385.801
Land	0.500	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	4.019	2.009

Note: GNC - Guidance, Navigation, and Control; RCS - Reaction Control System; Prop. - propulsion; VDT - video display terminal; ECLSS - Evironmental Control and Life Support Systems; S/C - Spacecraft; E. - Earth; LOI - Lunar Orbit Insertion; M. - Moon; Cap. - Capsule; FLO - First Lunar Outpost; nig. - night; TEI - Trans Earth Injection.

Table A.2 Spacecraft Power and Energy Consumed during Mission

	Allotted Power (kW)	Attitude Control	Commu nication	Compu-ters	GNC	RCS	Main Prop.	Lights &VDT	ECLSS	Thermal Control	Power	Electrol ysis	Power Consum tion	Energy Consum tion
		0.200	0.145	1.000	0.100	0.336	0.616	0.800	1.800	0.150	0.866	4.412		
S/C Ops	time (hr)	Ratio of power used to power alloted											(kW)	(kWh)
Pad	0.167	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	1.000	0.000	4.875	0.813
Launch	0.167	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	1.000	0.000	4.875	0.813
E. orbit	4.500	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	1.000	0.000	4.875	21.939
Transit	96.000	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	1.000	0.000	4.879	468.361
LOI brn	0.083	1.000	0.312	1.000	1.000	0.000	1.000	0.900	1.000	1.000	1.000	0.000	5.491	0.458
M. orbit	6.000	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	1.000	0.000	4.875	29.252
M. land	1.000	1.000	0.312	1.000	1.000	0.000	1.000	1.000	1.000	1.000	1.000	0.000	5.571	5.571
Cap. dy	120.000	0.000	0.312	1.000	0.000	0.000	0.000	0.900	1.000	1.000	1.000	0.647	7.428	891.390
FLO dy	216.000	0.000	0.312	0.333	0.000	0.000	0.000	0.400	0.200	1.000	1.000	0.647	4.922	1063.06
FLO ng.	336.000	0.000	0.312	0.333	0.000	0.000	0.000	0.500	0.200	1.000	1.000	0.000	2.149	721.974
FLO dy	288.000	0.000	0.312	0.333	0.000	0.000	0.000	0.400	0.200	1.000	1.000	0.000	6.481	1866.57
Cap. dy	48.000	0.000	0.312	1.000	0.000	0.000	0.000	0.900	1.000	1.000	1.000	1.000	8.988	431.415
Cap. ng.	72.000	0.000	0.312	1.000	0.000	0.000	0.000	1.000	1.000	1.000	1.000	1.000	4.655	335.189
Launch	0.500	1.000	0.312	1.000	1.000	0.010	1.000	0.900	1.000	1.000	1.000	0.000	5.495	2.747
M. Orbit	6.000	1.000	0.312	1.000	1.000	0.000	0.000	0.900	1.000	1.000	1.000	0.000	4.875	29.252
TEI brn	0.083	1.000	0.312	1.000	1.000	0.000	1.000	0.900	1.000	1.000	1.000	0.000	5.491	0.458
Transit	96.000	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	1.000	0.000	4.879	468.361
Land	0.500	1.000	0.312	1.000	1.000	0.010	0.000	0.900	1.000	1.000	1.000	0.000	4.879	2.439

Note: GNC - Guidance, Navigation, and Control; RCS - Reaction Control System; Prop. - propulsion; VDT - video display terminal; ECLSS - Evironmental Control and Life Support Systems; S/C - Spacecraft; E. - Earth; LOI - Lunar Orbit Insertion; brn - burn; dy - day; M. - Moon; Cap. - Capsule; FLO - First Lunar Outpost; ng. - night; TEI - Trans Earth Injection.

Preliminary Mission Timeline and Power Analysis

Mission timeline, power and energy requirements

S/C operation	Time (hr)	Power (kW)	Energy (kWh)
pad	0.167	5.165	0.861
launch	0.167	5.165	0.861
orbit	4.500	5.169	23.259
transit	96.000	5.165	495.863
LOI burn	0.083	5.781	0.482
orbit	6.000	5.165	30.991
land	1.000	5.861	5.861
in cap day	120.000	4.865	583.829
in FLO day	216.000	2.199	474.892
in FLO night	336.000	2.279	765.601
in FLO day	288.000	2.199	633.189
in cap day	48.000	4.865	233.532
in cap night	72.000	4.945	356.057
launch	0.500	6.117	3.059
orbit	6.000	5.165	30.991
TEI burn	0.083	5.681	0.473
transit	96.000	5.169	496.186
land	0.500	5.169	2.584
	1291.000		

Options Analysis

Fuel Cells

Fuel Cell Analysis	
18.85 kg/kWh	2.24 kg/kWh
H_2 (kg)	O_2 (kg)
0.046	0.384
0.046	0.384
1.234	10.383
26.306	221.367
0.026	0.215
1.644	13.835
0.311	2.617
30.972	260.638
25.193	212.005
40.615	341.786
33.591	282.674
12.389	104.255
18.889	158.954
0.162	1.365
1.644	13.835
0.025	0.211
26.323	221.511
0.137	1.154
219.553	1847.576

H_2 density	O_2 density
70.8	1141
Volumes	
3.10102844	1.61926041
Mass	
219.553	1847.576
Total	2067.129
Supporting mass	
percent of system	0.5
Total mass	4134.25789

Mass of Array	0

Total mass	4134.25789

Regenerative Fuel Cells

Fuel Cell Analysis (regenerative system)

18.85 kg/kWh	2.24 kg/kWh	Intermediate masses	
H2 (kg)	O2 (kg)	H_2	O_2
0.046	0.384		
0.046	0.384		
1.234	10.383		
26.306	221.367		
0.026	0.215		
1.644	13.835		
0.311	2.617	29.612	248.972
30.972	260.638		
25.193	212.005		
40.615	341.786		
33.591	282.674		
12.389	104.255		
18.889	158.954	59.504	500.740
0.162	1.365		
1.644	13.835		
0.025	0.211		
26.323	221.511		
0.137	1.154	28.291	238.077
	Total	87.796	738.818

Parameters for Electrolysis	
Fuel cell n	0.550
Electrolyzer	0.550
Electrolysis Power	
1st regen	
FC time (hr)	107.917
Elec time (hr)	336.000
FC Power	5.172
Electrolyzes	5.492
2nd regen	
FC time	336.000
Elec time	336.000
Fuel Cell power	2.580
Electrolyzes	8.527
Total fuel mass	826.613

H2 density	O2 density
70.8	1141
Volumes	
1.24005	0.6475
Mass	
87.796	738.818
Total	826.613
Supporting mass percent of system	0.5
mass	1653.2
Mass of Array	174.068
Total mass	1827.3

Rechargeable Batteries

Rechargeable Battery Analysis

Primary System

S/C Ops	Time (hr)	P (kW)	E (kWh)	Intermed. require. kWh	Require. 28V bus Ah	Time for (dis)charg hr
pad	0.167	5.165	0.861			
launch	0.167	5.165	0.861			
orbit	4.500	5.169	23.259			
transit	96.000	5.165	495.863			
LOI burn	0.083	5.781	0.482			
orbit	6.000	5.165	30.991			
land	1.000	5.861	5.861	558.178	19.935	107.917
in cap day	120.000	4.865	583.829			
in FLO day	216.000	2.199	474.892	1058.721	37.811	336.000
in FLO nig	336.000	2.279	765.601	765.601	27.343	336.000
in FLO day	288.000	2.199	633.189			
in cap day	48.000	4.865	233.532	866.721	30.954	336.000
in cap nig	72.000	4.945	356.057			
launch	0.500	6.117	3.059			
orbit	6.000	5.165	30.991			
TEI burn	0.083	5.681	0.473			
transit	96.000	5.169	496.186			
land	0.500	5.169	2.584	889.351	31.763	175.083

Options - Secondary Batteries				
NiCad	NiH2	NiZn	AgZn	AgCad
V/cell				
1.200	1.400	1.600	1.500	1.200
# of cells				
23.333	20.000	17.500	18.667	23.333
24.000	21.000	18.000	19.000	24.000
ED (Wh/kg)				
0.020	0.055	0.060	0.110	0.055
Battery mass				
52936.032	19249.466	17645.344	9624.740	19249.466
Array mass	77.184			
Total mass				
53013.216	19326.650	17722.528	9701.920	19326.650
Recharge Power kW				
3.781	3.781	3.781	3.781	3.781

Reentry Battery Analysis

Reentry Battery Options			Ag-Zn	NiCad	NiZn	AgCad
Energy (kWh)	2.440	kg/kWh	0.110	0.020	0.060	0.055
		kg	22.182	122.000	40.667	44.364

Final System Sizing

Fuel Cell Analysis (regenerative system)					
18.850	2.240	Intermediate masses		Parameters for Electrolysis	
H_2 (kg)	O_2 (kg)	H_2	O_2	Fuel cell n	0.550
0.043	0.363			Electrolyzes n	0.550
0.043	0.363			Electrolysis Power	
1.164	9.795			1st regen - post land	
24.846	209.103			FC time (hr)	107.833
0.024	0.204				
1.552	13.060			Elec time (hr)	336.000
0.296	2.487	27.968	235.375	FC Power	4.889
		electrolysis period		Electrolyzes	5.187
0.000	0.000	0.000	0.000	2nd regen - post FLO night	
				FC time (hr)	336.000
38.300	322.330	38.300	322.330	Elec time (hr)	336.000
		electrolysis period		FC Power	2.427
0.000	0.000	0.000	0.000	Electrolyzes	8.023
17.782	149.647	17.782	149.647		
0.146	1.227			Tot. Fuel mass	Water consumption
1.552	13.060			417.591	142.317
0.024	0.204				H_2
24.846	209.103			contingency factor	15.115
0.000	0.000			1.000	O2
0.000	0.000	26.568	223.594	fuel+water	127.202
	Water	15.115	127.202	673.908	
	Fuel	44.350	373.241	673.908	
Ratio=	Total	59.464	500.443	Water factor	1.000
8.416		59.464	500.443		
	1.000	59.464	500.443		
Add for return		78.794	663.114	741.908	
at all times		12.107	101.893	114.000	
Add for contingency		7.222	60.778	68.000	

Electrolyzes			
Assumed % of FC mass		1.000	unknown
	kg/cell	92.000	size
	cells	3.000	
	total	276.000	
Fuel cells			0.469
	kg/cell	92.000	0.356
Dimensions	cells	3.000	0.432
(last column)	total	276.000	1.016
H_2O	density	1000.000	
mass		360.631	
volume		0.361	
Additional Hardware		Component	552.000
Cabling and switches			
% of total	0.500		552.000
NEED TO ADD SOLAR ARRAY MASS TO THIS NUMBER =>		Preliminary Total mass	1104.000
	H_2	O_2	
cry density	70.800	1141.000	
Volumes	1.113	0.581	

Solar cell array analysis (based on GaAs solar cells)

Space solar intensity (W/m^2)	1358.000	Bus voltage	28.000	
Angle between sun and cell normal (rad)	0.000	Array voltage	33.600	
Solar cell efficiency	0.180			
Degradation factors	1.000	Parameters	fuel cell	
total degredation	0.000	Array	Specific W/kg	25.000
degredation coeff. (%/yr)	0.003	Weight (kg)	503.919	
time of exposure to rad. (yr)	0.085	Area based kg/m^2	3.300	264.000
time of exposure to rad (hr)	744.000	Area	80.000	
Thermal factors	0.745	Geometry		
thermal coefficient (%/degC)	-0.003	Assume	2.000	square panels
Maximum op. temp. (deg C)	130.000	Area/panel	fuel cell	
Reference temp. (deg C)	28.000		40.000	
Packing Factor	0.900	Side length	6.325	
		COM position	4.162	
Maximum Power Required	4575.400	Mass Moments		
Regeneration of Eclipse Power source		Perpen to face	11958.744	
		Perpen to axis	10344.458	
Electrolysers	8022.574	About axis	1614.286	
		fuel cell	Mass of support and other accessories assumed of array mass	0.500
Power required		12597.974		
Solar Array Area (m^2)		76.883		
				mass 132.000

TOTAL SYSTEM MASS	
Fuel Cell System	1777.90778
Solar Array System	396
Total	2173.90778

Descent Mass	948
Ascent Mass	1225.90778
	2173.90778
Masses	
Fuel	673.907782
Hardware	552
Support stuff	552

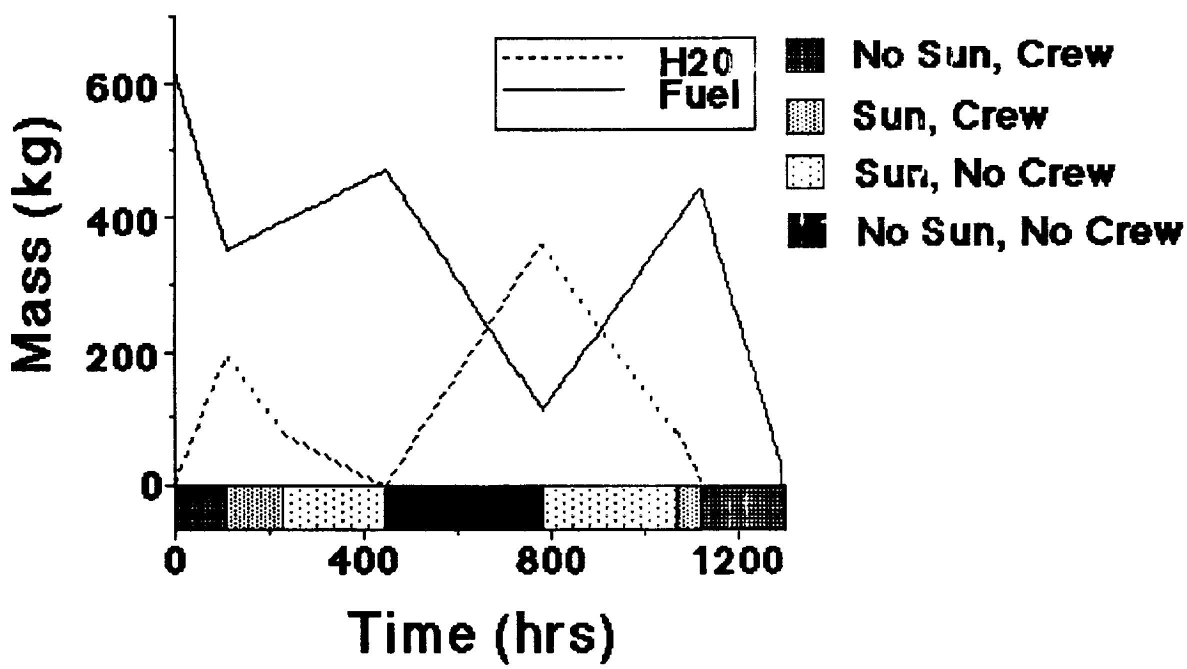
Total	1777.90778
Mass	
Accessories	132
Array	264
Total	396

Mass Timeline of Fuels and Water

Mass timeline						
Cum time	power	S/C	Energy	P. source	H_2	O_2
0.000	0.000	beginning	0.000	off	78.794	663.114
0.167	4.875	pad	0.813	fuel cells	78.750	662.751
0.333	4.875	launch	0.813	fuel cells	78.707	662.389
4.833	4.875	orbit	21.939	fuel cells	77.543	652.594
100.833	4.879	transit	468.361	fuel cells	52.697	443.491
100.917	5.491	LOI burn	0.458	fuel cells	52.673	443.286
106.917	4.875	orbit	29.252	fuel cells	51.121	430.226
107.917	5.571	land	5.571	fuel cells	50.825	427.739
227.917	7.428	in cap day	891.390	panels+rege	55.382	466.088
443.917	4.922	in FLO day	1063.062	panels+rege	63.584	535.116
779.917	2.149	in FLO night	721.974	fuel cells	25.284	212.785
1067.917	6.481	in FLO day	1866.571	panels+rege	55.367	465.963
1115.917	8.988	in cap day	431.415	panels+rege	60.381	508.159
1187.917	4.655	in cap night	335.189	fuel cells	42.600	358.512
1188.417	5.495	launch	2.747	fuel cells	42.454	357.285
1194.417	4.875	orbit	29.252	fuel cells	40.902	344.225
1194.500	5.491	TEI burn	0.458	fuel cells	40.878	344.021
1290.500	4.879	transit	468.361	fuel cells	16.031	134.918
1291.000	4.879	land	2.439	battery		
		tot energy	6340.066			
		av. power	4.911			
		peak power	8.988			
		min power	2.149			

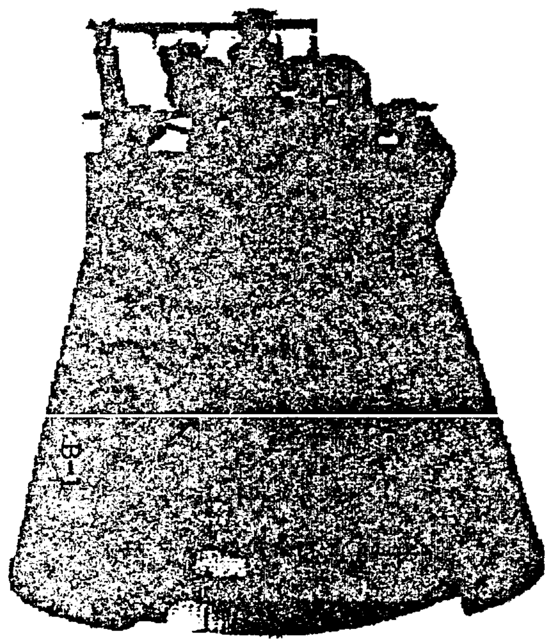
H_20 prod.	sum	H_20 to ECLSS	sum	H_20 elec.	H_20 stor.
0.000	0.000	0.000	0.000	0.000	0.000
0.406	0.406	0.105	0.105	0.000	0.301
0.406	0.812	0.105	0.209	0.000	0.301
10.959	11.771	2.828	3.037	0.000	8.131
233.949	245.720	60.320	63.357	0.000	173.629
0.229	245.949	0.052	63.409	0.000	0.176
14.612	260.560	3.770	67.179	0.000	10.842
2.783	263.343	0.628	67.808	0.000	2.155
0.000	263.343	75.400	143.208	42.906	-75.400
0.000	263.343	0.000	143.208	77.230	0.000
360.631	623.974	0.000	143.208	0.000	360.631
0.000	623.974	0.000	143.208	283.261	0.000
0.000	623.974	30.160	173.368	47.210	-30.160
167.429	791.403	45.240	218.608	0.000	0.000
1.372	792.776	0.314	218.922	0.000	0.000
14.612	807.387	3.770	222.692	0.000	0.000
0.229	807.616	0.052	222.744	0.000	0.000
233.949	1041.565	60.320	283.064	0.000	0.000

Water-Fuel Interaction



Appendix B: Propulsion Supplementary Information

...TIVE ENGINE OPTIONS



...10A-4	RL10B-X	RL10C-X	RL10M-X
...,600	22,000	25,000-35,000	15,000-31,000
...9	460 to 470	452 to 460	358 to 380
...:1	5.5:1 to 6.5:1	5.5:1 to 6.5:1	3.5
...0	450 to 700	450 to 700	320-700
...5	600	600	500
...:1	200:1 to 300:1	150:1 to 225:1	60:1 to 300:1
-	30 to 42	42 - 54	48-60

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COMPETITION SENSITIVE

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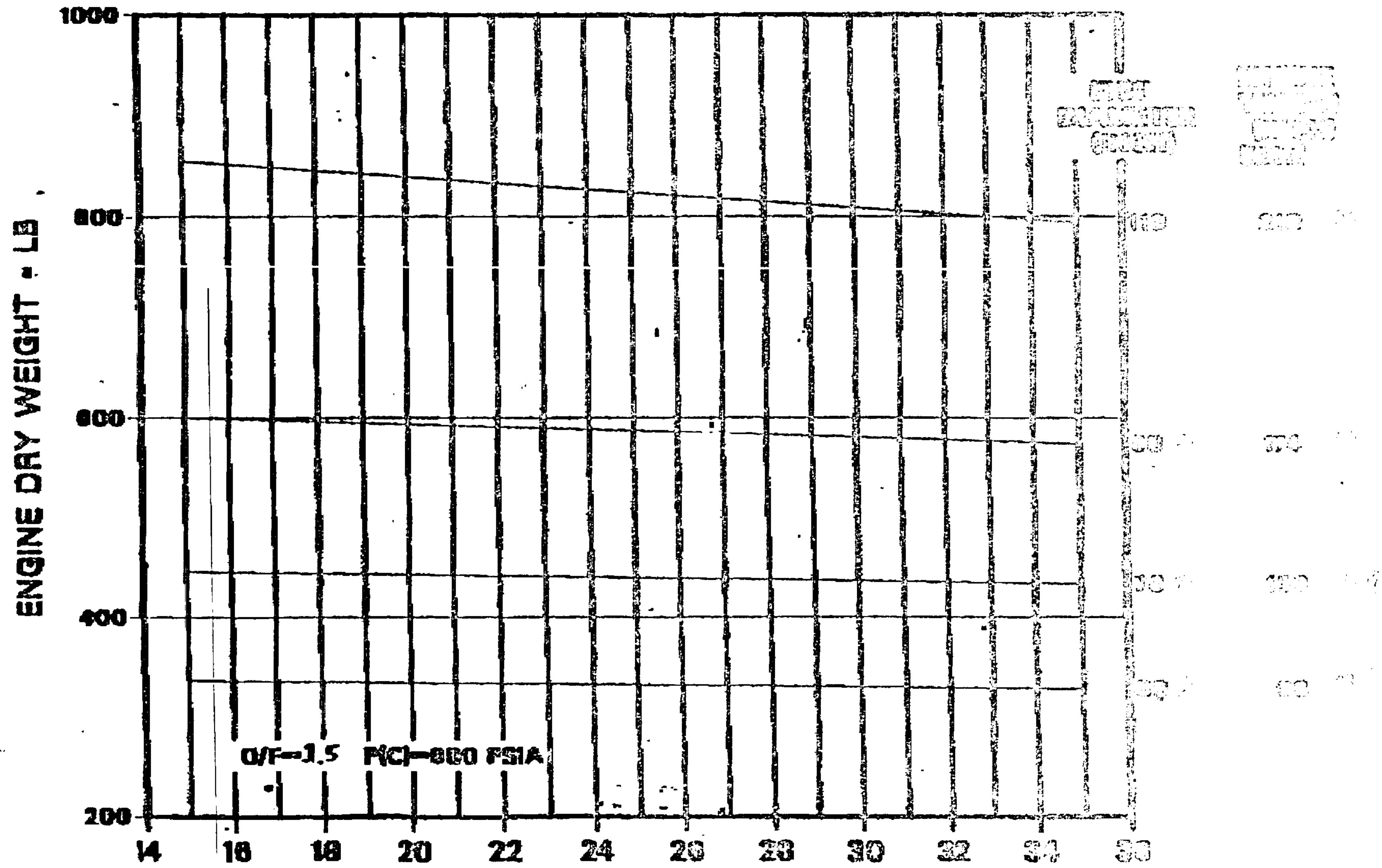
METHANE FUEL RL10

EFFECT OF THRUST LEVEL

MAX PAYLOAD NOZZLE



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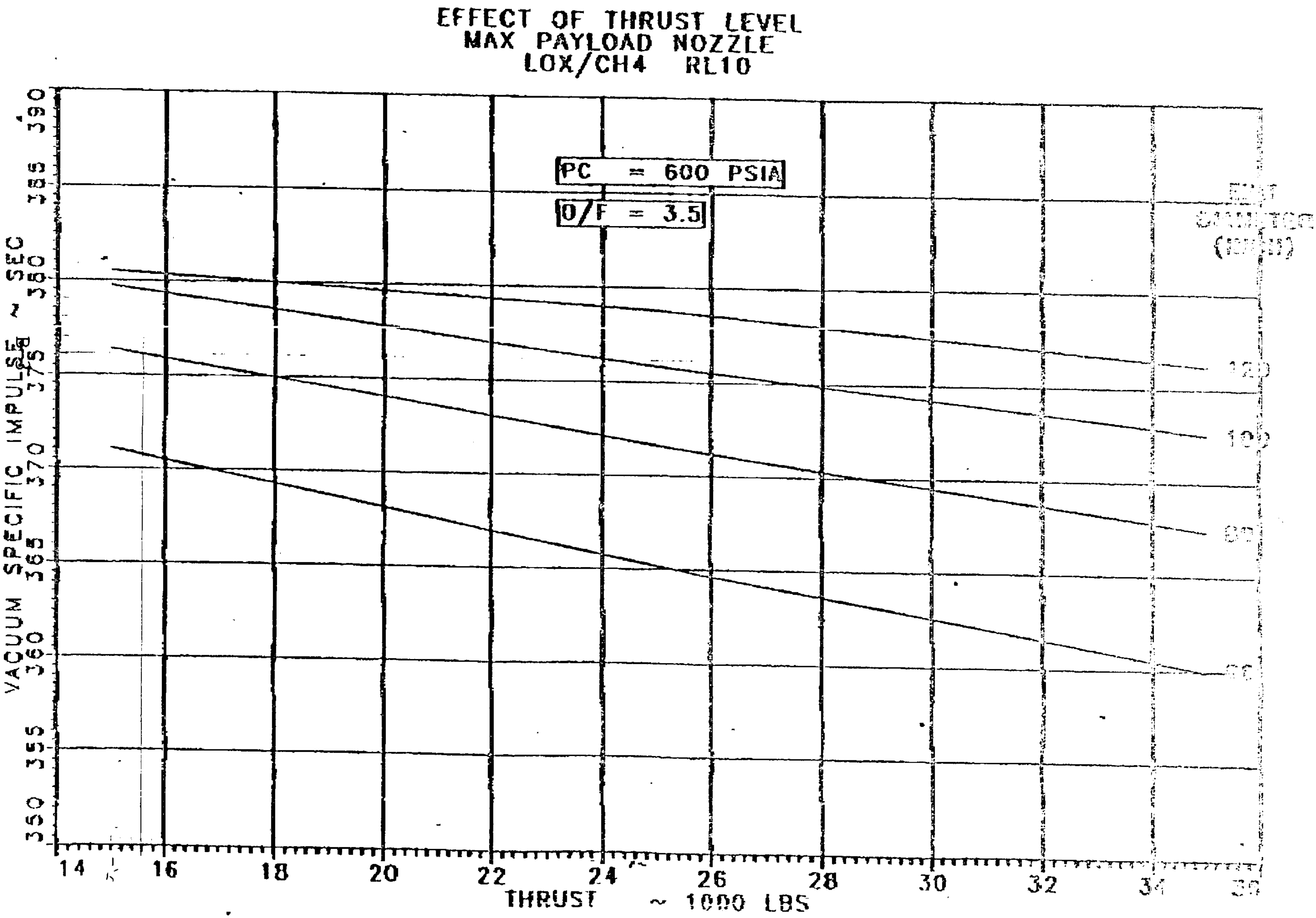
METHANE FUEL RL10



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tk! Solver Model for Propellant Mass Calculations

tk! Solver Variables Sheet

St	Input	Name	Output	Unit	Comment
	200	m0		mt	mass after launch into LEO
		mf1	104.78364	mt	mass after TLI burn
	95	m02		mt	mass after TLI stage separation
L		mf2	44.649348	mt	mass after LOI and Lunar descent
		m03	26.201848	mt	mass after sep. from Hab and LDS
L		mf3	12.499688	mt	mass after Lunar ascent and TEI burn
		mreentr		mt	mass at reentry
	3230	tlidelv		m/s	Δ V for TLI
	912	loidelv		m/s	Δ V for LOI
	1873	descdel		m/s	Δ V for descent
	1852	ascdelv		m/s	Δ V for ascent
	878	teidelv		m/s	Δ V for TEI
L		Isp1	509.35304	s	TLI Isp
L	376	Isp2		s	LDS Isp
L	376	Isp3		s	LAS Isp
	9.81	g		m/s^2	Acceleration due to grav. at Earth
		mtlidry	9521.6364	kg	dry mass of TLI stage
	15500	mdescdr		kg	dry mass of LDS
	0	mhab		kg	dry mass of habitat
		mascdry		kg	dry mass of LAS
	1.623	gMoon		m/s^2	grav. accel. at Moon surface
L		Tdescmi	27069.245	lbf	minimum thrust of descent stage
		Tascmin	9560.1384	lbf	minimum thrust of ascent stage
	1	T_to_Wt			Thrust to Weight Rat. at Lunar surf.
		eng#	.77344525		
	155680	maxthru		N	
		mp1	95216.364	kg	Propellant mass for TLI
L		mp2	50350.652	kg	Prop. mass for LOI and DESC
L		mp3	13702.159	kg	Prop. mass for ASC and TEI
L	6	MR1			mixture ratio for first stage eng.
L	46	rhoful		kg/m^3	density of fuel in stage 1
L	1140	rhoox1		kg/m^3	density of ox in stage 1
		mox1	81.614026	mt	mass of ox in stage1
		mful	13.602338	mt	mass of fuel in stage1
L		vox1	71.591251	m^3	volume of oxidizer in stage 1
L		vful	295.70299	m^3	volume of fuel in stage 1
L	3.6	MR2			mixture ratio for second stage eng.
L	445	rhofu2		kg/m^3	density of fuel in stage 2
L	1140	rhoox2		kg/m^3	density of ox in stage 2
		mox2	39609.902	kg	mass of ox in stage2
		mfu2	11002.751	kg	mass of fuel in stage2
L		vox2	34.745528	m^3	volume of oxidizer in stage 2
L		vfu2	24.725282	m^3	volume of fuel in stage 2
L	3.6	MR3			mixture ratio for third stage eng.
L	445	rhofu3		kg/m^3	density of fuel in stage 3
L	1140	rhoox3		kg/m^3	density of ox in stage 3
		mox3	13030.168	kg	mass of ox in stage3
		mfu3	3619.4912	kg	mass of fuel in stage3
L		vox3	11.429972	m^3	volume of oxidizer in stage 3
L		vfu3	8.133688	m^3	volume of fuel in stage 3
	262	mboilof		kg	boiloff from descent stage
	2947.5	mboilof		kg	boiloff from ascent stage
		m02_pos	74.189745	mt	

tk! Solver Rules Sheet

```
S Rule
  "first leg:  translunar injection
* call rocket(m0,Isp1,tlidelv;mf1)                "final mass after TLI burn
* m02=mf1-mtlidry-mboiloff1                       "mass after TLI separation
* mp1=m0-mf1                                      "prop. reqd. for Leg1

  "second leg:  lunar orbit insertion and descent
* call rocket(m02,Isp2,loidelv+descdelv;mf2)      "mass after LOI and descent
* m03=mf2-mdescdry-mhab-mboiloff2                 "mass after sep. of desc. stage
* mp2=m02-mf2                                     "prop. reqd. for Leg2

  "third leg:  ascent to orbit from lunar surface and TEI
* call rocket(m03,Isp3,ascdelv+teidelv;mf3)       "mass after ascent and TEI
* mreentry=mf3-mascdry                            "mass after ascent stage sep.
* mp3=m03-mf3                                     "prop. reqd. for Leg3

  "calculate minimum propellant volumes: stages 1, 2, 3.
* call vol(MR1,rhofu1,rhoox1,mp1;mox1,mfu1,vox1,vfu1)
* call vol(MR2,rhofu2,rhoox2,mp2+mboiloff1;mox2,mfu2,vox2,vfu2)
* call vol(MR3,rhofu3,rhoox3,mp3+mboiloff2;mox3,mfu3,vox3,vfu3)


  "estimation of dry masses for stages
* mtlidry=0.10*(m0-mf1)                     "estimation of dry mass of TLI
C mdescdry=0.1*(m02-mf2)                    "estimation of dry mass of descent
C mascdry=0.1*(m03-mf3)                     "estimation of dry mass of ascent s

  "estimation of minimum thrust for descent stage and ascent stage
* call rocket(m02,Isp2,loidelv;m02_postLOI)
* Tdescmin=m02_postLOI*T_to_Wt*gMoon
* Tascmin=m03*T_to_Wt*gMoon

  "compute the # of engines
* eng#=Tdescmin/maxthrust
```

tk! Solver VOL Function

```
Comment:              volume calculation
Parameter Variables:
Argument Variables:   MR,rhofu,rhoox,mp
Result Variables:     mox,mfu,vox,vfu
S Rule
  mfu*(MR+1)=mp
  MR=mox/mfu
  vfu=1/rhofu*mfu
  vox=1/rhoox*mox
```

tk! Solver ROCKET Function

```
Comment:              rocket equation
Parameter Variables: g
Argument Variables:   m0,isp,deltav
Result Variables:     mf
S Rule
  mf=m0*1/exp(deltav/(g*isp))
```

tk! Solver UNITS Sheet

From	To	Multiply By	Add Offset	Comment
m/s	km/s	.001		
kg	lbm	2.205		
ft/s	m/s	.3048		
mt	kg	1000		
m/s^2	ft/s^2	3.28083989501		
N	lbf	.224809024733		
m^3	ft^3	35.31		

Appendix C. ECLSS tk! Solver Model

RULE SHEET

```
S Rule
  "Values for open system (LIOH, Stored H2O, Stored O2)
* PRO=INT(N)*.195
* WHO=INT(N)*.210
* MSCO=((INT(N)*3131.0775+83.610)/90)*t
* VSCO=((INT(N)*3.2410)/90)*t
* MHWO=INT(N)*76.1-83.61
* LMO=MSCO+MHWO
* VHWO=INT(N)*7.4165
* LVO=VSCO+VHWO
* PIO=PRO*359.0
* HIO=WHO*109.0
  "Values for minimally closed system (Electrochemical Depolarized Cell,
  "Stored H2O, and Stored O2).
* PRM=INT(N)*.2475
* WHM=INT(N)*.3225
* MSCM=((INT(N)*617.6345+83.61)/90)*t
* VSCM=((INT(N)*.85850)/90)*t
* VHWM=INT(N)*1.60950
* MHWM=INT(N)*87.018-83.61
* LMM=MSCM+MHWM
* LVM=VSCM+VHWM
* PIM=PRM*359.0
* HIM=WHM*109.0
  "Values for partially closed system (Electrochemical Depolarized Cell,
  "Sabatier Reactor, Static Feed Electrolysis, Vapor Compression Distillation,
  "and Multifiltration).
* PRP=INT(N)*.4475
* WHP=INT(N)*.390
* MSCP=((INT(N)*152.4826+83.61)/90)*t
* VSCP=((INT(N)*.42202)/90)*t
* VHWP=INT(N)*.73353
* MHWP=INT(N)*108.3140-83.61
* LMP=MSCP+MHWP
* LVP=VSCP+VHWP
* PIP=PRP*359.0
* WIP=WHP*109.0
```

VARIABLE SHEET

St	Input	Name	Output	Unit	Comment
	4	N			Number of crewmembers
	38	t		day	Duration of mission (days)
					Results for Open System (LiOH, Stored H2O, Stored O2)
		PRO	.78	kw	Power Required
		WHO	.84	kw	Waste Heat Generated
		MSCO	5323.344	kg	Mass of Spares and Consumables
		MHWO	220.79	kg	Mass of System Hardware
		LMO	5544.134	kg	Total System Mass
		VSCO	5.4736889	m^3	Volume of Spares and Consumables
		VHWO	29.666	m^3	Volume of System Hardware
		LVO	35.139689	m^3	Total System Volume
		PIO	280.02	kg	Power Impact Penalty
		HIO	91.56	kg	Waste Heat Impact Penalty
					Results for Partially Closed System (Electrochemical Depolarized Cell (EDC Stored H2O, Stored O2, and Multifiltration (MF))
		PRM	.99	kw	Power Required
		WHM	1.29	kw	Waste Heat Generated
		MSCM	1078.418	kg	Mass of Spares and Consumables
		MHWM	264.462	kg	Mass of System Hardware
		LMM	1342.88	kg	Total System Mass
		VSCM	1.4499111	m^3	Volume of Spares and Consumables
		VHWM	6.438	m^3	Volume of System Hardware
		LVM	7.8879111	m^3	Total System Volume
		PIM	355.41	kg	Power Impact Penalty
		HIM	140.61	kg	Waste Heat Impact Penalty
					Results for Closed System (EDC, Sabatier Reactor, Static Feed Electrolysis, Vapor Compression Distillation, MF)
		PRP	1.79	kw	Power Required
		WHP	1.56	kw	Waste Heat Generated
		MSCP	292.82817	kg	Mass of Spares and Consumables
		MHWP	349.646	kg	Mass of System Hardware
		LMP	642.47417	kg	Total System Mass
		VSCP	.71274489	m^3	Volume of Spares and Consumables
		VHWP	2.93412	m^3	Volume of System Hardware
		LVP	3.6468649	m^3	Total System Volume
		PIP	642.61	kg	Power Impact Penalty
		WIP	170.04	kg	Waste Heat Impact Penalty

Appendix D. Structures Supplementary Information

Tank Sizing Spreadsheet

Units
Volume-m^3
Area-m^2
Length-m
Weight-kg

Propellant Tanks Sizing

	Ascent Stage		Descent Stage	
	Total Volume		**Total Volume**	
Oxidizer	11.43		34.746	
Fuel	8.134		24.726	
	Volume(Safety Factor 1.05)		**Volume(Safety Factor 1.05)**	
Oxidizer	12.0015		36.4833	
Fuel	8.5407		25.9623	
	Volume 1 Tank (2 Tanks Total)		**Volume 1 Tank (4 Tanks Total)**	
Oxidizer Tank	6.00075		9.120825	
Fuel Tank	4.27035		6.490575	

Dimensions

	Oxidizer Tank	Fuel Tank	Oxidizer Tank	Fuel Tank
Inner Radius	0.92	0.75	0.92	0.77
Outer Radius	1	0.83	1	0.85
Cylin. Height	1.03006518	1.41652351	2.20344728	2.45792426
Total Height	3.03006518	3.07652351	4.20344728	4.15792426

Tank Weights

	Oxidizer Tank	Fuel Tank	Oxidizer Tank	Fuel Tank
FLO				
Area	40	43.5	40	43.5
Weight	389	472.5	389	472.5
Selenium				
Area	19.0384631	16.0442066	26.4110411	22.2062598
Weight	185.149054	174.273279	256.847375	241.205925

Total Weights

		Ascent	Descent
Propellants	Oxidizer	13681.71	41590.962
	Fuel	3800.6115	11553.2235
Tanks	Oxidizer	370.298108	1027.3895
	Fuel	348.546558	964.823701

Fuel Cell Tanks Sizing

	Total Weight	Total Volume	Volume 1 Tank
Hydrogen	67.1039	0.9477952	0.4738976
Oxygen	550.252	0.48225416	0.48225416
Water	395.2749	0.3952749	0.3952749

Dimensions

	Hydrogen	Oxygen	Water
Inner Radius	0.48365084	0.48647715	0.45527122
Outer Radius	0.56365084	0.56647715	0.53527122
Total Height	1.12730169	1.13295429	1.07054243

Tank Weights

	Hydrogen	Oxygen	Water
Area	3.99236497	4.032503	3.60045753
Weight	43.3653436	39.2160916	35.0144494

Total Weights

Tanks+Hydro.	153.834587
Tanks+Oxygen	589.468092
Tanks+Water	430.289349

Lunar Lander Mass Breakdown

Descent Stage Mass Breakdown

Subsystem	Mass Each	Qty.	Total Mass
Structure			3500
Primary Structure			2000
Landing Gear			1500
Propulsion			2058
Pressurant Tanks			50
Fuel Tanks	220	4	880
Oxidizer Tanks	257	4	1028
Misc			100
Engines			0
Power			0
Water Storage			
Conditioning & Wiring			
Avionics			0
Sensors			
Misc			
Video System			
Attitude Control			99.97
Thrusters			
Plumbing, Valves, etc.			
Fuel Tanks			
Oxidizer Tanks			
DRY MASS			5657.97
Noncargo			0
Cargo			37181
Ascent Stage			32181
Misc. Cargo			5000
INERT MASS			42838.97
Consumables			500
Propulsion Helium			500
Propellant			55000
LOX			35000
Methane			20000
NTO			
MMH			
GROSS MASS			98338.97

Ascent Stage Mass Breakdown

Subsystem	Mass Each	Qty.	Total Mass
Structure			650
Secondary Structure			50
Primary Structure			600
Propulsion			1927
Pressurant Tanks	213		
Fuel Tanks	167	2	334
Oxidizer Tanks	195	2	390
Misc			115
Engines	272	4	1088
Power			120
Fuel Cells			0
Hydrogen Tanks			40
Oxygen Tanks			40
Water Tanks			40
Conditioning & Wiring			0
Avionics			195
INS			15
Sensors	10	2	20
Misc			28
Computer System			20
Displays and Controls			37
Communication			75
Environment			205
Active Thermal Cntrl			22.7
Misc. Tankage			205
DRY MASS			3097
Noncargo			300
			300
Cargo			10535
Crew Module			10,535
Mass Dumped			
INERT MASS			13932
Consumables			767
Spare O2&N2			50
Fuel Cell Hydrogen			67
Fuel Cell Oxygen			550
Propulsion Helium			100
Propellant			17482
Fuel			3800
Oxidizer			13682
GROSS MASS			32181

Command Module Mass Breakdown

Subsystem	Mass Each	Qty.	Total Mass
Structure			3920
Pressure Vessel Structure			1272
Heat Shield Substructure			2104
Secondary Substructure			544
Protection			718
Ablator			484
Tiles			162
Insulation			72
Propulsion			252
Pressurant Tanks			35
Fuel Tanks			15
Oxidizer Tanks			15
Misc			27
Thrusters	10	16	160
Power			841
Fuel Cells	147	3	441
Conditioning & Wiring			400
Avionics			542.56
INS			36.2
Sensors			4
Misc			9.66
Computer System			260
Displays and Controls			82.7
Communication			150
Environment			1753
Spares & Consumables			293
System Hardware			350
Active Thermal Control			450
Crew Systems			660
Landing			1454
Parachute System			934
Rocket System			152
Shock Absorption			368
DRY MASS			9480.56
Noncargo			608.5
Suits and Hardware			245.6
Crew			362.9
Cargo			200
Outbound			0
Inbound			200
INERT MASS			10289.06
Consumables			36
Oxygen			
Filters, Cartridges, Etc.			

Other Life Support			35
Propulsion Helium			1
Propellant			210
Fuel			80
Oxidizer			130
GROSS MASS			10535.06