NASW-4435

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Final Design Report

for the

Self-Unloading, Unmanned,Reusable Lunar Lander Project

Submitted to

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May 5, 1991

(NASA-CR-190005) SELF-UNLOADING, UNMANNED, REUSABLE LUNAR LANDER PROJECT Final Report (Texas Univ.) 94 p CSCL 22B

N92-21161

Unclas
G3/18 0073115

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Acknowledgements

We would like to thank our mothers and fathers for conceiving us, because otherwise we would not be here to write this acknowledgement. In addition to our mothers and fathers we would like to thank the following people for their help and support:

Dr. George Botbyl, Dr. Wallace T. Fowler, Dr. John Lundberg, Dr. Lyle Clark, Mr. Tony Economopoulos, Ms. Barbara Kozel, Mr. Tommaso Rivellini, John Connaly, Craig Hoskins, Tim Vinopal, Heather Dean, Lisa Bell, Todd McCusker, Peter Priest, Chris Varner, Richard Crawford, Richard Eanes, and Ruwan Arseculeratne.

Executive Summary

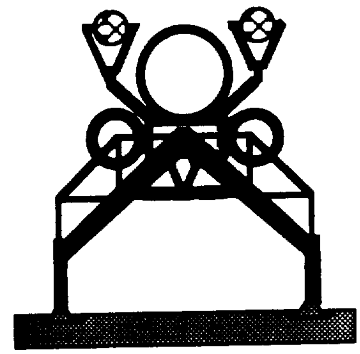
The future of the U.S. space program outlined by President Bush calls for a permanently manned lunar base. A payload delivery system will be required to support the buildup and operation of that lunar base. In response to this goal, RS Landers has developed a conceptual design of a self-unloading, unmanned, reusable lunar lander. The lander will deliver a 7000 kg payload, with the same dimensions as a space station logistics module, from low lunar orbit (LLO) to any location on the surface of the moon.

This executive summary briefly introduces the technical aspects of the design as well as the management structure and project cost.

La Rotisserie

This concept is a product of rigorous brainstorming followed by meticulous inspection of the resulting ideas. The payload delivery system consists of a lander, unloader, and payload.

As the figure shows, the payload and the unloader are loaded in an inverted position on top of the lander. After post-landing stabilization on the lunar surface, the entire structure will rotate 180° with respect to the legs. This rotation will take at least thirty minutes.



La Rotisserie

When the rotation is complete, the unloader will be lowered to the surface. The unloader will then drive out between the legs and deliver the payload to its desired location. The unloader has a range of 5 km when loaded with the payload. The 5 km range was needed because it was determined that the lander should land at least 1-2 km away from the lunar base. This is necessary in order to avoid excessive plume damage. Once the payload has been delivered, the unloader can return to low lunar orbit (LLO) with the lander, or it can remain on the surface to await the lander's return.

Main Engines

Solid core nuclear motors were chosen for use on the lunar lander. These motors have an optimistically projected specific impulse of 1200 sec. and thrust to weight ratio of 11.3. The maximum required thrust occurs during the descent phase of the mission, and it is 22,584 lbf.

It is not currently known whether a three motor configuration or a single motor configuration would be superior for use on the lander. For conventional motors, the three motor configuration is recommended for situations of engine out. There are studies being done to determine the effect of clustering nuclear motors. It may be necessary to use one nuclear motor with redundant turbopumps. All lander drawings in this document, however, show the three motor configuration.

Unloading Mechanisms

The detailed design of the mechanical components of the various payload unloading mechanisms is beyond the scope of this study, however, there are a few points that should be considered in their design. The areas that were considered are the types of electric motors, bearings, and drive train or gear reduction system that should be used.

The motors that are most promising for the RS Landers La Rotisserie concept use direct current, deliver moderate torque, medium rotation rates (around 1000 rpm), and are of a brushless design.

Coated bearings are suggested for use on the lunar lander. Lubricants will prove to be ineffective in the harsh lunar environment. They will either become filled with dust, freeze up, or boil off. Possible bearing coatings include Teflon®, Nomex®, and diamond. Diamond coatings can be applied using Chemical Vapor Deposition.

A harmonic drive system is recommended for use on the lunar lander. Harmonic drives have fewer moving parts than the conventional gear box. They are therefore less massive. Harmonic drive systems use flexible splines that wear faster than conventional gear box components, but with the advent of advanced materials, the harmonic drive can be designed to more than meet the lander's requirements.

Trajectories/Propulsion

The lander trajectories have been designed and optimized using a computer program called Lander. Program Lander was developed by Eagle Engineering in Houston, TX to simulate the ascent and descent phases of a lunar landing mission.

The landing site location of the Apollo 15 mission was chosen for the lunar lander simulation. The resulting total delta-velocities were 1.839 km/sec for ascent and 1.92 km/sec for descent. The flight times were 50 minutes for ascent and 63.25 minutes for descent. The use of the solid core thermal nuclear propulsion system greatly increased the efficiencies of the trajectories.

Structures and Materials

The lander structure provides connectivity and integrity to all of the lander's systems. The central box of the lander structure carries all of the loads generated by the subsystems. This box is a truss structure enclosed by honeycomb core panels. The truss structure is strong enough to support the loads generated by the subsystems, and its lightweight panels protect the subsystems from solar radiation, dust, and micrometeorites.

The landing gear is composed of four struts that are lightweight planar trusses with Apollo style Lunar Module landing pads. To enable the lander to remain level on an incline of up to 8°, a terrain adaptive system is incorporated into the landing gear.

Aluminum-lithium alloys were chosen as the main construction material for the lander. In addition to having the advantages of standard aluminum alloys, they can have a high tensile strength (over 100 ksi), along with increased weldability and a higher cryogenic strength.

Guidance, Navigation, & Control

The purpose of the guidance, navigation, and control (GNC) system is to determine the linear and angular position, velocity, and acceleration of the lander, to compare that data with the desired state, and to make corrections when necessary. The desired state of the lander will be provided by the predetermined trajectory analysis for each specific mission.

The GNC system is made up of three components: Sensors, Computer, and Control. The sensors provide information on the state of the lander. The computer evaluates the data from the sensors and instructs the control mechanisms. The control mechanisms then change the state of the lander by their action.

Three levels of sensors are used for redundancy. During optimum operating conditions, several components of each level of redundancy will be used. The primary, secondary, and emergency sensor arrays rely on a radar imaging/altimeter system, several sets of accelerometers and gyros, a transponder system, a close proximity altitude detection device, and the communications system. The communications system is only used as a sensor for emergency situations.

The onboard navigation computer will be a fault tolerant advanced computer that will be capable of out-performing today's most advanced Cray computer. The rapid pace of computer and software development has shown that a system of this type will not only be possible, but will have little mass and power consumption. The navigation computer will be responsible for monitoring the output and status of each sensor, monitoring the status of and providing input for each of the control devices, and providing an interface between the two.

The lander will utilize three control techniques: momentum exchange devices, small directional thrusters, and gimbaled/throttled main engines. While some redundancy exists using all three systems, the optimum operating conditions will use each technique where best suited.

The control of the unloader will be primarily automated with a remote control system as a backup. The unloader will have optical sensors which will inform the unloader's onboard computer of obstacles. The computer will then instruct the wheel motors to make the required adjustments. The unloader will be in constant communication with the lander, in case it becomes necessary to employ the back up remote control system.

Communications

The communications systems provide three basic functions: telemetry, command, and tracking. There are three areas of communications that will be performed by the lunar lander system: Lander/Earth,

Lander/OTV, and
Lander/Unloader.

S-band (2.3 GHz) will be used for direct communications between the lander and earth. The antenna on the lander will be a parabolic dish with steering capabilities similar to that on the Apollo spacecraft. The Apollo pointing system is more than sufficient for the communication link with earth.

It is also recommended that a communications satellite for Earth link capabilities be placed in a halo orbit about the L2 Lagrangian point. The satellite would allow transmissions to be made between the lander and earth when the lander is on the far side of the moon.

Communications between the OTV and the lander will be done with a VHF system. This system will be necessary for docking procedures. Once the lander is docked with the OTV a data feed umbilical will be connected to the lander by means of a manipulator arm on the OTV.

Communications between the lander and unloader will be done using a UHF system. The UHF receivers and transmitters are small, lightweight, and require little power. The UHF antennas are also small.

Power/Thermal Control

The energy for the power system is provided by the heat generated during engine cool down cycles. A power conversion loop transforms the heat into electrical energy, which is then stored in rechargeable Na-S batteries on the lander and the unloader. The conversion loop also serves to cool down the nuclear motors and keep the batteries at a higher operating temperature.

Two sets of batteries provide 11 kWh of power on both the lander and the unloader. The power for the unloader allows it to carry the payload 5 km at a speed of 2.5 km/hr. In the event that the unloader remains on the surface for an extended period, two solar arrays totalling 20 sq. m. are mounted on the unloader. These GaAs/Ge arrays are able to recharge the batteries fully in about one solar day. The solar arrays are also used to heat the unloader's batteries during payload transport.

Thermal control will be done using several methods. The first method will employ the use of a cryogenic refrigeration system that will be powered by the power generation loop. The second method will employ the use of 2 1/2" of multi-layer insulation on the propellant tanks and other areas that require thermal control. Heat exchangers on the power generation loop will also be used to keep certain areas of the lander warm.

The final method that will be used is the use of two radiation/thermal protection umbrellas. These umbrellas will be deployed from the landing struts after the complete rotation sequence has been performed. The umbrellas will help to reduce the work load on the refrigeration system considerably.

Management Structure

The management structure of RS Landers was designed for speed of communications between all group members. Progress and problems, for example, are reported directly to the C.E.O. and the Technical Supervisor. The final design is the compilation of input from all group members. All milestones for the project were completed on time.

Project Cost

The actual total cost of the project was $27,457. This is 2% less than the cost that was estimated at the start of the project.

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1.0 Introduction

The quest for a permanently manned lunar base has found its revival in President Bush's outline for the future of the U.S. space program. An efficient, reliable payload delivery system will be necessary to carry out the lunar base plan. The delivery system will transport the payload in three phases: launch from earth's surface to earth orbit, transfer from earth orbit to lunar orbit, and descent from lunar orbit to the lunar surface.

This report discusses the conceptual design of a self-unloading, unmanned, reusable lunar lander. The design presented in this report is an optional vehicle which can complete the third phase of the payload transportation from earth to the moon. This report highlights some of the options that were considered in each area of development. Finally, this report describes the management structure of RS Landers as well as the project's cost.

1.1 Mission Scenario

The lunar lander will deliver a payload from low lunar orbit (LLO) to the surface of the moon. Figure 1.1 shows the typical mission scenario for the lunar lander.

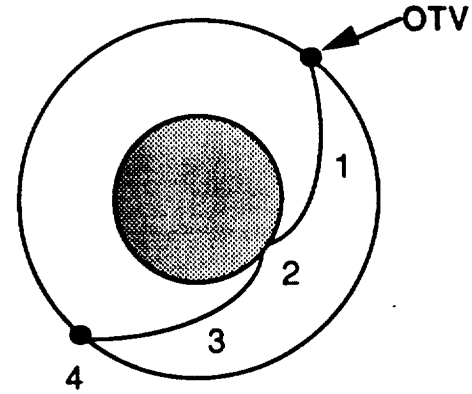


Figure 1.1 Lunar Lander Mission Scenario

While in LLO, the lunar lander will dock with an Orbital Transfer Vehicle (OTV). Phase 1 of the mission scenario is the descent from LLO to the lunar surface. Phase 2 is comprised of landing on the lunar surface and delivering the payload. Phase 2 could last anywhere from one hour to several weeks, depending on the arrival time of the next OTV. Phase 3 involves ascending back to LLO from the lunar surface. Phase 4 entails docking with the same OTV or a new OTV, refueling, performing minor repairs, recharging batteries (if necessary), entering program updates, and loading another payload aboard the lander. Phase 4 may require up to 24 hours. The ascent and descent phases of the mission scenario are discussed in detail in the Trajectory Analysis section of this document.

The four steps outlined by Figure 1 are considered one mission cycle. The lunar lander will be required to complete ten such cycles before major servicing is required.

1.2 Payload, Lander, and Unloader Requirements

The lunar lander system is comprised of three separate elements, the payload, the lander, and the unloader. The payload is the object which the lander transports from LLO to the lunar surface. The lander is the vehicle which transports the unloader and the payload from LLO to the surface. And the unloader is the device which unloads itself and the payload from the lander and delivers the payload to its designated location.

The maximum dimensions of the payload are the same as the space station logistics modules (4.42 m maximum diameter, 15 m maximum length). The maximum mass of the payload is 7000 kg. The payload will also be self-sufficient; that is, the lander is not required to provide any power, cooling, or any other support to the payload.

The lander must be able to descend from LLO to the surface of the moon and then ascend back to LLO. The lander must be refueled and reloaded in LLO by the OTV. Another requirement for the lander is that it must be able to complete ten delivery mission cycles before major servicing is required.

It was determined in an Eagle Engineering Lunar Base Launch and Landing Facility Conceptual Design Study that the lander must land one to two kilometers away from the lunar base in order to minimize damage to the base caused by engine plumes. Due to the probable plume damage, it will be mandatory to use a payload unloader to transport the payload from the landing site to the lunar base.

Since it is necessary to use an unloader, the lander must be capable of delivering the 7000 kg payload and the unloader to the lunar surface. The unloader must then be able to either remain on the lunar surface or return to LLO with the lander. If the unloader remains on the surface, the lander must then be capable of delivering a payload that has a mass of 7000 kg plus the mass of the unloader. The unloader will then be required to deliver the larger payload to its designated location on the lunar surface.

1.3 Additional Assumptions

It was necessary for RS Landers to make several additional assumptions for the lunar lander's general operational scenario. The first assumption was that RS Landers can specify the packaging of the payload. This was done so that a commanality could exist between all payload/lander/unloader interface mechanisms.

The next assumption was that the lunar lander will not be required to land on any inclination greater than ±8°. Inclinations greater than this could severely inhibit the unloader from removing itself and the payload from the lander if the lander was oriented improperly with respect to the inclination.

The third assumption was that the lander will not be required to land in an area that contains a large number of obstacles larger than 0.5 m in height. Obstacles larger than this could also inhibit the unloader from functioning properly.

The fourth assumption was that the lander can either be totally automated and/or remotely controlled. With the rapid advancements in computer hardware and software, automation becomes easier. However, mission control may wish to override a preprogrammed mission operation or take control of the lander if an emergency were to occur.

The next assumption was that there will be no restrictions on the types of fuels used near the OTV. The OTV is not expected to be a scientific vehicle that will have sensitive instruments that could be damaged by some types of rocket fuel.

The final assumption was that the necessary infrastructure for the construction, maintenance, and delivery of the lander to LLO will be available. It is, therefore, assumed that a heavy lift launch vehicle will be available to transport all of the lander's components to LEO in one launch. The lander will be assembled in LEO at a space station or a transportation node. Only after assembly will the lander be supplied with its nuclear and cryogenic fuels. The lander will then be transported to LLO by an OTV. It is also expected that the necessary support structure for the operation of the lunar lander will be in place before the lander's first mission.

2.0 Critical Areas of Development

With all of the initial designs that were considered during the conceptual design process, there are two critical areas of development. The first critical area is the main propulsion system, and the second is the unloading mechanism. These two areas greatly affect the final lander design, but they only affect the size of the lander configuration.

Graphical Simulation

The lander mission has been animated using a Personal IRIS™ computer. The simulation includes four phases: landing, rover vehicle and payload unloading, rover vehicle retrieval, and launch. The simulation will be available on video tape.

2.1 Lander Main Engines

The lunar lander's main propulsion system is one of the most critical subsystems that was considered. The propulsion system was not only the major contributor to the total mass determination process, it also affected the design of several other subsystems. Some of the design areas that were affected by the choice of the main propulsion system included attitude control, lander structure, thermal control, and power systems.

Two types of propulsion systems were considered. The first type utilizes chemical propellants and the second type relies on thermal nuclear propulsion. The items that were considered in the selection of the final propulsion system are the following:

- High I_{sp}
- High thrust to weight ratio
- Variable throttling capability
- Reliability
- Restart capability
- On-orbit refuelling
- 10° gimbal range
- Fuel storage considerations
- Effect on other lander systems
- Effect on the lunar environment

2.1.1 Chemical Rocket Motors

Only liquid chemical rockets were considered for use on the lunar lander. Solid core chemical rockets were not considered because they are not restartable or immediately reusable, and their specific impulse is low. Hybrid chemical motors were ruled out because they are not immediately reusable after each mission. New motors would have to be installed after each mission, or the hybrid motors would have to be excessively large in order to contain enough solid propellant to complete ten mission cycles.

After a review of various liquid chemical propellants that are available, it was determined that a liquid O_2 (LO_2) and liquid H_2 (LH_2) combination would be the most advantageous chemical propellant. This combination was considered because of its high specific impulse (480 sec.), its limited environmental impact, its high thrust to weight ratio, its variable throttling and restart capability, and because there are already several motor designs that use this combination. Table 2.1 contains the performance characteristics of LO_2 and H_2.

Table 2.1 Hydrogen and Oxygen Performance Characteristics

Mixture Ratio by Weight	3.5
by Volume	.21
Ave. Specific Gravity, (g/cc)	.26
Chamber Temp., (°F)	5870
Ratio of Specific Heats	1.22
Bulk Density, (g/cm^3)	.43

The use of LO_2 and LH_2 also has several negative aspects. One is that using the LO_2/H_2 combination would require the use of cryogenic storage methods. Cryogenic storage requires the use of a refrigeration system that would use large amounts of power. Insulation, thermoelectrics, and heat radiators are an alternative to a refrigeration system, but their use would most likely not be sufficient in preventing excessive fuel boil off.

Another drawback to the possible use of LO_2/LH_2 engines is that life expectancy of current motors is not long enough to satisfy the lander design requirements. The Space Shuttle Main Engines are only capable of approximately six minutes of burn time before servicing is required. The current burn time of LO_2/LH_2 engines is only sufficient for the descent phase of one mission cycle. More reliable motors with longer life spans will have to be designed, or an efficient space based maintenance system will have to be devised if LO_2/LH_2 motors are to be used.

2.1.2 Thermal Nuclear Rocket Motors

Thermal nuclear propulsion was considered because of its many desirable characteristics. Some of the desirable characteristics include a high thrust to weight ratio, restart capability, variable throttling capability, 10 hour burn time, high specific impulse (850 to 1200 sec for solid core and 2000 to 3000 sec for gaseous core), and power generating capabilities for both the lander and the lunar base. The development of nuclear motors is also critical for the advancement of the space program. No human will ever make it past Mars, and maybe not even to Mars, without the use of nuclear rocket motors. The lunar lander program could be a good proving ground for the use of thermal nuclear motors.

In a thermal nuclear rocket motor a nuclear reaction is used as the energy source rather than a chemical reaction. The binding energy of a proton or neutron is on the order of several million electron volts which is several orders of magnitude higher than the energy released by a chemical reaction. The increased energy per reaction allows much higher specific impulses to be achieved.

Choosing the propellant for a nuclear rocket is an important decision. The nuclear reaction in the core is used to heat the propellant so that it can be accelerated through a converging-diverging nozzle. Since the propellant does not have to supply its own energy, a wider range of propellants can be used. The propellant that will most likely be used in the designs that NASA is currently reviewing is cryogenic hydrogen. The low molecular weight of hydrogen makes it ideal for achieving higher specific impulses. Cryogenic storage of hydrogen will not be a problem if nuclear motors are used on the lunar lander. The nuclear motors can be used to generate power that can be used to run a refrigeration system as well as other onboard electrical systems.

The biggest negative aspect of nuclear motors is that there are no fully developed motors available at this time. With the cancellation of the NERVA project in the early 1970's, the USA's nuclear motor program was put on hold. Recently, because of the Mars mission plans proposed by the Bush administration, NASA has become interested in nuclear motors again. Currently there are several nuclear motor studies and tests underway.

The next negative aspect of nuclear motors is that heavy shielding is required to protect the payload and the lander's instruments. However, even with the added weight of the shielding, the nuclear motors are more fuel efficient than the chemical propellants that were considered. Appendix B discusses the results from the trajectory analysis program that was used to analyze the trajectories and the motors that were considered.

2.1.2.1 Solid Core Thermal Nuclear Propulsion

The first type of nuclear motor that was considered is the solid core thermal nuclear. It is depicted in Figure 2.1. There are several nuclear fuels that are under consideration by NASA, one being uranium carbide pellets coated with a variety of materials such as zirconium-carbide. Several reactor core designs are also being evaluated by NASA. The two most promising core designs are the particle bed reactor and a core, which would utilize a graphite matrix to house the fuel pellets.

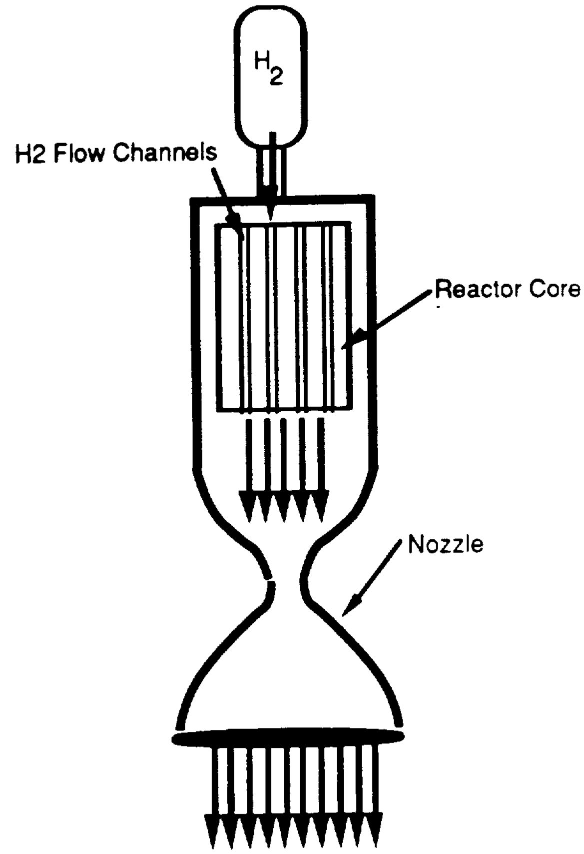


Figure 2.1 Solid Core Thermal Nuclear Rocket

The solid core would provide the energy source for the thermal nuclear rocket and turbopump, or possibly a pressure fed system would be used to feed the H_2 into the core chamber. The cryogenic H_2 would be pushed through flow channels in the reactor core, and around the reactor core, to maximize the heat transfer between the core and the propellant. The H_2 would not only serve as a propellant, it would serve as a core coolant so that thermal damage to the core materials could be controlled.

2.1.2.2 Gaseous Core Thermal Nuclear Propulsion

The second type of thermal nuclear rocket that was considered is the gaseous core motor. There are two types of gaseous core motor concepts, the open core and the light bulb core. Uranium 235 is one of the nuclear fuels that is currently being considered by NASA for both gas core uses.

Figure 2.2 depicts the open core gaseous nuclear motor. The nuclear plasma is contained in a vortex that is created by the flowing propellant. This arrangement allows for a greater amount of heat transfer between the fuel and the propellant. However, this type of motor allows small amounts of radioactive material to escape to the outside environment. Even though the projected ratio of propellant particle to radioactive particle expelled is 1500/1, the open core motor would not be acceptable for use on the lunar lander.

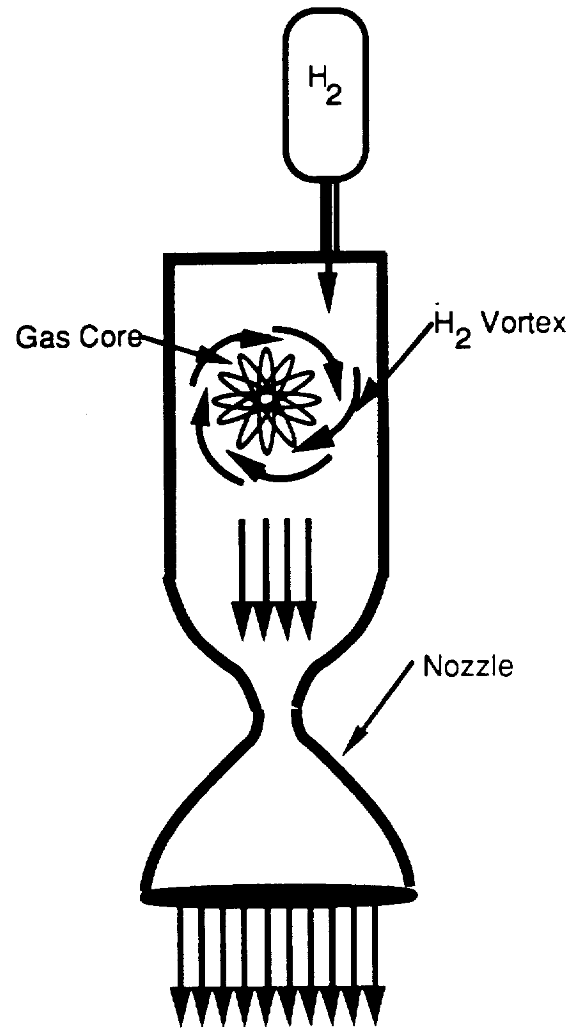


Figure 2.2 Gaseous Open Core Thermal Nuclear Rocket

The light bulb gaseous core nuclear motor concept is depicted in Figure 2.3. Unlike the open core motor the nuclear plasma is contained in a cylindrical structure that will most likely be made of advanced ceramics. The propellant is used to cool the cylindrical housing before it is expelled. Because the nuclear reaction is contained by the light bulb structure, it will not expel any radioactive material with the propellant as the open core motor would.

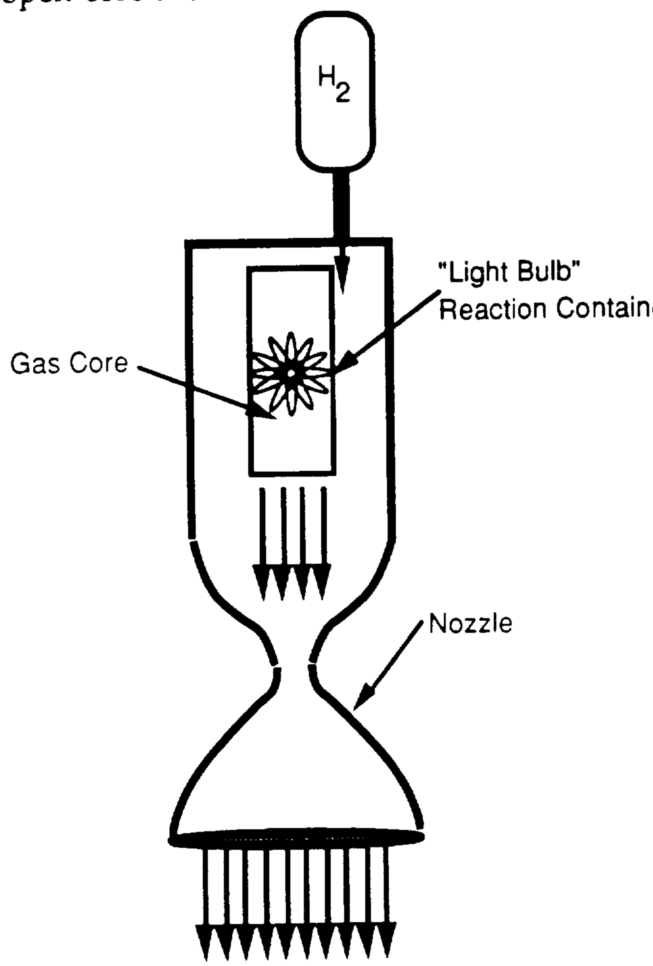


Figure 2.3 Gaseous Light Bulb Core Thermal Nuclear Rocket

The major drawback to the use of either gaseous core motor is that they will not be available for use on a lunar lander. The gaseous core motors will be a second generation system and they will not be fully developed until the mid twenty-first century.

2.1.3 Safety Issues Concerning Thermal Nuclear Motors

Even though the use of nuclear motors has several advantages, there are a few safety issues that must be seriously considered. Some of the issues can be easily solved, but others will require the advent of innovative ideas and methods.

The first issue that will be raised is the transportation of nuclear fuel to Low Earth Orbit (LEO) and beyond. This however, should not be a major issue for two reasons. The first reason is that fuel transport cannisters that can withstand a launch vehicle explosion have already been designed and have been in use for several years. Both the USSR and the USA have been transporting nuclear fuel to LEO for use in RTG's. The Soviets have also been using nuclear reactors in orbit with few mishaps. The second reason that the transport of nuclear fuel to LEO and beyond is not a major issue is that the nuclear fuel is not highly radioactive until after it has been placed in a reactor and used. If a launch vehicle explosion occurred that ruptured a transport cannister, the damage done to the environment by the release of small amounts of coated uranium pellets would be minimal.

The second safety issue that will be brought up is lander maintenance personnel safety. This is not a problem because the motors will have adequate shielding to make maintenance on most of the lander safe for humans in radiation protected space suits. Maintenance that is required on areas of the lander that are not sufficiently shielded (under the nozzles) can be done by robots.

Another safety issue that must be considered is the possibility of a lander crash on the lunar surface or with the OTV. A nuclear powered lander crash would not cause a nuclear explosion. However, if the crash was severe enough to rupture an engine core, radioactive material would be released. The amount of radioactive material released would depend on the severity of the crash and on the number of engines used by the lander. If one motor was used the possibility of more radioactive material being released is higher than if three motors were used. One motor uses approximately 25% less nuclear fuel than three motors, but all three motors on the lander would not necessarily be ruptured in a crash which is just severe enough to rupture one motor.

If a lander crash that was severe enough to rupture an engine core occured on the surface, that area of the surface would have to be quarantined until a clean up effort could be made. The possibility of the radioactive material damaging the lunar base would be small. This is because the landing area will be at least a kilometer away from the base, and the moon has no atmosphere to transport radioactive material.

If a lander crash occurred with the OTV and an engine core was ruptured, the severity of the crash would dictate the appropriate measures. In the case of a minor crash the safety of an OTV crew would be paramount. In the case of a major crash where both vehicles are obliterated, the addition of small amounts of radioactive material to the debris would be insignificant. The debris itself would be more dangerous to incoming spacecraft than the radioactive material.

2.1.4 Number of Engines

Several studies have been done on the advantages and disadvantages of using multiple engines or a single engine. Studies conducted by Rocketdyne, Pratt & Whitney, and Aerojet indicate using three is most efficient when considering weight and failure analysis. These studies were conducted with the use of chemical rockets in mind.

It may turn out that when thermal nuclear rockets are used, that the use of one engine with redundant turbopumps will be more efficient in terms of weight and reliability. Several studies on the clustering of thermal nuclear motors are currently underway. It is not yet known whether one engine in a cluster will adversely effect other engines in that cluster. With this in mind, using both one and three thermal nuclear motors was considered in the design process. All lander drawings, however, do show the three engine configuration.

2.1.5 Main Engine Selection

Due to the number and magnitude of the positive aspects of the solid core thermal nuclear motors they were chosen for use on the RS Landers lunar lander. The Eagle Engineering trajectory analysis program Lander was used to determine the maximum thrust required, delta velocities, propellant requirements, mass flow rates, and flight times using both one and three motors. Appendix B contains a discussion of the Lander program as well as the additional calculations that were used.

If three motors are used the maximum required thrust is 22,600 lbf. Three motors with 11,300 lbf thrust each would be used and run at 66.7% to achieve this thrust level. If engine-out occured, two engines run at 100% could be used to complete the deorbit burn. The projected total engine mass for the three motor configuration is 3000 kg. The expected engine performance characteristics for each motor in the three motor configuration are shown in Table 2.2. It is recommended that nozzle expanders be used in order to improve the efficiency of the motors. The length and diameter shown in Table 2.2 were used in order to facilitate the lander rotation sequence. The nozzles could be expanded up to an exit diameter of 3.33 m and length of 4.5m using expanders. The expansion ratio that is shown in Table 2.2 is based on current NASA approximations. This would correspond to the 2.0 m exit diameter.

Table 2.2 Engine Performance for the Three Motor Configuration

Propellant	LH_2
Thrust, (lbf, vac)	11,300
I_{sp} (sec.)	1200
Gimbal (°)	10°
Chamber Temp.(°R)	4840
Chamber Press.(psi)	3000
Exit Diameter (m)	2.0
Length (m)	3.0
Total Weight (kg.)	1000

If one motor is used the maximum required thrust is also 22,600 lbf. A motor with 25,112 lbf maximum thrust will be used and run at 90%. Three turbopumps will be used for redundancy, any two of which can be used to achieve the 22,600 lbf thrust level. Table 2.3 contains the performance characteristics for the one engine configuration. It is also recommended that a nozzle expander be used in the single engine configuration in order to improve the motor's efficiency. The length and diameter shown in Table 2.3 were used in order to facilitate the lander rotation sequence. The nozzle could be expanded up to an exit diameter of 12.0 m and length of 4.5m using expanders. The expansion ratio that is shown in Table 2.3 is based on current NASA approximations. This ratio would correspond to the 8.0 m exit diameter.

Table 2.3 Engine Performance for the Single Motor Configuration

Propellant	LH_2
Thrust, (lbf, vac)	25,112
I_{sp} (sec.)	1200
Gimbal (°)	10
Chamber Temp.(°R)	4840
Chamber Press.(psi)	3000
Expansion Ratio	500/1
Exit Diameter, (m)	8.0
Length (m)	3.3
Weight (kg.)	2222

The thrust-to-weight ratios and specific impulses for both the one and three engine configurations are based on several conflicting reports. These numbers are relatively optimistic.

2.2 Unloader Mechanisms

The detailed design of the mechanical components of the payload unloading mechanisms is beyond the scope of this study; however there are a few points which should be considered in their design. The areas that were considered in this study are the types of electric motors that should be used, the types of bearings that should be considered, and the types of drive train or gear reduction systems that should be considered.

2.2.1 Electric Motors

The first consideration for the payload delivery system is the type of motors that will be used. Key factors in the selection of a motor are maximum reliability and minimal weight. The motors that are most promising for the RS Lander concepts use direct current, deliver moderate torque, medium rotations rates (around 1000 rpm), and are of a brushless design.

Electric motors can utilize either Alternating Current or Direct Current electrical power. A D.C. motor will be more suitable since most space power systems provide D.C power. The use of A.C. motors would require the addition of power inverters, which would increase the weight of the spacecraft and reduce the efficiency of the electrical system. These inefficiencies and weight increases are second only to reliability in importance in the design of equipment for space applications.

The next major consideration in motor selection is the type of current commutation system. Conventional motors use brushes to skim along a commutator to switch the current in the windings on a rotor in a magnetic field. The commutation of the current causes the rotor to rotate. In brushless motors the the current is electronically commutated in the stationary windings which interact with a permanent magnet on the rotor to cause the rotor to rotate.

In brush motors, the friction between the brushes and the commutator limits the performance of the motor in several ways. One limitation is a result of the heat generated from the friction. The potential speed of the rotor is limited by the ability of the motor to reject the heat and by the thermal limitations of the materials in the motor. Since there is no atmosphere on the moon to allow the convection of heat from subsystems, heat generation is a significant problem in the design of mechanical systems. The friction also causes the brushes and the commutator to show wear rapidly. As a result, the design life of the system utilizing the brush motor will, among other factors, be limited by the motor. For extended life, the motor will require major servicing to replace the brushes and the commutator. Also, as the brushes wear, they create dust within the motor which can degrade the bearings and shorten their life. In either type of motor, the bearings supporting the rotor will eventually fail and require replacement; however, experience shows that the brushes are likely to be a limiting factor in the life of a motor.

Brushless motors also have failure modes. Their operation is dependent upon an electronic circuit which monitors the position of the rotor and controls the current in the windings to keep the rotor turning. The failure of the shaft sensor would not be a significant problem because a brushless motor can operate, albeit less efficiently, without the sensor information. A more serious problem would be the failure of the controller. However, this mode of failure is common to both types of motors, since position and speed control is important in most applications for the motors. Failure in the control system can be dealt with most efficiently through redundant systems. The control system can also be made modular so that the system could be "easily" replaced.

An additional problem could arise from the demagnification of the rotor magnets. This becomes a potential problem at temperatures above 100° C. There are at least three means of dealing with this problem. First, the magnet can be made from a material which is capable of with standing the 120° C maximum temperature on the moon. Most high energy magnetic materials can handle the high temperature, so this will probably be the best solution. A second solution would be to place the motor in a place where it is not likely to be exposed to the sun for long periods of time. The practicality of this solution depends on the application of the motor. The third solution would be to provide a means of removing the heat from the motor. Such a system will increase the weight penalty of using the motor; however, a system will be required to remove the heat generated in the windings despite the type of motor used. Actually the cooling problem is less severe in a brushless motor because the windings are stationary and can be mounted on a heat sink.

Space systems are designed for a given lifetime, and either type of motor could be designed to withstand the required amount of use. Another consideration is the ability of a system to perform beyond the expected life. Choosing systems which have fewer modes of failure and easier means of repair will increase the capability and hence, the survivability of the overall project. Due to these considerations,. brushless motors are preferred over conventional, brush-type motors.

2.2.2 Bearings

The next major area of consideration for the payload delivery systems is the types of bearings that should be used. The support of moving parts presents a problem in the lunar environment. In general, lubricants typically used on the earth will not perform in a vacuum. Such lubricants tend to evaporate or boil off in the vacuum. The temperature variations also pose a problem, since the temperature can vary from -150°C to +120 °C. Lubricants which can handle the low temperature are likely to boil off at the high temperature, while lubricants designed for the high temperature are likely to freeze at the low temperature. The best method is to use self-lubricating bearings. Such bearings utilize coatings which have low coefficients of friction, which facilitate the motion between the parts. An example of conventional self-lubricating bearing utilizes a braided layer of Nomex® and Teflon®.

Another possible surface coating is diamond. Through Chemical Vapor Deposition (CVD), a diamond or diamond-like coating can be applied to the surface of a bearing. A diamond-like coating has several beneficial qualities. The coating would have a low coefficient of friction to minimize heat build up and power dissipation; also, the coating is very hard, which provides a lower wear rate as compared to other surfaces. The thermal conductivity can be tailored, based on the material to provide optimum heat transfer from the surfaces. The coating can also be shaped, since it will conform to the coated surface.

2.2.3 Drive Train/Gear Reduction

There are two means of achieving the necessary gear reduction. The first type is a harmonic drive reduction unit, and the second is conventional gear box design. In a harmonic drive, a wave generator rotates within a toothed flexible spline, which engages a stiff circular spline that has a different number of teeth than the flexible spline. If the circular spline has one more tooth than the flexible spline, it will rotate counter to the wave generator at the relative rate of one tooth per revolution. If the difference is one tooth less, then the circular spline will rotate with the wave generator at one tooth per revolution. The rate of rotation increases as the difference in the number of teeth increases.

The advantage of the harmonic drive is that it has fewer moving parts than the gear box and is, therefore, less massive. A disadvantage to the system is the use of the flexible spline. The amount of torque which can be transmitted is dependent upon the shear strength of the teeth in the gears. Since flexible structures generally have lower shear strength than stiff structures, the available force or torque capacity of the harmonic drive is limited. The depth and width of the teeth, as well as the number of engaged teeth, can be increased to increase the available torque. There are, however, practical limits that must be observed.

An additional limitation is in the amount of reduction that can be achieved. The gear reduction is dependent upon the number of teeth in the flexible and circular spline. To achieve a gear reduction of 125:1, the flexible spline would require 125 teeth while the circular spline would have to have either 124 or 126 teeth. Similarly, in order to achieve a gear reduction of 1000:1, both splines must have approximately 1000 teeth. Increasing the teeth on the splines results in higher tooth pitch. Increasing the tooth pitch tends to increase the likelihood of a malfunction or jamming, which reduces the efficiency of the drive. These are not reasons to abandon the harmonic drive, but they must be considered when choosing a method of gear reduction.

It is also possible to combine these two designs to take advantage of the strengths of both. The exact configuration will depend upon the needs and cannot be generally addressed.

2.3 La Rotisserie Lander/Unloader Concept

Various lander/unloader concepts were considered by RS Landers during the lunar lander design process. The critical areas of design which were explained above, were a major factor in the sizing of each lander system. Appendix A contains a detailed description of the design process, as well as descriptions of each lander concept that was considered.

At the completion of the lander/unloader concept analysis process, the La Rotisserie Lander/Unloader concept was chosen by RS Landers as the best concept. Appendix A contains detailed CAD drawings of the La Rotisserie Lander/Unloader concept. These drawings include overall lander dimensions, unloader motion relative to the lander, and subsystem layout diagrams.

Figure 2.4 contains a front view of the La Rotisserie lander concept. The payload and the unloader are placed in an inverted position on top of the lander. Once the lander has landed and is stabilized, its entire structure will go through a 180° rotation about a central axle. This rotation will be performed with a motor that is mounted at one end of the axle. A second motor of a different type will be mounted at the opposite hub for redundancy.

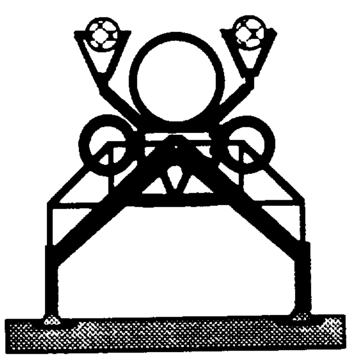


Figure 2.4 La Rotisserie Lander Concept Front View

The rotation sequence, which will take at least thirty minutes, is shown in Figure 2.5. After the spacecraft structure is inverted, the four male/female connectors (Sec. 2.4) will be electro-mechanically opened, and the unloader and payload will be lowered to the surface by a cable/winch system. The unloader will then be free to drive out from below the lander to deliver its payload to the desired location. Upon arrival at the desired location the unloader will lower the payload to the surface by means of a cable/winch system similar to that of the lander's. The unloader will then be able to return to the lander and be lifted aboard by the cable system, or to remain on the lunar surface a safe distance away from the lander's exhaust plume.

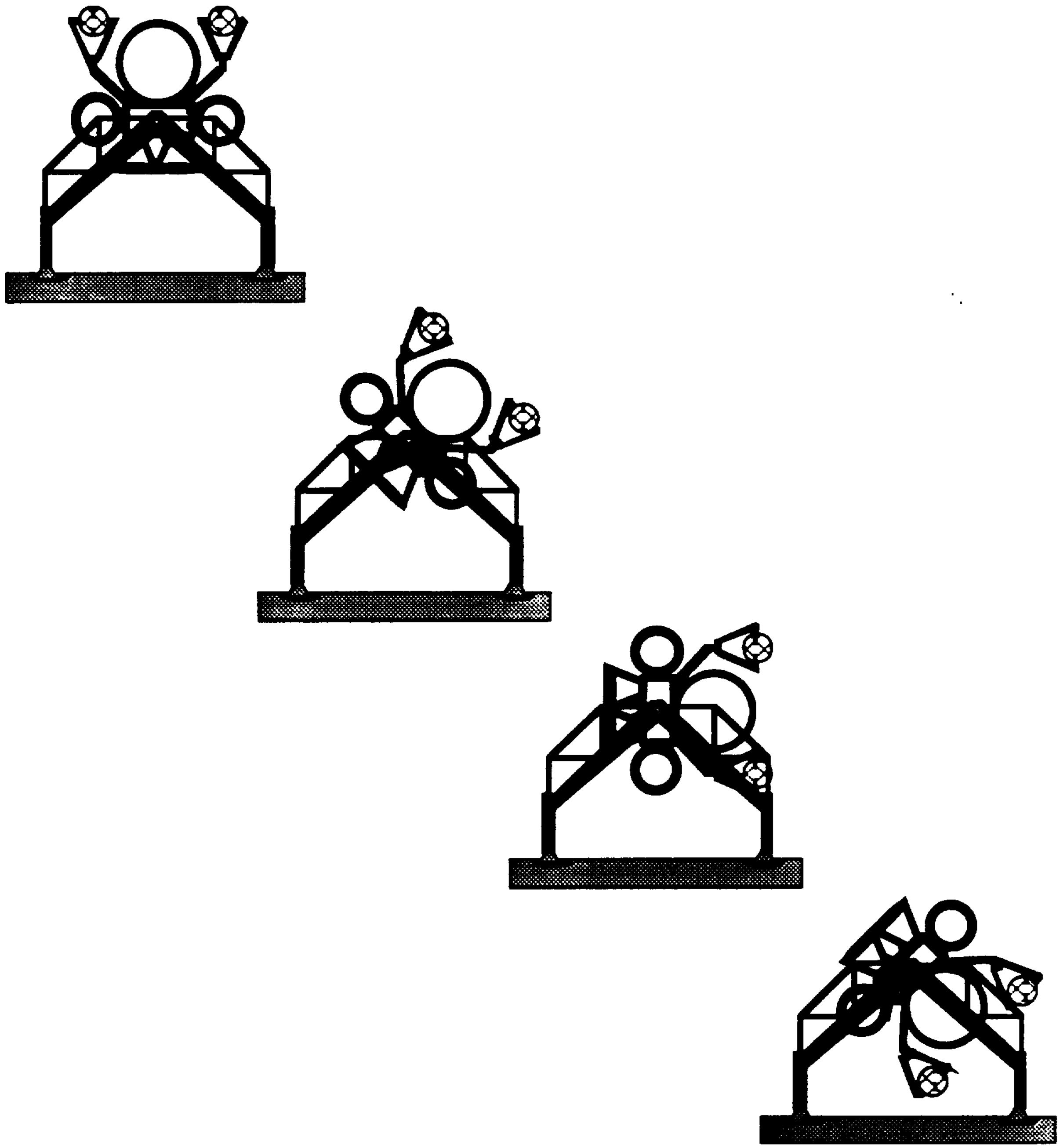


Figure 2.5 La Rotisserie Lander Rotation Sequence

Figure 2.6 contains a side view of the La Rotisserie concept. The unloader is shown to be a derivative of a log transport cart used in the logging industry. It will utilize an all wheel drive system for redundancy purposes and four wheel steering for maneuverability. Several different frame designs and wheel types were investigated. This unloader will give approximately one meter of clearance between the bottom of the payload and lunar surface during transport.

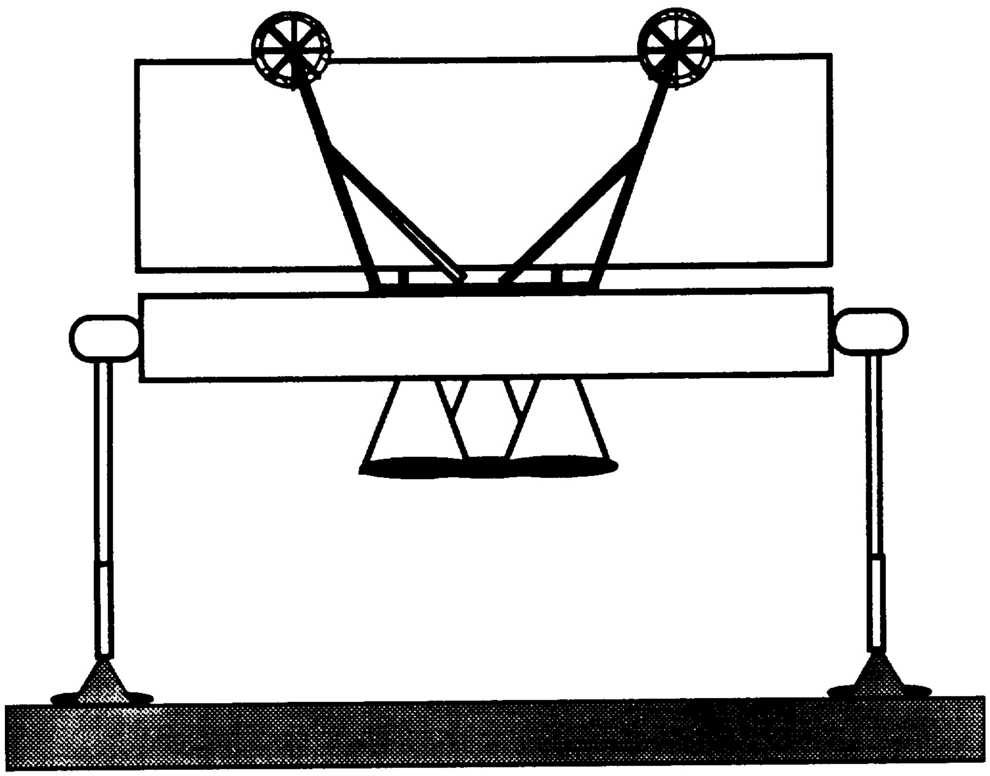


Figure 2.6 La Rotisserie Lander Concept Side View

2.4 Docking Procedures

The lander will dock with the OTV by latching directly onto the new payload. Figure 2.7 portrays the docking of the lander with the OTV while carrying the unloading cart. As shown in this figure, the unloading carts legs are folded down and out of the way to avoid damage that could possibly result from inadvertent contact with the OTV. The lander control system will be used to align the docking mechanisms and the lander Reaction Control System (RCS) burns will be performed to ensure a soft docking. Once the lander has successfully docked with the OTV, the lander will be refueled with refueling booms.

A data feed and power umbilical will also be attached to the lander by an additional OTV boom.

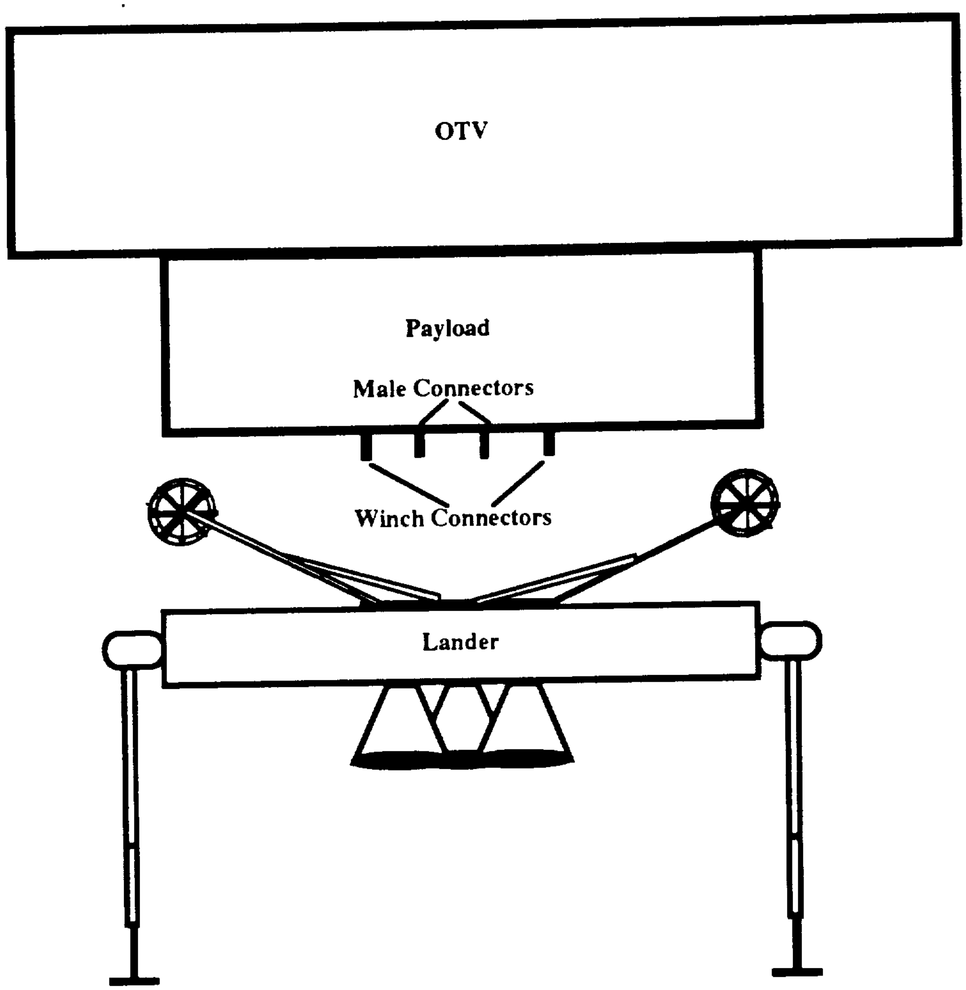


Figure 2.7 Lander/OTV Docking

The lander will latch onto the payload by using a series of four male/female connectors and two winch latching mechanisms. The female connectors and winch latching mechanisms are configured on the top of the lander and on the upper inside portion of the unloading cart as shown in Figure 2.8. Both the lander and unloader have these mechanisms to enable the payload to be attached either to the unloader or directly to the lander. Since the lander will be required to carry a wide range of payloads, each payload will be required to have compatible male and female winch connectors.

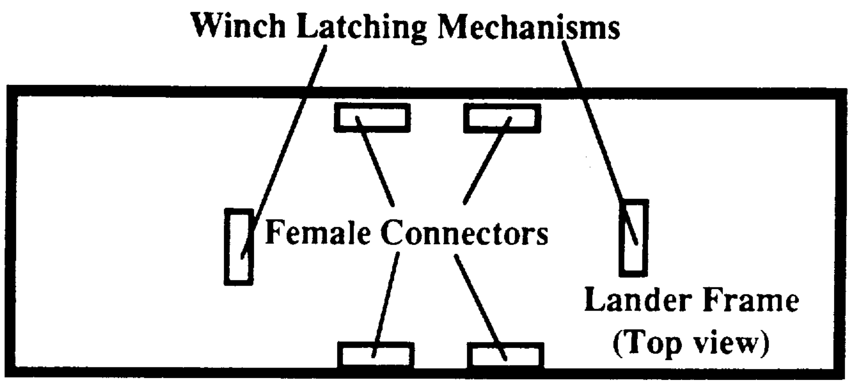


Figure 2.8 Lander/Unloader/Payload Interfaces

3.0 Additional Areas of Development

In addition to the critical areas of development for the lunar lander, there were other areas of development that are common to every spacecraft. Those areas included: trajectory analysis, structural design, materials, guidance, navigation, control, communications, onboard power systems, and thermal control. The trajectory analysis was used to help determine the optimum propulsion system for the lander. Once the optimal design for the propulsion system and payload delivery system were decided upon, it became possible to incorporate the optimal subsystems to fulfill the requirements for the other areas of development.

3.1 Trajectory Analysis

The lander trajectories have been designed and optimized using a computer program called Lander. A description of this program and the calculations for the ascent and descent trajectories is in Appendix B. The users manual for this program can be found in the NASA Document EEI Report 88-195. Program Lander was developed by Eagle Engineering to simulate the ascent or descent phases of a lunar mission to or from LLO. The landing site location of the Apollo

15 mission was chosen for the lunar lander simulation. This landing site has a latitude of 26.1011°. north and a longitude of 3.6528° east, which is located within the Hadley Rille-Apennine Mountain Range. The LLO for this mission is a 100 km altitude circular orbit with a 26.2° inclination. A small plane change of roughly 0.1° was therefore required for the ascent mission.

The ascent trajectory consists of three segments: a vertical takeoff and pitch-over maneuver, a gravity turn ascent trajectory, and an elliptical transfer to LLO. Figure 3.1 illustrates the takeoff, pitch-over, and gravity turn maneuvers for ascent. The ascent starts with a full thrust, vertical takeoff maneuver. When the local velocity reaches 30 ft/s, the spacecraft turns down range using a 70° pitch-over maneuver and proceeds to fly a gravity turn until Main Engine Cutoff (MECO). The spacecraft then coasts from pericynthion (altitude at MECO) to LLO on an elliptical transfer orbit. An orbital insertion maneuver is finally used to achieve a LLO of 100 km.

Figure 3.2 illustrates a scaled plot of the ascent gravity turn and the complete ascent trajectory with dimensions, delta-v's, and transfer times. The launch and gravity turn phases were found to require a delta-v of 1.81 km/s and a 5.7 minute flight time. The MECO altitude was found to be 22.2 km, which is the pericynthion of the elliptical transfer orbit. The transfer orbit was found to be a 22.15 x 110.24 km altitude, elliptical orbit with an eccentricity of 0.0244. The transfer time from pericynthion to LLO was found to be 44.35 min. The LLO insertion burn was found to require a 0.0297 km/s burn at a thrust angle of 0.8963° from the vehicle flight path. The total ascent delta-v and transfer time were found to be 1.839 km/s and 50 min respectively.

The descent trajectory consists of four phases: a Hohmann transfer from LLO to pericynthion, a gravity turn descent trajectory, a vertical pitch-up and hover maneuver, and a landing. Figure 3.3 illustrates the gravity turn, pitch-up, and hover/landing maneuver for descent. The descent starts with a deorbit burn in LLO to achieve the speed of a Hohmann transfer orbit at apocynthion. The spacecraft then coasts to pericynthion where it reignites its engines and performs a retrothrust burn to begin the gravity turn descent. When the local horizontal velocity reaches zero, the lander pitches up to a vertical orientation and begins to hover in search of a landing site.

Figure 3.4 illustrates a scaled plot of the descent gravity turn, and the complete descent trajectory with dimensions, delta-v's, and transfer times. The LLO deorbit phase was found to require a delta-v of -0.019 km/s, to reduce the velocity to 1.614 km/s. The transfer time from LLO to pericynthion is 57 min and the transfer orbit pericynthion altitude and eccentricity were found to be 16.85 km and 0.02313 respectively. The gravity turn descent and hover/landing maneuver were found to require a delta-v of 1.90 km/s and a 6.25 min transfer time. The Lander simulation used a one minute hover time and a 70° pitch-up angle. The total descent delta-v and transfer time were found to be 1.92 km/s and 1 hour, 3.25 min respectively.

The ascent and descent trajectories were compared to those of similar lunar lander missions. The *Apollo 15 Mission Report* (NASA Document MSC-05161) gives the spacecraft characteristics and trajectory data of the Apollo 15 mission. The Apollo 15 lunar liftoff mass was 4,951 kg, and the deorbit mass was 16,202 kg, which are 25 % and 57% less than our lift-off and deorbit masses, respectively. A detailed mass breakdown of our lander is contained in Appendix C. The Apollo 15 holding orbit was a 92.8 x 104.5 km orbit, with an inclination of 26.2°. The Apollo 15 delta-v's for ascent and descent were 1.92 and 2.0 km/s respectively. Our delta-v's are 4.2% and 5% less for ascent and descent. Since our masses are significantly larger than the Apollo 15's, our ascent and descent performances are much more efficient. This is due to the fact that our lander uses solid core thermal nuclear engines with a 1200 second I_{sp}, as opposed to the Apollo 15's Isp of 303 Sec. This performance loss was further studied using the Lander program for test cases using LOX. The corresponding trajectories are illustrated in Figures 3.5 and 3.6. The corresponding delta-v's for ascent and descent using LOX were found to be 2.010 and 2. 31 km/sec, which are roughly 8.5% and 16.9% larger than the corresponding Nuclear delta-v's

The Boeing report "Space Transfer Vehicle (STV) Concepts and Requirements Study" gives trajectory data for both gravity turn and low angle lunar trajectories. These trajectories are illustrated in Figure 3.7.

The Boeing Thrust to Weight ratios (T/W's) and I_{sp}'s were 0.2 and 465 Sec for ascent, and 0.33 and 465 Sec for descent. The gravity turn ascent and descent delta-v's were given as 1.867 and 1.933 Km/s. The low angle ascent and descent delta-v's were 1.861 and 1.891 Km/s respectively. The Boeing report concluded that low-angle descent trajectories required roughly 3% less delta-v than gravity turns. Since our trajectories used gravity turns, we should expect to improve our ascent and descent delta-v's by using low-angle trajectories. The low angle ascent and descent trajectories for our lander would require roughly 1.748 and 1.84 Km/s respectively.

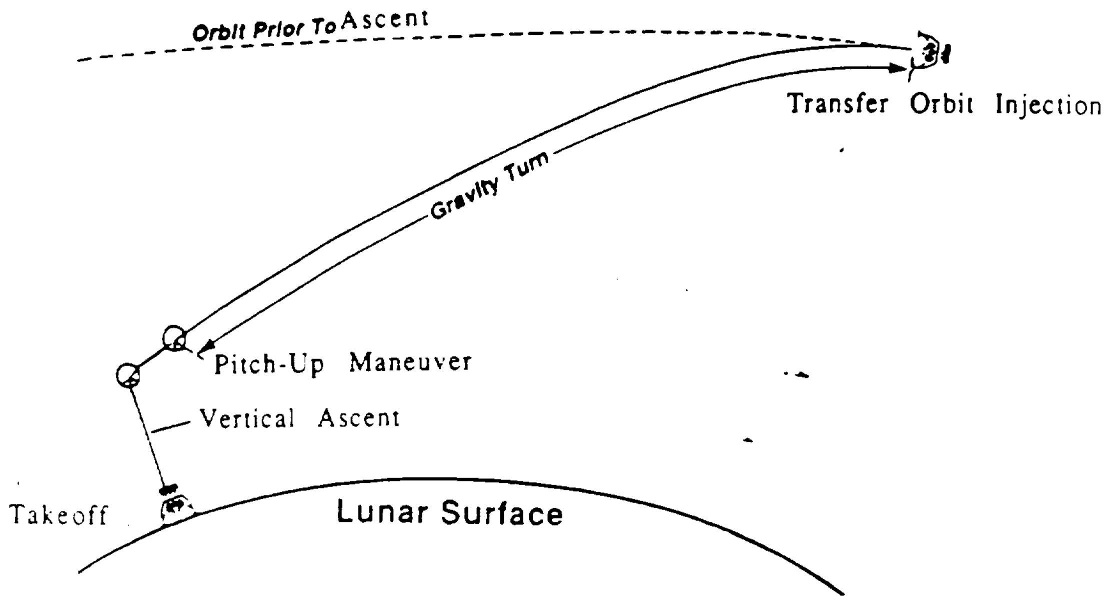
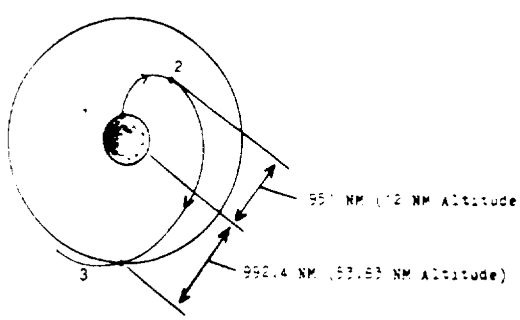


Figure 3.1: Gravity Turn Ascent Trajectory



TOTAL ASCENT TRAJECTORY

PHASE	DELTA-V (Km/Sec)	DELTA-TIME
1-2	1.81	5.7 Min
2-3	0	44.35 Min
3-4	0.0297	2.52 Sec
TOTAL	1.839	50 Min

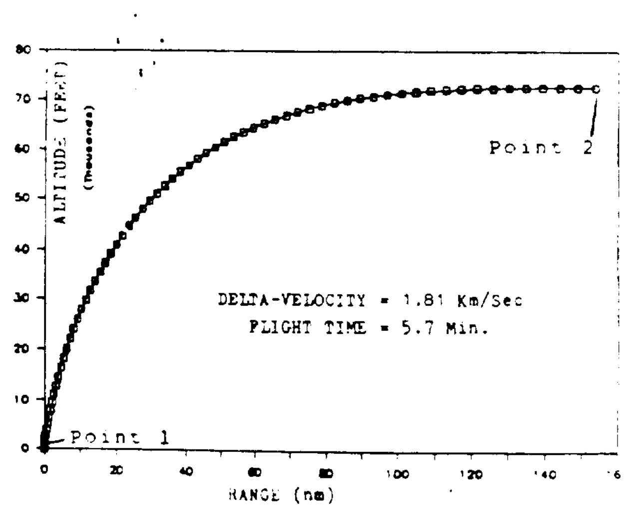


Figure 3.2: Scaled Ascent Trajectory Using Solid Core Thermonuclear Propulsion

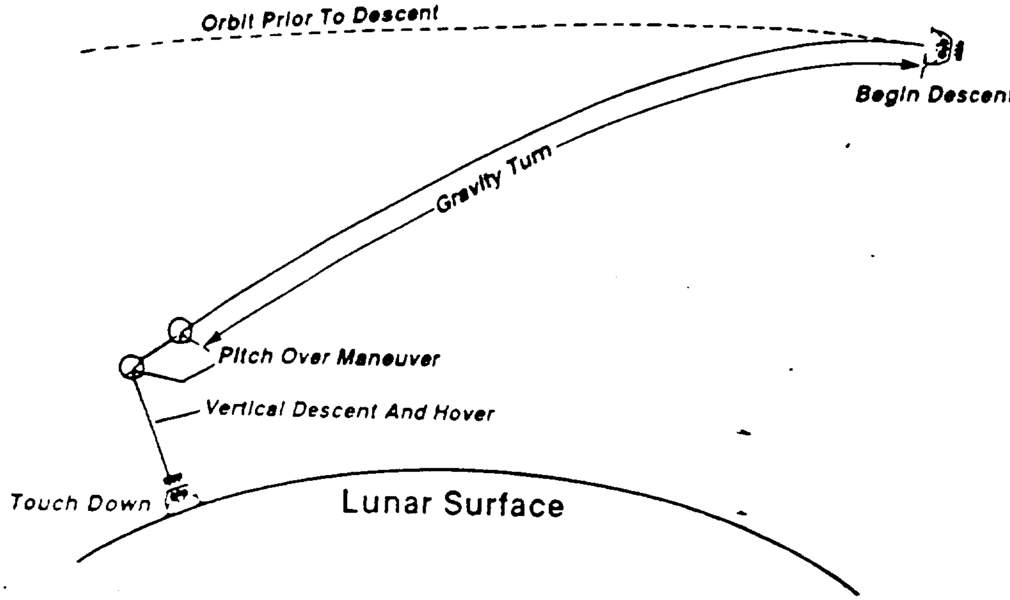
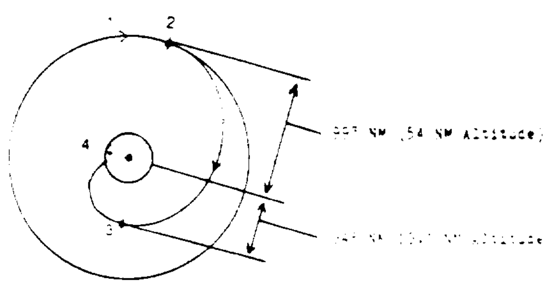


Figure 3.3: Gravity Turn Descent Trajectory



TOTAL DESCENT TRAJECTORY

PHASE	DELTA-V (Km/Sec)	DELTA-TIME
[illegible]	0.019	[illegible] Sec
[illegible]	[illegible]	56 Min. 57Sec
[illegible]	[illegible]	6 Min. 15 Sec
TOTAL	[illegible]	[illegible] Hr. 3.25 Min

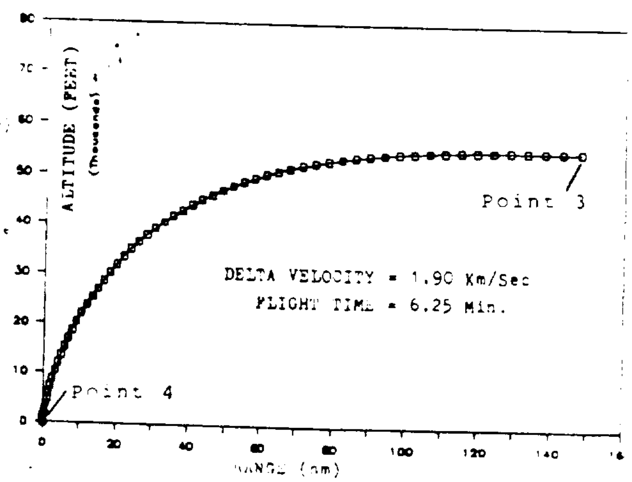


Figure 3.4: Scaled Descent Trajectory Using Solid Core Thermonuclear Propulsion

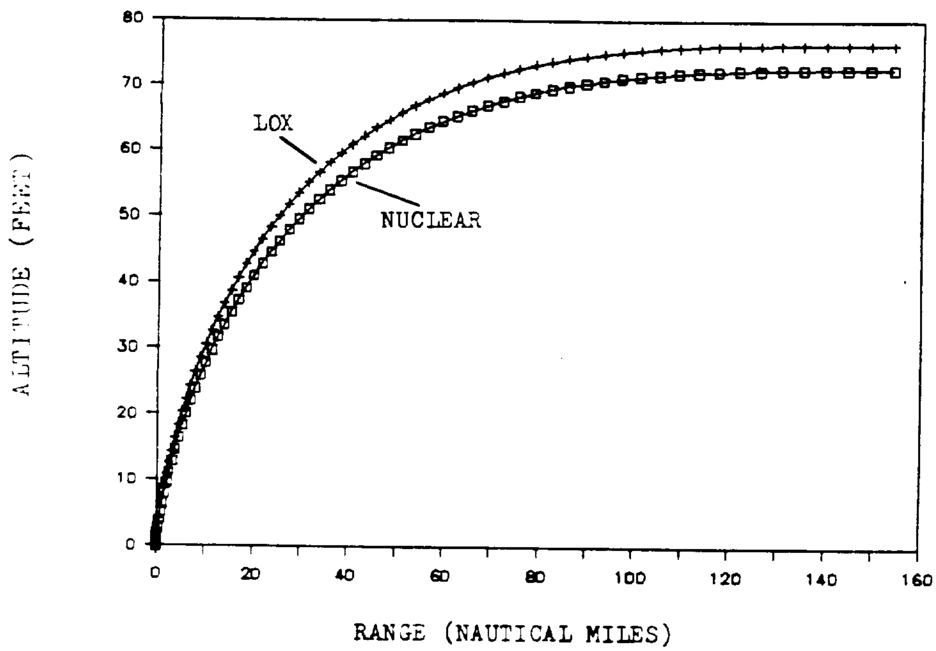


Figure 3.5: Ascent Trajectories for LOX and Nuclear Propulsion: Isp_{LOX} = 480 Sec, $Isp_{NUCLEAR}$ = 1200 Sec

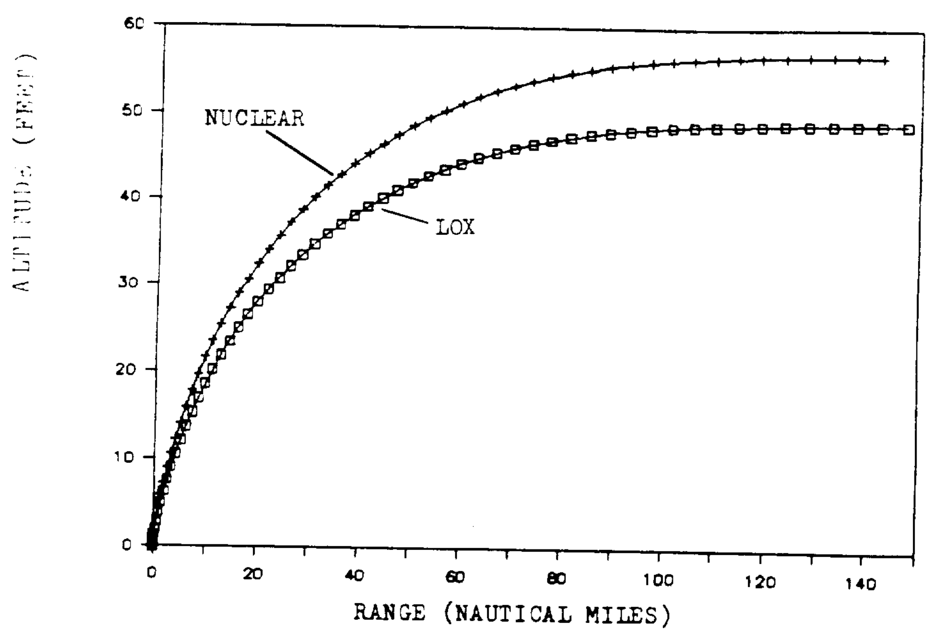
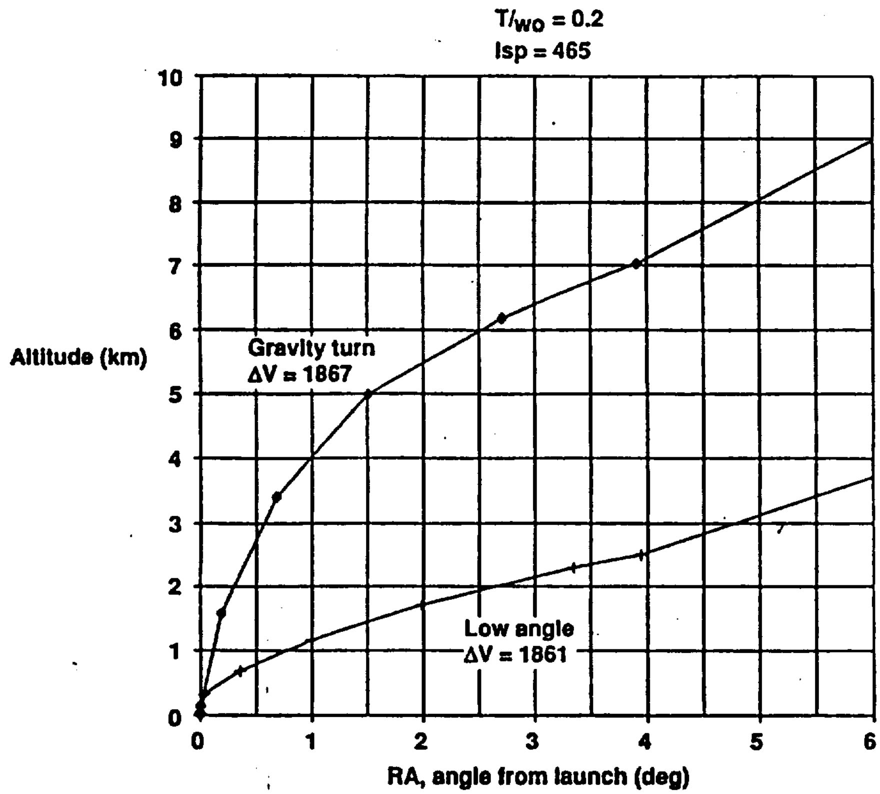


Figure 3.6: Descent Trajectories for LOX and Nuclear Propulsion: Isp_{LOX} = 480 Sec, $Isp_{NUCLEAR}$ = 1200 Sec

STV

MSFC - BOEING

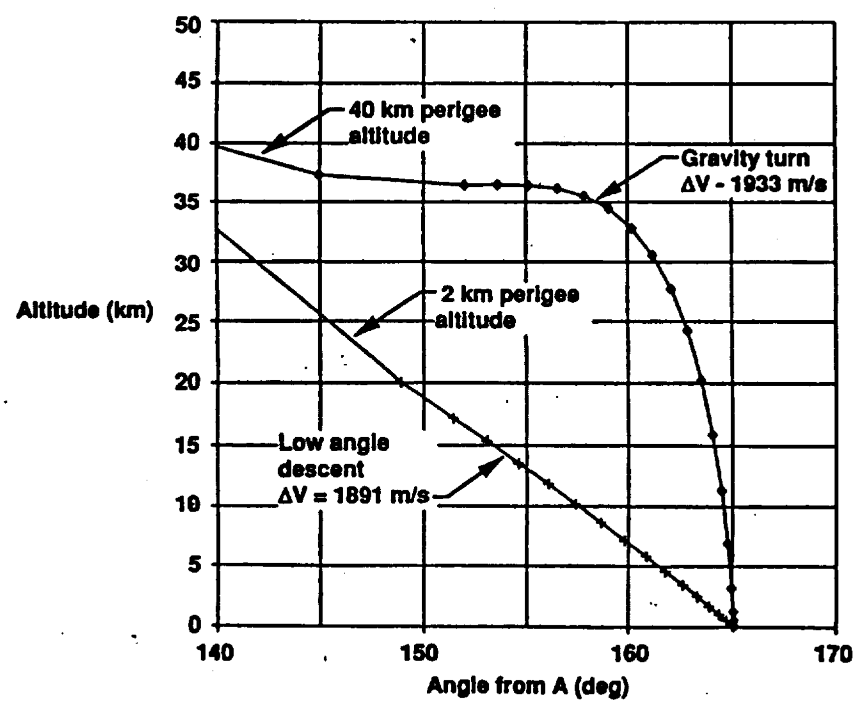
LOW ANGLE ASCENT TRAJECTORY CAN REDUCE ΔV



STV

MSFC - BOEING

DESCENT TRAJECTORY OPTIONS



- Start in 300 km orbit, T/W = 0.33 lb/lb
- Is = 465 sec, constant thrust
- Stop at 50m altitude, 5 m/s velocity

Figure 3.7: Boeing Gravity Turn and Low Angle Trajectories

3.2 Structures & Materials

The lander structure provides connectivity and integrity to all of the lander's systems. In addition to supporting the weight of the payload, the lander is required to house and support all of the necessary subsystems. The lander structure must also be able to withstand any torques or forces applied to it by any of the subsystems, particularly the propulsion system, the reaction control system, and the unloading mechanism. The basic design theory that was used in the structural design is the consolidation of most of the encountered forces into the same structural members to increase the overall efficiency of the structure. A complete set of dimensioned views of the lander are in Appendix A.

The main part of the lander structure is the central box, which carries all of the loads generated by the subsystems. As shown in figure 3.8, this box is a truss structure enclosed by honeycomb core panels. The truss structure is strong enough to support the loads generated by the subsystems and the lightweight panels protect the subsystems from solar radiation, dust, and micrometeorites.

As shown in Appendix A, the main fuel tanks are attached to the side of the lander's central box structure. In order to prevent damage from micrometeorites, the fuel tanks are encased in a 1 mm outer shell. The sizing of all the fuel tanks is outlined in Appendix D. It may be necessary to use cryogenic propellant to conduct main engine cool down. The propellant masses that are given in Appendix C and the tank sizes that are given in Appendix D take this into consideration. The propellant mass sizes are 5% larger than is required to perform the necessary delta-v's.

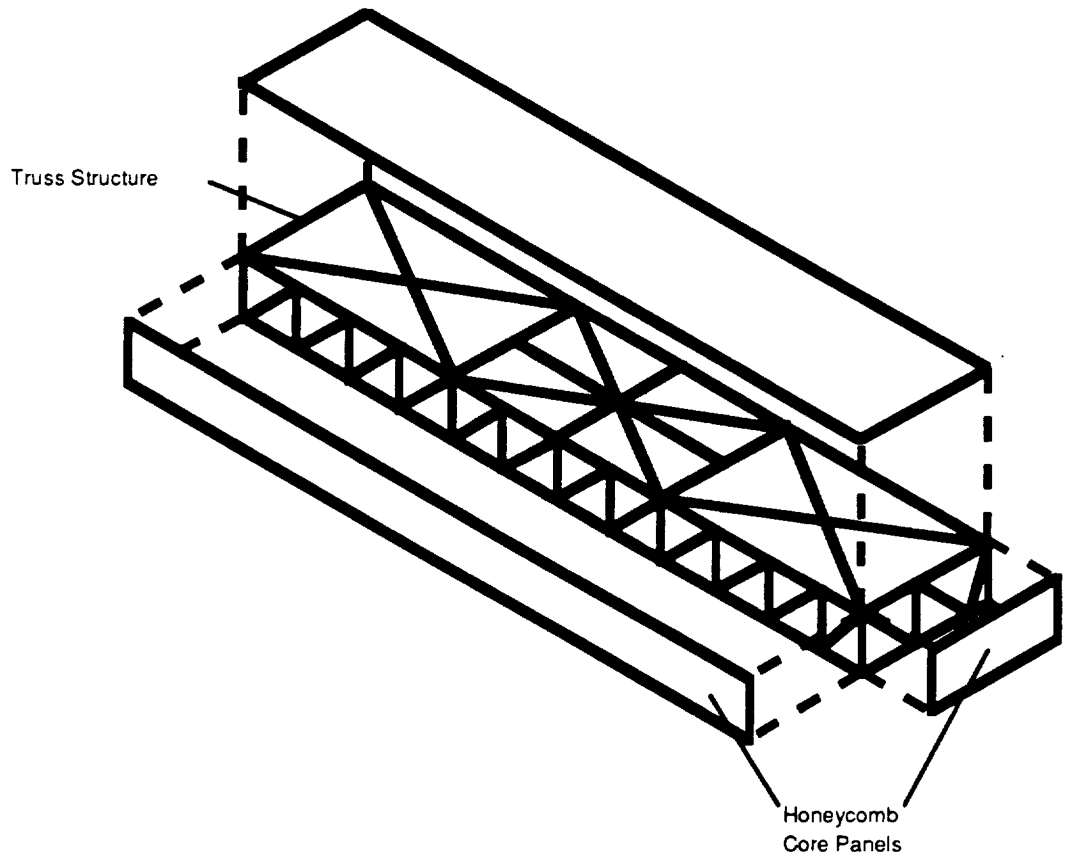


Figure 3.8 Lander Structure Central Box

The landing gear is composed of four struts that are planer trusses with Apollo style Lunar Module landing pads. A drawing of one set of the end struts is shown in figure 3.9. These struts are lightweight and strong enough to support the structure. The landing pads spread the lander's weight over a larger surface area, which reduces the contact pressure. Since these struts are not very flexible when loaded, a terrain adaptive system will be incorporated into the landing gear as depicted in figure 3.9. This system will allow the lander to remain level if the lander touches down on an incline of up to 8°. The landing gear was sized to enable the unloader to drive between the two end struts and to insure that the lander will not tip during the rotation sequence.

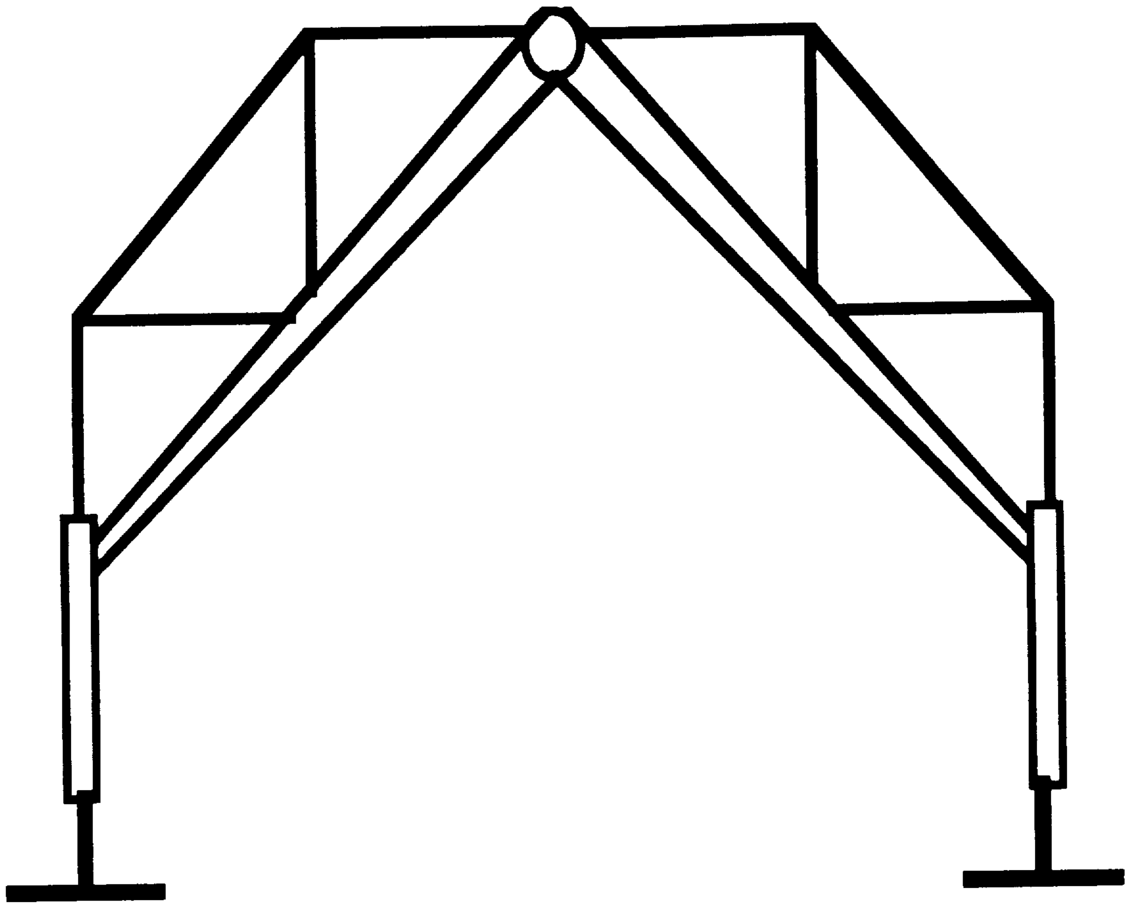


Figure 3.9 Landing Strut Design

The terrain adaptive system consists of a ground contact sensor and a brake/compression device on each strut which are electronically connected to the central computer. During the touchdown phase of the landing, the computer will unlock the compression device on each strut which will enable the landing pad portion of the strut to be compressed up to one meter. The lander will continue to descend until main engine cutoff is initiated by the contact of all four pads. If the terrain variation were greater than that which could be compensated for by the terrain adaptive system, the lander would continue to descend in a non-level attitude until all four pads made contact with the surface. This final scenario will be avoided because terrain of this type will interfere with the rotation phase of the mission.

In addition to being able to withstand the variety of loads the lander will be subjected to, the materials used in the construction of the lander will have to endure the extreme conditions of the lunar environment. Aluminum alloys are very good materials for spacecraft because they have many favorable properties. These properties include high stiffness-to-weight ratio, excellent workability, high ductility, high corrosion-resistance, insensitivity to radiation, non-magnetism, moderate cost, and availability in many forms. Typically, the only meaningful disadvantage of aluminum alloys is their low yield strength.

Aluminum-lithium alloys were also considered for use in the lander structure. They have been developed for high strength, weldability, and low weight. In addition to having the same favorable properties of typical aluminum alloys, aluminum-lithium alloys can have a high tensile strength (over 100 ksi), along with increased weldability and a weight reduction of up to 30%. Aluminum-lithium alloys also have a higher cryogenic strength than other aluminum alloys. Their high weldability and increased cryogenic strength makes them excellent choices for cryogenic fuel tanks.

Aluminum-lithium alloys are currently being produced by several manufacturers and can be used as sheet and structural members. Due to the availability and advantages of aluminum-lithium alloys, they will be the main material used in the lander's construction. In applications where the aluminum-lithium alloys do not possess the required strength, titanium will be used. It has a substantially greater yield strength and a higher stiffness to density ratio than aluminum alloys. The major disadvantage of titanium is that it is more difficult to machine than aluminum, making it more expensive. Ceramic materials will be employed for ultra high temperature applications, such as turbines or combustion chambers. Ceramics have an excellent ability to withstand high temperatures.

3.3 Guidance, Navigation, & Control

The purpose of the guidance, navigation, and control (GNC) system is to determine the linear and angular position, velocity, and acceleration of the lander, to compare that data with the desired state, and to make corrections when necessary. The desired state of the lander will be provided by the predetermined trajectory analysis for a specific mission and a study of the attitude requirements of the propulsion design. The phases of the mission in which GNC is essential include: descending from LLO to a selected sight on the lunar surface, hovering above the selected landing sight, ascending to LLO to rendezvous with an OTV, and docking procedures with the OTV.

A major assumption made in the design of the GNC system was that the lander would begin service before any operational structure was established on the lunar surface. Therefore no local remote control capabilities or transponders would be in place. This lead to the decision that the lander should be as automated as possible with remote control representing a backup, update, and/or emergency override system.

The GNC system will be made up of three components: Sensors, Control, and Computer. The sensors will provide the actual state of the lander at any given time. The control devices will be used to affect changes in attitude. The computer system will provide all the analysis on the data from the sensors and decide when to make use of the control devices based on the analysis. The order of the GNC system is shown in Figure 3.10.

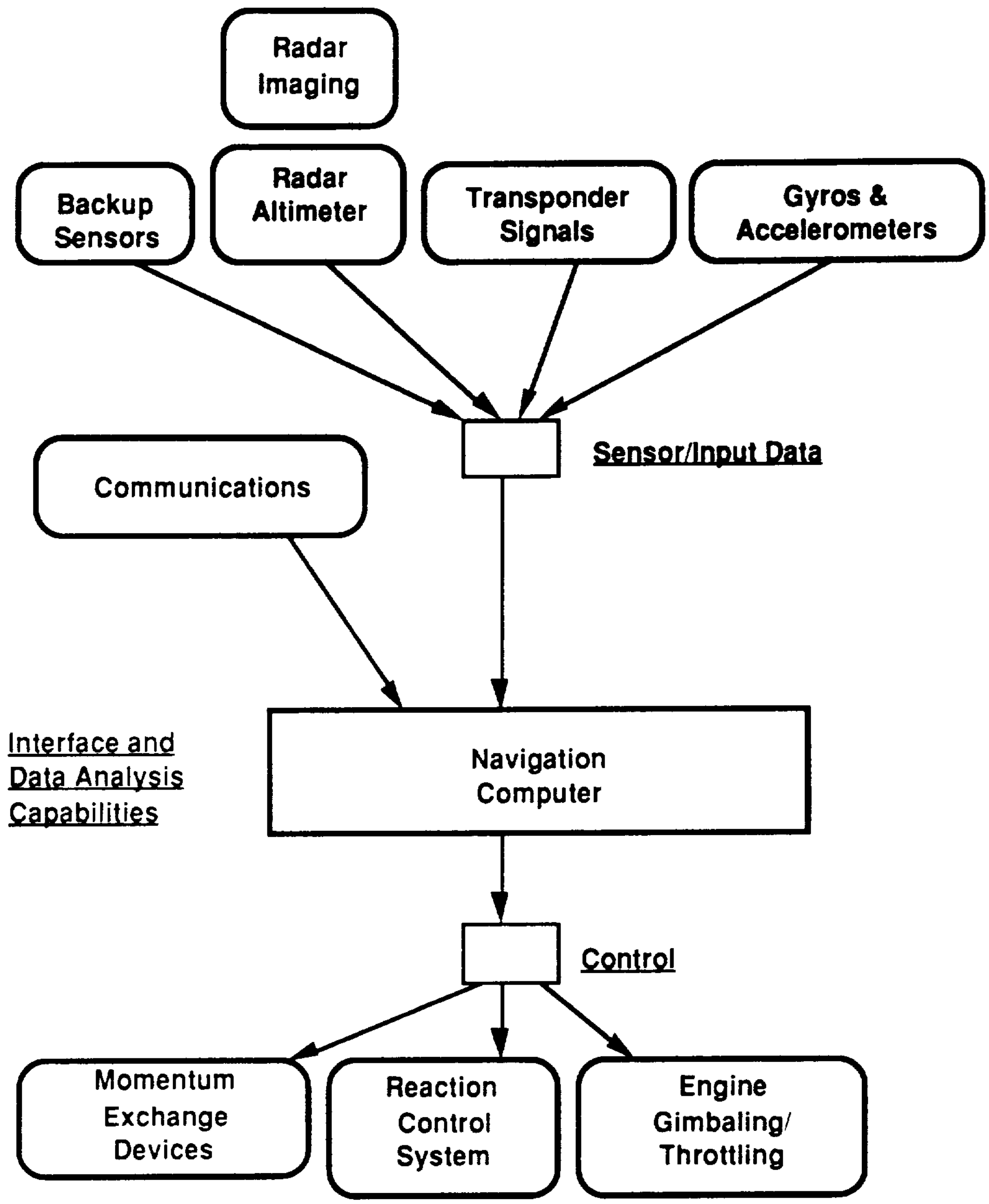


Figure 3.10 GNC Schematic

3.3.1 Sensors

The sensors used for guidance and navigation detect various aspects of the state of the lunar lander. There will be three levels of redundancy. During optimum operating conditions, several components of each level of redundancy will be used.

There will be more than one set of some sensors because their small size and weight do not present serious loading problems.

The primary sensor system will consist of a transponder system, a radar imaging/altimeter system, and a set of accelerometers and gyros. The transponder system will include an active seeker on the lander and a set of three (minimum) transponders at the landing sight. The transponder setup is shown in Figure 3.11.

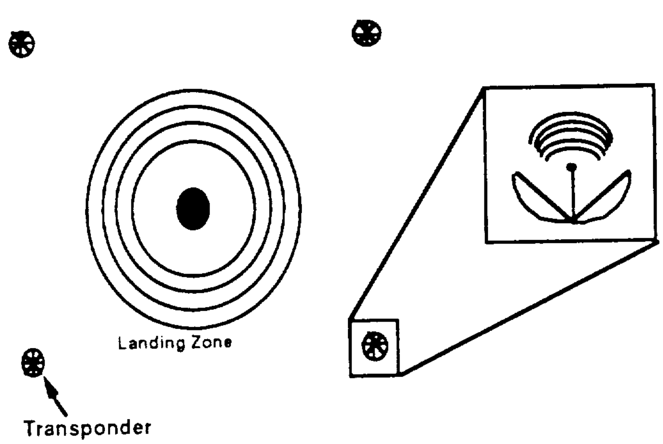


Figure 3.11 Transponders Distributed Around the Landing Sight.

The transponders are clam shaped so that when they are placed on the lunar surface, they will always open with the signal directed upward. This system will allow for precise orientation and location to be specified about the landing sight. It will also provide altitude information near the surface where the radar altimeter has difficulty. Otherwise, the radar altimeter will be used from orbit down to about 200 feet above the surface (a limit set by the physical limitations of radar technology). The radar will also be used to image the landing sight so that obstacles about the landing sight may be avoided by the lander and unloader. The accelerometers and rate and angle gyros will provide the remaining data needed for guidance.

For this system to function properly, a method of delivering the transponders to the landing sight will have to be devised. The current plan works under the assumption that the lander will be delivering many payloads to the same sights on the lunar surface. When landing at a new landing sight, the transponder system will be supplanted by the sensors from the backup system, along with a close proximity altitude detecting device. This device will be a small spring driven rod extending some length downwards from one of the legs of the lander. When the spring is compressed, the lander will know that it is a certain distance above the ground (a similar system was used on the Apollo missions). The unloader would then, after delivering the payload, distribute several transponders around the landing sight. This is a possible solution for delivering transponders to a new landing sight. Some other solutions might include pattern recognition software taking data from a camera for initial landing, or designing a probe to deliver the transponders well before the mission takes place.

The backup GNC system would include a set of two or more star/planet/Sun sensors, a radar imaging/altimeter system, a backup computer placed on the opposite side of the lander, and another set of accelerometers and rate gyros. The star/planet/Sun sensors will suffer greatly from occultation because of the Moon's proximity. Targeting several independent stellar bodies, preferably those very high or low in the orbital plane of the Earth-Moon system, will help to alleviate this problem. The same radar imaging/altimeter system will be used with both the primary system and the backups because of its large mass and volume.

The emergency system will be run from Earth based updates through the communications system, with backup from all the remaining functioning and trustworthy sensors. The Earth based updates could come from any applicable location determination systems established at the time, such as Extended GPS, a Lunar Positioning System or Earth based sightings. While the other sets of sensors are working, the information from these external sources would provide updates to those sensors prone to drifting. It will be very difficult to navigate using the third system, but this condition represents a level of critical failure of the other sensors and would only be used for emergencies.

3.3.2 Control

The lander will utilize three techniques for control: momentum exchange devices, small directional thrusters, and gimbaled/throttled main engines. While some redundancy exists using all these systems, the optimum operating conditions will use each technique where best suited.

The momentum exchange devices, a triaxial set of fly-wheels, will be used to make small precise changes in attitude, where time is not a constraint. Since these devices are driven by electrical power, they save on fuel usage. They are also useful during rendezvous, when it is not desirable to have exhaust against the OTV.

The small directional LH_2/LO_2 thrusters, are suited for making large, quick changes in attitude, as well as providing some translational capabilities. They will also be used to periodically bleed the excess momentum built up in the fly-wheels. Four direction thrusters will be mounted on either end of the body near the legs, with two direction thrusters mounted on the fuel tanks. These clusters are shown in Figure 3.12. The optimum mounting positions and thrust levels of these thrusters must be studied further.

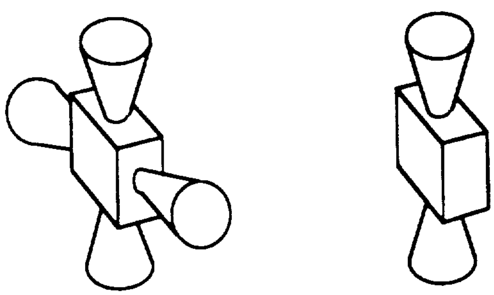


Figure 3.12 Four direction cluster and Two direction cluster

The gimbaling/throttling capabilities of the main engines will be used for stability maneuvers, cases of engine out, and to correct for center of mass changes which might occur with oddly shaped payloads. The maximum gimbal angle possible will be 10°. The throttling capabilities for the three engine configuration will range from 34% to 100% of maximum thrust, with the nominal operating condition at 67%.

The navigation computer will determine, based the differences between the actual and desired state of the lander, which set of systems will be used. This may require having a set of predetermined responses for each main type of attitude adjustment which could be encountered. Another possibility would be to allow the navigational computer to determine the best response on a case by case basis. Advanced artificial intelligence would be required for the latter.

3.3.3 Onboard Navigation Computer

The navigation computer will be responsible for monitoring the output and status of each of the sensors, monitoring the status of and providing input for each of the control devices, and providing an interface between the two. It will also have the ability to accept input from the communications system. Another primary use of the computer will be to take data from the imaging radar to get a picture of the landing sight. With this data and some pattern recognition/obstacle avoidance software, the choice of a safe precise landing sight can be made autonomously.

Choice of some of the more advanced systems mentioned above would depend on advanced artificial intelligence and pattern recognition software and hardware development. The rapid pace of computer and software development has shown that a fully automated and fault tolerant system of this type will not only be possible, but weigh little and consume relatively little power.

3.3.4 Unloader Navigation & Control

The control of the unloader will be primarily automated, with a remote control system as backup. The unloader's GNC system will be based on that of the lander, in that optical sensors will inform the navigational computer onboard the unloader of obstacles. The computer will then instruct the wheel motors to make the required adjustments. Again, the automated system will require artificial intelligence and/or obstacle avoidance software to make the decisions necessary for guidance.

The remote control system will require short range communication equipment to relay data to the Earth using the lander as a communications link. The detection of obstacles for the remote control system will be supplied by a video camera. The camera or cameras would also supply much of the public relations pictures which are always important to NASA projects.

3.4 Communications

The communication systems provide three basic functions: telemetry, command, and tracking. The telemetry function is the transmission of information from the spacecraft to earth. These transmissions consist of the status of the spacecraft instruments and systems. The information encoded in these transmissions needs to be of moderate quality.

The command function is the transmission of information from the ground (earth) to the spacecraft. These transmissions are comprised of information needed to control the spacecraft functions and to direct it to take specified actions, such as changing its flight path. The information encoded in the command function transmissions must be of high quality.

The tracking function is the transmission of information, used for trajectory monitoring and navigation, from the spacecraft to the ground. This consists of information such as spacecraft position and velocity, radio propagation medium, and properties of the solar system. This information needs to be extremely accurate.

3.4.1 Lander Communications

The lander's communication system will have the following characteristics, which are based on the Apollo spacecraft:

Input power: 130 W
Antenna diameter: .61 m
Transmitter power: 20 W
Transmitter aging factor: 1.5

The lander will be fully automated with the option for command override by mission control. The advantage of automation is that the response time will be greatly increased. The response time is very important in this type of mission. With the automated system, the navigational information can be input into the computer, and the flight path or attitude changes can be made immediately. If the lander were to be remotely controlled, the communication delays between the lander and earth would be excessive.

The main disadvantage of an automated lander is that a more complicated computer is required to run the lander systems. The rapid pace of computer and software development has shown that a fully automated and fault tolerant system, which can accomplish the lander's computational requirements, will not only be operable when needed, but will weigh little and consume relatively little power.

3.4.2 Lander/Earth Communications

The bands that are commonly used for explorer spacecraft are S-band and X-band (8.4 GHz), because at these frequencies the Earth's atmosphere is transparent. S-band (2.3 GHz) will be used for direct communications between earth and the lander.

The disadvantage to using S-band is that it will have a smaller gain than X-band. The advantage of S-band is that it will have a wider bandwidth, which will greatly reduce the pointing accuracy requirements for the antenna.

The antenna will be a parabolic dish type antenna with steering capabilities similar to that on the Apollo spacecraft. The Apollo pointing system is more than sufficient for the communication link needs of the lander.

It is suggested that a communications satellite for Earth link capabilities be placed in the L2 Lagrangian point. This point is located

60,000 km from the far side of the moon. The satellite should be placed in a halo orbit as suggested by R.W. Farquhar (NASA Goddard Space Flight Center), D.W. Dunham and S.C. Hsu (Computer Sciences Corporation). The satellite would allow transmissions to be made when the spacecraft is on the far side of the moon. This satellite could later be used by the Lunar base once it is operational. The communications between the satellite and the lander would be over Ku-band. Figure 3.13 illustrates the L2 communications satellite link concept.

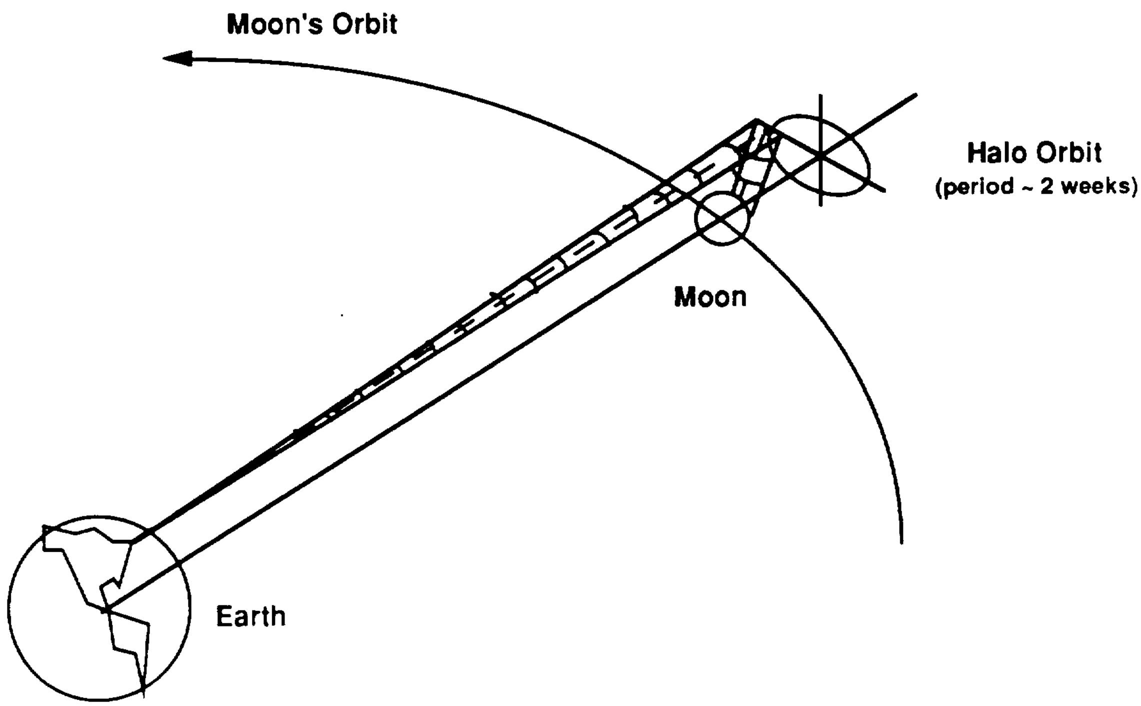


Figure 3.13 L2 Communications Satellite Link Concept

3.4.3 Lander/OTV Communications

Communications between the OTV and the lander will be necessary for docking procedures. These communications will be done using VHF. The VHF system will also be used for communications between the lander and the lunar base (after the base is operational). The advantage of VHF is that it has a wide bandwidth. The disadvantage is that it does not have a large range. The is not a problem since the OTV will be in LLO and the signal will only have to travel a distance of approximately 100 km.

3.4.4 Lander/Unloader Communications

The communications between the lander and the unloader will be over UHF. UHF receiver-transmitters, similar to those used on modern day aircraft, will be placed on both the lander and the unloader and will exchange the necessary data to operate the unloader. These receiver-transmitters are small, lightweight, require little power and need only a small antenna.

3.5 Power Systems and Thermal Control

Since solid core thermal nuclear motors where chosen as the lander's main propulsion system, the lander's primary power system will also be used as one of the thermal control systems. Conceptual details of how this system will work are provided in the following sections.

3.5.1 Lander Primary Power Systems

Four types of power systems were considered for primary power for the lander:

- Nuclear
- Fuel Cells
- Batteries
- Solar Power

The design of the primary power subsystem was driven by the choice of a propulsion system for the lander. Since a solid core thermal nuclear propulsion system was chosen, a power system which utilizes the reactor's energy was the favorable system.

The nuclear motors generate significant heat during engine firing. The energy for the engine-off phases (periods of time spent on the surface or docked with the OTV) of each mission will come from the excess heat generated by the reactor. The power generation loop shown in Figure 3.14 will be used to convert the surplus heat into electrical energy while simultaneously cooling the reactor and heating the batteries and other subsystems. Figure 3.14 is based on a closed Brayton cycle, but the choice of a specific conversion cycle for the power generation loop is dependent on the heat loads that will be dissipated. The heat loads will depend on the core type that will eventually be determined to be superior by NASA for the lander use. 500 kg was used as the mass estimate for this system. This mass estimate is base on the power generation needs of the lander. The working fluid of this system will most likely be liquid He or liquid Xe. It may be determined that it will also be necessary to flow cryogenic propellant over the reactor core of each engine during the operation of the primary power system in order to facilitate main engine cool down.

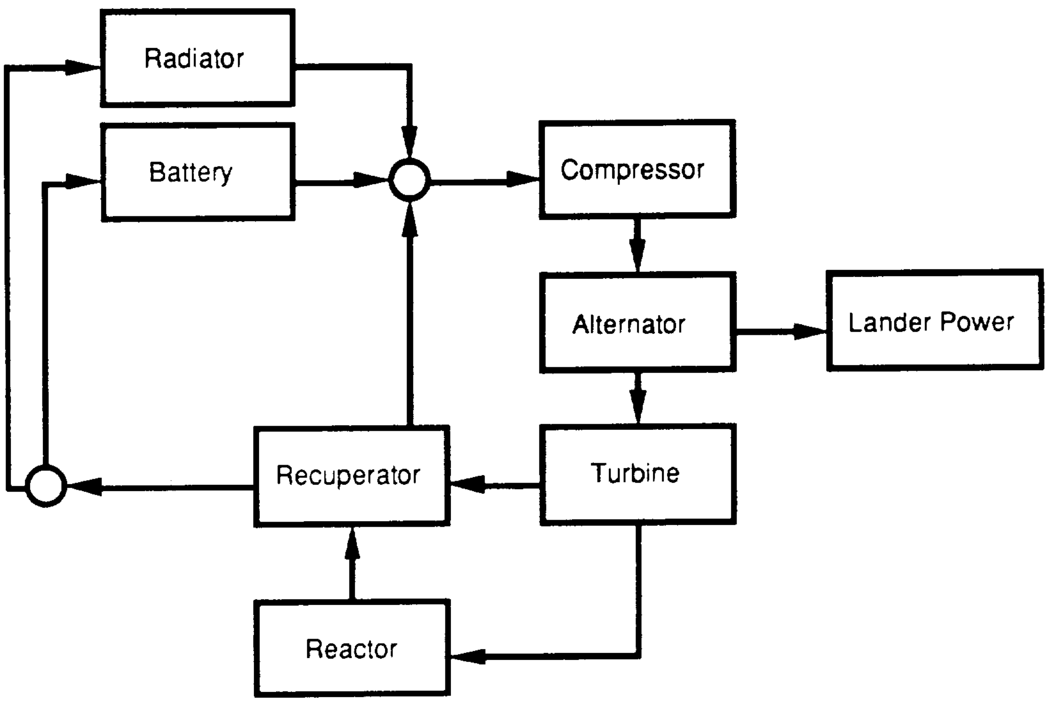


Figure 3.14 Lander Power Conversion Loop

3.5.2 Lander Secondary Power Systems

The lander's secondary power system will be used during engine-on phases of each mission (periods of time spent in flight). The secondary system will be recharged by the primary system during engine off phases of each mission. This arrangement was chosen for two reasons. The first reason is that the operation of the primary power system during lander flight time would interfere with the landers inertial stability. The second reason is that the primary power system will be used for main engine cool down and peak power consumption phases of the missions. The peak power consumption phases consist of the lander rotation sequence, unloader lowering and retrieval phases, cryogenic refrigeration system operation, and secondary power system recharging.

Four types of power systems were considered for the lander's secondary power system:

- Nuclear (RTG's)
- Fuel Cells
- Batteries
- Solar Power

RTG's were rejected for their relatively low power output and inability to be recharged by the primary power system. Fuel cells were rejected due to the problems associated with regeneration. Solar arrays were not selected as the secondary power system because they are not rechargeable.

Rechargeable batteries were chosen as the secondary power source for the ascent/descent phase of the lander's mission. Batteries have no moving parts, which contributes to their high reliability. Batteries also require less energy for recharge than other systems because of their relatively high charge/discharge efficiencies (about 80% for Na-S batteries). The only significant environmental constraint on batteries is the operating temperature, which can easily be accommodated for by using excess heat from the nuclear reactor.

The following battery types were considered: $Ni-H_2$, Ag-Zn, Ag-Cd, $Li-FeS_2$, and Na-S. The Na-S (sodium-sulfur) batteries were chosen because of their superior characteristics, which are summarized in Table 3.1.

Table 3.1 Na-S Battery Specifications

Mass	100 kg
Specific energy	0.11 kWh/kg
Energy density	0.15 kWh/l
Charge efficiency	80%
Depth of discharge	80%
Ideal operating T.	300+°C
Nominal cycle life	1000
No. freeze/thaw cycles	20

The batteries on the lander will provide 11 kWh of power and have a total mass of 100 kg. This power supply exceeds the peak power demands of the lander, which occur during main engine startup. Main engine startup requires approximately

1 kW of power. The size of the batteries will give enough power to run the lander's communication systems and the cryogenic refrigeration system once the primary power system is no longer able to operate.

3.5.3 Unloader Power Systems

Along with RTG's and batteries, power for the unloader beamed from the lander was considered as a possible power source, but this method of power supply was not chosen due to its relative lack of technological readiness.

The primary power system that will be used by the unloader are rechargeable Na-S batteries. These batteries will also be recharged by the lander while the unloader is still attached to the lander. The Na-S batteries, identical to those on the lander, will have a mass of 100 kg and deliver 11kWh of power. This power will allow the unloader to travel near 2.5 km/hour with the payload and to deploy the payload once it has reached its desired location.

The unloader's secondary power system will be comprised of solar arrays. Even though solar cells would not make a good primary power source for the lander or unloader, they are an excellent secondary power source for the unloader. Solar cells are quite modular and have few moving parts. They are also a proven technology. Although the arrays on the unloader will undergo deterioration during extended idle periods, it has been assumed that the arrays will degrade 30% over 15 years before requiring replacement. GaAs/Ge cells were chosen primarily due to smaller size of the required array, when compared to Si cells.

The unloader's solar arrays will be used for two purposes. When the unloader is in the active mode (delivering a payload), the solar arrays will be used to keep the batteries warm. When the unloader is in the passive mode and is away from the lander, the solar arrays will also be used to recharge the Na-S batteries.

The nuclear reactor propulsion system provides sufficient energy to support both the lander and the unloader power subsystems. Since the purpose of the power subsystem is to make use of the surplus energy generated by the reactor as well as cool the reactor, the above design incorporates power systems which make the best use of this energy during all phases of the mission.

3.5.4 Thermal Control

There are several areas of the lunar lander that will require thermal control and protection. Some of those areas include cryogenic fuel tanks, navigation sensors, communication equipment, and computer systems. In addition to the method used by the power generation loop to perform main engine cool down, battery heating, and heat rejection other types of thermal control will be employed on the lunar lander.

The main thermal control system that will be employed is a cryogenic refrigeration system using liquid He or Xe as the working fluid. This system will be powered by the primary power generation system that uses the excess heat from the main engines. Heat exchangers and radiators will be used in the cryogenic refrigeration system and the power generation systems. The heat exchangers will be used to keep the Na-S batteries at their operational temperature (above 300°C), and the propellant in a cryogenic state. The radiators will be used to dump excess heat to the shaded environment.

Radiator sizing has not been completed at this time. Once the best core design for the thermal nuclear motors is determined by NASA, it will be necessary to perform a detailed heat load analysis on the lander systems. This analysis should take into consideration variables such as Earth and Sun view factors, shaded area temperatures, bright side temperatures, and each particular mission profile. A mass of 500 kg is used as an estimate for the size of the cryogenic refrigeration system. This mass will be governed by the heat loads of the lander and the efficiency of the secondary thermal control systems.

The second thermal control method that will be employed is the use of 2 1/2" thick multi-layer insulation on all of the cryogenic fuel tanks. Insulation will reduce the work load of the cryogenic refrigeration system that will be used to prevent propellant boil off. Insulation will also be used on other areas of the lander, such as on the batteries and the GNC instrumentation.

The third device that will be used is a thermal/radiation protection umbrella. Two umbrellas will deployed after the second rotation sequence is performed and the lander is in the upright position. The umbrellas will be deployed from the lander's leg sections. The umbrellas will also reduce the work load of the cryogenic refrigeration system.

4.0 Management Structure

The management structure of RS Landers, shown in Figure 4.1, was designed with speed of communications in mind. Progress and problems were reported directly to the C.E.O. and/or the Technical Supervisor. Although, each group member was responsible for at least one area of the design, all members participated in both the conceptual and the technical design of the lunar lander. Cooperation between the team members was good because they chose to be lead engineers for the areas/subsystems they were most interested in. In Figure 4.1, the lead engineer of each group is italicized.

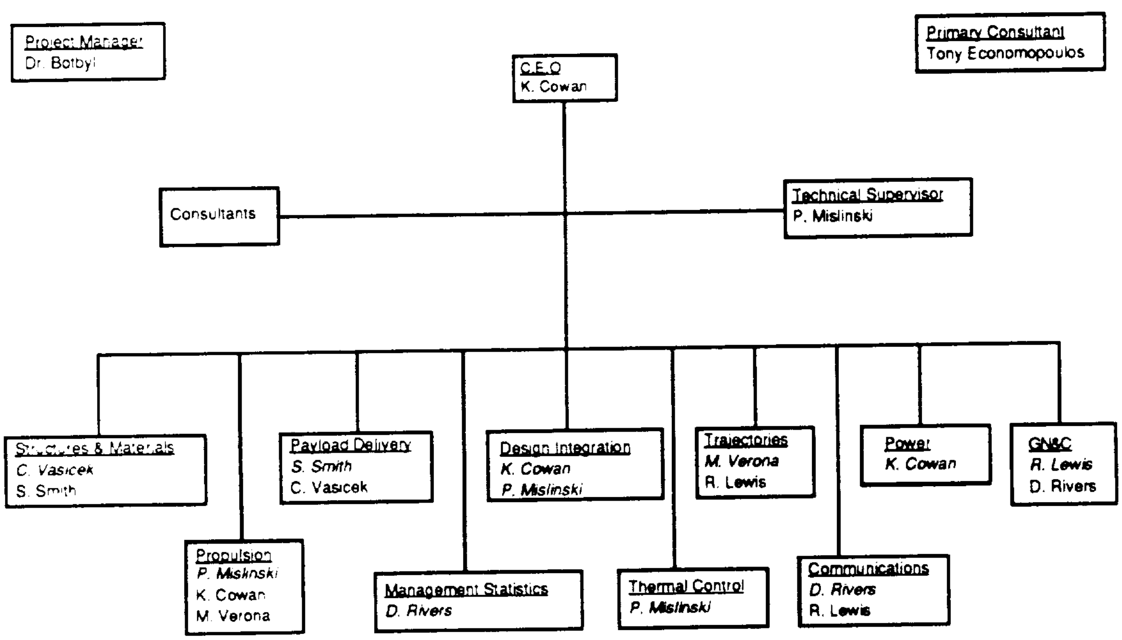


Figure 4.1 RS Landers Management Structure

The integration of the final design was supervised by the C.E.O. and the Technical Supervisor, but the design is the culmination of input from all group members. The final design for the lander was chosen using a decision matrix. Appendix A contains a detailed description of the design process and the decision matrix used. Figure 4.2 shows the timeline by which major deadlines were met. All major deadlines were met on time.

Figure 4.3 shows the task flow chart for meeting a more detailed set of project milestones. Note the order in which the tasks were met and note also that integration occurred after the options for each set of tasks were narrowed.

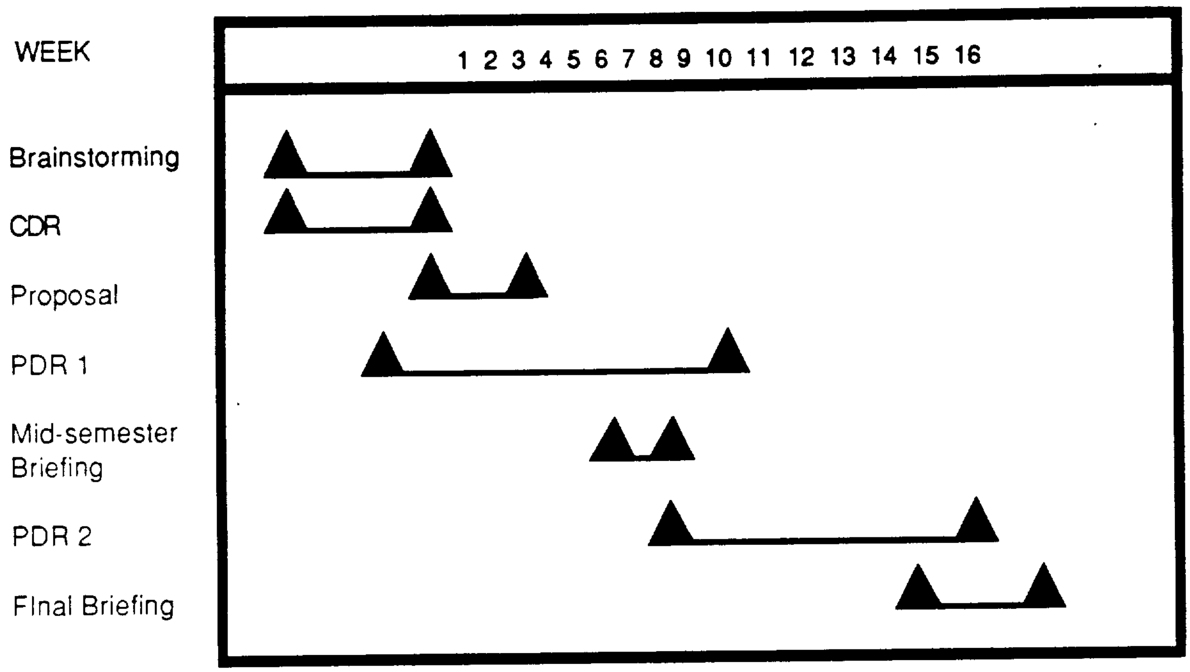


Figure 4.2 RS Landers Project Timeline

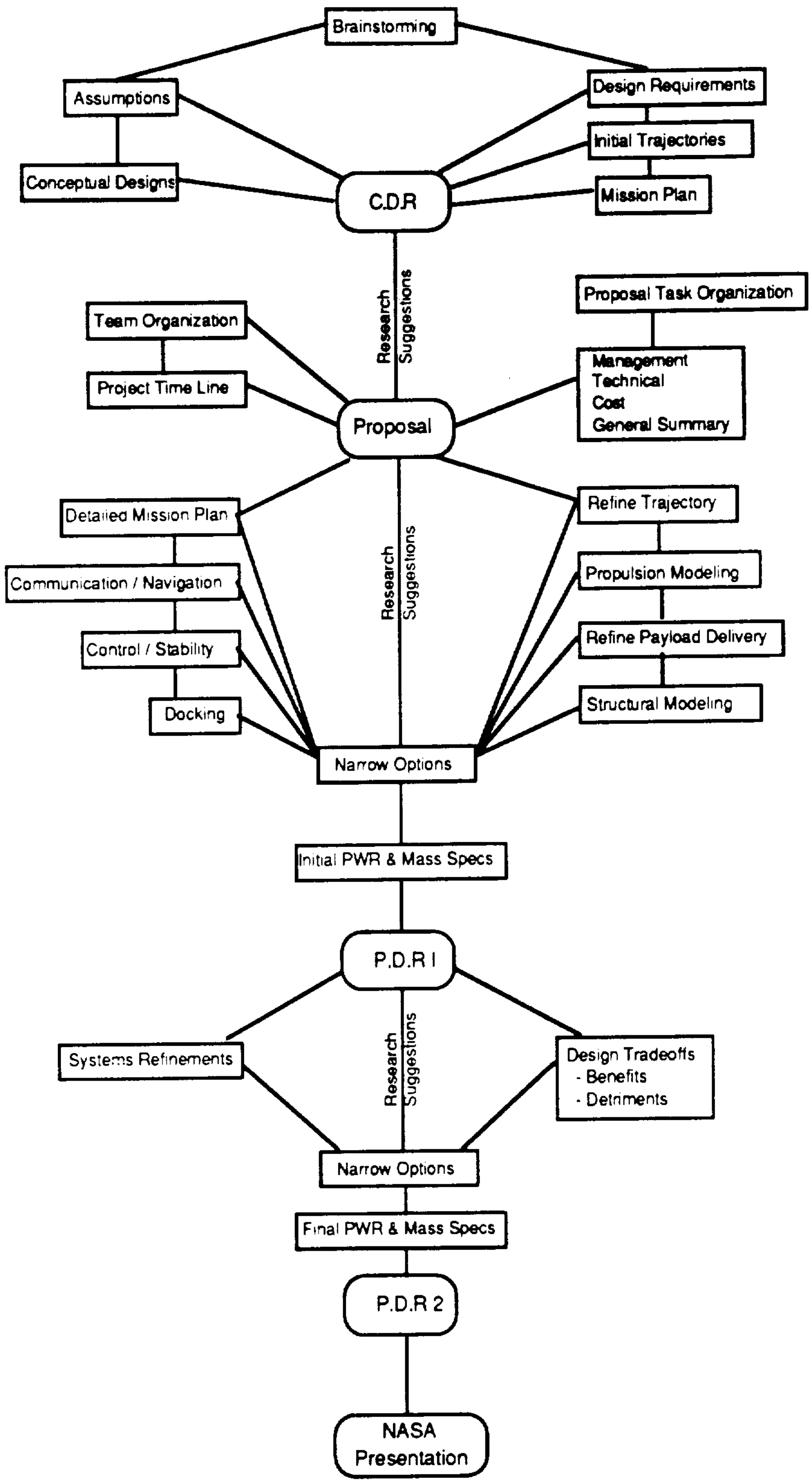


Figure 4.3 Task Flow Chart

5.0 Project Costs

The personnel costs for the first sixteen weeks are shown in Figure 5.1. Figure 5.1 shows that the personnel costs were very near the projected cost, which was $23,328.00 for the sixteen week period. The actual cost for that time period was $23,603.00. The actual costs were 101.2% of the projected personnel cost.

The actual total cost for the sixteen week period is shown in the Figure 5.2. The actual total cost is also near the projected total cost. The projected cost for the sixteen weeks was $27,922.84, while the actual cost was $27,457.20. This is 98.3% of the projected cost.

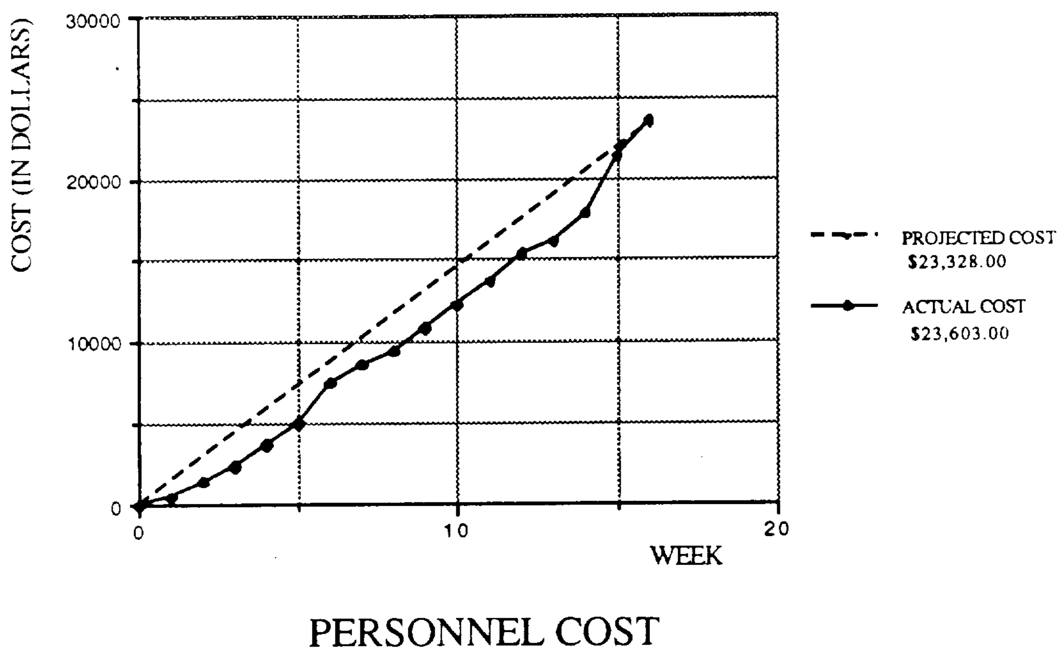


Figure 5.1 Personnel Costs, Projected and Actual

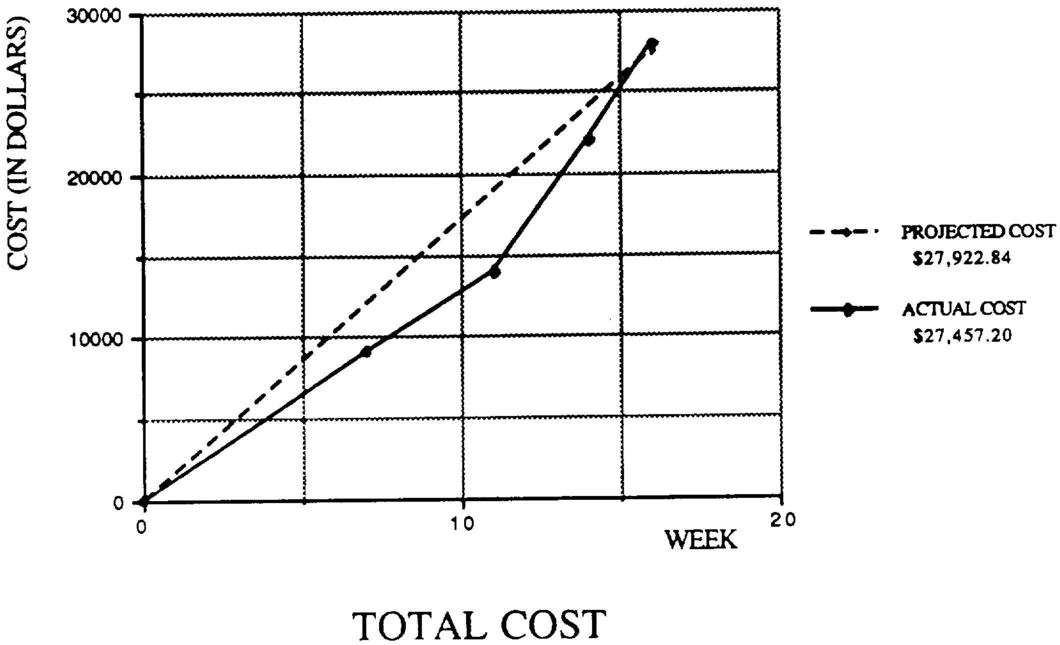


Figure 5.2 Total Costs, Projected and Actual

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Appendix A - Design Process

Appendix A: Design Process

This appendix presents the process by which the final design of the lunar lander was selected. The process began with initial brainstorming sessions, out of which came a myriad of contraptions for conveying a payload to the lunar surface. Some of these systems were feasible while some were completely outrageous. More brainstorming followed in order to narrow this tremendous spectrum of vehicles into a handful of feasible landers. Five preliminary lander concepts remained after the dust had settled. This was the first generation of self-unloading, unmanned, reusable lunar landers.

A.1 The First Generation

The first generation of landers were developed with primarily one goal in mind, the payload location. Hence, the emphasis in the drawings and descriptions that follow is on payload location.

Conceptual Design #1

As Figure A.1 indicates, two payloads hang on either side of the lander's legs. The hanging payloads cause significant structural problems. To ensure stability in case the payloads have different masses, the center of mass of the lander can be kept in the center of the lander by moving the payloads in or out on the payload support arms. The payloads are unloaded using a winch system.

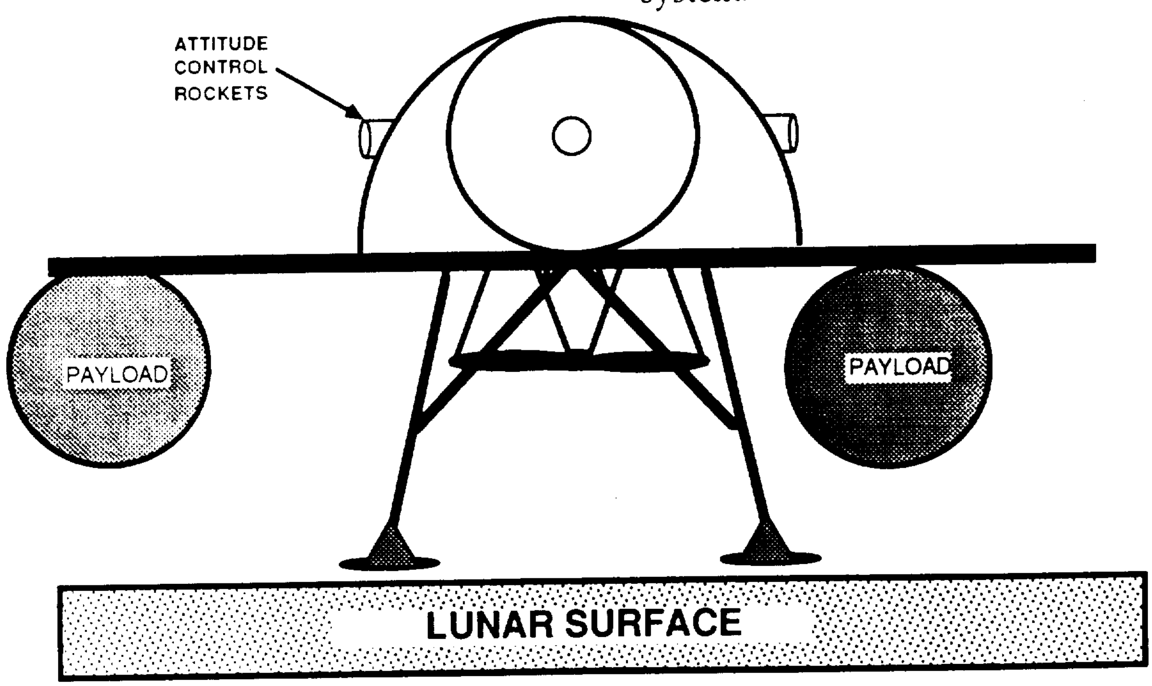


Figure A.1 Conceptual Design #1

Conceptual Design #2

Figure A.2 shows that the second conceptual design carries one payload on top of the lander. The payload is deployed using a track system and a ramp. After the payload reaches the top of the ramp, it then slides or rolls down the ramp to the lunar surface.

Although this lander allows the payload's mass to be located near the center of mass, the unloading procedure will not ensure correct orientation of the payload or a predictable location of the payload relative to the lander, due to unforeseen surface debris.

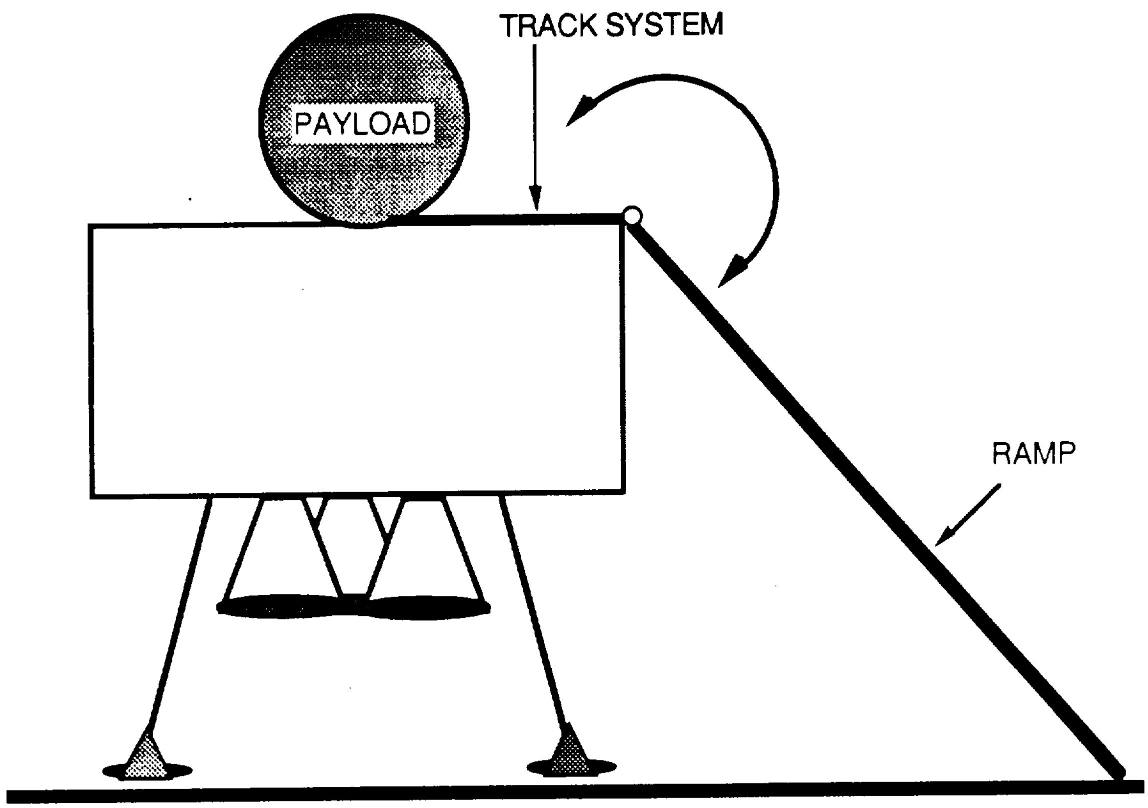


Figure A.2 Conceptual Design #2

Conceptual Design #3

This design can carry two payloads, which are unloaded by rotating the mounting structure 180° and lowering the payload to the surface using a winch system. This design has structural and stability problems.

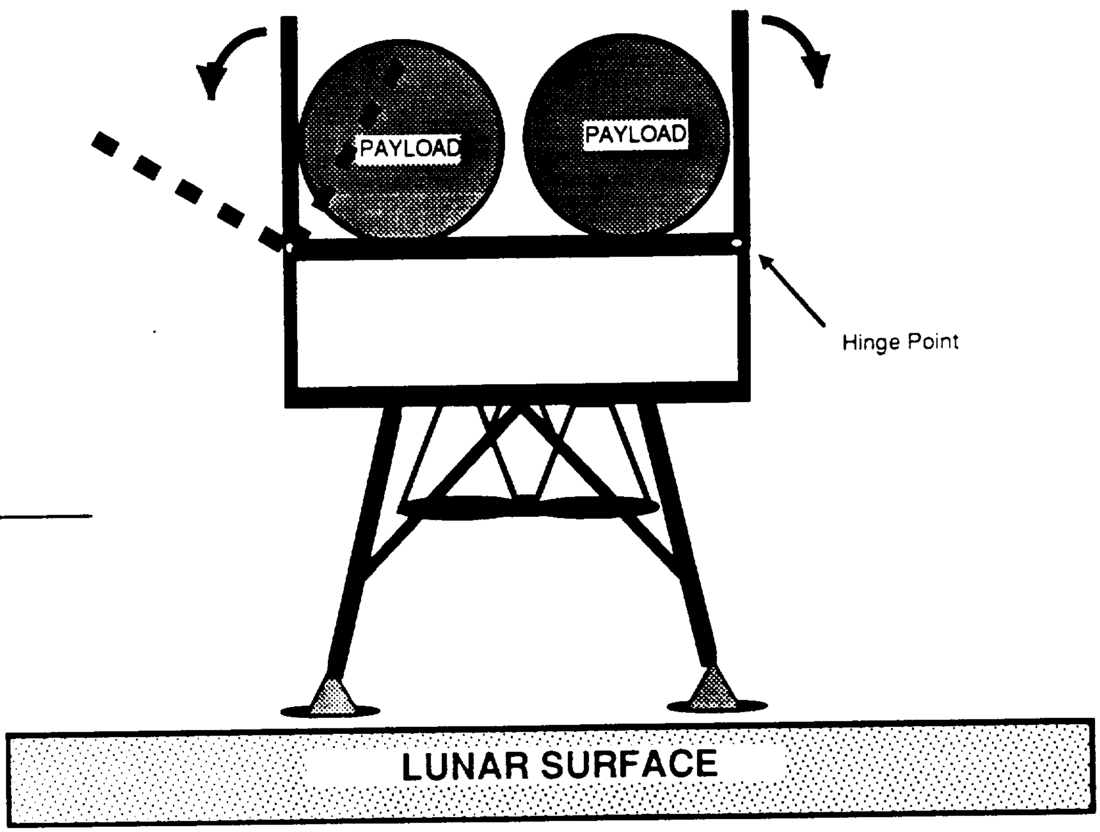


Figure A.3 Conceptual Design #3

Conceptual Design #4

Figure A.4 depicts the fourth conceptual design, which also carries two payloads. This design does not allow for differing payload masses and has, therefore, stability problems. The unloading mechanism uses a forklift device to lower the payload to the lunar surface. This forklift device adds complexity to the lander and is not compatible with the lunar environment.

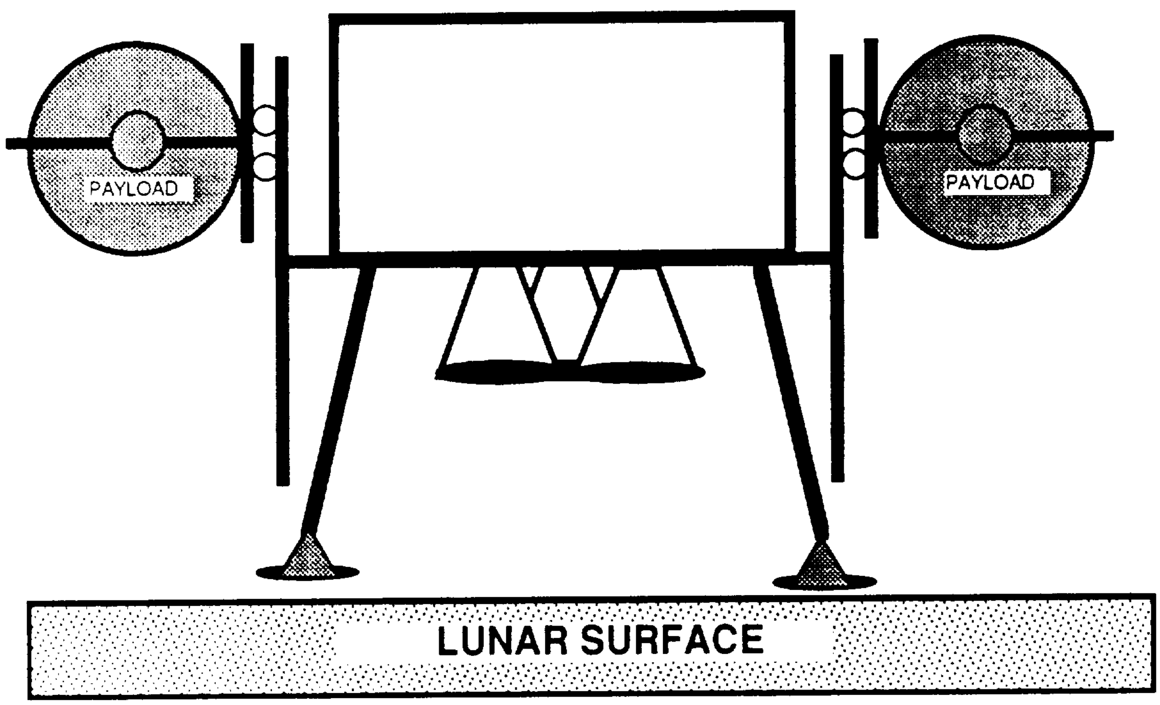


Figure A.4 Conceptual Design #4

Conceptual Design #5

The fifth conceptual design depicted in Figure A.5 is called La Rotisserie. In this design the major portion of the lander assembly rotates 180° and the lowers the payload with a winch system. The payload is centered on the lander, but the rotation sequence requires redundancy in the motors which turn the platform.

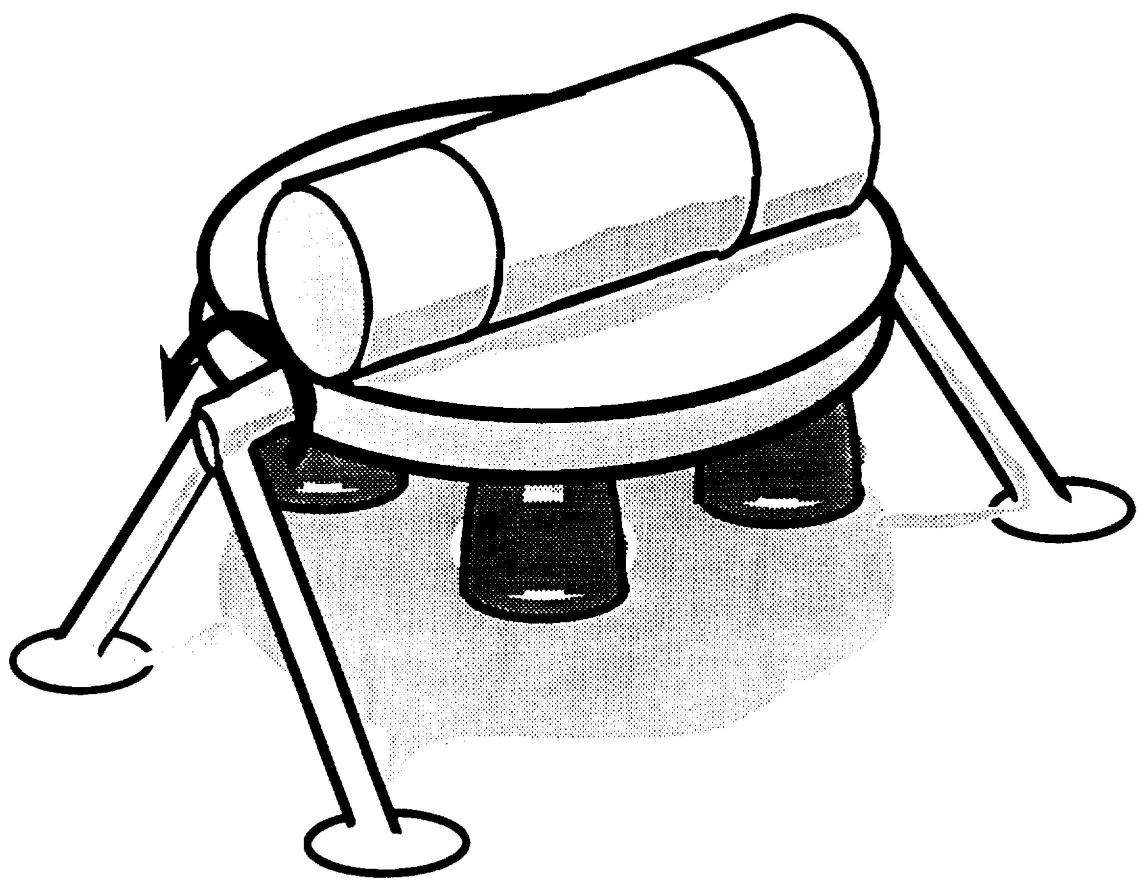


Figure A.5 Conceptual Design #5

A.2 The Second Generation

The next step in the design process involved evaluating the first generation designs according to the following criteria:

- Stability (center of mass location)
- Method of unloading
- Simplicity (mechanical simplicity)
- Size/Mass
- OTV interface

Two vehicles remained after applying the above criteria. The first vehicle was called the Ramp Lander, which evolved primarily from Conceptual Design #2. The second vehicle was an improved version of Conceptual Design #5. It remained La Rotisserie.

It was determined by an Eagle Engineering Lunar Base Launch and Landing Facility Conceptual Design study that it would be necessary to land at least one to two kilometers away from a lunar base in order to avoid damaging the base with the rocket plume debris. This study made it evident that a payload unloader which was capable of traversing the distance between the landing site and the base would be required for any lunar lander design. As a result, both of the landers that were still under consideration were improved upon and given a payload unloader.

The Ramp Lander

The Ramp Lander design is shown in Figures A.6 and A.7. The ramp will be a rigid part of the landing structure. This design requires the payload to include a Payload Support Structure (PSS) in its packaging. The PSS is used during the unloading process. In the Ramp Lander design, the payload and PSS are mounted on top of the payload unloader, which is mounted on top of the lander. The unloader for this design will be a short flat cart approximately five meters in width.

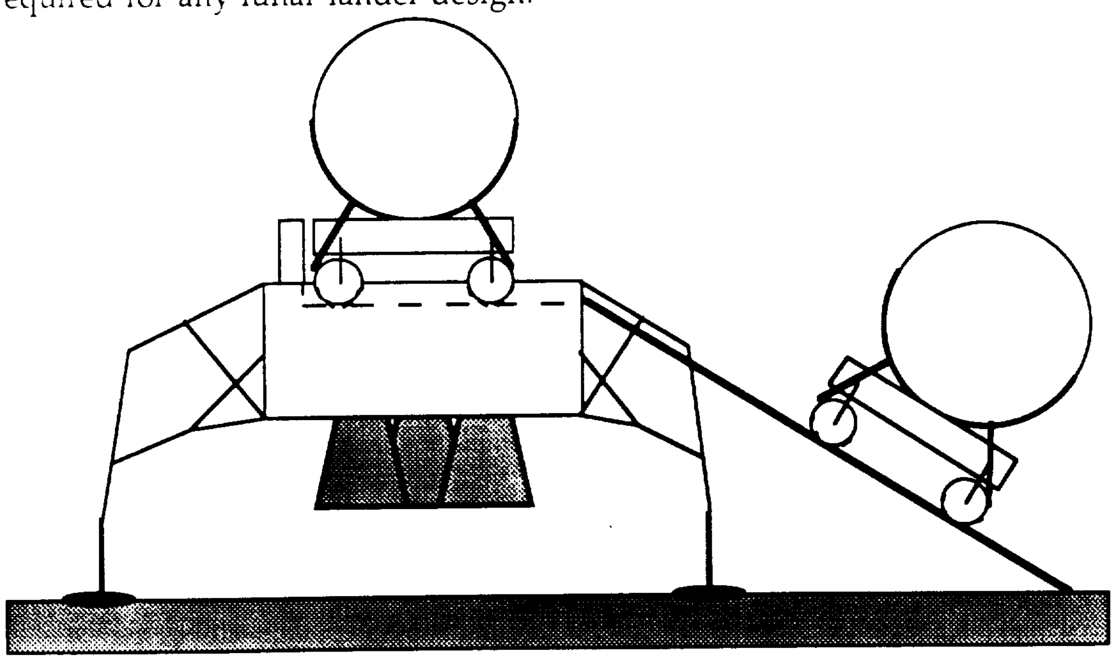


Figure A.6 Ramp Lander Concept Side View

Upon landing, the unloader will detach itself from the lander and drive down the ramp. The direction of motion of the unloader is with the wide face of the payload facing forward. The unloader will be supported with a tether to prevent tipping during the descent down the ramp. The unloader will then deliver the payload to the desired location. To unload the payload, the unloader will detach itself from the payload, lower itself from under the payload, and drive out from below the payload, which is supported by the PSS. The unloader will then remount the lander via the ramp or remain on the surface to await the return of the lander with another payload.

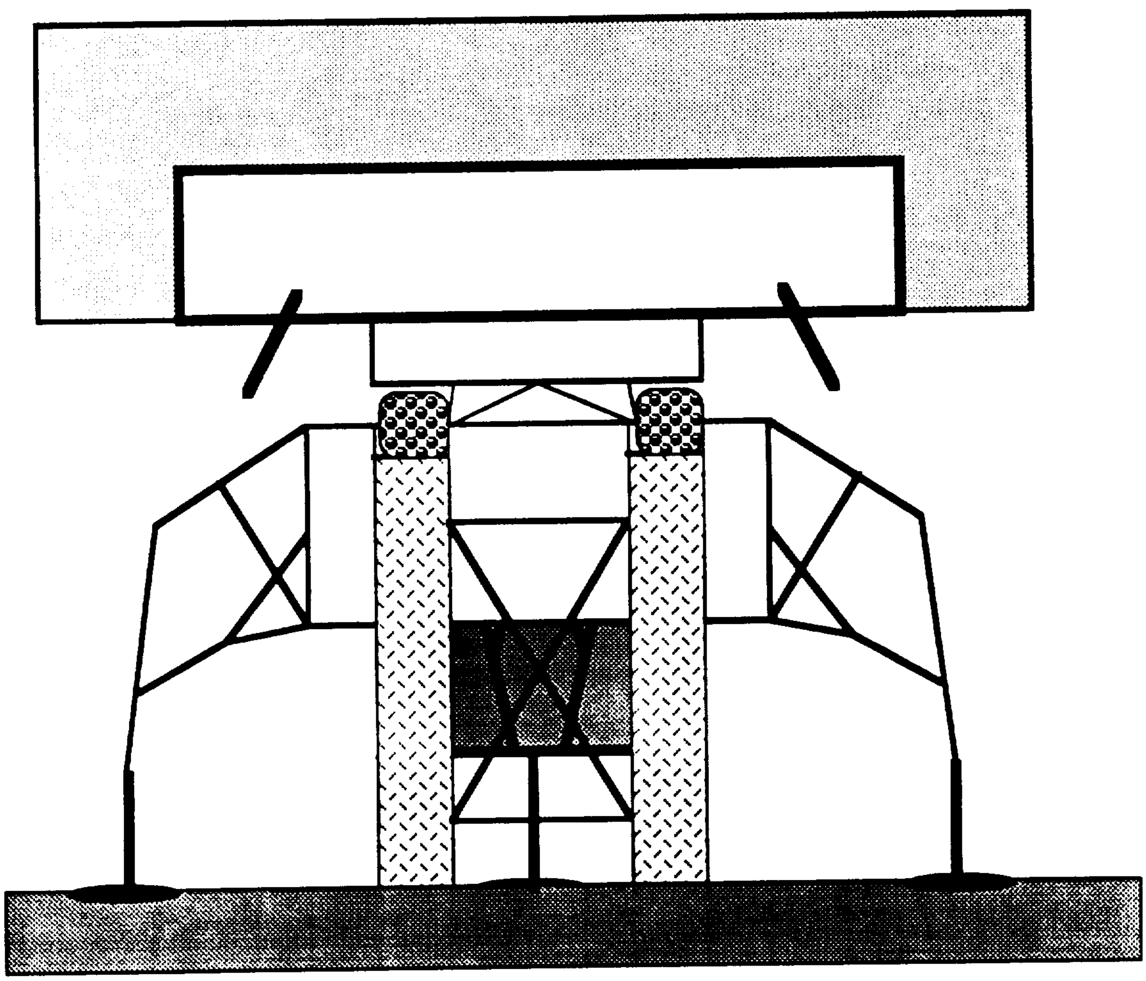


Figure A.7 Ramp Lander Concept Front View

La Rotisserie

Figure A.8 contains a front view of the La Rotisserie lander concept. The payload and the unloader are loaded in an inverted position on top of the lander. Once the lander is stabilized after landing, its entire structure will go through a 180° rotation about point A.

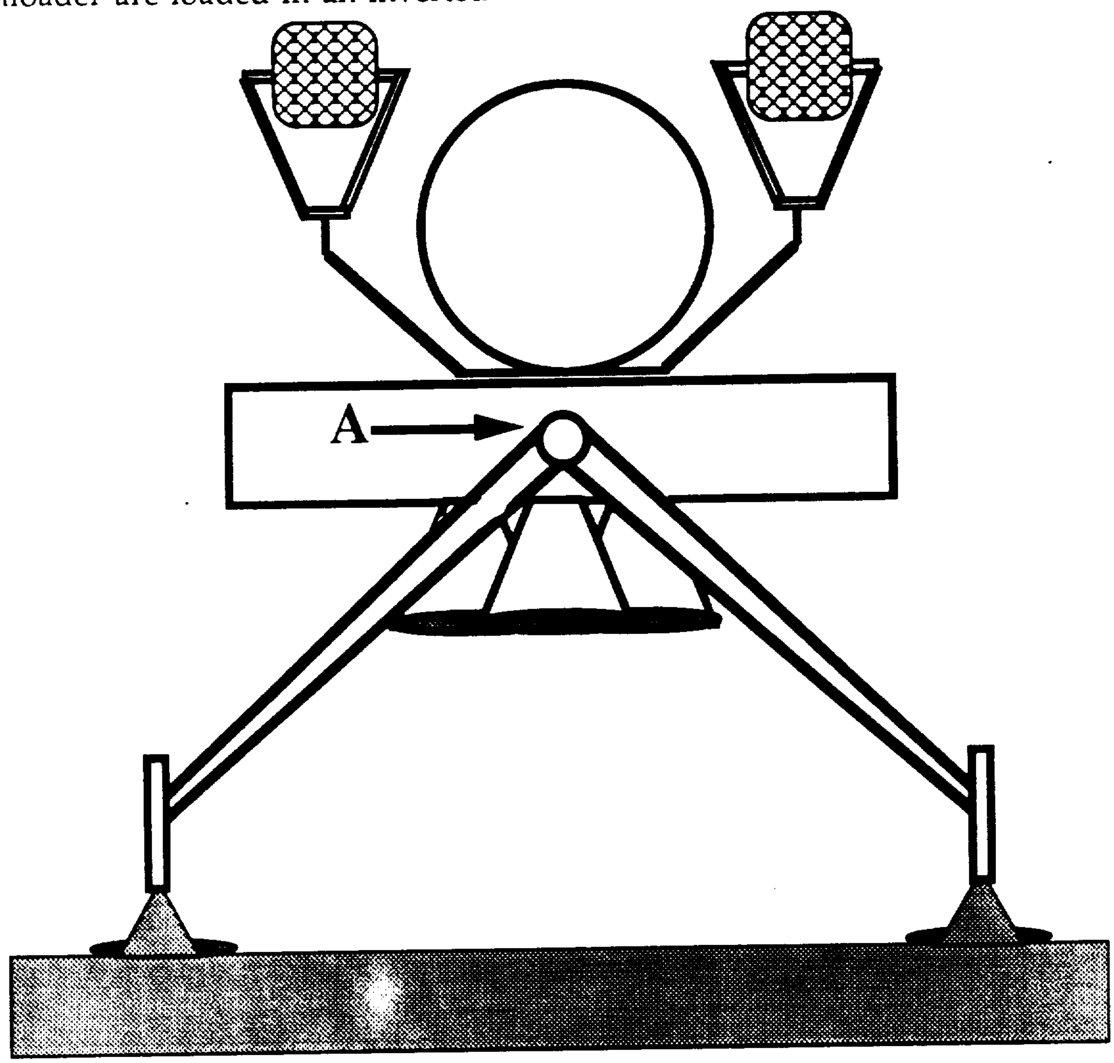


Figure A.8 La Rotisserie Lander Concept Front View

The rotation sequence, which will take at least thirty minutes, is shown in Figure A.9. After the spacecraft structure is inverted the unloader will be lowered to the surface by a cable system. The unloader will then be free to drive out from below the lander to deliver its payload to the desired location. Upon arrival at the desired location the unloader will lower the payload to the surface by means of a cable system similar to that of the lander's. The unloader will then be able return to the lander and be lifted aboard by the cable system, or to remain on the lunar surface a safe distance away from the lander's exhaust plume.

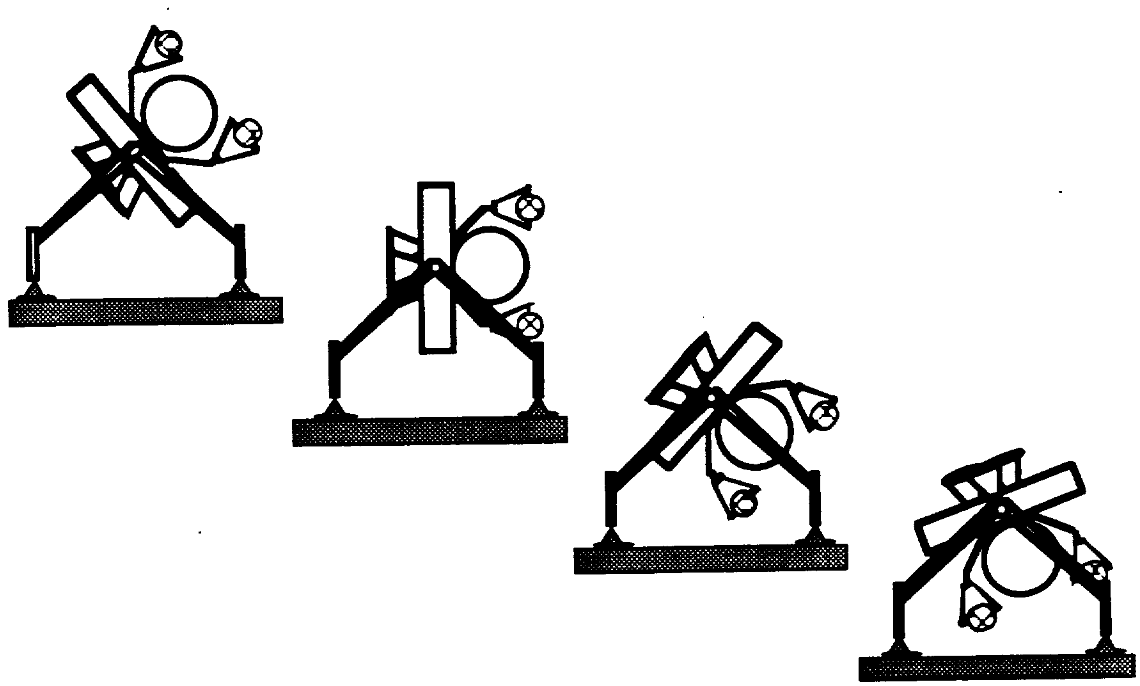


Figure A.9 La Rotisserie Lander Rotation Sequence

Figure A.10 contains a side view of the La Rotisserie concept. The unloader is shown to be a derivative of a log transport cart used in the logging industry. It will utilize an all wheel drive system for redundancy purposes and four wheel steering for maneuverability. Several different frame designs and wheel types were investigated. This unloader will give approximately one meter of clearance between the bottom of the payload and lunar surface during transport.

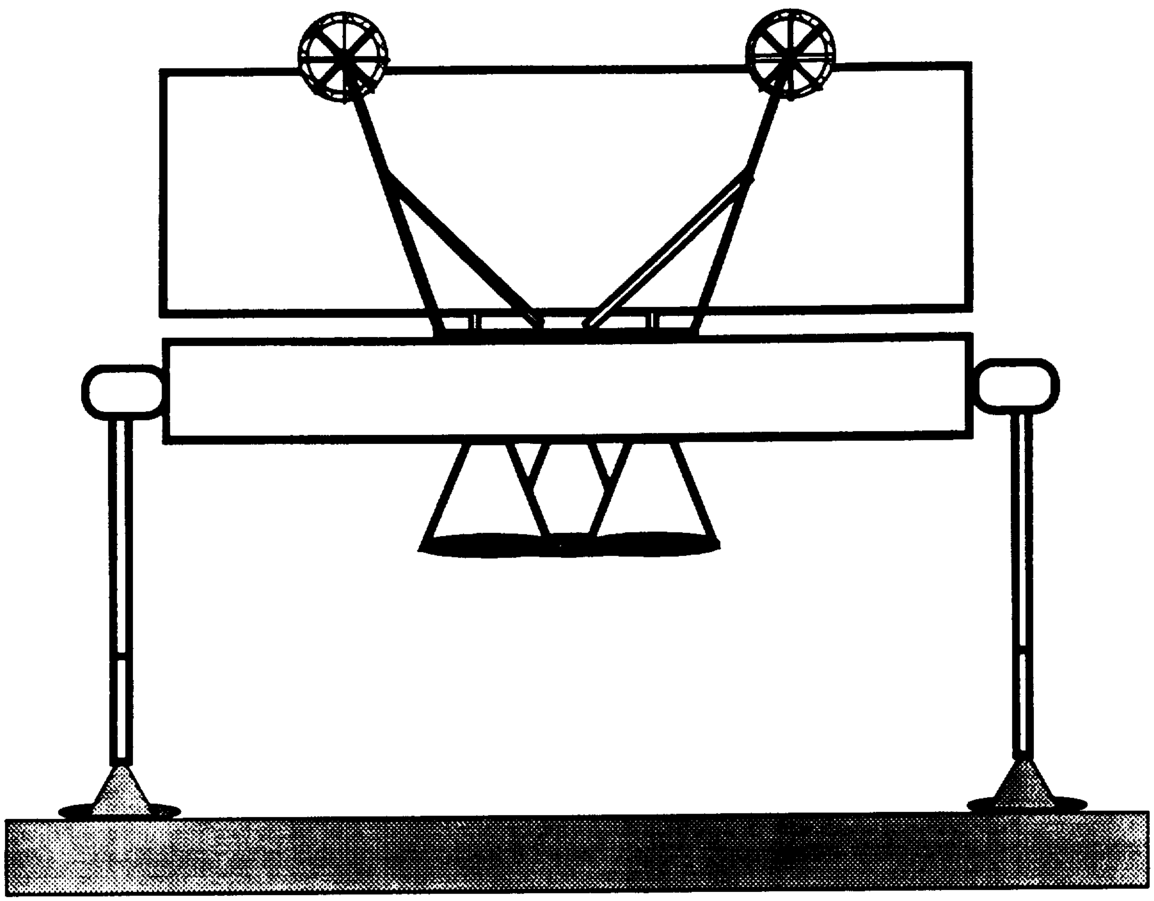


Figure A.10 La Rotisserie Lander Concept Side View

Design Decision Matrix

A decision matrix was used to determine which of the two remaining lander concepts would be chosen by RS Landers. Figure A.11 contains the decision matrix that was used. Each member of RS Landers was involved in determining the design considerations and the weighting methods that were used in the matrix. Each member was also involved in applying the value of each design consideration to each lander concept. Once the decision matrix was completed, it became apparent that the La Rotisserie lander concept was superior.

Design Characteristics	Weight	Designs			
		Ramp		La Rotisserie	
		Score	Total	Score	Total
Innovation	8	2	16	10	80
Simplicity (Mechanical Complexity)	9	8	72	6	54
Unloader Performance	7	6	42	10	70
Size/Mass	6	4	24	5	30
OTV Interface	4	6	24	5	20
Lander Stability	5	4	20	6	30
			198		284

Note: The possible scores ranged from 1 to 10 with 10 being the best.

Figure A.11 Lander Concept Decision Matrix

Once it was determined that the La Rotisserie Concept was superior, the advanced designs that are shown in CAD drawings at the end of this appendix were generated. Additional attention was given to the lander's structure, subsystems locations, engine placement, propellant tank locations, dimensions, and antenna placement.

Subsystems Placement

The various subsystems of the lander are arranged inside the lander structure as shown in Figure A.12 In addition to the engines, the engine compartment contains the power system. Also, all of the GNC and communications equipment are carried in the two subsystem bays. The locations of the subsystems were selected in an effort to keep the center of gravity of the lander in the center of the engine compartment.

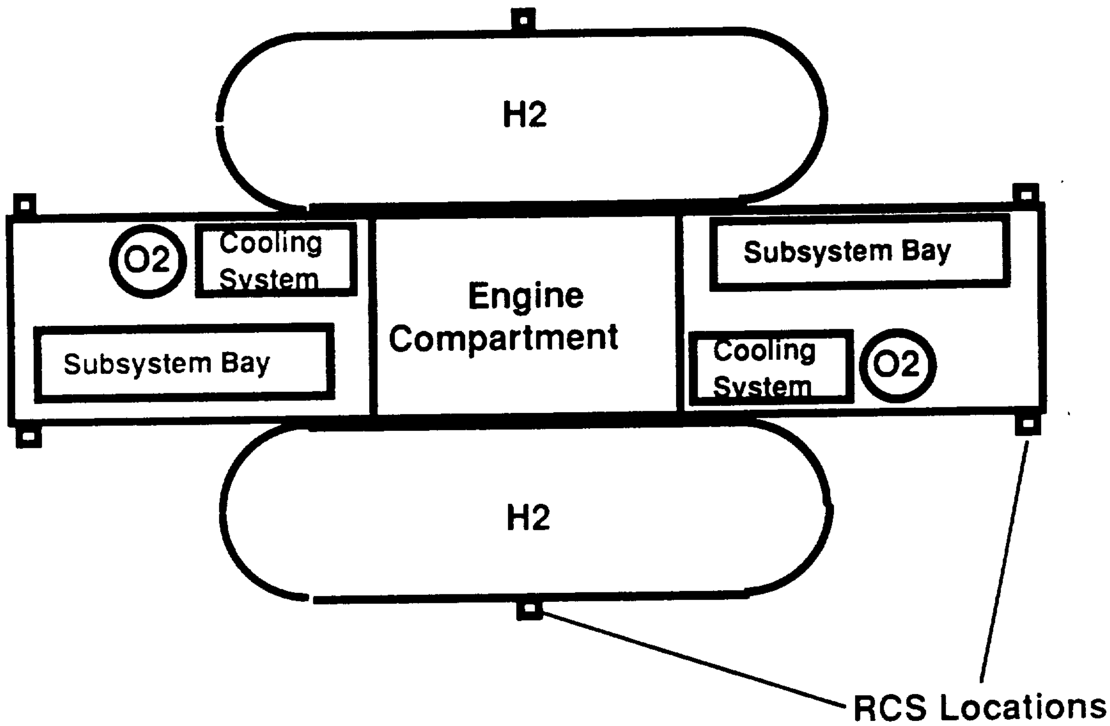
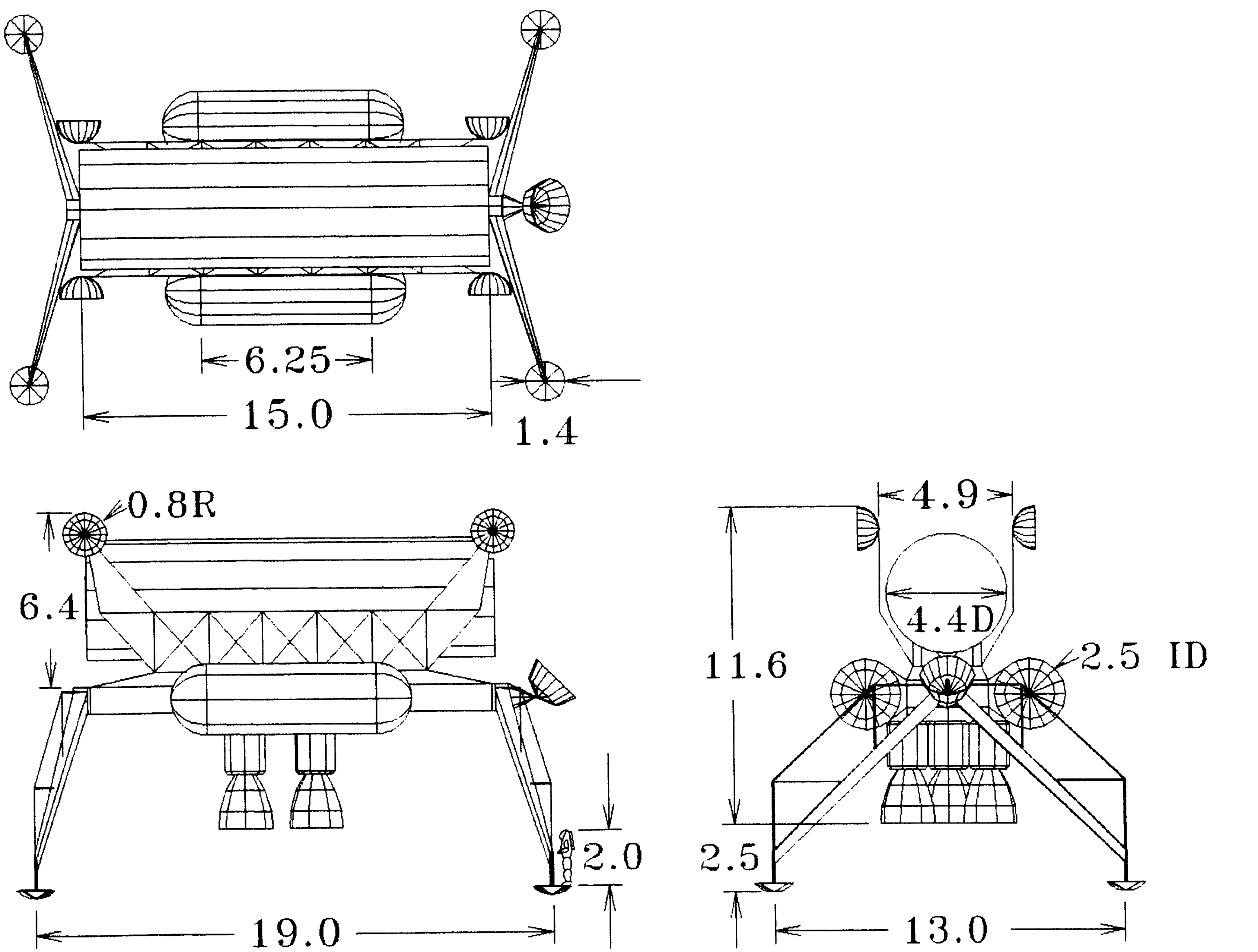
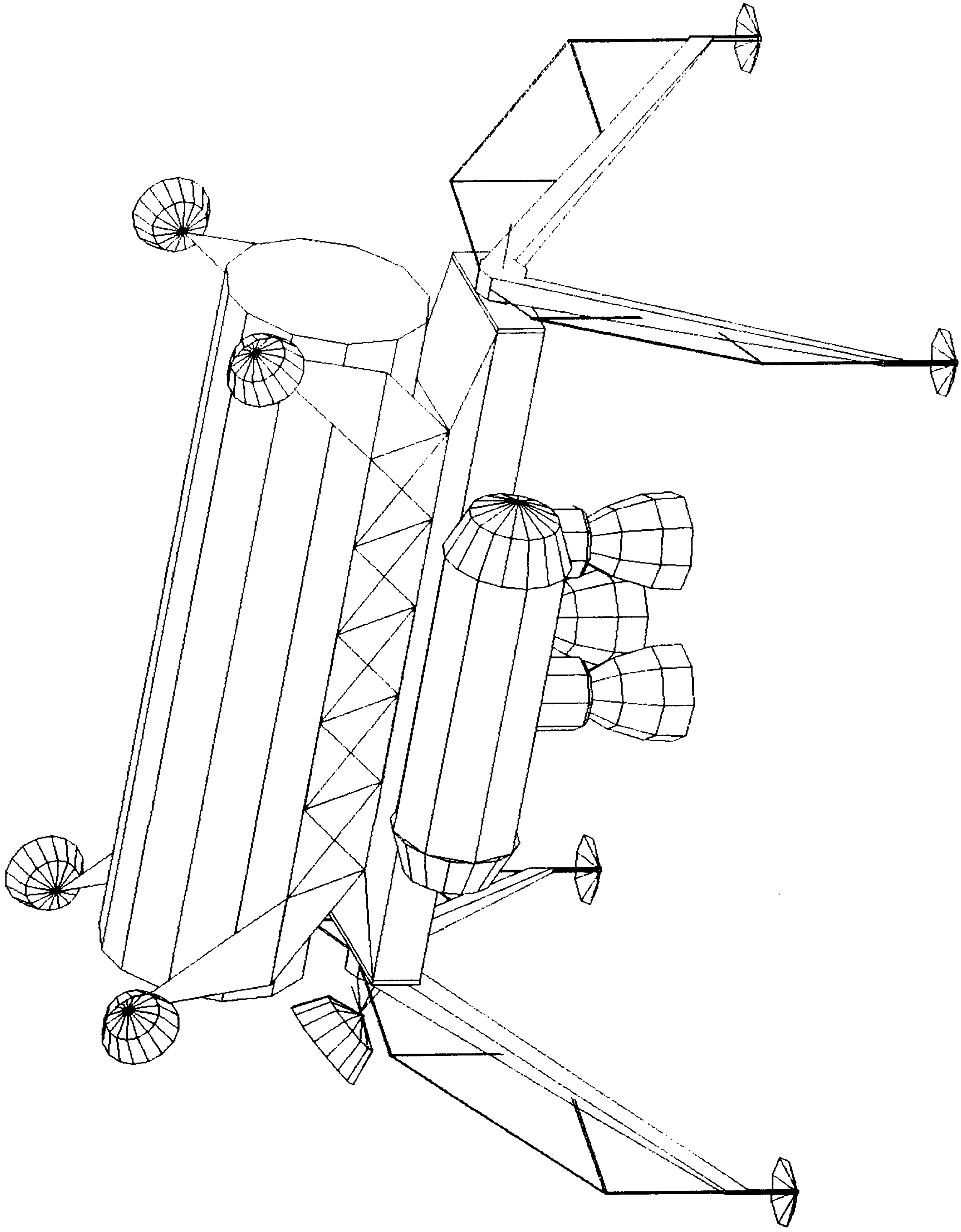
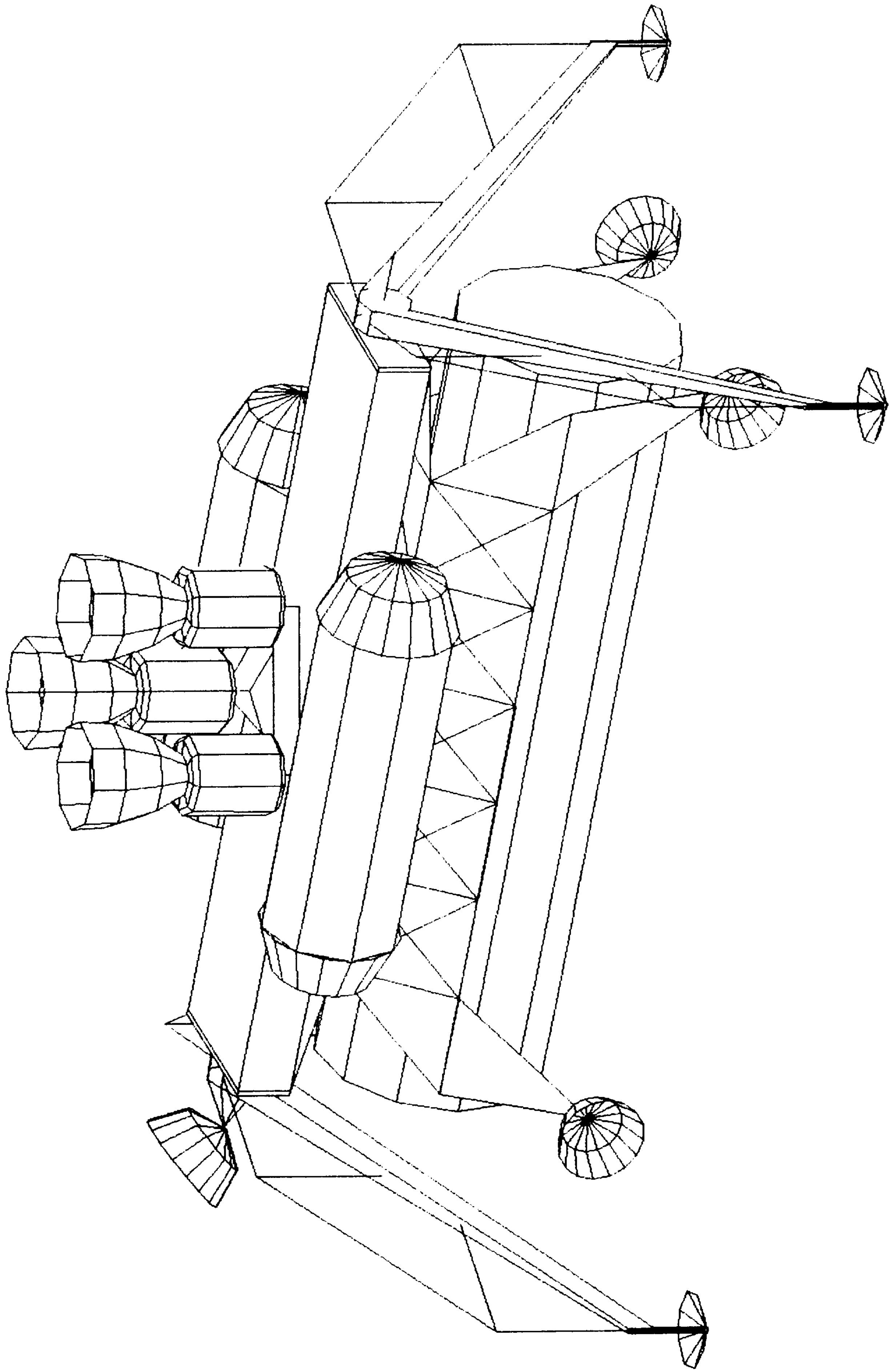
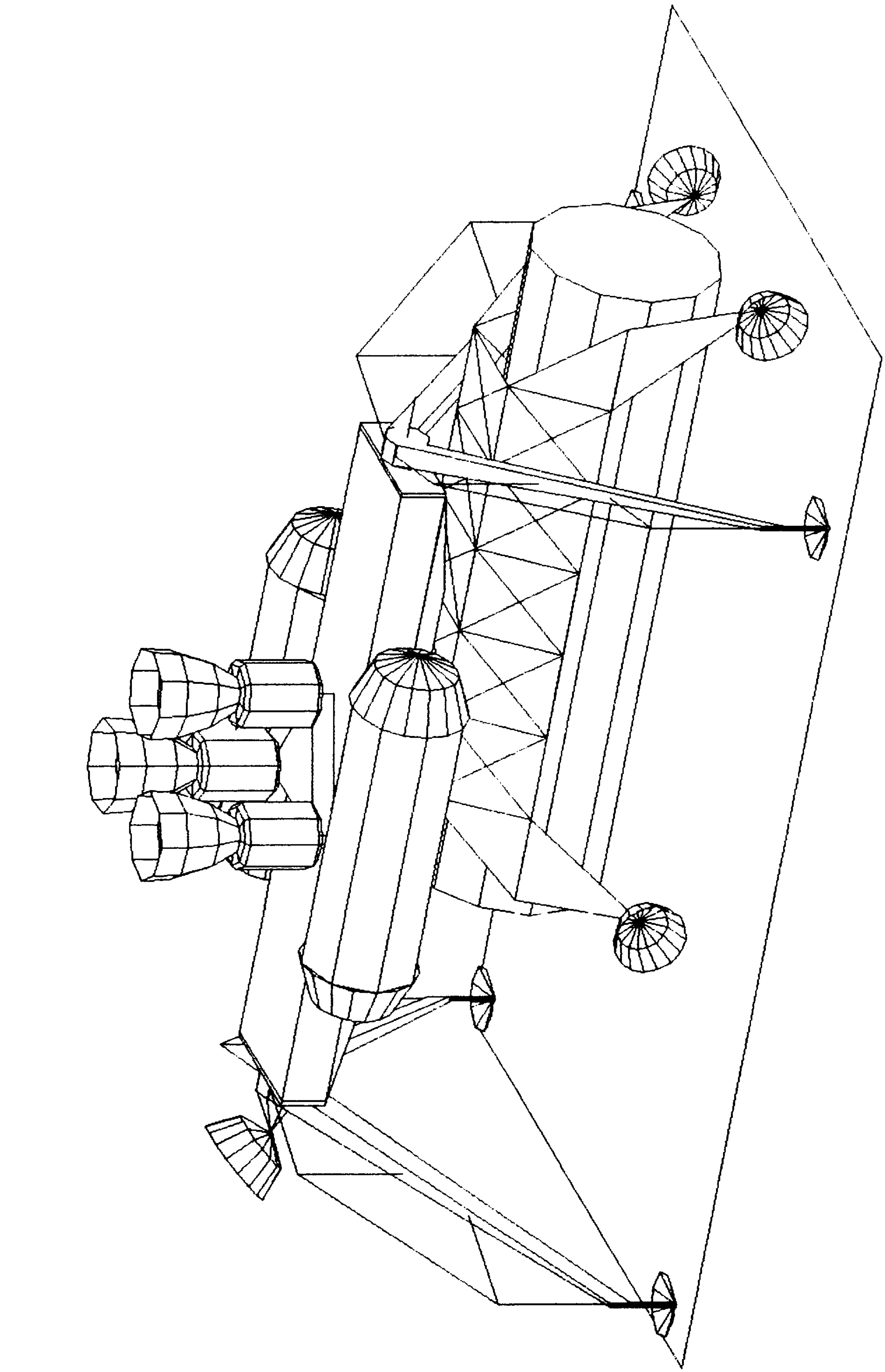


Figure A.12 Subsystems Locations


6.25
15.0
1.4
0.8R
6.4
2.0
19.0
4.9
4.4D
2.5 ID
11.6
2.5
13.0







Appendix B - Trajectories

Ascent Trajectory Calculations:

(See Figure 3.3 for orbit numbering scheme)

Given:

$\mu_L = 4902.2\ km^3/sec^2$

$Isp = 1200\ sec$

$r_1 = R_L = 1738.3\ km$

$h_2 = 11.958\ nm = 22.147\ km$

$h_3 = 54\ nm = 100\ km$

$h_4 = 59.525\ nm = 110.241\ km$

$r_2 = 1760.45\ km$

$r_3 = 1838.31\ km$

$r_4 = 1848.54\ km$

$\Delta v_{12} = 5936.5\ ft/sec = 1.8095\ km/sec$

Compute Δv_3:

$a_{24} = (r_2 + r_4)/2 = 1804.5\ km$

$v_{3_{2\text{-}4}} = \sqrt{\mu_L(2/r_3 - 1/a_{2\text{-}4})} = 1.61762\ km/sec$

$e_{2\text{-}4} = (r_4 - r_2)/(r_4 + r_2) = 0.02441$

$p_{2\text{-}4} = a_{24}(1 - e^2_{2\text{-}4}) = 1803.41883\ km$

$h_{2\text{-}4} = \sqrt{\mu_L p_{2\text{-}4}} = 2973.324\ km^2/sec$

$v_{3_{LLO}} = \sqrt{\mu_L / r_3} = 1.633\ km/sec$

$\cos(\varnothing_3) = h_{2\text{-}4}/(r_3 \bullet v_{3_{2\text{-}4}}),\ \varnothing_3 = 0.89628°$

$\Delta v_3^{\,2} = v_{3\ 2\text{-}4}^{\,2} + v_{3\ LLO}^{\,2} - v_{3_{2\text{-}4}} \bullet v_{3_{LLO}} \bullet \cos(\varnothing_3)$

$\Delta v_3 = 0.02971\ km/sec$

$\Delta v_{3_{Ascent}} = \Delta v_{12} + \Delta v_3 = 1.8392\ km/sec$

Fuel Mass for Δv_3:

$\Delta m_3 = m_{Inert} \bullet [\exp^{\{\Delta v_3/(I_{sp} \bullet g_e)\}} - 1]$

$m_{Inert} = = 18739\ lbm$

$I_{sp} = 1200\ sec$

$g_e = 0.00981\ km/sec^2$

Δm_3 = 47.36 lbm

Time of Flight Δt_{2-3}:

$\cos(E_3) = \{[e_{2-4} + \cos(f_3)]/[1 + e_{2-4} + \cos(f_3)]$

$f_3 = \cos^{-1}[(p_{2-4} - r_3)/(e_{2-4} \bullet r_3)] = 141.034°$

$\cos(E_3) = -0.767681$, $E_3 = 140.1461° = 2.44601$ rad.

$\Delta t_{2-3} = \sqrt{a^3/\mu_L} \bullet [2k\pi + \{E_3 - e_{24} \bullet \sin(E_3)\} - \{E_0 - e_{24} \bullet \sin(E_0)\}$

Δt_{2-3} = 2661 sec = 44.35 min

Descent Trajectory Calculations:

(See Figure 3.5 for orbit numbering scheme)

Given:

$\mu_L = 4902.2\ km^3/sec^2$

Isp = 1200 sec

R_L = 1738.3 km

$h_2 = h_1$ =54 nm = 100 km

h_A = 52.35 nm =96.96 km

h_3 = 9.098 nm = 16.85 km

$h_4 = 0$

$r_1 = r_2 = h_1 + R_L$ = 1838.31 km

$r_A = h_A + R_L$ =1835.26 km

$r_3 = h_3 + R_L$ = 1755.2 km

$r_4 = R_L$ =1738.3 km

Compute Δv_{12}:

$v_1 = \sqrt{\mu_L / r_1}$

$v_2 = \sqrt{\mu_L(2/r_2 - 1/a_{23})}$

$a_{2-3} = (r_2 + r_3)/2$ = 1796.8 km

v_2 = 1.614 km/sec

$\Delta v_{1-2} = v_1 - v_2$ = 0.01898 km/sec

Compute Δv_{23}:

$v_{3_{A-3}} = \sqrt{\mu_L(2/r_3 - 1/a_{A3})}$

$a_{A-3} = (r_A + r_3)/2 = 1795.23$ km

$v_{3_{A-3}} = 1.6897$ km/sec

$\Delta v_{23} = v_{3_{A-3}} - v_{3_{2-3}} = -0.000744$ km/sec

$v_{3_{2-3}} = \sqrt{\mu_L(2/r_3 - 1/a_{23})} = 1.6904$ km/sec

$\Delta v_{34} = 6232.24$ ft/sec $= 1.90$ km/sec

$\Delta t_{34} = 375$ sec

$\Delta t_{12} = \Delta m_{12}/m_{12}$

Let $M_1 = M_{Inert} + M_{Fuel\ Descent} + M_{Payload} = 43606$ lbm

$\Delta m_{12} = M_1 \bullet [\exp^{\{\Delta v_2/(I_{sp} \bullet g_e)\}} - 1] = 70.363$ lbm

$\Delta t_{12} = \Delta m_{12}/m_{12} = 3.74$ sec

$M_3 \cong M_1$

$\Delta m_{23} = M_3 \bullet [\exp^{\{\Delta v_3/(I_{sp} \bullet g_e)\}} - 1] = 2.76$ lbm

$\Delta t_{23} = \pi/\sqrt{\mu_L} = 3417$ sec

maximum mass flow rate and maximum thrust required

$\Delta m/\Delta t_{burn} = 0.584944$ lbm/sec

$T_{max} = (\Delta m/\Delta t_{burn}) * I_{sp} * g_c = 22{,}584$ lbf

Program Lander Documentation:

Program Lander has the following five basic assumptions:

- Lander descends from or ascends to a LLO of 15 to 500 nm.
- Attitude dynamics and rotational motion are not simulated.
- A spherical moon model is used.
- A lunar atmosphere is not present.
- Gravitational harmonics are not modeled.

Lander is a three-degree-of-freedom simulation which can be used to simulate the descent from LLO to the lunar surface or an ascent from the lunar surface to LLO. For ascent simulation, the spacecraft vertically accelerates away from the ground at full thrust. When the local velocity reaches 30 ft/s, the vehicle turns down range using a pitch-over maneuver and proceeds to fly a

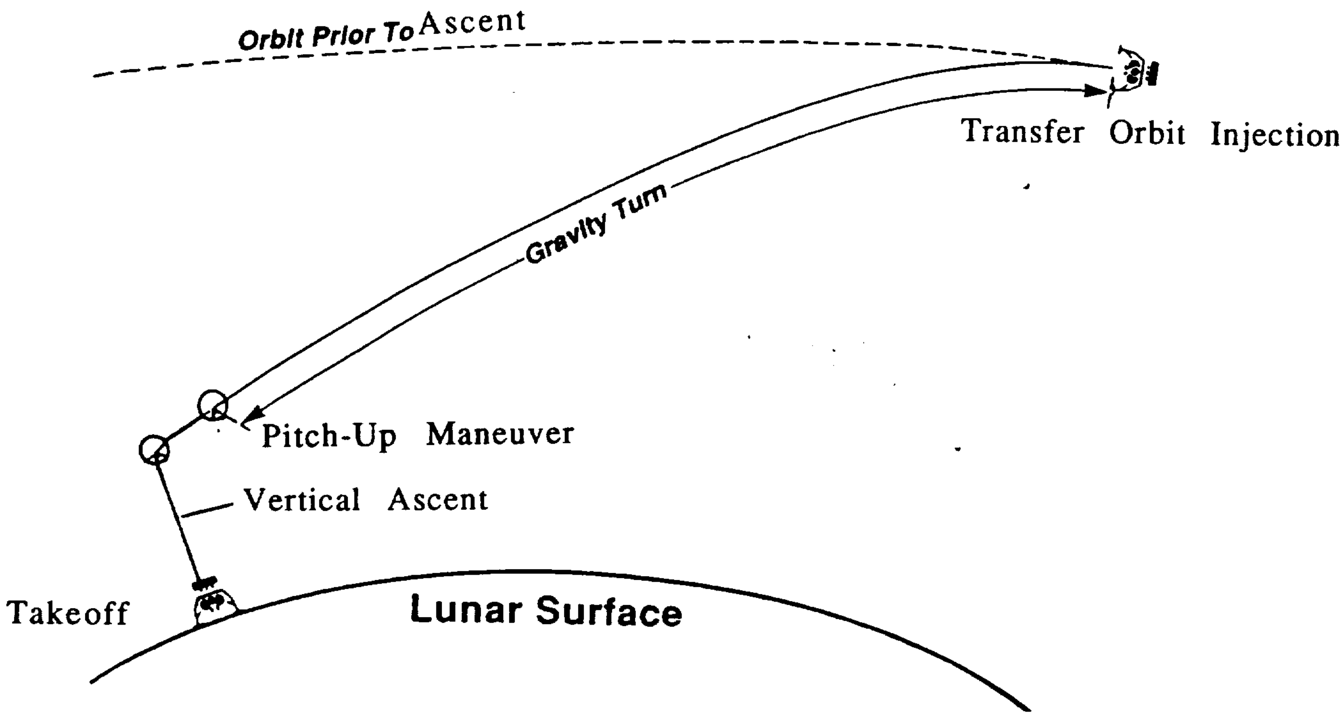


Figure B.1 Gravity Turn Ascent Trajectory

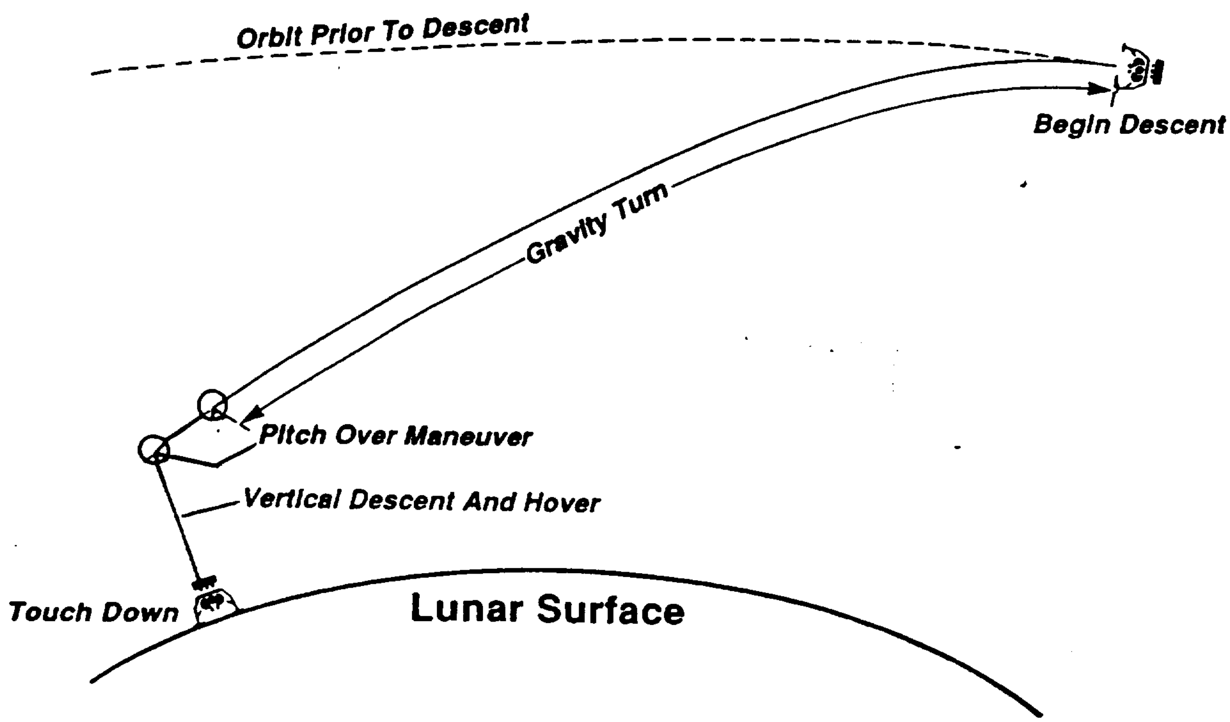


Figure B.2 Gravity Turn Descent Trajectory

gravity turn until Main Engine Cutoff, as shown in Figure B.1. The spacecraft then coasts until it reaches the required altitude, where it performs an orbit insertion burn.

For descent simulation the lander initially performs a deorbit burn in LLO. The vehicle then coasts to pericynthion, where it reignites its engines and begins a gravity turn descent, as shown in Figure B.2.. When the local horizontal velocity reaches zero, the lander pitches up to a vertical orientation and begins to hover in search of a landing site.

The Lander program uses a spherical coordinate system which is depicted in Figure B.3.

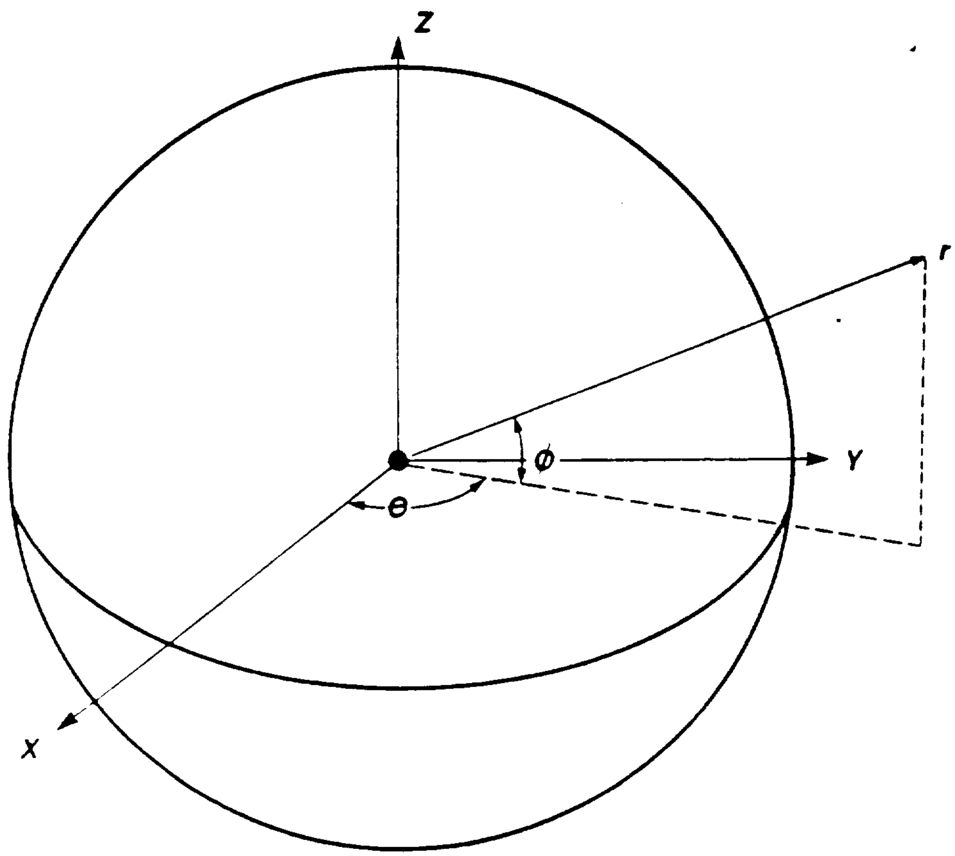


Figure A.3 Spherical Coordinate System Used by Lander

The spherical coordinate system equations of motion of a spacecraft of mass (m) under the influence of thrust (T) and gravity (g) are given by:

$$\frac{d^2r}{dt^2} = r[\frac{d\theta}{dt}]^2[\cos(\phi)] + \frac{dr}{dt}\frac{d\phi}{dt} + \frac{T}{m}\sin(\gamma) - g$$
,

$$\frac{d^2\theta}{dt^2} = 2\frac{d\theta}{dt}\frac{d\phi}{dt}[\frac{\sin(\phi)}{\cos(\phi)}] - \frac{2\frac{dr}{dt}\frac{d\theta}{dt}}{r} + \frac{T\cos(\gamma)\sin(\psi)}{mr\cos(\phi)},$$

$$\frac{d^2\theta}{dt^2} = \frac{T\cos(\gamma)\cos(\psi)}{mr} - [\frac{d\theta}{dt}]^2\cos(\phi)\sin(\phi) - \frac{2\frac{dr}{dt}\frac{d\phi}{dt}}{r},$$

where γ = Flight path angle and Ψ = Heading.

The Lander program uses a Newton-Raphson technique to optimize the pitch-over maneuver and the MECO time for proper LLO insertion. Integration of the equations of motion is performed using a Runge-Kutta fourth order integrator.

Table A.1 shows a test matrix of orbital and propulsion data, which provided four test cases for the Lander program.

Orbital/Propulsion Parameters	DESCENT Nuclear	DESCENT LOX	ASCENT Nuclear	ASCENT LOX
Landing site latitude (Deg)	26.1011	26.1011	26.1011	26.1011
Landing site longitude (Deg)	3.6527	3.6527	3.6527	3.6527
Hover time (Sec)	60.0	60.0	N/A	N/A
Time to MECO (Sec)	440	440	440	440
Holding orbit pericynthion (nm)	54	54	54	54
Holding orbit apocynthion (nm)	54	54	54	54
Pitch-over angle (Deg)	70	70	70	70
Holding orbit inclination (Deg)	26.2	26.2	26.2	26.2
Payload mass (Lbm)	15,432	15,432	0	0
Inert Mass (Lbm)	21,560	21,560	21,560	21,560
Propellant mass (Lbm)	9,665	21,710	3,325	8,953
Specific impulse (Sec)	1,200	480	1,200	480
Maximum thrust (Lbf)	22,584	32,131	11,008	13,724
	CASE 1	CASE 2	CASE 3	CASE 4

Appendix C - Mass Statement

Appendix C Mass Statement

When delivering a payload of 7000 kg, the total deorbit mass of the lander will be 21584 kg. In addition to the payload mass, this deorbit mass includes 9780 kg of inert mass and 4804 kg of fuel. A mass of the lander is broken down in Table C-1. The masses of the inerts were either calculated or obtained from various references.

Table C-1 Mass Statement

Item	Mass (kg)
Payload	7000
Inerts	
Structure	
(Lander)	2290
(Unloader)	1200
3 Engines w/Shielding	3000
RCS	600
Fuel Tanks w/Insulation	820
Power System	700
Refrigeration System	500
Rotation Motors & Winches	300
GN&C	150
Data Processing	40
Communication	50
Thermal Control	130
Total Inert Mass	9780
Fuel	
Descent	2876
Ascent	1508
RCS	420
Total Fuel Mass	4804
Deorbit Gross Mass	21584

Appendix D - Fuel Tank Sizing

Appendix D Fuel Tank Sizing

The lander has four fuel tanks that are sized according to the total mass of cryogenic hydrogen and oxygen required for descent/ascent and attitude control. The hydrogen tanks are cylindrical with spherical end caps and are a total of 8.75 meters long with a radius of 1.25 meters. The two oxygen tanks are spherical with a radius of .341 meters. Both hydrogen and oxygen tanks are reinforced with baffles and are insulated with 2.5 inches of multi-layer insulation. The hydrogen tanks also have a 1 millimeter outer shell for protection against micrometeorites. All four tanks are constructed out of an aluminum lithium alloy with a density of .093 lb/in^3 and a yield stress of 689 MPa. The total mass of the four fuel tanks, including insulation and micrometeorite protection is 815.5 kilograms. The various fuel tank parameters are in Table D-1

Table D-1. Fuel Tank Parameters

	Hydrogen Tanks	Oxygen Tanks
Fuel Stored(kg)	2222	180
Fuel density(kg/m^3)	70.9	1140
Number of tanks	2	2
Tank ullage	5%	5%
Tank volume(m^3)	32.907	.166
Tank Radius(m)	1.25	.341
Cylinder Length(m)	6.25	-----
Internal Pressure(MPa)	.10	.10
Al-Li yield stress(MPa)	689	689
Allowable stress(MPa)	516.75	516.75
Tank thickness(mm)	.3	.3
Tank shell volume(in^3)	1257.3	26.76
Al-Li density(lb/in^3)	.093	.093
Tank mass(kg)	53.15	1.13
Mass of 2.5 in. of MLI(kg)	159.44	3.39
Mass of baffles(kg)	1.59	.03
Outer shell thickness(mm)	1	-----
Outer shell volume(in^3)	4471.5	-----
Outer shell mass(kg)	189.02	-----
Total mass of tanks(kg)	**806.4**	**9.1**