Project Selene: AIAA Lunar Base Camp

AIAA Space Mission System 2019-2020

Virginia Tech Aerospace Engineering

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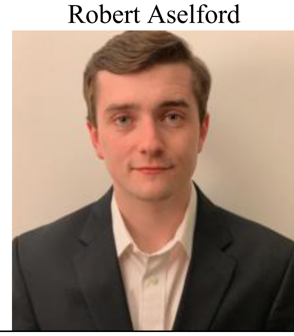
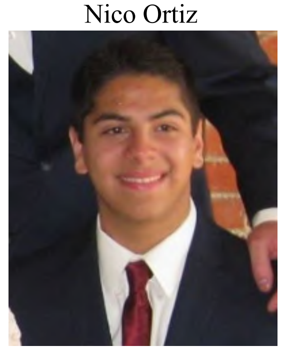
Maedini Jayaprakash CCDH/Operations 	**1085663**	Robert Aselford 	**1109195**
Michel Becker Attitude, Trajectory, Orbits and Launch Vehicles 	**1109194**	Nico Ortiz 	**1109533**

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1. Symbols and Acronyms

Δv	=	Velocity change in orbital mechanics
ISP	=	Rocket Specific Impulse
g	=	Earth's gravitation constant = 9.81 m/s^2
μe	=	Gravitational Constant * Mass of Earth
μm	=	Gravitational Constant * Mass of Moon
re	=	Radius of the Earth
rm	=	Radius of the Moon
AHP	=	Analytical Hierarchical Process
DT	=	Direct Transfer
DTE	=	Direct to Earth
ECLSS	=	Environmental Control and Life Support System
EVA	=	Extra Vehicular Activity
ISRU	=	In-Situ Resource Utilization
LCT	=	Lunar Communications Terminal
LEO	=	Low Earth Orbit
LET	=	Low Energy Transfer
LLO	=	Low Lunar Orbit
TOF	=	Time of Flight
PBR	=	Pebble Bed Reactor
RFP	=	Request for Proposal
SEV	=	Space Exploration Vehicle
SPR	=	Small Pressurized Rover
WEH	=	Water Equivalent Hydrogen
VSD	=	Value System Design

2. Executive Summary

Due to technological advancements, commercial opportunities, and societal pressure the advancement of space exploration has seen a renewed emphasis approaching that of the Apollo era. The current administration's initiative is to send humans back to the Moon by 2024 with the goal of establishing a permanent presence on the Moon. NASA has also emphasized the importance of proving technology and procedures in the lunar environment where the risk to life and capital is greatly reduced when compared to missions that would require greater travel time, such as Mars. Testing new technologies and techniques for future space exploration and colonization on the Moon allows for systems to be tested and proven. Compared to testing on the Martian surface the human risk at the lunar surface is lower. In the event of an accident or failure aid can be provided to crew members much quicker. Lunar testing will enable the development of technologies for future exploration and settlement on celestial bodies. The RFP for the following mission requires the construction of a sustainable and expandable lunar base that is able to support a crew of four astronauts for 45 days while performing relevant scientific experimentation that will allow for the continued development of the space environment within a budget of $12 B by 2031. Due to the low budget allocated for this mission, when considering the infrastructure that must be delivered to the surface and the long timeline for getting all necessary materials to the moon, low energy options were evaluated for transporting the habitat and other materials to the Moon. Typically, spacecraft traveling to the Moon use Direct Transfers (DT) which optimizes the amount of time by using large amounts of Δv. The short transit time was a requirement for many of these missions since many had humans on board. This mission requires only the delivery of materials and equipment to the Moon therefore transfers that require less Δv at the price on longer times of flight (TOF) were selected. From the research done, Low Energy Transfers (LET), shown in Figure 1 below, were concluded to be the optimal way to deliver the materials to the Moon. They require about 15% less Δv for the transfer while also offering extended launch windows. Low energy transfers also allow the spacecraft to target any orbit around the moon with minimal Δv changes while in the transfer orbit. The only drawback of these transfers is that they can take between 70 and 130 days to get to the Moon from low Earth orbit (LEO) depending on the exact orbit [3, 4].

Using this low energy transfer allows rockets to deliver more payload to the lunar surface. This mission plans to use primarily Falcon Heavy rockets along with one Falcon 9 rocket since they are proven technologies, provide the best performance for the lowest cost, and are quickly manufacturable. In total there will be 10 launches spanning from 2023-2030 to deliver components needed to the lunar surface or lunar orbit. The Falcon Heavy rocket can land up to 10,000 kg of payload on the Lunar surface and costs $150 M per rocket [5]. Larger rockets are currently being developed, such as SLS and Starship, and would be better suited for a mission to the Lunar surface since they can transport a larger payload, but because they are still early in their development and the reliability is unknown, they were not used. Landing the upper stage of the Falcon Heavy rockets will need development and infrastructure on the lunar surface to support the success of this operation and plans to implement NASA's strategies for eliminating the boil off rates for LOX systems. As the Falcon Raptor engines use methane there is little boil off but a strategy will be developed to further eliminate the

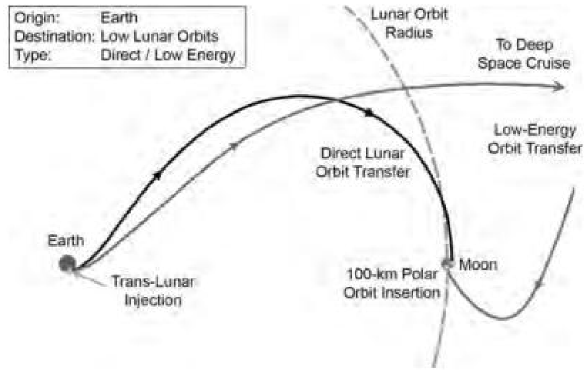

Figure 1. Comparison of a direct transfer (black) and a low energy transfer (grey) targeting an orbit around the moon. The spacecraft travels deep into space in the low energy transfer[3].

boil off for methane if necessary. Additional power sources might need to be implemented into the Falcon Heavy second stages so that the rocket sustains power during the long transfer. A landing pad with sensors will need to be built on the surface along with a self deploying crane to unload the payloads and a rover to transport the payloads and move the empty rockets off of the landing pad. Together this infrastructure and development will help lower the overall mission cost and promote a long lasting and sustainable lunar base.

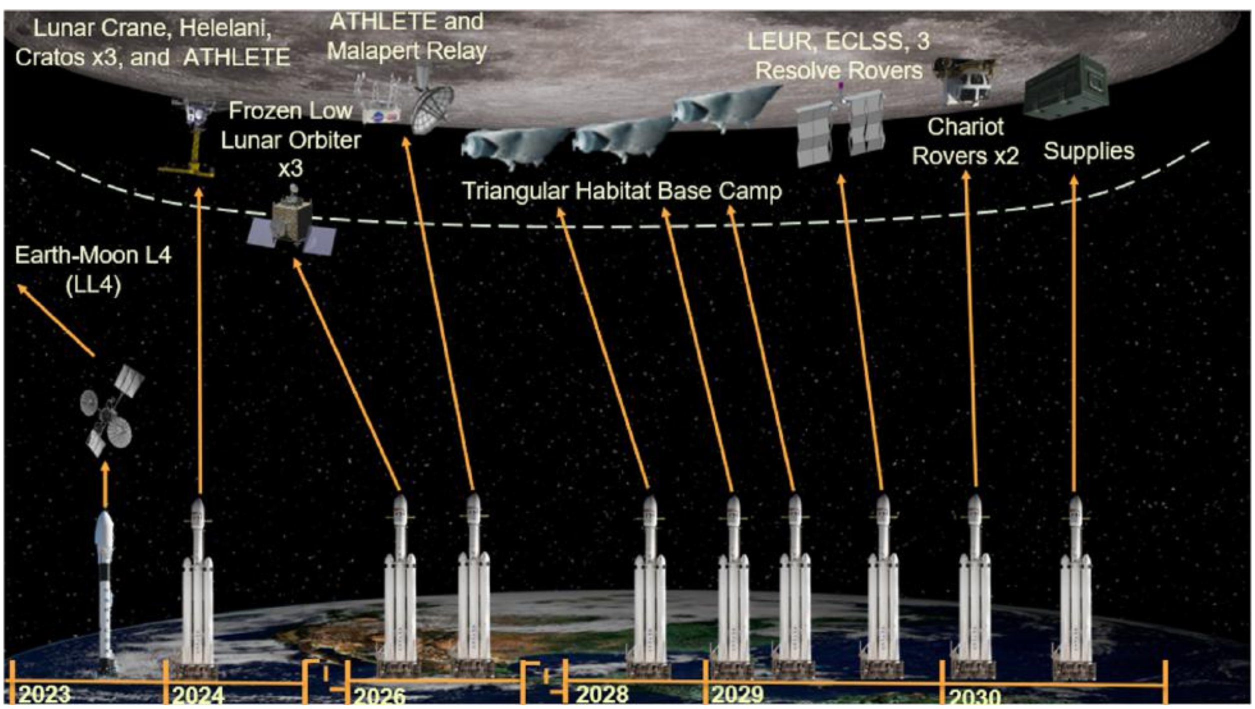

Figure 2. Concept of Operations.

Another primary requirement for this long duration manned mission was a reliable communication system capable of direct to Earth (DTE) communication [1, 2]. In order to provide coverage for the astronauts and ground assets, a three level redundant system was designed. The primary link is located at Malapert Mons, a mountain of high elevation at the lunar south pole. Each relay was designed to satisfy NASA and International space communication requirements, while considering the constraints of the lunar environment. The link architecture is shown in Figure 27 and displays the three tiers of communication systems. A lunar communications terminal with a K/S-band frequency antenna and transponder at Malapert will be in direct line of sight with Earth. This terminal will be the focal point for all surface operations as well as for relaying received signals to and from Earth. Also located at Malapert will be the optical terminal, MAScOT. MAScOT will be responsible for relaying all science data collected on the surface back to Earth. The secondary method of providing communication will be through a constellation of three satellites in a polar orbit around the Moon. The final method will be through a satellite relay orbiting in the Earth-Moon LL4 Lagrange point. The lunar communications terminal (LCT) will act as a lunar ground station, with a redundancy located at the habitat. Both the polar constellation and Lagrangian satellite are capable of dual Ka/S band. Through the three systems of redundancies the astronauts will be better protected and supported on the moon, resulting in a safer mission.

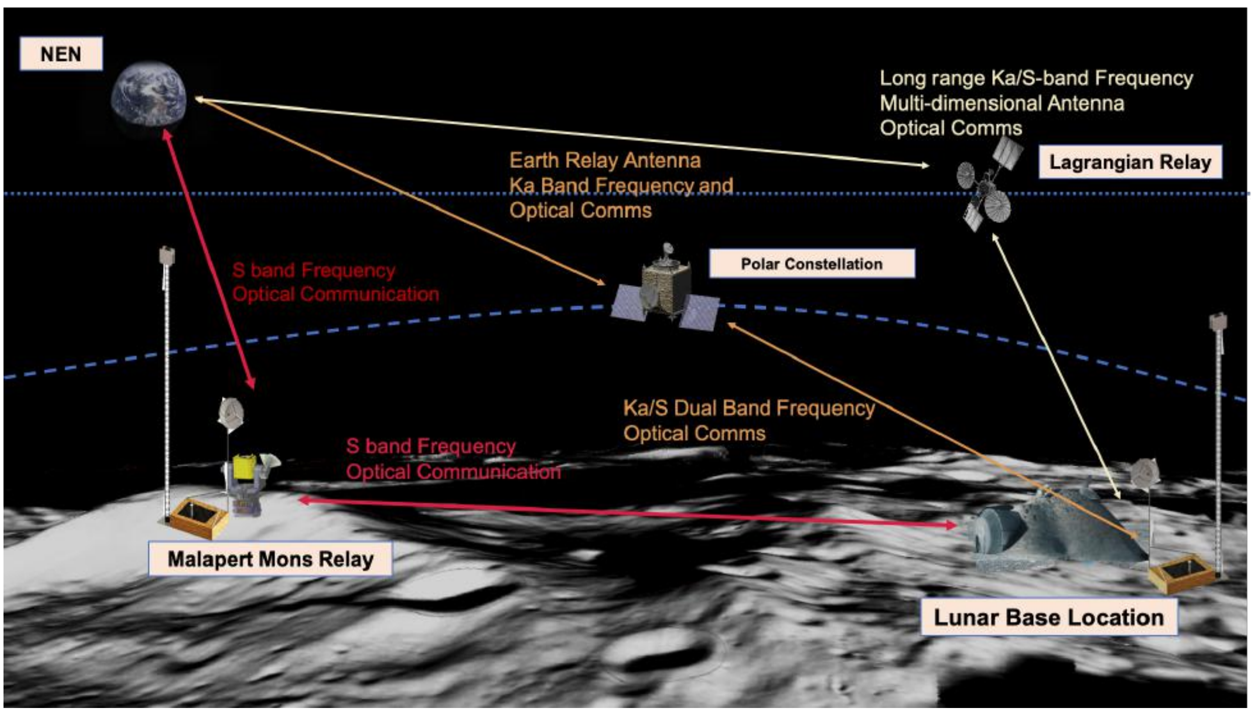

Figure 3. Communication Architecture.

To support the scientific focus of this mission each relay will also be capable of optical communications. The use of optics will allow for large amounts of data to be relayed back to Earth and lay the foundation for setting up a lunar communication network for future missions. Each redundancy will have both RF and optical capabilities to ensure a reliable and high speed communication system.

The location of the lunar habitat determined the ISRU resources available, the type of geological experiments that can be performed, and the environmental conditions that the crew and mission hardware will be exposed to. Risk was the main driver of the habitat location while other factors such as complexity of mission design, longevity of the habitat hardware, expandability of the base, and the potential scientific return were of significant importance. The environmental factors and ease of ISRU acquisition determined the longevity of the habitat. The density of water equivalent hydrogen (WEH) was linked to the potential expandability of the base. The distance from Malapert Mons and the sites of interest were considered as the differences in risk associated with the mission. Finally, the significance and amount of scientific discovery associated with the base was the basis on which the scientific return was measured. After a trade study was conducted, the ridgeline between Cabeus and Haworth craters was chosen as the base location. This was decided because of its close proximity to the two craters which both contain dense amounts of water equivalent hydrogen while also being close enough to Malapert Mons which is the location of our primary communications array. Table 6 details the specifics for the ridgeline between Cabeus and Haworth Craters [6]. Scientific experiments will be conducted to further detail the solar wind composition which is responsible for stripping the atmosphere from Mars. Insights into this area of study will help determine future experiments that will be performed on Mars and provide a more detailed understanding of the influence of solar wind. Specific craters such as Cabeus crater will also be studied and sample returns will be made to increase our understanding of the formation of our solar systems, specifically the Moon's influence on the formation of the Earth. Cabeus crater is the primary target of this mission's sample return goals since it is one of the oldest craters in the south pole that is within range of the chariot rovers which will transport the astronauts to the site. The use of regolith to construct structures will improve our capacity for ISRU. One such use of ISRU could be a shielding structure to help protect against micrometeoroid impacts, thermal cycles, and radiation exposure. This shielding structure can be paired with radiation detectors to better understand the effects of the Earth's magnetosphere in shielding radiation and lead to developments in the understanding of the space environment. To complement the discoveries and advancements made with respect to ISRU, science will be performed that focuses on the effects of long duration mission on the human body, particularly with a focus on the exposure to solar and cosmic radiation when there is little to no magnetosphere present to shield the body from harmful rays. To safely and ethically analyze the extent of radiation exposure, synthesized tissue will be isolated to controlled conditions where radiation dosage will be monitored. At various time intervals the samples will be returned to Earth for an analysis of the effects. This also advances the capabilities of regenerative medicine and availability of organs for transplants since the complex and detailed geometry of organs can be better replicated in low gravity environments.

Reliable surface power is vital to support base operations, maintain communication with the Earth, and power life support systems for the crew. Solar arrays, fuel cells, and nuclear reactors were considered for the power architecture. Even the most illuminated regions of the Moon still can have eclipses of up to 120 hours during which no solar power can be produced [7]. Architectures relying on solar power need batteries or hydrogen fuel cells to continue to support base operations during an eclipse. The main driving factors in choosing a power system are reliability and complexity of the system. The chosen power architecture is a pebble bed nuclear reactor core with two redundant 25 kW Brayton cycle generators. The pebble bed reactor is fueled with low enriched uranium with reduces security concerns associated with nuclear material. The life support and thermal control systems are powered by the reactor system, so reliability is one of the driving factors of the power architecture. An architecture with two redundant generators was chosen to decrease the risk of a power failure. This architecture offers the least complex system with high specific power. As this architecture relies on nuclear power, it can be used anywhere regardless of illumination conditions making it the most applicable to other exploration missions. The pebble bed reactor architecture is sized to provide more power than the base requires, allowing the single power system to power future additions to the lunar base.

3. Preface and Introduction

3.1 Project Management

The following report is the culmination of a request for proposal (RFP) proposed by the American Institute of Aeronautics and Astronautics (AIAA) and the senior capstone project for Virginia Tech's senior design class for aerospace engineering [8]. As such the project was subject to deadlines for both overhead organizations and the development of the project was planned accordingly. The three major deliverables are a fall semester report, a spring semester report, and an AIAA project submission. The fall report is to demonstrate the conceptual mission design that has been developed in response to the RFP demonstrating the 12 key principles of systems engineering. The spring semester report and the AIAA project submission are both similar in scope and requirements sharing close deadlines and expectations. The second phase of the project plan, which began in the early spring semester, builds upon the fall report and explores a detailed subsystem analysis.

The spring semester report is the following paper which details the systems engineering process, the methods used to develop a concept of operations, the justification of the process, and many of the components that are required to support a lunar base camp. To develop the best mission architecture mission and system synthesis was considered throughout all aspects of the project. Due to the technical nature and varied disciplines of the project, sub-teams were developed according to the organization chart. Each sub-team had a leader who was also a part of the ConOps sub-team which was responsible for the development of a cohesive system integration. In the conceptual design phase of the mission all team members joined three sub-teams to ensure a broad understanding of the project and assist with mission synthesis. During the detailed design phase members focused on specific systems to deepen their understanding and were required to be a part of two sub-teams.

Table 1. Spring semester organization chart of Project Selene on a sub-team basis for the detailed mission design phase showing the teams in bold and team leads were italicized.

Attitude, Trajectory, and Orbits	**Communications and Data Handling**	**Power, Thermal, and Environmental**	**Structures and Launch Vehicle**	**Systems Engineering**
Michel Becker	*Maedini Jayaprakash*	*Patrick Crandall*	*Logan Lark*	*Brendan Ventura*
Olivia Arthur	Bobby Aselford	Heidi Engebreth	Brendan Ventura	Michel Becker
Bobby Aselford	Olivia Arthur	Matt Pieczynski	Michel Becker	Maedini Jayaprakash
Brendan Ventura	Logan Lark	Nico Ortiz	Nico Ortiz	Patrick Crandall
			Heidi Engebreth	Matt Pieczynski
				Logan Lark

3.2 Problem Definition

3.2.1 Background and Motivation

Within the past 10 years there has been an increase in public desire for space exploration coupled with significant advancements in the aerospace industry enabling cheaper access to space environments. Much of the conversation revolves around a permanent Mars colony and such a leap would require a large advancement and validation of technology. Returning to the Moon seems a likely first step towards achieving the goal of landing on and colonizing Mars since humanity has landed on the Moon before. The Moon offers an ideal environment to test deep space systems due to its relative ease of access and its comparably harsh environment. SpaceX has begun planning its #dearMoon project, the European Space Agency (ESA) designed its Moon Village concept, and NASA seeks to establish a permanent settlement on the lunar surface. As such, lunar colonization piqued the interest of the AIAA which is a professional society that sponsors design competitions and research in the Aerospace field.

3.2.2 RFP and Description

“Project Selene” presents the challenge of designing an extended manned lunar stay as contracted by NASA and international partners. The RFP asks that the project utilize system engineering processes to present a detailed mission to design and plan a lunar habitat as well as research projects for the crew. The crew of 4 men and/or women shall live on the lunar surface for a minimum of 45 days and perform scientific tests with a focus on those which shall benefit future missions and advance technology for deep space exploration and colonization [8]. The habitat shall be sustainable and expandable and will serve as the cornerstone for future infrastructure development. Although “Project Selene”, named after the Greek goddess of the Moon, is not responsible for the transportation of the crew to the Moon, the mission architecture does need to ensure the habitat is ready for habitation within 72 hours of the crew landing. The habitat design must take measures to maintain the safety of the crew for the duration of their stay and outline the scientific experiments that will be performed. More specific requirements are outlined in Section 8 of this paper.

3.2.3 Project Scope

The scope of this project is broad, seeing as many industries and expertise would be necessary to achieve the requirements laid out in the RFP and would require significant time and monetary investments. The goals of the proposal are to create the most effective lunar habitat for NASA's needs. As such there are many requirements for manned and unmanned systems which must be satisfied. The requirements which must be met include: the transportation of materials and structures, maintaining constant communications between the Earth ground station and Moon base, reliable life support and environmental protection for the crew, and ensuring scientific return from the mission. Optimizing these systems, while remaining below the budget of 12 B 2019 USD and within the landing deadline of December 31st, 2030, is the baseline for a successful design.

We also must deliver certain documents to the AIAA within a schedule they provide. These deliverables include a Letter of Intent by February 9th, 2020 and a Proposal due on May 14th, 2020. Within our final report shall be: an executive summary, a requirements definition, a concept of operation, system level design specifications with trade studies to justify the decisions made, seamless design integration, science goals and crew operations, a cost estimate which includes mass, power, and volume budgets, a schedule for deployment, and references. Should our proposal pass the initial evaluation, we would present our design project during the 2020 AIAA Propulsion and Energy Forum on August 24-26th, 2020. Our design seeks to deliver a fully functional and expandable base camp on the Moon, as well as the transportation and communication systems needed for the mission, by the AIAA provided mission start date of December 31st, 2030 [8, 9].

The transportation of the lunar crew lies outside the scope of this mission design and is the responsibility of NASA. As such the transportation of the crew shall not be included in the costs and development of our project. As specified by the AIAA, the lander will have the necessary systems to support the crew for a duration of 72 hours.

3.2.4 Disciplines

The design team for this project includes only aerospace engineering students. The project, however, is multidisciplinary, and thus the creation of sub-teams was necessary to encompass the design requirements for AIAA. The required disciplines as outlined by the AIAA include aerospace subdisciplines such as structures, propulsion, orbital and flight mechanics, thermal and environmental control, power, communications, sensors, and systems optimization.

3.2.5 Societal Sectors

The primary stakeholders of this mission include NASA and its international partners, who are providing personnel and funding for the project, The secondary stakeholders include the taxpayers who are indirectly paying for the project, the aerospace companies that will be contracted for parts and systems of the mission, and all future space engineers who can look on our lunar colony as a precedent for future space colony missions.

3.2.6 Assumptions

As described in the RFP we are assuming that our budget for the total mission is $12 B 2019 USD and that the funds for the project will be continuous at a steady rate. Another assumption made in this project at the high mission planning level is that there is low risk of cancellation and as such the crewed part of the mission does not have to occur before December 31^{st}, 2030. Additionally, as advised by the AIAA committee, there will be little to no dependence on prior missions to the lunar surface to assist in the set up and maintenance of assets deployed to the lunar surface before the state of the crewed mission.

3.2.7 Relevant Capital and Resources

One of the AIAA requests was to utilize as much existing technology as possible while advancing space exploration with the purpose of utilizing the findings for Mars exploration. While not everything used in this design already exists, there are a number of resources at our disposal which have been researched and developed by other engineering entities. A pre-existing system in the design is the Space Exploration Vehicle (SEV). This concept, as designed by NASA, is a pressurized vehicle designed to operate in a vast number of off-Earth terrestrial missions. The chassis is also detachable for an unpressurized, lighter vehicle configuration which can be driven with spacesuits. The SEV has numerous attachable tools such as cranes and bulldozer blades which will be very useful for scientific missions our crew will undertake as well as construction endeavors [10–13]. Many communications satellites and relays are already developed which can be utilized to speed up production. This is covered in greater detail in the communications portion of this proposal in Section 9.12. A large part of the habitat architecture is based upon pre-designed habitats such as concepts outlined by ESA. Our launch vehicles are all readily available as well, including Space-X's Falcon-series launch vehicles such as the Falcon 9 Expendable and Falcon Heavy Expendable. This project will utilize both existing and developing technologies to increase the use of in-situ resources and in turn minimize cost.

One of the most important aspects of Project Selene's mission is the incorporation of In-Situ Resource Utilization (ISRU). A specific portion of the project design focused on using the resources avaiable on the Moon. Due to the taxing lunar environment, there are few resources available to us. However, an example of ISRU is the use of lunar regolith for structures and tools that can be constructed on the Moon as well as water that can be synthesized by regolith gathered from specific locations. This process will be detailed more fully in the habitat architecture section 8. Since the mission is partially to demonstrate humanity's capability of colonizing other celestial bodies, the use of ISRU is integral to setting a precedent that will make colonization feasible.

4. Value System Design

4.1 Introduction

After considering the problem definition and identifying the necessary requirements, a value system design (VSD) was developed to quantifiably score the merit of design. The following factors, shown in Figure 4, were identified as driving factors for the design. Once the VSD was complete the weighting factors were determined and the numerical merit of the design was determined using the analytical hierarchical process (AHP).

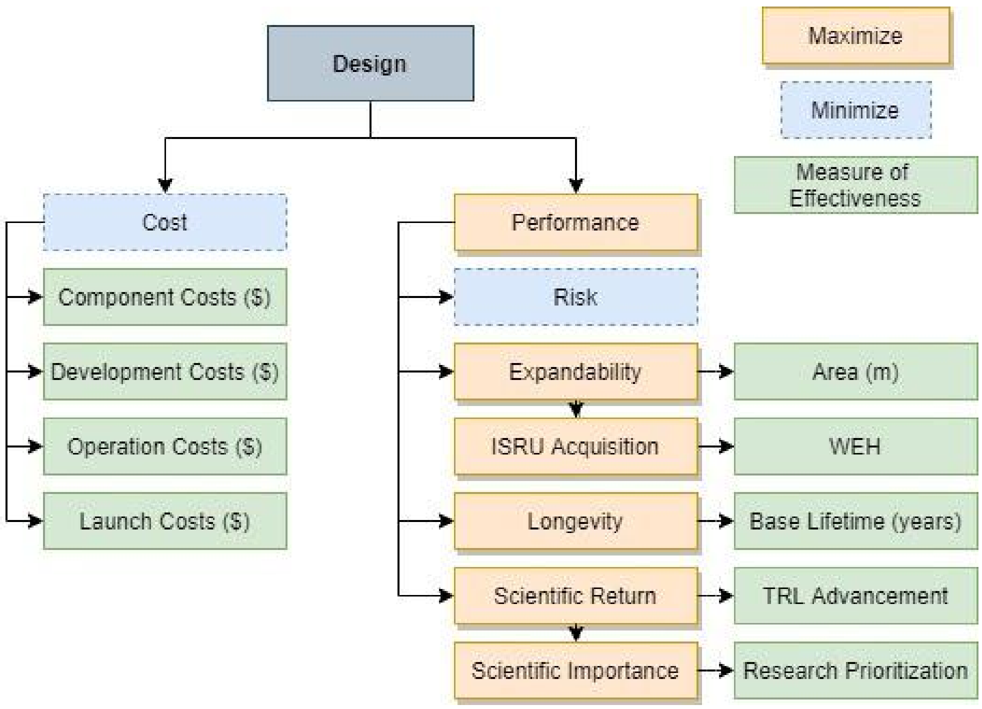

Figure 4. AHP that was used to develop factors of importance for the mission design.

4.2 Analytical Hierarchical Process

In an AHP, the factors considered are rated on relative importance when compared to each other for the mission. When comparing the factors along the rows to the factors along the columns the scores are determined using the scoring system in Table 10.

For the mission the factors considered were the longevity of the lunar habitat system, expandability of the habitat and human presence on the Moon, scientific return of the experiments performed during the mission, and the risk that an unfavorable event would occur. Longevity was used as the baseline factor for comparison. Expandability was determined to be of equal importance as longevity and received a relative score of 1. Scientific return was determined to be weakly less important than longevity and received a score of 1/3. Risk was determined to be weakly more important than longevity of the habitat due to the challenges involved with creating the first lunar base and received a score of 3.

Table 2. Definition of scores for the AHP are described above. The higher the score, the more important the factor on the left of the matrix is compared to the factor along the top of the matrix.

Score(factor (i) compared to factor (j))	Description
1/3	Option i is weakly less important than option j
1	Option i is as important as option j
3	Option i is weakly more important than option j
5	Option i is strongly more important than option j
7	Option i is very strongly more important than option j
9	Option i is absolutely more important than option j

The completed AHP matrix can be seen below:

Table 3. The scores for the AHP factors are shown above. The score compares the factor in the column to the factor in the row.

AHP	Longevity	Expandability	Scientific Return	Risk
Longevity	1	1	3	1/3
Expandability	1	1	3	1/3
Scientific Return	1/3	1/3	1	1/5
Risk	3	3	5	1

These scores were then normalized with respect to the sum of the column to find the weighting of each cell within the matrix. This matrix was then squared twice to ensure that no factors unreasonably dominated the decision process. Next the summation of the matrix and the rows were determined and the summation of the rows were normalized to the summation of the matrix, yielding the weighting factors shown below.

Table 4. Weighting percentages for each of the factors considered for the mission. These value came from the AHP and are used in the decision making process and trade studies.

Longevity	Expandability	Scientific Return	Risk
18.82%	18.82%	8.83%	53.53%

4.2.1 Longevity

The longevity of the lunar habitat system will be measured in years and a habitat unit will be determined as having expired once the maintenance cost required to sustain operations surpasses the original manufacturing and emplacement cost. Furthermore, if a habitat suffers a critical failure and needs to be replaced the habitat will be deemed as having expired when assessing the longevity of the base design. The cost for the habitat unit in 2019 fiscal year dollars for this proposed mission design is $300 M [9]. The listed cost does not account for communication systems and scientific equipment. However, when assessing the longevity of the base these associated costs would be included. The base was designed to meet the RFP goal of assisting in future exploration of space specifically focusing on missions to the Moon and Mars. The longevity of the lunar base camp will determine how long it will function as a resupply point and proving ground for technologies. For this reason longevity was included as the parameter of reference when scaling for the AHP for the initial mission.

Parameters that will extend the longevity of the lunar base camp can be used to score the longevity of the initial lunar base. These parameters include in-situ resources available for extraction, water equivalent hydrogen (WEH) reserves, and outside environmental factors. These parameters necessitate the use of ISRU technologies to supplement the resources consumed and produced via environmental controls while the base is operational. Therefore higher efficiency ISRU technologies will need testing and development for use by the launch date to maximize base longevity. This also factors into the expandability of the initial base.

4.2.2 Expandability

The expandability of the lunar base will be measured by the number of potential crew members that a given location is able to support. This includes the WEH available to be processed, the potential access to various means of power production, and the maximum number of habitats that can feasibly be deployed to the lunar surface. WEH can supplement fuel, be used in the construction of infrastructure, or be used to supplement the water supply brought to the Moon. Furthermore the available WEH is determined by the range of a Resolve rover. This was determined to be limited to 100 km for the early stage of the lunar base camp. This 100 km range was determined by the maximum range of a Chariot rover since it is the transportation vehicle that the crew will use to perform maintenance actions on any malfunctioning assets greater than 10 km away from the base camp. Power generation directly impacts the expandability of the base camp since more power will be needed to support additional habitats and the surrounding topography of the base camp will determine how many habitats can be supported. Base locations surrounded by a steep slope or large boulders will inhibit expansion. The expandability of a habitat was determined to be of equal importance to longevity since a sustained and expandable lunar presence is desired for potential colonization.

Given the RFP, the lunar base's expandability must be constructed such that future developments and construction is enabled. This requires a level of sustainability that can be achieved through deployment of ISRU technologies. Use of these technologies can supplement the (WEH) needs of the lunar base for the initial 45 day mission duration and beyond. The location of the initial base must also accommodate expansion through additional habitat units as well as the expansion of the initial communication network. For these reasons, the base expandability was reasoned to be of equal importance to the longevity given how each constraint is closely related with regards to the initial base design.

4.2.3 Scientific Return

The quality of scientific return will be measured based on the impact to the advancement of space exploration, potential for monetary gain, and utility to human health and quality of life. The potential for monetary gain is a significant factor in the expansion of space exploration and travel. By increasing the potential for monetary gain, interest in the private sector for space exploration will increase while decreasing NASA and other international partner's dependence on governmental bodies for funding, which can be unstable at times. The increased involvement of the private sector will drive innovation and the development of the space environment. Incorporating ISRU into the lunar base design will conserve monetary resources that would otherwise be spent on additional launch vehicles. As humanity continues to explore new planets and moons various resources will become more available as long as there is the capability for accessing and processing them. As such, technology must be developed to process the resources found and allow the safe transportation of astronauts.

The advancement of human health and quality of life is another major factor in determining the quality of scientific return. Many of the technological advancements made to assist in the exploration of space have led to improvements in everyday technologies such as those found in healthcare and communication systems. The exploration of space has also led to enlightening discoveries about our solar system, the workings of the universe, and has influenced the way in which we perceive our place in the universe. These discoveries enrich and aid many peoples' lives through entertainment and learning.

To access these resources we must better understand the space environment so that astronauts and explorers are kept healthy. The return on investment for such missions is large but requires that astronauts be adequately protected from radiation. This can only be achieved by understanding the dosage and effects of the radiation they are exposed to. To better understand these impacts the astronauts on the lunar base will be creating synthesized muscle tissue and organs using a 3D bioprinter. These printed samples will then be placed in environments where radiation levels are monitored and can be altered. In addition to understanding the types of radiation and the effects on the human body, the synthesis of biological material will also enable the advancement of shielding from radiation and eventually the development of detailed organ synthesis for transplant patients.

Scientific return for this mission was determined to be of weakly less importance than longevity since it will take time to yield any significant results and will not significantly impact the proposed 45 day mission. However, given the nature of the base scientific returns will remain important beyond the initial mission timeline.

4.2.4 Risk

Risk will be measured on a relative scale considering the likelihood of a detrimental event occurring and the impact that such an event would have on the mission. The complexity of a mission will in many cases increase the likelihood of a detrimental event occurring unless the complexity is introduced to mitigate the likelihood or consequence of a failure. As such the complexity of the mission was partially used to determine the associated risk. In addition to complexity certain events were identified as increasing the risk of the mission and mitigation plans were developed. The identified risks along with the plan of action associated with those risks can be found in Section 12 of the report. Since the loss of life and capital could cripple the confidence in industry and government organizations pursuing the exploration of space, risk was determined to be weakly more important than longevity seeing as a decrease in funding or public support could halt further plans of significant manned space exploration. The risks identified for this mission include the time in which assets will be deployed to the lunar surface as well as the 45 day mission operations. However, the transportation of the crew to and from the Moon was not considered as it does not fall in the scope of the mission RFP.

4.2.5 Cost

The monetary cost of the initial lunar base is based off of the costs to develop, manufacture, and launch all necessary materials and equipment to the base location, as well as the operation costs for the initial 45 day mission. Operation costs associated with the base camp after the initial 45 day mission are therefore not included within the estimate. The overall budget for the mission architecture is \$12 B 2019 USD. Ensuring that the overall cost does not exceed the given limit has determined many of the technologies chosen for development and the incorporation of ISRU technologies into the mission lifetime. Development costs were determined by analysis of the generational design, mass, quantity required, and difficulty of transport. Launch costs were reduced due to the cheap manufacturing of the chosen launch vehicle and efficient transfer of equipment to the lunar surface. Currently the overall cost amounts to \$11.54 B USD based on 2019 values.

5. Initial Concept of Operations

The purpose of this section is to give a high level overview for the concept of operations (ConOps) for Project Selene. The Falcon rockets produced by SpaceX will be the primary launch vehicles for Project Selene due to the short manufacturing time, high reliability, and low cost. This will be elaborated upon further in section 5.2 where a detailed analysis of the launch vehicles considered will be given. These rockets will then use a low energy transfer (LET) from low Earth orbit (LEO) to low lunar orbit (LLO). These maneuvers require around 3.25 km/s of ΔV and have a time of flight (TOF) of approximately 85 days. There will be a total of 10 launches consisting of one Falcon 9 and 9 Falcon Heavy rockets. The Falcon Heavy rockets will be used to launch supplies, hardware, and infrastructure to the lunar surface while the Falcon 9 rocket will transport a communications satellite to the Earth-Moon L4 point (LL4) where it will also conduct scientific experimentation. A visualization of these operations can be seen in Figure 31.

The Lunar South Pole region was selected as the regional location of the lunar base camp due to its high water equivalent hydrogen signatures, the presence of the Aitken basin which is of particular scientific interest, and the long periods of daylight and darkness that various regions experience. While the North Pole also has a large amount of WEH, the WEH is spread out over a larger area which would make collection and processing a riskier and time consuming process.

5.1 Orbital Analysis

In historical missions, most spacecraft have gone to the Moon using a direct transfer similar to a Hohmann transfer which is typically the most efficient way to transfer from one orbit to another. This is based on the assumption that this is a two-body restricted problem but there are more efficient ways to get to the Moon if other conditions are considered in the calculation. Some of these trajectories include bi-elliptical transfers, low thrust transfers, and low energy transfers. Hohmann transfers typically use approximately 4.0 km/s of Δv for insertion and injection into a circular LLO from LEO [14]. The most efficient transfer to the Moon is the low energy transfer which typically saves around 15% of the Δv needed for the transfer while bi-elliptical and low thrust transfers only save between 5-10% [3]. These savings come from calculating the transfer as a 4-Body problem between the Sun, Earth, Moon and spacecraft since all three of these bodies have a significant effect on the spacecraft.

Low energy transfers target a high apoapsis, approximately one to two million kilometers away from Earth, where the Sun has more influence on the spacecraft and uses the Sun's gravity to help raise the periapsis of the spacecraft. This allows for ballistic capture of the spacecraft in the Moon's gravity requiring less Δv for the insertion and injection around the Moon. An example of this type of transfer can be seen in Figure 1. Additional savings in Δv can come from targeting low angle approaches to the Moon so that the spacecraft can land directly without needing to circularize first. The Δv savings from this transfer allow for more supplies to be sent to the Moon in each rocket that is used. Additional advantages of this type of transfer is that there are extended launch windows and more opportunities to launch to the Moon form a LEO parking orbit as well as being able to target any orbit around the Moon. The only downside of the low energy transfers is that the transfer can take anywhere between 70 and 120 days depending on the exact transfer orbit [3].

Due to the fact that there will be no humans on board the spacecraft, this type of transfer is possible and favorable because of the large amounts of mass needed for base construction on the Moon. The deadline for the mission to be complete is over ten years away, which leaves enough time for these transfers. Since the low energy transfers require less Δv, which will save money, and stays within the required timeline a selection was made to use low energy transfers for this mission over traditional direct transfers.

5.2 Launch Vehicles

Currently there are many launch vehicles in development for travel to the Moon and other bodies the the solar system such as SLS and Starship. Neither of these launch vehicles are very far in their development and little information is known about them, so they were not considered. Table 5 compares many of the rockets that can be contracted by the public. SpaceX's Falcon Heavy delivers the most payload for the lowest cost when compared to the other available rockets. The only drawback is the limited volume for the payload, although SpaceX is currently working on a larger payload fairing. Since the Falcon Heavy will greatly reduce launch costs compared to the other rockets and because most of the payloads are mass limited instead of volume limited Falcon Heavy was selected as the launch vehicle that will be used.

Table 5. Launch cost in millions of dollars and the payload capability to GTO for different rockets available to contract to the public. Again, SLS and Starship were not included [15].

Rocket	Launch Cost $M	Payload to GTO (kg)	Payload Volume (m^3)
Falcon 9 Reusable	50	5,800	120
Falcon 9 Expendable	62	8,300	120
Falcon Heavy Reusable	90	8,000	120
Falcon Heavy Expendable	150	26,700	120
Vulcan Centaur	20,000	7,400	300
Vulcan Heavy	20,000	16,300	300
Delta IV Heavy	350	14,220	240
Atlas V	153	8,900	375
New Glenn	TBD	13,000	TBD
Ariane 6 A64	99	11,500	TBD

SpaceX has also shown their ability to produce Falcon 9 and Falcon Heavy rockets in a timely manner. They can build a Falcon 9 in under 3 weeks and a Falcon Heavy in under 12 weeks which will help the mission reach the deadlines. The Falcon 9 and Falcon Heavy are also two of the most reliable rockets available. Out of 84 launches, there have only been 3 failures which is a failure rate of about 3.5%. Of the three failures, one was a partial failure, another was a preflight failure and only one was a total in flight failure. SpaceX has also not had a failure since 2016 and their only total failure happened in 2015. The reliability and capabilities of the Falcon series rockets makes it the best selection for this mission. Additionally, SpaceX is in development of their Raptor engine which is more powerful and efficient than the Merlin engine which is currently used on the Falcon 9 and Falcon Heavy. Elon Musk has stated that this engine will primarily be used for Starship, but that it will also be retrofitted onto the Falcon Heavy which will increase the payload capabilities of this rocket [16]. With the Raptor engine, the Falcon Heavy will be able to deliver over 10,000 kg of payload to the Lunar surface shown in the calculations below [17].

$$\Delta v = Isp * g * log\frac{m_i}{m_f} \tag{1}$$

Calculating the Δv of the rocket is done using the ideal rocket equation shown above where m_i is the initial mass of the rocket, m_f is the mass after burnout of the rocket, Isp is the specific impulse of the rocket, and g is the gravitational acceleration on Earth. This calculation is done for both stages of the rocket, assuming that the rocket has been fitted with Raptor engines, where the ISP is 355 seconds for stage one (averaged between sea level and vacuum) and 382 seconds for stage 2 [18]. Next, the Δv to land on the Lunar surface is needed. To get into LEO at a 180 km circular orbit approximately 9.2 km/s of Δv which will be Δv_1. [14] To get into a higher more stable orbit, a Hohmann transfer is used [19].

$$\Delta v_2 = \sqrt{2(\frac{\mu_e}{r_1} - \frac{\mu_e}{r_1 + r_2})} - \sqrt{\frac{\mu_e}{r_1}} \tag{2}$$

$$\Delta v_3 = \sqrt{\frac{\mu_e}{r_2}} - \sqrt{2(\frac{\mu_e}{r_2} - \frac{\mu_e}{r_1 + r_2})} \tag{3}$$

In the Hohmann calculation using equations 2 and 3, μ_e is the gravitational constant times the mass of Earth, r_1 is the initial radius of the orbit which would be 180 km plus the radius of the Earth r_e, and r_2 is the final radius which is 400 km plus re. From plugging in these numbers, the Δv needed to get into a 400 km circular orbit above Earth from a 180 km circular orbit is 0.125 km/s. Adding this to the velocity needed to get into LEO the Δv needed to reach this orbit is 9.325 km/s. Next, the spacecraft needs to go into a trans-lunar injection. The exact Δv varies depending on the exact transfer orbit, but the average value is around 3.2 km/s to insert into a 100 km circular orbit around the Moon [3]. The calculations for these transfers are very complex and hard to model, so they will not be written out. Finally, the Δv for landing on the Lunar surface is needed which is calculated below [20].

$$\Delta v_{L1} = \sqrt{\frac{\mu_m}{r_{LLO}}} * [\sqrt{2(\frac{r_{LLO}}{100 + 2 * r_m})} - 1] \tag{4}$$

$$\Delta v_{L2} = \sqrt{\frac{\mu_m}{r_m}} * [1 - \sqrt{2(\frac{r_m}{100 + 2 * r_m}}] \tag{5}$$

$$v_{moon} = \sqrt{\frac{\mu_m}{r_m}} \tag{6}$$

From adding the values from the equations above the Δv for landing on the lunar surface from LLO can be calculated. In the equations above r_{LLO} is the radius the spacecraft is at when in low lunar orbit of 100 km and μm is the gravitation constant times the mass of the Moon. The first and second equation which calculate Δv_{L1} and Δv_{L2} is effectively a Hohmann transfer to put the spacecraft on a collision course with the surface of the moon. The last equation for moon is the velocity on the surface of the Moon. Together these three velocity changes makes Δv_5. The total velocity needed to get from the surface of Earth to the surface of the Moon is found by summing all five of the changes if velocity.

$$\Delta vtotal = \Delta v_1 + \Delta v_2 + \Delta v_3 + \Delta v_4 + \Delta v_5 \tag{7}$$

Putting in the values for the equation above results in a total Δv of 14.25 km/s. By setting equation 7 equal to equation 1, the maximum payload can be found for a Falcon Heavy Rocket. While the maximum calculated payload was just under 13,000 kg, as an added safety precaution maximum payload was reduced to 10,000 kg.

6. Habitat Location

6.1 Introduction

The location of the lunar habitat is the second of the three key trades identified as driving the conceptual design of a lunar base established by 2031 [8]. Preliminary analysis determined that the density of ice water and WEH was higher in the lunar south pole than in the lunar north pole while the north pole was predicted to have more overall ice water and WEH [21]. To mitigate the distance that rovers and astronauts would have to travel to gather ISRU for water the south pole was selected to be the region in which the lunar base would be developed. Next, several craters in the area were determined to be the locations of large amounts of ice water and WEH. The craters identified were Shackleton crater, Sverdrup crater, Shoemaker crater, Haworth crater, and Cabeus crater [22, 23]. Shackleton and Sverdrup craters were eliminated early on in the selection process because they exceeded the maximum range of 100 km of the Chariot rover operating on one charge [11, 13]. Another consideration was the complexity and risk associated with attempting maintenance on the Malapert Mons relay if the base was located at either of these sites. They both were eliminated them from consideration since additional stop off bases would have to be established to allow for maintenance actions. The specifics for why Malapert Mons was chosen as a relay location is described in more depth in Section 9.8. The lengthy travel distance coupled with the lower WEH concentration made it apparent that neither Shackleton nor Sverdrup craters would be the optimal choice for a base camp. The remaining craters in addition to a few points located between the craters were then used as part of the location trade study. The factors considered were the terrain surrounding the ISRU and habitat locations, ISRU proximity, risk, environmental factors, and the value of potential science experiments. These factors were weighted using the AHP and the result of the trade study determined that the ridgeline between Malapert Mons, Cabeus crater, and Haworth Crater was the best choice as seen in Table 6. For reference the location at which data was gathered was -86.35 degrees latitude and of -23.19 degrees longitude [6].

6.2 Region Selection

The three regions of the lunar surface considered for habitat emplacement were the equatorial region of the Moon as well as both the North and South poles. The equatorial region offers little in the way of due to long periods of exposure to sunlight and also experiences significant periods during which DTE communication isn't possible. Furthermore, the equatorial region of the Moon has fewer craters dating back to the early formation of the solar system and as such offers less significant insight.

The North pole region of the Moon is projected to contain the largest amount of WEH. However, the WEH located at the North pole is in low concentration and spread out over vast distances. As such, rovers and crew members would be exposed to an increase in risk failures occurring over the extended EVA reduced this location's desirability.

The South pole region of the Moon has a large amount of WEH, albeit less than the North pole, in high concentrations making these locations ideal for setting up a base nearby. In addition to the highly concentrated WEH the Aitkenson

Basin is located in the South pole which contains many craters projected to date back to the pre-Nectarian era of the Moon's geology which corresponds to the time frame in which the early formation of our solar system was forming. These considerations combined with the DTE communications that are able to be established through the use of Malapert Mons makes the South pole the ideal region to establish a lunar base camp.

6.3 Locations of Interest

Within the South pole region several locations were identified as being potential candidates due to their accessibility, WEH concentration, proximity to Malapert Mons, environmental factors, and potential for scientific return. Accessibility included the distances and slopes needed to be transversed to access areas of high scientific interest or WEH density. Accessibility and the environmental factors, such as temperature variation, combined made up the longevity score when the merit of the design was related back to the AHP. The concentration of WEH factored into how many crew members could be accommodated and in turn related back to expandability. The significance of the scientific return will be based on the access to various types of regolith, specifically that which dates back to the early formation of the solar system. Finally, risk is determined by the distance from Malapert Mons and the complexity of the associated architecture. Since Malapert Mons is an optimal location for a primary relay system it was determined early on in the design process that this would be a key location and the location of the primary terminal. Given these parameters the craters identified were Shackleton crater, Sverdrup crater, Shoemaker crater, Haworth crater and Cabeus crater.

6.4 Eliminated Locations

Shackleton crater was identified due to its high WEH density when compared to the surrounding region and due to the fact that the ridge line of Shackleton receives near constant light from the Sun. This made it an ideal location for the use of solar panels since many other regions of the lunar south pole located near craters can experience darkness for 55-100 hours [6]. There were, however, concerns surrounding the method in which the habitat would be constructed and path rovers would take to collect the ISRU from inside the crater due to its steep slopes. As such the habitat would likely have to be built outside of the rim of the crater and cables would run the electricity to the habitat from the solar panels on the rim of the crater. Sverdrup crater was found to have less WEH density, was lower in altitude and as such received less sunlight. However, the terrain around Sverdrup was found to have a much more gradual slope than that of Shackleton [6, 24].

After the communications system was determined both sites were eliminated due to the DTE limitations in addition to poor mission synthesis which would either increase the risk or the complexity associated with Project Selene. The range for a Chariot rover is 100 km on one full charge while the distance to Shackleton crater and Sverdrup crater is 119 km and 163 km respectively. This would mean that to make repairs another habitat would have to be established to shelter the astronauts on their journey. This would drastically increase the complexity of the design since the extra habitat would need to be stocked with additional supplies and have maintenance and inspection actions conducted before a mission to Malapert could be conducted.

6.5 Remaining Locations

The remaining craters where Cabeus, Haworth and Shoemaker craters. All of these locations were within 80 km of the communications relay on Malapert Mons, had high WEH density, and inclines of less than 30 degrees. Midpoints between these craters were also considered in order to optimize IRSU and the distances to each of the mid-point locations were found along with the largest slope to those points.

Cabeus crater has the highest WEH density in the lunar south pole at 0.55 wt% [6]. Cabeus crater was also associated with past NASA Missions and was the subject of several experiment conducted to further detail material composition and abundance [25]. The temperature variation at the Cabeus habitat location is average for the lunar south pole, has a distance of 43 km to the highest water density, and has the longest distance to Malapert with 80.4 km. In addition to Cabeus crater's high WEH, it is also one of the largest "craters" in the south pole. Geological scans have shown that the crater is composed of several smaller craters and one large crater. The craters created by these impacts provide great sites for WEH due to the low amount of sunlight they receive. One concern with Cabeus crater is related with visibility for the Malapert relay. A large mountain range exists between Cabeus and Malapert which would disrupt communications. If a base was placed near Cabeus crater a large antenna or additional communications relay would be required. Both of the prior solutions increase the complexity associated with this configuration. Shoemaker crater is slightly larger than Haworth but significantly smaller than Cabeus and as such both Shoemaker and Haworth would have less WEH than Cabeus. The specifics for each habitat location ar e given in the following Figure where the values for the midpoint locations are broken up by 1^{st} crater, 2^{nd} crater.

Table 6. Statistics of the remaining lunar south pole craters used in determining habitat location [6]

Crater	Distance to WEH (km)	Distance to Malapert (km)	Summer Max Temperature (K)	Winter Min Temperature (K)	Max Temperature Variation (K)	Max Slope to WEH (degrees)	WEH Density (wt%)
Cabeus	43	80.4	260	80	185	20.9	0.55
Haworth	17.2	61	230	50	180	28.1	0.44
Shoemaker	14.6	63.4	200	30	170	23.6	0.5
Haworth & Shoemaker	20.0, 20.1	56.8	180	30	150	18.5, 27.8	0.44, 0.5
Cabeus & Haworth	69.4, 36.1	55.6	265	55	210	21.8, 21.3	0.55, 0.44

These specifications were used to determine scores for the various craters. The distance to the WEH, distance to Malapert Mons, Max temperature variation, and max slope to the WEH are values than need to be minimized and as such were scored by making a vector of their inverse, summing the vector, and then normalizing the vector to its sum. Since WEH Density in wt% is desirable, the same process was done excluding inverting the values.

The weighting factors were determined using the AHP of relative scoring. The baseline factor was the ease of WEH acquisition and as such was given a relative score of 1. The density to WEH when compared to the ease of acquisition of WEH was given a relative score of 3. ISRU is key to the creation of a sustainable and expandable presence on the lunar surface by cutting down launch costs and enabling greater manufacturing on the lunar surface. As such a large presence would extend the longevity of the base and allow accommodation of a larger crew. Furthermore, technology could be improved to allow for greater ease of WEH and ISRU acquisition more easily than relocating a base of operations. The importance of the risk, when compared to the ease of ISRU acquisition, was determined to be 5 since loss of assets or crew could drastically reduce the productivity of the mission, prevent the goals of the mission from being achieved, and cause a loss in public support for further space exploration. The environment of the chosen habitat location was based on the temperature variation between the summer maximum and the winter minimum and was given a relative weighting when compared to the ease of WEH acquisition of 1/3. While the environmental conditions such as temperature variation could expose the habitat to increased wear and tear the temperature variation in the lunar South pole region is largely uniform and will not have large impact on the location of the habitat. The importance/impact that science goals had on the choice of location for the base camp was slightly less when compared to longevity. The low impact of science on the location was primarily due to the fact that all of the considered locations are regions of interest and are expected to have similar regolith composition and origin.

The eventual weighting of risk was 55.2% which is close to the 53.52% that was determined in the AHP. The ISRU ease of acquisition and environmental considerations were determined to have a weighting factor of 10.6% and 6.1% respectively and determine the longevity of the base which made up 18.83% of the AHP. The ISRU density was linked to the expandability of the habitat made up 21.9% of the location trade study which is close to the 18.83% that was determined in the AHP. Finally, the science return was 6.1% of the location trade study weighting while it was 8.82% in the AHP weighting.

Table 7. Comparison between the AHP factors of merit and the location trade study's weighting factor's expressed in terms of the factors of merit.

	AHP (%)	**Location Trade Study (%)**
Longevity	18.83	16.7
Expandability	18.83	21.9
Science Return	8.82	6.2
Risk	53.52	55.2

6.6 Chosen Location

Using the weighting factors previously discussed and the values found for each location a trade study was conducted to determine the best choice between the sites considered.

The sources of WEH and sample return for the selected location will be Cabeus and Haworth craters. The details of

Table 8. The resulting out puts of the trade study for each considered location and factor. The total score can be seen in the final column. Based on this trade study it can be seen that the ridge line between Cabeus and Haworth carers is the best option for a lunar base camp.

	ISRU Ease of Acquisition	Risk	Environmental	ISRU Density	Science Goals	Totals
Weighting Factor	10.6	55.2	6.1	21.9	6.1	–
Cabeus	0.12	0.14	0.19	0.16	0.15	14.79
Shoemaker	0.19	0.14	0.21	0.15	0.05	14.7
Haworth	0.16	0.14	0.20	0.13	0.08	13.93
Cabeus and Haworth	0.22	0.36	0.17	0.29	0.63	33.54
Haworth and Shoemaker	0.32	0.21	0.24	0.27	0.08	23.04

the location can be seen below.

Table 9. The environmental, WEH, slopes, and distances associated with placing the lunar base on the ridge line between Cabeus and Haworth craters, Lat. -86.3, Long. -23.3 [6]. Were regolith is classified as Nectarian or from pre-Nectarian terra.

	Base Location	Base to Cabeus	Base to Haworth	Base to Malapert Mons
Temperature Change (K)	210	50	80	170
Slope to WEH (degrees)	2.55	21.28	21.89	8.51
Distance to WEH (km)	0	69.42	36.12	52.01
WEH Density (wt%)	0.25	0.55	0.44	0.32
Regolith Classification	NpNt	pNbm	IpNt	NpNt

7. Operations

7.1 Introduction

The daily operations of crew and semi-autonomous rovers will follow a prescribed set of prioritized goals and actions scheduled over the mission duration. These actions include, but are not limited to, construction of the lunar base camp, construction of supporting infrastructure, performance of scientific experimentation, collection of lunar samples and resources, maintenance of the lunar base systems, and outreach to educational institutions and the public to promote awareness for the exploration of space. The Helelani rover will construct a landing pad for the delivery of supplies from upper stage Falcon rockets to the surface. To ensure reliable descent, the tiles ATHLETE makes and landing pad Helelani constructs will have embedded sensors, enabling communications with the Falcon rockets and communication systems in place. The Cratos rovers will assist in the construction and maintenance of the lunar base and its constituents by polymerizing the lunar regolith with PTFE. Such structures include a shielding dome for the habitats deployed to the lunar surface as well as ramps and “roads” that lead from the landing pad to the lunar base which will aid in the transportation of hardware and supplies. The Resolve rovers will collect regolith for the primary purpose of ISRU acquisition and to allow for the gathering of lunar regolith samples that are beyond the operational range of the crew which can be analyzed on the lunar surface or returned to Earth. Crewed actions will include EVAs up to 100 km away from the lunar base camp during lunar day cycles for the purpose of lunar base camp maintenance and scientific experimentation. The science goals include solar wind analysis, radiation exposure, and lunar sample return among others. For EVAs further than 10 km the crew shall use a Chariot rover which will serve as the transportation vehicle on the lunar surface. The Chariot rover will also be equipped with a small pressurized rover (SPR) which can maintain livable conditions for up to 4 days, comes with deployable solar panels for charging purposes, and has a volume that can accommodate 2 crew members comfortably for 4 days. Their range for one charge is 100 km and travel at approximately 5 km per hour. This section will detail the requirements and assumptions associated with mission operations as well as the science experiment that will be conducted and the infrastructure that will be deployed to the lunar surface.

7.2 Requirements

The requirements laid out in this section will describe the inherent and derived requirements for rover and crew operations. While all requirements and standards will be met those listed below were deemed to be of specific concern.

Nominal Metabolic Intake

<u>EER for men 19 years old and older</u>

EER (kcal/day) = 622 – 9.53 × Age [y] + 1.25 × (15.9 × Body Mass [kg] + 539.6 × Height [m])

<u>EER for women 19 years old and older</u>

EER = 354 – 6.91 × Age [y] + 1.25 × (9.36 × Body Mass [kg] + 726 × Height [m])

Figure 5. Equations showing the caloric intake for men and women based on age, height and weight [2]

7.3 Assumptions

The following section outlines the assumptions made that most directly impact the mission operations.

1) The average EVA duration will consist of 10 hours unless travel via the Chariot rovers is required. This is consistent with NASA-STD-3001 Vol.2 Sec. 6.3.2.7.
2) Chariot rovers dry mass includes solar panels that will be able to recharge the vehicle within 12 hours during any point within the lunar day. In the case that additional solar panels are necessary to achieve this recharge time the return mass will be decreased to accommodate the difference in vehicle dry mass.
3) The Resolve rovers will be equipped with solar panels that will enable recharging at any point during the lunar day.
4) Decreasing the mass of the landing legs that will be installed on the upper stage of the Falcon rockets results in a decrease of max stress by half.
5) Maximum of 4000 kg of propellant in the upper stage of a Falcon Heavy when the rocket reaches lunar orbit.

Table 10. Mission Inherent and Derived Requirement along with reference

Inherent Requirements			
Requirement Name	**Description**	**Requirement Source Number**	**Implementation**
Caloric Intake	The caloric intake ***shall*** follow the equations in Table 9 of NASA-STD Vol.2 with an activity factor of 1.25	NASA-STD-3001 Vol.2 Sec. 7.1.1.3	See Figure 5 for a breakdown of the caloric intake.
Caloric Intake per hour of EVA	200 kcal ***shall*** be provided for each extra hour of EVA per person.	NASA-STD-3001 Vol.2 Sec. 7.1.1.4	See Figure 5 for a breakdown of the caloric intake.
Physiological Counter Measures	The Physiological health of the crew ***shall*** be maintained through an exercise schedule and proper care.	NASA-STD-3001 Vol.2 Sec. 7.4	See Section 8 for the breakdown of exercise equipment.
Level of medical attention given mission	The medical care available to the crew ***shall*** adhere to the standards laid out in Table **??**	NASA-STD- Vol.2 Sec. 7.5.1 [2]	This will be explored in greater detail in further design iterations.
Suited Operations Water Quantity	During EVAs there ***shall*** be 240 ml or 8oz of water per hour of EAV. It can be assumed that an EAV will last 10 hours on the lunar surface.	NASA STD-3001 Vol.2 Sec. 6.3.2.7	Accounted for in the mass budget Table 42 which shows a breakdown of water quantity for the mission.
Habitat Maintenance	All maintenance checks shall be scheduled and able to be conducted in a timely manner.	NASA STD-3001 Vol.2 Sec. 9.7.2.1	See Section 7 for a description of maintenance actions.
Assembly Time	The habitat shall be constructed within 72 hours of crew deployment to the lunar surface.	AIAA RFP	Refer to Section 8 for estimated set up time based on the systems chosen.
Applicable to Future Missions	The processes and technology shall aid in the development of science with a focus on space exploration.	AIAA RFP	See Section7 for a description of the science experiments that will be conducted.

7.4 Scientific Experimentation

Many experiments have been conducted which look into the use of extra-terrestrial regolith as a building material for habitat construction. If this reaches full maturity in the sense that the synthesized structure would be able to maintain an atmosphere this could yield significant mass savings. Many approaches have been considered ranging from fusing the sulfur found in lunar regolith, using elements of concrete as a binding agent, or polymerization of the lunar regolith. Thus far, no approach has yielded a small enough porosity to result in a structure able to self-contain an atmosphere. Furthermore, the experiments conducted to fuse the sulfur using Xenon lights focused with mirrors, found that many of the samples had uneven heating which caused peeling of the layers. Additionally the size of the instruments required, such as a 52 m^2 mirror, made practical use impossible at this time [26]. Other approaches, such as using elements of cement mixtures, have masses that would be so large that transportation to the lunar surface would be impractical for broad use such as the construction of a domed wall. However, the potential to reduce the mass of habitat structures could be a significant advancement in the exploration of space. Polyethylene is a lightweight polymer that has great potential for low mass radiation shielding. As such, several polymers were considered for selection as the binding agent that would be used in the Cratos rovers to construct the habitat's protective shell. The first step was to determine a stress to density ratio that would be needed for the structure to be self-supporting. To begin narrowing down the search for materials a stress to density ratio was found assuming a 1 m thickness for a structure 8 m long and 5 m wide. The yield stress to density ratio was determined to be 4.623 J/kg. Using the material database software "CES EduPack" [27] the yield stress and density of Butyl Rubber was found and the minimum thickness was determined to be around 1.73 mm if the structure had been completely built out of the Butyl Rubber. These were established before going in depth into the material selection to serve as sanity checks since few materials were unable to fulfill these requirements. Next was to find a material that could operate in the temperature ranges that would be experienced during the mission. For the location selected, the winter minimum was determined to be 55 K (-218.15 C) and the summer maximum temperature was 265 K (-8.15 C). Thus, the max temperature difference between seasons is 210 K. This operational range drastically reduced the materials that could be used for such a mission. As illustrated in Figure 6 and 7, the only two polymers with such a wide range of temperature ranges were PEEK 6 and 7 these only PTFE had low enough operating temperatures to be used on the Moon. PTFE is also resilient to UV radiation exposure which after the visible spectrum is the wavelength of light that the Sun gives off the most. For these reasons PTFE was chosen as the binding material for lunar infrastructure construction.

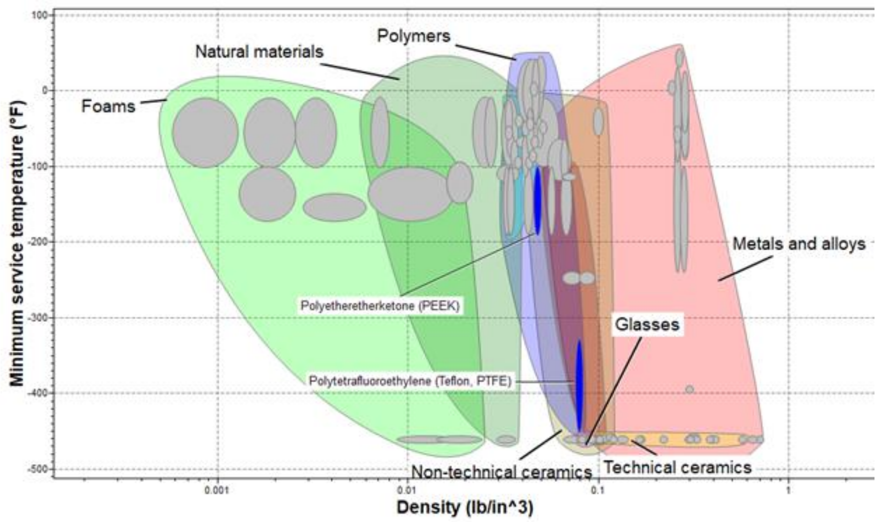

Figure 6. CESEduPack software showing the temperature vs density of various materials. This information was used for material selection of the polymerizing material.

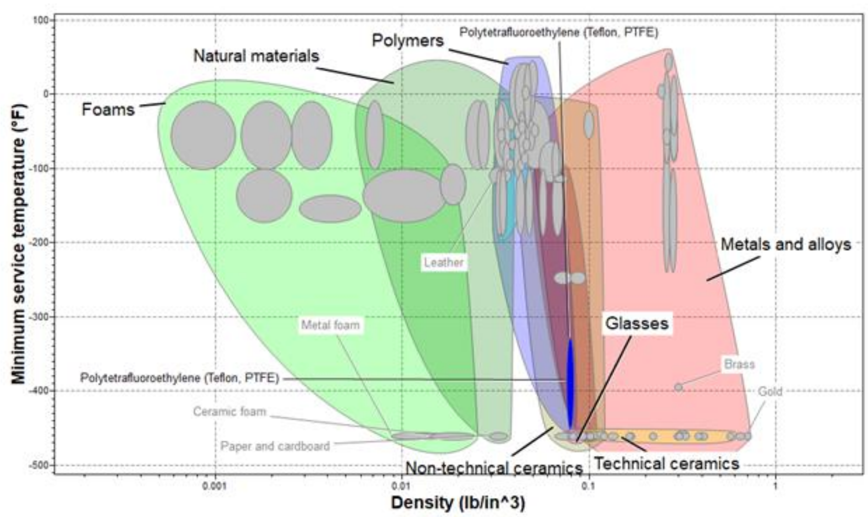

Figure 7. CESEduPack software showing the temperature vs density of various materials. This information was used for material selection of the polymerizing material.

The properties for lunar regolith were modeled after lunar highland simulant 1 (LHS-1) which is synthesized to be comparable to the regolith that will be present near the lunar base camp. An in-house code was developed using MATLAB to approximate the actual stresses and thermal expansion that will be seen on the Lunar surface. Virginia Tech associate professor Dr.Gary Seidel recommended that approximations of physical properties could be determined through the law of mixtures while the inverse law of mixtures should be used to determine thermal properties. Mixture ratios were incremented with a step size of 10%, in order to calculate for the minimum required wall thickness. The max stress seen during the mission was determined by summing the stress associated with a Cratos rover's weight on the structure, the max thermal stress due to temperature variation, and the stress due to weight of the structure itself.self-weight. Results from the code concluded that mixtures as low as 10% PTFE with a thickness of 1 mm could withstand the max stresses seen with a factor of safety of 1.2. A mixture ratio of 10% PTFE with a thickness of 1mm was decided upon due to the large increase in polymerizing mass needed to increase the thickness further and since a thickness of 1mm was sufficient for the expected loads. The goal of this project was to verify the simulated approximations with experimental data collected from lab experiments in several Virginia Tech facilities.

However, due to the restrictions put in place to control the spread of COVID-19 access to Virginia Tech facilities was revoked and the experiments were unable to be conducted. The experiments that would have been conducted on mixture samples would have included:

- dog-bone tensile stress test
- compressive strength tests
- Charpy impact tests
- thermal conductivity tests
- cyclic loading tests

The data collected would have been used to update the approximations in the code, such as thermal expansion coefficient and thermal conductivity. The test results would have been compared with the predicted results to better improve other material properties and to develop a higher performing structure. These experiments would also help size the dimensions of the shell to protect the habitat from micrometeoroid impacts, large temperature fluctuations, and would supplement the habitat's radiation shielding capabilities.

A satellite will be sent to the Earth-Moon L4 point (LL4) to serve as part of the communication relay and to perform relative science at this location. The LL4 point is a region of gravitational stability and therefore an area for dust particles to collect. The dust is of interest because of the insight it could provide into the formation of the Earth, Moon, and our solar system. Only high energy collisions between asteroids and the Earth or Moon could propel dust to reach the LL4 point. Many of the high energy impacts to the Moon occured early in the solar system's development so it is believed that much of the regolith found in this region would be from the pre-Nectarian to Imbrian time periods. To perform analysis on the dust particle composition a Venitia-Burny-Student Dust Counter (VBSDC) will be housed on the satellite [28]. If the dust in this region yields information about the formation of the Earth and Moon it could allow geological missions on the surface of the Moon to be better prepared and more developed. For this mission a lunar communication network is being developed to assist with the need for reliable constant communication and to handle the large amount of scientific data that will be collected. The use of optical communication in addition to RF will meet the growing demands for faster data rates and be able to handle the large amount of data that will need to be transmitted back to Earth. Data transmission will be sent using light as the medium through highly reliable infrared lasers. Light waves are much shorter than radio waves and allow data to be transmitted across narrower beams, which directly reduces the amount of possible loss. The laser from the optical terminal points directly to the recipient ground

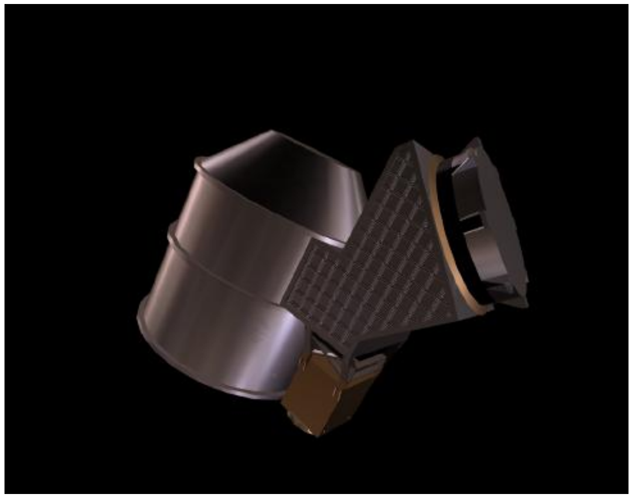

Figure 8. Cosmic Dust Analyzer that was used on the Cassini mission, that will be placed on the LL4 communication satellite [28].

station telescope with extreme accuracy and precision, thus reducing the need for a large antenna as compared to RF. Having optical capabilities will help towards setting up the foundation for better communications for future manned missions reducing the risks.

As discussed in Section 6 the ridgeline between Cabeus and Haworth craters also offers significant opportunities for scientific return due to its proximity two both craters. Both Haworth and Cabeus craters are in the Aitken Basin which accounts for craters that are from the pre-Nectarian period which corresponds to 3.92-4.533 GYA. This time period is relatively early when considering the formation of our solar system and is of great interest to geologists and astronomers since sample returns from this period could provide insight into the early formation of the Earth and its relationship with the formation of the Moon and asteroid belt [29–31]. Regolith samples from Cabeus and Haworth craters could be compared to find the differences and similarities in regolith composition. Missions to both of these sites will be conducted and sample return will be established based on the capabilities of the lunar lander and the projected needs of the ground-based lab. To access the regolith from the pre-Nectarian period the top 3-5 meters of regolith will have to be excavated since the average depth of "top soil", three meters in mare regions and up to five meters in the highland regions, will contain regolith from recent meteoroid impacts. This will be done using the various attachments, such as a plow, that are compatible with the Chariot/SEV rover.

While conducting EVAs the Astronauts will also make stops at pre-selected locations to deploy updated versions of the passive seismic experiment that was conducted during the Apollo 14 mission. The improvements to this experiment are sensitivity at a given range and the duration for which the probe can be deployed since the Apollo 14 version had comparatively poor sensitivity and over five year recorded a handful of significant impacts [32]. When conducting EVA's through the use of the Chariot rovers, data will be collected monitoring the radiation of the lunar surface using a battery-operated independent radiation detector (BIRD) which interfaces with the vehicle's telemetry system to provide detailed radiation maps. This will further aid in the understanding of the lunar environment since radiation is poorly understood and modeled in the lunar environment. Forty-eight radiation area monitors (RAMs) will also be used throughout the three habitat modules to monitor the intensity and wavelength of radiation that permeates through the base. These RAM sensors will determine the effectiveness of the habitat walls and shell at blocking various classifications of radiation and will also assist in the prolonged human exposure to space environments (PHESE) experiment.

The lunar habitat will have a section designed to study regenerative medicine and additive manufacturing. Bioprinters will be used to study the effects of radiation on human muscle tissue, myocytes, by printing tissue samples. There is an advantage to printing away from earth because the printing material, bioink, is normally composed from a combination of stem cells and support material. The support material acts similar to a scaffolding but is thick and viscous and often hinders the rate of printing or increases stress on myocytes causing improper growth. The Moon's gravity is one-sixth that of Earth's and provides an environment that is ideal for bioprinting because the support material is not needed. This allows the bioink solution to be less viscous and enables printers to print with a more fine tip, creating precise and detailed structures [33]. These low viscosity inks have not been extensively tested as their Earth based counterparts, and during these missions inks of varying low viscosities will be tested for endurance, strength, and longevity [34]. A company named Techshot, at the request of NASA, has designed a zero gravity bioprinter, the Biofabrication Facility (BFF). The BFF is currently being tested on the International Space Station, shown in Figure 9.

Figure 9. Biofabrification Facility onboard the ISS, a similar one will be used on at the lunar habitat to print tissue samples in various conditions.

The astronauts at the lunar base will continue these tests and return the samples to Earth once their mission concludes [35]. The TRL of this process can increase from a 7 to a 9 only through extensive research and testing. These tests will be performed with the future goal of printing complete organs in space for patients waiting for organ transplants, bioink can be composed of their own stem cells, thus preventing any chance of the body rejecting the implant. As future space missions shift toward further exploration, it is important to create Earth-independent systems. The risk of injury when performing such missions always exists but, by creating bio-printing systems such as these we can better prepare astronauts. Another major consideration is the risk of a solar particle event and the issues that can happen from exposure to radiation. To study the effect of radiation on muscle tissue there will be a specialized chamber designed for radiation testing. Similar to the phantom torso experiment performed on the ISS, astronauts will take printed tissue samples from the bioprinter and assemble a "torso" with radiation detectors embedded within. The detectors will monitor the way radiation spreads through muscle tissue, and provide a safe way to study the effects of radiation without harm to the astronauts. The ability to sample the radiation within the tissue better predicts how radiation moves through the body [36]. For this reason this scientific mission has been named PHESES. Experiments and scientific missions such as this one serve as the precursor to human exploration missions and are essential to human health and safety.

Around the vicinity of the lunar base camp an experiment analyzing solar wind will also be conducted. This experiment will analyze the solar wind composition as a function of the Moon's location to Earth's magnetosphere. This solar wind analyzer will be deployed when the Moon is in the tail of the Earth's magnetosphere, exiting the Earth's magnetosphere, outside of the Earth's magnetosphere, and entering the Earth's magnetosphere. Due to the duration of the mission this experiment can be repeated twice. Potential sensors include a solar wind spectrometer [32] and a foil stand which would be returned to Earth for further analysis. The sensor is very accurate because it measures both the composition and velocity of particles in the solar wind by capturing the molecules when they are embedded into the foil [36].

7.5 Infrastructure

7.5.1 Needs

Traditionally, supplies sent to the Moon is put on a Lunar lander which typically separates from the launch vehicle either in geostationary orbit (GEO) or in (LLO). From here, the lander delivers the payload to the surface with a soft landing. Unfortunately, the cost for doing this is very expensive and historically has only been done with small payloads typically under a few thousand kilograms. Most companies predict the cost for Lunar landers to be around $1.2 M per kg of payload [37]. This mission is predicting to send around 75,000 kg of mass to the Lunar surface with the largest single payload around 9,500 kg. In total, the costs of this would be close to $90 B just to launch the materials to the moon. Since this exceeds the required $12 B budget, another solution would need to be found. The solution that was developed was to land the upper stage of the launch vehicle on the lunar surface, which in this case is the upper stage of a Falcon Heavy. If this solution was used, then the cost for landing payload on the surface would be reduced to $0.017 M per kg of payload. This value comes from the maximum payload solved for in Section 5.2 of 10,000 kg and the cost of a Falcon Heavy being $150 M per launch [5]. Since the Falcon Heavy is not designed for this type of operation, some modifications would need to be made and some supporting systems would need to be established on the Lunar surface. First, the Falcon Heavy currently uses the Merlin 1D engine which is fueled by LOX and chilled RP-1 which could lead to RP-1 boil off during a long flight [5]. A solution to this is already in development in the

Raptor engine from SpaceX which is fueled by LOX and liquid methane [18]. Both this fuel and oxidizer already have solutions to help prevent boil off and since the spacecraft should remain under the boiling temperature for both liquids by coating the tanks with a material called Solar White [38]. This engine is also being designed for the Starship and Big Falcon Rocket which are meant for long interplanetary travel along with delivering payload to other celestial bodies [18]. Additionally, the Raptor engine uses spark igniters with a duel redundant system which eliminate the pyrophoric mixture of triethylaluminum-triethylborane for repeated ignitions. Elon Musk has also stated that this engine can and will be retro fit onto Falcon Heavy rockets and SpaceX was also contracted to test this by the Air Force [16]. Next, the power supply for the rockets control systems would not last long enough to survive the transfer orbit, A possible solution to this problem would be to put a small solar array on the rocket that can deploy once the rocket reaches orbit. Next, landing legs would need to be added to the upper stage of the Falcon Heavy. SpaceX already has a solution for this problem as well because SpaceX has been landing and reusing their rockets for years. The same landing legs used for the first stage of the Falcon 9 can be used for this system. More research and development would need to go into preparing the Falcon Heavy for this long journey, the development and testing cost for these changed was calculated from the Johnson Space Center development reference to be approximately to be $1.6 B. Finally, a few other systems will need to be placed on the Lunar surface to support the landing of the launch vehicles.

7.5.2 Ground Systems

To help with landing the Falcon Heavy upper stages, a few different systems will need to be in place on the Lunar surface. First, a flat area will need to be made for the rocket to land on. Next, a rigid landing pad will need to be placed on the surface to support the weight of the rocket and prevent regolith from flying away and damaging nearby systems. Then, there needs to be a crane that can unload the rocket's payload which will sit over 12 m high. Next, there needs to be a rover that can transport the payload from the landing pad to the habitat location. Finally, a rover will also be needed to move the empty rocket off of the landing pad so the landing pad can be used again. This procedure can be seen in Figure 10. All of these systems will need to be controlled remotely since the habitat needs to be completed before the crew arrives.

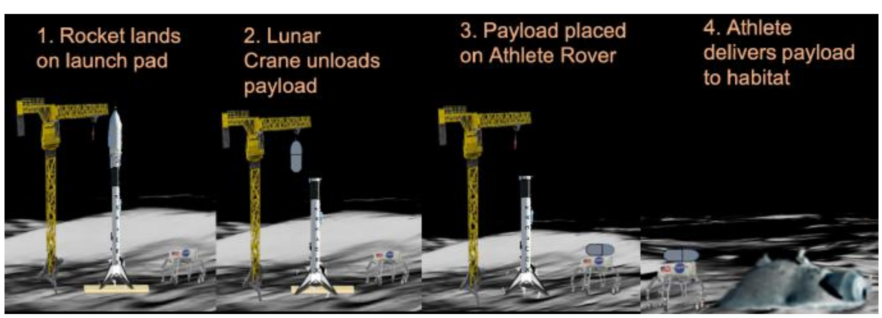

Figure 10. Steps 1 through 4 demonstrates how the lunar crane will transfer the payload from the rocket to the Athlete rover. The rover will then transport the payload to the correct location.

7.5.3 Communication

The rovers on the lunar surface will need to operate for years before the arrival of humans in 2031 to set up the infrastructure needed on the Moon. To help with their operation and autonomous control from Earth a communication system will be established. In 2023, a satellite will be launched to the Lunar Lagrange 4 point between the Moon-Earth system. This satellite will have communication capabilities and will have full view of the habitat. Next, a three satellite constellation will be placed in a low frozen Lunar orbit in 2026 which will constant coverage of the Lunar base. Finally, in late 2026 a communications relay will be deployed on Mons Malapert. All three of these systems will be able to communicate with the rovers operating on the surface which will allow them to be controlled from Earth. The communication architecture will be discussed in more detail in section 9. This autonomous control is necessary since the astronauts can only survive for 72 hours in their lander before the base needs to be finished and because the rovers need to help with the setup of the infrastructure on the Lunar surface that will support the landing of the launch vehicles.

7.5.4 Set Up

The first payload that will be sent to the Lunar surface will be in 2024. A Falcon Heavy will carry two MX-9 landers from Moon Express, shown in Figure 11, which will be inserted into LLO. These landers are advertised to deliver 6.5 km/s of Δv with 2350 kg of total mass and 2000 kg of fuel [39]. From equation 1 the Isp can be solved for the lander, which is approximately 321 s. The lander needs to have at least 2 km/s of Δv to land the supplies from LLO. Again using equation 1 with 2500 kg of payload, the lander has 2.3 km/s of Δv meaning it can land a payload of 2500 kg or less. One of the MX-9 landers will carry three Cratos rover, a Helelani rover and the lunar crane. The second lander will deliver the first All Terrain Hex Limed Extra Terrestrial Explorer (ATHLETE rover). The Cratos rovers are primarily there to help cover the habitats with Lunar regolith, but they will also function to help level out the area for multiple landing pads. The landing pad will be constructed by both the ATHLETE rover and the Helelani rover.

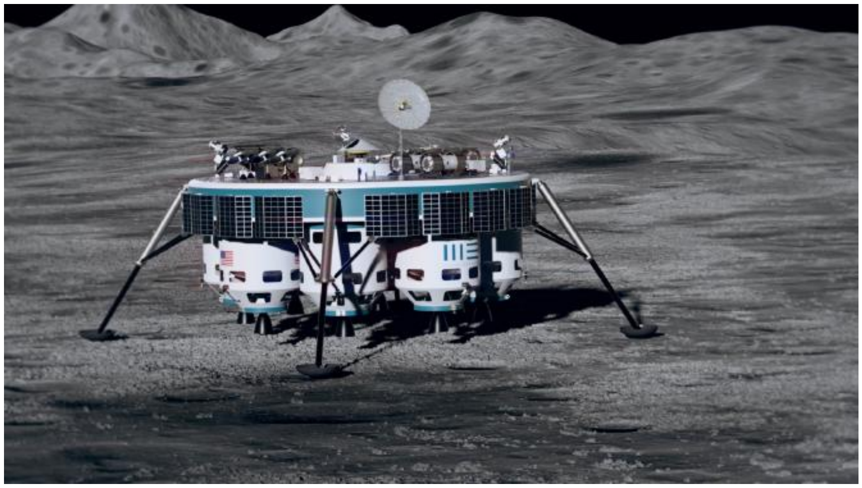

Figure 11. Above is the MX-9 Lunar lander from Moon Express. The lander is seen in a sample return configuration, one of the three different configurations available.

The Helelani rover, as shown below in Figure 12, stands at approximately 1580 mm tall with a width of about 1650 mm and a length of roughly 1450 mm. It has a gross vehicular weight of 329 kg including two 24 V DC lead-acid battery packs [40]. These battery packs are located in the chassis and are all that's needed to power the rover, each weighing approximately 36 kg. Included on the Helelani are rover sensors for voltage, temperature, GPS, and more. The rover is operated via RF receivers and a remote control. Helelani is capable of carrying approximately 110 kg of payload and is equipped with an imaging system useful for maneuvering the rover [41]. The movement of the rover is powered by two independent Magmotor Electrical motors that operate with a steering mechanism that resembles a skid and steer design. In addition to the remote control capability, Helelani can also be controlled over the internet using the GUI and a wireless link communication system [41].

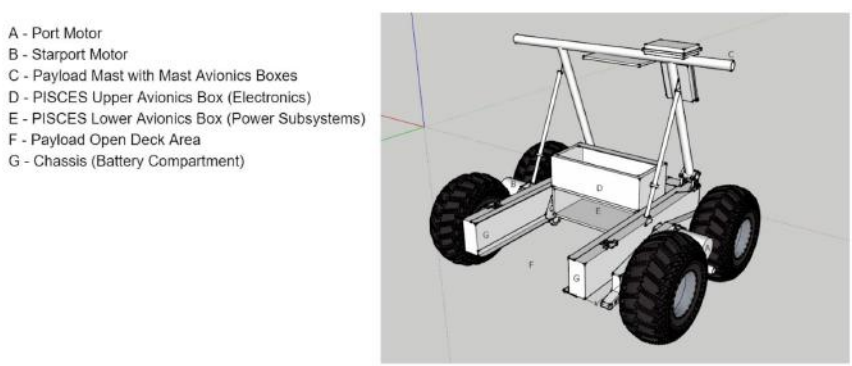

Figure 12. The Helelani rover's components are shown above. The primary function of this rover is to carry and place the pavers for the landing pad. The rover interlocks the pavers together [40].

The landing pads will be constructed with pavers fabricated by sintering basalt, which is lunar soil. The pavers lock together and will be placed by the Helelani rover, which can be seen in Figure 13. The pavers will be sintered using a

3D printer, which operates via microwaves, that attaches to the ATHLETE rover [42]. The landing pads consist of two regions: the bullseye and the outer apron. The bullseye, which is the preferred landing location, is constructed with the pavers, which have a TRL of 5 [41]. Without the pavers in place, the rocket plume could cause craters to form under the lander, as well as ejecting regolith dust and debris at approximately 2000 m/s [41]. The outer apron is an excavated area surrounding the bullseye made by compacting the regolith surrounding the landing area. This area surrounding the bullseye is safe for landing, but multiple landing and take-off maneuvers are not ideal due to the damage that the force of the rocket would cause to the excavated soil. Having multiple launches and landings on the outer apron could cause high velocity regolith that could cause damage to the rocket, as well as anything surrounding the pad.

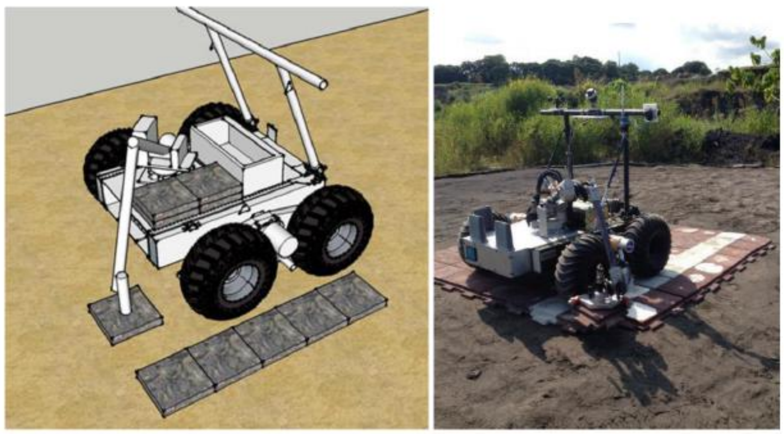

Figure 13. Helelani rover placing the pavers for a landing pad. This testing was done in Hawaii making pavers out of the basalt material on the ground [42].

Once the landing pad has been constructed, the Cratos rovers will add a resin made from 1880 kg of PTFE and regolith to the joints between the pavers. This will ensure that the pavers stay connected while a rocket is landing. The rovers will also place sensors around the landing pad to help track the rocket during landing. With the landing legs, RCS thrusters on the rocket, and the landing pad in place, the rocket will have no problem landing. The rocket will decelerate using the main engine until it is 20 m above the lunar surface so that the rocket engine does not blow away any of the pavers or melt the resin. The RCS thrusters, along with short bursts from the main engine will continue to slow the rocket during its descent, counteracting the Moon's gravity. Once the rocket has landed, the ATHLETE rover will move the crane over to the rocket to help unload it. The crane used is based on the NASA designed Lunar Surface Manipulation System (LSMS) which has been tested to work with the ATHLETE rover shown in Figure 14 [43]. The mass of this crane was designed to be only 3% of the mass being carried at the tip. Additionally the height can be scaled up to be able to unload cargo from the top of the payload fairing of the Falcon Heavy at 24 m high. With the highest mass single piece of cargo being the habitat at 8000 kg, the mass of the crane was found to be 1000 kg with the scaling for the height as well. The original crane was 8m high so the crane needed to be three times taller. This mass also includes the growth factor of 20% for the mass of the habitat along with a 1.2 factor of safety. This crane was also designed to be low cost and is approximated to cost $1000 per kilogram of material [43].

Figure 14. The LSMS and ATHLETE rover working together to perform offloading of simulated ISRU. This testing was done by NASA Langley at Moses Lake Washington [43].

The ATHLETE rover, shown in Figure 15 is capable of carrying over 14,000 kg on the lunar surface and can traverse angles of up to 35 degrees [44]. ATHLETE will be responsible for moving the payloads from the landing pad to the location of the base for setup. There will be two ATHLETE rovers on the surface of the Moon so that one can move the crane and lower the payloads down onto the second ATHLETE. The ATHLETE rovers also have the capability of separating into two separate Tri-ATHLETEs and will use this capability to approach the empty rockets on the landing pad, pick them up, and move them to a safe location out of the way. Before the payloads can be delivered to the landing pad, it will needs to be tested to ensure the technology is ready.

To prevent any loss of payloads, the landing of the Falcon Heavy will be tested twice with no payload and once with a minimal payload. The first test will occur in 2024 with the launch vehicle that delivered the MX-9 landers. This rocket will stay in its orbit for approximately 72 hours while the landing pad is built since the landing pad takes 72 hours to build with the ATHLETE and Helelani rovers [40]. This rocket will go through the landing procedure outlined above in the paper and will then be moved off of the landing pad. Any necessary adjustments that need to be made to the technology will occur over the next two years. The next test will take place in 2026 with the launch vehicle that is delivering the three polar satellites to LLO. This rocket will go through the same procedures as the first test and adjustments will be made again. Then, there will be the first test with a real payload. Still in 2026, a rocket will deliver the communications relay for Malapert Mons and the second ATHLETE rover. The relay will be on another MX-9 lander and will separate from the launch vehicle once it is in LLO. Once they are far away enough from each other, both the launch vehicle and the MX-9 lander will go through landing procedures. Again, the rocket will follow the procedure outlined above to confirm that the technology works. Once the rockets lands, the ATHLETE rover will be unloaded using the crane and the first ATHLETE. Now that all of the infrastructure is in place on the Lunar surface and the technology has been tried and proven, it will be possible to deliver the habitats needed for the mission. The

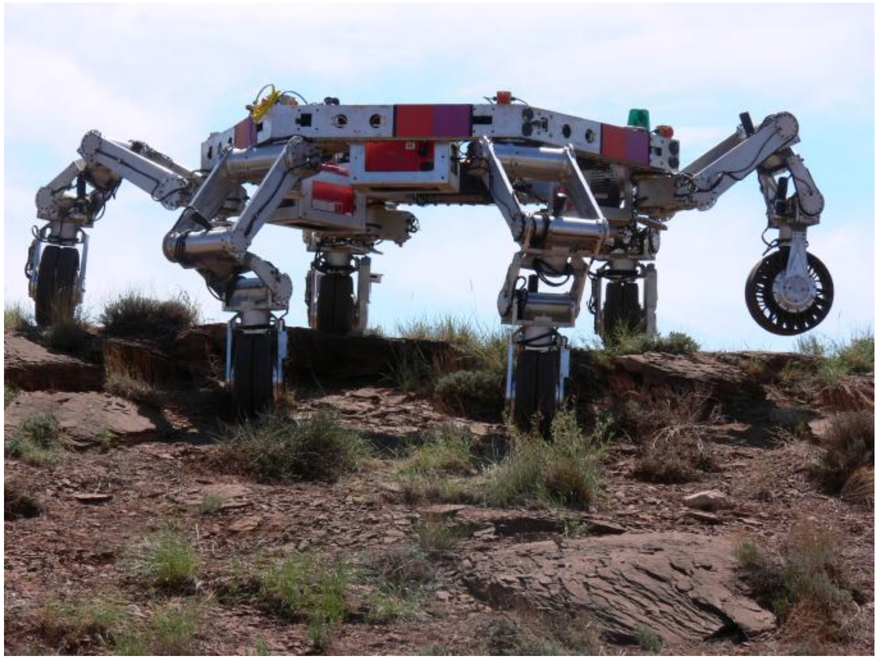

Figure 15. The ATHLETE rover is a six legged rover capable of stepping over rocks, which makes it very mobile and capable of climbing steep terrain [44].

cost comparison of using this system vs lunar landers can be seen in Figure 16. The costs of this system include the development costs, the costs of the crane, rovers, and Lunar landers. This system's cost was also compared against buying lunar landers and launching them on Falcon Heavies. This also drove up the cost because of the increased mass of each launch and made it impossible to get the habitat to the Lunar surface.

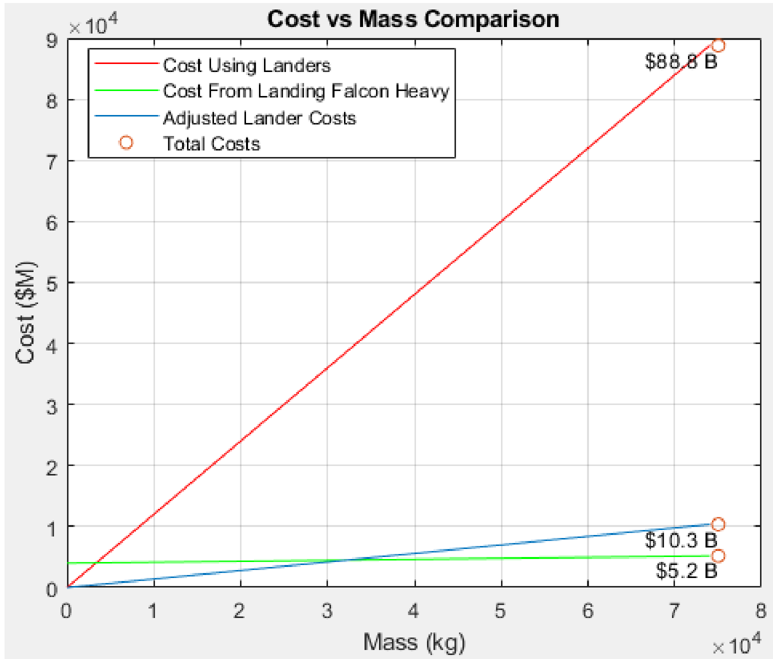

Figure 16. Cost per kilogram mass delivered to the Lunar surface using three different methods is shown. These methods are using Lunar landers, the system outlined in this section, and purchasing both the Lunar landers and Falcon Heavies separately. The costs associated with these methods are $90 B, $5.2B, and $10.3B respectively to deliver 75,000 kg to the Moon showing the savings made and the potential savings in the future when the base is expanded upon.

The system discussed above allows the mission to be complete within the budget of $12 B and promotes expandability for the Lunar base. As time goes on and more is added to the Lunar base the savings for launching and landing mass on the Moon will continue to increase. If this is to become a permanent habitat constantly crewed with humans, there will need to be resupplies of food, water, and oxygen. This system allows for larger payloads to be delivered and cuts costs drastically. Without this system, it would also be impossible to land large habitat components needed for human survival.

8. Habitat Architecture

8.1 Summary

The lunar environment is one of the most inhospitable enviroments. With no atmosphere and no magnetic field, the Moon is constantly bombarded with micrometeorites, solar wind, and ionizing radiation. To add to the danger, some locations have temperature differentials of more than 210 K. The goal of the habitat architecture is to provide the crew with protection against this dangerous environment. To ensure the safety of the crew for manned missions NASA has outlined explicit standards and requirements. To meet these standards. The lunar habitat outlined in this section was carefully designed to meet all requirements while best suited for the mission directives. The habitat consists of three ovoid inflatable modules connected by rigid octagonal nodes. Each module is covered in a PTFE-regolith polymerized matrix to provide additional protection from the lunar elements. The octagonal nodes have openings on each side to serve as an air lock during EVAs whil also serving as potential connections for future inflatable habitats.

Within the habitat is a robust life support system, based on the system aboard the ISS, to ensure the internal atmosphere is breathable. The system will be modified to processes oxygen provided from ISRU collection. Oxygen will also be brought from Earth and stored in cryogenic tanks, with a few non powered backup oxygen production systems to ensure redundancy in the event of an emergency. Through the use of this complex network of habitat systems, the crew will be comfortable while on the Moon and able to complete their science missions as required by the AIAA.

8.2 Standards and Requirements

Most aspects of space mission design are subject to established standards and requirements, which are outlined in the NASA Spaceflight Human-System Standards (NASA-STD-3001). The document specifies in detail the uniform basis by which all spaceflight missions must abide to. NASA-STD-3001 provides a strong foundation to place the safety and comfort of our crew upon. When considering the human element of a mission, many requirements are a constant across all missions. Although the environmental problems can be unique, the solutions are not quite so. Each crew member has their own power and volume requirements, namely caloric intake and personal space. The solution then is to provide the crew with satisfactory sustenance and living space within the habitat. The human system requires breathable oxygen and thermal control; however, these are not naturally present on the lunar surface. Thus, a life support system using ISRU and power is necessary to provide ample air and a comfortable temperature. Radiation sickness presents a real danger to the crew, thus radiation blocking is required. Preventing human system breakdown is a balancing act of providing the numerous needs and wants of the crew. As such, the physical and mental health of our crew is not dissimilar to the maintenance of a machine. The human crew is a system within the greater mission. In detail below in Table 8.2.2 are the numerous requirements provided by NASA-STD-3001 and how this mission accommodates for them.

8.2.1 Volume and Area Requirements

Volume 2 of NASA's system standards states that the habitat must provide sufficient volume to accommodate the crew of four, their activities during their mission, and their overall behavioral health [1]. This requirement from NASA is purposefully vague due to the fact that there is no agreed upon standard for required habitable volume [45]. Habitable volume, by itself, is also a poor measurement of the habitat's acceptability without interior layout considerations [45]. However, rough estimates of the volume required per crew member can be made by looking at past missions. Looking at the volume per crew for the Skylab, Salyut, MIR, and ISS missions gives a range of total volumes required for a crew of four [46]. These total volume estimates range between 120 m^3 for Salyut up to 568 m^3 for the ISS [46]. Another analysis conducted by NASA used five estimation methods – historical spacecraft volumes, standards and design guidelines, Earth-based analogs, parametric sizing tools, and conceptual point designs – found that the total pressurized volume required to support a crew of four ranged between 160 and 280 m^2 and that the total habitable volume per crew member ranged between 40 to 70 m^3 [46]. More volume is beneficial to the crew and the overall expandability of the base but more volume means more habitat mass so a compromise must be made to achieve acceptable volume and mass numbers.

In addition to the overall internal volume requirements, breaking down the volume and area requirements by functional areas was important to obtain a more detailed breakdown. Since NASA does not specify volume or area requirements for these functional areas, a study by the University of Houston in collaboration with NASA was used as the primary source for deriving the volume and area requirements [47]. Most papers on the subject only state volume requirements, but square footage is also important for applications such as a lunar base where gravity requires that there be a designated floor. Table 11 shows the volume and area requirements for each of the functional area systems in the habitat.

Table 11. Volume and area requirements for different habitat functional areas derived from the Partial Gravity Habitat Study conducted by the University of Houston in collaboration with NASA [47]. Totals for area and volume do not include circulation space that must also be added so that astronauts are able to easily move throughout the structure.

Functional Area	Required Volume (m^3)	Required Area (m^2)
Galley	18	4
Dining	16	5
Recreation	15	9
Exercise	20	11
Health	14	4
Storm Shelter	20	11
Crew Quarters	28	10
Hygiene/Waste	5	3
Maintenance/Work Area	10	5
Communication Stations	12	6
ECLSS	30	8
ECLSS Backup	30	8
Science	50	19
Total	**268**	**103**

8.2.2 Environmental Standards

From the NASA-STD-3001, the relevant environmental standards are outlined in Table 8.2.2 These provide the baseline for what is necessary for mission success, and drove the decision-making process for what environmental systems to utilize in the habitat. Along with the standards Table is Table 13, which defines the temperature to humidity ratio the environmental systems must aim for, and Figure 17, which defines the radiation doses the crew cannot exceed on a 30-day, 60-day, and lifetime basis.

Table 12. NASA-STD-3001 Environmental Standards. These outline the imperative functions of environmental control systems, and must be followed for the crew to live comfortably in the habitat.

Section	Requirement
6.1 Trend Analysis of Environmental Data	The system shall provide environmental monitoring environmental water, data such as atmospheric composition, temperature, humidity, and radiation. The crew shall be able to interpret this data to prevent harmful conditions.
6.2 Internal Atmosphere	The system shall provide a clean, breathable atmosphere consisting of oxygen in addition to inert diluent gasses such as nitrogen and limited carbon dioxide. The system shall maintain ppO2 to within the physiological range of 20.7 kPa and 50.6 kPa. The internal atmospheric pressure shall be between 20.7 kPa and 103 kPa at all time and shall not decrease by more than 207 kPa/min or increase by 93.1 kPa/min. Average relative humidity shall not fall below 25% nor above 75%. Atmospheric temperature shall be between 18 °C and 27 °C. Atmospheric temperature/humidity ratio shall fall within the "Comfort Zone" outlined in Figure 17. The system shall provide a display, recording, alerting, and remote adjustment of atmospheric conditions, including possible combustion or contamination
6.3 Water	The system shall provide clean, sterile, potable water capable of supporting any human use. There shall be at least 2.0 kg of water per crew member per day specifically for drinking. The system shall provide methods of heating and cooling water. The system shall provide sufficient water for food rehydration, personal hygiene, immediately available eye irrigation, medical contingency, suited operations, and post-landing recovery.
6.4 Contamination	The system shall only use chemicals of Toxic Hazard Level Three or below as defined in the Guidelines for Assessing the Toxic Hazard of Spacecraft Chemicals and Test Materials (JSC 26895). The system shall use chemicals that would not decompose into hazardous chemicals. The system shall limit air- and waterborne contaminants as defined in the SMAC tables found in current JSC 20584. The system shall limit the levels of lunar dust smaller than 10 μ in size to an average of 0.3 mg/m^3. The system shall provide a means to remove contaminants
6.8 Radiation	Planned career exposure to ionizing radiation shall not exceed 3% of Risk of Exposure-Induced Death. Planned radiation dose shall not exceed career and short-term limits defined in Table 13

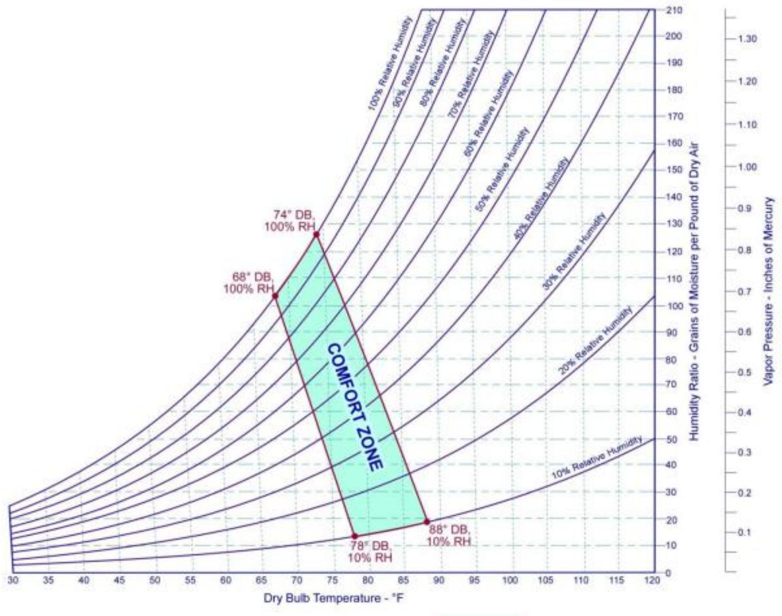

Figure 17. Above is the NASA-STD-3001 defined "comfort zone" temperature/humidity ratio of habitat atmospheres. Environmental control systems such as ECLSS must always abide by this range of ratios to maintain a comfortable habitat for the crew.

Table 13. Dose limits for Short-Term or Career Non-Cancer Effects as provided by NASA-STD-3001. Units are in miliGray-Equivalent (mGy-Eq) quantity and miliGrays (mGy). The limits to radiation on the crew informed some of the architectural decisions detailed in this section.

Organ	30-day Limit	1-year Limit	Career Limit
Lens	1000 mGy-Eq	2000 mGy-Eq	4000 mGy-Eq
Skin	1500	3000	6000
Blood-Forming Organ	250	500	N/A
Circulatory System	250	500	1000
Central Nervous System	500 mGy	1000 mGy	1500 mGy

8.3 Habitability Standards

NASA-STD-3001 also defines the habitability standards by which the mission operations must abide, the most relevant of which are outlined in Table 14. These standards are meant to preserve the physical and mental health of the crew by offering them a high standard of nutrition, hygiene, personal space, medical capability, and recreation. The systems to meet these standards success are detailed in the habitat architecture and layout. In summary, two of the three subhabitats are devoted to the living and recreation of the crew during their 45 day stay. The third subhabitat is meant to hold all the equipment the crew will use for their scientific experiments.

Table 14. Habitability Standards from NASA STD 3001[1, 2]

Section	Requirement
7.1 Food	Food shall be of minimum quality, have a minimum caloric content, and have Nutrition a minimum amount of micronutrients. There shall be a viable way to make food and clean the area where the food was made. There shall be a way to control food spill and waste.
7.2 Hygiene	The habitat shall be capable of offering personal bodily hygiene facilities.
7.3/7.8 Waste Management	The habitat shall have waste management for bodily and consumable waste.
7.4 Physiology	The habitat shall have suitable volume and environmental accommodations, and countermeasures for physiological emergencies.
7.5 Medical	A medical system shall be provided to the crew for all four levels of physiological emergencies, from level 1's motion sickness and first aid to level 4's sustainable life support and limited surgical care.
7.6/7.7 Storage	The system shall provide accessible stowage of hardware, supplies, personal items, and other inventory management.
7.9 Sleep	The system shall accommodate for privacy and sleep requirements of each crew member.
7.10 Clothing	The system shall have sufficient provisions and storage for exclusive crew clothing.
7.11 Housekeeping	The habitat shall provide sufficient volume for cleansing access and materials for cleansing capabilities.
7.12 Recreation	The habitat shall have personal and team recreational capabilities for the crew.

8.3.1 NASA Architectural Standards

NASA-STD-3001 also details the architectural standards of the habitat. These drove the final external design of the habitat, which was meant to encompass the navigable volume the crew would need to live comfortably. Many of the architectural standards are only relevant to zero-gravity living quarters, and therefore do not apply to a lunar stay.

Table 15. Requirements for providing comfort for the crew[2].

Section	Requirement
8.1 Volume	The habitat shall provide the volume and equipment necessary to ensure crew safety, maneuverability, and usability for experiments.
8.4 Hatch Operability	Hatches and doors shall be operable at all times for use in operations, contingencies, and emergencies. facilities.
8.7 Lighting	The system shall provide internal and external lighting to support all expected crew tasks. In the event of an emergency, the system shall provide emergency lighting. The lighting within the system shall allow for circadian processes such as sleep. The system shall provide lighting controls and glare prevention.

8.3.2 NASA Hardware and Equipment Standards

Due to the scientific nature of the mission, the crew must be provided with all the hardware and equipment they will need, as well as all the protective equipment to keep them safe. NASA-STD-3001 provides the broad standards which shall apply to all the hardware and equipment. When choosing scientific missions the crew would undertake during their lunar stay, these guidelines were kept in mind to ensure the hardware provided for the crew would meet NASA standards.

Table 16. Requirements for Crew Safety[2].

Section	Requirement
9.1 Standardization	The hardware and equipment shall have interfaces common between all crew members. The interfaces shall be differentiable based on what equipment they are for and functions they serve
9.2 Training Time	The habitat and equipment shall have a minimized amount of time to train for all system, hardware, and equipment operation.
9.3 Hazard Minimization	The system shall protect the crew from mechanical, energy, structural, heat, gas and fluid, shock, and sharpness hazards. All PPE shall be provided for the crew
9.4 Durability	The systems, hardware, and equipment shall have protective provisions and shall be capable of withstanding forces imposed by the crew and environment.
9.5 Assembly and **Disassembly**	All system hardware and equipment shall be designed such that it cannot be installed improperly. All mating and demating of equipment shall be made simple and nonhazardous by the equipment without the use of tools.
9.6 Cable Management	The system shall provide space to keep the cables organized for easy access and maintenance.
9.7 System Maintainability	The crew shall have a standardized method of maintaining and operating the system functions.

8.4 Habitat Design

8.4.1 Constraints

In addition to the requirements imposed by NASA, there were several other factors that constrained the design of the habitat. The mass and stowed dimensions of the habitat were constrained by the available launch vehicles and landers. The Falcon Heavy fairing constrained the stowed dimensions to no more than 4.6 meters in diameter and 6.7 meters in length. In order for the habitat to be landed on the lunar surface, the mass was also constrained by the lander to be no more than 9000 kg.

8.4.2 Inflatable Habitat Design

The goal of the habitat design was not only to meet the requirements for this particular mission but to create a multipurpose and modular design that could be used to expand upon the initial base in the future. Given the requirements and constraints, an inflatable habitat design was selected over a more traditional rigid structure. An inflatable design allows for more internal volume with less mass than what is possible with a rigid structure.

There are several models of inflatable habitats available through NASA's private partners. Bigelow Aerospace and the Sierra Nevada Corporation both have their own models of inflatable habitats. Bigelow's large inflatable habitat is known as the B330 whereas Sierra Nevada's habitat is known as the Large Inflatable Fabric Environment (LIFE). Using a commercial option that has already been developed would be ideal for saving costs, however, the commercial habitat options are designed to be space station modules rather than lunar base modules. Due to being designed as space station modules, the habitats have a high mass of more than 20,000 kg that would make them incredibly difficult and economically infeasible to deliver to the lunar surface. For this reason, it was not possible to utilize a pre-existing habitat design. Due to the constraints imposed by the launch vehicles available, it was necessary to use several smaller habitats than a single large habitat. To meet the volume requirements and constraints on launch vehicles, it was necessary to design three smaller habitats.

The custom habitats use the same technology as the Bigelow and Sierra Nevada habitats. The walls of the habitat are made of multiple layers of different materials. The outer layers of the habitat make use of materials such as Kevlar and Vectran to provide ballistic and abrasion resistance. Inner layers are made up of thermal insulation and polyurethane bladders used to inflate the habitat. The innermost layer, which is exposed to the crew, is a Nomex fabric which is fire and chemical resistant. In total, the walls of the habitat are approximately 0.5 m thick with over 20 layers of material to provide ballistic, radiation, and thermal protection.

Bigelow's B330 and Sierra Nevada's Large Inflatable Fabric Environment (LIFE) habitat are both cylindrical in shape. All inflatable habitats currently being produced use a cylindrical or ovoid shape due to the shape eliminating corner stresses that are present in other shapes. One downside with using the cylindrical design in an environment with gravity is that the maximum floor area can only be achieved by placing the floor at the middle of the cylinder which leaves roughly half of the habitat's volume unusable. This problem is not a concern in a space station environment where everything is in free fall and there is not enough gravity to distinguish a floor. This problem can be remedied by using an ovoid shape for the habitat. As shown in Figure 18, an ovoid shape allows for eliminating corner stresses like the cylindrical option but also allows for optimizing usable floor area and internal volume. Figure 19 shows a cutaway of the actual habitat structure.

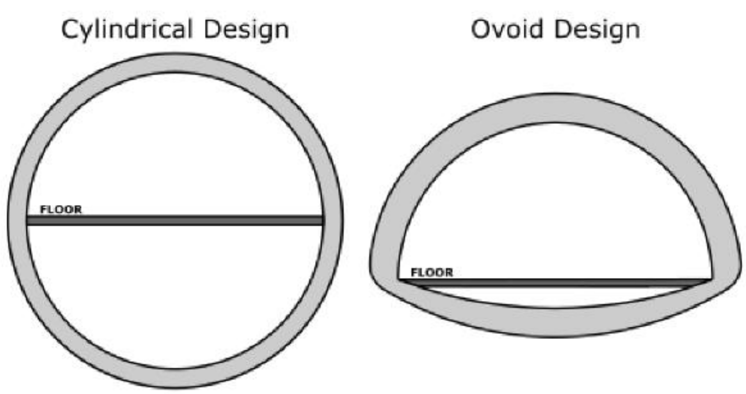

Figure 18. Diagram showing how the ovoid design allows for optimizing the usable volume and floor area. For maximum area, the floor must be placed in the middle of the cylindrical design which causes a significant loss in habitable volume. The ovoid design avoids this by minimizing the volume under the floor while maintain the structural advantages of the cylindrical design.

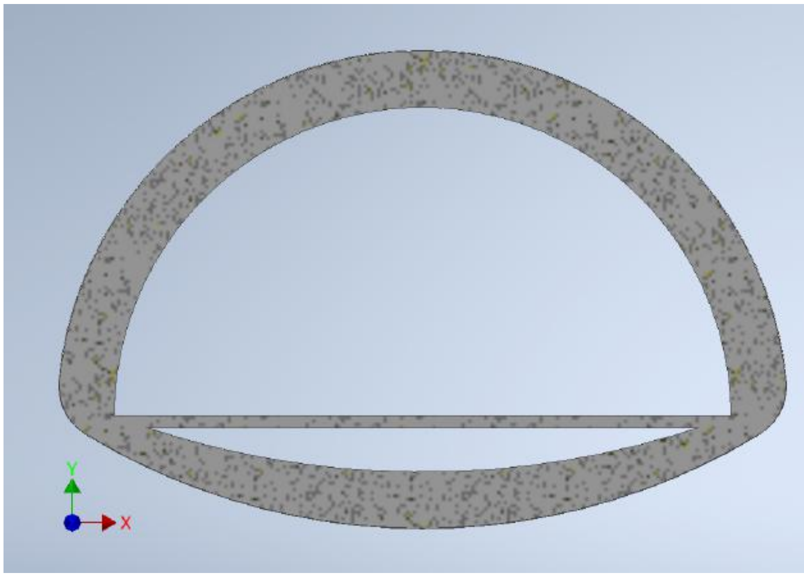

Figure 19. Diagram showing a cutaway view of the habitat structure's shape. Using an ovoid-shaped design allows for optimization of floor area and usable internal volume.

Using the ovoid design, the habitat's inner diameter is 5.5 m and outer diameter is 6.5 m with an internal length of 10 m. This provides approximately 120 m^3 of internal volume and roughly 55 m^2 of floor space per habitat which is enough to satisfy the volume and area requirements and have enough volume and area left over for circulation within the habitats. Figure 20 shows a model of the completed habitat design.

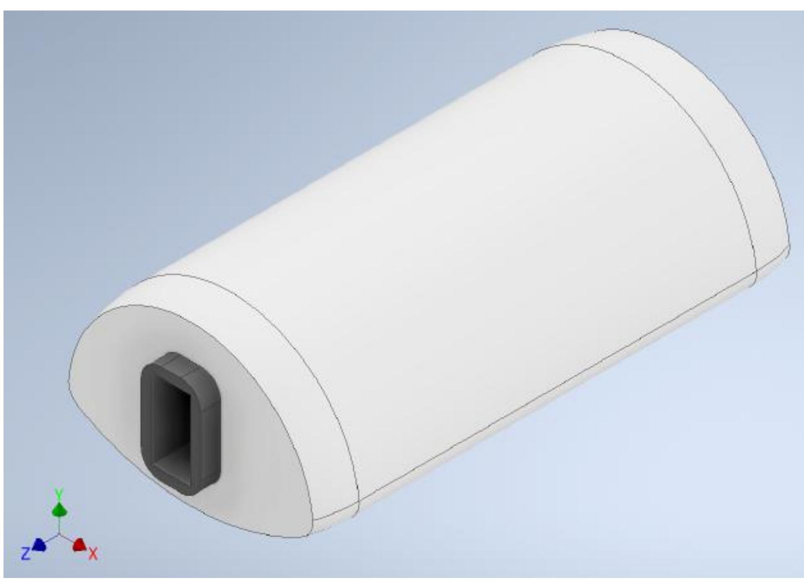

Figure 20. Computer model of the inflatable habitat structure. The inflatable portion, shown in white, has a length of 10 m and the entire structure has a length of 12 m after accounting for the rigid mating adapter, shown in gray.

8.4.3 Connecting Node Design

The connecting nodes for the base serve to connect the habitats and function as the ingress/egress points. Unlike the habitats, the connecting nodes are traditional rigid structures made of aluminum. The connecting nodes are aluminum due to the fact that structures such as airlocks and suit ports are significantly easier and cheaper to integrate into rigid structures than inflatable structures.

The connecting node design is directly tied to the external habitat layout. Given the relatively high mass of rigid structures, it is beneficial to minimize the number of connecting nodes that must be utilized in the base structure. NASA also requires that each habitat allows for dual egress meaning that each habitat should have more than one point of entry and exit. A triangular habitat layout allows for minimizing the number of nodes needed and allows for dual egress from each habitat. Figure 21 shows a diagram of the initial base layout and an example of how the base could be expanded upon in the future.

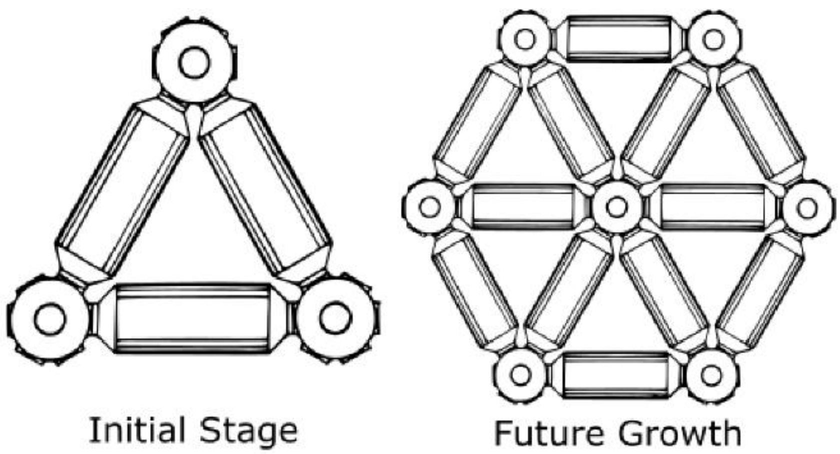

Figure 21. Diagram (left) showing the initial three habitats and the three connecting nodes between them. Diagram (right) showing one example of future growth for the habitat with the triangular configuration.

The rigid connecting nodes are octagon-shaped and make use of semi-rigid extendable connectors to mate the habitats and nodes. The extendable connectors allow for flexibility in the spacing of the habitats and the overall layout which contributes to the expandability of the design. The flexible connectors also serve to make it easier to connect the habitats on the uneven lunar terrain. Figure 22 shows a model of the initial habitat layout with the three rigid connecting nodes and the three inflatable habitats.

8.4.4 Habitat Floor Plan

When designing the habitat floor plan, it was important to consider the cross compatibility of functional areas. One added benefit of having multiple habitats rather than a single habitat is that separating incompatible functional areas is much easier. The functional areas were first separated by work and living areas. Each habitat contains a one-meter wide

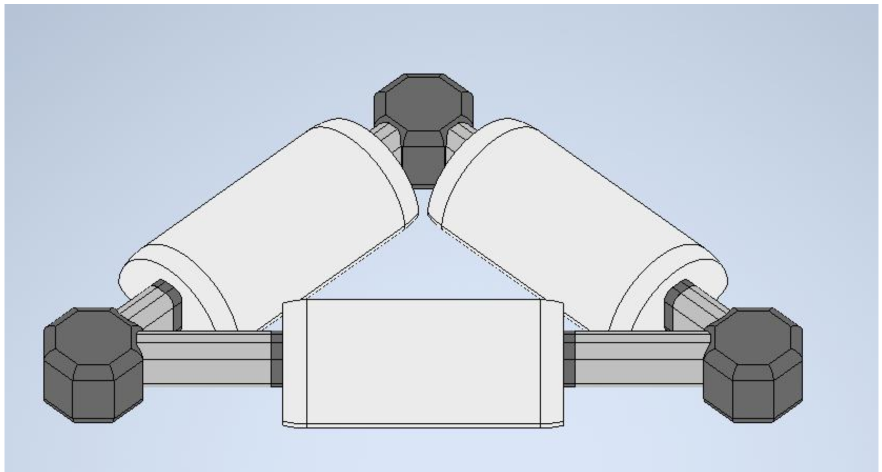

Figure 22. Model of the three habitats, shown in white, and the three rigid connecting nodes, shown in dark gray. The rigid nodes make use extendable, flexible connectors, shown in light gray between the habitats and nodes, to allow for flexibility in the spacing of the habitats and to make it easier to connect the habitats on uneven terrain.

central corridor to allow for easy movement of people and equipment between habitats.

The first habitat includes the crew quarters, galley, dining, and recreation areas. This category includes areas that should produce minimal noise to not interfere with the crew's sleeping, eating, and recreation activities. The galley consists of four International Standard Payload Racks (ISPR) containing the equipment such as a water chiller, water heater, rehydration ports, a small trash compactor, a small oven, and food storage space. The International Standard Payload Rack is a modular steel framework used on the International Space System to allow for interchangeability of hardware and experiments [48]. The ISPR allows for the lunar base to re-use hardware that has been tested and used on the ISS. The ISPR also allows for workstations to be easily upgraded in the future with new components which aids to the expandability of the design. The crew quarters each contain storage, bunk beds to save floor space, and a small personal workstation. The dining and recreation areas provide space for the crew to sit and eat or enjoy their downtime.

The second habitat contains the health maintenance facilities, exercise areas, hygiene and waste facilities, and storm shelter. The second habitat also houses the primary environmental control and life support system (ECLSS). The health maintenance facility also makes use of three payload racks to house basic medical equipment and tools such as monitoring equipment and terminals. Newer NASA documents on waste management do not specify the number of restrooms required per crew, but older NASA documents (NASA STD-3000) state that there should be one bathroom per four crew members [49]. The exercise area contains enough space for room for a stationary bicycle and treadmill like used on the ISS [50]. The habitat also includes a 11 m^2 storm shelter to protect the crew during solar particle events.

The third habitat primarily includes the laboratory space and maintenance work area. The work habitat also includes a backup ECLSS system. The third habitat contains four additional payload racks to store the scientific equipment and experiments that will be run during the mission. The habitat also contains general work space and storage areas for samples. The maintenance work area will provide astronauts the space and tools to repair any of the habitat's systems that may need repair. Table 17 shows a detailed breakdown of the habitat subsystems and the area breakdown for each system.

Table 17. Table showing a breakdown of habitat subsystems for each of the three habitats.

Crew Habitat		Health Habitat		Laboratory Habitat	
System	**Area (m^2)**	**System**	**Area (m^2)**	**System**	**Area (m^2)**
Crew Quarters	10	Storm Shelter	11	General Lab Space	10
Recreation	9	Exercise	11	Payload Racks	4
Dining	5	ECLSS	8	Comms Racks	1.5
Galley	4	Waste/Hygiene	3	ECLSS Backup	8
Comms	3	Comms	1.5	Maintenance	11
Storage	4	HMF	4	Storage	4

8.4.5 Architectural ISRU

Although the inflatable habitat is fundamentally strong, and insulates heat and radiation well, the thermal insulation it offers can be improved through the use regolith shielding. While previous spacecraft examples such as the ISS use thin reflective insulation for heat rejection, such a solution is unviable in the shaded regions of the moon between Cabeus and Haworth, where the temperature can reach an astoundingly frigid 55 K, or -218 °C. Even the maximum temperature of 265 K, or -8 °C, goes well outside the NASA-STD-3001-required minimum of 18 °C. Thermal control can be accomplished by bringing heavy elements such as lead or aluminum from Earth, but this solution is equally nonviable due to the massive cost of transporting such materials. Therefore, looking inside the box, or perhaps the Moon, may be the optimal solution to the insulation problem.

8.4.6 Lunar Regolith for Environmental Protection

Lunar regolith has strong potential to act as structural material. The substance's main attributes include extreme abundance and ease of manipulation for architectural purposes. Utilization of lunar regolith will allow for many of the structural requirements to be met while testing additional technologies and reducing the mass of resources that would otherwise need to be transported from Earth. Analysis from the trace samples recovered from previous lunar excursions have identified the general composition of the regolith and has allowed for the creation of a lunar simulant, JSC-1A, that can provide approximations on structural and thermal properties of the regolith. As such, the synthesis of regolith-based structural mixtures, known colloquially as lunarcrete, has been lab-proven before, shedding light on the structural and thermal properties of what may be the most important resource to this mission. In-Situ Resource Utilization (ISRU) of regolith will allow for reduced costs and mass required to create structures and ideal testing conditions for certain lunar experiments [51]. Environmental protection in space is a broad subject, encompassing protection from temperature fluctuations, micrometeorites, solar wind, ionizing radiation, and local moondust particles. All of these hazards pose a danger to the crew, and must be mitigated to the fullest extent possible. Lunarcrete, as an outer barrier to the habitat, is capable of warding off all of these dangers, keeping the crew safe while inside the habitat at all times while also allowing the ECLSS system to provide internal environmental control.

Many versions of lunarcrete exist, including the three considered for this mission: sulfur-based, water-based, and polymer-based regolith mixtures. Each has their advantages and disadvantages. However, with the addition of the Cratos rovers, the utilization of polymers seemed the most optimal. To justify this decision, a scientific experiment was planned which tested the structural and thermal properties of JSC-1A-based concrete using the testing equipment available at the Virginia Tech Aerospace Structures and Materials Lab in Randolph Hall. While the experimental portion of the exercise was rendered impossible by the COVID-19 pandemic, the theoretical portion could justify our decision to utilize polytetrafluorethylene, also known as Teflon. From the code developed for the experiment, the necessary thickness of Teflon to block incoming radiation and micrometeorites is 1 mm [52]. With a density of 2170 kg/m^3, and assuming full coverage of the habitat and additional uses of Teflon totals to 0.96 m^3, 2080 kg of Teflon must be transported from Earth. Using this Teflon, the Cratos rovers will 3-D print Teflon-based lunarcrete, totalling 1 cm of outer walls. This additional material from the regolith will be enough for thermal, solar wind, and lunar dust insulation.

8.4.7 Habitat Power Requirements

The habitat is one of the main entities of the mission that requires a substantial amount of power. NASA-STD-3001 has numerous requirements for the personal hygiene, waste management and recreational capabilities as well as the lighting of the habitat. All of the ways to fulfill these requirements need power to function. In addition to the required power for the appliances and modules, the average power was calculated using a duty cycle fraction. The duty cycle was determined by using the ISS crew schedule to estimate how long the crew would be using each appliance and therefore how much power would be used within a 24 hour period [53]. An overview of the habitat power requirements can be seen in Table 18.

Table 18. An overview of the required and average power of the habitat systems. The average power was found by multiplying the required power by a duty cycle fraction. The duty cycle fraction was determined by estimating how much time each system was used during 24 hours.

Functional Area	Required Power (W)	Average Power (W)
Galley	4100	807
Lighting	400	400
Recreation	500	60
Health	5000	50
Crew Quarters	500	60
Hygiene/Waste	50	7.5
Communications	500	500
Science	12000	6960
Total	23050	8845

In the galley, the crew prepares their meals and disposes of waste. The stove requires 1000 W of power and has an average power of 110 W. The trash compactor requires 400 W and has an average power of 44 W. The 20 gal W heater and the 10 gallon water chiller requires 1500 W and 750 W, respectively. The water heater's average power is 165 W and the water chiller's average power is 82.5 W. The rehydration ports for the dehydrated food requires 50 W and has an average power usage of 5.5 W. The lighting of the habitat is used 24 hours of the day and therefore has a required power and an average power of 400 W. The communication system is also used for 24 hours and so has a required and average power of 500 W [47].

The dining area requires no additional power. The crew recreation area requires 500 W with an average power use of 60 W per day. The first aid station and the medical equipment could use up to 5000 W of power but have an average power of 50 W. The crew quarters, including the sleeping areas and personal space, has a requirement of 500 W and an average usage of 60 W per day. The hygienic facilities of the habitat, including the toilets, showers and waste disposal, requires 50 W and an average power of 7.5 W. The equipment used for science experiments in the work space needs 12000 W and uses on an average day 6960 W. The total required power of the habitat is 23050 W. The total average power usage of the habitat is 8845 W. The average power of the habitat was used to determine the power source because it is more accurate to the actual amount of power used in the habitat [47].

8.5 Environmental Systems

The best option for the life support systems for the lunar base camp is the ECLSS that is used on the ISS. The ECLSS is comprised of seven subsystems that would be used to ensure that a human crew shall survive for an extended period of time on the Moon. These subsystems are the Atmosphere Control and Supply (ACS), Atmosphere Revitalization (AR), Fire Detection and Suppression (FDS), Temperature and Humidity Control (THC), Vacuum System (VS), Water Recovery Management (WRM) and Waste Management (WM) systems [54]. Each of thse subsystems operates to satisfy the requirements for human survival set by NASA-STD-3001. For example, the ACS, AR and THC systems ensure that the NASA-STD-3001 Volume 2 section 6.3 on Water requirements [2]. For this mission, these subsystems will be restructured to function properly and efficiently in a lunar environment. The mass of the ECLSS is 2200 kg and the volume is 7.14 kg [54]. The ECLSS requires 2000 W of power [54].

8.5.1 Life Support Systems Reliability

Life support systems are an essential part of designing a habitat and must be reliable and redundant because the failure case for this system is the loss of human life. One of the top-level requirements is that the crew shall survive the mission. To achieve this, any singular failure cannot lead to loss of crew (LOS). The reliability of a system can be increased by spare parts and redundant systems. To decrease the likelihood of common cause failure, one or more of the redundant systems should be designed independently. The failure rate for the overall life support system is 0.0159 per day, meaning there is a 61.67% probability of failure over the 45 day mission [55]. Using Equation 8 with F being the failures per day and N being the number of spares, the failure rate with three spares and an additional ten percent was calculated to be 6.45% [54]. It can be decreased by adding a redundant system, which squares the failure rate for a value of 0.85% failure. This value is sufficient for the safety of the mission.

$$FailureRate = \frac{F^2}{N} + 0.1 * F \tag{8}$$

In addition to the redundant system, there will be independent backup systems in place to provide emergency life support to the crew. These life support systems will provide the crew with enough time to either fix the failure in the ECLSS or to return to the lander safely. One of these systems is the Backup Oxygen Candle System (BOCS). The BOCS produce oxygen and sodium chloride. Each unit has a mass of 12 kg, a volume of 0.01 m^3 and produces 3.4 kg of oxygen [53]. To have a contingency supply of 45 days of breathing oxygen for four men, the total mass and volume of the system is 496 kg and 0.44 m^3, respectively.

8.5.2 Thermal Control System

The conditions on the lunar surface are extreme and must be dealt with for a lunar base to be possible. The ridgeline between the Cabeus and Haworth Craters can reach temperatures of 55 to 265 K. A thermal control system will be in place to maintain a proper temperature range in the habitat for the survival and comfort of the crew and for the operability of electronics and other habitat systems. A thermal control system consists of heat collection, heat transport and heat rejection [56].

A passive and an active system were researched for use as the thermal control system on the base. Conventionally, passive systems such as heat pipes are utilized as the thermal control system for space missions. A passive system is one that has no moving parts and can function independently from a power source [57]. Heat pipes are essentially a small vapor cycle that uses a temperature gradient of a fluid to transport heat [58]. An average aluminum heat pipe radiator has a specific mass of 5 $\frac{kg}{m^2}$ and an emissivity of 0.85 [59]. An advantage of heat pipes is that each pipe is independent from another. In case of the failure of one heat pipe due to micrometeorite impact, the ability of the others to function properly is not affected [58]. An active system that was researched was the liquid droplet radiator (LDR). Active systems

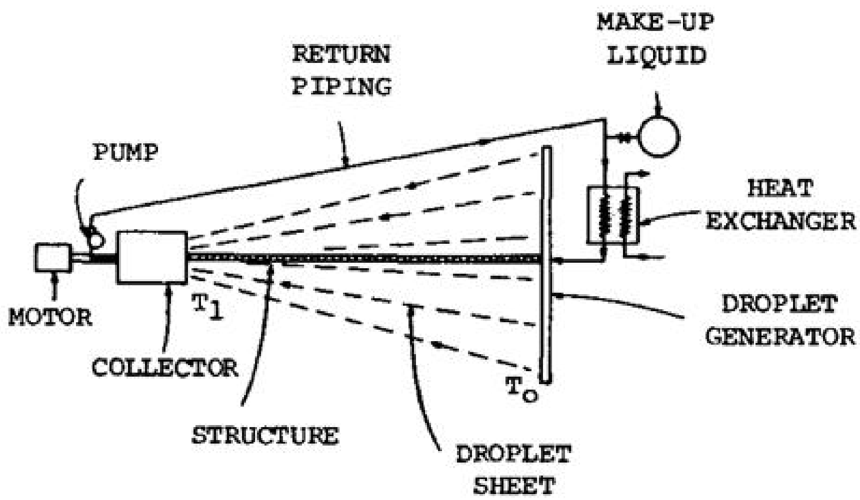

Figure 23. A diagram of the LDR. The droplets are generated and radiate heat as they are sprayed from the generator to the droplet collector. The droplets are then pumped through the returning pipe to the heat exchanger and the cycle begins again [60].

require input power and are on average more precise and effective than passive systems [57]. In this active system, droplets are generated and then are sprayed in a controlled manner across a region where the droplets radiate the heat. A LDR can be seen in Figure 23. The droplets are then collected and pumped back to the hotter part of the system [58]. The LDR can produce higher $\frac{kW}{kg}$ than the conventional heat pipe, as seen in Figure 24. LDRs are a lighter alternative to the heat pipes because the fluid mass is very small compared to that of the heat pipes. Also, because of the nature of the system, the radiating surface is not affected by micrometeorite impact and therefore requires less mass for protection [60]. The droplets in the LDR provide great amounts of surface area for radiation and can also be stored at a low volume for transportation [59]. Due to the better performance measures of power to weight ratio and specific mass and the ability to be stored at a low volume, the LDR was chosen as the thermal control system of the lunar base.

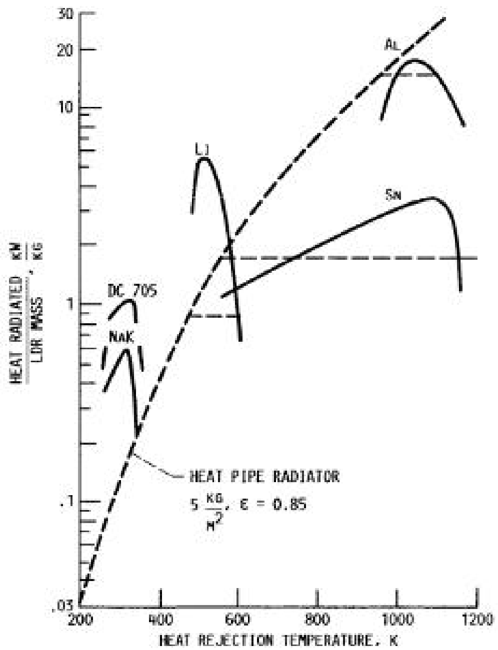

Figure 24. Comparison of the specific powers for LDRs with various fluids and a heat pipe radiator [59].

The LDR system for the mission was based off of specifications of a Lithium 100 megawatt system composed of multiple LDRs. In this system, the droplet radius is 50 μm and the velocity of the droplet is 22 $\frac{m}{s}$ [60]. The LDR is triangular with a base of 33 m and a length of 83 m [60]. The LDR power to weight ratio is 6.8 $\frac{kW}{kg}$ and the specific mass is 0.6 $\frac{kg}{m^2}$ [60]. One radiator has a mass of 815 kW and is able to dissipate 5550 kW of heat [60]. The radiator will exist in a vacuum. The hot droplets do not vaporize due to the low vapor pressure of Lithium[53]. Using a low vapor fluid, the evaporation of the fluid over the course of the mission is negligible [61]. The radiator will be able to handle the needs of the mission and allows for future expansion because more radiators can be installed. In order to assure that a thermal control system failure will not cause a mission failure, a redundant LDR will be present on the base.

8.5.3 ISRU for Oxygen Production

While the ECLSS system is ideal for atmospheric control, it does not by itself supply oxygen. While bringing breathable air to the Moon is possible, it is not viable nor cost-effective in the long term. Perhaps the most readily available source of oxygen on the Moon is what is already there, embedded in the lunar regolith. ISRU is by far the mission's best primary method of providing oxygen for the crew. According to NASA-STD-3001, the internal atmospheric pressure of the habitat shall be between 20.7 kPa and 103 kPa at all time. To generate enough oxygen without spending the extra budget to launch it to the Moon, In-Situ Oxygen Production is necessary for mission success. Knowing the volume requirements of the habitat, it is possible to find the exact amount of oxygen which must be produced to provide 50.6 kPa of pressure throughout the habitat. As stated in section 3.2 Inflatable Design, each subhabitat has a volume of 120 m^3. With 3 subhabitats, there are 360 m^3 of space that needs a ppO2 pressure of 50.6 kPa at most. Using the worst-case temperature of 18 °C, also provided by NASA-STD-3001, and the ideal gas law P*V = n*R*T, finding the total molar value n is possible. (103e3 kPa)*(360 m^3) = n*(287.053 J/kgK)*(300.15 K), making n equal to 217.96 moles of oxygen. The mass of a mole of oxygen is 32 g, thus the necessary mass of oxygen which shall be in the system at any time is 6.974 kg. According to NASA, the average person consumes about 0.84 kg of oxygen per day [62]. To maintain the 6.974 kg of oxygen in the atmosphere, along with the 151 kg used by a crew of 4 across the 45 day mission duration, it is safe to assume at least 200 kg of oxygen should be available at the start of the mission. Knowing the amount of oxygen necessary for the crew, the next step is how to generate and store this oxygen with ISRU.

8.5.4 Cryogenic Tube-On Oxygen Tank

A number of oxygen storage methods were considered for the mission, including a Linde Cycle Cryocooler, Inline Liquefaction, Conduction Cooling, and a Tube-On Tank. [63] As each method of storage could hold much more than the necessary amount of oxygen for the mission, the main criteria for selection were the system's mass and power draw. Table 19 below goes through these criteria for each system. The power usage is broken down into the input power and the cooling power necessary to prevent boiloff of stored oxygen.

Table 19. Summary of mass and power usage for each oxygen storage option. As the primary criteria are minimized mass and power, the lowest of both, Tube-on Tank, is the clear choice[63]

	Tube-on Tank	**Linde Cycle**	**Inline Liquefaction**	**Conduction Cooling**
Empty Mass (kg)	68	198	173	173
Input Power (W)	2873	2790	3570	4263
Radiator Power (W)	3250	3335	4100	4634
Total Power (W)	6123	6125	7670	8897

As demonstrated in 19, the tube-on tank method of storage was the clear optimal choice due having the least mass and total power of all the options. Figure 25 below details the inner mechanisms and cooling functions of the tube-on tank cryocooler. This tank can store 7.9 m^3 of liquid oxygen, which has a density of 1141 kg/m^3. Therefore, the tank can store 9014 kg of oxygen. This tank should be more than necessary for the 45 day mission duration, as well as cover for some possible future missions utilizing the habitat. Now, with a way to store the oxygen, there must be a way to collect the oxygen from in-situ resources. As regolith is the most common resource on the moon, and it contains many chemicals containing oxygen, some form of extraction must be used.

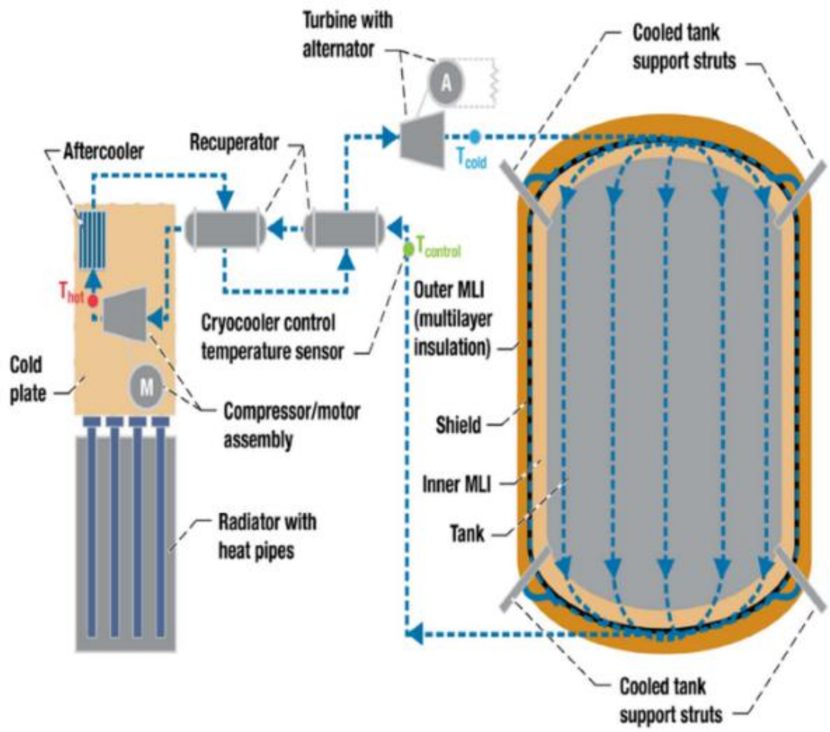

Figure 25. Inner workings of a Tube-On Cryogenic Oxygen Tank. Empty, the tank weighs 68 kg and can store 7.9 m^3 of liquid oxygen, which comes out to 9014 kg of oxygen. At a power cost of 6.123 kW, it can keep the oxygen cold enough to never boil off, offering high reliability and more than enough breathable oxygen for the entire 45 day mission [63].

8.5.5 NASA RESOLVE

While a few methods of oxygen extraction from regolith were theorized, by far the most developed and reliably tested was NASA's Regolith and Environment Science Oxygen and Lunar Volatiles Extraction Rover, also known as NASA RESOLVE[64]. The rover works by collecting lunar regolith and feeding hydrogen through the soil. This hydrogen would react with the Oxygen to create water, which would be stored in local tanks. The water generated is notably multipurpose, however must be purified before use due to its mixture with regolith. RESOLVE comes equipped with a non-regenerative purifier, although this may not be reliable enough to work long-term. With the water produced, a cathode feed electrolyzer separates the hydrogen and oxygen into base separate tanks, where the hydrogen can be reused and the oxygen is stored in gaseous form.

In all, RESOLVE gathers 660 kg of oxygen per year, or 1.8 kg per day. The oxygen, for the purposes of the mission, is transported to the cryogenic tube-on oxygen tank of the habitat. However, the output of RESOLVE is not able to fill this tank by itself unless it had 14 years to do so. The mission cost and volume budgets, fortunately, allow for 10 rovers to be brought to the moon for simultaneous oxygen production. These 10 rovers can fill the main tank in 500.8 days, which is enough time for the tank to be filled before the crew reaches the moon in 2031. Each RESOLVE uses 200 W of power, has a mass of 60 kg, and a volume of 3.8 m^3 . When including the cryogenic tank, which uses 6.12 kW to cool the oxygen to prevent boil off, has an empty mass of 68 kg, and a volume of 7.9 m^3, the total power usage, mass, and volume dedicated to ISRU oxygen and water production is 8.12 kW, 668 kg, and 37.9 m^3 respectively.

8.6 Manufacturing Timeline

The manufacturing timeline is derived from the manufacturing timelines of past missions. The primary objects of comparison are the ISS modules and the Bigelow B330 production times. Table 20 shows the build times for the Bigelow habitat and several ISS modules and their habitable volume as a general representation of their size. While the Sierra Nevada Corporation also have an inflatable habitat design, they have not yet disclosed any information on the production times. For the habitable volume provided, inflatable habitats have a shorter production time than the comparable rigid habitats.

Table 20. Build time for several inflatable and rigid structures and their habitable volume. The Bigelow B330 is listed three times to show the scaled production that Bigelow Aerospace claims to be possible. The first habitat would be produced in 48 months, the second in 28 months, and the third in 12 months [65].

Structure	**Build Time (months)**	**Habitable Volume** (m^3)
Bigelow B330 - First Hab	48	330
Bigelow B330 - First Hab	28	330
Bigelow B330 - First Hab	12	330
Destiny Module (ISS)	60	105
Zarya Module (ISS)	48	72
Columbus Module (ISS)	48	75
Leonardo Module (ISS)	24	31

Given that the Lunar Habitat uses the same technology and production processes as the Bigelow B330, it is likely that the production times are similar. While each of the Lunar Habitats is smaller than a single Bigelow B330, the maximum production time of 48 months per habitat was used for scheduling purposes. This is likely an overestimate of the production time but an overestimate helps prevent schedule slippages that may occur with an underestimate.

8.7 System Selection

Three types of primary habitat structures were considered for meeting the previously stated requirements: rigid, inflatable, and hybrid structures consisting of a mixture of inflatable and rigid components. Each of these structural options has its own advantages and disadvantages that must be accounted for in the overall design of the base. Research was done on each of these options to determine which would be the most suitable for mission's requirements and constraints.

8.7.1 Rigid Structures

Rigid habitation modules, like the ones used on the ISS, are a proven technology. They are primarily made of aluminum to decrease mass and the associated launch costs. In addition to an aluminum shell, the modules on the ISS also make use of several layers of Kevlar and Nextel shielding to provide ballistic protection from micrometeorites [66]. One of the primary benefits of using a rigid structure made of aluminum is that it very durable and has a long lifespan. Rigid structures are not as susceptible to deterioration from abrasion or radiation as some the fabric materials used in inflatable habitats.

Although rigid modules are a proven option for habitation, they possess several drawbacks. It is difficult to build rigid structures with large amounts of internal volume due to the form factor and mass of the structure. The mass of the structure severely limits the number of launch vehicles that can deliver the habitat and drastically increase the launch costs. Likewise, the form factor of the habitat must enable it to fit inside of the rocket fairing of the launch vehicle. This combination of factors makes assembling a large lunar base with rigid structures expensive due to the number of launches required.

8.7.2 Inflatable Structures

One of the newest innovations in habitation technology is the use of inflatable structures to decrease launch costs, increase packing efficiency, and to increase the internal volume that can be utilized by missions. The most notable company that produces inflatable habitats is Bigelow Aerospace. Their inflatable BEAM module which is currently attached to the ISS serves as a technology demonstration for their larger habitat options. The structure of the habitat is made of a mixture of different materials. The outer layers of the habitat make use of materials such as Vectran and Kevlar to provide ballistic and abrasion resistance. Inner layers are made up of thermal insulation and polyurethane bladders used to inflate the habitat. The innermost layer which is exposed to the crew is a Nomex fabric which is fire and chemical resistant. In total, the walls of the habitat are approximately 0.46 m thick with over 20 layers of material to provide ballistic, radiation, and thermal protection [67]. Using data obtained from the BEAM's stay on the ISS, inflatable habitats appear to have ballistic protection comparable to that of the station's rigid modules [68]. The fabric materials used in the inflatable shell were also found to drastically reduce secondary radiation when compared to the traditional aluminum modules [68].

While inflatable habitats possess many advantages, they also possess several disadvantages. Due to the nature of inflatable structures, it is quite difficult to add features such as windows or suit ports to the habitat. Inflatable habitats are also complicated by issues with materials. A large variety of materials are needed to make the inflatable shell of the habitat, but each material has its own strengths and weaknesses. For example, materials such as Vectran are useful for abrasion resistance and ballistic protection but are not resistant to radiation. This makes it difficult to determine the correct materials and layer pattern for the habitat.

8.7.3 Hybrid Structures

Prior to the development of fully inflatable habitats, NASA experimented with hybrid structures to combine the benefits of inflatable and rigid structures. NASA's TransHab program made use of a rigid central structural core made up of carbon-fiber composite materials [69]. The outer core was an inflatable shell made up of layers of Kevlar, Nextel, combitherm bladders, and Nomex cloth [69]. While the final design was chosen to use carbon-fiber composites for the structural core, an aluminum core was also considered to decrease costs and streamline production [69]. While carbon-fiber composites are lightweight and strong, they are also expensive and difficult to manufacture which is why many hybrid structure proposals since TransHab have chosen to use aluminum as the structural support material.

While hybrid structures were meant to combine the benefits of both rigid and inflatable structures, they actually ended up combining many of the disadvantages as well. Since the hybrid structures still make use of a large number of rigid structural components, they end up weighing more than fully inflatable habitats but less than fully rigid structures. Likewise, because they rely on an inflatable structure for the majority of the internal volume, they essentially have the same durability and lifespan as an inflatable structure instead of the more desirable durability of a rigid structure.

8.7.4 Trade Study

The trade study for the habitat architecture selection was conducted using a set of decision matrices. The weighting factors for the decision were safety, specific volume, and longevity. Safety incorporates the ballistic and radiation shielding abilities of each habitat. The specific volume is a measure of the average amount of internal volume that can be provided per habitat mass. This metric accounts for both the mass and volume of the habitat by favoring a design that can provide more volume for less mass. The longevity of each of the habitat options was also considered since the goal of the mission is to establish a long-term lunar base past the initial 45-day mission duration. The results of the decision matrix can be seen in Table 21.

Table 21. The inflatable habitat architecture decision matrix used to pick between inflatable, hybrid, and rigid habitat architectures. The criteria for selection were safety, longevity, and specific volume.

	Safety	**Longevity**	**Specific Volume**	**Score**
Weighting Values	0.60	0.20	0.20	N/A
Inflatable	0.43	0.20	0.65	0.43
Hybrid	0.43	0.20	0.23	0.34
Rigid	0.14	0.60	0.11	0.23

Rigid structures scored the lowest due to providing a very low amount of pressurized volume per structural mass when compared with the inflatable and hybrid architectures. Rigid structures also provide less radiation protection than either inflatable or hybrid structures. The inflatable habitat architecture scored the highest in the process due to having the most favorable safety and specific volume.

9. Communication and Data Handling

9.1 General Introduction

The purpose of the communication subsystem is to transmit and receive commands from ground stations and relay commands between other communication systems and return scientific data [70]. The Command and Data handling system processes all incoming data and controls the operations of the spacecraft.

9.2 Introduction for Communications

An ideal communication system is one that maximizes reliability while minimizing cost and errors. Prior to designing, it was important to establish necessary requirements. To do this, the communication requirements for space missions were obtained from NASA's Space Flight Human-System Standards Volumes 1 and 2 and from the Intentional Communication System Interoperability Standards (ICSIS). Additional factors for each communication system are reliability, operational simplicity, power usage, data storage, mass and costs. Reliability is defined as the ability of having direct to Earth (DTE) communications. This is deemed of most importance, as this is a long duration manned mission. Operational simplicity defined the complexity of a communication system. It needed to utilize advanced technologies while reducing the risk of faults. Telemetry data and data collected from the science performed during the mission needs to be transferred back to Earth efficiently so the storage capabilities of the satellite system are also considered. Reference was drawn from legacy and proposed NASA missions for determining mass, cost, and power requirements.

9.3 NASA and International Standards

There are several baseline requirements for the communications architecture which are defined by NASA's Space Flight Human-System Standards Volume 1 and 2. Volume 2 of NASA's system standards contains the primary requirements for mission communication systems. Of this list of requirements, there are several worth highlighting, but every requirement must be fulfilled, the primary ones are displayed in Figure 26. First, the communication system must provide audio, text, and video uplink and downlink capabilities to support crew performance and behavioral health [2]. For audio and video communications, the communications quality must be high. This involves controlling for operational parameters such as background noise, volume levels, digital encoding, spatial resolution, lag times, and other similar parameters [1]. In addition to quality controls, the communication system must also be intuitive and simple for the crew to use [71]. Each of these system requirements help ensure that the communications system is capable of providing high quality, reliable, and easy-to-use communications for the crew and mission. Volume 1 of NASA's system standards primarily defines the private crew communications that the system must be capable of supporting. Two-way voice and video communications are necessary for supporting private medical communications, private psychological conferences, and private family conferences. Private family conferences, psychological conferences, and medical communications are to be scheduled weekly, biweekly, and as required respectively [1]. The communication system must also support capabilities for crew relaxation, recreation, entertainment, news services, and social communication [1]. These requirements help support the physical and psychological health of the crew during the mission.

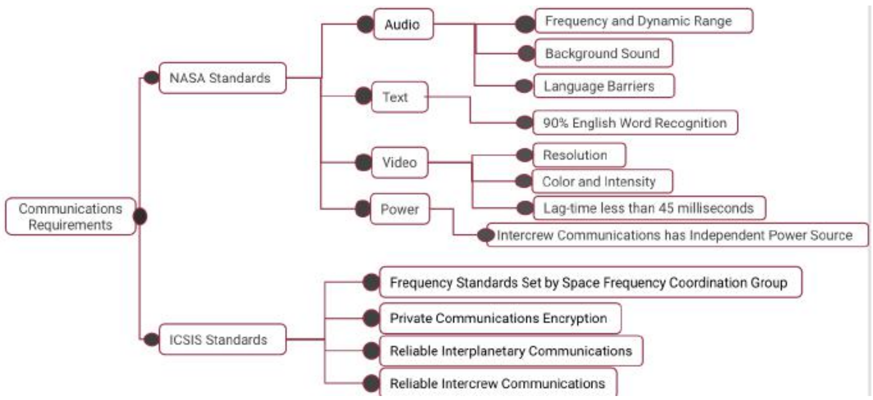

Figure 26. Communications Requirement breakdown flow chart showing requirements origins of requirements.

The International Communication System Interoperability Standards (ICSIS) provides detailed specifications for data requirements. The ICSIS states that all dimensions for data are required to be in the International System of Units (SI units). The Near-Earth frequency bands, also known as Near Space frequency bands, are used when the spacecraft is at most 2 million km away from Earth, which is allocated by the International Telecommunication Union (ITU). Deep Space frequency bands are used when the spacecraft is at least 2 million km away from Earth, also allocated by the ITU [71]. The bit numbering convention is also specified by the ICSIS. The convention is used in the identification process of each bit in an N-bit system. The first bit is defined as 'Bit 0' and continues counting up to 'Bit N-1' with the Most Significant Bit (MSB) being the first bit to be transmitted [71]. Data fields are generally grouped into 8-bit words, known as octets, in a manner of conforming with the data-communications practice. In accordance with the Consultative Committee on Space Data Systems (CCSDS) convention, all spare bits are set to zero.

9.4 Intro to Radio Frequency and Lunar Constraints

In addition to meeting the requirements above, the lunar surface possesses some unique challenges. The geological environment of the south pole is an area composed of craters. Many of the craters are exposed to periods of irregular sunlight with only parts of the rim in continuous sunlight. This would prove challenging for a rover powered by solar panels, for it would be limited in only surveying areas with suitable sunlight.

Due to this uneven terrain and Earth-Moon geometry communication systems must be designed to overcome these limitations. For example, a gimbaled rover antenna attempting to communicate with Earth in this area would be restricted to an approximate 10° range of motion [72]. Additionally, a transceiver would be susceptible to interference from many factors. The curvature of the Moon's surface can contribute to communication interference, resulting in destructivity and noise. Described as the Fresnel Zone, this happens to any communication signal over a 10 km range. For DTE communication the transmitter and receiver must be in direct line of sight with each other, this is especially challenging over large distances [2]. Distance causes time delays and allows signals to fade, this is known as space loss. Radio waves travel at the speed of light and take approximately 2.56 seconds to travel to the Earth and back. Often times these challenges are mitigated by increasing the size or power capabilities of the antenna. Recently the scientific community had been looking into using optical communications as an alternative to the standard RF communication. The overall signal and strength of a communication system is calculated by using a link budget in decibels. The link budget uses an equation to relate all the parameters needed to compute the signal to noise ratio, Equation 9.

$$\frac{E_b}{N_o} = \frac{P * L_t * G_t * L_s * L_a * G_r}{k * T_s * R} \tag{9}$$

Where P is the transmitter power, L_t is the transmitter to antenna line loss,G_t is the transmit antenna gain, L_s is the space loss, L_a is the transmission path loss, G_r the receiver antenna gain, k Boltzmann's constant, Ts is the system noise temperature and R is the data rate.

The factors that contribute to the strength of the signal are the transmitter power, the data rate and antenna gains of the transmitter and receiver respectively. The remaining factors contribute to noise and signal interference. Loss can occur due to inefficiencies between the transmitter and antenna, the distance between transmitter and receiver, and from atmospheric interference. For each communication relay and system the link budget will be calculated in order to determine the feasibility of the design. RF communication operates through the use of radio waves that are set to specific wavelengths. The deep space bands that are most frequency used for space missions are S, X, K and Ka bands. Table 22 displays the bands that will be used along with the frequencies for each.

Table 22. Most commonly used RF bands that will be used for this mission and their respective frequencies.

Band	Frequency
S band	2- 4 GHz (7.5-15 cm)
X band	8-12.5 GHz (2.4 – 3.8 cm)
K Band	18-26.5 GHz (1.1 – 1.7 cm)
Ka Band	26.5-40 GHz (0.75-1.1 cm)

With S-band, the frequency range for satellite downlinks are between 2.2 to 2.3 GHz [72]. S-band is reserved for space to Earth transmissions. In comparison, Ka-bands are in the range of 26 to 40 GHz [73]. According to JPL, the

Ka-band is now considered to be the "spectrum of the future" [72]. There are many benefits to using Ka-band. There is a great cost efficiency, they have smaller terminals, and a greater resilience to interference. Some of the challenges that are faced with Ka bands are that they have a large rain attenuation as well as facing the difficulty of meeting satellite interference regulations [72]. In comparison, S-band also has many advantages. Some of which are that it performs well in adverse weather and it has better ground penetration [74]. For every advantage, there is a disadvantage of S-band, including lower throughput, narrow band spectrum, and a larger antenna size compared to other bands [75].

9.5 Introduction to Optical Communication

One solution to the ever increasing amount of data requirement is to use optical lasers to send signals instead of radio waves. There have been several recent missions such as the Lunar Laser Communications Demo (LLCD) that successfully tested the potential of optical communication [76]. Light is used as the medium for data transmission using highly reliable infrared lasers. Laser waves are much shorter than radio waves and allow data to be transmitted across narrower beams, which directly reduces the amount of possible loss. The laser from the optical terminal points directly to the recipient ground station telescope with extreme accuracy and precision, thus reducing the need for a large antenna as compared to RF. The LLCD mission proved that downlink data rates as high as 662 Mbps are attainable. Onboard the spacecraft was a 10 cm telescope in the optical module, a 0.5 W laser at wavelength of 1550 nanometers that used a 16-ary PPM. Uplink capabilities were at 20 Mbps and used a 4-ary PPM [77]. This method of communication is optimal for scientific missions where large amounts of data are being collected and returned to Earth. Having optical capabilities will help towards setting up high frequency and strong signals for future manned missions.

9.6 Redundancy Outline

Communications between an Earth based command center and a satellite is always important for keeping that satellite functional and executing station-keeping maneuvers; however, communications is even more important if there are human lives relying on them. Maintaining continuous communications between Earth and the lunar habitat is the most important aspect of making sure astronauts are safe and can perform their duties. The use of astronauts on any mission means that additional communications capabilities such as two-way audio/video communications with Earth must be included for private medical or psychological conference and communication with family members. Since these astronauts' physical and psychological health relies heavily on being able to maintain constant contact between them and Earth, there have been three communications redundancies included in this mission design. Under ideal conditions each communication method will maintain constant contact with both Earth and the lunar habitat, however if there is an abnormality such as a solar flare that inhibits the communications signal, or if a relay goes down due to a mechanical failure there will be two other active relays, ensuring that the astronauts never lose contact with Earth. These relays will each be outfitted with radio frequency (RF) communications and some will have optical communications capabilities as well. The link architecture showin in Figure 27 displays each communication relay and capabilities of each. The primary form of contact is a relay housed on top of a mountain close to the habitat called Malapert Mons. This mountain is at an elevation of 5 km allowing it to maintain a constant view of all activities on the lunar surface as well as never losing view of Earth, giving it constant contact with both astronauts and Earth control. The Malapert station will be outfitted with the optical MAScoT terminal, shown in Figure 30, and an RF lunar communications terminal (LCT), there will also be a redundant LCT located nearer to the habitat for emergency communications. The secondary communications relay is a constellation of satellites in polar orbit around the Moon; this constellation features three satellites in circular orbit separated by 120 degrees meaning the habitat will always be in view of at least one satellite. Each of these satellites will have both RF and optical capabilities. The final method of communication is a satellite orbiting in the Earth-Moon LL4 Lagrange point. This location was chosen because it continuously affords the satellite a view of both Earth and the habitat. This satellite will only have RF capabilities. This specific communications architecture was chosen because it allows for high data rate transfers between the Malapert relay or the polar satellites and Earth ground terminals when the Earth's position and atmospheric conditions allow for optical ground terminal access. When this access is restricted then any of the relays can still communicate between the habitat and Earth using RF, but this eliminates the benefits that come from optical communications. Together this means that there will always be contact between the habitat and the Earth, and during certain access windows those communications are capable of high rate data transfers.

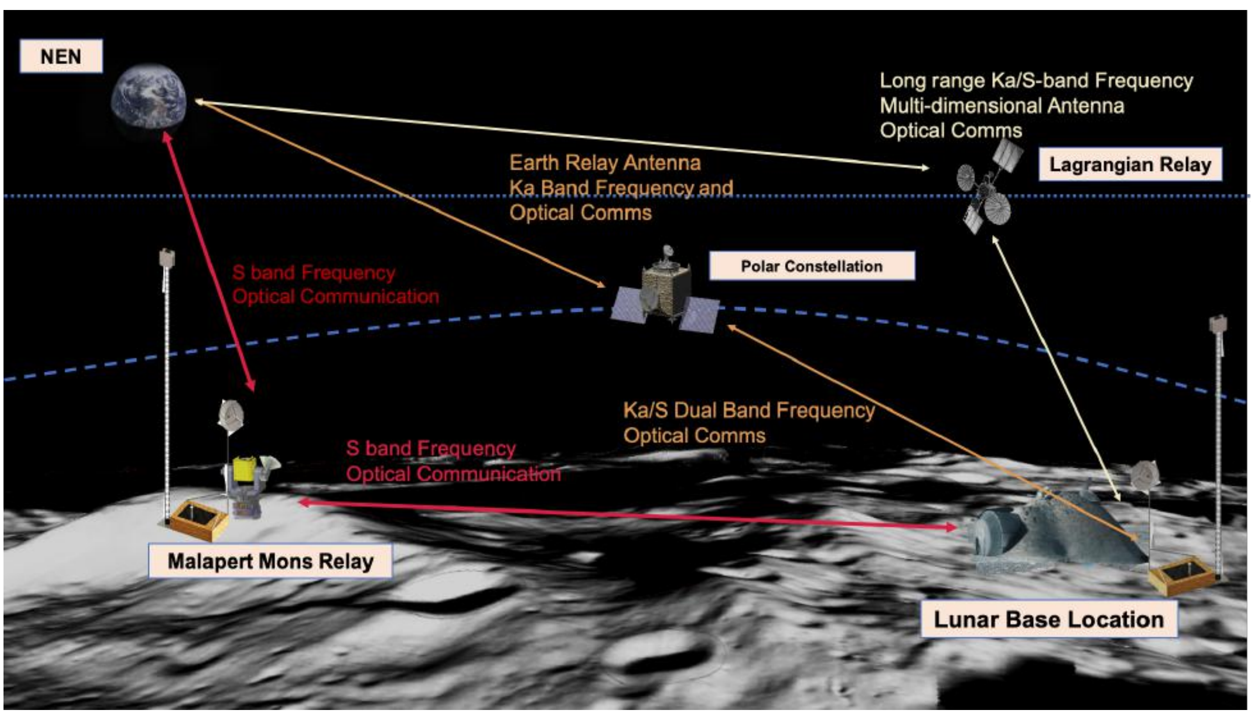

Figure 27. Communications Link Architecture: Depicting the three redundant communication architectures with the locations and frequency operations of each.

9.7 Polar Satellites

The secondary mode of communication relies on a constellation of three satellites separated by 120 degrees in a polar orbit with an inclination of around 90 degrees around the Moon. These three satellites will be in a circular orbit with an altitude of 10,000 km. With this orbit and the antenna on the satellite being housed on a gimbal capable of rotating +/- 90 degrees there will be complete coverage of the habitat by at least one satellite at all times and constant view of the Earth. This means that no matter how the satellites are positioned along their orbit, the habitat will always be able to communicate with at least one of them, and that satellite can then relay that communication to Earth. Another benefit of having three of these satellites is that the antenna housed on the lunar surface should never have to rotate to communicate with at least one of them, cutting down on power usage and communications transfer time. The design of these satellites will be based on NASA's proposed COMPASS Lunar Relay Satellite (LRS) program, which lays out plans for a constellation of satellites used for communications and data transmission at long distances in space. Each satellite will include 3 amplifiers, 2 for X-band transmission at 100W and 1 for Ka-band transmissions at 35 W as well as 2 transponders which serve as a primary and a back-up for long range communications. On the front face of the satellite there is 1-m dish with an antenna that provides Ka/K band (23/26 GHz) communications with the moon. In addition to the Ka/K band, the antenna is capable of integral Cassegrainian S-band (2 GHz) communications, which it uses to aid in navigation. With these characteristics, the polar constellation will ensure that if the Malapert Mons relay should be in-operational at any time, there will be an immediate and effective backup relay providing constant contact between the habitat and Earth.

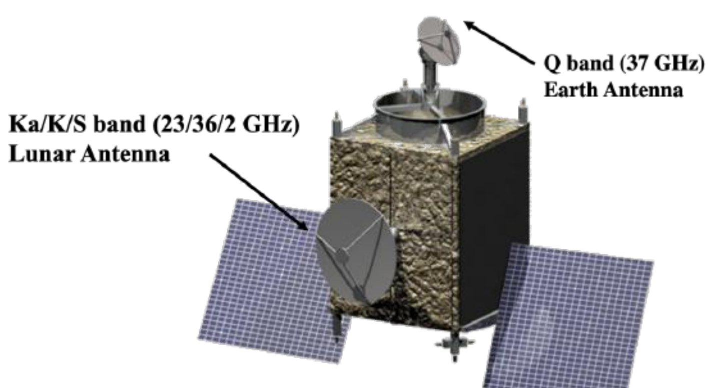

Figure 28. Model of the Polar Satellite with labeled primary communication bands based of NASA Compass Program.

9.8 Malapert Station

The Malapert Mons relay will serve as the primary communications relay for the base camp and mission. Malapert Mons is a mountain located at the lunar south pole approximately 50 km from the base camp site. Due to the mountain's location and height of 5 km, it has constant line-of-sight communications access to Earth. This allows for a relay to be placed on top of the mountain to serve as the primary communications link for the base camp and south pole region. Since this mountain overlooks the majority of the south pole region, it will also be able to support the majority of extravehicular and rover activities that occur in the area. The relay itself will utilize solar panels and batteries for power since Malapert Mons receives near constant illumination conditions [78].

The communications station at Malapert will utilize a Lunar Communications Terminal (LCT) along with a modified Small Deep Space Transponder (SDST) for direct-to-Earth communications and communications with the orbiting satellites. The modified Small Deep Space Transponder will operates on S and Ka frequency bands [79]. The transponder is capable of data rates of up to 6 megabits per second [79]. The Lunar Communications Terminal consists of a pallet containing the communications and avionics equipment, surrounded by a thermal control system (radiator), a deployable 10 meter tower, and a secondary deployable dish to provide communications between the habitats, polar relay satellites, and Earth [80]. In addition to radio frequency communications, the Malapert relay will also incorporate laser communications using the Modular, Agile, Scalable Optical Terminal (MAScOT). MAScOT is capable of achieving direct-to-Earth data rates of 80 to 250 megabits per second and forward data rates of up to 20 megabits per second [80]. Additional benefits of the MAScoT terminal lie in its modular design. The architecture of the terminal is fully modular which allows for the telescope, latch and gimbal, and back-end optics to be developed independently as long as subassembly interface requirements are followed [81]. The design itself is also scalable in that larger telescopes can be accommodated without significant changes to the terminal's subassemblies [81]. The optical payload will house a space terminal with will contain an optical module, modem, beacon and controller electronics [82].

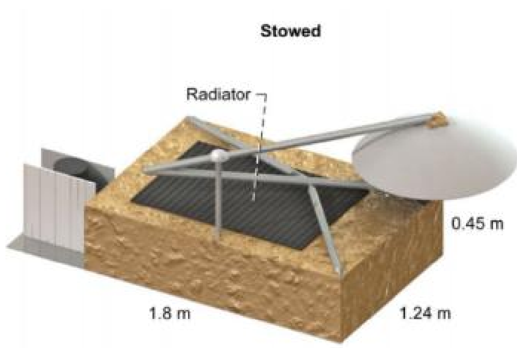

Figure 29. Diagram showing stowed Lunar Communications Terminal. This is how it will be packaged in spacecraft to protect instrumentation.

9.9 Lagrangian Satellite

If the Malapert and Polar relays are unable to send a signal, data will be routed through a satellite located at the fourth Earth-Moon lagrange point. This satellite will perform similarly to NASA's third- generation Tracking and Data Relay Satellites (TDRS) by relaying data, voice and video communications from the lunar station back to Earth. A trade study was performed when choosing between the LL4 and LL5 points. Both regions provide a stable orbit for a satellite

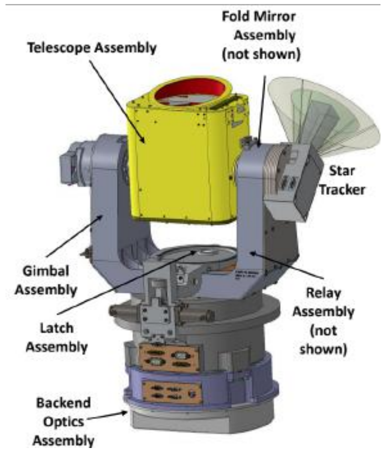

Figure 30. MAScoT Optical terminal that will be at Malapert Mons.

reducing the need for stationkeeping and both are areas of scientific interest because they will provide access to particles trapped in these regions since the formation of the Earth and Moon. The LL4 point was chosen because a satellite at this location will maintain a constant view of both the lunar habitat and the Earth, while the LL5 point only has an intermittent view of the habitat. Figure 31 shows the location of the satellite with respect to the Moon and Earth.

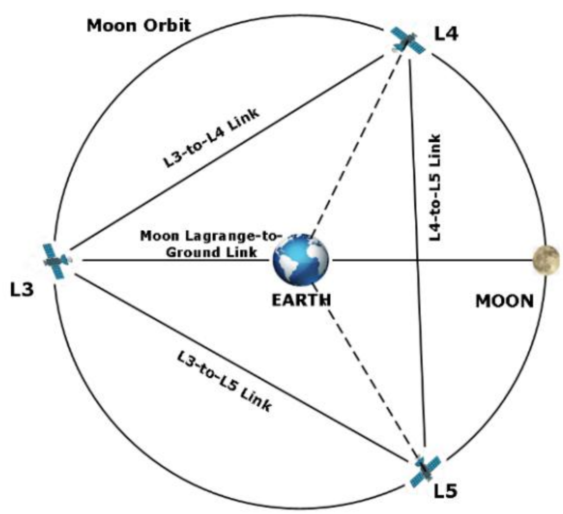

Figure 31. Diagram showing the Earth-Moon Lagrangian Constellation and position of satellite.

This satellite will contribute to the lunar network of communication satellites that will support future missions. It's dimensions are similar to those of the polar satellites but modified to contain a cosmic dust analyzer to collect data on the type of particles caught in the LL4 orbit. Data will be primarily relayed through S-band frequencies through the use of a large high gain antenna [83] this includes immediate uplink and downlink for telemetry and command data between the base and Earth.

9.10 Ground Stations

The Near Earth Network (NEN) is a network of Earth-based ground stations located around the world to provide constant communications and tracking services to missions operating in the near earth region including to lunar orbit and the lunar surface [84]. The overwhelming majority of Near Earth Network ground stations support both S-band and X-band radio frequency communications but only one ground station supports Ka-band communications. The White Sands Ground Station in New Mexico is the only NEN ground station that supports Ka-band communications [85]. Therefore, the S-band and X-band ground stations will serve as the primary points of contact for the lunar base camp. The Ka-band station at White Sands will serve to augment communications and increase data transmission rates in the window that it has contact with the lunar base.

Different ground stations will be used for receiving optical communication, which will be used solely for receiving large amounts of scientific data from the mission. Currently the number of ground stations that are capable of receiving optical transmission is limited, but as the optical network is developed around the moon, the ground stations on Earth will also evolve. The Lunar Lasercom Ground Terminal (LLGT), located at Nasa's White Sands Complex in New Mexico, will be used as the primary optics receiver. A backup ground station which is still in the developmental phase and currently has limited functionality is the Optical Communications Telescope Laboratory (OCTAL) located in Table Mountain, California.

The LLGT was selected as the primary ground station because it is scalable and equipped with the necessary equipment to receive optical communications at the data rates they will be sent, and it was successful when used for the LLCD mission. It utilizes four 15 cm diameter refracting telescopes for uplink and four 40 cm diameter reflecting telescopes for downlink [86]. All eight downlink telescopes are positioned together on a single gimbal, and signals that the telescopes receive are sent to the control room through fiber optic cables to be processed [87]. The biggest risk involved with optical communication is the extreme amount of precision required from the system, therefore the entire telescope station is enclosed in a temperature controlled fiberglass dome for further protection and stabilization. Despite the best efforts to stabilize the ground stations, the atmosphere is a little more difficult; storms, cloud cover, or even high enough humidity can interfere with an optical signal, which is why only non-essential high data rate scientific transfers will be made via optical signal. This way its large data capabilities are utilized, while never risking the crucial communications between Earth ground stations and the lunar habitat.

Figure 32. Diagram of the Lunar Laser Ground Terminal that shows all 8 telescopes,followed by an depiction of the telescopes encased in fiberglass layer.

LLOT will serve as the backup ground station and later will serve as the primary station once the facility has been fully established. LLOT is part of the NASA owned Optical Communications Telescope Laboratory (OCTAL) [88]. The terminal contains a 1m diameter telescope that is capable of 34 Mb/s downlink. A 1558 nm laser beacon will be used to transmit from OCTL to the lunar optical terminals, which then will use an acquisition and tracking detector to locate the beacon and return a 1550 nm downlink laser back to Earth [88]. The system computer will mainly process the incoming data once the data has been fully received in order to save on cost and power. However, the system will still provide limited real-time diagnostics to ensure that the signal was correctly recorded and received.

9.11 Data Rates

Table 23. Data Rates that will be used to send signals to and from Earth for the LCT and Polar Satellites.

Band	**Description**	**Applicable System(s)**	**Data Rates (without explicit margin added)**		
			Low Rate (Mbps)	High Rate (Mbps)	Total Rate (Mbps)
Ka/S	Aggregate Peak Rate to Earth	PS and Earth Ground System	3.9	151	155
Ka/S	Aggregate Peak Rate from Earth	PS and Earth Ground System	1.1	66	67.1
Ka/S	Aggregate Peak Rate Up to PS from Lunar Surface	PS and LCT	6.4	216	222.4
Ka/S	Aggregate Peak Rate Down from PS to Lunar Surface	PS and LCT	6.1	141	147.1
Ka/S	Aggregate Peak Rate Across Lunar Surface	LCT	8.7	143	151.7

Table 24. Optical Data Rates from the Laser Lunar Ground Station to MAScoT Terminal

Description	**Data Rates**
Downlink (from Satellite to LLGT)	622 Mbps down to Earth 177 Mbps to Earth through thin clouds
Uplink	20 Mbps

9.12 Command and Data Handling

Each satellite will have a command and data handling (CDH) subsystem which will operate the spacecraft autonomously. This includes automatic state switching and execution of time-tagged commands. An onboard spaceflight computer and solid-state recorder will manage data and perform spacecraft housekeeping [89]. The computer will run flight software, provide tracking and telemetry data, monitor spacecraft health and protect faults from spreading. Fault handling is defined as the ability to debug and operate in safe mode when certain errors are detected to prevent the propagation of errors to the entire spacecraft [90]. The computer will have 128 megabytes of high-speed random-access memory and a X2000 RAD 5500 microprocessor. All components will undergo rigorous testing to perform better in the space environment. The RAD is the successor to the one used on the Mars Reconnaissance Orbiter. Scientific data collected by the Lagrangian satellite will be stored on the solid-state recorder. It is capable of storing up to 160 gigabits of information, until it can be sent to Earth for further analysis. An atomic clock onboard each satellite will assist with navigation and signal accuracy. The Lagrangian satellite will be launched first and will house the master clock, to which all subsequent mission computers and atomic clocks will synchronize to [91]. The CDH system on the lunar surface will operate at a higher and more complex level because it is responsible for processing all incoming signal, overseeing all lunar operations and sending outgoing data back to Earth. The computer system will focus heavily on computation and logic and will operate similarly to the system for the upcoming Orion mission [92]. Operations will be prioritized through the use of time-triggered gigabit ethernet [92] which allows users to prioritize data that will be sent through the onboard network. Time-triggered data such as anything relating to vital systems such as navigation and life support will have the highest priority and is guaranteed bandwidth. This is especially necessary for a long duration manned mission such as this one, where in case of an emergency situation the crew can communicate with ground control immediately. Rate-constraints data is data that is essential but not time constrained and sent when time-triggered data is not present. Lastly non-critical data categorizes non-critical tasks such as crew videoconferencing and uses remaining conventional ethernet and will be delivered with remaining bandwidth.

10. Power Systems

10.1 Overview

Reliable surface power is vital to support base operations, maintain communication with the Earth, and power life support systems. A trade study was conducted to choose the primary power system. Solar and nuclear power options were considered. The nuclear option was chosen as the superior power system and a description of this system is included.

10.2 Requirements

10.2.1 Needs

The power generation and storage system must be able to operate reliably in the lunar environment for the duration of the mission and for longer periods of time to support future habitation of the lunar base.

10.2.2 Alterables

The RFP emphasizes base expandability and extensibility to future missions. With this in mind, the base camp's power system should be expandable to accommodate greater power draws from future base expansions. The power system should also be extensible to future missions, such as base camps at other locations on the lunar surface, Mars colonies, or other interplanetary missions. Therefore, the choice of power system should not be constrained to use at the chosen location at the lunar south pole.

10.2.3 Constraints

The power system must be able to be deployed reliably on the lunar surface. It must also be small enough and light enough to fit inside the launch vehicle and be landed on the lunar surface.

10.3 Systems With Independent Power Systems

Many of the lunar base camp systems include their own independent power system. The three polar orbit communication satellites and the communication satellite located at the Lagrange point all draw power from onboard solar panels. NASA-STD-3001 10.5.3.8 requires that all communication systems must have independent power sources, so the ground station communication system and the Malapert transponder are powered by refillable hydrogen-oxygen fuel cells similar to the fuel cells used on the Space Shuttle. The ATHLETE rovers are powered by onboard solar panels and refillable hydrogen-oxygen fuel cells. Finally, the Cratos rovers are powered by onboard solar panels.

10.4 Power Requirements

Five systems were identified that have large power draws and no form of independent power system: habitat modules, the life support systems, habitat thermal control, the Chariot rovers, and the ISRU system consisting of ten oxygen collecting rovers, and a cryogenically cooled oxygen storage tank. The expected power draw from all of these systems is 23 kW. A breakdown of the individual power draws from each of these systems is shown in Table 25. Safety precautions will be implemented to make sure that the power system is not overloaded by too many systems drawing power at once.

Table 25. A breakdown of the estimated total 23kW power draw on the primary power system.

Component	Power [W]	AIAA Growth Factor [%]	Power With Growth Factor [W]
Life Support	1920	15	2300
Thermal	1200	15	1400
ISRU	8120	15	9400
Habitat	8500	15	9800
Chariot Rovers	150	15	180
		Total	23000

10.5 Assumptions

The Chariot rovers were assumed to have a 25% duty cycle as the astronauts will not be using the Chariot rovers constantly. Components in the habitat associated with things like cooking, washing, waste management, etc. were given duty cycles based on their assumed usage. In the event of a total power failure, it was assumed that the crew could evacuate the base in 48 hours.

10.6 Considered Power Architectures

10.6.1 Solar Power

Solar panels have been the backbone of spacecraft power systems since the early days of space exploration. They provide consistent power in sunlight, have a low failure rate, and a long life time. Generally solar panels are used in conjunction with rechargeable batteries to provide power when the spacecraft is in shadow and the solar panels produce no power. At the lunar south pole there are many areas which receive sunlight for extended periods of time, as much as 89% in some locations [7]. Solar power systems are very reliable and have extensive flight heritage making solar power an attractive prospect for lunar surface power.

10.6.2 Batteries

Since solar panels only provide power while in sunlight, energy storage methods, such as batteries, are required to supply power while the panels are in shadow. The lunar day is 29 Earth days long, so most locations on the lunar surface are exposed to roughly 14 Earth days of sunlight and 14 Earth days of darkness. Some locations at the lunar south pole receive consistent sunlight for extended periods time, but periods of eclipses for up to 120 hours can occur [7]. With a power draw of 23 kW, 14 tons of batteries would be required to power the base through an eclipse using state of the art rechargeable LiP batteries [93]. This high mass due to batteries introduces considerable difficulties for manufacturing, deployment, and safety.

10.6.3 Fuel Cells

Closed loop hydrogen fuel cells are a form of energy storage which could be used in place of batteries. Open loop hydrogen fuel cells have been used on the space shuttle to provide power to onboard electrical systems [94]. Hydrogen fuel cells create energy by combining molecular hydrogen and oxygen into water and converting the resultant heat into electrical power. Closed loop hydrogen fuel cells take input power to separate the water back into hydrogen and oxygen so that it can be reused for power generation at a later date [95]. Lab tests have shown that hydrogen fuel cells can store as much as six times as much energy per unit mass as traditional rechargeable batteries [95].

10.6.4 Fission Power

Nuclear fission is a well established source of power in terrestrial applications, and has been used in the form of radioisotope thermoelectric generators (RTG) for many years in space applications. Nuclear reactors with turbine generators are commonly used on Earth, but have never been used in space before. The massive power requirements of terrestrial bases has prompted much interest in fission generators in recent years. NASA is currently developing the 1 kW "Kilopower" reactor for use in both terrestrial bases and spacecraft. However, Kilopower uses highly enriched uranium as its fuel which poses security and political concerns as it is used in the production of nuclear weapons [96].

Due to these issues, low enriched pellet bed reactors were considered instead. Pellet bed reactors (PBR) use spherical uranium and graphite pellets as the reactor fuel. The advantage of using pellets is that the fuel is more separated as opposed to a traditional uranium fuel rod making it less prone to overheating [97]. Pellet bed reactors can operate with various fuel enrichment levels with higher enrichment leading to less reactor mass [97]. These types of reactors also do not release harmful radiation into the environment if there is a launch failure and the reactor is dropped into the ocean [97]. These properties make the pellet bed reactor worth considering for the power architecture.

10.7 Power System Selection

The power architectures considered were: solar panels and batteries, solar panels and hydrogen fuel cells, one 23 kW PBR and batteries for backup, one 23 kW PBR and hydrogen fuel cells for backup, and one 23 kW PBR with an additional redundant 23 kW generator loop. The initial estimate for surface power requirements from the power budget was 23 kW. The solar options require battery or hydrogen fuel cell power for a 120 hour eclipse. The nuclear and battery/hydrogen fuel cell backup options assume that batteries/fuel cells will power the base long enough for the crew to abort the mission in the event of a reactor failure. The base can operate on a minimum of 5 kW in the event of an emergency power failure by providing power only to components essential to crew survival. The final architecture utilizing a PBR with an additional redundant generator loop remains fully operational if one of the generator loops fails. The crew has the option to abort the mission or continue the mission with accepted increased risk in the event of a single power generator failure.

The key considerations in choosing an appropriate power architecture were: specific energy [W/kg], production cost [$], reliability, development costs, and complexity. High specific energy is favored as launch and landing costs per kilogram to the lunar surface are extremely high. Also, for extensibility to future missions to Mars, low specific energy systems would be prohibitively expensive. Nuclear options have higher specific energy than solar options because solar options require massive batteries to survive the 120 hour eclipse. Low production cost is favored as this keeps the overall mission cost lower and reduces the cost to future missions that utilize the same power architecture. Solar options are more expensive than nuclear options because of the sheer amount of solar panels and batteries required for the solar options. Reliability is one of the most important considerations because the power architecture supports all life support and habitat functions, so a loss of power puts the crew at significant risk. Solar panels and batteries have the highest reliability because they have extensive flight heritage. Hydrogen fuel cell options also have high reliability because they do not have a single failure point like nuclear options do. Nuclear PBR options have lower reliability because they have not been used in space before. The reliability of the two generator system is deemed acceptable as the system is fully redundant.

The estimated development costs associated with each power architecture was based off of the TRL level of the technologies involved in each power architecture. Development cost to raise the TRL of specific technology is high, but it is only a one-time cost so future missions relying on similar power architectures will not incur the same costs. The solar and battery option has the highest TRL, and thus the lowest development cost, because this technology has flown many times. Hydrogen fuel cells are at TRL 6, but they are a relatively small system, so the development costs are assumed to be reasonable. Space based nuclear power is TRL 5 based on the Kilopower system. The development cost of a PBR were assumed to be high due to the large size of the system. The final consideration, complexity, takes into account the difficulty of deploying the system on the lunar surface. Options involving solar power are exceedingly complex as they require many solar panels to be deployed on the surface, and massive batteries or hydrogen fuel cells must be landed on the surface as well. The nuclear option with battery backup is of intermediate complexity as this option still requires landing massive batteries on the surface, but it does not require deploying solar panels. The two generator nuclear option and the one nuclear reactor with hydrogen fuel cell backup both have low complexity because only a few components need to be deployed, and the mass of the components is not exceedingly large. Table 26 shows the result of the trade study on power architectures.

Table 26. The power systems trade study. This study finds that a nuclear reactor with one redundant generator loop is the best power architecture considered.

	Energy Density	**Production Cost**	**Reliability**	**Development Cost**	**Complexity**	**Score**
Weighting Factors	0.09	0.04	0.41	0.05	0.41	
One reactor and One redundant generator loop	0.58	0.34	0.19	0.17	0.39	0.31
Solar and Batteries	0.03	0.00	0.52	0.30	0.04	0.25
One reactor and H cell backup	0.06	0.31	0.04	0.17	0.37	0.20
Solar and H fuel cells	0.24	0.00	0.19	0.20	0.06	0.13
One reactor and Battery backup	0.07	0.34	0.05	0.17	0.13	0.11

It can be seen from the Table that the 25 kW reactor with a redundant 25 kW generator loop is the best architecture with a high score in most categories.

10.8 Pebble Bed Reactor Power System Overview

The PBR reactor system is modeled after a baseline PBR system published by The University of New Mexico [98]. The primary components of the PBR system are: the reactor core, a He-Xe Brayton cycle generator, and Na-K heat rejection panels. Figure 33 shows the configuration of one of the power generation loops. The second redundant generator loop will be identical to the first and both will share the single PBR core.

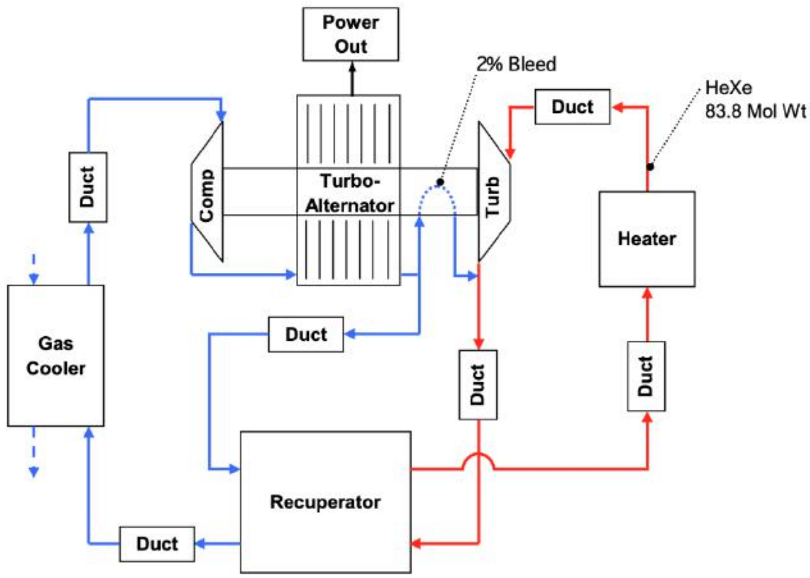

Figure 33. A diagram of one of the generator loops of the power system[99]. The block labeled Heater in this diagram is the PBR core which supplies heat to the system, and the Gas Cooler block dissipates heat through the NaK heat rejection panels.

The reactor core generates heat through nuclear fission. The core has heat pipes to carry heated He-Xe from the core to the Brayton cycle converter. A mixture of He and Xe was chosen as the working fluid because it offers a high heat transfer coefficient along with high molecular weight[98]. A diagram of the PBR reactor core is shown in Figure 34. The mass of the reactor core was estimated to be 450 kg based off of the baseline reactor design[98].

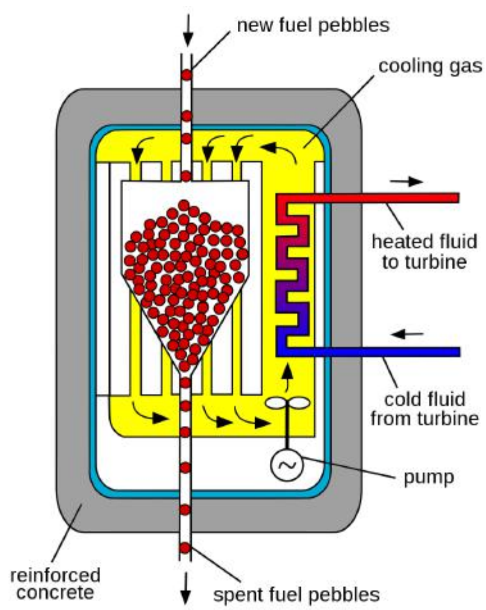

Figure 34. A diagram of a pebble bed reactor[100]. The PBR used on the lunar surface will not drop spent fuel pellets out of the bottom as shown in this diagram. All fuel pellets will remain in the reactor core for the lifetime of the reactor.

The Brayton cycle generator produces electrical power from the heated gas from the reactor core. NASA has done considerable testing with Brayton cycle converters and has shown success in using them for power generation in space applications[99]. Brayton cycle generators are the standard method of power generation for terrestrial power stations. The mass of the Brayton generator was estimated to be 440 kg, 220 kg per generator, based off of the baseline reactor design[101].

The Na-K heat rejection panels dissipate waste heat from the system into space. The design for the heat panels is based on the heat rejection panels used on the International Space Station. The advantage of this design is that the panels can be placed in a stored configuration for launch and then extended on the lunar surface. Figure 35 shows the deployment sequence of the thermal panels for the International Space Station which will be employed in the PBR system's heat rejection panels. The mass of the heat rejection panels was estimated to be 600 kg, 300 kg per generator loop, based off of the baseline reactor design[101].

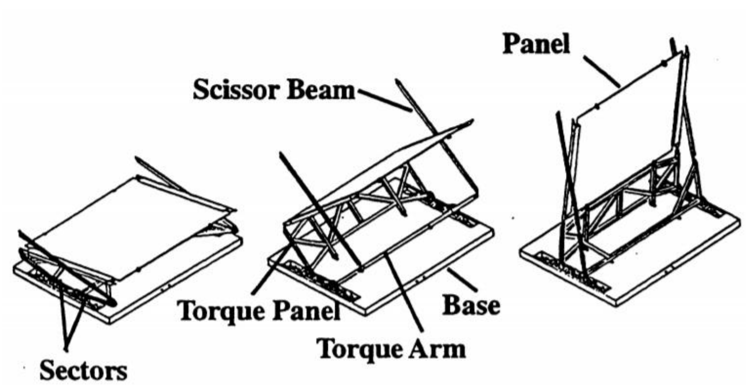

Figure 35. A diagram showing the deployment sequence of the heat rejection panels for the PBR reactor [102]. This is the same way that the heat rejection panels on the ISS were deployed.

CAD Models of the PBR system in deployed and stored configurations are shown as Figures 36 and 37 respectively. The reactor core must be stored 2 m below grade for adequate neutron deflection[98]. Cratos rovers deployed on the surface prior to the arrival of the reactor system will dig a 2 m deep hole for the reactor core to sit in. The rovers will then fill the hole in with regolith covering the reactor. A chute will extend up to the surface allowing the astronauts to fill the reactor with the fuel pellets upon arrival.

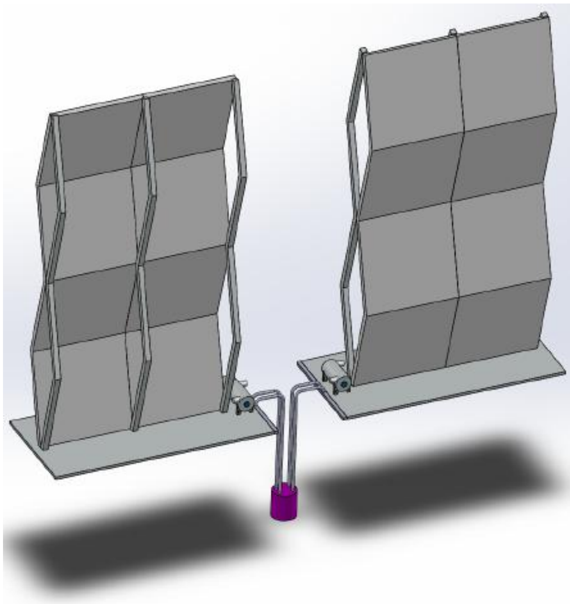

Figure 36. CAD model of the PBR power system in the deployed configuration for the lunar surface

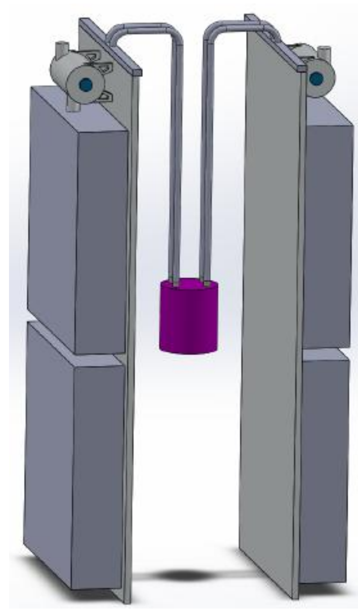

Figure 37. CAD model of the PBR power system in the stowed configuration for launch

The system is launched and placed on the lunar surface unfueled so the core will not be active while in transit. The fuel is in the form of small uranium pellets which will be transported in canisters lined with a neutron absorber to keep the fuel subcritical[98]. The crew will need to dump the fuel into the reactor core upon arrival. This setup ensures system safety during launch and emplacement.

The total mass of the system was estimated to be 1900 kg, including all working fluid and coolant duct work[98]. In the deployed configuration, the system is 10 m across and stands 7 m tall. In the stored configuration, the system is 4.5 m tall, 2.2 m wide, and 2 m deep. The reactor core is capable of generating a consistent amount of power for 66 years, but the core remains radioactive for long after this and must remain buried for 300 years before it is safe to remove[98].

The PBR system is capable of providing reliable, consistent power for the lunar base camp for the duration of the mission and far into the future. The system is invariant to illumination conditions making it well suited for locations where solar power is unavailable or inconsistent. This same architecture could be used to power lunar bases at other locations on the moon or inside lunar lava tubes, or it could power a Mars base consistently through the day/night cycle. High extensibility to other missions and low complexity make the PBR system ideal for this mission.

11. Research and Development

11.1 Overview

Several systems selected for use in the lunar base camp require further development before they can be successfully utilized. This section will detail development timelines as well as how the development costs were estimated for each system.

11.2 Estimating Development Costs

The development costs associated with each of these components was estimated using the Johnson Space Center Advanced Missions Cost Model [54]. This model uses quantity of units produced [Q], mass of the system [M], the year that the system must be launched [Y], the technological generation of the design [G], the type of system [T], and the estimated difficulty of the system [D], on a scale of -2.5 to 2.5, to estimate the development costs associated with the system [54]. The equation for cost estimation is shown as equation 10 [54].

$$Cost = 5.65 * 10^{-4} Q^{0.59} M^{0.66} 80.6^{T} (3.81 * 10^{-55})^{(1/(Y-1900))} G^{-0.36} 1.57^{D} \tag{10}$$

11.3 Landing the Upper Stage of a Falcon Heavy

The Falcon Heavy will need to see a few changes before it is able to deliver payloads directly to the Lunar surface, which was discussed in detail in Section 5. SpaceX has already developed the propulsion system that will be used for the launch vehicle, but changes will need to be made to the power systems, the problem of possible boil off will need to be fixed, and the landing legs from the Falcon 9 will need to be modified and adjusted for the upper stage. For calculating the development costs, the dry mass of the second stage and the landing legs is 5500 kg, the number of rockets needed will be 7, the year the system will be launched first is 2024, the generation was considered the number of landing attempts by SpaceX which is 55, the type of system was planetary lander, and the difficulty was determined to be -1.5 since all of these systems already exist but need to be put together. The equation gave a value of \$1.63 B for the development of this system. This money will be spent to make initial changes to the rocket as well as making new changes as the problem occur.

Table 27. The parameters used in the Advanced Missions Cost Model for modifying the Falcon Heavy payload to land on the Moon.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
7	5500	2.46	2024	65	-1.5	1626

11.4 Nuclear Power System

Space based fission power has seen increased interest from NASA in recent years, but most relevant to the system proposed in this study is NASA's Brayton Power Conversion Unit. This system was developed by NASA to test components for a surface based nuclear power plant. The system incorporates all components present in the power system proposed for the lunar base camp except for the pebble bed core itself and the heat rejection panels [103]. These two components are therefore the two main areas of development for the PBR power system.

Table 28. The parameters used in the Advanced Missions Cost Model for the nuclear power system.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
2	1848	2.13	2029	1	0	816

Table 28 shows the parameters that were used with the Advanced Missions Cost Model to estimate the cost of the power system. A quantity of 2 was used because there will be two generator loops although there will only be one core. The mass was estimated from the sum of the estimated masses of the major components of the system. The type used was the value for a habitat module because there is no type parameter for power systems in particular, it seemed like the closest fit for this system. This is a first generation system because no previous nuclear power system of this type has been used in space before. The difficulty was set to zero because although this is a large and complicated system, much of the development work has already been done by NASA. These parameters result in an estimated development cost of $816 M.

11.5 Habitat Modules

Table 29 shows the parameters that were used with the Advanced Missions Cost Model to estimate the cost of the habitat modules. A quantity of 3 was used to meet the requirements for area and volume. The mass was estimated from similar inflatable modules such as the Bigelow B330 and NASA's past TransHab experiment. This habitat is a new generation of system building upon a long history of different inflatable module prototypes and flight tested models. The difficulty was set to -1.75 due to several NASA partners, such as Bigelow Aerospace and the Sierra Nevada Corporation, already having the manufacturing infrastructure for inflatable habitats. Since the lunar habitat modules use the same technology, collaborating in their development could help save development costs and time.

Table 29. The parameters used in the Advanced Missions Cost Model for the habitat modules.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
3	8000	2.13	2028	7	-1.75	660

11.6 Lunar Crane

The crane being used on the surface of the Moon is based on the LSMS which was developed by NASA. According to NASA the system can be sized to accommodate any mission [43]. A crane for this mission would need to be taller and stronger than the prototypes that were made and tested. Based on the data NASA provided, the crane needs to have a mass of 3% the maximum tip mass it will carry. The most massive component being sent to the Moon is the habitat which has a mass of 8,000 kg. With the 20% mass growth allowance, this mass becomes 9,600 kg. The crane also needs to be 3 times taller than the prototype, but the thickness and mass also scales with this. Adding in a factor of safety of

1.2, this brings the mass of the crane to around 1,000 kg. Before deploying this crane, it should be developed, built, and tested. For the development equation, only one unit is needed, the mass is 1,000 kg, the system type is planetary, the launch date is 2024, there have been 5 generations tested, and the difficulty was a 0 since the system is only being resized. From the equation, the development cost comes out to be $609 M.

Table 30. The parameters used in the Advanced Missions Cost Model for the lunar crane.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
1	1000	2.39	2024	5	0	610

11.7 Life Support

The parameters associated with the ECLSS development are shown in Table 31. The ECLSS needs to be modified to function efficiently in lunar gravity and needs development to increase the reliability of the system. The ECLSS will be a second generation system due to the system being currently in use on the ISS. While there will be two units present at the base camp, only one unit is required for the development costs. The difficulty is zero because the system is known to work already and has already been developed a lot by NASA. The total development cost comes out to be $460 M.

Table 31. The parameters used in the Advanced Missions Cost Model for the life support system.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
1	2200	2.13	2025	2	0	460

11.8 RESOLVE Rover

Table 32. The parameters used in the Advanced Missions Cost Model for the liquid droplet radiator.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
10	60	2.14	2028	4	0	140

11.9 Cratos Rover

Table 33. The parameters used in the Advanced Missions Cost Model for the liquid droplet radiator.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
3	100	2.14	2026	3	-2.5	40

11.10 Liquid Droplet Radiator

The LDR development cost is small in comparison to other aspects of the mission but is still an important entity. LDRs have not been implemented on a mission outside of a testing capacity and therefore is a first generation system. The total mass for the system was estimated to be 14.7 kg and the launch date is in 2025. The difficulty was set to -1.5 because although the system has not been made before, the system will be relatively straight forward to manufacture using existing manufacturing technologies. The LDR parameters and resulting development cost can be found in Table 34. The development cost of the LDRs is $40 M.

Table 34. The parameters used in the Advanced Missions Cost Model for the liquid droplet radiator.

Q	M	T	Y	G	D	Cost in Millions of FY19 Dollars
6	14.7	2.13	2025	1	-1.5	40

11.11 R&D Timeline

A research and development (R&D) timeline has been developed to identify goals for when R&D of these components must be completed. It was assumed that R&D and production of a component must be completed one year in advance of the launch of the component to account for vehicle integration. Table 35 lists the completion dates for all the components which require development.

Table 35. Research and Development Completion Dates.

Lander	Power System	Habitat	Crane	Life Support	RESOLVE	Cratos	LDR
Aug. 2027	Aug. 2028	Aug. 2027	Jan. 2023	Aug. 2027	Aug. 2028	Jan. 2023	Aug. 2027

The development timeline is more aggressive for smaller, less complex systems, such as the lunar crane and the Cratos rovers requiring completion by 2023, but the deadlines for larger, more complex systems, are farther off. This allows these large components the most possible time for development.

11.12 R&D Budget

Table 36. R&D Budget Breakdown.

Component	Development Cost [Millions of FY19 Dollars]
Lander	1630
Power System	820
Habitat Modules	660
Crane	610
Life Support	460
RESOLVE	140
Cratos	40
LDR	40
Total	**4400**

12. Risk Analysis

12.1 Format and Definitions

The purpose of this section is to assess the key risks associated with Project Selene. These risks were assessed in consistency with NASA standards. The risks were grouped and identified based on the risk to asset classification: human, cost, schedule, and technical and were scored accordingly. The likelihood was determined in accordance with Figure 37 and the consequence was determined in accordance with Figure 38.

Table 37. Likelihood rating based on risk classification in accordance with NASA standards.[1, 2]

Likelihood Classification		
5-Very High	Qualitatively	Likely to occur
	Quantitatively	Human: 10<P
		Program: 50%<P
4- High	Qualitatively	Probably will occur
	Quantitatively	Human: 1<P<10
		Program: 33<P<50%
3- Medium	Qualitatively	May occur
	Quantitatively	Human: 0.1<P<1
		Program: 10<P<33%
2- Low	Qualitatively	Unlikely to occur
	Quantitatively	Human: $10^{-}3$<P<0.1
		Program: 1<P<10%
1- Very Low	Qualitatively	Occurrence Improbable
	Quantitatively	Human: P<$10^{-}3$
		Program: P<1%

Using these classifications and factors a risk chart and list were compiled.

Table 38. Consequence rating based on risk type and effect according to NASA standards [104].

Likelihood Classification		
Rank	**Asset Class**	**Description**
5-Very High	Human	Death, permanent debilitating injury, or chronic illness
	Cost	Cost > 10%
	Schedule	Major slip: launch delay, delay to next mission, exceeds reserves
	Technical	New Technology required, key performance parameters (KPP) not met
4- High	Human	Severe illness or occupational injury
	Cost	7% < Cost <= 10%
	Schedule	Moderate critical path slip or non-critical paths slip of 3 < Months <= 4
	Technical	Significant impact to KPPs, minimum mission success criteria achievable, moderate technological development required
3- Moderate	Human	Minor illness or occupational injury
	Cost	5% < Cost <= 7%
	Schedule	Non-critical path slip 2 < Months <= 3, accommodated by reserves
	Technical	Moderate impact to KPPs, minimum mission success criteria achievable within margin, requires some technological development
2- Low	Human	Need for minor first aid treatment
	Cost	2% < Cost <= 5%
	Schedule	Non-critical path slip 1 < Months <= 2, accommodated by reserves
	Technical	Minor impact to KPPs, minor impact to minimum mission requirements, requires modification to existing technologies
1- Very Low	Human	Negligible or no safety impact
	Cost	Cost <= 2%
	Schedule	No impact to schedule reserve
	Technical	No technological modification required

12.2 Risk Identification and Mitigation

Table 39. Risks such that the primary risk is to a crew member.

Risk ID	Risk Name	Risk Description	Risk Mitigation	Likelihood		Consequence	
				Initial	**Final**	**Initial**	**Final**
HR-1	Solar Particle Event (SPE)	If a solar particle event (SPE) occurs, then there is the potential for severe injury or loss of life.	A SPE is mitigated by incorporating a radiation storm shelter into the habitat architecture as well as planning a mission during times of low solar activity.	2	1	5	4
HR-2	Chariot Rover Malfunction	If there was a Chariot rover malfunction, then there is a potential for injury or loss of life.	A Chariot rover malfunction will be mitigated by bringing spare parts, deploying two Chariot rovers so that the remaining crew can perform a rescue operation, and determining routes with agreeable terrain prior to mission operations.	2	1	5	4

HR-3	Coms. Failure	Mechanical failure of a communication instrument causing loss of link, there is might be decrease in speed of data relayed.	There are three tiers of communication redundancies greatly reducing the possibility of loss of communications. Additionally each instrument had been radiation hardened tested for space environment further reducing chance of failure.	4	2	4	2
HR-4	Micrometeorite	If there is a micrometeorite impact, then there is a potential for injury or loss of life depending on the mass of the micrometeorite.	The lunar regolith structure around the habitats will serve to protect them from micrometeorite impacts.	2	1	5	4

Table 40. Risks relating to mission timeline or operations.

Mission Risk							
Risk ID	**Risk Name**	**Risk Description**	**Risk Mitigation**	**Likelihood**		**Consequence**	
				Initial	**Final**	**Initial**	**Final**
SR-1	Cratos Rover Failure	If a Cratos rover malfunctions, then there is the potential for schedule delays when constructing infrastructure to two years.	A Cratos rover malfunction was mitigated by deploying an additional rover to assist with operations and stay on schedule.	2	2	5	1
SR-2	Launch Failure	If a launch vehicle malfunctions prior to or during take off, then there is the potential for schedule delays in deployment of assets to the surface. This ranges from 3 months to 2 years.	A take off malfunction was an accepted risk since there is little to be done to mitigate it.	2	2	5	5
SR-3	Lunar Landing Failure	If an upper stage of a Falcon Heavy fails to land on the lunar surface and damages the payload, then there is the potential for schedule delays in deployment of assets to the surface. This ranges from 3 months to 2 years.	A landing malfunction is being mitigated by adding a landing pad, sensors, and landing legs to the upper stage of the Falcon Heavy that will work the RCS thrusters to ensure it lands upright.	4	2	5	5
SR-4	Habitat Deployment Failure	If an inflatable habitat fails to deploy, then there is likely to be significant schedule delays of up to 2 years.	This risk is accepted since there is little that can be done other than requiring that if one habitat doesn't deploy there will be compensation.	2	2	5	5

TR-1	Resolve Rover Failure	If a Resolve rover malfunctions, then dependency on resources brought to the surface will increase.	A Resolve rover malfunction was mitigated by deploying an additional rover to assist with the acquisition of ISRU such as WEH.	2	2	3	1
TR-2	Landing Tile Displacement	If a paver comes loose during a landing of supplies on the Lunar surface it could damage the lander or some of infrastructure around it.	By coating the connection points of the pavers with a resin, using short firings of the main engine, and only using the main engine above the surface this will reduce the likelihood and amount of pavers being dislodged.	5	2	4	2
TR-3	Reactor Failure	A reactor failure would leave many base components powerless, including the ECLSS, and the habitat modules.	A second, redundant generator loop will be included in the power system to mitigate single point failures and reduce the chance of a total failure.	2	1	5	5
TR-4	ECLSS Failure	An ECLSS failure would leave the crew without a source of oxygen or water, so the base would become uninhabitable.	A second, technically diverse, life support system will be developed and included in the base. Emergency oxygen canisters will also be supplied to provide oxygen to the crew for long enough to abort the mission.	5	1	5	5
TR-5	Thermal Failure	A habitat thermal failure would cause heat to build up inside the habitat making it unsafe for the crew and increasing the risk of an ECLSS malfunction.	A second identical thermal control system will be included in the habitat modules to reduce the chance of a thermal system failure.	2	1	5	5

13. Conclusions

13.1 Project Overview

In response to significant development in the aerospace industry, increased support for space exploration, and current missions the AIAA has released an RFP calling for designs of a lunar base camp. The requirements will be fulfilled through the use of low cost launch vehicles, existing and developing technologies, and infrastructure created though ISRU. The primary launch vehicles will be the Falcon series rockets which have demonstrated high reliability in recent years and are the lowest cost launch vehicle that can reach the lunar surface. These rockets will use LETs to reach the lunar surface and LL4 point while implementing NASA's emerging solution to boil off rates to retain fuel. The

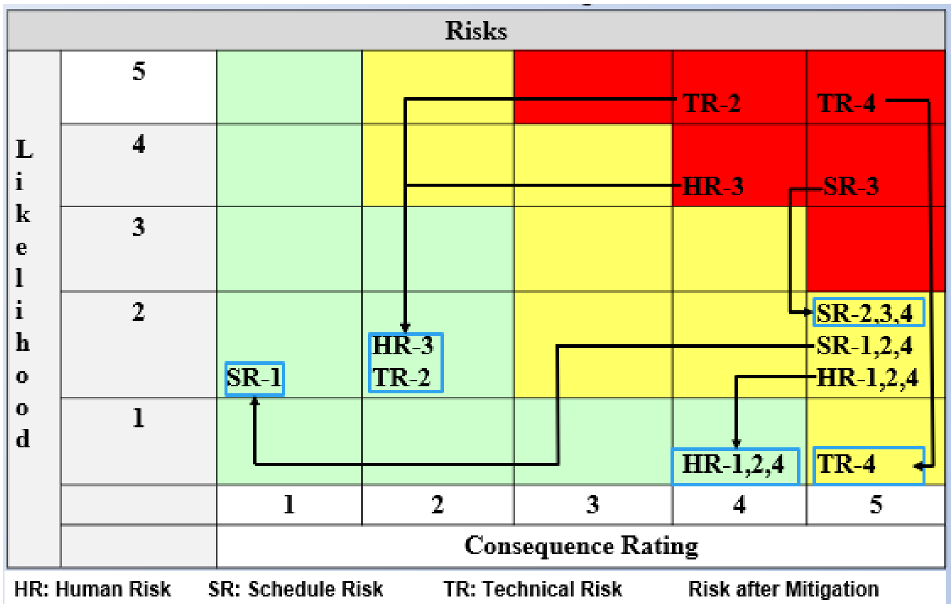

Figure 38. Risk mitigation matrix of the 10 highest scoring risks.

initial payloads will establish a three tier redundant communications system and develop a landing pad on the lunar surface. The assets that will land on the lunar surface include ATHLETE rovers, Cratos rover, a Helelani rover, crane, and Malapert Mons relay which will be delivered on a MX-9 lander. The rovers will construct a lunar landing pad 30 m in radius upon which the remaining Falcon rockets can land. This landing system will be verified and tested with the Falcon rockets used to get the hardware to the lunar surface and adjustments will be made if necessary. Furthermore, after the Falcon rockets have landed they will be removed from the launch pad through the ATHLETE rovers exceptional carrying capacity.

Once the initial infrastructure is in place and operational, three inflatable habitats will be deployed to the surface where the Cratos rovers will then construct a protective barrier from regolith and 200 kg of PTFE to increase the life span of the habitat and aid in the shielding of radiation, micrometeorites, and thermal cycles. The protective barrier will also serve as an experiment to further test the capabilities of ISRU technology. The habitats selected will fulfill NASA standards by providing 55 m^2 in floor space, and 120 m^3 in volume per habitat.

Next the LEU nuclear pellet bed reactor will be deployed to the surface and safely set up through the use of the ATHLETE rover. The reactor will be the primary source of energy for the habitat and will supplement other systems dedicated power systems.

Finally, shortly before the deployment of the crew, supplies including will be delivered to the lunar surface including 4646 kg of food and 436 kg of water accounting for extended EVA activities and including a factor of safety of 1.25 allowing for up to 56 days of habitation without rationing.

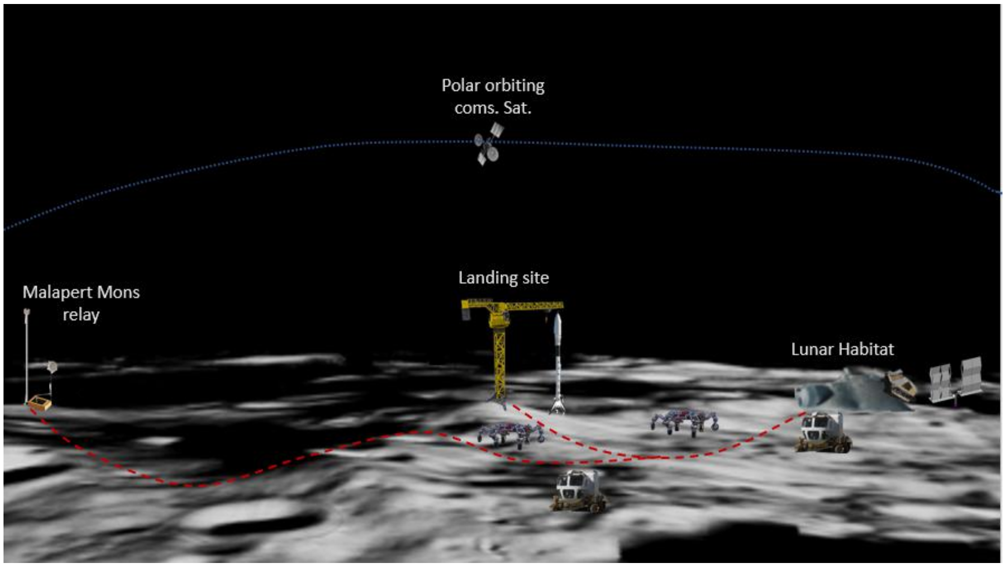

Figure 39. Lunar surface with showing common routes in red that will be used to travel from the landing pad to the habitat and from the habitat to Malapert Mons. The Chariot, ATHLETE, and Cratos rovers can be seen performing their functions. There will also be 10 resolve rovers which will be collecting WEH from craters. All of these systems will have constant communication due to the Malapert Mons and polar satellite constellation that can also be seen.

13.2 Budgets

Table 41. Total power budget showing the predicted power requirements, the growth factors, and power source.

Component	Power (W)	Growth Factor (%)	Power with Growth Factor (W)	Power Source
Life Support	1920	15	2300	PBR Generator
Chariot Rovers	150	12	180	PBR Generator
ISRU	8120	15	9400	PBR Generator
Thermal	1200	15	1400	PBR Generator
Habitat	8500	15	9800	PBR Generator
Other Rovers	500	15	580	Onboard solar panels
Polar Satellites	1482	15	1800	Onboard solar panels
Lagrange Satellite	494	15	570	Onboard solar panels
Malapert Transponder	350	15	410	Hydrogen oxygen fuel cell
Ground Station	137	15	160	Hydrogen oxygen fuel cell
BIRD Sensor	77	15	90	PBR Generator
ATHLETE	1000	15	1200	Hydrogen oxygen fuel cells and onboard solar panels
Total			28000	

Table 42. Total launch mass budget showing the component manifest, predicted component mass, mass growth per component, total launch mass, and the margin for the launch.

Launch Order	Components	Mass (kg)	Growth Factor (%)	Mass with Growth (kg)	Total Mass per Launch (kg)	Mass Margin (kg)
1	Lagrangian Sat.	1100	5	1155	1115	500
2	Helelani	460	6	488		
	Crane	1000	6	1060		
	3 Cratos Rovers	300	15	345		
	2 MX-9 Landers	4700	15	5405		
	ATHLETE Rover	2340	6	2481	9778	222
3	Polar Sats.	3300	5	4620	3465	6536
4	Primary Coms.	1124	25	1405		
	MX-9 Lander	2350	15	2703		
	ATHLETE Rover	2340	6	2481		
	PTFE	2080	8	2247	8834	1166
5-7	Habitat 1-3	24000	20	28800	9600	400
8	Life Support	2200	8	2376		
	Resolve Rovers	300	15	345		
	Power	1848	10	2033		
	Supplies	3500	4	3640	8394	1607
9	Chariot Rovers	8000	4	8320	8320	1680
10	Food & Water	5082	55	7875		
	Supplies	1500	4	1560	9435	565
	Total	**67523**	**227**	**76622**	**7662**	

Table 43. Total costs for all primary compnemnts of mission in FS 2019 USD.

Millions of Fiscal Year 2019 USD	
Allowed Budget	**12000**
2 Chariot Rovers	500.00
3 Cratos Rovers	60.00
10 Resolve Rovers	200
ECLSS	674.50
Communications System	1090.00
3 Habitats	300.00
Launch Vehicles	1403.00
ATHLETE	20.00
3 MX-9 Moon Express Landers	1200.00
Science Equipment	0.05
Helelani	0.50
Crane (LSMS)	600
Development Costs	4379.00
Operation Costs	959
Total	**11387**
Margin	**613**

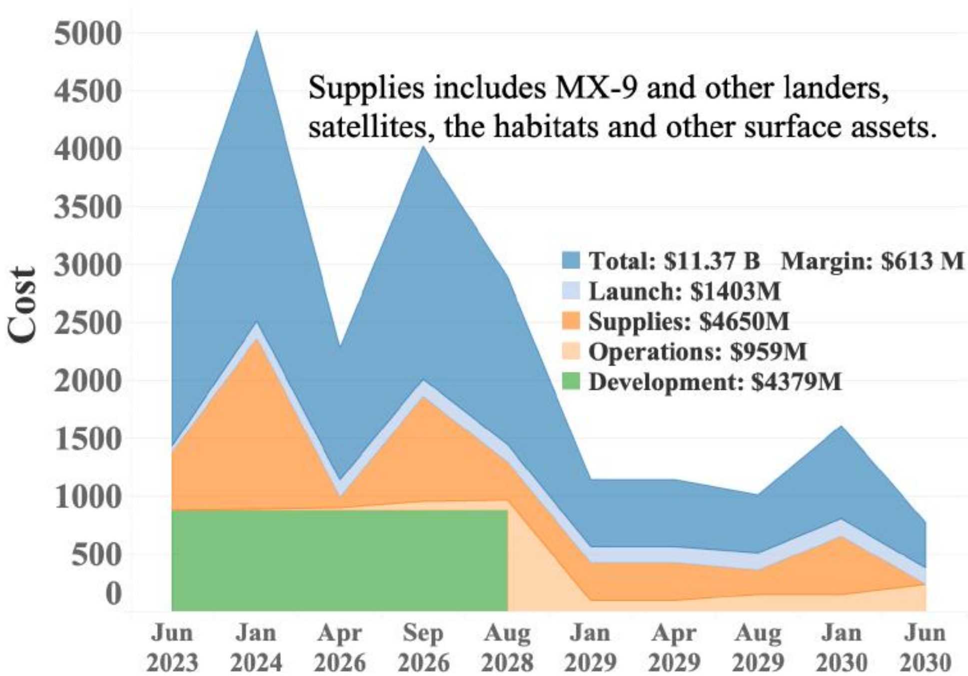

Figure 40. Cost Budget in FS 2019 USD showing Launch, Supplies, Operations and Development costs for mission duration.

13.3 Conclusion

The development of the space environment will increase access to valuable resources that are rare on Earth. The countries that are able to access these resources early on will increase their economic standing while establishing a secure footing in space environments. The past has proven that humanity is capable of the extraordinary feat of safely landing people on the Moon and that the construction and development of stations in the space environment is possible. Twenty-plus years of scientific experimentation, testing, and habitation on board the International Space Station (ISS) has proven that long duration missions to space are possible. Currently there is a drastic increase in the interest and support for space exploration. This support should be leveraged to conduct the next logical phase of manned space exploration, a sustained and long duration habitability of the lunar surface. This will be used as a proving ground for further technological and operational development that will grant access to new resources, yield further insights into the development of our solar system, and pave the way for further space exploration to Mars and beyond. NASA's 2020 FYB is $22.6 B USD, assuming that this funding remains stable and objectives remain fairly consistent, the cost of Project Selene will make up a mere 5.3% of NASA's budget while managing to drastically advance humanity's presence in space. The time to take our next big leap and secure our footing in space is now.

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