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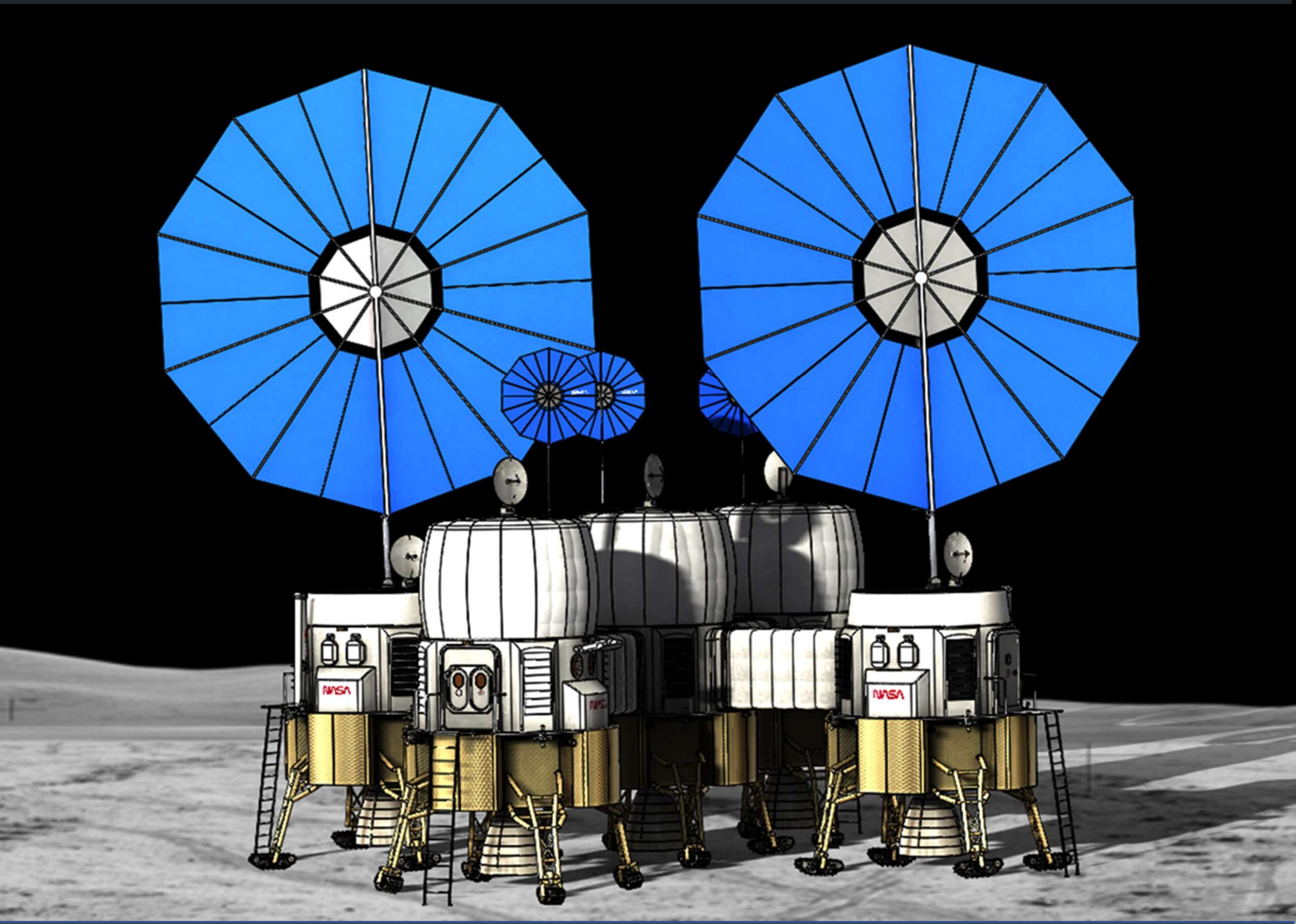

RESPONSE TO 2020 AIAA LUNAR BASE CAMP DESIGN COMPETITION

Lunar Base Outfitted With Interchangeability and Expandability

Lunar BOWIE

May 2020

California State Polytechnic University,
Pomona Aerospace Engineering Department

Space Oddities

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SHAPING THE FUTURE OF AEROSPACE

Aerbwong Chitamitara
Project Lead
AIAA: 935170

King Saba
AIAA: 965135

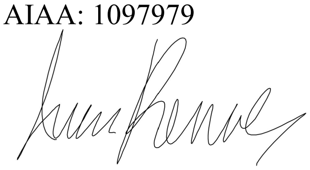
Brandon Delannoy
AIAA: 1083625

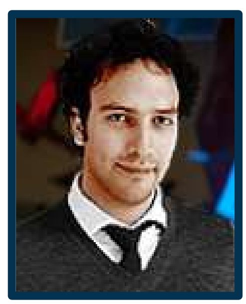
Bryce Satterfield
AIAA: 984131

Christopher Ligrano
AIAA: 981285

Joshua Barnum
AIAA: 1084989

Allan Ramos
AIAA: 1097979





Dr. Navid Nakhjiri
Project Advisor

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List of Acronyms

ACS	Attitude Control System	**MCR**	Mission Concept Review
ACE	Attitude Control Electronics	**MIB**	Minimum Impulse Bit
ACR	Alternative Concept Review	**MUSTANG**	Modular Unified Space Technology Avionics for Next Generation
BOL	Beginning of Life	**MCR**	Mission Concept Review
BOWIE	Base Outfitted With Interchangeability and Expandability	**NAC**	Narrow Angle Camera
CDR	Critical Design Review	**NDL**	Navigation Doppler Lidar
CDRA	Carbon Dioxide Removal Assembly	**ORR**	Operational Readiness Review
CR-1	Connecting Ridge Location 1	**PDR**	Preliminary Design Review
CTBE	Cargo Transfer Bag Equivalent	**PFU**	Particle flux unit ($Particle/cm^2\ sec\ sr$)
DDT&E	Design Development Test and Evaluation	**Pmp**	Maximum power point
DLC	Descent and Landing Computer	**PSE**	Power System Electronics
DR	Decommissioning Review	**SC**	Spacecraft
DSN	Deep Space Network	**SDR**	System Design Review
ECLSS	Environmental Control and Life Support System	**SEPIA**	Separation Interface Assembly
EOL	End of Life	**SKG**	Strategic Knowledge Gaps
ESA	European Space Agency	**SPLICE**	Safe and Precise Landing - Integrated Capabilities Evolution
FOV	Field Of View	**SRM**	Solid Rocket Motor
FRR	Flight Readiness Review	**SRR**	System Requirements Review
FY	Fiscal Year	**TDS**	Terminal Descent System
IAU	Integrated Avionics Unit	**TFU**	Theoretical First Unit
IMU	Inertial Measurement Unit	**TLI**	Trans lunar Injection
I/O	Input/output	**ToF**	Time of Flight
ISPR	International Standard Payload Rack	**TRN**	Terrain Relative Navigation
ISS	International Space Station	**UPA**	Urine Processing Assembly
LOX	Liquid Oxygen	**VIBES**	Vitality Base Element System
LH2	Liquid Hydrogen	**WPA**	Water Processing Assembly
LV	Launch Vehicle	**TT&C**	Telemetry Tracking and Command
LBC	lunar Basecamp	**REID**	Risk of Exposure-Induced Death
LOLA	lunar Orbiter Laser Altimeter	**xEMU**	Exploration Extravehicular Mobility Unit
LRO	lunar Reconnaissance Orbiter	**VHF**	Very High Frequency

1.0 Executive Summary

Space Oddities is seeking to join the global effort in expanding human borders and boundaries beyond Earth by utilizing thoughtful innovation, engineering creativity and competitive partnerships. The first step in Space Oddities' pursuit of expanding human reaches beyond Earth is to assist the National Aeronautics and Space Administration (NASA) in establishing sustainable human presence on the lunar surface. Space Oddities is proposing the Lunar **B**ase **O**utfitted **W**ith **I**nterchangeability and **E**xpandability (Lunar **BOWIE**) in response to the 2020 American Institute of Aeronautics and Astronautics (AIAA) Lunar Base Camp (LBC) Design Competition. Space Oddities' goal is not only to put humans back on the Moon but also lay the foundations for a permanent lunar base and support future deep space science and exploration missions to Mars and other targets. The Lunar BOWIE will serve as a first phase lunar base camp that will ensure the survivability and habitability for a crew of 4 for nominal mission period of 45 days, with considerations of both their physical and mental well-being. The Lunar BOWIE will also be capable of supporting scientific missions that will initially focus on in-situ resource utilization and support extra vehicular activities for lunar surface exploration. Moreover, Space Oddities has designed the Lunar BOWIE with modularity in mind to allow for future lunar base camp expansion and resupply for extensibility of crewed missions. Space Oddities is looking to send and deploy the Lunar BOWIE by the 31st of December 2030 and within the budget of $12B US Dollars (Fiscal Year 2019). To meet scheduling and cost goals, Space Oddities will maximize the use of current technology or technology that is in NASA's development portfolio and any other technology with Technical Readiness Level high enough to be developed and tested for flight readiness within the Lunar BOWIE First Article Inspection (FAI) phase. Putting humans back on the Moon still presents many challenges but sustaining human life on the lunar surface adds a new set of difficulties that Space Oddities is ready to meet.

The most important challenge concerns crew health and safety, particularly protecting them from the harsh conditions of the lunar environment, sustaining crew life for longer durations on the lunar surface, and maintaining their psychological health. The Moon is relatively close to Earth compared to other celestial bodies, but the overall distance for human space flight is quite long. Sending humans to the Moon means taking them outside the natural protections of Earth from harmful radiation, impacts, extreme thermal environments and lunar dust. The crew on the International Space Station (ISS) still operate within Earth's magnetic field that protects them from harmful solar radiation. On the lunar surface, the crew will be exposed to the full effects of solar winds with solar particle events and cosmic rays that are an ever-present threat as well. Although a crewed lunar mission does not experience a debris field like those on the ISS, the chances and sizes of impacts to a lunar base structure still have to be considered. The thermal environment on the lunar surface also goes through extreme temperature differences that the crew will need to be protected from. In addition, the lunar dust is pervasive which can affect crew health if steps are not taken to prevent dust from entering the living space.

The distance to the Moon also presents logistical support challenges for crew survivability and habitability. A lunar base camp must be able to sustain livable conditions for the 4 crew members during the 45-day mission period without relying on the chances of resupply of consumables or replacement of life support components. The crew must not only have enough air, food and water to survive, but reliable waste disposal systems and basic amenities in order to properly live on the base. Staying in constant communication with the crew, where there is currently no robust communication infrastructure, also poses a challenge to ensuring crew survivability. Although there is significant gravity on the Moon, it is only a sixth of that on Earth and may still lead to musculoskeletal atrophy of the crew. Hopping around on the Moon itself may provide enough resistance, but more exercise will be needed to keep the crew healthy for the duration of the mission as the crew may not perform EVAs every day.

Exercise can also address the challenges of maintaining the psychological health of the crew. Individual crew members also require large enough personal space for themselves with enough accommodations for livable conditions. Delivering a base to the lunar surface with enough volume and amenities for the crew will require careful design considerations. The crew will also need other things such as enough lighting and even enough windows. Maintaining social connections for the crew over the distance of Moon also poses difficult challenges. A space for promoting social interactions between crew members and providing video capabilities for crew members to connect with their loved ones on Earth must also be considered in the design.

Many of the challenges that the lunar environment presents to sustaining crew health and safety also affects the condition of the lunar base camp as well. In addition, there is already existing challenges of the space environment during transit. Radiation can cause severe damage to avionic components if not protected properly. Any failures cannot rely on resupply to be fixed or replaced and the thermal environment can also cause component failures without proper temperature control. The lunar base camp must also be able to withstand the loads experienced during launch, maneuvers during transit and landing on the lunar surface. Moreover, lunar dust plumes during landing can hamper safe landing of the spacecraft which is just among other challenges lunar dust presents to the mission.

Addressing many of the challenges creates the challenge of being able to deliver everything the base needs to the target location. In addition, the base must be able to support the crew in their science and exploration missions. A proper analysis and selection of the target location must be conducted in order to maximize the sustainability of the crew's presence on the lunar surface, but also maximize scientific and exploration opportunities that the target location may provide. Thus, mass limits and scientific interests of the target location requires careful selection of scientific payloads to support the crew in their mission. NASA's scientific goals and strategic knowledge gaps must also be considered when choosing both target location and science payloads.

In order to ensure all the design constraints are being met for the lunar base camp, Space Oddities has derived requirements based on both the challenges previously discussed and governing system standards that also addresses the same challenges. Furthermore, derived requirements should be traceable to the RFP requirements which Space Oddities have adopted as the system-level requirements for the lunar base camp design. Space Oddities' approach to meeting requirements developed 2 architectures, where one represented a more conventional design, dubbed the BalLunar, and the other a more "radical" design, dubbed the LunaBago. Both architectures were developed up to conceptual design where the down selected architecture by trade study would move on through preliminary design. Trade studies were also used to down select subsystems and subsystem components for the Lunar BOWIE. Trade studies were performed using figure of merits that best met the derived requirements for each subsystem and the highest weighted score for the subsystems and subsystem components are selected for integration into the Lunar BOWIE design.

The Lunar BOWIE, in response to the AIAA LBC Design Competition, is a modular multi-launch design with 4 variants and 5 total modules to be delivered. The 4 variants consist of an inflatable habitat module made for 2 occupants, the Environmental Control and Life Support System (ECLSS) module, the cargo module, and an inflatable central module known as the **VI**tality **B**ase **E**lement **S**ystem (VIBES) module. The ECLSS subsystem, modeled after the ISS ECLSS, will provide open loop air, closed loop water, and waste management systems to support crew life with enough spare components to ensure a 0.1% failure rate over 450 days. Accounting for margins in the 45-day mission period, the base camp will be able to supply the crew with 60 days of air and food. The VIBES module and 2 habitat modules combined, will provide all necessary living accommodations for a crew 4 for the mission. The VIBES module serves as the central location for crew members to gather and socialize as the galley and fitness equipment are located in the module. The combined 5 modules will provide more than the required 17.6 cubic meters of habitable volume per crew member with lighting and windows for livable conditions.

The 5 modules to be delivered will consist of 2 habitats and one of each other variants using 5 total SpaceX expendable Falcon Heavy launches. The first module will be launched in June 2029 with 4 subsequent launches to follow every 2 months. The target location for delivery will be the Connecting Ridge near Shackleton Crater via direct lunar transfer. The Falcon Heavy will perform the Trans-Lunar Injection (TLI), followed by solid rocket motors to perform the Lunar Orbit Insertion (LOI), after which will be jettisoned, and then a liquid bi-propellant rocket engine will take over for descent landing. A combination of solid rocket motors and a liquid bi-propellant rocket is used to meet the fuel mass and volume requirements for the propulsion system. Four clusters of four attitude control system thrusters will be used to perform all the maneuvers and any correction needed to successfully deliver modules in transit. A barbecue roll will be used as a means to protect components from the thermal environment. NASA's Safe and Precise Landing-Integrated Capabilities Evolution (SPLICE) will be used as the main descent landing sensor, which is expected to be TRL 6+ by 2022, and the Terminal Descent Sensor (TDS), used for the Mars Science Laboratory, will be used for safer landing in the dust plumes.

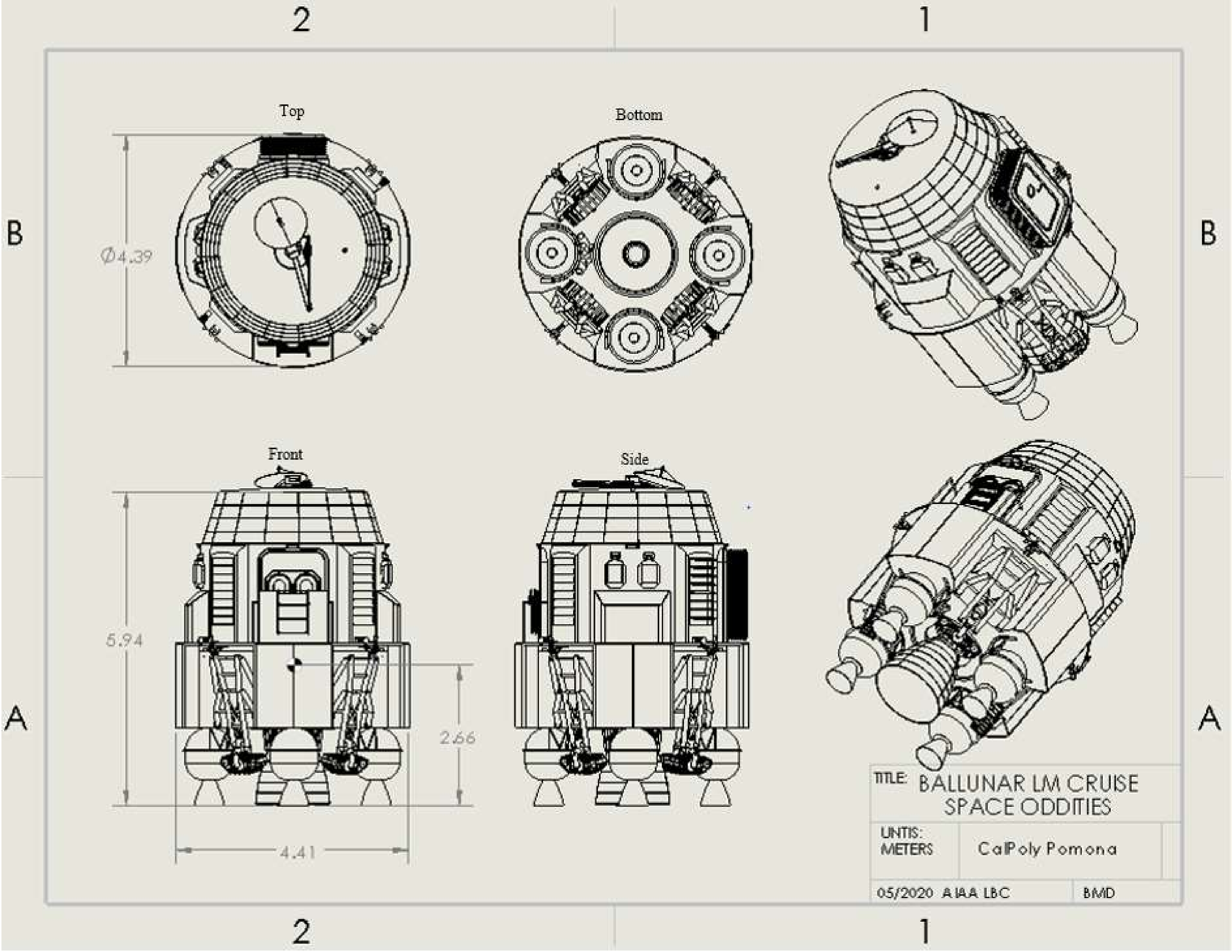
The 5 Lunar BOWIE modules will be capable of performing autonomous rendezvous and docking to minimize any crew operations for base camp deployment and ensure the safest transition from their lander to a fully deployed base. SPLICE will allow the precision landing of the Lunar BOWIE modules to within 50 meters of each other and treads on self-adjustable hydraulic landing legs, along with range and level sensors on the modules will allow leveled docking over the lunar terrain. The first three modules will also be equipped with deployable 2-meter solar arrays to provide temporary power for the autonomous docking. After all modules are docked, the main solar arrays will deploy, the base will pressurize, and the inflatable modules will inflate.

The elevation of the target location will have 88.1% to 91.4% of illumination with the longest continuous period of darkness of 3-5 days. Passive thermal controls using white thermal coatings, Multilayer insulation (MLI) blanket, and polyurethane foam, along with active thermal controlled radiators and louvers will be used to maintain the thermal environment for habitability and protect the interior components. The walls of the module will also include layers of 7075-aluminum, Ultra-High Molecular Weight Polyethylene (UHMWPE), and a 7075-aluminum honeycomb interior to protect the crew from harmful radiation and impact from small debris while maintaining interior pressures. The illuminated elevation of the target location will also provide enough sunlight for almost 40kW of base power through the two 10-meter solar arrays and batteries will be used for back-up power.

The telecommunications system will consist of a lunar relay constellation and multiple ground stations to maintain constant communication with the crew. The lunar relay constellation will consist of the Lunar Gateway and another telecom satellite which will be launched prior to the launches of the modules. The acquisition, launch, and operations of the second telecom satellite are included in the budget. The Lunar BOWIE is also capable of lunar surface communications to support EVA and surface exploration missions. The telecom systems will also be capable of allocating a minimum of 400Mb/s to allow video communications with Earth. The primary flight computer used for command and data handling will be the Modular Unified Space Technology Avionics for Next Generation (MUSTANG). The MUSTANG is equipped with a processor board that is triple modular redundancy fault tolerant and Class S rad-tolerant capable.

Some modules will also be equipped with International Standard Payload Racks (ISPR) for both storage and science support capabilities. Science payloads will initially accommodate in-situ resource utilization experiments, plant growing chambers, and 3-D printers. ISPR's can be used to adapt to other variety of science payloads as necessary and ensure the continuation of supporting science missions under the expansion of the base.

Taking on such a large and ambitious project does not come without its programmatic challenges which most importantly includes schedules and cost overrun. It is important to properly manage program schedules and balance budgets to prevent delays and unnecessary expenditures. Ensuring a steady manufacturing supply chain and building margins onto the cost can further prevent schedule delays and overbudgeting. Understanding of technical and programmatic risks along with proper mitigation is also important to running a successful program. The proposal also includes mission operation plans that consists of the program schedule and cost breakdown, mission lifetime estimates and manufacturing and supply chain of the Lunar BOWIE, and detail for the End of Mission and Disposal with consideration of planetary protection concerns. In addition, a risk analysis of the program and the design are also included.



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SPACE ODDITIES
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2.0 Mission Overview

This section provides a summary of the customer's needs, the mission objectives to be accomplished, and system design requirements identified for our proposed lunar base camp. We wanted to distinguish between long-term goals and shorter-term goals to better understand the challenges to be solved. Design constraints were also written as requirement statements to ensure that our design complies with the customer's requests

2.1 Needs Analysis

There is a need to support future deep space science and exploration missions, most notably future missions to establish human presence on Mars. In order to support any future mission successes, NASA and international and commercial partners are looking to establish permanent assets near or on the lunar surface. The establishment of the lunar assets will begin with the NASA Artemis Program which includes the Lunar Gateway, an orbital lunar station. Part of meeting the need to support future deep space science and exploration missions is also establishing assets on the lunar surface to test how astronauts will live, perform, and operate on the Moon in preparation for the future missions. A lunar base camp will be the first step toward establishing a permanent lunar base on the Moon that can support crew life and support them in their missions.

2.2 Mission Objectives

The mission objective of the lunar base camp will be to ensure the survivability and habitability of a crew of four for a nominal mission period of forty-five days and support the crew in their science and exploration missions. As the first phase of establishing a permanent lunar base, the base camp will also be capable of expanding to accommodate potential growth of crew size and extending the crew's ability to safely live and work on the lunar surface. Moreover, the lunar base camp will also be delivered by the 31st of December 2030 and within the budget of $12B USD (FY19) To meet the schedule and cost objective of the mission, the lunar base camp will maximize the use of proven technology and technology rated at TRL 6+ by development.

2.3 RFP/System Level Requirements

System-level requirements have been identified and listed verbatim from the RFP to better understand design requirements, derive and decompose lower requirements, establish traceability, and ensure design compliance. The system level requirements, in *Table 2.3-1*, are categorized as Technical, Management, and Cost requirements denoted by T, M, and C, respectively. Although, requirements have been sectioned off and rewritten with *shall* words, the rest of the statements are taken verbatim from the RFP. The RFP reference numbers, in the first column, refers to the RFP section that was renumbered during bursting for the team's own references. In addition, the requirements are ordered according to the order that the constraint is found in the RFP for quick reference.

Table 2.3-1 System Level Requirements

RFP Ref #	Type and Req #	Requirement Statement
1.1	T0.0-1	Lunar Base Camp (LBC) shall be designed to support a crew of 4 for a nominal mission duration of 45 days on the lunar surface.
1.2	T0.0-2	Location of LBC shall be chosen to maximize crew survivability, scientific research, and support for future deep space missions
1.3	T0.0-3	The LBC shall have the potential for the base camp to be expanded to accommodate more crew for longer duration in subsequent expeditions
1.4	T0.0-4	The design shall consider the various activities, resources, and systems that future exploration missions to other solar system destinations would require and how the base camp would help enable those missions
2.1	T0.0-5	The design shall include all the necessary systems to launch and deploy the base camp elements to the lunar surface
2.2	T0.0-6	LBC shall detail the necessary tasks that are required to bring the base camp to operation to sustain the crew for the duration of the expedition
2.3	T0.0-7	The crew lander shall only sustain the crew of 4 for a period no longer than 72 hours after landing; any payload capability that the crew lander has will be fully utilized to support the crew for the 72-hour period and for crew ascent. The crew lander and ascent stage is not part of the base camp design.
3.0	M0.0-1	Trade studies shall be performed on system options at the system and subsystem level to demonstrate the fitness of the chosen base camp design.
3.0	M0.0-2	Design shall maximize the use of technologies that are already demonstrated on previous programs or currently in the NASA technology development portfolio.
3.0	M0.0-3	Advanced technology use shall consider cost, schedule, and risk associated with the development.
4.0	M0.0-4	Design shall detail subsystem components, including mass, power, and volume, and how the design requirements drove the selection of the subsystem
5.0	M0.0-5	Design shall detail the estimated lifetime of each of the components, the lifetime of the system and number of surface expeditions the basecamp can sustain, and the potential upgrades/expansions that are available with the design and how extensibility and longevity considerations impacted the design choices
6.0	M0.0-5	Design shall detail how the base camp components will be packaged, launched, and deployed to the lunar surface, whether any on-orbit or on-surface assembly or rendezvous of components will be required, and what systems would be required to assist in the delivery of the components to the lunar surface
7.0	C0.0-1	The initial cost for the lunar base camp shall not exceed $12 Billion US Dollar (FY19) from the start of the program to the human expedition, including design development test and evaluation (DDT&E) and theoretical first unit (TFU) costs of all of the base camp elements.
8.0	C0.0-2	Technology advancement cost shall be included if advanced technology options are utilized in the design.
9.0	C0.0-3	The cost cap includes launch costs to deploy the base camp systems, but does not include the cost of the human expedition mission and its associated lander/ascent stage
10.0	M0.0-7	The base camp shall be ready to receive the first expedition crew no later than December 31, 2030.

3.0 Mission Design

In this section, we will go over the design methodology of our design process from the beginnings in System Requirements Review (SRR) up to Preliminary Design Review (PDR). In addition, this section covers the primary two architectures developed for this design process, and the down selection process used to narrow the scope of the project down to a single architecture. Once down selected, an overview of the concept of operations, followed by different mission phases with different mission configurations are presented. Finally, an overview of the landing location, trajectory, and launch vehicle are presented.

3.1 Design Methodology and Process

For our design team, we were tasked with carrying two different architectures up until SDR. The designs were following the ideas of having one design follow conventional design philosophy and using historical models to iterate upon for the design. The other architecture was more of a "radical" design, allowing us to use our creativity to solve the problems presented in the RFP in a unique manner.

The total life cycle of our design process started in December with SRR. This review gave us feedback on our two designs and allowed us to refine the architectures. We then engaged in an active design process with our faculty advisors, requiring us to present our corrections and additions roughly every two weeks. This process culminated in our System Design Review (SDR), where we presented our designs to both Lockheed Martin and Northrop Grumman, receiving industry feedback on our designs. Using this feedback, our architectural trade study was developed, and we down selected to our selected architecture. After this point, we continued to advance our design until we presented our single architecture design to JPL in our PDR.

The process of developing the architectures was primarily conducted by using trade studies. These trade studies were conducted by researching multiple solutions to each problem presented, constructing a trade study to compare each option to another, and down select to the most optimum option. This method was used throughout the development of both architectures presented below, in addition to refining subsystems past the point of architecture down selection.

3.2 Summary of Architecture #1: BalLunar

The first architecture, named BalLunar, is the conventionally designed architecture. The primary namesake to the architecture is the inclusion of an inflatable upper story for half of the variants. The architecture is composed of 5 total modules, with 4 unique variants among the modules. *Figure 3.2-1* shows the completed setup of the BalLunar modules on the lunar surface.

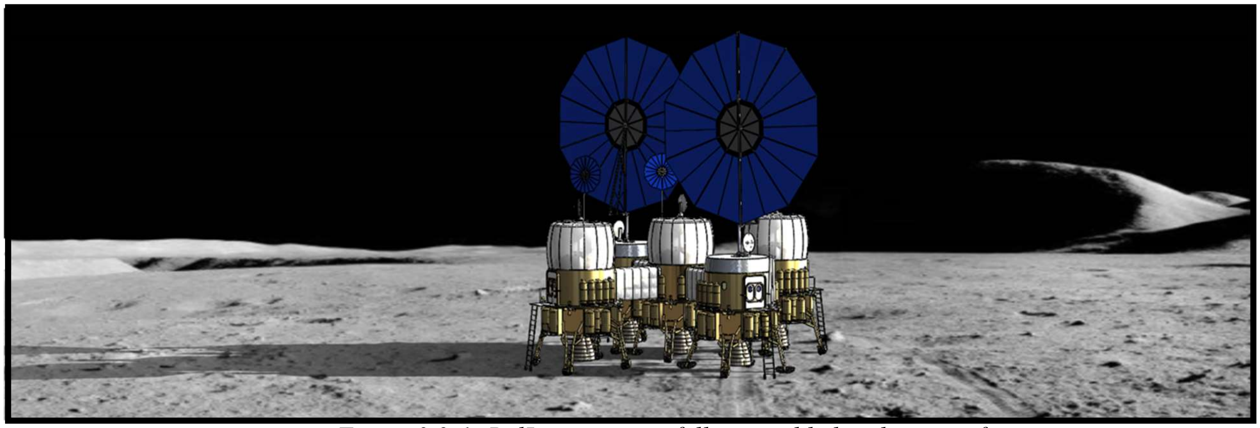


Figure 3.2-1: BalLunar system fully assembled on lunar surface

The 4 variants of the BalLunar architecture are as follows: Living Modules, ECLSS Module, Cargo Module, and the VIBES Module. The details of each variant's subsystems, in addition to detailed views of each variant, are explored later in this paper, but a general summary of the variants is included here.

Variant #1, Living Module (LM)

The Living Modules (LM) are the 2 modules that the crew will reside in during the nominal mission duration. Each LM is equipped with 4 International Standard Payload Racks (ISPR) for use of storage of personal items and crew accommodations. It is also equipped with 2 beds, one for each crew member, which are located in the upper inflatable section of the module. The inflatable section is split between the two crew members to allow for personal space and is separated by curtains.

Variant #2, ECLSS Module

Due to the large mass of the Environment Control and Life Support System (ECLSS), the ECLSS module houses the majority of that subsystem and provides the name of the variant. A major difference from the LM is the change of an inflatable upper section, which is replaced by a solid structure used to support the large solar arrays of the primary power system. The solar arrays and the ECLSS systems are detailed in their respective breakdowns further down in this report.

Variant #3, Cargo Module

The Cargo Module is a modified ECLSS module and contains extraneous maintenance equipment in addition to a large portion of the ECLSS backups. Like the ECLSS module, it utilizes a solid upper structure to support the large solar array.

Variant #4, VIBES Module

The VItality Base Element System (VIBES) module is the central module for the BalLunar architecture. It functions as the social gathering and gym for the LBC, as it contains the crew and galley system and gym equipment.

3.3 Summary of Architecture #2: LunaBago

The second architecture, named LunaBago, is the "radical" architecture designed. The primary design difference is the maneuverability of the LunaBago, compared to the permanent structure of the BalLunar. In addition to this difference, the LunaBago architecture requires 10 modules, although with a similar number of unique variants at 4. *Figure 3.3-1* shows the fully linked setup of the LunaBago modules, which is used to resupply each module and allow for the crew to socialize. All modules of the LunaBago have an autonomous driving capability, but also can be manually piloted by crew members.

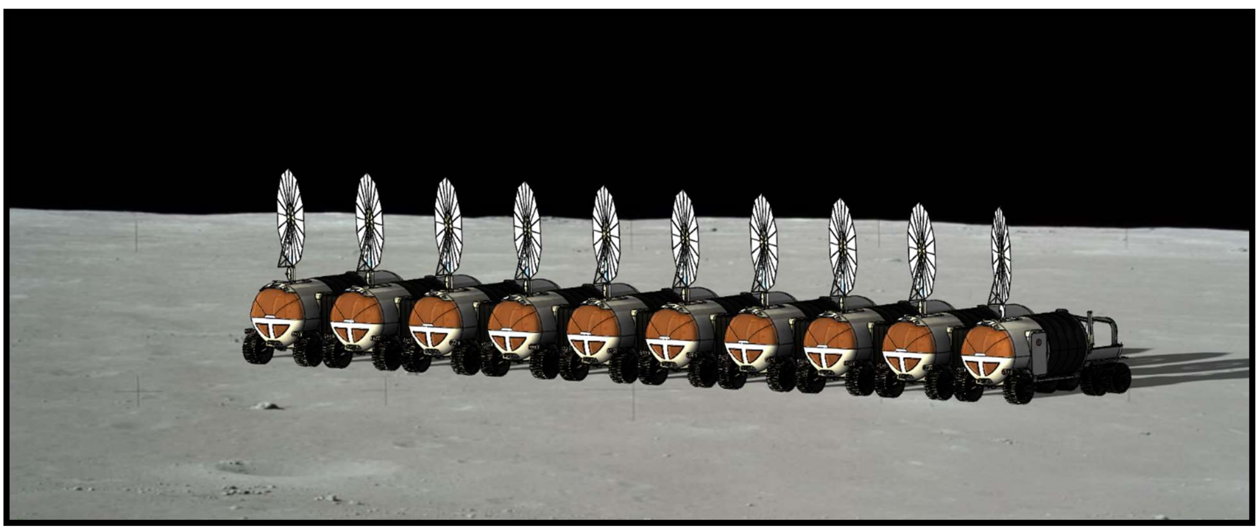


Figure 3.3-1: All 10 LunaBago modules assembled to form singular LBC

Another key difference in the architectures is the landing method. The BalLunar can use its legs as its landing mechanism, but the LunaBago has no such legs. Thus, a different landing mechanism was required to be developed to allow for a safe landing of each module. The LunaBago Lander was developed to accomplish this goal and is shown in the following *Figure 3.3-2*.

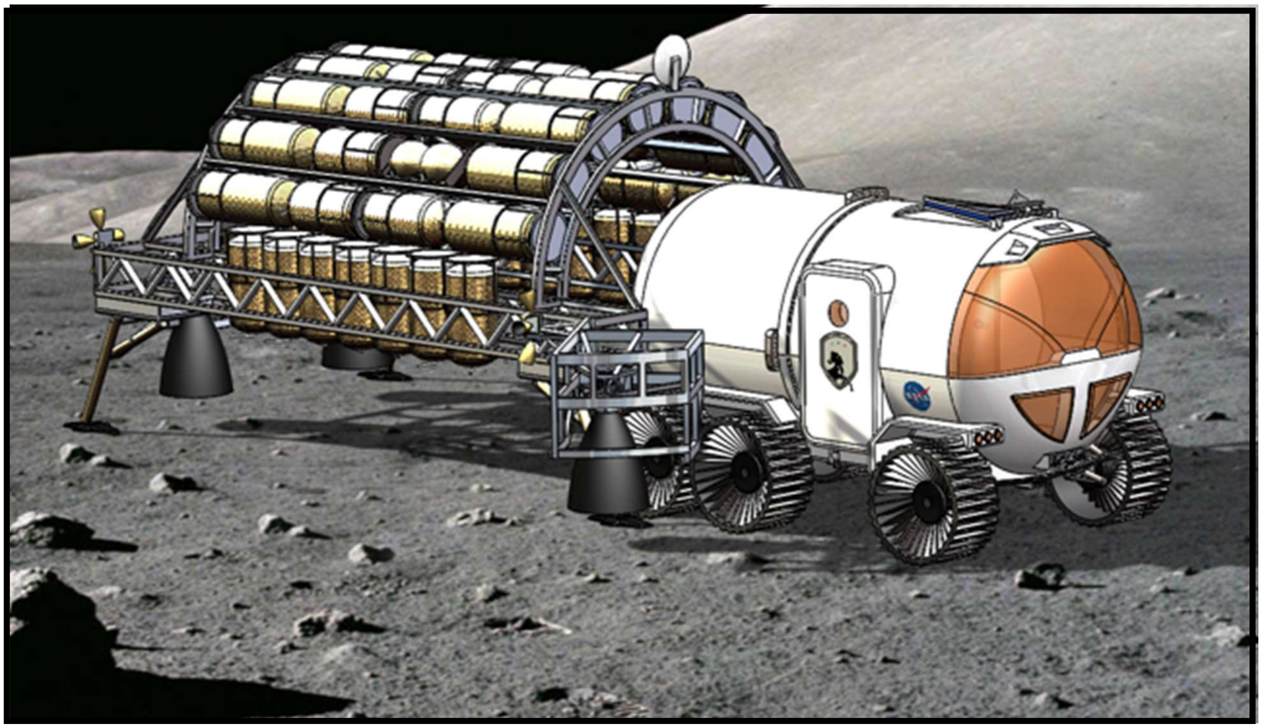


Figure 3.3-2: LunaBago Lander successfully landed on lunar surface.

The variants of the LunaBago are the following: Living Modules, ECLSS Modules, Crew Accommodations Modules, and Utility Module.

Variant #1, Living Modules

The LM's of the LunaBago are used to house a single crew member versus the 2 crew members of the BalLunar LM's. This means that the LunaBago LM's can be independently maneuvered by single crew members to accomplish unique scientific missions. The module, once fully supplied from the ECLSS modules, can operate for up to 72 hours before needing another resupply. There is a total of 4 LM's for the LBC.

Variant #2, ECLSS Modules

The ECLSS Modules of the LunaBago serve the same purpose as the ECLSS module of the BalLunar: to house the primary ECLSS system of the entire LBC. Due to the mass of the ECLSS system, the total system was distributed among 3 separate modules These modules stay connected during total LBC operations, but can autonomously follow single LM's to extend their individual mission operation capabilities.

Variant #3, Crew Accommodations Modules (CAM)

The CAMs are used during unified LBC operations to function as the crew social setting and a crew dining area. There is a total of 2 CAMs: one module function as the kitchen and dining area, with the other module storing the gym and social gathering area.

Variant #4, Utility Module

The Utility Module sacrifices pressurized volume to accommodate an exterior crane and winch for use in servicing other LunaBago modules or other scientific objectives. There is only a single Utility Module for the LBC, and as such is generally kept with the other modules, and only driven if needed.

3.4 Down selection of Architectures: Trade Study

To down select to a single architecture, multiple Figures of Merit were compared between the two architectures. The major FOM's are discussed below.

Total Spacecraft Wet Mass: Ensuring the LBC modules were meeting the mass budget is a requirement, as the launch vehicle can only launch a certain mass. Any additional margin is also recommended for scientific instrumentation and equipment, which is discussed later.

Mission Cost: Given the total mission budget of $12 billion USD2019, ensuring the architecture was capable of completing the mission within the budget is required.

Number of Launches: With both architectures utilizing multiple launches, a major Figure of Merit was minimizing the total number of launches. This was done to reduce the chance of a launch failure, causing a module to fail to land on the lunar surface.

The final trade matrix is shown below in *Figure 3.4-1*, and *Table 3.4-1* shows the totaled unweighted and weighted scores of both architectures. Due to the BalLunar architecture presenting less launch risk and a lower mission cost, the BalLunar architecture was chosen for our design.

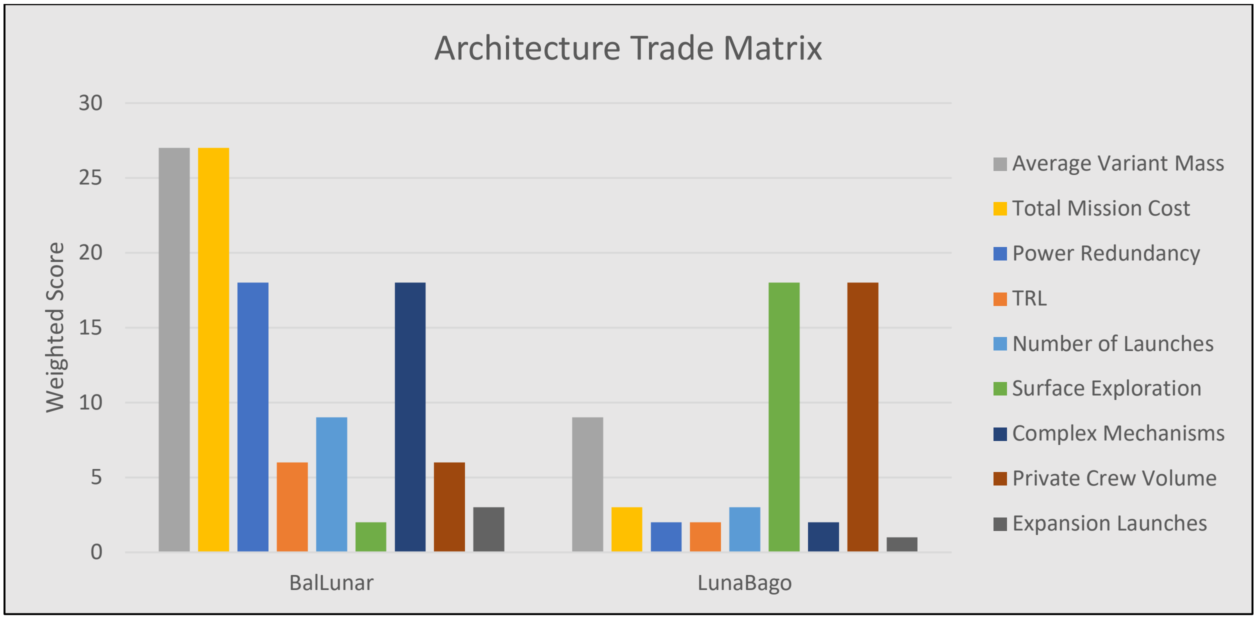


Figure 3.4-1: Architecture Down Selection Trade Study

Table 3.4-1: Weighted and unweighted scores for architecture down selection

	BalLunar		LunaBago	
	Unweighted	**Weighted**	**Unweighted**	**Weighted**
Total Score	**49**	**116**	**27**	**58**

The selected BalLunar architecture as the Lunar BOWIE is discussed in further detail under subsequent sections below. The Concept of Operations for the BalLunar architecture is provided under section **3.5 Lunar BOWIE Concept of Operations** in *Figure 3.5-1* and *Figure 3.5-2*.

3.5 Lunar BOWIE Concept of Operations

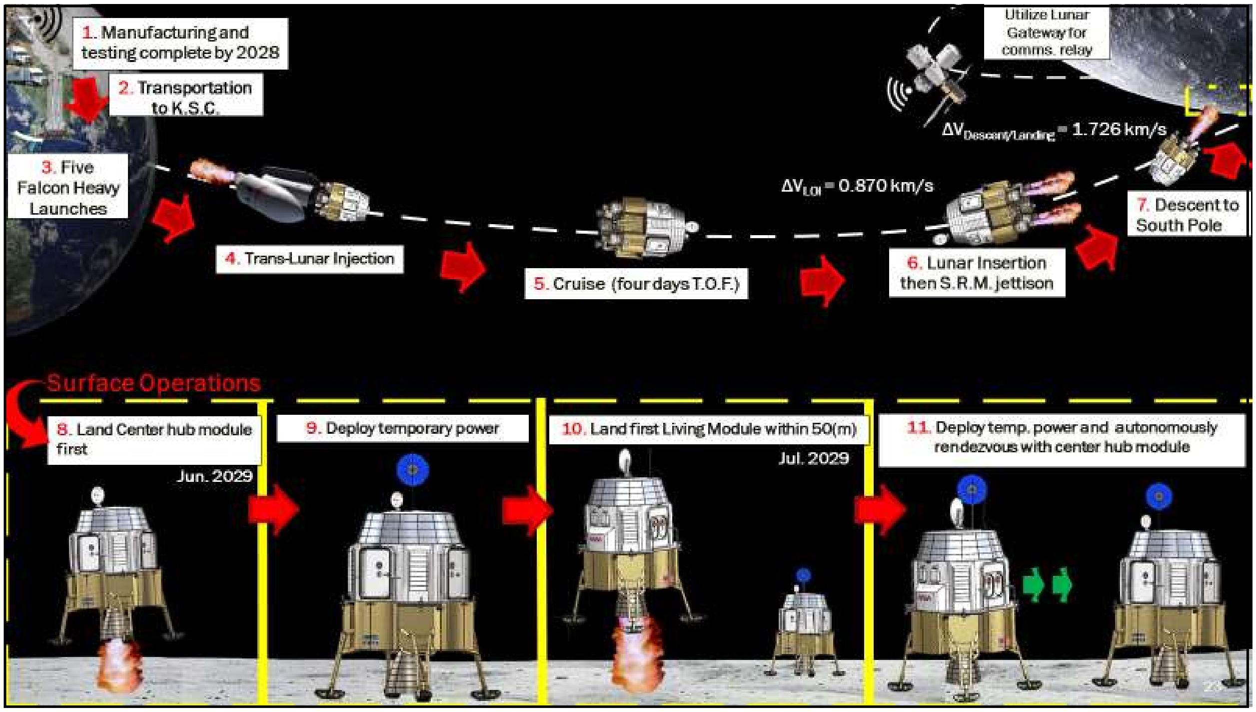


Figure 3.5-1: Conceptual Operations of Lunar BOWIE part 1

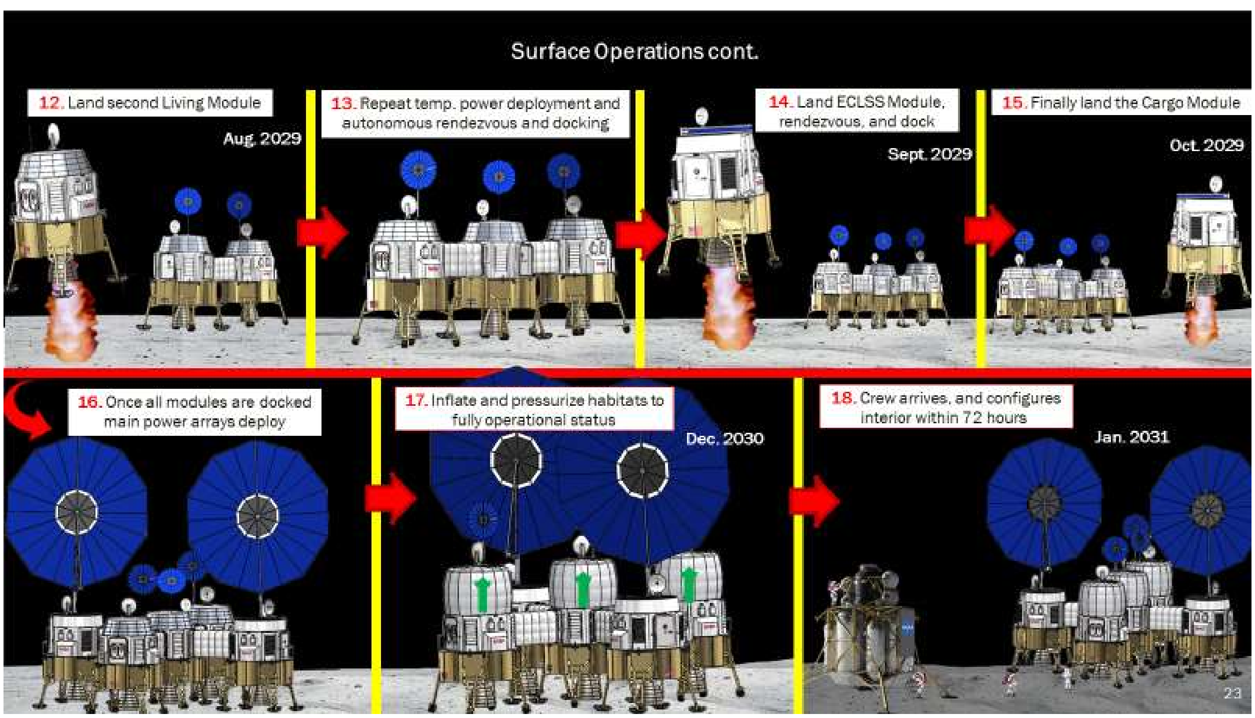


Figure 3.5-2: Conceptual Operation of Lunar BOWIE part 2

3.6 Lunar BOWIE Configurations

3.6.1 Mission Configuration

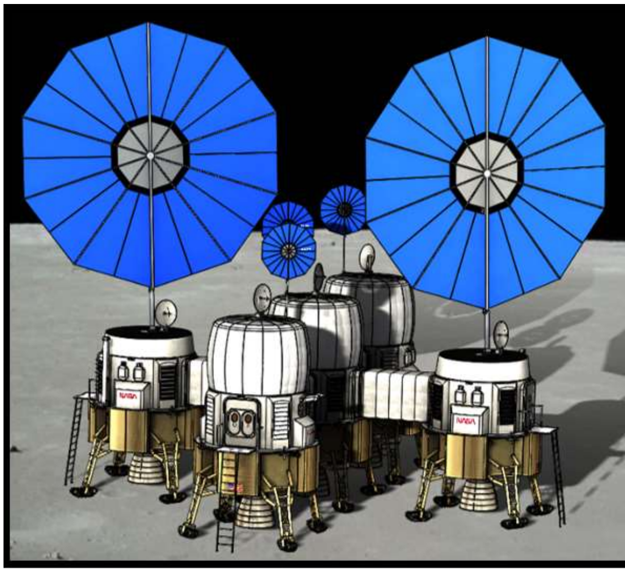


Figure 3.6.1-1 Mission Configuration

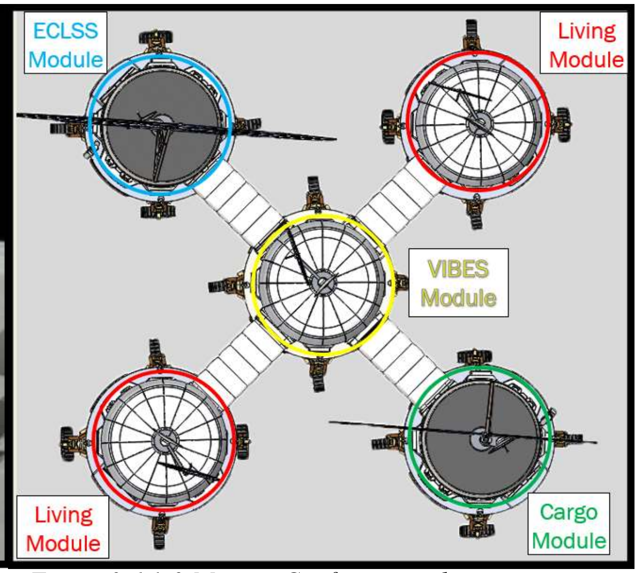


Figure 3.6.1-2 Mission Configuration layout - top view

The Lunar BOWIE architecture consists of five total BalLunar modules with four different variations, all of which are joined through semi-rigid extendable docking bridges. The design provides the maximum potential for expandability into a larger lunar base by including multiple airlocks per module that are designed to be interchangeably used either for access to the outside environment through the Suit Port Airlocks or configured to accept additional modules delivered for future expeditions. We chose to incorporate the use of Suit Port airlocks on the Living Modules for the purposes of preventing lunar regolith contamination of the interior. The color-coded top down view of the final mission configuration shown above (*Figure 3.6-2*) illustrates how the initial variants of these modules will be distributed. With one ECLSS Module, one Cargo Module, and two Living Modules all linked together through a common center-hub module called the VIBES Module.

Each module shares 90% common subsystems and primary structural elements for streamlined manufacturing and cost reduction in mind. Their commonality provides redundancy in the case one module becomes disabled then the expedient reconfiguration of another module to accept any mission critical equipment will not result in a total mission failure.

3.6.2 Stowed Configuration

The four BalLunar variants are stowed within SpaceX's Falcon Heavy fairing with the same configuration. The main challenge presented here stemmed from our propulsion design choice of implementing four Star-37GV Solid Rocket Motors (SRM) on the underside of our module's primary structure, leaving limited clearance and interface area for any commercially available Launch Vehicle Adapter (LVA). We determined designing a custom LVA would be necessary for staying clear of all vibration envelopes as per SpaceX's requirements. We performed trade studies using the maximum design loads and conditions SpaceX has provided within the Falcon Payload User's Guide [1] to investigate multiple designs and possible materials. Resulting in a selection of a Titanium (Ti13V-llCr-3A1) truss design which provided the best performance to overall mass. The FEA of the load conditions investigated for this selected design is included in the structural analysis section.

The BalLunar modules have been designed with axisymmetric featuring wherever possible to remain in compliance with the mass properties requirements of no more than 12.7 cm C.G. offset from the launch vehicle's centerline and the maximum allowed C.G. height of a payload of this weight class at no more than 2.9 m from the interface plane. With the smallest C.G. offset margin of all four variants being that of the Living Module at 10.9 cm off the centerline and only 1.5 m height above the interface plane.

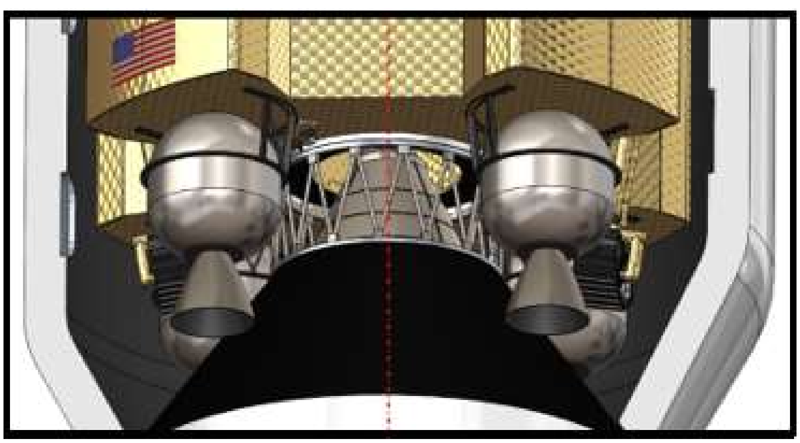


Figure 3.6.2-1 Custom Launch Vehicle Adapter

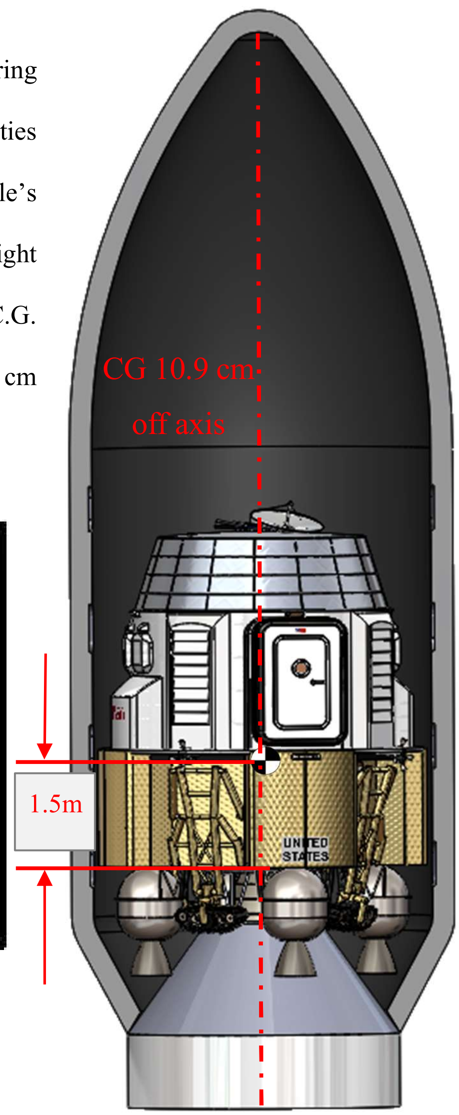


Figure 3.6.2-2 Stowed configuration within Falcon Heavy fairing

3.6.3 Cruise Configuration

For the cruise phase of our mission we have incorporated an Aluminum protective cover which completely encapsulates the upper inflatable structure on the two Living modules and VIBES module variants to mitigate the risk of impact or exposure damage prior to its deployment on the lunar surface.

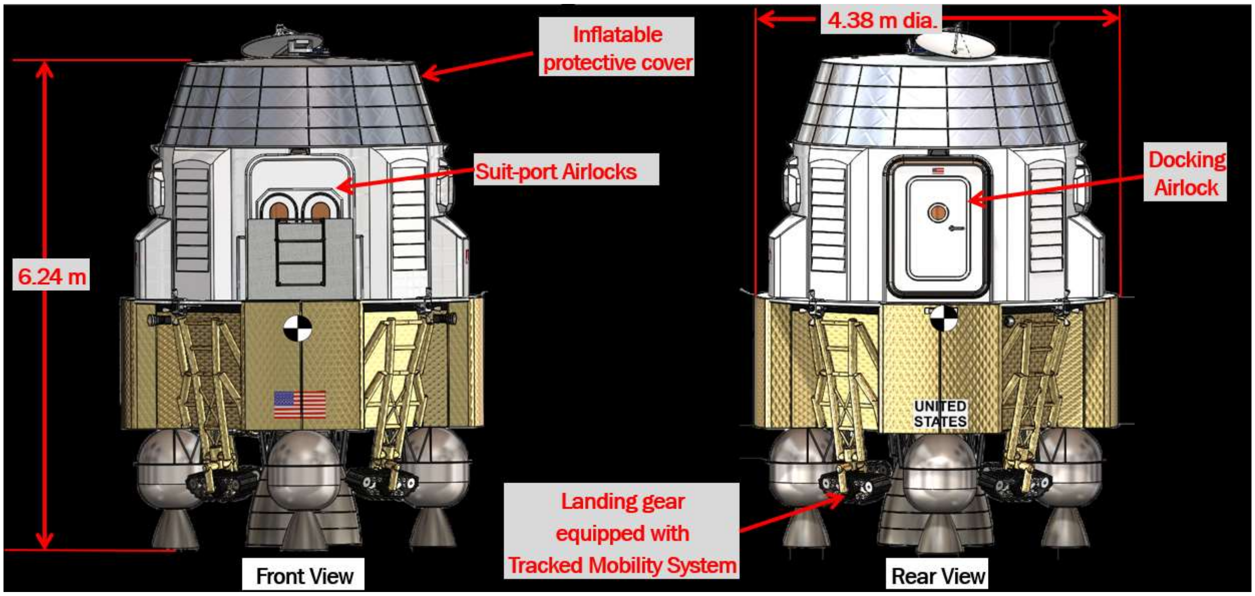


Figure 3.6.3-1 Cruise configuration (Living Module)

All variants shall be able to maintain communication with ground operations during all phases of the mission. To give our design this capability without the need to include multiple systems for cruise and surface operations we have implemented a pointing mechanism that has 360° rotation and 15° tilt capabilities at the top of each module's structure. During the cruise phase our spacecraft will maintain an attitude with the top of the module pointed roughly towards Earth and stay within these limits provided for maintaining communication.

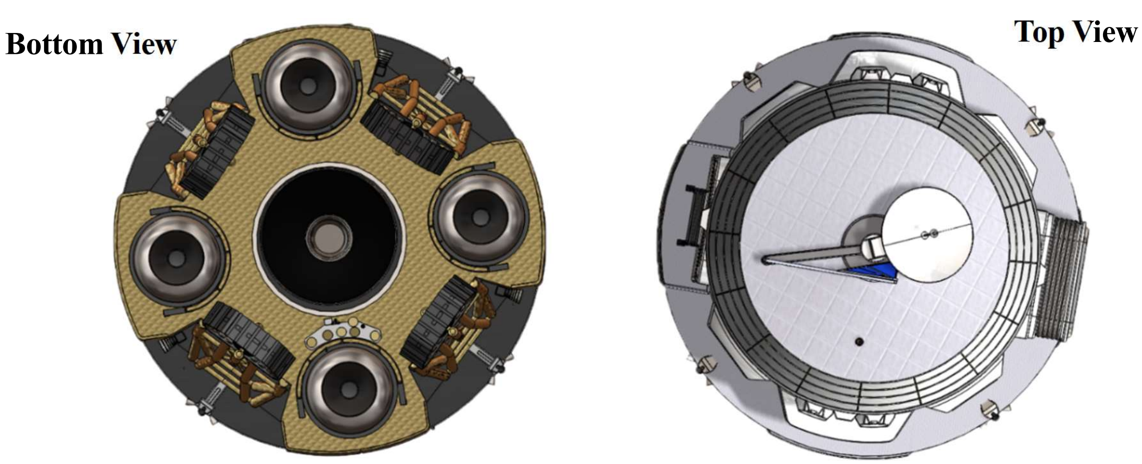


Figure 3.6.3-2 LM Cruise configuration Top and Bottom Views

3.6.4 Surface Deployed Configurations

3.6.4.1 Living Module

Once the Living Module (LM) has landed on the surface it will deploy its 2 m diameter temporary power array located at the top of the module like our telecom system it also utilizes a custom mechanism that can rotate 360° with a 15° tilt capability for the purpose of maintaining the incident lighting angle for maximum efficiency. Simultaneously the LM will communicate with the awaiting VIBES Module to obtain the precise location of its designated vacant docking port. The modules will utilize their Tracked Mobility Systems for transiting the approximate 50 m distance to the VIBES Module to begin their docking sequence. The decision for designing a tracked system versus a more traditional wheeled design was made after comparing estimates of traction capabilities in soft sand on slopes of up to 15° as an approximation of lunar regolith. The tracked design though more complex and susceptible to the abrasive lunar regolith showed far superior traction capabilities for a vehicle dry mass estimated at 10,000 kg. The wider footprint area of the tracked design allows the dispersion of the vehicles weight while a wheeled design would sink and become bogged down. We gave priority to the traction capabilities believing the complexity and possibility of mechanical issues from regolith intrusion would pose less risk to the mission considering the short transiting distances and one time use of the system. Two electric motors per system are utilized in the design one powering the racks themselves and the second providing the 360° precision steering capability built into each "ankle" of the landing gear system seen in *Figure 3.6-8.*

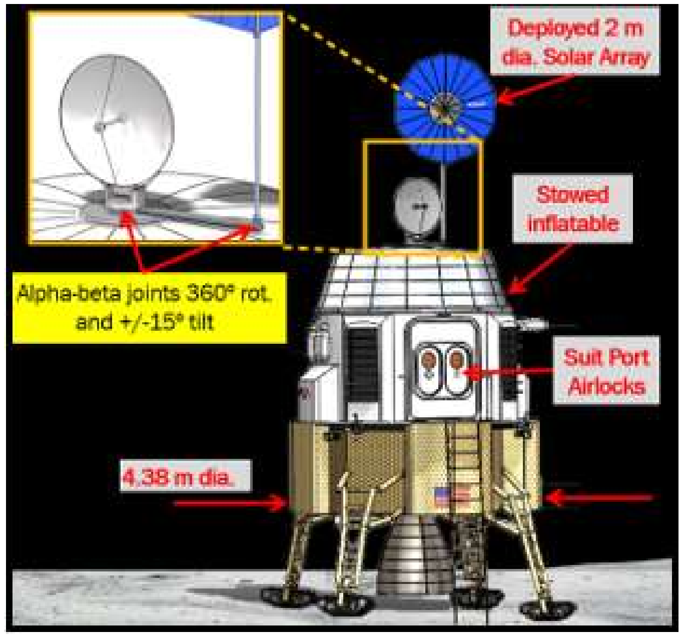


Figure 3.6.4.1-1 LM power deployment (pre-inflation)

Figure 3.6.4.1-2 Tracked Mobility System

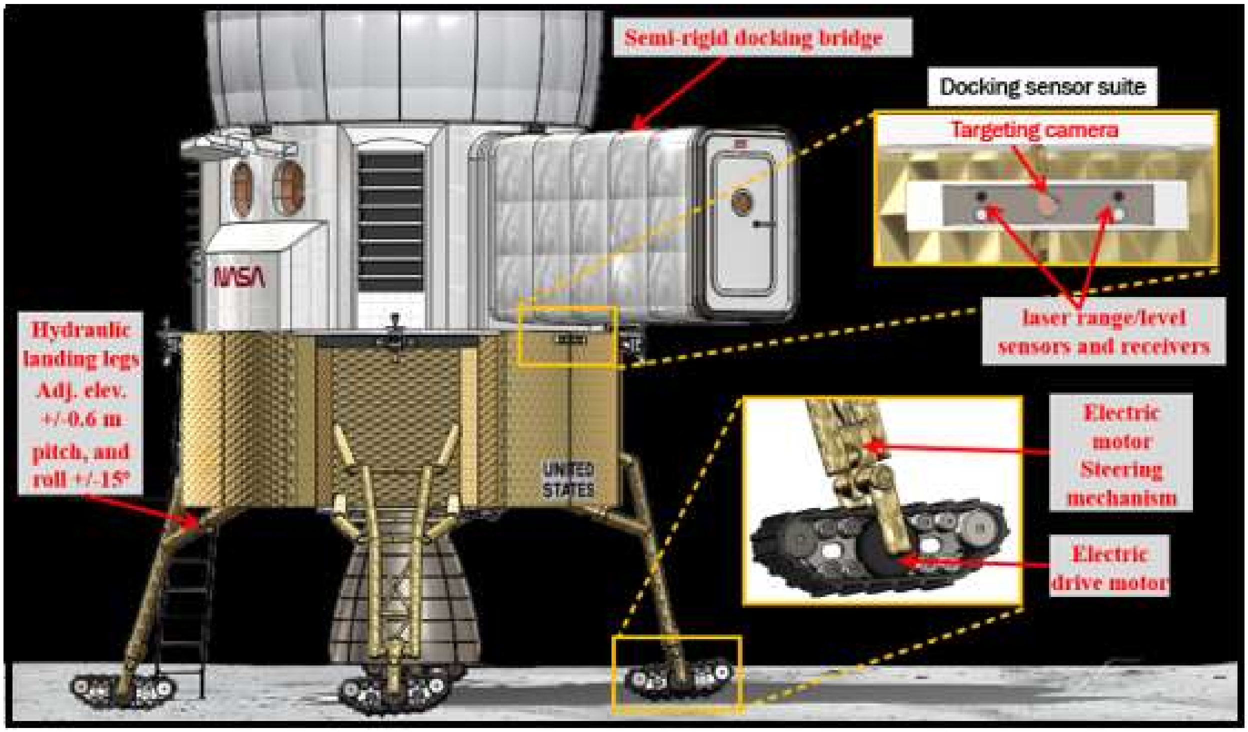


Figure 3.6.4.1-3 Lunar Bowie Autonomous Rendezvous mechanisms

The Hydraulic Landing Gear System is a vital mechanism providing the capability to adjust the elevation, and automatically level out the module to establish conformity between all modules for precision docking. Based off topology maps of our target location and stability calculations we have designed the system to compensate for up to a 15° slope and still successfully complete the planned final mission configuration. An in-depth FE analysis is included in the Structures section detailing the material, sizing, axial and lateral load strength tests for this selected design.

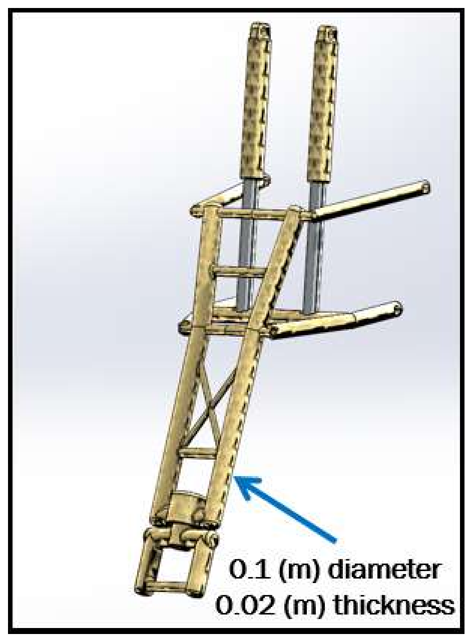


Figure 3.6.4.1-4 Hydraulic Landing Gear (adjustable height)

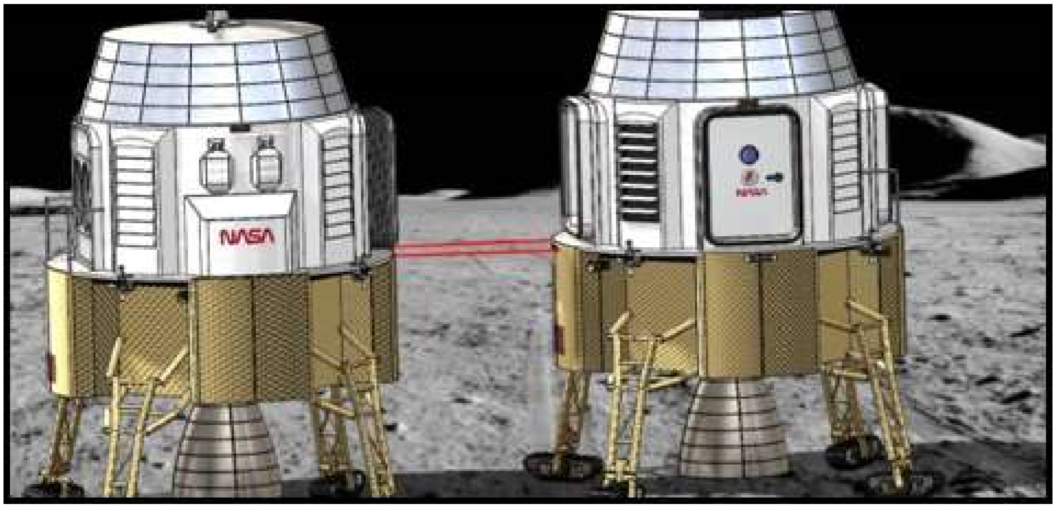


Figure 3.6.4.1-5 Autonomous Rendezvous Docking Sensor Suite operation

Once the LM has achieved a roughly parallel orientation and 10 m distance from the VIBES module the Docking Sensor Suite (DSS) will take over for the final stage of completing the autonomous docking sequence. The DSS consists of a docking camera and two laser range and level sensors built into each modules structure located just below each extendable docking bridge or airlock as shown in *Figure 3.6-9*.

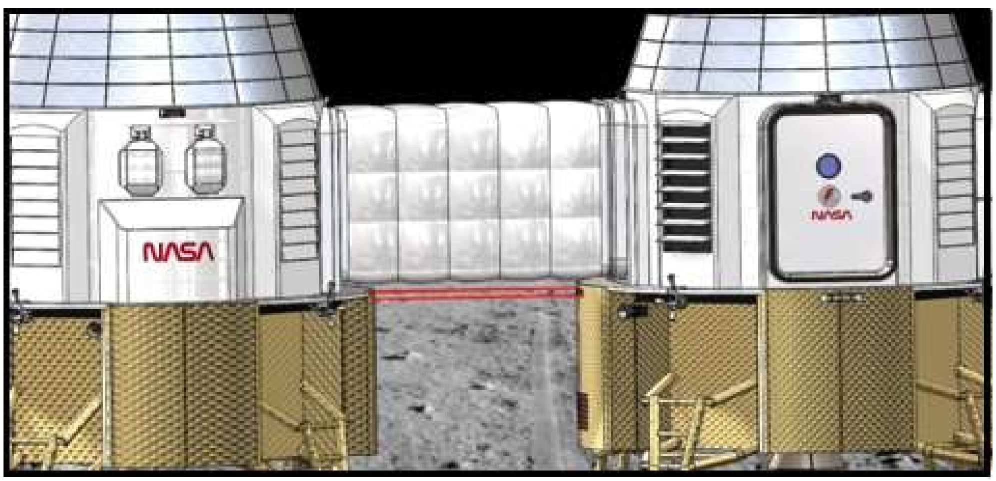


Figure 3.6.4.1-6 Autonomous Rendezvous Bridge extension and connection

The docking camera will aid in the initial measurements for conforming the LM's elevation and leveling to match the VIBES module's orientation. Once the two projected lasers are lined up correctly and received by the opposing DSS the LM will proceed in fine tuning the orientation between the modules including beginning to close the stand-off distance between the modules to the designed bridge extension distance of 2.4 meters. Finally, the Semi-Rigid Docking Bridge (SRDB) extends from within the living module and completes the docking sequence when the VIBES module communicates a successful lock has been made.

A closer look at the Semi-Rigid Docking Bridge seen in *Figure 3.6-13* with the exterior inflatable walls removed shows the mechanism within to be a Hydraulic Telescoping Bridge and from this bridge is suspended a Telescoping Suspension Walkway which deploys when pulled in tandem by the overhead beams as they extend.

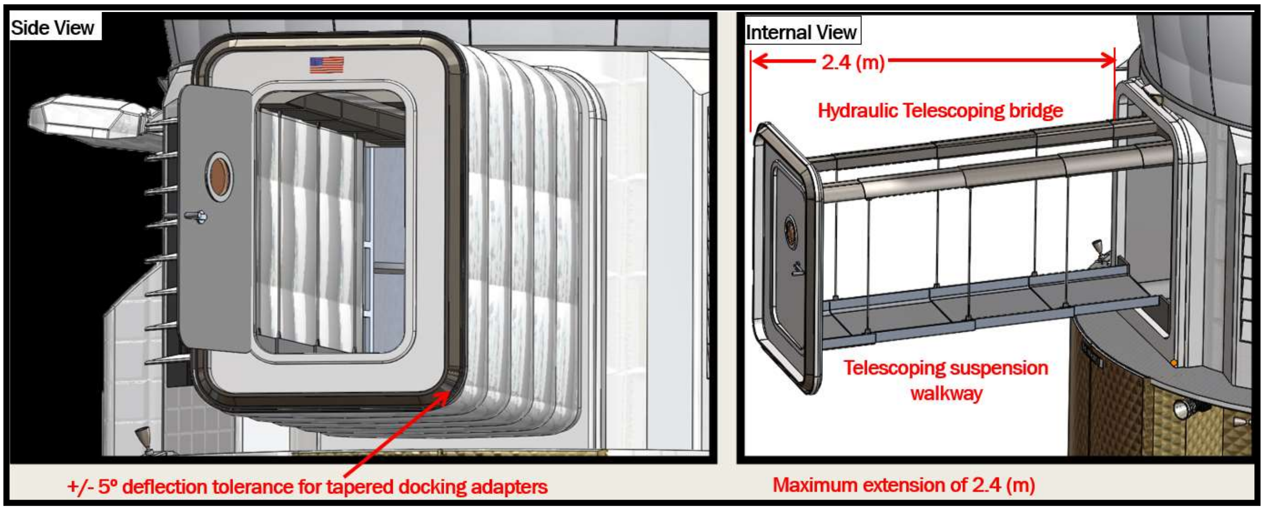


Figure 3.6.4.1-7 Semi-rigid Docking Bridge and internal mechanism

The design choices behind the Semi-Rigid Docking Bridge (SRDB) were driven by the need for a compact stowable lightweight bridge while continuing to remain compliant with radiation protection requirements as well as our LV mass-properties requirements for maximum C.G. offset. Traditional methods for radiation protection within solid exterior walls did not trade well against the alternative of utilizing an inflatable bridge design incorporating built-in radiation protection layers. The inflatable design provides superior compactness at minimum mass and therefore the minimum impact in offsetting the spacecrafts C.G. as well as enabling these BOWIE modules to provide sufficient radiation protection throughout, so the Astronauts can work efficiently without the need to rely on wearing restrictive spacesuits for radiation protection. The SRDB was designed such that the exterior inflatable walls carry no structural loads only the ΔP loads as they "float" around the telescoping bridge mechanism. The FEA of the internal mechanism evaluated for maximum deflection is included under the Structural analysis sub-section.

Both Living Module's lower levels are capable of being reconfigured for supporting multiple scientific payloads along the lines of these primary science objectives (but not limited to) lunar surface exploration, lunar volatiles research, regolith uses for construction or farming, and human habitability of the lunar surface. Some specific instrumentation we have designed to support are the lunar Volatiles Scout, Advanced Plant Habitat (APH), and Regolith 3D Printer. Each module holds three International Standard Payload Racks (ISPRs) with an allocated 800 kg mass budget for carrying scientific payloads and other supporting equipment.

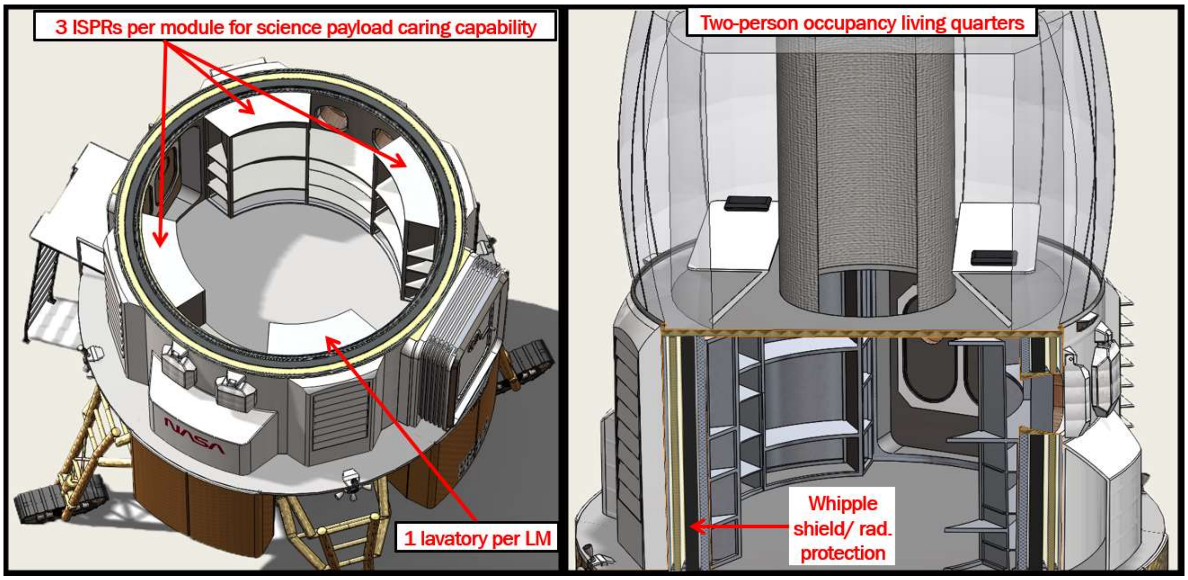


Figure 3.6.4.1-8 Living Module interior layout

The upper level of the LM is the inflatable upper structure as seen in the provided cross section above (*Figure 3.6-14*) which first inflates only after all modules have landed and docked to the VIBES module. Implementing this inflatable structure increases the living volume where the crew's living quarters are located, these modules have been configured as two person occupancy living modules with a usable living volume of 36 m^3 which is slightly in excess of NASA's requirement for a minimum of 17 m^3 of usable living space per crew for a nominal mission duration of 45 days. This custom inflatable structure includes radiation shielding layers built to our specific radiation shielding requirements. The center column seen dividing the interior of upper structure is a critical structural element not just for the inflatable itself but it provides the support for our crucial telecom and power systems allowing the utilization of that key vantage point atop the module. The center column stows on the lower floor and deploys upwards in tandem with the inflation of the upper structure locking into place when the inflatable is fully deployed, another important benefit provided by this center column is it divides the living quarters such that some personal private space can be afforded to the Astronauts promoting better phycological health over the 45 day mission duration.

3.6.4.2 Cargo/ECLSS Module

The Cargo and ECLSS Modules are structurally identical and only differ from each other by how we have configured their interiors to carry different payloads and subsystems. These modules primary purposes are to serve as platforms that host some of the more vital subsystems that are essential to the Lunar BOWIE's functionality.

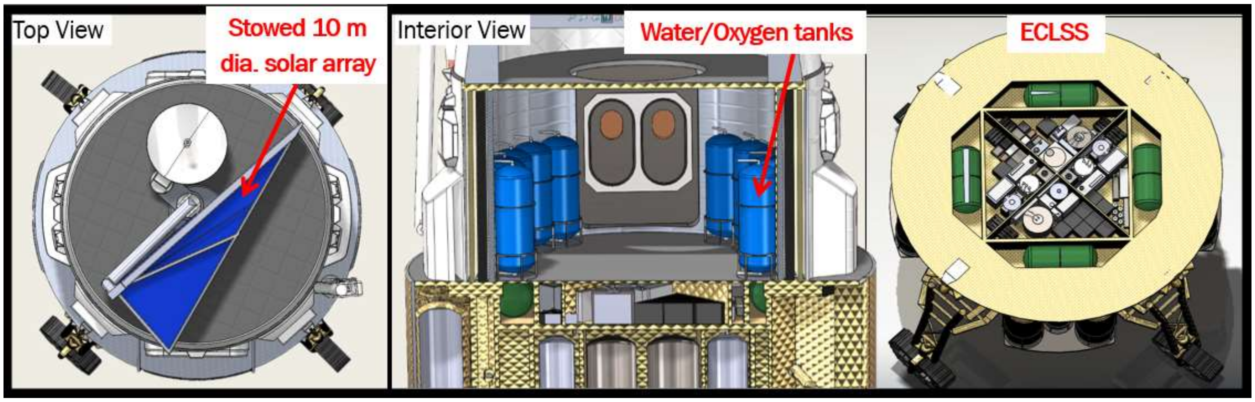


Figure 3.6.4.2-1 ECLSS Module top view (left), and interior views of ECLSS components

The most apparent difference is these modules have solid aluminum upper structures as opposed to the inflatables. The main driver behind that decision was these modules both must host the base's main power system, which consists of two 10 m diameter MegaFlex Solar Arrays (*Figure 3.6-15*). We conducted trade comparisons of several different main power supply systems and deployment options. We determined our final choices considering our preliminary numbers for mass margins, internal payload capacities, illumination availability at our target location, and our preference for developing an autonomously assembled base. A lightweight externally stowed entirely solar power system was the best fit for powering our design. The ability to stow these massive arrays externally would save us vital internal volume and mass savings by trading the heavier inflatable for a lightweight reinforced aluminum structure. We were able to reallocate that much needed mass margin towards other developing subsystems. The *Figure 3.6-15* above is showing cross sections of the ECLSS Module and how we are distributing most of our components and spares parts of our ECLSS subsystem. From the top down view of the ECLSS Module shows the mechanism developed for stowing, deploying, and maintaining the correct solar incidence angles of our massive MegaFlex Solar Arrays.

Both the Cargo and ECLSS Modules are equipped with a Remote Manipulator System or what we like to call the "BalLunarm" as seen in *Figure 3.6-16* below. We opted for incorporating the BalLunarm rather than an elevator lift system because the arm provides multifunctionality such as large sample retrievals meaning any object too large to climb a ladder with can be lifted to the airlock or the other way around. We also most importantly gain the capability of heavy lift maintenance jobs, for example without the BalLunarm we would not be able to have redundancy for our main power arrays since a disabled 10 m diameter array could not be removed and replaced by human hands alone.

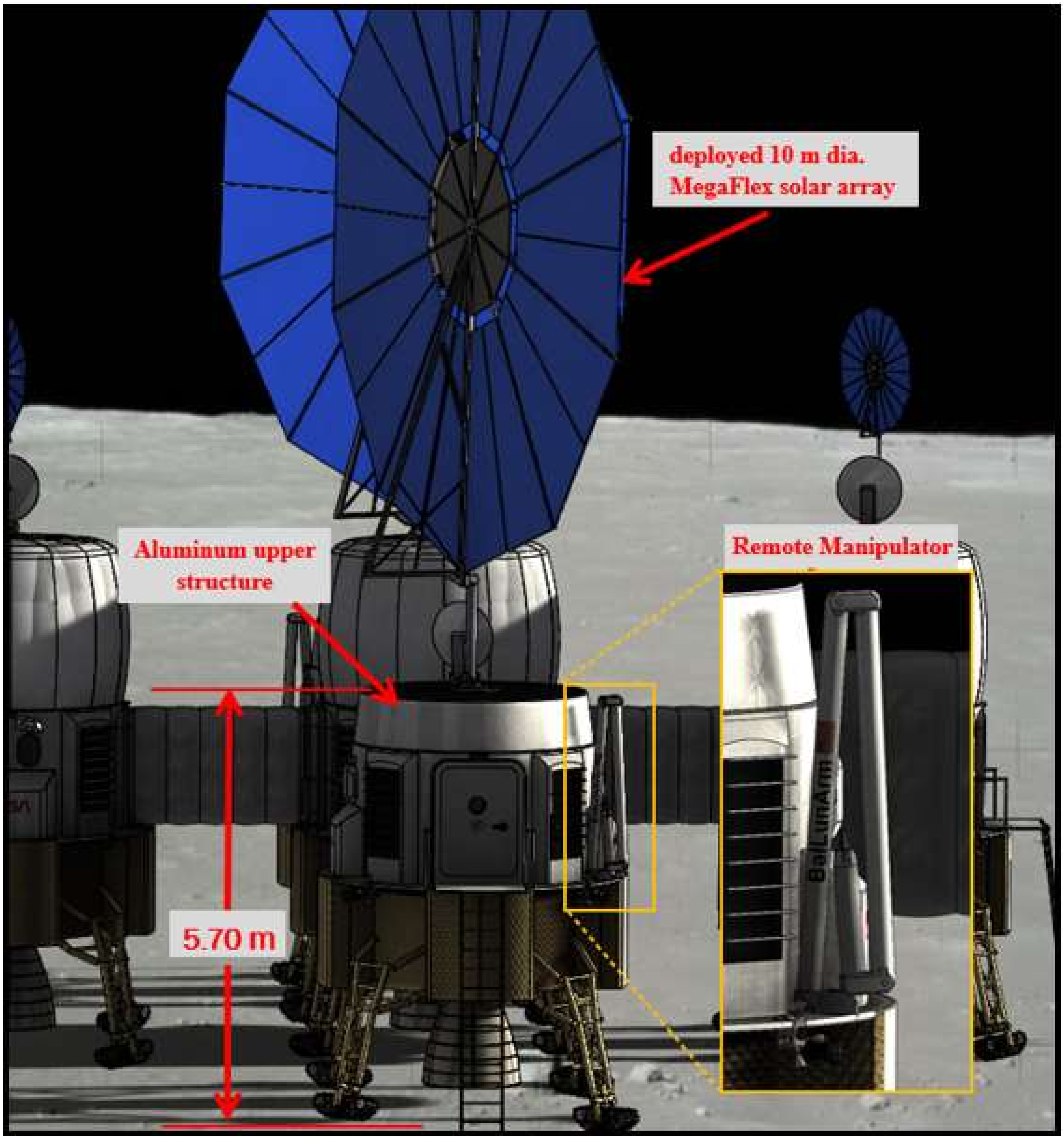


Figure 3.6.4.2-2 Cargo/ ECLSS Deployed view, with the BalLunarm System

3.6.4.3 VIBES Module

The VIBES Module stands for the "VItality Base Element System" named for the fact that it is essentially the central hub module that links all the other modules together. For the initial first mission it would be the only one of its kind with no redundant pair. Though the loss of the VIBES Module would not spell certain mission failure considering it carries no vital subsystems and the other modules could still all successfully dock to each other without it and continue the mission. Though the VIBES Module provides what we have determined to be the most efficient spatial layout in comparing existing designs and listening to NASA Astronauts who continually emphasize the importance of an efficient interior layout designed not just for maximizing productivity but an Astronaut's entire agenda. With a relatively short time of 45 days on the lunar surface time is money.

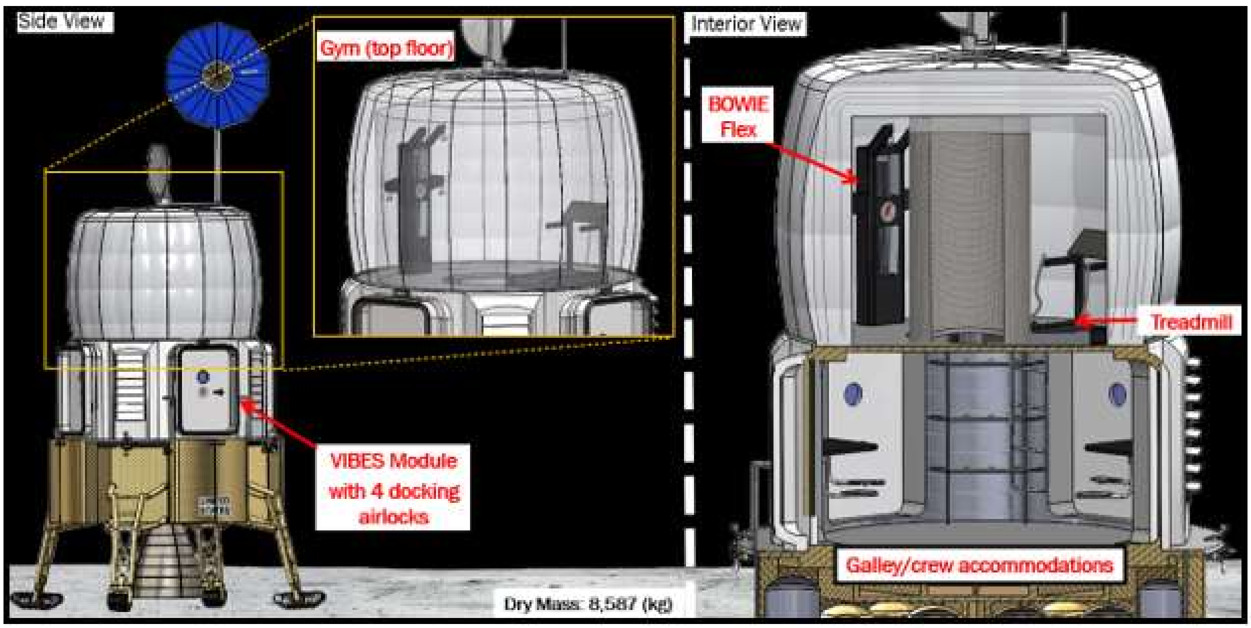


Figure 3.6.4.3-1 VIBES Module Deployed view, and internal layout

Shown in *Figure 3.6.4.3-1* above the interior design of the VIBES Module consists of the galley and crew accommodations on the lower level. Here is where Astronauts would be able to socialize with the rest of the crew. The upper floor includes a gym as per NASA requirements for 100 min. a day of physical fitness for a mission of this duration. The Vibes Module shares the same style inflatable upper structure as the two Living Modules though within the Gym of the VIBES Module there will be some necessary assembly required for the fitness equipment.

3.7 Target Location

The lunar surface presents an environment with vast opportunities for scientific discovery and exploration. The establishment of a lunar base camp should consider a target location with a large potential for scientific return, crew survivability, and extensibility that will enable future deep space missions, as specified by the RFP. The lunar Science for Landed Missions Workshop Findings Report provides a summarized analysis of priority lunar landing sites, referred to as Regions of Interest (ROI), and key science and exploration goals determined by NASA and the scientific community. The report addresses Strategic Knowledge Gaps (SKG) in lunar science and exploration, encompassing gaps in the data and technologies necessary for the successful presence of humans on the Moon and other solar system destinations. Three broad lunar science and exploration SKG themes are identified [2]:

SKG 1 - Understand the lunar resource potential: Composition and distribution of lunar volatiles

SKG 2 - Understand the lunar environment and its effect on human life: Biological impact of solar activity and radiation, lunar dust, and low gravity environment at the lunar surface

SKG 3 - Understand how to work and live on the lunar surface: Development and integration of a lunar infrastructure with resource production while maintaining human safety

In addition to addressing SKGs, the target location must also consider conditions of illumination, topography, and proximity to other ROIs, all of which characterize the Figures of Merit (FOM) used to determine the adequacy of each region. To determine the target location for the Lunar BOWIE, trade studies were performed using the aforementioned FOMs, quantified by percent illumination, slope and terrain roughness, and concentration of ROIs. A high percent illumination correlates to consistent power generation for the Lunar BOWIE, small slopes and terrain roughness allow for the safe landing of spacecraft, and a high concentration of ROIs enables base expandability and resource production.

Having selected the lunar south pole as general target, a trade study of specific regions of interest was performed to determine the target location for the lunar base camp. Percent Illumination proves to be the most weighted FOM since it is directly correlated to electrical power generation and thermal cycling of structures, thus ROIs with less than 70% illumination were omitted from the trade study. *Figure 3.7-1* includes the ROIs considered for the target location, which were the following: Shackleton Rim Location 1 (SR1), Shackleton Rim Location 2 (SR2), and Connecting Ridge Location 1 (CR1) [3]. *Figure 3.7-1* also displays an optimal transverse path between

ROIs shown in green. Based on the trade study results shown in *Table 3.7-1*, Connecting Ridge Location 1 was chosen as the target location for the Lunar BOWIE.

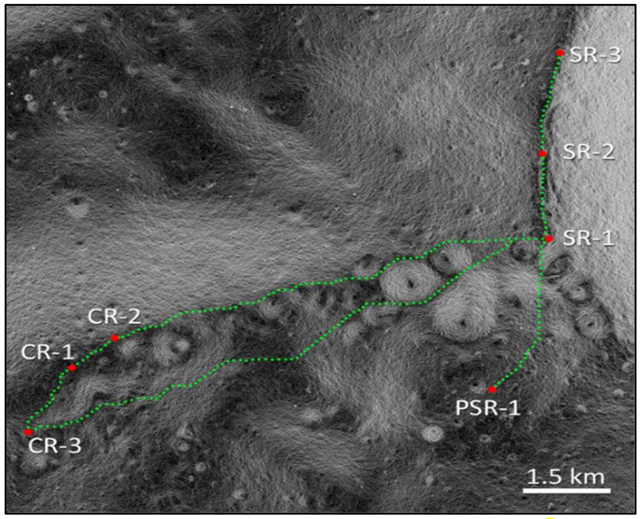


Figure 3.7-1: South Pole Regions of Interest [3]

Table 3.7-1: Target Location Trade Study

FOM's / Alternative Target Locations	T0.0-1 % Illumination WF = 1		T0.0-1 Slope WF = 1		T0.0-3 Roughness WF = 1		**Weighted Total**
	U	W **=WF* U**	U	W	U	W	
Connecting Ridge Location 1	9	9	3	3	3	3	**15**
Shackleton Rim Location 1	3	3	1	1	3	3	**7**
Shackleton Rim Location 2	1	1	1	1	3	3	**5**

Using data from the Lunar Reconnaissance Orbiter (LRO), Phillip Gläser from the Technical University of Berlin evaluated the conditions of illumination, topography, slopes, and surface roughness at CR1 over a 19-year period, accounting for varying conditions throughout the lunar precession cycle [4]. Illumination maps developed by Gläser are shown in *Figure 3.7-2*, where one pixel represents an area of 400 m^2. *Figure 3.7-2A* shows an accumulated illumination map at 2 meters above ground, where CR1 receives an average illumination greater than 80%, represented by the pixels outlined in black. The points of greatest illumination within CR1 are referred to as Spot 1 and Spot 2 (outlined in white in *Figure 3.7-2A*), which has 88.1% average illumination at 2 meters above ground, with a maximum continuous period of illumination of 233.87 days and an eclipse period of 4.58 days. *Figure 3.7-2B* shows the accumulated illumination at 10 meters above ground, where CR1 receives an average illumination greater than 90%, represented by the pixels outlined in black.

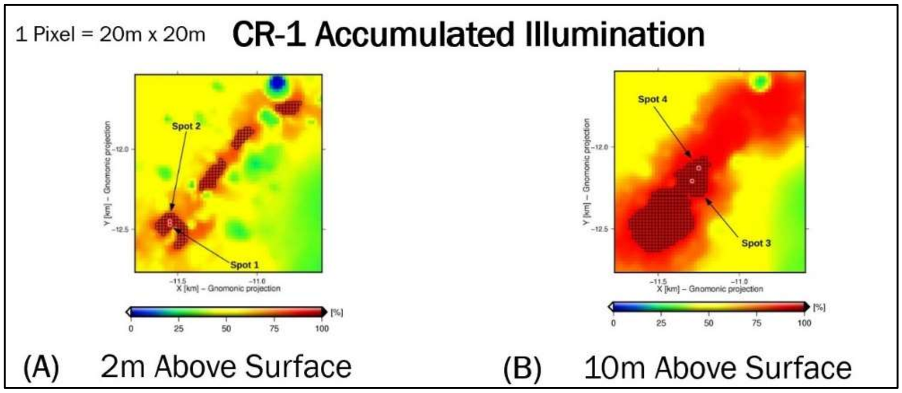


Figure 3.7-2: Connecting Ridge Location 1 Accumulated Illumination [4]

Slope and terrain roughness maps developed by Gläser are shown in Figure 3.7-3, where one pixel represents an area of 400 m^2. The slope map in *Figure 3.7-3A* was constructed using a 150 m baseline measurement, where only slopes less than 5º are color coded, showing that Spot 1 is surrounded by slopes less than 2º. Terrain roughness maps were developed using two approaches: the Sigma-Z approach *(Figure 3.7-3B*) which analyzes the standard deviation from a reference plane, and the Pulse Width Derivation approach *(Figure 3.7-3C*) which measures changes in the reflected laser pulse emitted from the LRO. The area surrounding Spot 1 displays a Sigma-Z roughness of under 2 m for a 150 m baseline measurement, and a Pulse Width Derived roughness of under 30cm for a 5m baseline measurement. By superimposing data and maps for illumination, slope, and terrain roughness, Gläser developed a landing site map (*Figure 3.7-3D*) measuring the percent safety for a hypothetical mission based on conservative lander design constraints, representing a realistic lander. For a landing dispersion area with a 200 m diameter (31,415 m^2), the average landing safety at Spot 1 is 97.82%. CR1 attributes are summarized in *Table 3.7-2*

The favorable conditions at Connecting Ridge Location 1 enable the safe landing of spacecraft over a large area and its close proximity to other ROIs yields the potential for extensibility regarding future missions, establishing CR1 as the prime target location for the lunar base camp.

Table 3.7-2: Connecting Ridge Location 1 Attributes [4]

Lunar Coordinates	Illumination @ 2m Above Ground	Slope (deg)	σ_z Roughness (m/m)	PWD Roughness (cm/m)	Landing Dispersion Area (m^2)
89.4395ºS, 222.8066ºE	88.1%	< 2º	0.013	6	129,825

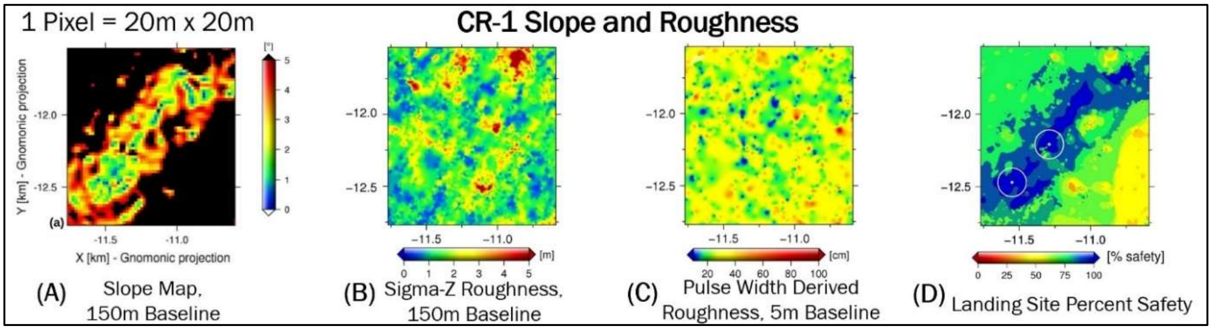


Figure 3.7-3: CR1 Slope, Terrain Roughness, and Landing Safety [4]

3.8 Trajectory

The establishment of a lunar base camp inherently requires the transport of highly massive systems to the lunar surface. Given the lunar south pole as the target location, the trajectory must be designed to efficiently deliver the modules comprising the Lunar BOWIE while considering the required launch C_3, required spacecraft ΔV, and time of flight (TOF), all of which characterize the figures of merit for the orbital trajectory. A lower launch C_3 corresponds to a higher deliverable mass by the launch vehicle, a lower spacecraft ΔV allocates mass to the base camp, and a shorter time of flight leads to less propellant boil off, saving mass in terms of propellant margins and thermal protection systems.

The orbital trajectories considered for the mission consist of direct ballistic lunar transfer orbits and low-energy ballistic lunar transfer orbits. Jeffrey Parker of NASA's Jet Propulsion Laboratory has performed a survey of ballistic transfers to low lunar orbit, evaluating direct and low-energy ballistic transfer orbits with widely varying performance parameters [5]. In his survey, Parker used a back-propagation method to generate efficient transfer orbits of both types. The process used to generate each trajectory begins by defining a target lunar orbit and lunar orbit insertion (LOI) state. The state is then propagated backwards in time for 200 days, from which the trajectory perigee and perilune passages are identified. Finally, if the trajectory displays favorable conditions, the trajectory performance parameters are characterized. Using this method to develop either type of trajectory may lead to the inclusion of Earth phasing orbits and/or lunar flybys, causing the spacecraft to pass through the Van Allen radiation belts [5], risking degradation of vital spacecraft components. Thus, trajectories involving Earth phasing orbits or lunar flybys were excluded from consideration. The spacecraft is assumed to be launched from Kennedy Space Center (KSC), thus limiting the possible low earth orbit (LEO) equatorial inclination within the range of 28.5^{o} to 57^{o}. The transfer orbits considered for the trade study begin by launching into a 185 km circular LEO parking orbit from which a trans-lunar injection is performed, followed by a ballistic coast period with a subsequent lunar orbit insertion into a 100 km polar low lunar orbit (LLO). The direct and low-energy transfers are shown in *Figure 3.8-1* and their corresponding performance parameters [5] are shown in *Table 3.8-1*. The direct lunar transfer orbit was selected as a result of the trade study shown in *Table 3.8-2*.

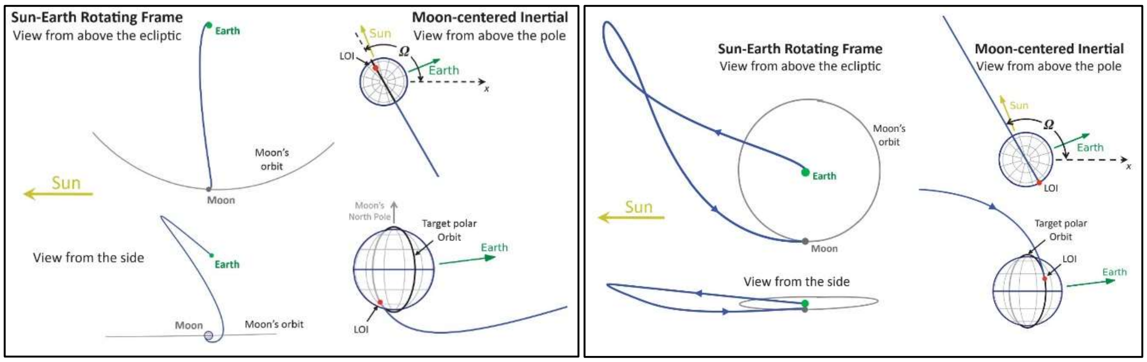


Figure 3.8-1: Direct lunar Transfer (left) and Low-Energy lunar Transfer (right) [5]

Table 3.8-1: Transfer Orbit Performance Parameters [5]

Orbital Trajectory	**Ω [deg]**	**ω [deg]**	**ΔV_{LOI} [m/s]**	**TOF [days]**	**LEO Equatorial Inc. [deg]**	**C_3 [km^2/s^2]**
Direct Transfer	120	326.4	860.4	4.155	43.459	-2.058
Low-Energy Transfer	120	169.2	669.3	83.483	29.441	-0.723

Table 3.8-2 Orbital Trajectory Trade Study

FOMs / Alternative Trajectories	FC0.0-1.1 ΔV_{LOI} WF = 2		M0.0-7 TOF WF = 1		FC0.0-1 Launch C_3 WF = 2		Weighted Total
	U	W **=WF* U**	U	W	U	W	
Low-Energy lunar Transfer	9	18	0	1	3	6	**25**
Direct lunar Transfer	3	6	3	3	9	18	**27**

Due to the geometry of the Sun-Earth-Moon system, the direct ballistic lunar transfer orbit presents itself in in two variations: arrival at first quarter Moon and arrival at third quarter Moon, corresponding to times of arrival when the Sun-Earth-Moon angle is approximately 90º. Both transfer orbits display similar performance and orbital parameters [5] shown in *Table 3.8-3*, with the exception of the right ascension of the ascending node, which is shifted by 180º since arrival at first and third quarter Moon correspond to lunar phases that are in direct opposition of each other. The spacecraft was designed based on the upper limit of the constraining performance parameters to ensure the capability of arrival at both first and third quarter Moon. The corresponding ΔV's from lunar orbit insertion to landing are shown in *Table 3.8-4*.

Table 3.8-3: Performance Parameters for Arrival at 1st and 3rd Quarter Moon [5]

Arrival at:	Ω (deg)	ω (deg)	ΔV_{LOI} (m/s)	Duration (days)	LEO Equatorial Inc. (deg)	C_3 (km^2/s^2)
1st Quarter Moon	120	326.4	860.4	4.155	43.459	-2.058
3rd Quarter Moon	300	330	867.5	4.004	43.459	-2.045

Table 3.8-4: Mission ΔVs

Maneuver	ΔV (m/s)
Lunar Orbit Insertion	870
Inclination Change	15.2
Descent Orbit Insertion	19.46
Descent/Landing	1,692
Total	2,596

3.8.1 Launch Windows

Ballistic, two-burn lunar transfer orbits with no Earth phasing orbits or lunar flybys repeat on a monthly basis. The most efficient performance parameters present themselves when the Sun-Earth-Moon angle is nearly orthogonal at lunar orbit insertion, focus was thus placed on arrival at first and third quarter Moon [5]. An itinerary of possible launch opportunities was developed, spanning from June 2029 throughout December 2030 with two launch windows per month (arrival at first and third quarter Moon). This expansive itinerary of possible launch opportunities helps mitigate against launch delays that may occur due to adverse weather effects or launch schedule conflicts. The extensive list of launch opportunities along with their corresponding trans-lunar injection dates, times, and times of flight between orbital maneuvers are listed in the appendix, with the first two opportunities listed in *Table 3.8.1-1*.

Table 3.8.1-1: Launch Opportunities and Times of Flight

Launch Window	Arrival Moon Phase	Quarter Moon Date/Time	TLI Date/Time	TLI to LOI TOF	LOI to Inc. Change TOF	Inc. Change to DOI TOF
June 2029	1st Quarter	06/19/2029 09:54	06/15/2029 06:10:48	4d 3h 43m 12s	10m 59s	29m 26s
June 2029	3rd Quarter	06/03/2029 18:19	05/30/2029 14:35:48	4d 3h 43m 12s	10m 59s	29m 26s

3.9 Launch Vehicle

Table 3.9-1 Launch Vehicle Requirement Statements

Launch Vehicle Requirement Statements
Mass properties requirements of no more than 12.7 cm C.G. offset from the launch vehicle's centerline and maximum allowed C.G. height of a payload of this weight class at no more than 2.9 m from the interface plane
The spacecraft shall be designed to withstand the maximum design loads of +6.0 to -2.0 gs axial and ±3.5 gs lateral

The launch vehicle used for delivering the Lunar BOWIE modules was chosen to be the expendable Falcon Heavy System by SpaceX. A total of five expendable Falcon Heavy launches will be necessary.

Chosen LV: Falcon Heavy

- Falcon Heavy (Expendable)
- $150 M per Launch
- Fairing constraints: 4.6 m Ø x 11 m height *Figure 3.9-1*
- Flight Proven
- 21,000 kg Payload Capacity to TLI

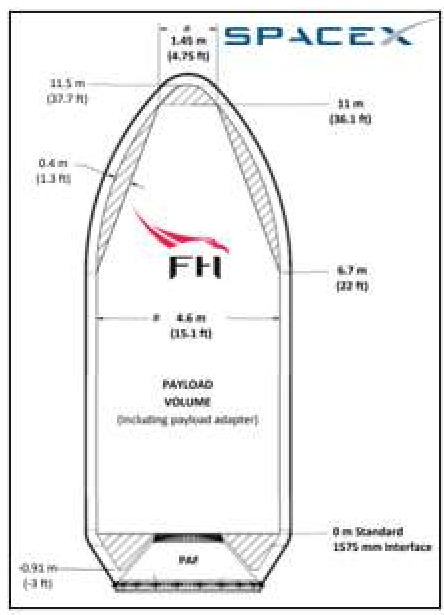


Figure 3.9-1 LV Vibrations Env.

L.V.	Max. Payload Capacity to TLI	Cost per Launch FY2019	Fairing Dimensions	Status
Falcon Heavy 	21,000 kg	$150 M USD	Ø4.6m x h'6.7m h_{max}= 11m	Operational
SLS Block 1B Cargo 	40,000 kg	~ $2 B USD	Ø7.5m x h'9.8m h_{max}= 19.1m	In Development
Starship 	100,000 kg (w/ refuel)	≤ $100M	Ø8.4m x h'17.8m h_{max}= 21.6m	In Development

Figure 3.9-2 Launch Vehicle Trade Study

4.0 Lunar BOWIE Subsystems

The following section details the design of the subsystem and subsystem components for the BalLunar modules and the lunar base camp. Subsections include the main propulsion system, ADCS, TCS, Structures, Telecommunications, CD&H, Power, ECLSS, Crew Accommodations, and Payloads. Each subsystem is governed by their respective derived requirements. Trade studies are performed to down select subsystem components that best meet the requirements.

4.1 Propulsion System

Table 4.1-1 Propulsion Requirements

Propulsion Requirement Statements
The Propulsion system shall be restartable for multiple mission deterministic maneuvers.
The Propulsion system shall possess a minimum of four degrees of thrust vectorability.
The Propulsion system shall possess deep throttling capability for soft landing.

The design choice for our main propulsion system was driven by limited internal volume capacity and our technical derived requirements that state our propulsion system shall be capable of throttling for soft lunar landing, thrust vectoring for the necessary landing accuracy required, and multiple restarts to account for the minimum three mission deterministic maneuvers. Multiple hybrids and bi-propellant engines using cryogenic or storable propellant options along with possible supplementation of these bi-prop. engines with SRMs where investigated in our trade studies. Our final down selected propulsion system consists of a combination of Aerojet Rocketdyne's RL-10C utilizing cryogenic bi-propellants (LOx/H_2) supplemented by four Northrop Grumman Innovation System's Star 37GVs.

This combination was selected because the addition of utilizing the four SRMs reduced the required ΔV that would have been provided solely by the bi-propellant engine therefore reducing overall bi-propellant volume needed while simultaneously utilizing previously unused space in affixing the four SRMs to the underside of our primary structure. We chose the Star 37GV model because it provides the proper combined ΔV capability to mass ratio of the few possible models that utilize the crucial vectorable nozzle [6]. This thrust vectoring capability of the SRMs allows the thrust vector to be aligned nearly through the C.G. of the spacecraft when ignited to minimize the stability torque issues that arise with having four solid rocket motors not all igniting in perfect unison. The SRM's perform approximately 600 m/s of the initial ΔV burn for lunar Orbit Insertion and will be subsequently jettisoned afterward for mass reduction and most importantly providing a clear field of view for our landing sensors during the decent phase. This dual propulsion configuration saves 9 m^3 of internal volume which helps to solve the design challenges presented when utilizing multiple smaller modules with limited internal payload capacities and meeting the NASA minimum living space requirements. The choice for the RL-10C model was justified by its deep throttling capability and efficiency provided by its optimization for use in vacuum [7].

4.2 Attitude Determination and Control Systems

The Attitude Determination and Control System (ADCS) for each BalLunar module is responsible for performing attitude and state vector determination along with attitude reorientation and stabilization throughout the entire mission spaceflight phase. Control modes are specifically determined for the Lunar BOWIE mission, shown in *Table 4.2-1*.

Table 4.2-1: Attitude Control Modes

Control Mode	Description
Orbit Insertion	No spacecraft control, control provided by launch vehicle upper stage
Attitude Acquisition	Initial attitude determination and stabilization
Translunar Cruise/ Station Keeping	Vehicle reorientation as necessary in response to environmental disturbances, and maintaining a passive thermal control mode (BBQ roll)
Slew	Reorientation for proper attitude during pre-specified orbital maneuvers
Descent/Landing	Pitch up maneuver, attitude hold during descent/landing

Requirements were iteratively derived by analyzing each phase of spaceflight while considering ACS sensor constraints, maneuvering requirements, and environmental disturbance torques. Spacecraft stability and accurate attitude determination and control are paramount for mission success and thus must be maintained throughout spaceflight. The descent and landing sensors are located underneath the BalLunar module and consist of lidar and radar systems which need an unobstructed view of the lunar surface; thus, the ACS must be capable of reaching any attitude withing 30º of nadir. To obtain proper thrust vector placement for each orbital maneuver, the BalLunar module must be capable of performing a 180º rotation about any of its axes, mitigating unforeseen situations that lead to a misaligned thrust vector. The attitude determination sensors place constraints on the rotational rates according to their operating ranges, thus rotational rates must be maintained under 3º/s. The BalLunar modules require accurate attitude knowledge and control relative to an inertial reference frame, thus each module must have a pointing accuracy of 0.1º [8]. An undesired state that exceeds the target attitude may occur as a result of attitude reorientation maneuvers, thus to avoid excess attitude and the unnecessary expenditure of propellant, each module must have a maximum angular rotation overshoot of 10% during target acquisition or upset recovery maneuvers. The requirements for the BalLunar attitude control system are specified in *Table 4.2-2*.

Table 4.2-2: Attitude Control System Requirements

Requirement Statements
The spacecraft shall be capable of obtaining any attitude within 30º of nadir
The spacecraft shall maintain rotational rates < 3º/s for attitude acquisition
The spacecraft shall be capable of performing 180º rotation about any S/C axis for each pre-determined propulsive maneuver
The spacecraft shall have a pointing accuracy of 0.1º
The spacecraft shall have a maximum angular rotation overshoot of 10% during target acquisition or upset recovery

4.2.1 Attitude Determination Sensors

A suite of sun sensors, star trackers, and inertial measurement units (IMUs) was chosen for attitude and rate determination. To perform full three-axis attitude determination, a minimum of two external, non-parallel vector measurements are required for inertial sensing [8]. Sun sensors measure the angle between their mounting point on the spacecraft and incident sunlight and are generally characterized by their accuracy and reliability. Star trackers use a charge-coupled device to image a star field which is internally processed by an onboard computer that determines three-axis attitude based on a pre-loaded star catalog. Inertial measurement units measure linear and angular motion measurements using a combination of accelerometers and gyroscopes, respectively. The Solar MEMS D60 sun sensor, and the Terma T2 star trackers were selected due to their high accuracy, low mass, low power consumption, flight heritage, and radiation hardened components. The selected IMU is the Honeywell HG1700SG, consisting of three ring laser gyroscopes, and three quartz resonating beam accelerometers, which have a large gyroscope and accelerometer operating range and has an extensive flight heritage. The low mass and power consumption of the selected sensors allowed for a configuration of 4 star-trackers, 6 sun sensors, and 2 IMUs, providing redundant sensors of each type.

Table 4.2.1-1: Inertial Attitude and Rate Sensor Performance

Item	Quantity	Mass/unit (kg)	Power/unit (W)	FOV (deg)	Accuracy (deg)	Supplier / Model
Star Tracker	4	0.76	3	20 Circular	0.0058	Terma / T2
Sun Sensor	6	0.1	1	120 Circular	< 0.06	Solar MEMS /D60
Item	**Quantity**	**Mass/unit (kg)**	**Power/unit (W)**	**Gyroscope Op. Range**	**Accelerometer Op. Range**	**Supplier / Model**
IMU	2	0.7	5	±358 deg/sec	±12 g	Honeywell / HG1700SG

4.2.2 Descent and Landing Sensors

The Lunar BOWIE consists of five separate BalLunar modules joined via autonomous rendezvous and docking on the lunar surface. To successfully perform the rendezvous, the modules must land within close proximity of each other, requiring a suite of guidance, navigation, and control (GN&C) technologies that enable high precision landing. NASA's Safe and Precise Landing – Integrated Capabilities Evolution, known as SPLICE, is a multi-directorate, multi-center project, part of NASA's Precision Landing and Hazard Avoidance (PL&HA) domain that enables landing at locations that pose a high risk for lander missions [9]. SPLICE is developed for high landing accuracy via sensors and algorithms that significantly reduce navigation errors while performing accurate hazard detection and avoidance.

The SPLICE system consists of a Navigation Doppler Lidar (NDL), Hazard Detection Lidar (HDL), Terrain Relative Navigation (TRN) system, and Descent and Landing Computer (DLC). The NDL performs measurements along three different laser beams, providing range measurements beginning at a line of sight (LOS) range of 7.3 km, and velocity measurements beginning at a LOS velocity of 200 m/s, with LOS errors of 2.2 m and 1.7 cm/s, respectively [10]. The NDL uses trajectory and navigation filter algorithms to support precision landing. The HDL is a hybrid scanning-imaging lidar that performs surface imaging and generates Digital Element Maps (DEM) in 2 seconds with a cm-level precision at its nominal operational altitude of 500 m, with altimetry starting at 10 km with less than 10 cm precision [10]. The HDL uses hazard detection algorithms based on those previously developed for the ALHAT (Autonomous precision Landing and Hazard Avoidance Technology) project, modified for the new HD Lidar DEMs. The TRN system uses a camera to generate real-time, precise maps of the lunar surface which are superimposed over preloaded onboard maps, which are then used in conjunction with TRN algorithms. The DLC is designed to house NASA's High-Performance Spaceflight Computing (HPSC) processor, which in configured to process SPLICE sensor data and algorithms via a Field Programmable Gate Array (FPGA) [10].

The use of lidar in the SPLICE system proves to be problematic for the end portion of the BalLunar module's landing phase; lidar is highly susceptible to lunar dust raised by surface-engine plume interactions. In addition to the SPLICE system, the modules are equipped with JPL's Terminal Descent Sensor (TDS) as a redundant descent and landing sensor suite that uses Ka-band radar which has little susceptibility to lunar dust and engine plumes. The TDS has flight heritage on the Mars Science Laboratory (MSL) and uses 36 GHz, Ka-band, pulse-doppler radar consisting of a 6 beam configuration with 3º antenna beam widths, having 3 beams canted 20º off nadir, 2 beams canted 50º off nadir, and 1 nadir beam. Altogether, the beams provide range measurements beginning at a max LOS range of 3.5 km with an error of 2% the instantaneous slant range, and velocity measurements at a max LOS velocity of 200 m/s with an error equivalent to 0.75% the instantaneous velocity plus 0.2 m/s [11]. The SPLICE system and TDS provide high-accuracy, redundant systems for guidance, navigation, and control, allowing multiple modules to land within close proximity of each other to form the Lunar BOWIE.

4.2.3 Attitude Control Actuators

The BalLunar module is a lander type spacecraft that must maintain control authority throughout all phases of spaceflight. To provide the module with the most appropriate hardware for attitude control and stabilization, the following methods were evaluated: spin, dual spin, three-axis, momentum bias, and gravity gradient stabilization. High maneuverability and accuracy are essential for landing a spacecraft; therefore, the gravity gradient, spin, dual spin, and momentum bias stabilization methods were discarded from consideration. The BalLunar module performs hazard avoidance maneuvers during the landing phase, requiring rapid and reliable maneuverability; the mechanically limited lifetime, proneness to failure, and relatively low speed of reaction wheels and control moment gyros exclude them from consideration [12]. This led to the selection of the three-axis attitude control and stabilization method consisting of reaction control thrusters.

The reaction control system (RCS) must be capable of counteracting disturbance torques, maintaining pointing requirements, and performing slew maneuvers with varying slew rates for each phase of spaceflight. To reduce the mass and complexity of the system, the RCS was designed to minimize the number of thrusters while providing sufficient redundancy for malfunctions. The thrusters were sized and configured in accordance with the

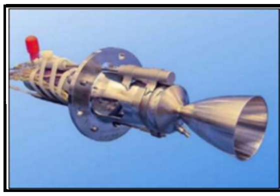
BalLunar module's moments of inertia (MOI) and geometry. The thrusters must display a thrust range capable of performing large and quick slew maneuvers during descent and landing mode, while not exceeding maximum slew rate and overshoot requirements during translunar cruise and slew modes. They must also have a low enough minimum impulse bit (MIB) to provide small control torques for fine attitude control. A trade study was performed based on the MIB, total impulse, number of impulses, and technology readiness level (TRL) as the figures of merit, shown in *Table 4.2.3-1*. The minimum impulse bit determines fine control capabilities and fuel consumption minimization, the specific impulse quantifies the efficiency in terms of impulse provided per unit mass of propellent, the number of pulses relates to the thruster operational lifetime in terms of on-off cycles, and TRL assesses the thruster technology maturity.

Table 4.2.3-1: RCS Thruster Trade Study

FOMs / Alternative Thrusters	T0.0-20 Minimum Impulse Bit WF = 3		T0.0-21 Total Impulse WF = 2		FT0.0-20 Number of Pulses WF = 1		M0.0-2 TRL WF = 2		**Weighted Total**
	U	W	U	W	U	W	U	W	
Moog Monarc-22-12	3	9	9	18	3	3	9	18	**48**
Aerojet Rocketdyne MR 106-L 22N	9	27	3	6	3	3	9	18	**57**
Bradford ECAPS 22N HPGP	1	3	1	2	1	1	1	2	**8**

The Aerojet Rocketdyne MR 106-L 22N hydrazine monopropellant thruster was chosen as a result of the trade study; its performance characteristics are shown in *Table 4.2.3-2*. Several configurations were evaluated, varying in number of thrusters and structural configuration. A 16-thruster configuration was implemented, consisting of four clusters of four thrusters each, separated by 90º about the major axis of the spacecraft. The chosen configuration minimizes fuel consumption and allows full three-axis maneuverability with up to any three thrusters malfunctioning, providing a level of redundancy of 3 [13]. To prevent impingement by the thruster plumes on the spacecraft, the thrusters were canted by 17º. The thruster configuration, thruster plume cones, and ACS sensor fields of view (FOV) plots are shown in *Figure 4.2.3-1*, with thruster plumes shown in orange, sun sensor FOVs in green, and star tracker FOVs in blue. All spacecraft surfaces and sensor fields of view are clear from plume impingement.

Table 4.2.3-2: Aerojet Rocketdyne MR 106-L-22N Thruster Performance Characteristics

	Mass (kg)	Valve Power (W)	Thrust (N)	Feed Pressure (Bar)	I_{sp} (sec)	MIB (N-sec)	Total Impulse (N-sec)	# Pulses
	0.59	25.1	10 - 34	5.9 - 27.6	228 - 235	0.015	561,388	120,511

To ensure the selected thrusters were within the appropriate thrust range, their performance was evaluated for 180º and 30º rotations about each spacecraft axis via response curves for angular position, velocity, and acceleration vs time. It can be seen from *Figure 4.2.3-2* and *Figure 4.2.3-3* that the selected thrusters perform the maneuvers with rotational rates under 2º/s while acquiring the desired attitude with less than 2% overshoot for each axis, meeting the derived requirements detailed in *Table 4.2-2*.

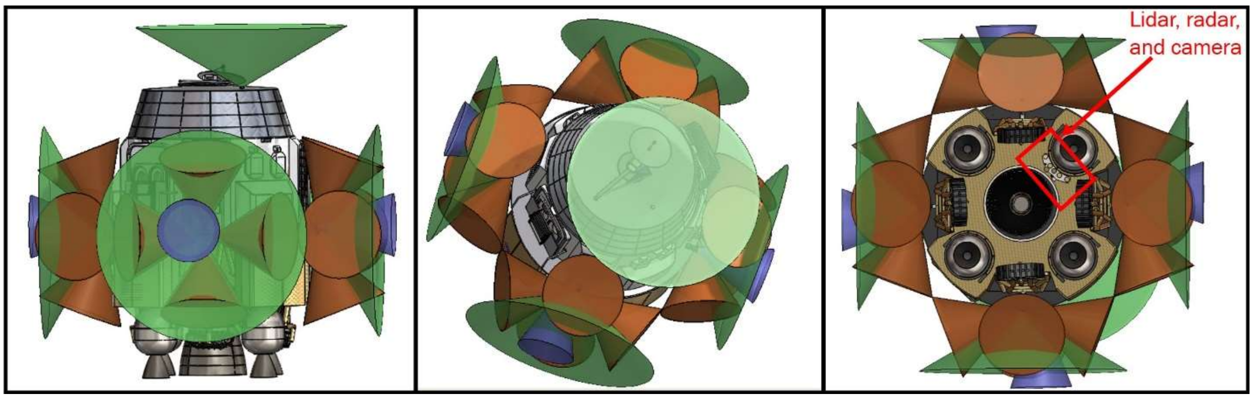


Figure 4.2.3-1: Thruster Plume Cones and ACS Sensor FOV plots

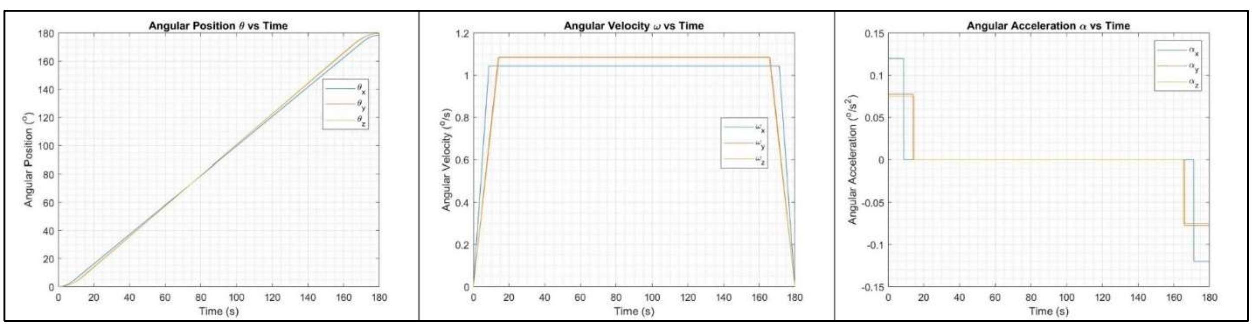


Figure 4.2.3-2: Response Curves for 180º Rotation About Each Spacecraft Axis

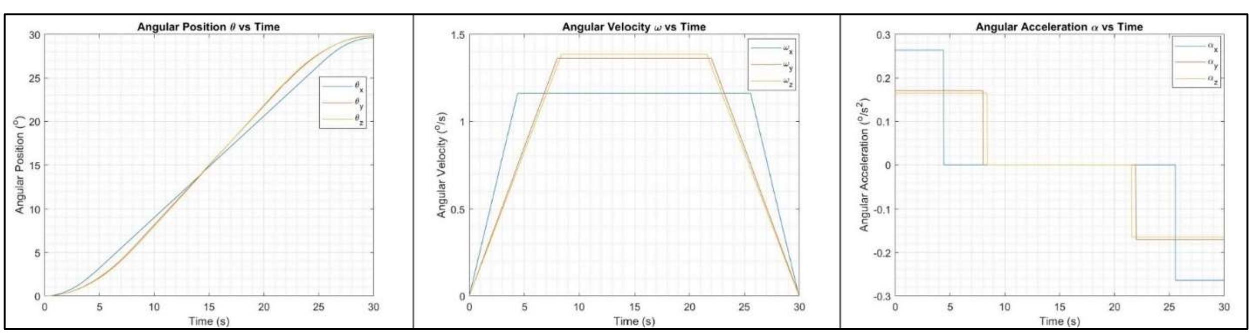


Figure 4.2.3-3: Response Curves for 30º Rotation About Each Spacecraft Axis

Once separated from the launch vehicle, the spacecraft will encounter environmental disturbance torques which need to be counteracted to maintain stability. The spacecraft will encounter environmental torques of varying type and magnitude depending on the spaceflight phase. Once in orbit about Earth, aerodynamic drag and magnetic torques will be experienced in orbits under 500 km and ranging from 500 to 35,000 km, respectively. Once an altitude above 35,000 km is reached and in low lunar orbit, the aforementioned torques can be neglected since the Moon lacks an atmosphere and a substantial magnetic field. Gravity gradient torques need to be accounted for in both LEO and LLO due to the exponentially higher gravity force experienced by the extremities located further away from the gravitational primary. A resulting torque from solar radiation pressure is experienced for each moment a photon strikes the surface of the spacecraft, which corresponds to the entire spaceflight phase [8]. Worst-case values of torques, the resulting angular momentum transferred to the spacecraft, and propellant required to counteract the disturbance torques are shown in *Table 4.2.3-3.* Worst-case disturbance torque calculations are based on the geometry of the spacecraft and its largest dimensions, assuming the spacecraft center of mass was be offset by a distance equal to 10% the largest area.

Table 4.2.3-3: Worst-Case Environmental Disturbance Torques

Disturbance Environment	Earth Environment Torque (N-m)	Earth Environment Angular Momentum (N-m-s)	Moon Environment Torque (N-m)	Moon Environment Angular Momentum (N-m-s)
Solar Pressure	8.06E-05	28.95	8.06E-05	28.95
Aerodynamic	1.87E-05	0.10	0	0
Gravity Gradient	8.12E-03	43.12	4.55E-03	32.15
Magnetic	4.29E-04	2.28	0	0
Total Propellant Mass for Disturbance Torques (kg)				0.021

In total, the ACS equipment listing is composed of the RCS thrusters, inertial attitude and rate sensors, descent and landing sensors, and propellant required for all ACS maneuvers with their corresponding margins. The equipment mass and power breakdown is shown below in *Table 4.2.3-4*.

Table 4.2.3-4: ACS Equipment Listing

Item	Quantity	Mass (kg)	Power (W)
Thrusters	16	9.44	401.6
Star Tracker	4	3.04	12
Sun Sensor	6	0.6	2.4
IMU	2	1.4	10
SPLICE (NDL)	1	15	85
SPLICE (HDL)	1	N/A	N/A
SPLICE (DLC)	1	N/A	N/A
TRN Camera	2	0.512	5
TDS	1	25	120
Propellant Mass		33	
Total		**87.99**	**636**

4.3 Thermal

Table 4.3-1 Thermal Control System Requirements Statements

Requirement Statement
The Thermal Control System shall maintain interior temperatures within the range of 291-299K (65-80°F) while inhabited.
The Thermal Control System shall be able to reject a minimum of 340W while inhabited
The Thermal Control System shall maintain minimum operating temperatures for during eclipse times

4.3.1 Worst Case Temperature Study

Considering the Lunar BOWIE will not be inhabited during any eclipse periods we will provide simple heaters for maintaining vital components during the worst-case cold conditions. The crew will be present for the worst-case hot conditions in which we shall maintain the required internal temperature range of 291-299K (65-80°F) for crew comfort. A study was performed on the module's solid structure to determine our thermal material thicknesses and the best internal equilibrium temperatures that could be achieved with passive thermal control techniques [14].

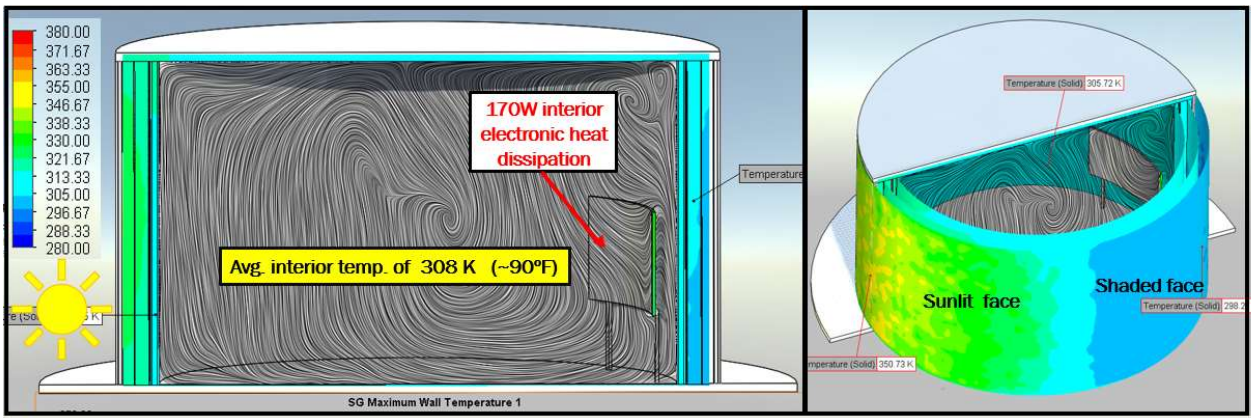


Figure 4.3.1-1 Worst-case equilibrium temperatures for solid structure

The study uses worst case solar irradiance at the lunar South Pole as well as End Of Life (EOL) parameters for the materials being tested which we detail in the materials section of this report. Included in the heat transfer simulation shown in *Figure 4.3.1-1,* is conduction through solids, convection in the interior (air), solar radiation, and a 170W internal heat dissipation source that represents the estimated peak power dissipation from internal components. This resulted in an average interior equilibrium temperature of 308K (90°F), so it was determined an active thermal control system would need to be designed to be able to bring the internal temperature into the required comfort range.

4.3.2 Active Thermal Control System Design

After determining that a minimum of 340W would need to be rejected from the structure to bring the internal temperatures to the mid-range of the thermal requirements, it was determined the minimum radiative area (1.5 m^2) required to dissipate the heat required. Multiple designs were compared which included deployable and/or fixed radiators with multiple placement options on the spacecraft. We selected a design that consists of four total radiators like the ones shown in *Figure 4.3.2-1* which are dispersed equally around the circumference of the spacecraft, guaranteeing the minimum area required for heat dissipation would always be located within the the shaded side of the module. This allows for the simplest design while providing reliable heat dissipation with a healthy margin provided for possible higher internal heat dissipation of certain scientific payloads.

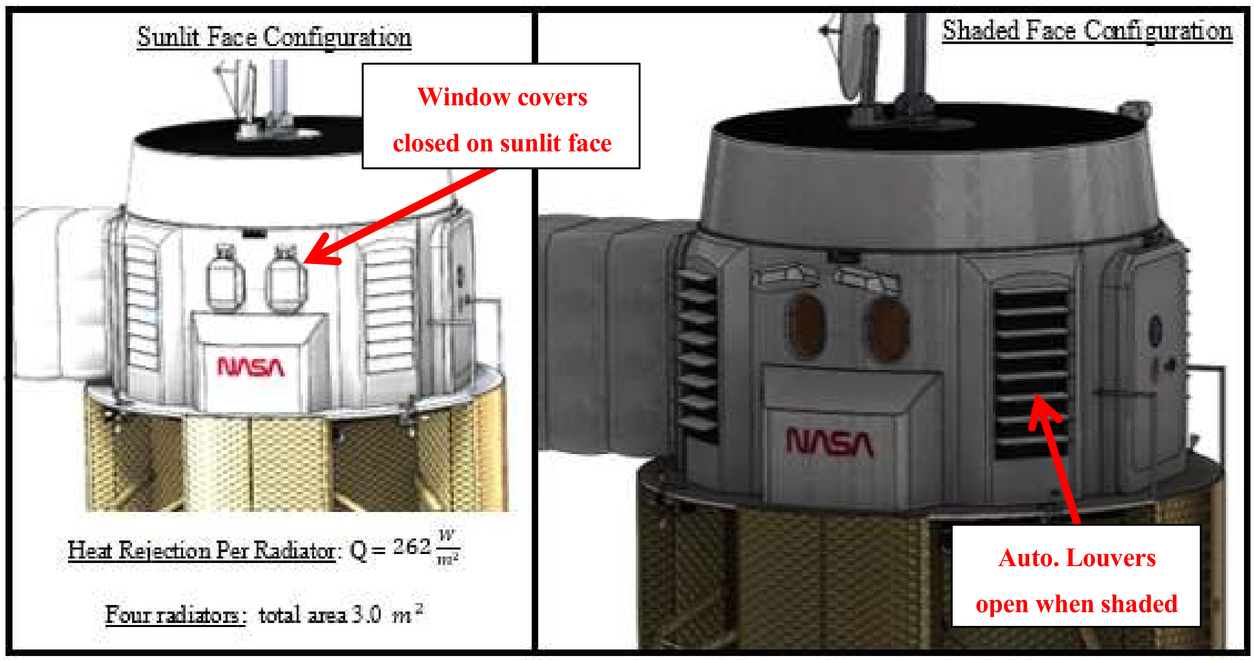


Figure 4.3.2-1 Active Thermal Control System – Sunlit and Shaded configurations

The issue of having some radiators exposed to direct sunlight has led to implementing an automatically actuated louvered design shown in *Figure 4.3.2-1* which will close to cover and insulate the radiators, preventing overheating when exposed on the sunlit side of the module. This prevents the system from absorbing unwanted heat into the system and reducing its heat rejection efficiency. Also because of the very low sun angles experienced at the poles it was determined necessary to provide thermal covering for the windows as seen in *Figure 4.3.2-1,* which when located on the sunlit face these covers can be closed to prevent the additional heat source from overheating the interior.

4.3.3 Solar Array Efficiency

For the purposes of determining what kind of efficiency we should be expecting from our main power arrays a study was conducted to determine their worst-case hot equilibrium surface temperatures. Again, using the worst-case solar irradiance expected at our target location as well as EOL parameters for the materials of these MegaFlex arrays.

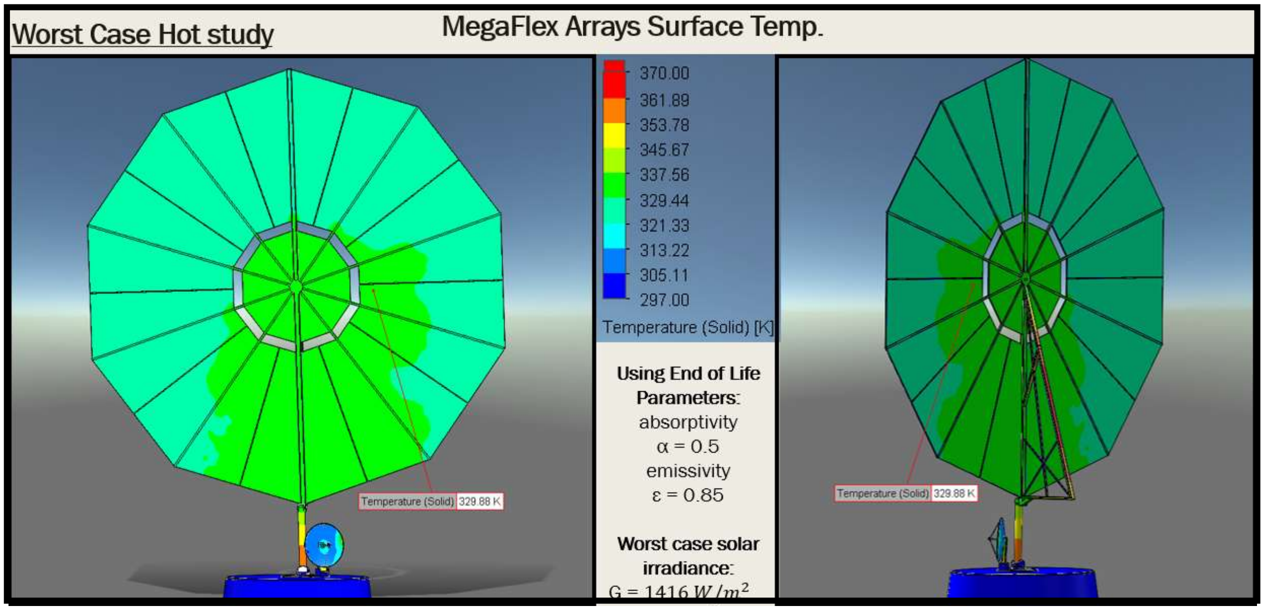


Figure 4.3.3-1 Equilibrium Surface temperatures for MegaFlex arrays

The simulation resulted in an equilibrium surface temperature of 330K (130°F) (*Figure 4.3.3-1*). At these higher temperatures we can expect at least a 10% decrease in efficiency. Reducing our power budget slightly from the maximum 40-kW power production capabilities of only these two arrays together down to 36-kW. We have also applied the same 10% decrease in power to our 2 m diameter supplemental solar arrays which are located on the VIBES and Living Modules and are detailed in the Power section of the report.

4.4 Structures

Table 4.4-1 Structural Requirements Statements

Requirement Statements
The design shall show positive MS against ultimate loads using a FS of 2.6
The LBC shall provide appropriate shielding to limit radiation to 1,000 millisieverts (mSv) for the duration of the mission
The LBC shall ensure that the Risk of Exposure Induced Death does not exceed 3%.

4.4.1 Structural, Thermal, and Radiation Protection Materials

This section refers mostly to the materials used within the solid structure of the modules. We have designed a multiple layer shielding system within the exterior wall that consists of five total layers as shown in *Figure 4.4.1-1* included below. Each layer being comprised of a different material and designed for different specific purposes. In *Figure 4.4.1-1* a cross section of the actual shielding within the module shows the spacing we have allowed to account for the expected thermal expansion and contraction of these materials which for longevity purposes must be accounted for when exposed to such a large range of temperature changes.

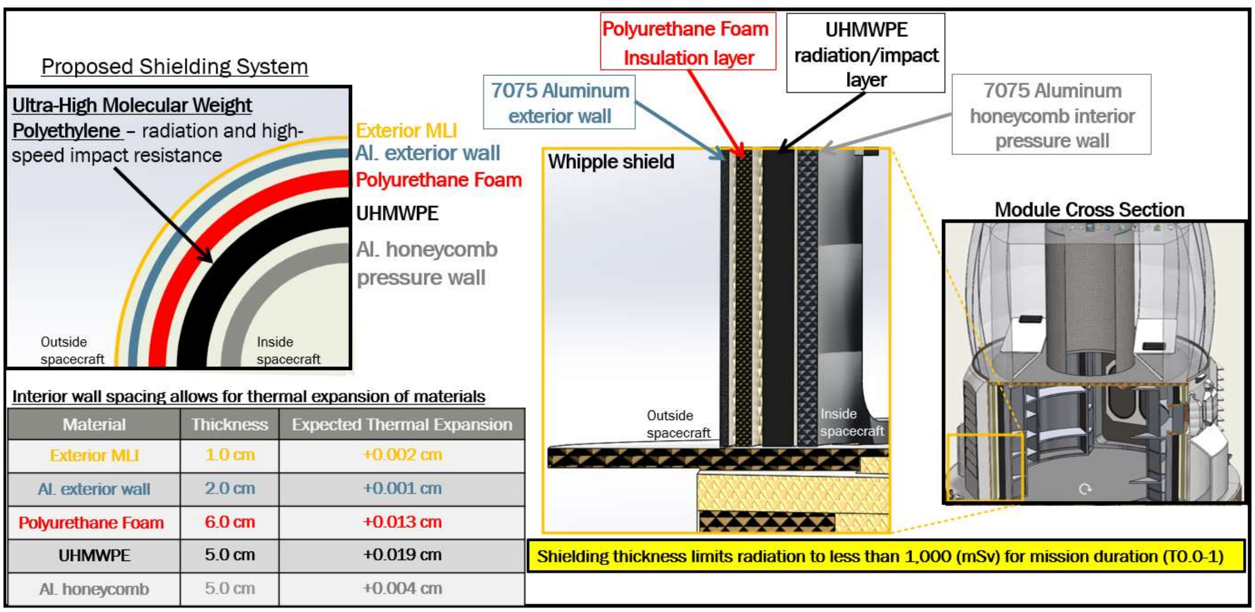


Figure 4.4.1-1 Materials within solid structure shielding

The outer-most layer is Multi-Layered Insulation (MLI) consisting of Aluminized Polyester and Kapton thin sheets. The MLI is only included on the lower half of the solid structure which is housing our cryogenic propellants where the extra insulation was warranted. It was determined by our thermal analysis that the complexity of covering the upper first level of the modules with MLI was not necessary considering an active thermal control system was needed regardless. The third layer from within is also for thermal insulation the material being Polyurethane Foam at 6.0 cm thickness which we determined through thermal analysis that even at this maximum thickness we could provide the interior temperatures were still not meeting the requirements range.

Within the shielding are two different 7075 Aluminum layers that provide structural strength as well as protection from the elements. The outer-most layer being 2.0 cm in thickness it contributes to resisting the axial loads on the spacecraft though once on the surface its main function is for high speed projectile protection and the first line of defense against radiation [15]. We chose 7075 Al. because of its excellent strength to weight ratio, the tradeoff of 7075 Al. is the manufacturing complexity that comes with losing weldability. Though considering our then two-billion-dollar budget margin versus our slim mass margins it was determined to not be as significant as the need for minimizing our mass. The inner-most layer is also a 7075 Al. though it exists as a honeycomb core sandwiched between two thinner sheets. This layer being responsible for resisting the ΔP loads of the pressurized interior.

The last layer to mention is Ultra High Molecular Weight Polyethylene (UHMWPE) this layer serves two purposes first for radiation shielding and second for high speed impact resistance which explains its considerable thickness of 5.0 cm. As opposed to using Kevlar layering for high speed impact protection the UHMWPE can provide equivalent dual-purpose protection at only a 25% increase in mass. The NASA standard for radiation protection comes from NASA-STD-3001 which states in section 4.2.10.1 "Planned career exposure to ionizing radiation shall not exceed 3 percent Risk of Exposure-Induced Death (REID) for cancer mortality at a 95 percent confidence level." [16, pp 21-22) A study into using UHMWPE for whipple shielding protection [17] found that it has the same radiation protection as standard Polyethylene (PE). To meet the NASA requirement while minimizing the mass, 5 g/cm^2 area density was chosen for the UHMWPE. A study of using PE as radiation shielding found that at that density, the REID level is at 2.25% [18]. While higher density coats provide higher protection, we are complying with the concept of As Low As Reasonably Achievable (ALARA), such that the mass requirement of the system does not enable a massive radiation protection system. This coating is found in every module, meaning every module has radiation protection from the normal space environment. The mass required for this is 1040 kg per module and is attributed under the mass subsystem in the mass breakdown. However, the modules do not contain a bunker for use in extreme radiation conditions. To ensure that the crew can survive a large radiation event, the following options can be utilized for safety:

- The crew embarks into their suits, and then returns to the interior of the modules while suited. This improves the radiation shielding as it utilizes both suit and base radiation shielding.
- If the landing capsule for the crew contains a higher level of radiation shielding, a similar strategy to can be implemented by the crew to embark and stay inside the capsule while suited for protection.
- If the storm is too great for either of these options, the crew may use the landing capsule to launch to join the Lunar Gateway system in orbit and stay there for protection.

4.4.2 LVA Design

We compared multiple designs and materials using the maximum design loads and conditions outlined by SpaceX within their Falcon Payload User's Guide. Resulting in a selection of a Titanium (Ti13V-llCr-3A1) truss design which provided the best performance to overall mass [19]. The FEA of the three load conditions investigated for this selected design are the maximum dynamic pressure (Max-Q), main engine cut-off (MECO), and the maximum lateral loads condition. We are highlighting the maximum lateral load condition here which is where we determined this LVA design is experiencing its maximum loads.

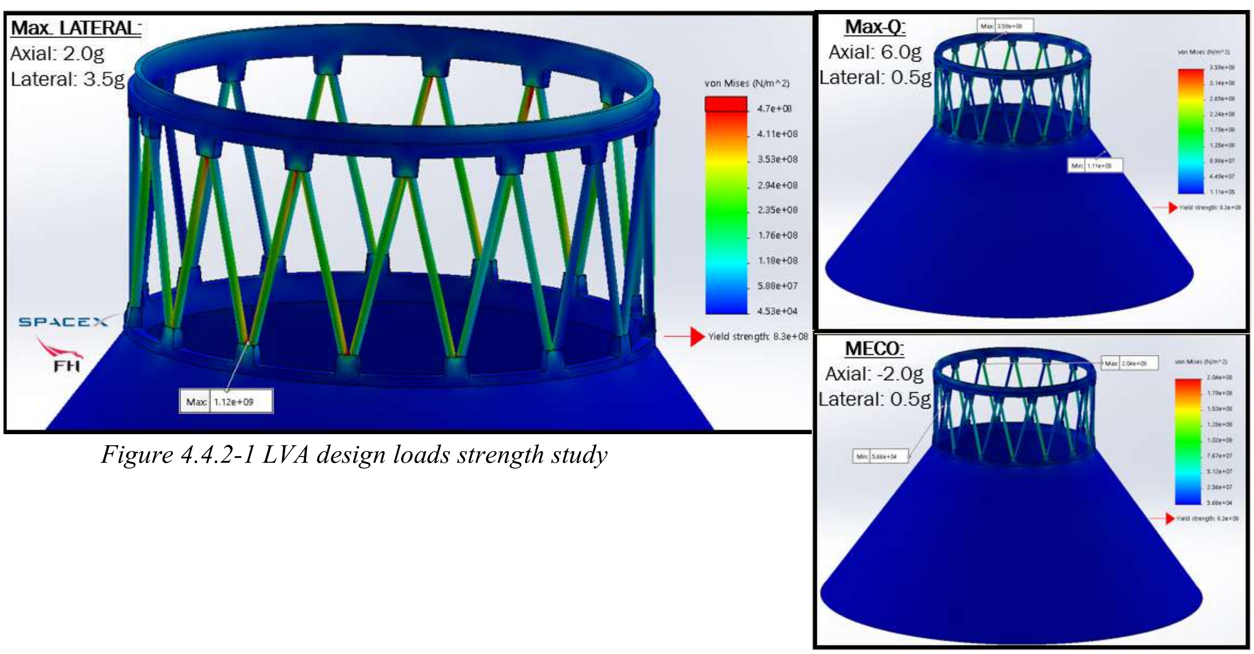


Figure 4.4.2-1 LVA design loads strength study

Table 4.4.2-1 Margins of Safety for LVA design loads conditions

LVA Material: Titanium (Ti13V-IICr-3A1)		
Event	M.S. Yield $F_{ty} = 8.30e{+}8\ (N/m^2)$	M.S. Ultimate $F_{tu} = 1.172e{+}9\ (N/m^2)$
Max. Lateral	-0.259	0.0464
Max-Q	1.31	2.26
MECO	3.07	4.75

We optimized this design by adjusting the material, truss layout, and sizing until we showed just barely positive Margins of Safety against ultimate, which can be seen in *Table 4.4.2-1* shown above along with the critical location of where the stress is peaking up to its maximum in between the struts of the Titanium truss members.

4.4.3 Hydraulic Landing Gear Design

A study of the two maximum loads cases were performed to evaluate the strength of this design for landing and its autonomous transit phase across the lunar surface for rendezvous. First was the maximum ultimate static load to be experienced, followed by combining that maximum static load with our estimated maximum lateral load condition to simulate the landing leg encountering an obstacle in transit such as a large rock that induces a large bending moment from the lateral force experienced at the end of the leg.

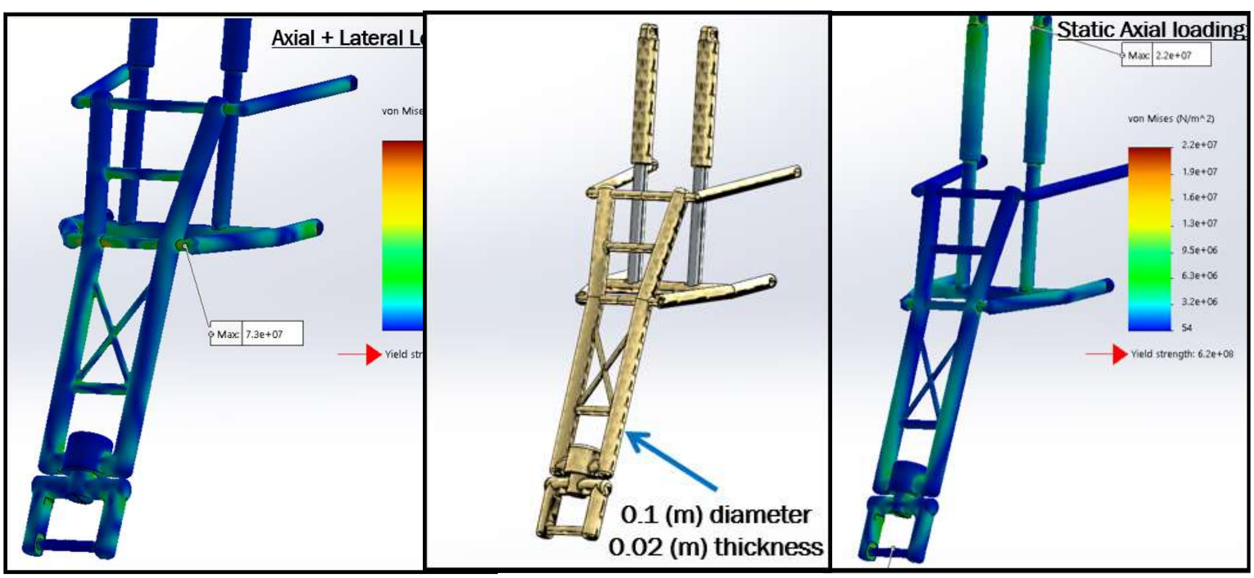


Figure 4.4.3-2 Hydraulic Landing Gear maximum axial and lateral design loads

Table 4.4.3-2 Margins of Safety against Ultimate for Axial and Axial+Lateral load conditions

Material: AISI 4340 Steel, normalized	
Static Loading	M.S. Ultimate $F_{tu} = 1.045e+9 \ (N/m^2)$
Axial	16.5
Axial + Lateral	4.3

In this study we chose to use AISI 4340 Steel which is a very dense steel alloy commonly seen used in aircraft landing gear. We used a Factor of Safety of 2.6 times our limit loads to determine the ultimate loads used. We are showing high Margins of Safety as you can see in *Table 4.4.2-2* above, even for the combined load condition. So, for further optimizing the design some drop test simulations will be performed to evaluate what worst case hard landing [20] can be survived before any reduction in material or simplifications to the design will be made.

4.4.4 Semi-Rigid Docking Bridge Displacement

To confirm our Semi-Rigid Docking Bridge could successfully dock to another module we investigated the expected maximum deflection we should expect from the design as it deploys on the lunar surface. To give the docking operations a slight margin of error we designed the docking adapters as tapered male to female connections allowing a for anything less than a 5° offset would be corrected by the tapered adapters when engaged.

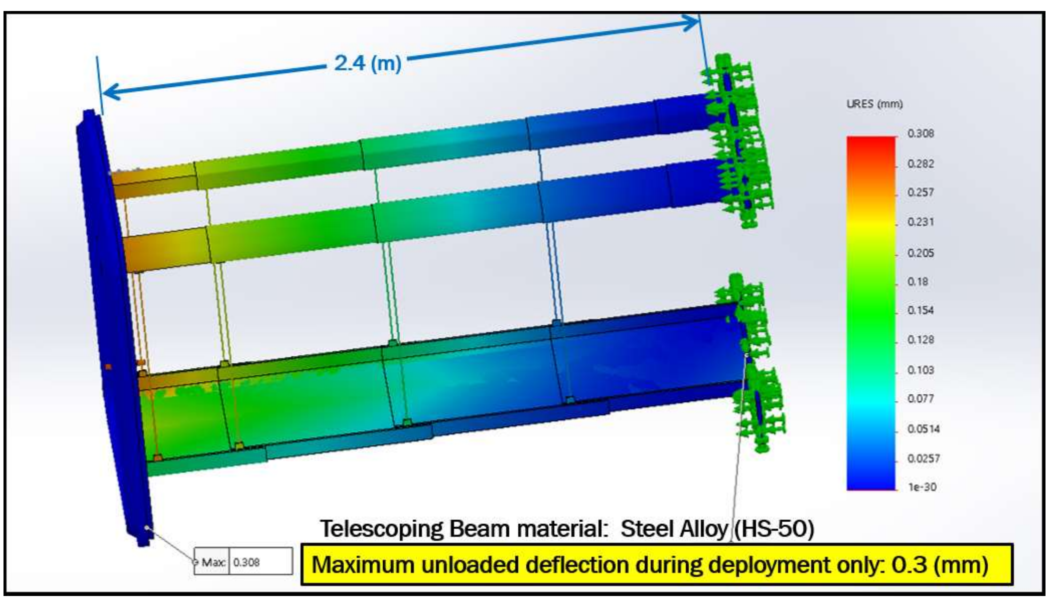


Figure 4.4.4-1 – Semi-Rigid Docking Bridge maximum deflection study

This study uses HS-50 Steel for the Telescoping beam's material along with lunar gravity and no external applied loads to simulate the maximum deflection of this design while experiencing its maximum bending moment due to its own weight during deployment. The maximum deflection of the beams shown in *Figure 4.4.4-1* were less than 1 mm which should be expected since the design was sized for withstanding a minimum of two larger sized astronauts laden with gear.

4.5 Telecommunication

Table 4.5-1 Telecommunication System Requirements

Requirement Statement
LBC and supporting systems shall be capable of 24 hours per day of communication with Earth.
The LBC's telecommunications system shall contain a minimum bandwidth of the system of 400 Mb/s to allow for video communication of the crew with Earth.

The telecommunication subsystem is driven by the main requirement of providing constant communication. Being a human spaceflight mission, keeping the astronauts alive is top priority. The Lunar BOWIE's communication subsystem is designed to meet this requirement by utilizing multiple ground stations and two satellite orbiters.

4.5.1 Ground Station

The Lunar BOWIE will be using the multiple ground stations from the Near-Earth Network (NEN) [21]. NEN is comprised of numerous ground stations located around the world that supports satellites in multiple orbits, including lunar orbit. Therefore, at any given time, the Lunar BOWIE will be able to communicate with at least one ground station.

4.5.2 Satellites

Due to our location being in the lunar South Pole, the line of sight with Earth is limited. In order to meet the communication requirement, the Lunar BOWIE is designed to be supported by two orbiting satellites. The main orbiter will be the Lunar Gateway, which is expected to be operational by 2026, using its Near-Rectilinear Halo Orbit [22]. This orbit can constantly provide coverage the back side of the Moon, but it will not be enough to cover the South Pole. Therefore, an additional satellite will be included to account for the lunar Gateway's orbital period.

4.5.3 Transit Communication

Launching from Cape Canaveral, the orbiting satellites will be assisting the modular launches during transit. The satellites will provide support through navigation by performing system checks prior to the Lunar Orbit Insertion.

This will prevent the occurrence of any navigational mishaps due to lack of communication, ensuring the modules will arrive at the targeted location safely.

4.5.4 Lunar Relay System

The communication relay system is inspired from the NASA SCAWG Report [23] and will be utilizing two radio frequencies, S-band and Ka-band. As shown in *Figure 4.5.4-1*, S-band will be used for emergency links and TT&C (telemetry tracking and command), and Ka-band will be used for relaying mission data from the lunar surface to the orbiters and then back to Earth. Additionally, the capability to communicate directly from the Moon back to earth is implemented in any case of orbiter malfunctions. We designed this relay system towards mission expendability as the orbiters will remain in orbit after the 45-day mission requirement. In addition to supporting the Lunar BOWIE mission, the relay system will also be capable of supporting future deep space missions.

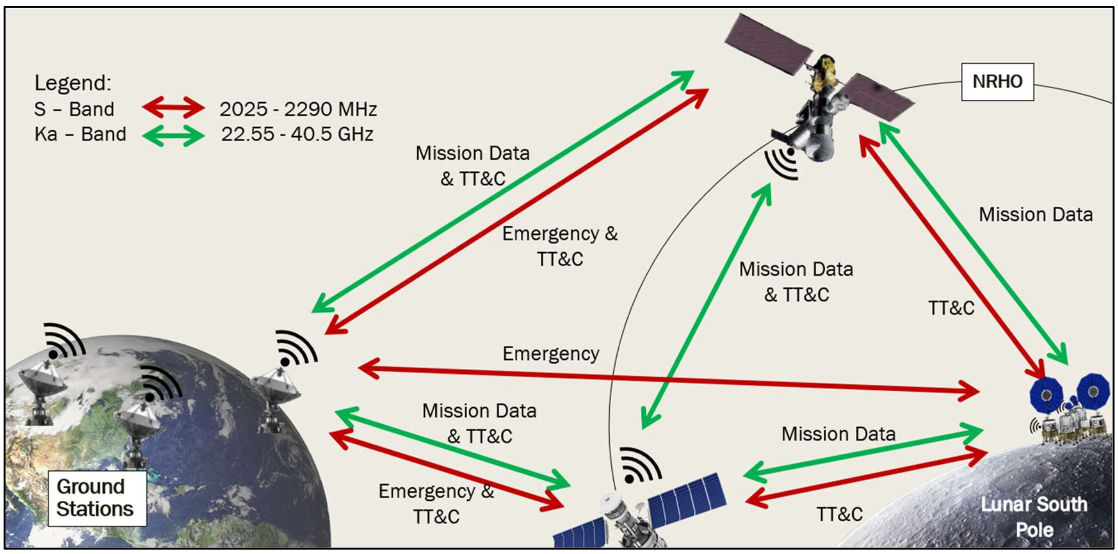


Figure 4.5.4-1 Communication Relay System

4.5.5 Surface Communication

Once all the modules land on the lunar surface, each module will use their individual 1-meter antenna in order to rendezvous with each other. Currently, the Lunar BOWIE is assumed to be using the new generation xEMU's for the upcoming Artemis mission [24]. The space suit is assumed to use very high frequency (VHF) to communicate with one another and back to the modules. Mission data collected by the EVAs will be relayed back to the module and then sent to the orbiters.

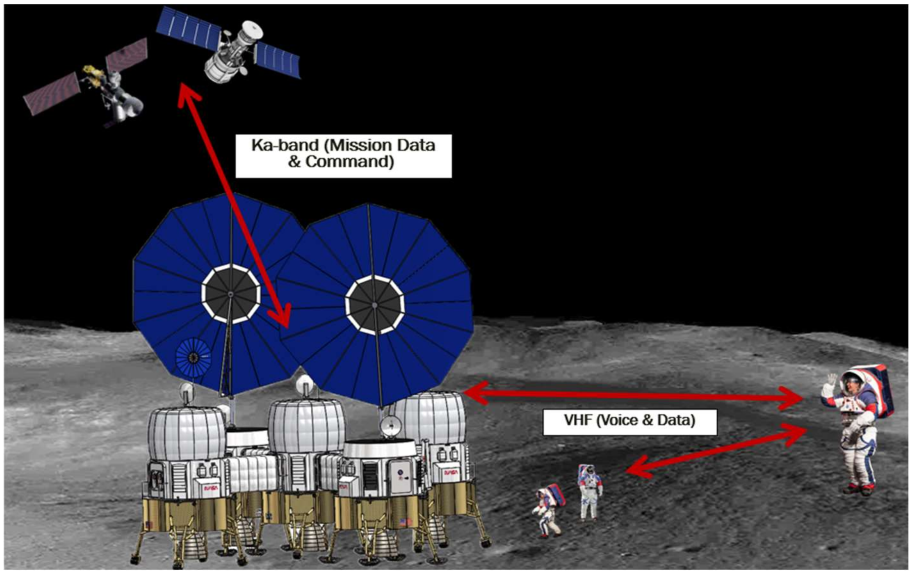


Figure 4.5.5-1 Surface Communications

4.5.6 System Data Rates

The system data rates are calculated for the entire communication relay system. As shown in the *Table 4.5.6-1*, both the uplinks and downlinks achieve high data link margins of over 10dB.

Table 4.5.6-1 Communication Data Rates

Downlink from Moon to Satellite	**Apoapsis**		**Periapsis**	
Frequency	2.22	GHz	2.23	GHz
Range	70000	km	1500	km
Eb/N0 Achieved	20	dB	24	dB
Eb/N0 Required	5	dB	5	dB
Data Link Margin	15	dB	19	dB
Uplink from Moon to Satellite	**Apoapsis**		**Periapsis**	
Frequency	27	GHz	27	GHz
Range	418000	km	1500	km
Eb/N0 Achieved	22	dB	31	dB
Eb/N0 Required	10	dB	10	dB
Data Link Margin	12	dB	21	dB
Max Range	**Uplink Earth to Satellite**		**Downlink Earth to Satellite**	
Frequency	40	GHz	38	GHz
Range	418000	km	418000	km
Eb/N0 Achieved	22	dB	23	dB
Eb/N0 Required	10	dB	10	dB
Data Link Margin	12	dB	13	dB
Emergency Uplink	**from Satellite to Moon**		**from Earth to Moon**	
Frequency	2.21	GHz	2.25	GHz
Range	70000	km	418000	km
Eb/N0 Achieved	23	dB	20	dB
Eb/N0 Required	5	dB	10	dB
Data Link Margin	18	dB	10	dB

4.5.7 Hardware List

Altogether the telecommunication subsystem is comprised of antennas, transmitters, transponders, transceivers, and cabling. The equipment listing has a total mass of 81.2 kg and total power of 136 Watts per module.

Table 4.5.7-1 Telecommunication Equipment list

Item	Quantity	Mass (kg)	Power (W)
Antennas	2	60	110
Transmitters	2	5	10
Transponders	2	4.2	10
Transceiver	2	2	6
Cabling	--	10	--
Total		**81.2**	**136**

4.6 Command and Data Handling

The Command and Data Handling (C&DH) system manages data transmitted to and from each module. It also controls and processes information from the module's subsystems. The primary system used in this design is the Modular Unified Space Technology Avionics for Next Generation (MUSTANG) from NASA's Goddard Space Flight Center (GSFC). In addition to C&DH, the MUSTANG also includes the module's Power System Electronics (PSE) to control loads from the electrical power to the module throughout mission phases. MUSTANG is a modular construction of different NASA boards [25]. The boards included for this mission can been seen in the system breakdown in *Figure 4.5-1*, under either CDH or PSE MUSTANG systems.

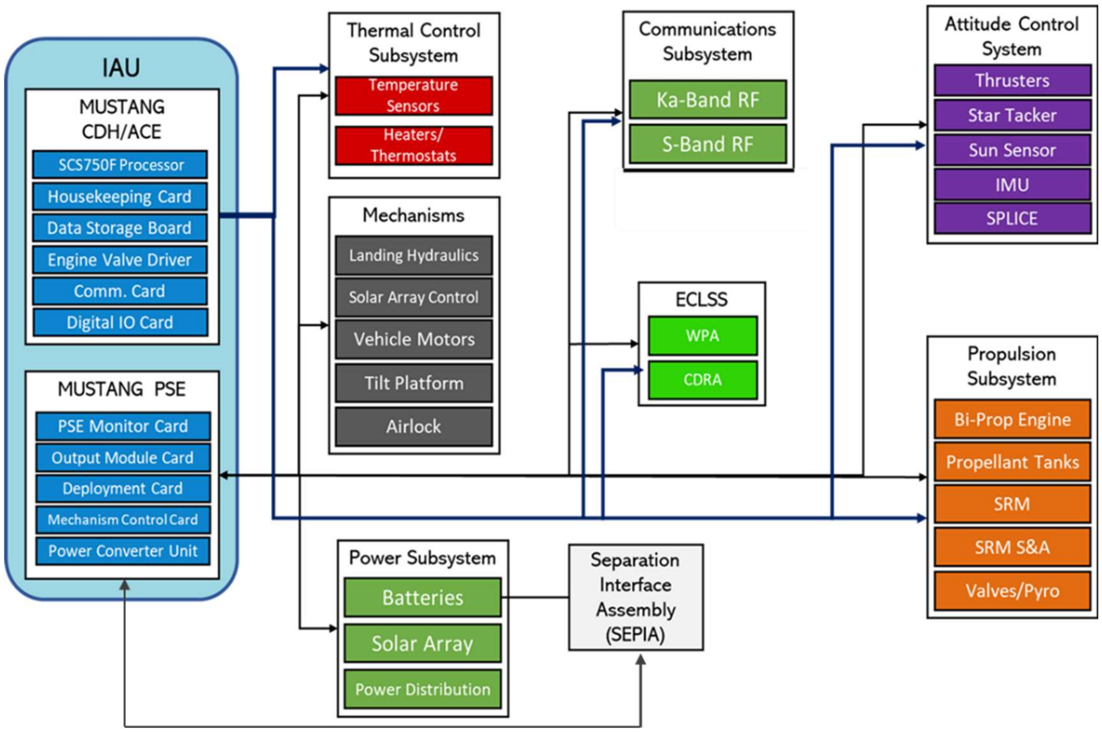


Figure 4.6-1 C&DH MUSTANG System Block Diagram

The SCS750F space computer is manufactured by Maxwell. It is rad-tolerant class S and demonstrated only one board upset every 100 years in simulated LEO or GEO testing. It has triple modular redundant processing and in the unlikely event of an upset, can flush, restore, and re-synchronize the three processors in 1ms [26]. With this processor board, errors due to upsets are highly unlikely and it is certainly prepared to fix the error.

4.6.1 Software

The primary operating system used is the real time operating system (RTOS) VxWorks, as it is the same system used for the SCS750F. Any additional software will use pre-existing NASA software or be built with the Core Flight Software (CFS), which is compatible with and supports the Maxwell SCS750 platform. CFS was selected as it and the MUSTANG are both produced from GSFC and any developed software will be easily applied to the different MUSTANG boards [27]. Software for BOWIE is divided between four major components, Core Flight executive services, CFS configurable applications, mission-specific software, and mission-specific devices input/output. The system can use pre-existing software developed from previous NASA missions, and use CFS to develop the new, original software necessary for this specific mission [27].

4.7 Power Subsystem

The power system for the Lunar BOWIE addresses the individual power system design for each BalLunar module and the assembled basecamp power. The system was designed considering the different power requirements for the different phases including the spaceflight phase, surface rendezvous phase, and occupied operations phase.

Defining powering requirements for individual subsystems proved to be problematic. Additionally, the RFP did not define NASA's payload or payload power requirements. Due to these factors, an estimation was used when designing the power system. NASA estimates 40 KWe baseline for a lunar surface basecamp and a basecamp on Mars, which includes a 30% margin [28]. Their 40 KWe estimate was used as baseline requirement for this mission. As seen in *Figure 4.6-1*, with a 40Kwe power level and nominal mission duration of 45 days, NASA recommends a solar or fission reactors to serve as the primary basecamp power source [29]. NASA's Kilopower reactor was considered for use, but due to its mass-to-power ratio, deployment, and safety concerns, solar arrays were selected as the primary power source.

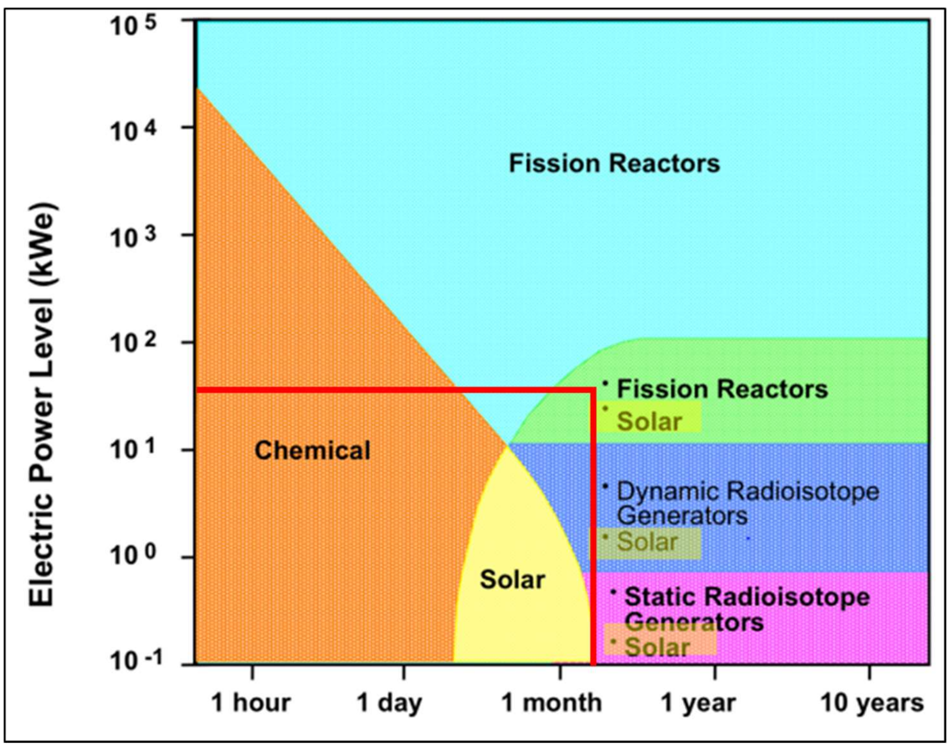


Figure 4.7-1 NASA Power System Recommendation

The design for the orbit to landing power requirements utilized NASA's 2019 "Lunar Lander Reference Design" as reference. The reference design followed a 3-6 day transit and allocated power required for cruise stage, TCMs, braking, landing, and immediate post landing. The reference design also accounts for systems engineering growth and an uncertainty percentage. The estimated energy requirement is 1,973 Wh and peak power requirement of 4,000W [30].

The C&DH systems' Separation Interface Assembly (SEPIA) limits the power system operations until the module separates from the LV second stage [30]. After separation, the module is completely powered by batteries until landing on the lunar surface. The battery selected is the space qualified MA190-210 modular lithium-ion battery manufactured by GS Yuasa. A single battery provides sufficient power, but four are included in each module. The inclusion of four batteries provides several functions. They ensure adequate redundancy, are used as a power source for surface rendezvous, and can act as a backup power source for the basecamp.

Table 4.7-1 MA190-210 Modular Lithium Ion Battery

Power (W)	Energy (Wh)	Mass (kg)	Quantity	Total Mass (kg)	Total Power (kW)	Total Volume (m^3)
4,290	7,144	59	4	236	17.1	0.185

After landing on the surface, each module requires power to rendezvous with the other basecamp modules. As landing sensors provide accuracy of 50m, the distance each module must travel is 50-100m. The MA190-210 batteries will be used to power the surface transit. The first three modules scheduled to land (2 Living, 1 VIBES) are each supplied with a small solar array which will deploy after landing. The array provides an additional 721 W to each module. The inclusion of solar arrays was to ensure enough power was supplied for the rendezvous and to account for any scheduling discrepancies which could result in delayed landing dates, requiring longer duration until complete basecamp assembly.

The arrays used are the Orbital ATK UltraFlex arrays. This array is compatible with all solar cell technologies [31]. The cells used in analysis are ZTJ triple junction cells [32]. The array was selected because it is lightweight, compact, maneuverable, and has high strength. The turning mechanism included with the array seen in *Figure 4.7-2*, is not standard but specifically designed for this mission. It is capable of 360 rotation and 15 tilt independent of the communications dish, ensuring the arrays are always facing the sun. The deployment of the array is seen in *Figure 4.7-3* The mass calculations includes the additional mechanism masses, and the power calculation accounts for 10% efficiency decrease due to operating temperature.

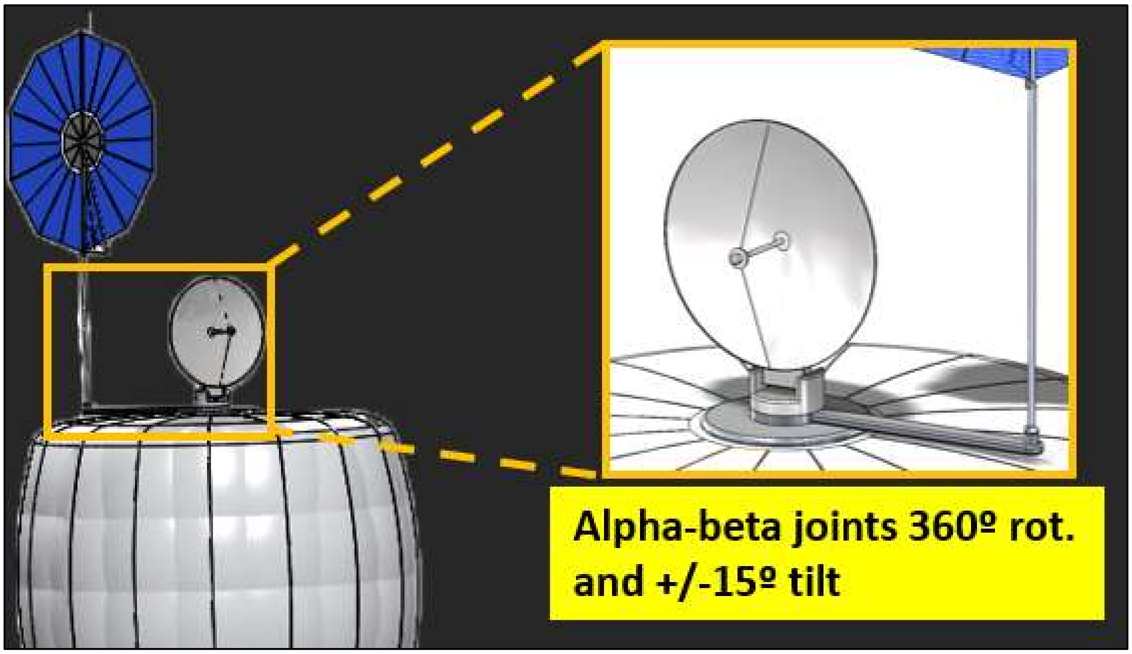


Figure 4.7-2 Solar Array Rotation & Tilt

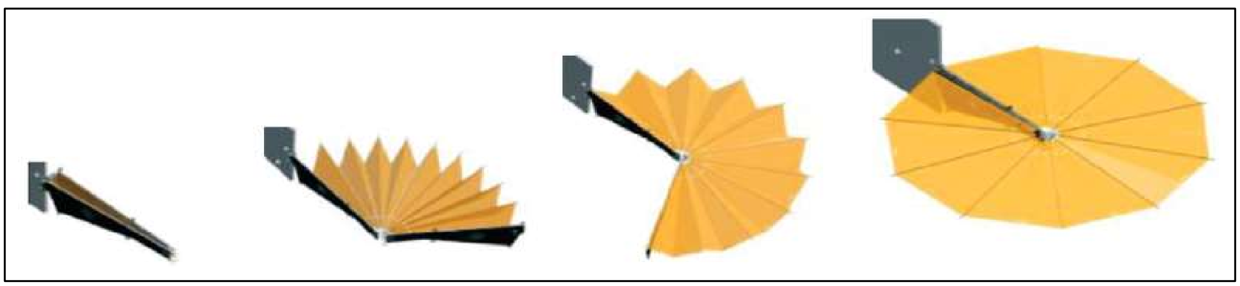


Figure 4.7-3 UltraFlex & MegaFlex

The primary solar arrays are a larger version of the UltraFlex arrays, the MegaFlex array. While following the same architecture, the MegaFlex name is applied to sizes of 10m diameter or larger, and this design utilizes two 10m arrays. The arrays are included with the last two modules to land, the ECLSS module and Cargo module. The larger arrays also exhibit the rotation and tilt capabilities of the smaller array. When deployed, the larger arrays are expected to completely block 1 of the 3 smaller arrays, while the remaining 2 will still be functional. The power for individual modules and the completed basecamp is specified in *Table 4.7-2.*

Table 4.7-2 Module Power System Breakdown

Module	Battery Power (kW)	Battery Energy (kWh)	Solar Array Power (kW)
Living Module 1	17.16	28.576	0.721
Living Module 2	17.16	28.576	0.721
Vibes Module	17.16	28.576	0.721
ECLSS Module	17.16	28.576	18
Cargo Module	17.16	28.576	18
Basecamp Total *	**85.5**	**142.88**	**37.442**

** Total Solar Array Power excludes 1 UltraFlex array due to being blocked by the Megaflex array after rendezvous*

The selected location, CR-1, provides 88.1% Illumination 2m above ground and 91.94% Illumination 10m above ground. The arrays are positioned atop each module, with the bottom of each array set 6.5m above ground, receiving from continuous illumination 88-91% of the time. The modules share power through adapters included in the bride docking system.

4.7.1 Power System Lifetime

The MA190-210 modular lithium-ion battery has had extensive use, in mostly LEO and GEO missions. GEO missions showed a lifetime of 18 years. After 4,000 cycles of 80% depth of discharge (DOD), the battery capacity was only reduced to 85% of its BOL capacity [33]. The solar array advertised lifetime is 15 years, but more important is the lifetime of the solar cells. The equivalent fluence approach was used to analyze radiation received by the cells and its effect on efficiency compared to testing following AIAA-S-111 Radiation Performance Standards. The analysis focused on the effects of the continuous solar wind, considering 96% of particles being photons and 4% of particles as alpha-particles [34]. Data from NASA's WIND satellite provided mean energy levels of the particles and NOAA solar cycle prediction allows to determine what flux level to use. The mean energy level used for protons is 12.7eV , and alpha-particle mean energy is 23.9 eV [35]. The flux ranges from 10^{10} to 10^{12}pfu; As seen in *Figure 4.6.1-1*, lower activity is expected during mission time in 2030, therefore a flux level of 10^{11}pfu was used [36]. Results can be seen in *Table 4.6-3*, noting negligible degradation after 45 days and degradation of - 10% efficiency after 4,400 days or 12 years.

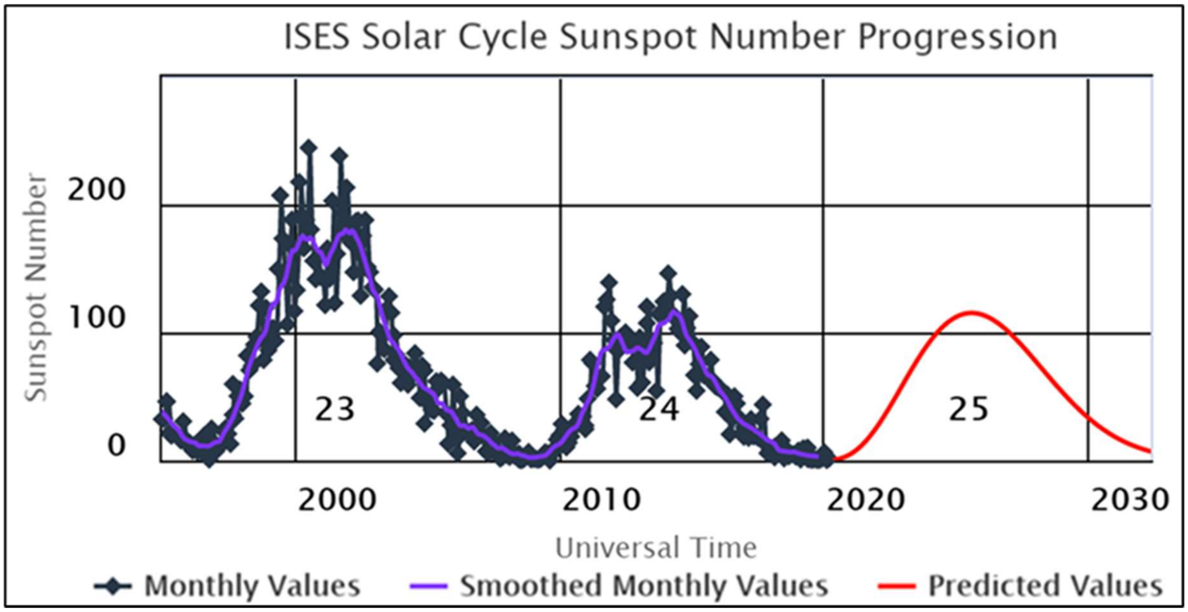


Figure 4.7.1-1 NOAA Solar Cycle Prediction

Table 4.7.1-1 Solar Array Efficiency Degradation Using Equivalent Fluence Approach

Fluence (e/cm^2)	Day	Pmp	Power Output (kW)
4.9+E12	45	-	36.4
3+E13	264	0.99	36
1+E14	880	0.96	34.9
5+E14	4401	0.9	32.76

Solar storms and high energy particles are sporadic and low in frequency. If a large energy event were to occur, the basecamp would receive ample notice from the ground station; the arrays are capable of stowage and re-deployment after the storm passes. The Cargo module is also equipped with a spare 10m MegaFlex array if an array needs to be replaced. The batteries can provide 143 kWh for emergencies or to adjust for shading of the arrays. The power system is capable to provide power for extended mission durations.

4.8 Environmental Control and Life Support System

As a human spaceflight mission, the safety of the crew is paramount. To provide necessary Environmental Control and Life Support System (ECLSS) monitors and controls the internal basecamp environment and the water recovery system. The system included in this design was derived from the ECLSS on the International Space Station and designed according to NASA-STD-3001, "Spaceflight Human-System Standard". While there are a multitude of system requirements, the driving requirements can be seen in *Table 4.7-1.*

Table 4.8-1. ECLSS Requirements

Requirement Statement
Each module shall allow to display, monitor, and control its own atmospheric conditions such as pressure, humidity, temperature, and ppO2, ppCO2
The system shall maintain the pressure to which the crew is exposed to between 3.0 psia < pressure ≤ 15.0 psia
The system shall be able to maintain thermal conditions in the comfort zone
The system shall maintain the atmospheric temperature within the range of 18 ºC to 27 ºC during all nominal operations
The system shall provide potable water that is safe for human use, including drinking, food rehydration, personal hygiene, and medical need
The food system shall provide each crewmember with a minimum number of calories per day, based upon estimated energy requirements (EER)
The system shall provide a minimum of 2.0 kg of potable water per crewmember per mission day for drinking
CO2 levels shall be limited to the values stated in the tables located in JSC 20584, Spacecraft Maximum Allowable Concentrations for Airborne Contaminants
A fire protection system comprised of detecting, warning, and extinguishing devices **shall** be provided to all spacecraft volumes during all mission phases without creating a hazardous environment
The LBC shall provide level III medical care capabilities to the crew for the full duration of the mission

[38]

Each module contains proper equipment to allow the display, monitor, and control its own atmospheric conditions, but all modules must be connected to allow full ECLSS operations. The piping for the shared system is included within the inter-module bridges. An extended mission would require a re-supply of oxygen, supplied with an additional BalLunar module.

While the ISS ECLSS includes a closed-loop oxygen reclamation system, this design instead supplies the required oxygen for 60 days. "The current ISS Oxygen Generation System (OGS) would pay back the mass of the initial system and three spares in 557 days" [39]. Being a shorter mission duration than ISS, it is more mass efficient to supply the necessary oxygen than to include the OGS. The ECLSS does utilize the same Carbon Dioxide Removal System (CDRS) and Water Recovery System (WRS). Also included is component spares to provide 0.1% total failure rate over 450 days [39]. The ECLSS hardware, consumables, and packaging can be seen in *Table 4.7-2*. The operating hardware is contained in the ECLSS module, while consumables and spares are divided between the other base camp modules. The storage of the system hardware can be seen in *Figure 4.7-1*.

Table 4.8-2 ECLSS Mass Breakdown [39]

System	Mass Including Spares (kg)	Mass Breakeven Days
Carbon Dioxide Removal System (CDRS)	428	61
Water Recovery System (WRS)	3,649	196
Item	**Mass (kg)**	
Water	500	
Oxygen	208	
Nitrogen	174	
Rodnik Water Tanks	70	
Nitrogen/Oxygen Recharge System Tanks (NORS)	523	
Total Mass (kg)	**5,407**	

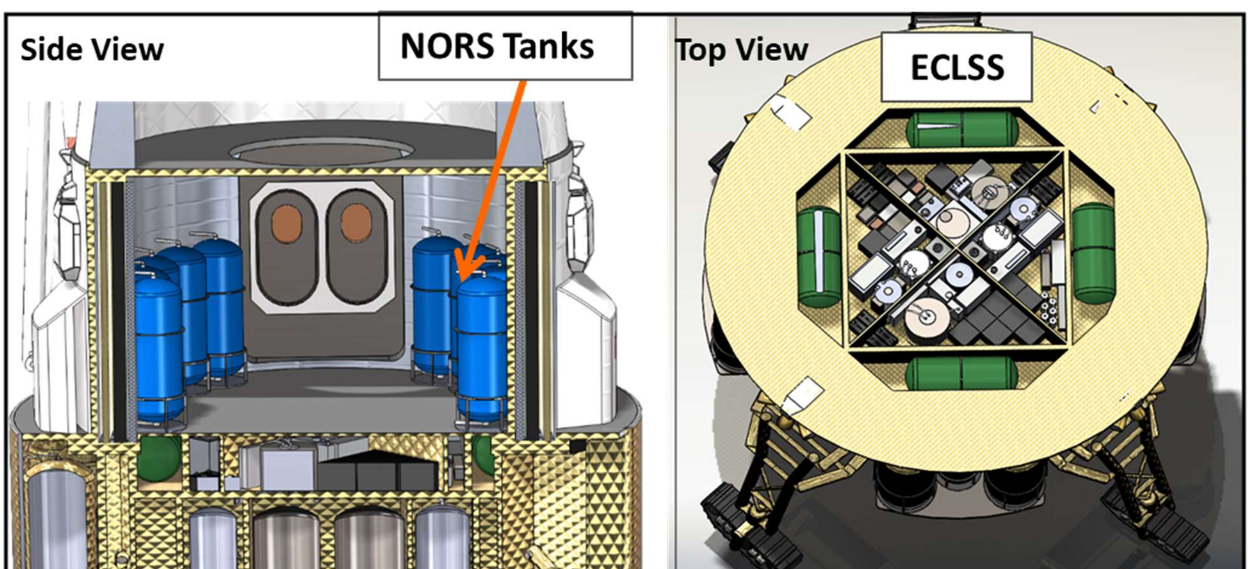


Figure 4.8-1 ECLSS Hardware Storage

The WRS recycles a 30-day water supply, which is sufficient to meet the potable water requirements. While majority of operating hardware is in the ECLSS module, other modules also contain necessary elements. Such as, each module carries 1 nitrogen tank for initial pressurization. Modules will be pressurized to 14.7 psi, or 1 atm, and the air will consist of 21% oxygen & 79% nitrogen.

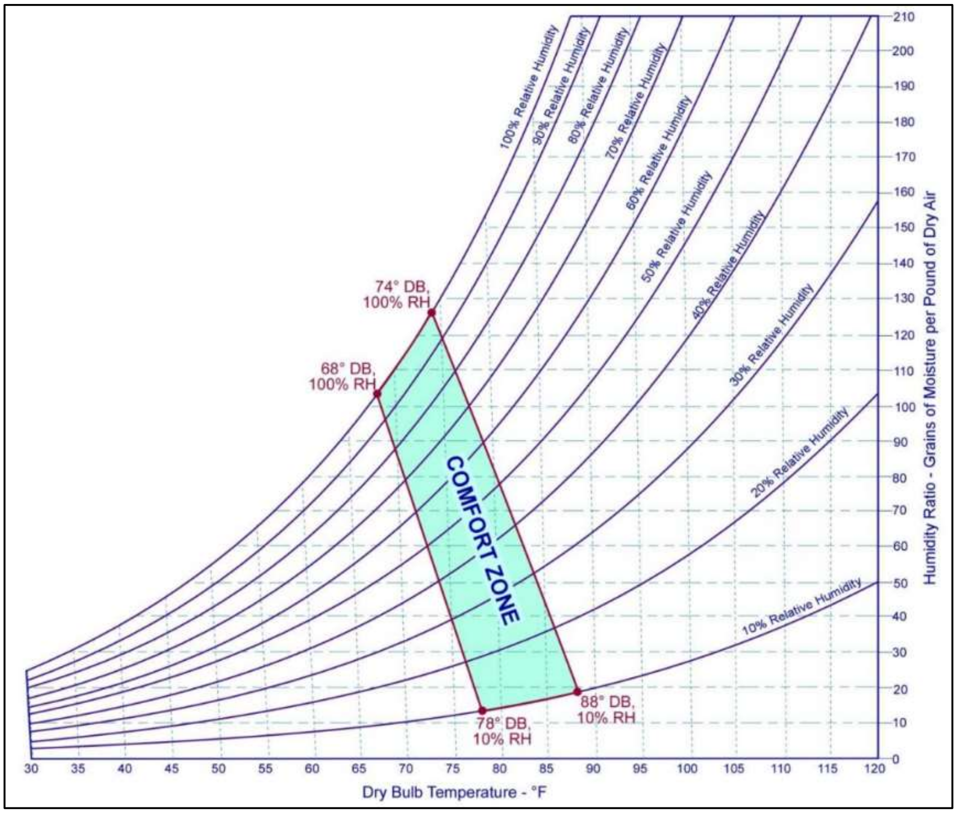


Figure 4.8-2 NASA Comfort Zone for Crewed Missions

The Comfort Zone determined by NASA can be seen in *Figure 4.7-2* [37]. It should be noted the required temperature shown is the dry-bulb temperature. The WRS maintains proper interior humidity. The worst-case-hot thermal analysis determined the interior ranges from 31.6 ºC (88.8 F) to 41.6 ºC (106.88 F), while requirements are to maintain 18 ºC (64 F) to 27 ºC (80 F). To meet the temperature requirements, the Active Thermal Control System (ATCS) includes 4 sets of radiators for each module, as seen in *Figure 4.7-3*. The mass of the radiators is included in the Thermal Protection system.

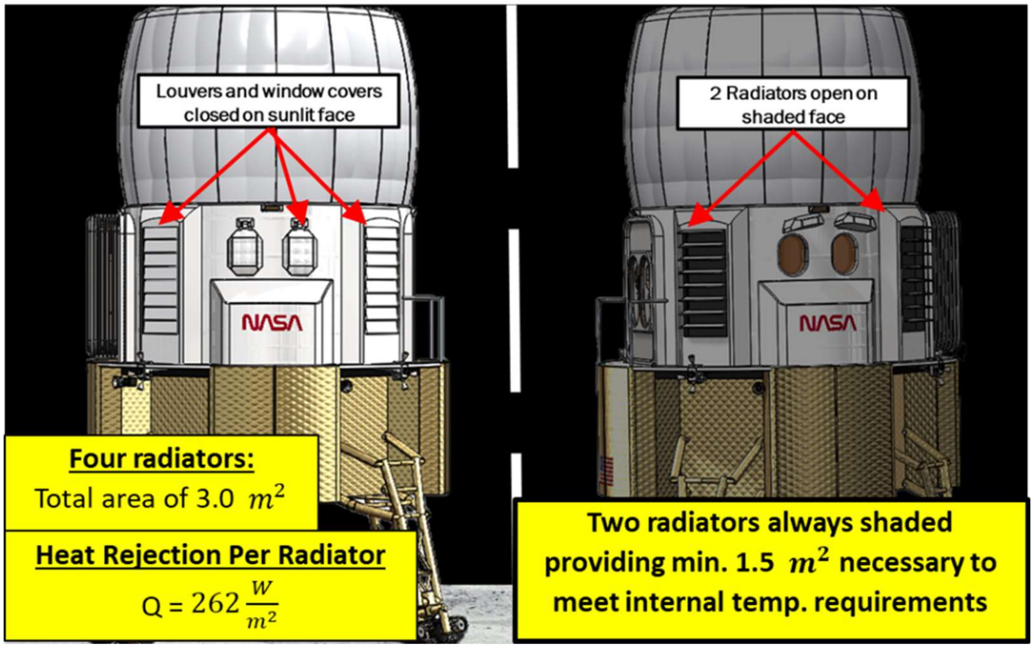


Figure 4.8-3 Thermal Protection System Radiators

4.9 Crew Accommodations

The Crew Accommodations are the "elements of mission hardware, software, and even procedures that most directly serve the needs of the crewmembers". The mass of accommodations was derived from a resource model from "Human Spaceflight Mission Analysis and Design" and was modified to better follow NASA STD-3001 and historical usage data from the ISS. The most notable masses are galley and food, to accommodate for a baseline 1.8 kg/person/day; maintenance, to ensure enough tools and equipment for repairs; and crew health care, to meet the requirement of level III medical care capabilities, required for the nominal mission duration of 45 days.

Table 4.9-1 Crew Accommodations

Crew Accommodation	Mass Subtotal (kg)
Galley and Food	728
Waste Collection	100
Personal Hygiene	90
Clothing	46
Recreational Equipment	100
Housekeeping	137
Operational Supplies and Restraints	130
Maintenance	750
Photography	90
Sleep Accommodations	36
Crew Health Care	567
Cargo Transfer Bag Equivalent	117
Total	**2,891**

The accommodations are divided in different modules, and the exact mass division can be seen in the final module mass statements. The galley or kitchen for the crew is contained in the VIBES module. For waste collection, each living module includes its own bathroom including toilet and shower. An extended mission would require re-supply of accommodations such as food, and other consumables for personal hygiene, housekeeping, and healthcare.

4.10 Scientific Instruments and Payloads

Due to the lack of any specified scientific mission in the RFP, we have developed multiple possible scientific missions, in addition to a few scientific instruments that could be used for those specified missions. However, these should be viewed as suggestions, as the scientific goals of the LBC will most likely change after the Artemis program finishes their research goals. Thus, to accommodate this uncertainty, the BOWIE modules have International Standard Payload Racks (ISPRs) located on all modules except the ECLSS module. The total potential equipment mass for the BOWIE modules is 1025 kg, split among the 4 modules containing ISPRs. These are broken down further in the mass breakdowns.

The primary science objectives for the LBC mission follow similar mission directives for Artemis. Those missions, in addition to a few scientific equipment capable of completing those missions, are:

- Research Lunar Volatiles
    - Catalogue and research the composition and usability of volatiles, such as lunar ice, on the lunar surface.
    - Can be completed using the lunar Volatiles Scout (LVS)
        - A 2-meter drill that using spectroscopy to analyze regolith samples
- Lunar Surface Exploration
    - Catalogue exploration missions across the lunar surface to better understand the lunar environment, in addition to creating a more detailed map of the lunar surface.
- Lunar Regolith uses for Construction and Farming
    - Testing of the lunar regolith for uses as construction material or farming soil. Primary experimentation for future permanent lunar habitation and lunar base construction.
    - Can be completed by an Advanced Plant Habitat (APH)
        - Fit for an ISPR, the APH tests plant growth and soil composition
    - Can be completed by a use of a regolith 3D printer

        - By using lunar regolith as the base material for 3D printing, larger structures and other parts can be manufactured on the lunar surface for a lower cost

- Human Habitability on the Lunar Surface
    - Observation of lunar environmental effects on human body, and experimentation of ways to mitigate negative effects

5.0 Mission Management and Operations

In this section, an overview of the total program schedule from beginning to lunar deployment is presented, followed by comparison of the total mission lifetime with resupply missions and without them. A summary of the preliminary manufacturing techniques and a list of suppliers is discussed. The identification and mitigation of program risks are discussed, to show that preparations have been made to account for delays or other possible problems encountered during the 10-year duration of the project. The next section details the end of mission plans, and the disposal plans for the BOWIE modules. Finally, a cost breakdown of the entire project is presented, followed by a mass breakdown of each variant of the spacecraft modules.

5.1 Program Schedule

In order to deliver a fully operational lunar base camp by the 31st of December 2030, Space Oddities will implement the schedule for the Lunar BOWIE program shown below in *Figure 5.1-1*. Readiness Reviews are scheduled as a program milestone to be achieved by the set dates. As of this proposal, Preliminary Design Review for the Lunar BOWIE was completed as scheduled and the program is currently on schedule to deliver the base by the schedule requirement.

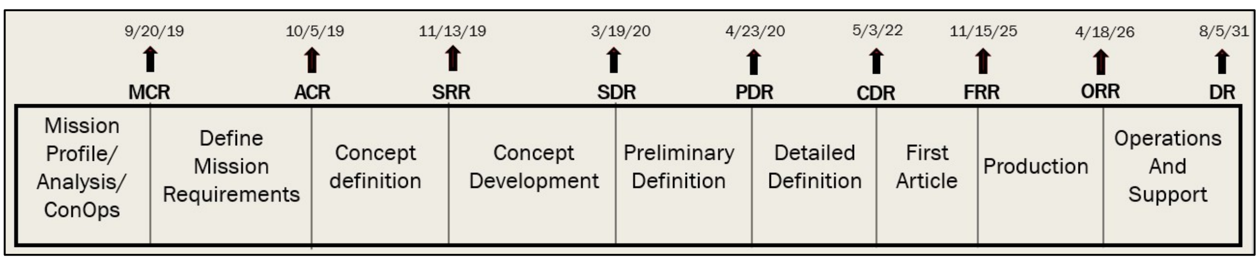


Figure 5.1-1 Lunar BOWIE Program Schedule

5.2 Mission Lifetime Summary

Estimates of the majority of the systems and component lifetimes are based on using the ISS as a historical analog. The lifetime of the solar arrays is based on the analysis of degradation found in the Power Systems subsection. The main factor in determining the lifetime of the Lunar BOWIE depends on whether the base will be continuously expanded and resupplied and if so, the shortest lifetime of the base's components that cannot be resupplied or replaced. The main propulsion and attitude control systems do not play a factor in the lifetime assessment as the base will no longer make use of them once fully deployed. If there is no mission to expand and resupply the base, then the original mission will only last as long as the margins considered for the air and food supply of the crew. The crew will only be able to remain on the base 15 days beyond the nominal mission period of 45-days. If there are missions to expand the base, then the base can be resupplied and the base's lifetime will last as long as that of the inflatable structure, as that cannot be replaced. Solar arrays, batteries, telecom and avionics components can be preplaced. Therefore, the estimated maximum lifetime of the Lunar BOWIE with expansions and resupplies is 12 years.

Table 5.2-1 Summary of Mission Lifetime Estimates for System and Components

System Lifetime	Duration
Northrop Grumman Star 37 G SRM	4 Days
Aerojet Rocketdyne RL 10C-3	4.5 days
ACS	4.5 days
ECLSS: Open Loop Air Supply System	60 Days
ECLSS: Food and Food Management	60 Days
ECLSS: Closed Loop Water System	8 Years
ECLSS: Waste Management	8 Years
Main Structure	15 Years
Inflatable Structure	12 Years
TCS	15 Years
Solar Arrays	12+ Years
Batteries	5 Years
Telecoms	15 Years
CD&H	15 Years
Total Base Lifetime w/out Resupply	**60 Days**
Total Base Lifetime w/ Resupply	**12 Years**

5.3 Manufacturing and Supply Chain

The main body of all Lunar BOWIE variants will be manufactured and assembled at Space Oddities headquarters in Malibu, California, USA. The assembly of the modules include all subsystem components that will not be included in the assembly of the lander system. The main propulsion system used for the lander system will be delivered to a rented-out facility at the Kennedy Space Center facility to be assembled with the lander system. The assembled modules will be transported to the KSC facility for integration with the lander system. A list of manufacturers for the main subsystem components is provided in *Table 5.3-1*. Due to the COVID-19 pandemic, Space Oddities original planned contract with Bigelow Aerospace to manufacture and supply the inflatable structure has been changed, due to Bigelow Aerospace having laid off all of its employees. To ensure that Space Oddities meets the schedule requirements, a secondary supplier of inflatables, Redline Aerospace is now the main contractor for the Lunar BOWIE's inflatable structure.

Table 5.3-1: List of manufacturers for Lunar BOWIE

Manufacturer	Component	Location
Northrop Grumman ATK	Star 37G SRM	Ogden, UT
Aerojet Rocketdyne	RL 10 C-3	El Segundo, CA
Aerojet Rocketdyne	MR 106-L-22N	El Segundo, CA
NASA Langley Research Center	SPLICE	Langley, VA
Astrobotics	Lunar Relay Communication Satellite	Pittsburgh, PA
NASA Goddard Space Flight Center	MUSTANG	Greenbelt, MD
Northrop Grumman ATK	MegaFlex Solar Arrays	Goleta, CA
GS Yuasa	Batteries	Roswell, GA
NASA Marshall Space Flight Center	ISPR	Huntsville, AL
Redline Aerospace	Inflatable Structure	Chilliwack, BC, Canada
MDA Robotics and Automation	BalLunarm	Brampton, Ontario, Canada

5.4 Risk Analysis

A prevented detailed risk analysis was performed for mission safety and assurance purposes. Risk analysis creates awareness of hazards and identifies risks to the program. Risks that are identified are written into risk statements and then mitigated using risk mitigation waterfall steps. Worst-case scenarios are also looked at to ensure proper contingencies are taken.

5.4.1 Risk Statements

Table 5.4.1-1 includes some of the most important risk factors that were considered during the design process. Technical Risks (TR) and Management Risks (MR) are both considered to help ensure that design requirements are met and are written into to 'if, then' statements to better perform risk analysis. For design coherency, each risk identified can be traced back to related design requirements. The most important technical risks identified involve risk to crew health and safety and the management risk presented involves risk to scheduling delays.

Table 5.4.1-1: Risk statements with related requirement

Related Requirement	Risk #	Risk Statement
T0.0-3 to 6	TR1	If life support systems were to fail due to component failures, then crew life will be in danger
LBC shall minimize crew radiation exposure to 1,000 mSv	TR2	If radiation levels exceed shielding for average radiation at surface target location due to SPE, then crew life will be in danger
Maintain 24-hour coverage between crew and Earth	TR3	If 24-hour coverage with the crew is interrupted due to operational failure of lunar relay constellation then risk to crew life will increase.
System Level: LBC shall be fully operational by 31-12-2030	MR1	If manufacturing delays occur due to disruption in the supply chain, then delivery of lunar base will not comply with RFP schedule

Risks are also quantified and organized into a risk cube as shown below in *Figure 5.4.1-1*. The risks are quantified by the likeliness of the risk to occur versus the severity of consequences of the risks if they were to occur. A risk that is very likely to occur and yields high consequences is considered a high risk and a risk that is not likely to occur and yields minimal consequences is considered low risk. Any risk that is considered high or medium risk must be mitigated to low risk status in order for the risk to be fully addressed. Any risk that are already considered low do not need to be addressed.

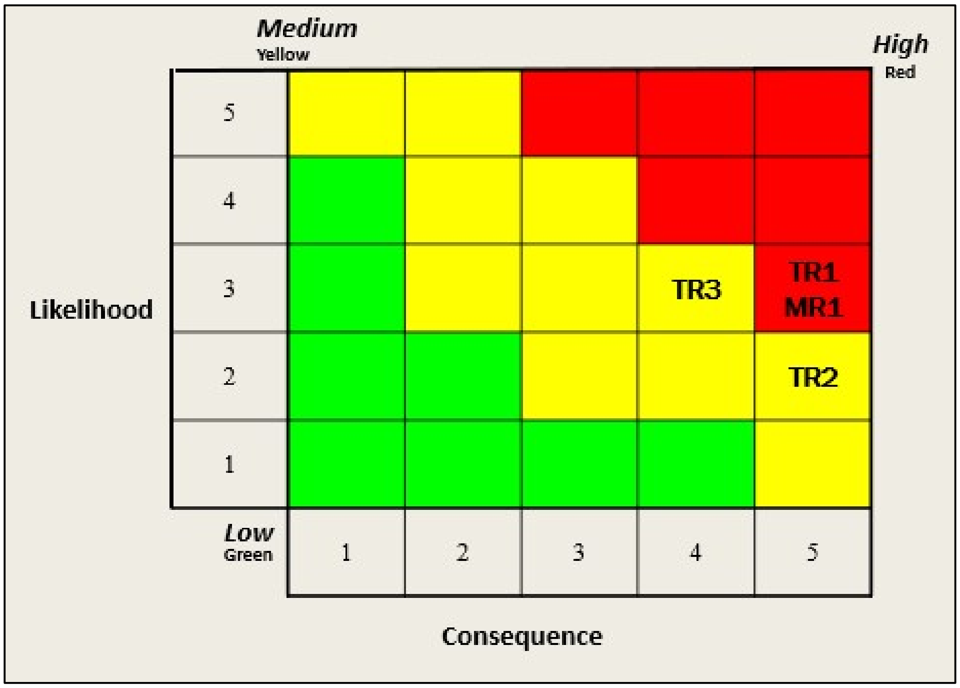


Figure 5.4.1-1 Risk Cube

5.4.2 Risk Mitigation

To mitigate high and medium risks, risk mitigation waterfalls or risk burndowns are used to identify the necessary steps to bring down the risks to acceptable levels. Each risk that requires mitigation also requires the risk to be mitigated within a targeted timeline. For Space Oddities, program milestones are used as target dates to bring down the risks to acceptable levels. Subsequent mitigation approaches for each risk stated in *Table 5.4.1-1* are shown below in *Figure 5.4.2-1* to *Figure 5.4.2-4*.

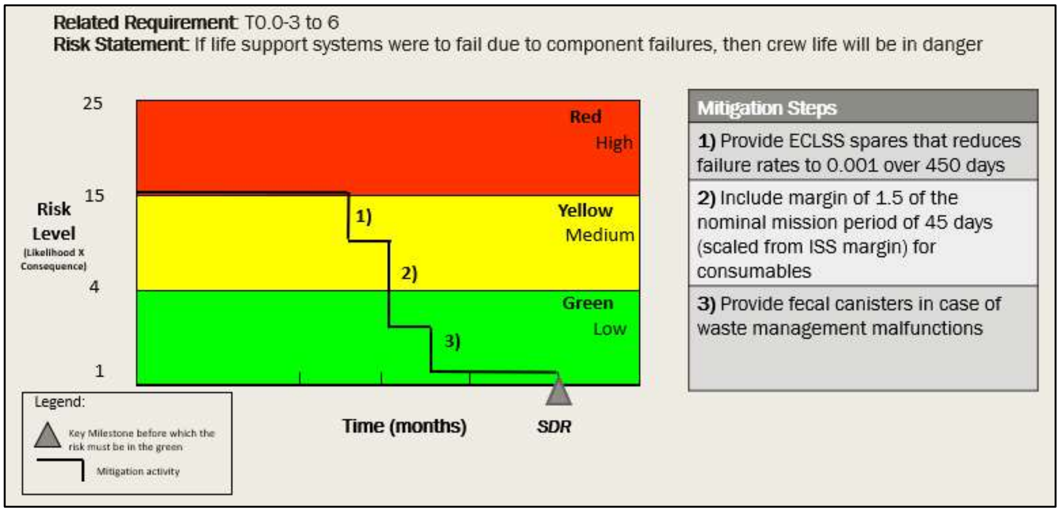


Figure 5.4.2-1 Risk Mitigation for Technical Risk 1

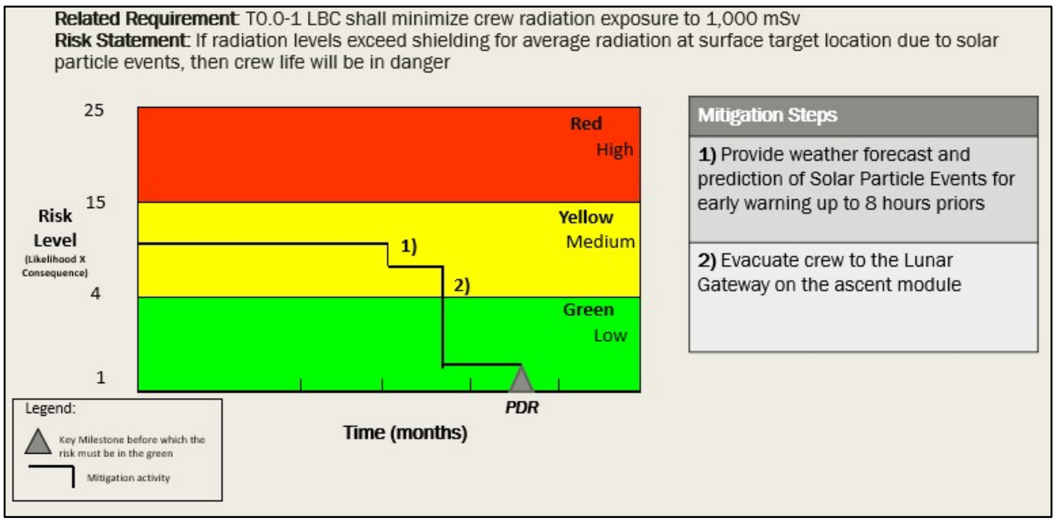


Figure 5.4.2-2 Risk Mitigation for Technical Risk 2

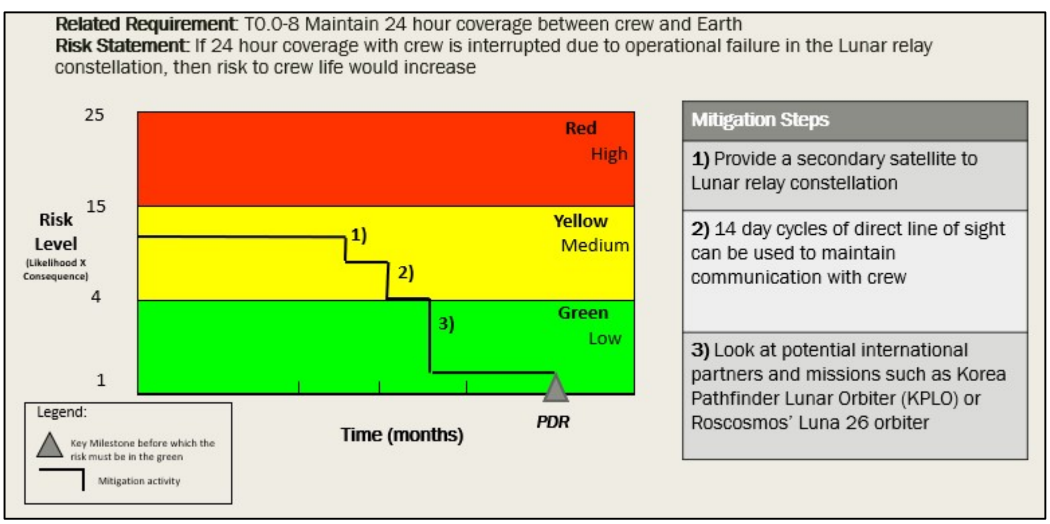


Figure 5.4.2-3 Risk Mitigation for Technical Risk 3

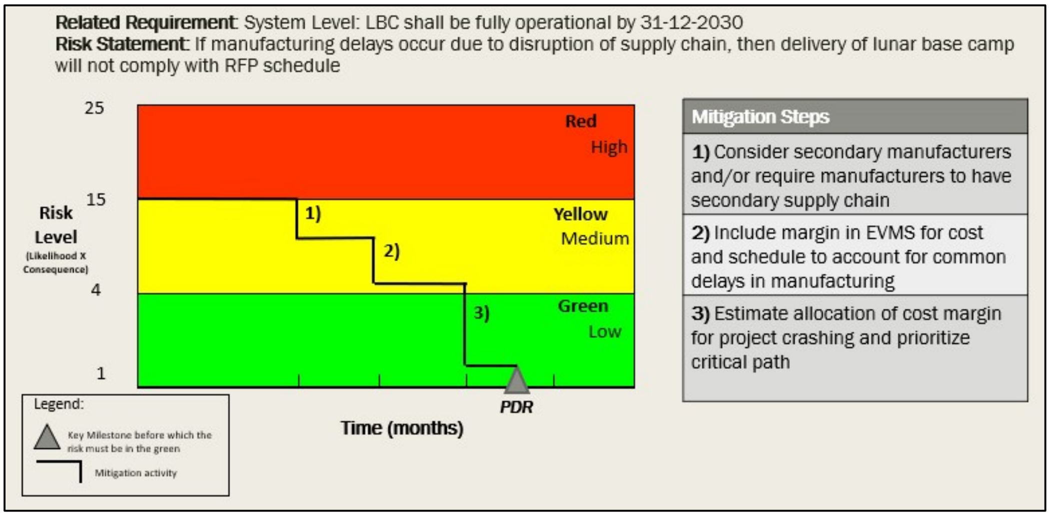


Figure 5.4.2-4 Risk Mitigation for Management Risk 1

5.4.3 A Worst-Case Scenario

Worst-case scenarios were also considered to ensure crew health and safety even during catastrophic events. In this particular case, Loss of Vehicle scenarios were considered with plans and procedures discussed to ensure the safety of the crew and possible results of the mission. Space Oddities have determined that any LOV to the VIBES Module or the ECLSS Module would require the evacuation of the base to the lunar ascent module for the crew to be returned home or shelter in the Lunar Gateway. The remaining modules will be able to function independently and sustain the crew long enough to prepare for evacuation by way of suit ports and their space suits. The loss of the VIBES and ECLSS modules will result in a Loss of Mission. LOV of 2 habitat modules will also result in LOM. However, the loss of a single habitat module will not result in a Loss of Mission. The remaining modules will be able to sustain crew life, however some of the crew will have to occupy other modules. The 60-day margin for food and air will allow the crew to continue the mission for the nominal 45 days. The xEMUs from the lost habitat module will be recoverable by the crew members to be stored in the airlocks of other modules. A LOV to the cargo module will not significantly impact the crew or the mission.

5.5 End of Mission and Disposal Plan

End of Mission plans will follow NASA's procedure for Planetary Protection Category II [40]. Under NASA guidelines, an End of Mission report will be produced with the assessment of the Lunar BOWIE's impact on the lunar environment and the ability to conduct future science and exploration missions. However, following NASA's guidelines, there is a remote chance that contamination from the Lunar BOWIE will affect future science interests and returns. Further plans were also made based on two cases considered. Upon the success of the 45-day mission period, the Lunar BOWIE will be capable of extended mission support and operations. The Lunar BOWIE will have the 15-day margin of consumables for a short-extended mission, but Space Oddities will also have the operational capabilities to send resupply modules for longer extended mission periods that also serve as the expansion of the lunar base. In the case that the 45-day mission period is not met with the early return of the crew or the mission is not extended, the base will be powered down for hibernation and can remain on the lunar surface.

5.6 Cost Analysis

In order to meet the cost requirements and stay within the budget of $12B USD for FY 2019, a detailed cost analysis was performed using NASA's cost estimating software, Project Cost Estimating Capability (PCEC). Total cost estimation of the Lunar BOWIE totaled $10.73B USD for FY 2019. A cost Work Breakdown Structure (WBS) of the program functions and product are provided in *Figure 5.6-1***.**

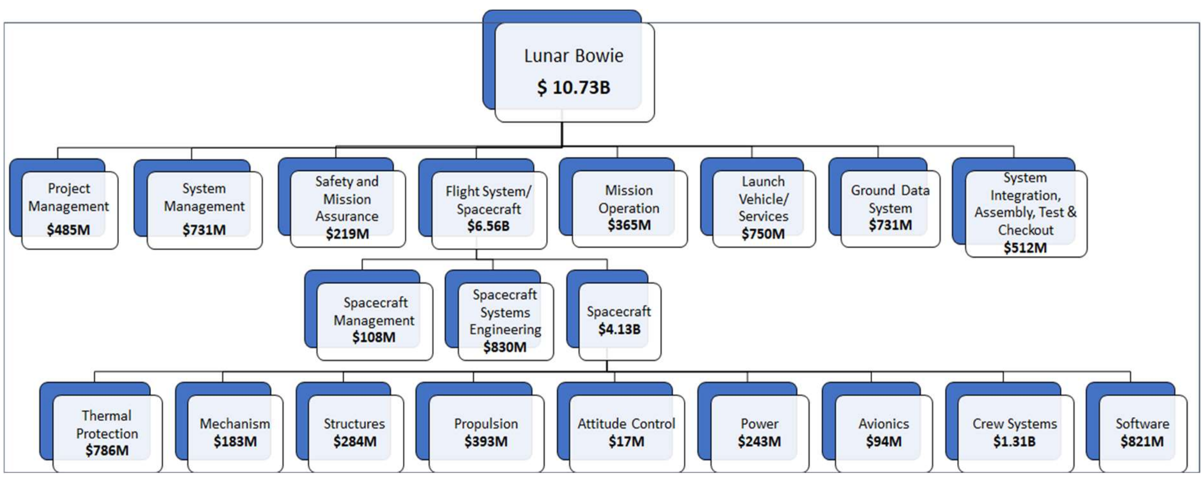


Figure 5.6-1: Cost Work Breakdown Structure for the Lunar BOWIE program.

To ensure confidence in the cost estimation of the Lunar BOWIE, direct input and analogous methods within PCEC was used. Using the direct input method required actual cost data of all subsystems and subsystem components, along with functional cost estimates. Analogous method utilizes past NASA missions as a reference on estimating cost for each subsystem and functional aspects of the program. The main reference mission used for the analogous method is the Apollo Mission and certain cost parameters were adjusted using the ISS. The cost of 5 Falcon Heavy launches was directly inputted as an adjustment for launch vehicle cost. At a total cost of $10.73B USD, there is margin of $1.27B USD out of the $12B USD that Space Oddities will use to for the possible acquisition and launch of another lunar relay constellation satellite for communication redundancy and for production of possible back-up modules to be used. The cost margin will also be considered for any increases in cost production and for program crash costing to prevent delays.

5.7 Variant Mass Breakdown

Using the Falcon Heavy Expendable, the mass budget for a TLI mission is 20,946 kg. In order to successfully complete the mission, all modules must meet or be under this budget to ensure a successful launch. The following tables are a detailed breakdown of each variant's mass, with subsystem masses.

Variant #1, Living Module

The LM shares the inflatable upper section with the VIBES module, and the inflatable material required to create this large upper section has a larger mass than the solid structure. Thus, for the LM and the VIBES modules, the structure mass is almost three times as large as first estimated using Human Spaceflight: Mission Analysis and Design[41]. Due to the dated nature of the initial mass estimation, the estimates for Avionics, Power and ACS systems are all too high, thus allowing for other subsystem masses to increase.

The LMs also contain personal crew accommodations, as seen on *Table 5.7-2* (Note: some of the listed crew accommodations have been split evenly among the two LMs, resulting in the mass per module being halved). These accommodations would be stored in the three ISPRs on the LMs

Table 5.7-1 Living Module Mass Estimates

Subsystem	Allocation %	Budget (kg)	Current (kg)	Status
Structure	22%	1663	4595	C
Mechanisms	8%	605	950	C
Thermal Protection	9%	680	680	E
Propulsion System	15% (prop)	1073	740	C
Attitude Control	2%	151	88	C
Power	15%	1134	261	C
Avionics and Controls	10%	756	254	C
Environmental control and life support	8%	605	805	C
Crew Accommodations	8%	605	684	C
Landing	18%	1361	1361	E
Other	n/a	n/a	n/a	
Dry Mass w/o Prop		7561	9677	E
Dry Mass		**8634**	**10417**	**E**
Payload		3000	312	C
Inert Mass		**13793**	**10729**	**E**
Propellant Mass		7153	9672	C
Wet Mass		**20946**	**20401**	**E**

The LMs also contain some ECLSS system backups, and packaging for the ECLSS system. This results in 200 kg per module of extra ECLSS mass, besides the estimated 605 kg for pipes and pumps throughout each module.

The mechanism mass consists of the drive system and treads used for autonomously rendezvousing on the lunar surface. Treads are heavier than the estimated mass for wheels, and thus that subsystem mass is larger than the estimate.

Table 5.7-2 Crew Accommodations per Living Module

Crew Accommodations (For 1 LM)	Mass (kg)
Personal Hygiene	45
Clothing	23
Recreational Equipment	50
Housekeeping	68
Photography	45
Sleep Accommodations	18
Cargo Transfer Bag Equivalent	50
Crew Health Care	283.5
Waste Collection (Lavatory)	100
Total	**682.5**

The minor payload mass is listing only the known ISPR's on the LMs. The LM has a capacity of 300 kg of scientific equipment in its current configuration, and the equipment would be stored in the ISPRs on the LM, in addition to the crew accommodations. Scientific equipment would have priority storage, as the crew accommodations can be stored elsewhere if needed. Each ISPR has a mass of 104 kg.

Variant #2, Cargo Module

Table 5.7-3 Cargo Module Mass Estimates

Subsystem	Allocation %	Budget (kg)	Current (kg)	Status
Structure	22%	1663	2999	C
Mechanisms	8%	605	1006	C
Thermal Protection	9%	680	680	E
Propulsion System	15% (prop)	1073	740	C
Attitude Control	2%	151	88	C
Power	15%	1134	551	C
Avionics and Controls	10%	756	254	C
Environmental control and life support	8%	605	1954	C
Crew Accommodations	8%	605	750	C
Landing	18%	1361	1361	E
Other	n/a	n/a	n/a	
Dry Mass w/o Prop		7561	9638	E
Dry Mass		**8634**	**10378**	**E**
Payload		3000	416	C
Inert Mass		**13793**	**10794**	**E**
Propellant Mass		7153	9725	C
Wet Mass		**20946**	**20524**	**E**

The Cargo Module shares a solid upper structure with the ECLSS Module, contrasted with the inflatable upper structure LM and the VIBES Module. This mass is substantially lower than the inflatable mass, thus causing the structural mass the almost be double, instead of tripling the mass of the estimate.

For the ECLSS subsystem, the Cargo Module contains a large majority of the ECLSS backups, in addition to packaging, that for mass purposes could not fit on the ECLSS module. The estimate for pipes and pumps is lower than the LM estimate, due to the lower requirement of water flow (no lavatory or accessible drinking area), thus allowing for a larger amount of backup ECLSS equipment to be stored. The Cargo Module contains 1649 kg of ECLSS backups and packaging, and an estimated 305 kg of pipes and pumps.

The mechanism mass is slightly higher due to the presence of the BalLunarm, as the treads and drive system exist on every other module. In addition, the only crew accommodation stored on the Cargo Module is the maintenance gear for the crew, which is estimated at 750 kg using Human Spaceflight Mission Analysis and Design. The Cargo Module contains 4 ISPRs, with a total mass of 412 kg. The scientific equipment capacity of the Module is 225 kg.

The Cargo Module contains 2 MegaFlex solar arrays: 1 deployable on the solid upper structure, and another space MegaFlex array located within the module itself. The total mass for the 2 arrays is 315 kg, which combined with the 236 kg battery power results in the largest mass for the power subsystem among all four variants.

Variant #3, ECLSS Module

Table 5.7-4 ECLSS Module Mass Estimates

Subsystem	Allocation %	Budget (kg)	Current (kg)	Status
Structure	22%	1663	2999	C
Mechanisms	8%	605	1006	C
Thermal Protection	9%	680	680	E
Propulsion System	15% (prop)	1073	740	C
Attitude Control	2%	151	88	C
Power	15%	1134	389	C
Avionics and Controls	10%	756	254	C
Environmental control and life support	8%	605	3358	C
Crew Accommodations	8%	605	0	C
Landing	18%	1361	1361	E
Other	n/a	n/a	n/a	
Dry Mass w/o Prop		7561	10135	E
Dry Mass		**8634**	**10875**	**E**
Payload		3000	0	C
Inert Mass		**13793**	**10875**	**E**
Propellant Mass		7153	9741	C
Wet Mass		**20946**	**20616**	**E**

The ECLSS module is the most specialized module, as it contains no ISPRs for scientific equipment or crew accommodations. All of the mass is transferred to storing the primary ECLSS system, the consumables required, the remaining ECLSS backups, and the packaging for the whole system. The mass includes all of the pumps and pipes for the system. The structural mass and mechanism mass' are the same for the Cargo Module as it is for the ECLSS module.

Variant #4, VIBES Module

Table 5.7-5 VIBES Module Mass Estimates

5Subsystem	Allocation %	Budget (kg)	Current (kg)	Status
Structure	22%	1663	4595	C
Mechanisms	8%	605	950	C
Thermal Protection	9%	680	680	E
Propulsion System	15% (prop)	1073	740	C
Attitude Control	2%	151	88	C
Power	15%	1134	261	C
Avionics and Controls	10%	756	254	C
Environmental control and life support	8%	605	605	E
Crew Accommodations	8%	605	875	C
Landing	18%	1361	1361	E
Other	n/a	n/a	n/a	
Dry Mass w/o Prop		7561	9669	E
Dry Mass		**8634**	**10409**	**E**
Payload		3000	416	C
Inert Mass		**13793**	**10825**	**E**
Propellant Mass		7153	9745	C
Wet Mass		**20946**	**20570**	**E**

With the lack of a BalLunarm and the addition of the inflatable upper structure, the structural mass and the mechanism mass is the same as the LMs. The VIBES module contains the crew and galley system for crew accommodations in addition to some additional operational supplies and transportation bags. The crew and galley system is the large majority of the crew accommodations in this module, with that equipment alone being 728 kg. The VIBES module contains no excess ECLSS equipment but does contain the estimated 605 kg of pipes and pumps to be used for sinks and other kitchen equipment, in addition to pumping the water and air into the other modules. The VIBES module is equipped with 4 ISPRs and has the potential to carry up to 200 kg of scientific equipment. However, the ISPRs will be primarily used for storage of the crew and galley system, and as such volume may be an issue in assigning scientific equipment to this module for launch.

As shown, all 4 variants of the BalLunar architecture are below the maximum mass allowance for the Falcon Heavy Expendable, making our system compliant with the mass limit.

6.0 Conclusion

Lunar BOWIE's key design feature is its modular variants. The combined variants utilize the ECLSS and a variety of crew accommodations adapted from the ISS to ensure survivability and habitably of 4 crew members for 45 days and support them in their science and exploration missions. Careful consideration was taken to not only sustain crew physical health, but also their psychological well-being. Moreover, the Lunar BOWIE's modular design also allows for future missions that expand the overall base and can resupply it for larger crew housing and extended missions. Space Oddities also considered the advantages in versatility of a modular design for future base expansion. Although no detailed design is presented at this time, a Lunar BOWIE module has potential to be repurposed for other uses as a base component, such as a Cupola Module similar to the Cupola on the ISS. The initial mission will make use of ISRU experiments to help understand future utilization of sustainable lunar resources for extended crewed mission and supporting future deep space missions. However, the base is capable of accommodating a variety of scientific missions as needed. The establishment of the base at CR-1 near the Shackleton Crater will further support the use of ISRU and support future science. Modularity allows each base component to be sent in smaller masses using 5 SpaceX's Falcon Heavy System, but overall, allows more mass to be delivered at a fraction of the cost of the SLS. In addition, the Lunar BOWIE maximizes the use of current technology to further keep costs within the budget of $12B USD for FY 2019 and have the base fully operational by the 31st of December 2030.

6.1 Compliance Matrix

A compliance matrix is provided below in *Table 6.1-1* as a quick reference for verifying that all design constraints are met by the Space Oddities proposal of the Lunar BOWIE.

Table 6.1-1: Lunar BOWIE Compliance Matrix

Req ID	Requirement Statement	Rationale	Comply	Page # (s)
T0.0-1	Lunar Base Camp (LBC) shall be designed to support a crew of 4 for a nominal mission duration of 45 days on the lunar surface.	Crew life support system with mission lifetime assessment is included in LBC design	Yes	62-66,68, 69
T0.0-2	Location of LBC shall be chosen to maximize crew survivability, scientific research, and support for future deep space missions	Location is based on available sunlight, safe landing and high scientific interest.	Yes	27-30

T0.0-3	The LBC shall have the potential for the base camp to be expanded to accommodate more crew for longer duration in subsequent expeditions	Modular designs allow for expansion and resupply of Lunar Base Camp.	Yes	9-11, 16
T0.0-4	The design shall consider the various activities, resources, and systems that future exploration missions to other solar system destinations would require and how the base camp would help enable those missions	ISPRs allow adaptability for various science missions and available ISRU payload allows research for future use of lunar resources.	Yes	66-67
T0.0-5	The design shall include all the necessary systems to launch and deploy the base camp elements to the lunar surface	Extensive analysis and selection of trajectory, LV, propulsion systems, and ACS.	Yes	17,18, 27-43
T0.0-6	LBC shall detail the necessary tasks that are required to bring the base camp to operation to sustain the crew for the duration of the expedition	Autonomous rendezvous and docking are discussed in detail	Yes	15-23
T0.0-7	The crew lander shall only sustain the crew of 4 for a period no longer than 72 hours after landing; any payload capability that the crew lander has will be fully utilized to support the crew for the 72-hour period and for crew ascent. The crew lander and ascent stage is not part of the base camp design.	Autonomous assembly of the LBC prior to crew arrival will minimize time of transfer. The LBC will provide necessary supplies and systems for the entire expedition without utilizing the crew lander.	Yes	15-23
M0.0-1	Trade studies shall be performed on system options at the system and subsystem level to demonstrate the fitness of the chosen base camp design.	Functional analysis trade studies performed to verify system and subsystem performance	Yes	14, 28, 31, 34, 40
M0.0-2	Design shall maximize the use of technologies that are already demonstrated on previous programs or currently in the NASA technology development portfolio.	Preference will be given to technologies of the highest TRL levels	Yes	33, 38, 41, 47, 59, 62, 65
M0.0-3	Advanced technology use shall consider cost, schedule, and risk associated with the development.	Risk analysis of and cost breakdown of system will be provided.	Yes	69-74
M0.0-4	Design shall detail subsystem components, including mass, power, and volume, and how	Payload, volume, mass, and power statements included and	Yes	75-78

	the design requirements drove the selection of the subsystem	justified through requirements compliance		
M0.0-5	Design shall detail the estimated lifetime of each of the components, the lifetime of the system and number of surface expeditions the basecamp can sustain, and the potential upgrades/expansions that are available with the design and how extensibility and longevity considerations impacted the design choices	The LBC will be designed to end of life component specifications and fatigue life analysis of components presented	Yes	60, 61, 68, 69
M0.0-5	Design shall detail how the base camp components will be packaged, launched, and deployed to the lunar surface, whether any on-orbit or on-surface assembly or rendezvous of components will be required, and what systems would be required to assist in the delivery of the components to the lunar surface	A detailed comprehensive concept of operations including all necessary systems presented	Yes	15-26
C0.0-1	The initial cost for the lunar base camp shall not exceed $12 Billion US Dollar (in FY19) from the start of the program to the human expedition, including design development test and evaluation (DDT&E) and theoretical first unit (TFU) costs of all of the base camp elements.	Comprehensive cost estimation analysis will be performed cost of project management and mission operations.	Yes	74
C0.0-2	Technology advancement cost shall be included if advanced technology options are utilized in the design.	The design will minimize mass and the inclusion of costly undeveloped advanced technology.	Yes	74
C0.0-3	The cost cap includes launch costs to deploy the base camp systems, but does not include the cost of the human expedition mission and its associated lander/ascent stage	Comprehensive cost estimation analysis will be performed including LBC launch costs, minus human operations and lander/ascent stage costs	Yes	74
M0.0-7	The base camp shall be ready to receive the first expedition crew no later than December 31, 2030.	Complete manufacturing schedule and mission timelines presented for readiness before December 31, 2030	Yes	15, 68

References

[1] "Falcon User Guide." *SpaceX*, SpaceX, Jan. 2019, https://www.spacex.com/sites/spacex/files/falcon_users_guide_10_2019.pdf

[2] Jawin, E. R., Valencia, S. N., Watkins, R. N., Crowell, J. M., Neal, C. R., & Schmidt, G. (2019). "lunar science for landed missions workshop findings report," *Earth and Space Science*, 6, 2–40. URL: https://doi.org/10.1029/2018EA000490

[3] Speyerer, E. J., Lawrence, S. J., Stopar, J. D., Gläser, P., Robinson, M., & Jolliff, B. L. (2016). "Optimized traverse planning for future polar prospectors based on lunar topography," *Icarus*, 273, 337-345. URL: https://doi.org/10.1016/j.icarus .2016.03.011

[4] Gläser, Philipp. (2014). "Evaluation of topography, slopes, illumination and surface roughness of landing sites near the lunar south pole using laser altimetry from the lunar reconnaissance orbiter," Ph.D. Dissertation, Technical University of Berlin, Berlin, Nov. 2014, URL: http://dx.doi.org/10.14279/depositonce-4306

[5] Parker, Jeffrey S, Rodney L Anderson, and Andrew Peterson. *A Survey of Ballistic Transfers to Low lunar Orbit*. Pasadena, CA: Jet Propulsion Laboratory, NASA, 2011. Print.

[6] *Propulsion Products Catalogue*. Northrop Grumman, 2018.

[7] Hulbert, Eric. "Lunar Lander and Return Propulsion System Trade Study." *NASA-TP-3388*, National Aeronautics and Space Administration, Aug. 1993.

[8] Wertz, J. R., Puschell, J. J., & Everett, D. F. (2011). *Space mission engineering the new SMAD*. Microcosm Press.

[9] Robertson, E. A., & Carson, J. M. (2016). "PL&HA Domain, Precision Landing & Hazard Avoidance", NASA Technical Reports Server, URL: https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20170001719.pdf

[10] Carson, J. M., Sostaric, R. R., Pedrotty, S. M., Estes, J. N., Amzajerdian, F., Cianciolo, A. D., Blair, J. B., Restrepo, C. I., Rutishauser, D. K., Tse, T., Hines, G. D. (2019). "PL&HA and SPLICE Overview", NASA, 2019, URL: https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20190027507.pdf

[11] Pollard, B. (2012), "Radar Terminal Descent Sensor" *Radar Science and Engineering Friday Forum*, NASA JPL, Pasadena, California, 2012

[12] Brown, C. D. (2002). *Elements of spacecraft design*. Reston, VA: American Institute of Aeronautics and Astronautics.

[13] M. Pasand, A. Hassani and M. Ghorbani, "A study of spacecraft reaction thruster configurations for attitude control system," in *IEEE Aerospace and Electronic Systems Magazine*, vol. 32, no. 7, pp. 22-39, July 2017. doi: 10.1109/MAES.2017.160104

[14] Harris, Robert S. "APOLLO EXPERIENCE REPORT - THERMAL DESIGN OF APOLLO LUNAR SURFACE EXPERIMENTS PACKAGE." *NASA TN D-6738*, National Aeronautics and Space Administration, Mar. 1972.

[15] Badavi, Francis F, et al. "Validity of the Aluminum Equivalent Approximation in Space Radiation Shielding." *NASA/TP-2009-215779*, National Aeronautics and Space Administration, July 2009.

[16] "NASA Space Flight Human-System Standard Volume 1, Revision A: Crew Health." *NASA Technical Standard*, National Aeronautics and Space Administration, 30 July 2014.

[17] Cha, Ji-Hun, et al. "Ultra-High-Molecular-Weight Polyethylene as a Hypervelocity Impact Shielding Material for Space Structures." *Acta Astronautica*, vol. 168, Mar. 2020, pp. 182–190., doi:10.1016/j.actaastro.2019.12.008.

[18] Cucinotta, Francis A, et al. "Evaluating Shielding Approaches to Reduce Space Radiation Cancer Risks ." *NASA TM-2012-217361*, National Aeronautics and Space Administration, 1 June 2012.

[19] Gannon, Brian Bernard. "DESIGN AND ANALYSIS OF THE LAUNCH VEHICLE ADAPTER FITTING FOR THE PETITE AMATEUR NAVY SATELLITE (PANSAT*)"*. Naval Postgraduate School , 0AD, 1994.

[20] Lee, Joo Ahn, et al. "Lunar Lander Conceptual Design." *NASA-CR-186233*, National Aeronautics and Space Administration, 17 May 1989.

[21] Schaire, Scott H, and David L Carter. "Near Earth Network (NEN) Users' Guide." NASA, explorers.larc.nasa.gov/HPMIDEX/pdf_files/18_[Near_Earth_Network_(NEN)_Users'_Guide_Revision_4_Redacted_]453-UG-002905.pdf

[22] Gerstenmaier, William, and Jason Crusan. "Cislunar and Gateway Overview." NASA, www.nasa.gov/sites/default/files/atoms/files/cislunar-update-gerstenmaier-crusan-v5a.pdf.

[23] "Space Communication Architecture Working Group (SCAWG) ." Syzygy Engineering, 15 May 2006, syzygyengineering.com/2007/SCAWG_Report.pdf.

[24] Mahoney, Erin. "A Next Generation Spacesuit for the Artemis Generation of Astronauts." NASA, NASA, 4 Oct. 2019, www.nasa.gov/feature/a-next-generation-spacesuit-for-the-artemis-generation-of-astronauts.

[25] Art Azarbarzin. "MUSTANG Applications Modular Avionics" NASA Goddard Space Flight Center. 2019 IEEE Space Computing Conference Pasadena, CA. 2019

[26] Maxwell Technologies, Inc "SCS750 Super Computer For Space" Doc. # 1004741. Rev. 7. Maxwell Technologies, Inc

[27] NASA Goddard Space Flight Center. "core Flight System (cFS) Background and Overview" https://cfs.gsfc.nasa.gov/cFS-OviewBGSlideDeck-ExportControl-Final.pdf

[28] Rucker, Michelle A. "Surface Power for Mars" NASA Johnson Space Center. AIAA Paper 2016-5452. December 01, 2016.

[29] Lyons, V. J., Gonzolez, G. A., Houts, M. G., Iannello, C. J., Scott, J. H., and Surampudi, S., Space Power and Energy Storage Roadmap, NASA, April 2012

[30] Kennedy, L. D. "NASA Lunar Lander Reference Design". NASA Marshall Space Flight Center. NASA/TP-2019–220391. November 2019

[31] "Spacecraft Components." *Northrop Grumman*, 2020, www.northropgrumman.com/space/spacecraft-components/.

[32] "ZTJ Space Solar Cell DATASHEET APRIL 2018" 2018. SolAero Technologies Corp. https://solaerotech.com/wp-content/uploads/2018/04/ZTJ-Datasheet-Updated-2018-v.1.pdf

[33] "MA190 Modular Lithium Ion Battery For Satellites" Specification Sheet. GS Yuasa Lithium Power, Inc. http://www.gsyuasa-lp.com/SpecSheets/MA190.pdf

[34] Wurz, P. "Solarwind Composition" Proceedings of the 11th European Solar Physics Meeting "The Dynamic Sun: Challenges for Theory and Observations (ESA SP-600). September 2005, Leuven, Belgium.

[35] Bader, G. Stenberg Wieser. "Proton temperature anisotropies in the plasma environment of Venus". American Geophysical Union. 2018

[36] "Solar Cycle Progression." *Solar Cycle Progression | NOAA / NWS Space Weather Prediction Center*, www.swpc.noaa.gov/products/solar-cycle-progression.

[37] NASA STD-3001. "NASA Space Flight Human-System Standard, Volume 2, Revision B: Human Factors, Habitability, and Environmental Health" NASA-STD-3001, Volume 2. National Aeronautics and Space Administration: Washington, DC, 2019

[38] "NASA Space Flight Human-System Standard Volume 1 , Revision A: Crew Health" NASA-STD-3001, Volume 1, Revision A Change 1. National Aeronautics and Space Administration Washington, DC. 2015

[39] Jones, H., "Would Current International Space Station (ISS) Recycling Life Support Systems Save Mass on a Mars Transit?" ICES-2016-109, 46th International Conference on Environmental Systems, 10-14 July 2016, Vienna

[40] "NASA Procedural Requirements, NPR 8020.12D: Planetary Protection Provisions for Robotic Extraterrestrial Missions." *NASA*, NASA, 20 Apr. 2011, nodis3.gsfc.nasa.gov/npg_img/N_PR_8020_012D_/N_PR_8020_012D_.pdf.

[41] Larson, Wiley J., and Linda K. Pranke. *Human Spaceflight Mission Analysis and Design*. McGraw-Hill, 2000.

Appendices

WBS #	Level	WBS Element	DDT&E	Design & Development	System Test Hardware	Flight Unit	Production	Non-Allocated	Operations	Total
0	1	**BalLunar**	**$ 5,218.97**	**$ 4,996.65**	**$ 215.22**	**$ 188.70**	**$ 1,346.92**	**$ -**	**$ 4,161.58**	**$ 10,727.47**
1.0	2	Project Management		$ -	$ -	$ -	$ -	$ -	$ 485.22	$ 485.22
2.0	2	Systems Engineering		$ -	$ -	$ -	$ -	$ -	$ 731.59	$ 731.59
3.0	2	Safety and Mission Assurance		$ -	$ -	$ -	$ -	$ -	$ 219.48	$ 219.48
4.0	2	Science/Technology		$ -	$ -	$ -	$ -	$ -	$ 365.79	$ 365.79
5.0	2	Payload(s)								$ -
6.0	2	Flight System \ Spacecraft	$ 5,218.97	$ 4,996.65	$ 215.22	$ 188.70	$ 1,346.92	$ -	$ -	$ 6,565.89
6.01	3	Crewed Vehicle Management	$ 98.28	$ 98.28	$ -	$ 2.02	$ 10.11	$ -	$ -	$ 108.40
6.02	3	Crewed Vehicle Systems Engineering	$ 774.11	$ 774.11	$ -	$ 11.29	$ 56.45	$ -	$ -	$ 830.56
6.03	3	Crewed Vehicle Product Assurance	$ -	$ -	$ -	$ -	$ -	$ -	$ -	$ -
6.10	3	Crewed Vehicle	$ 2,903.68	$ 2,681.36	$ 215.22	$ 165.55	$ 1,231.18	$ -	$ -	$ 4,134.86
--	4	Thermal Protection	$ 444.81	$ 356.00	$ 88.81	$ 68.32	$ 341.58	$ -	$ -	$ 786.39
--	4	Mechanism	$ 109.18	$ 89.79	$ 19.39	$ 14.91	$ 74.56	$ -	$ -	$ 183.74
--	4	Structures	$ 170.04	$ 140.30	$ 29.74	$ 22.88	$ 114.39	$ -	$ -	$ 284.43
--	4	Propulsion	$ 5.03	$ -	$ -	$ -	$ 388.40	$ -	$ -	$ 393.43
--	4	Attitude Control	$ 2.07	$ -	$ -	$ -	$ 15.02	$ -	$ -	$ 17.09
--	4	Power	$ 224.38	$ 219.42	$ 4.97	$ 3.82	$ 19.10	$ -	$ -	$ 243.48
--	4	Avionics	$ 29.06	$ 12.15	$ 16.90	$ 13.00	$ 65.02	$ -	$ -	$ 94.07
--	4	Crew Systems	$ 1,097.98	$ 1,042.57	$ 55.41	$ 42.62	$ 213.11	$ -	$ -	$ 1,311.09
--	4	Software	$ 821.14	$ 821.14	$ -	$ -	$ -	$ -	$ -	$ 821.14
--	5	Flight Software	$ 547.42	$ 547.42	$ -	$ -	$ -	$ -	$ -	$ 547.42
--	5	Ground Software	$ 273.71	$ 273.71	$ -	$ -	$ -	$ -	$ -	$ 273.71
6.60	3	Integration, Assembly, Checkout	$ 178.53	$ 178.53	$ -	$ 9.84	$ 49.18	$ -	$ -	$ 227.70
6.70	3	System Test Operations	$ 89.51	$ 89.51	$ -	$ -	$ -	$ -	$ -	$ 89.51
6.80	3	Ground Segment	$ 1,174.86	$ 1,174.86	$ -	$ -	$ -	$ -	$ -	$ 1,174.86
6.80.01	4	Ground/Test Support Equip	$ 948.69	$ 948.69	$ -	$ -	$ -	$ -	$ -	$ 948.69
6.80.02	4	Tooling	$ 226.17	$ 226.17	$ -	$ -	$ -	$ -	$ -	$ 226.17
6.80.03	4	Facilities								$ -
6.80.04	4	Launch Operations								$ -
6.80.05	4	Flight Operations								$ -
7.0	2	Mission Operations System (MOS)		$ -	$ -	$ -	$ -	$ -	$ 365.79	$ 365.79
8.0	2	Launch Vehicle/Services							$ 750.00	$ 750.00
9.0	2	Ground Data System (GDS)		$ -	$ -	$ -	$ -	$ -	$ 731.59	$ 731.59
10.0	2	System Integration, Assembly, Test & Check Out		$ -	$ -	$ -	$ -	$ -	$ 512.11	$ 512.11
11.0	2	Education & Public Outreach		$ -	$ -	$ -	$ -	$ -	$ -	$ -

1st Quarter Moon					
Launch Window	1st Q.M. Date	TLI Date/Time	LOI Date/Time	Inc. Change Date/Time	DOI Date/Time
Jun 2029	06/19/2029 09:54	06/15/2029 06:10:48	06/19/2029 09:54:00	06/19/2029 10:05:00	06/19/2029 10:34:26
Jul 2029	07/18/2029 07:14	07/14/2029 03:30:48	07/18/2029 07:14:00	07/18/2029 07:25:00	07/18/2029 07:54:26
Aug 2029	08/16/2029 11:55	08/12/2029 08:11:48	08/16/2029 11:55:00	08/16/2029 12:06:00	08/16/2029 12:35:26
Sep 2029	09/14/2029 18:29	09/10/2029 14:45:48	09/14/2029 18:29:00	09/14/2029 18:40:00	09/14/2029 19:09:26
Oct 2029	10/14/2029 04:09	10/10/2029 00:25:48	10/14/2029 04:09:00	10/14/2029 04:20:00	10/14/2029 04:49:26
Nov 2029	11/12/2029 16:35	11/08/2029 12:51:48	11/12/2029 16:35:00	11/12/2029 16:46:00	11/12/2029 17:15:26
Dec 2029	12/12/2029 09:49	12/08/2029 06:05:48	12/12/2029 09:49:00	12/12/2029 10:00:00	12/12/2029 10:29:26
Jan 2030	01/11/2030 06:06	01/07/2030 02:22:48	01/11/2030 06:06:00	01/11/2030 06:17:00	01/11/2030 06:46:26
Feb 2030	02/10/2030 03:49	02/06/2030 00:05:48	02/10/2030 03:49:00	02/10/2030 04:00:00	02/10/2030 04:29:26
Mar 2030	03/12/2030 01:48	03/07/2030 22:04:48	03/12/2030 01:48:00	03/12/2030 01:59:00	03/12/2030 02:28:26
April 2030	04/10/2030 19:57	04/06/2030 16:13:48	04/10/2030 19:57:00	04/10/2030 20:08:00	04/10/2030 20:37:26
May 2030	05/10/2030 10:11	05/06/2030 06:27:48	05/10/2030 10:11:00	05/10/2030 10:22:00	05/10/2030 10:51:26
June 2030	06/08/2030 20:36	06/04/2030 16:52:48	06/08/2030 20:36:00	06/08/2030 20:47:00	06/08/2030 21:16:26
July 2030	07/08/2030 04:02	07/04/2030 00:18:48	07/08/2030 04:02:00	07/08/2030 04:13:00	07/08/2030 04:42:26
Aug 2030	08/06/2030 09:43	08/02/2030 05:59:48	08/06/2030 09:43:00	08/06/2030 09:54:00	08/06/2030 10:23:26
Sep 2030	09/04/2030 14:55	08/31/2030 11:11:48	09/04/2030 14:55:00	09/04/2030 15:06:00	09/04/2030 15:35:26
Oct 2030	10/03/2030 20:56	09/29/2030 17:12:48	10/03/2030 20:56:00	10/03/2030 21:07:00	10/03/2030 21:36:26
Nov 2030	11/02/2030 03:56	10/29/2030 00:12:48	11/02/2030 03:56:00	11/02/2030 04:07:00	11/02/2030 04:36:26
Dec 2030	12/01/2030 14:57	11/27/2030 11:13:48	12/01/2030 14:57:00	12/01/2030 15:08:00	12/01/2030 15:37:26

3rd Quarter Moon					
Launch Window	3rd Q.M. Date	TLI Date/Time	LOI Date/Time	Inc. Change Date/Time	DOI Date/Time
Jun 2029	06/03/2029 18:19	05/30/2029 14:35:48	06/03/2029 18:19:00	06/03/2029 18:30:00	06/03/2029 18:59:26
Jul 2029	07/03/2029 10:58	06/29/2029 07:14:48	07/03/2029 10:58:00	07/03/2029 11:09:00	07/03/2029 11:38:26
Aug 2029	08/02/2029 04:26	07/29/2029 00:42:48	08/02/2029 04:26:00	08/02/2029 04:37:00	08/02/2029 05:06:26
Sep 2029	09/30/2029 13:57	09/26/2029 10:13:48	09/30/2029 13:57:00	09/30/2029 14:08:00	09/30/2029 14:37:26
Oct 2029	10/30/2029 03:32	10/25/2029 23:48:48	10/30/2029 03:32:00	10/30/2029 03:43:00	10/30/2029 04:12:26
Nov 2029	11/28/2029 15:48	11/24/2029 12:04:48	11/28/2029 15:48:00	11/28/2029 15:59:00	11/28/2029 16:28:26
Dec 2029	12/28/2029 01:49	12/23/2029 22:05:48	12/28/2029 01:49:00	12/28/2029 02:00:00	12/28/2029 02:29:26
Jan 2030	01/26/2030 10:15	01/22/2030 06:31:48	01/26/2030 10:15:00	01/26/2030 10:26:00	01/26/2030 10:55:26
Feb 2030	02/24/2030 17:58	02/20/2030 14:14:48	02/24/2030 17:58:00	02/24/2030 18:09:00	02/24/2030 18:38:26
Mar 2030	03/26/2030 02:51	03/21/2030 23:07:48	03/26/2030 02:51:00	03/26/2030 03:02:00	03/26/2030 03:31:26
April 2030	04/24/2030 11:39	04/20/2030 07:55:48	04/24/2030 11:39:00	04/24/2030 11:50:00	04/24/2030 12:19:26
May 2030	05/23/2030 21:57	05/19/2030 18:13:48	05/23/2030 21:57:00	05/23/2030 22:08:00	05/23/2030 22:37:26
June 2030	06/22/2030 10:20	06/18/2030 06:36:48	06/22/2030 10:20:00	06/22/2030 10:31:00	06/22/2030 11:00:26
July 2030	07/22/2030 01:07	07/17/2030 21:23:48	07/22/2030 01:07:00	07/22/2030 01:18:00	07/22/2030 01:47:26
Aug 2030	08/20/2030 19:15	08/16/2030 15:31:48	08/20/2030 19:15:00	08/20/2030 19:26:00	08/20/2030 19:55:26
Sep 2030	09/19/2030 12:56	09/15/2030 09:12:48	09/19/2030 12:56:00	09/19/2030 13:07:00	09/19/2030 13:36:26
Oct 2030	10/26/2030 13:17	10/22/2030 09:33:48	10/26/2030 13:17:00	10/26/2030 13:28:00	10/26/2030 13:57:26
Nov 2030	11/18/2030 00:32	11/13/2030 20:48:48	11/18/2030 00:32:00	11/18/2030 00:43:00	11/18/2030 01:12:26
Dec 2030	12/17/2030 16:01	12/13/2030 12:17:48	12/17/2030 16:01:00	12/17/2030 16:12:00	12/17/2030 16:41:26