Project CAVEMAN:

***CA**RGO **VE**HICLE AND **MAN**NED TRANSPORT*

AIAA 2018-2019 Undergraduate Team Space Systems Design Competition

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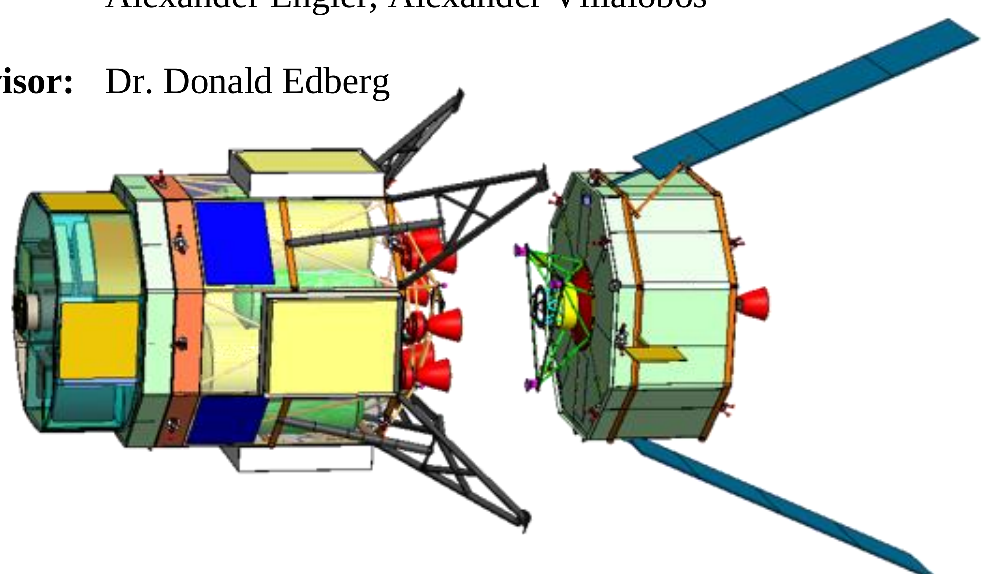

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List of Acronyms

CAVEMAN	Cargo Vehicle and Manned Transport
DSG	Deep Space Gateway
NRHO	Near Rectilinear Halo Orbit
ISS	International Space Station
LLO	Low Lunar Orbit
BLuR	Big Lunar Rocket
OM	Orbital Module
LM	Landing Module
FOV	Field of View
ECLSS	Environmental Control and Life Support System
CLUB	Cargo Loading and Unloading Bot
MLI	Multi-Layer Insulation
P&ID	Piping and Instrumentation diagram
LEO	Low Earth Orbit
NICS	Non-Interlayer-Contact Spacer
C&DH	Command and Data Handling
ACS	Attitude & Control System
KSC	Kennedy Space Center
SLS	Space Launch System

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Executive Summary

With the space industry inching closer and closer to extending mankind's reach beyond Earth, the Moon has become a staging point for future deep space missions. So much so, NASA has been collaborating with industry partners to develop what is known as the Deep Space Gateway (DSG) in a Near Rectilinear Halo Orbit (NRHO) around the Moon. This orbit was chosen by NASA for its various benefits that are not within the scope of this paper. The DSG will be similar to the International Space Station (ISS) but with its positioning in the space around the Moon, it provides a great environment for testing deep space systems. Alongside this, the DSG will grant mankind a "home-base" for lunar exploration that will provide astronauts and engineers with experience in extraterrestrial exploration. However, since the DSG is to remain in orbit about the Moon, the necessity for a vehicle to deliver enough cargo and/or crew becomes apparent. This objective of this paper is to deliver a proposal for such a vehicle.

Requirements for this vehicle were provided by the American Institute of Aeronautics and Astronautics. The main system requirements are the capability of taking 4 crew members and/or 15,000 kg of cargo from the DSG to anywhere on the lunar surface and back. The system must make multiple trips to and from the Moon utilizing the DSG as a propellant refill station. This system cannot cost more than $10B and must make its maiden voyage from the DSG to the Moon's surface by December 31, 2028. The full list of system level requirements can be seen in Section 1.4.0.

Various architectures were explored prior to down select to our current architecture. The main focus of the trade study between these architectures, which can be seen in Section III, was the total mass of propellant necessary for a single mission. This drove the design from a single stage "Big Lunar Rocket (BLuR)" to the current design of a three-module system deemed the CAVEMAN Transport which aims to minimize the amount of mass taken down to the surface, thereby reducing propellant consumption overall. An image of the three-module system follows. Note that cargo is carried by the LM within 3 of the 4 bays located radially on the exterior of the module.

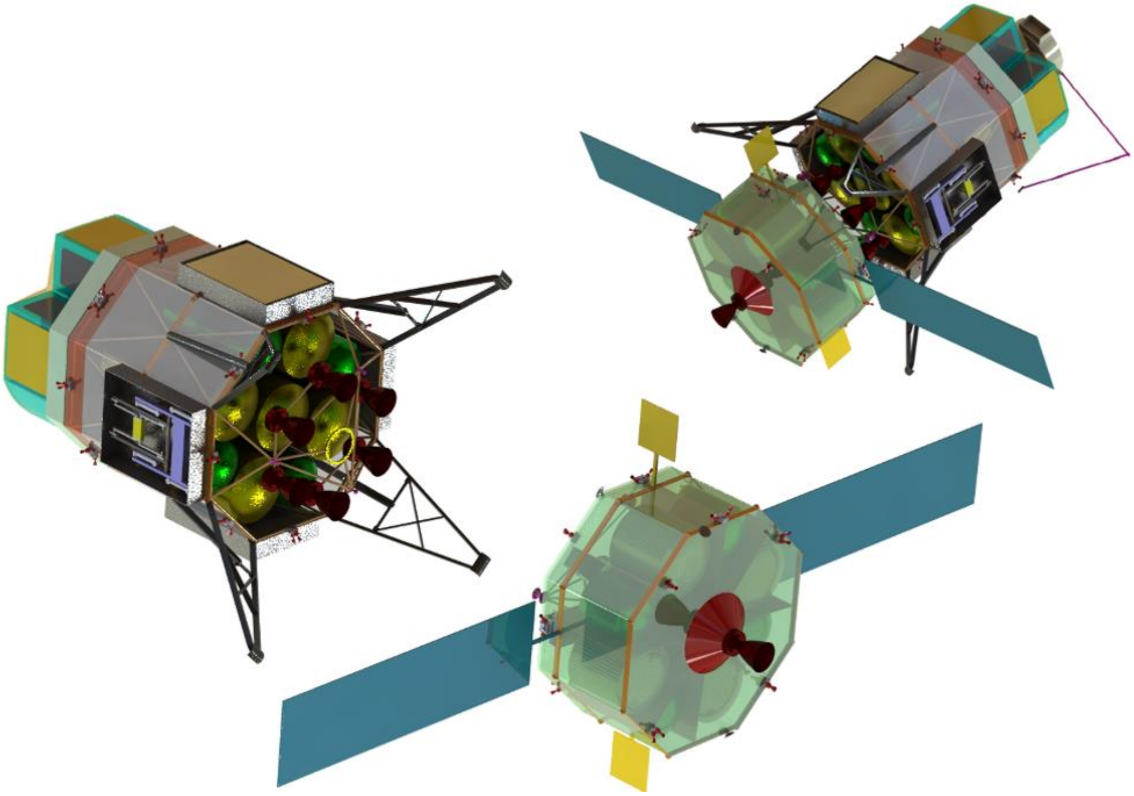


Figure ExSum-1: CAVEMAN Transport, In-Flight and Docked Configurations

The modules are the Inhabited Module (Hab), the Landing Module (LM), and the Orbital Module (OM). The Hab remains at the DSG for cargo missions, thereby saving approximately 5 tons on its trip. This, coupled with the lack of cargo requirement during crew missions, allowed for some creativity during mission design. Two different trajectories, one for cargo and one for crew, were implemented and can be seen in detail in Section IV. The two trajectories, calculated through a custom 3-Body Problem MATLAB Script, can be seen side-by-side below.

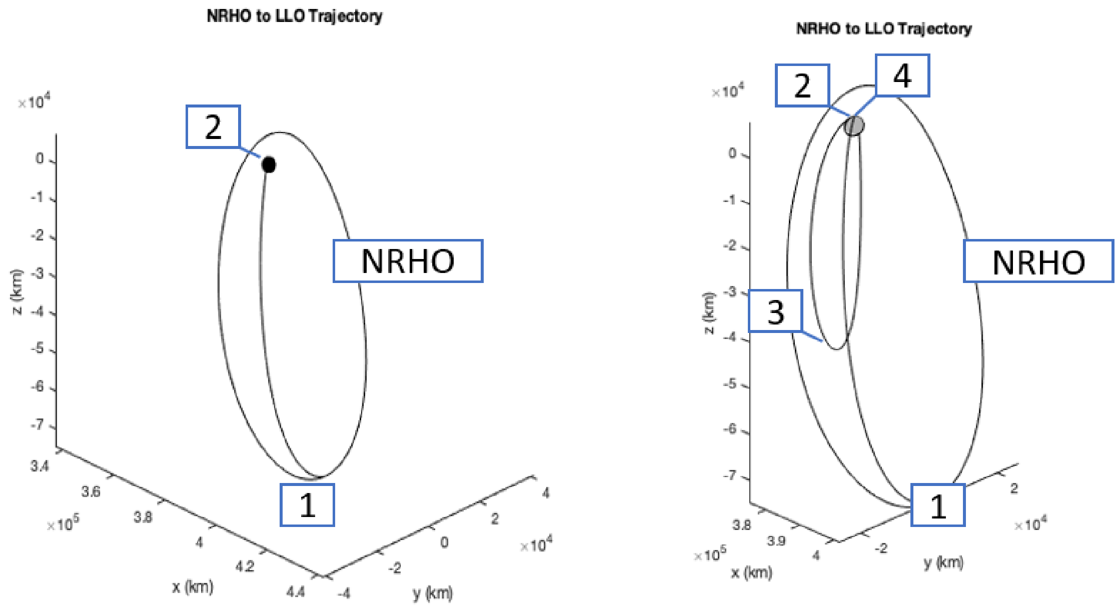


Figure ExSum-2: Cargo (Right) and Crew (Left) Mission Trajectories

The cargo mission's trajectory was the driver in terms of mission ΔV used to size the propulsion system. This trajectory found a burn necessary for the spacecraft to reach a low lunar altitude [1], then burned again to circularize the orbit [2]. The system now waits for the Moon to rotate until the desired landing site is beneath and can then begin descent. From there, the OM and LM can separate and the LM, filled with cargo and void of the Hab, performs its 1 Earth-G descent burn. The system then unloads its cargo utilizing what is essentially a lunar forklift, deemed the Cargo Loading and Unloading Bot (CLUB), tethered to the system and stored in one of the 4 cargo bays located on the outside of the LM. Once surface operations have concluded, in less than 24 hours, the OM performs a minor plane change to account for the rotation of the Moon, and the LM begins its ascent with the CLUB and different, slightly less massive cargo. The two modules rendezvous and the OM then performs the burn necessary to take the CAVEMAN back to the DSG. The crew mission's trajectory is similar to the cargo mission, except the mass savings supplies the system with extra ΔV which is used to shorten the mission duration by modifying and adding maneuvers at [2], [3], and [4] in the above image to the right. Burn [2] now moves the orbit into an elliptical orbit with a high altitude apoapsis, allowing for a plane change burn [3] to occur before coming down to periapsis and burning again [4] to circularize as necessary. A full concept of operations can also be seen in Section IV.

The lunar environment was examined for its effects on the mission. Of those effects, the main issues focused on in this paper are the ionizing radiation which would prove detrimental to the CAVEMAN's crew, the thermal environment which entails two very different temperature scenarios depending on the exposure to sunlight, and lunar dust which proved to be a problem in the Apollo missions. Details on the lunar environment that influenced this proposal are found in Section III.

In order to combat the effects of the lunar environment, the Hab is equipped with a full Environmental Control and Life Support System (ECLSS) and Environmental Protection System (EPS). The ECLSS, found in Section 5.2.2, control the interior of the Hab's crew chambers with various water management, air management, and fire suppression components. The EPS passively guards the crew against the extreme temperatures and harsh radiation of the environment with the use of radiating louvers and various layers and coatings. Details on the Hab's EPS can be found in Section 5.2.5.

As stated previously, the propulsion system design was based off the ΔV required by the cargo mission's trajectory. Alongside this, the necessity for restart and variable thrust capability, and a 1-G descent drove the design to its current configuration. The propulsion system utilizes LOX/LH2 as propellants for both the LM and OM. This propellant combination was decided on after a trade study that weighted total propellant mass and the ability to use fuel cells for power highly. From here, the propellant mass calculated through boil-off, fuel cell necessity, and margin were added to the total mass of propellant. The choice of engine came down to the range of thrust that could be supplied by a single engine. This led to the use of the RL-10C for its 6% deep throttle capability and max thrust of 101.3 kN. This engine's operational O/F ratio was used to split the mass and volume of the two propellants, then the cross-sectional packing of the tanks was mapped out and can be seen in the following images. The tanks were then sized based off those volumes and the radii which the tank packing produced. Details on the propulsion system design of the LM and OM can be found in Sections 5.3.1 and 5.4.1 respectively.

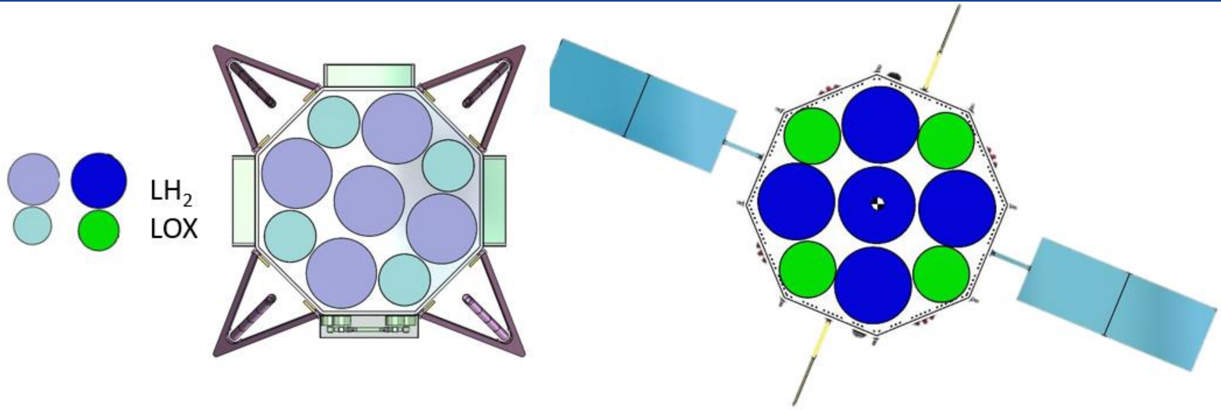


Figure ExSum-3: Tank Packing for LM (Left) and OM (Right)

The power system utilizes batteries for all three modules, but charges those batteries in different ways. The OM uses solar panels since it remains in orbit for the entire mission, while the LM was granted the capability of utilizing fuel cells thanks to the previously mentioned propellant selection. This allows the system to land on the dark side of the moon and still continue to generate electricity. The Hab cannot generate electricity as instead powered by the other modules throughout its mission. Details on the power subsystem can be seen in Sections 4.1.1, 4.2.4, 4.3.8, and 4.4.8.

A preliminary schedule can be found in Section 6.6. This schedule would place our first cargo mission departure from the DSG on December 01, 2028. The crew mission would launch a few months later so long as the cargo mission is successful.

Cost estimations for the system were done using two cost models, both NASA's Project Cost Estimating Capability (PCEC) and the Human Spaceflight: Mission Analysis and Design (SMAD) model. The two cost models estimated $8.7B and $9.3B respectively. A breakdown of this cost estimate can be seen in Section 6.7.

The entirety of the past school year has been dedicated to the development and refinement of the CAVEMAN Transport's design in hopes of satisfying all requirements. A compliance matrix explaining how the requirements have been addressed can be found in Section 6.8. The following proposal aims to capture all the effort poured into this vehicle throughout the course of its design. We thank the reviewers for their time spent examining our work.

I. Mission Overview

1.0 RFP Background

The American Institute for Aeronautics and Astronautics (AIAA) sent out a Request for Proposal (RFP) which addresses the imminent presence of the Deep Space Gateway (DSG) in lunar orbit. The DSG (currently known as the Lunar Orbital Platform Gateway but referred to as the DSG throughout this paper) will serve as a means of access to the lunar surface and deep space by stationing itself in a Near Rectilinear Halo Orbit (NRHO) around the Moon, acting as a “staging point” for missions to Mars and beyond. The RFP then provides requirements for a Reusable Lunar Surface Access Vehicle (RLSAV) to serve as a delivery system between the DSG and the moon for potential utilization of the celestial body’s resources and the establishment of a lunar base.

2.0 Needs Analysis

The proposed vehicle serves as a stepping stone for mankind’s leap into deep space exploration. This vehicle has the potential to serve NASA by demonstrating the capability of deep space transportation, modernizing interplanetary landers, and establishing the feasibility of cryogenics in space beyond Earth’s orbit. In addition to the benefits towards NASA, the RLSAV would promote deep space exploration to the next generation of aspiring engineers across the world, leading to innovation within the aerospace industry and development of technology for the advancement of civilization. It is also of interest for the American government to back such a project in order to help maintain a technical aptitude across the national space industry. Also, with the future of the space industry leaning towards the colonization of celestial bodies, this program would allow the United States to cement their position in becoming one of the first nations to colonize our closest neighbor.

3.0 Scope

The RFP provided by AIAA entails not only the requirements of the mission which can be seen in the section that follows, but also delivery of certain information detailing how objectives of the mission are met. This proposal will layout the design of the Caveman Transport which includes mission design, life support, structures, propulsion, thermal control, attitude control, telecommunication, command and data handling, and electrical power. This proposal also takes economics and schedule into account in order to meet the objectives of the RFP. Alongside the physical design, timeline, and economics of the system, an operational procedure concept is provided which outlines the entirety of the mission from launch through the system’s first mission.

4.0 System Requirements

The RFP's requirements for the system are summarized in Table 1.4.0-1 below.

Table 1.4.0-1: System Level Requirements

REQ #	REQ Statement
T 0.1	The system shall deliver crew and/or cargo to **anywhere** on the surface of the Moon from the Deep Space Gateway.
T 0.2	The system shall operate in crew or cargo delivery mode • Crew Mode: **4 astronauts, 24-hour** life support on surface • Cargo Mode: **15,000 kg** to surface, **10,000 kg** from surface
T 0.3	Vehicle shall make **multiple trips** to and from the DSG utilizing propellant refill.
P 0.1	Cost shall not exceed **$10B** (FY17) • Includes launch cost, design development test and evaluation (DDT&E), and theoretical first unit (TFU)
P 0.2	The vehicle shall make its first trip from the Deep Space Gateway to the lunar surface no later than **December 31, 2028**

II. Architecture Down Select

Starting from the most basic system and working towards complexity as infeasibilities crept in, the proposed system started as a single stage to orbit vehicle and slowly became a module system leaving assets in Low Lunar Orbit (LLO) or at the DSG. The different descriptions can be found in the table below. Architecture 4, the Three Module Fully Reusable concept was down selected due to low mass and full reusability with cryogenic propellants.

Table 2-1: Architecture Down Select

N	Desc.	Dry Mass (mT)	Wet Mass (mT)	Reusable Hardware
1	Single Stage to Orbit	45.5	190	Yes
2	Disposable	33.5	120	No
3	Orbital Asset	24.5	94	Yes
4	3 Module Fully Reusable	19	80	Yes

1.0 Single Stage to Orbit:

The first mission profile analyzed was a “single stage to orbit” vehicle that was capable of the entirety of cargo operations and crewed operations. This system essentially represented the least complexity and was a good baseline comparison for future designs. After doing a literature search of available or proposed vehicles, it became instantly clear that the currently proposed systems were incapable of being integrated with the 15-ton payload requirement. One specific case that was startling was the massive Lockheed Martin design with a very minimal cargo requirement.

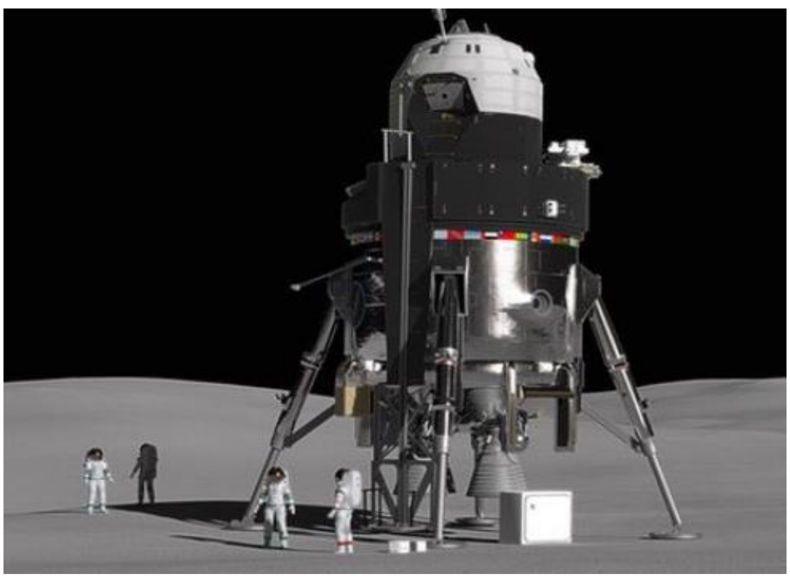


Figure 2.1.0-1: Lockheed Martin Single Stage to Orbit Lunar Lander (Credit: Lockheed Martin)

2.0 Disposable:

Taking a page out of Shuttle's book, we decided to look at strap on boosters that would assist in the large thrust requirement. This not exactly high Isp was offset but the simplicity of solid systems and the ability to completely jettison the associated mass. Although it moved the needle in the direction of feasibility, it was deemed too much of an un-reusable technology. Without having more concreate requirements surrounding the reusability of hardware the path of least regret became shelfing it for more reusable technology.

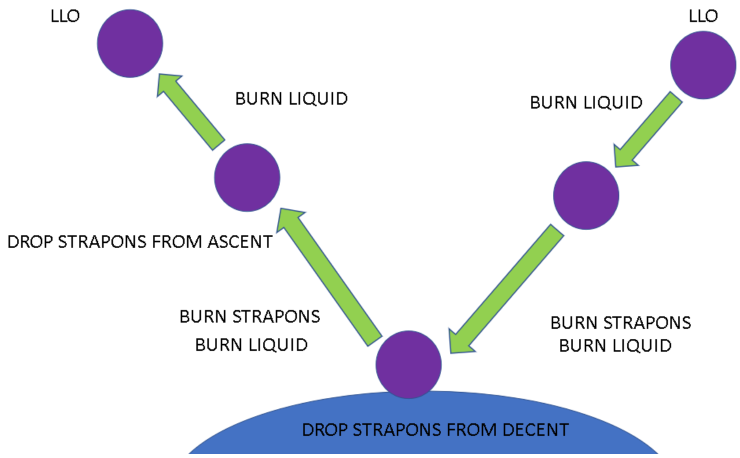


Figure 2.2.0-1: ConOps for Solid Booster Strap-ons

3.0 Orbital Asset:

Seen as a direct successor of the Apollo architecture and a natural evolution from the previously mentioned single stage to orbit vehicle, our double module vehicle left an asset in space as a "lunar tug". This vehicle was capable of cargo and crewed operations with the same vehicle, meaning that the crewed habitat was brought to the surface even when doing cargo missions.

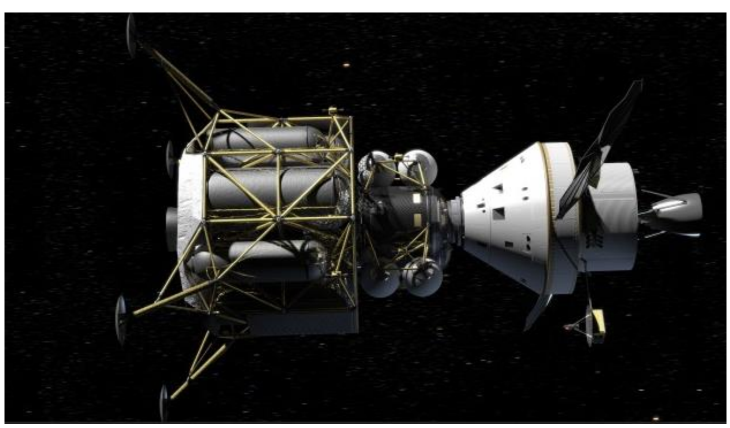


Figure 2.3.0-1: Lunar Tug Docking (Credit: NASA)

4.0 Three Module Fully Reusable:

Expanding on the good ideas presented within the Double Module, the Multi Module attempted to separate the crewed habitat from the nominally landed vehicle to save on cargo mission mass. It was quickly determined that if this approach was taken then, as long as the Hab stayed under 10 tons, the vehicle would be theoretically capable of bringing it to and from the lunar surface.

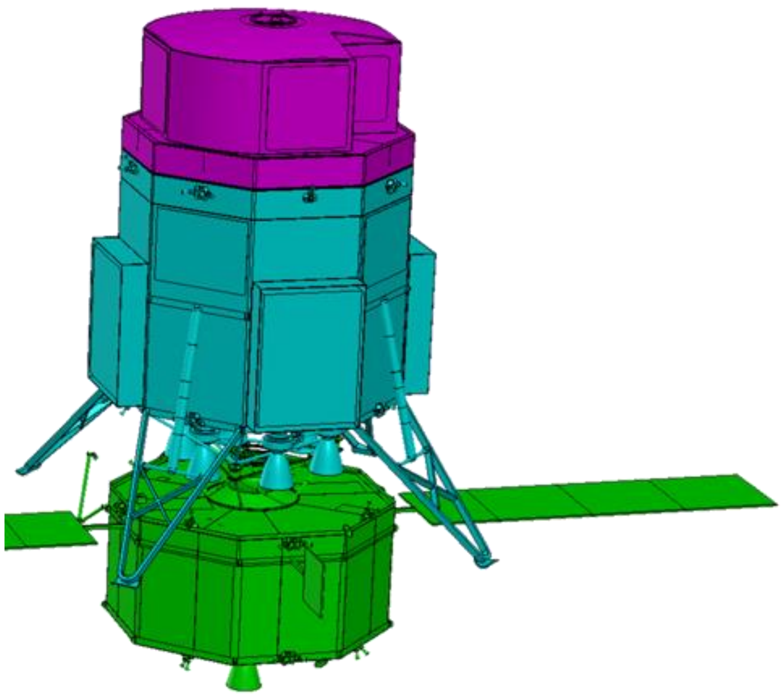


Figure 2.4.0-1: 3 Module Vehicle Designed by Lunacy Solutions

III. Lunar Environment

The entire operational lifetime of the vehicle will be spent in the lunar environment since it will only be making trip to and from the lunar surface. To design a reusable vehicle requires that we address all the major issues that arise when you are operating a spacecraft in this harsh environment. The moon is significantly less massive than Earth which means there is less gravity allow for a spacecraft to use less energy to escape from it. This lack of gravity also means it has virtually no atmosphere causing it to have very extreme temperatures. Unlike the Earth, the Moon lacks a magnetic field which protects those inside from exposure to large doses of radiation found in space.

1.0 Temperature Concerns

The lunar surface temperatures are extreme and range from boiling hot to freezing cold depending on its exposure to sunlight. This is caused by the lack of an atmosphere on the moon which would usually help insulate the surface. A spot on the moon typically has 13 and 1/2 days of sun, then that is followed by 13 days of night. Therefore, if we land somewhere on the lunar surface, the temperature will most likely remain the same by the time we depart back to the DSG. Even with these temperatures, it still presents the difficult problem of keeping both the crew, cargo, and propellants in their design temperatures. The orbital module will be in a polar orbit above the lunar surface, so it will be cycling through periods of sunlight and darkness. A depiction of the temperature cycle on the moon can be seen below. Note that one lunar day is equal to approximately 27 earth days.

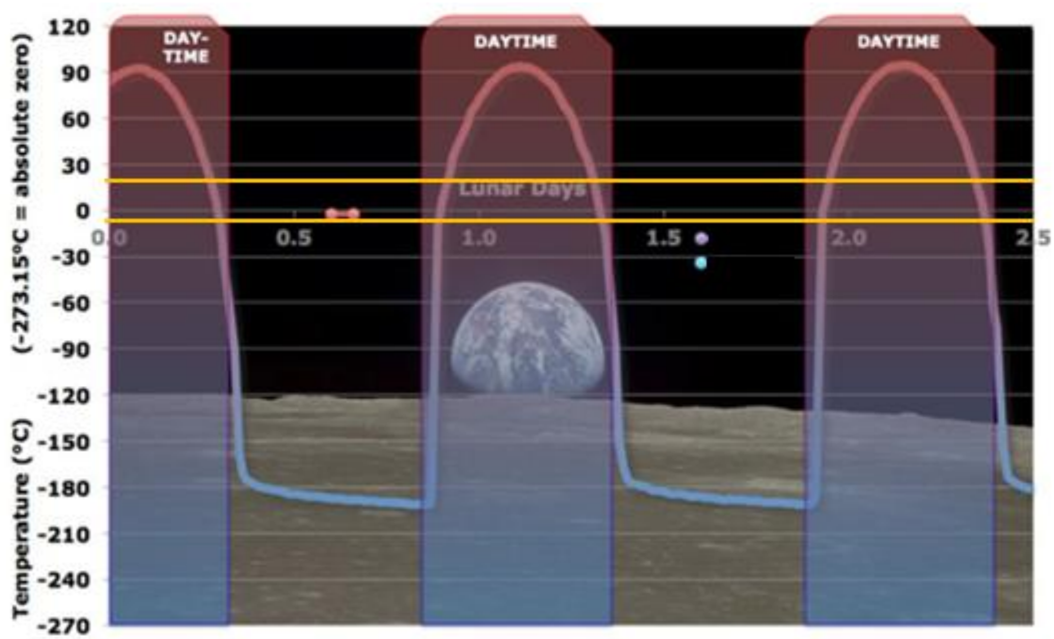


Figure 3.1.0-1: Lunar Surface Temperature Ranges

2.0 Ionizing radiation

Since we are handling crew during select missions, we must keep them alive and healthy during the entire duration. The exact details of the DSG are unknown but it should be safe to assume that they have designed it to protect the astronauts between missions. Due to the system's short mission duration, the decision was made not to design around Galactic Cosmic Radiation (GCR) because it should not pose a large risk to the astronaut's health. Our main concerns for radiation exposure source are solar winds and solar flares which have been well documented by NASA. As seen below in Figure 3.2-1 there are spikes in solar activity every 10 years with peaking at the start of the decade and reaching a low point at roughly the half decade. Our team will not completely ignore GCRs, but rather allow for a factor of safety in order to provide ample protection to the crew in case of there being heightened solar activity or a heavy bombardment of GCRs to the spacecraft. To meet this requirement, we will need extra radiation protection on our Hab module along with the structure to provided adequate protection.

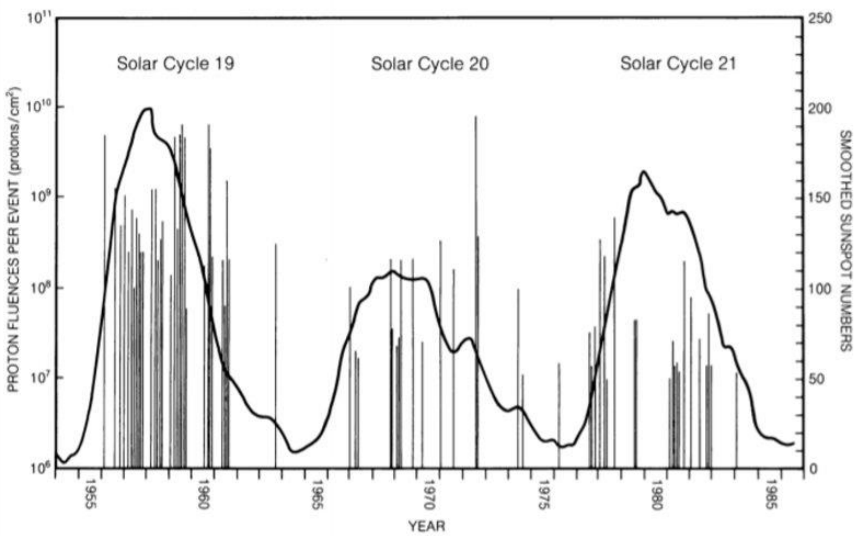


Figure 3.2.0-1: Solar Cycle

3.0 Lunar Surface

With the moon being bombarded by meteorites, there are numerous craters raging in size and rock composition. The spacecraft will need to be equipped with robust landing legs to allow for landing anywhere on the lunar surface with little damage to any important components of the spacecraft. With these deep craters means that during landing if we land on a slope our spacecraft could slide down either way and must be able to do so without damaging the landing legs.

The lunar surface is covered by a layer of reactive dust which can contaminate any surface that encounters the lunar surface. The issue this causes is when landing on the lunar surface the spacecraft can be contaminated through the dust that is stirred up by the landing burn. This dust is very fine with most of it been less than 0.02 mm in size which makes it very difficult to filter and remove from surfaces. These particles once stuck to a surface can cause adverse effects to the thermal properties, scratch the surface, or block the surface leading to a decrease in effectiveness the surface can provide. The lunar dust is so fine that if it gets ingested by an astronaut there is very serious health concerns that can arise with a direct exposure to the particles.

4.0 Minor Concerns

4.1 Mascons

The lunar surface contains many gravitational anomalies across its surface that can tug satellites in low orbits out of their desired orbit. These small perturbations in an orbit can cause them to complete distort the orbit that it could lead to a collision with the Moon. Our short mission duration doesn't allow for these small forces to have a negligible effect on our orbit allowing us to orbit at which ever inclination we desire.

4.2 Torques

4.2.1 Solar Torques

Any object in view of the Sun will experience some forces as the light particles reflect off the surface and for larger spacecraft that can be a real issue since it can push a circular orbit into more of an elliptical orbit. For any mission lasting less than a month these forces have a very small effect on the orbit and should not cause any design decisions.

4.2.2 Gravitational Torques

With the Moon being significantly small and less dense than the Earth the total pulling force of gravity is almost negligible. Since our spacecraft will be in orbit around the Moon for less than a month this constant tugging towards the surface is negligible and will not drive any trajectory design decisions.

IV. Mission Design

As described by the architecture down selection, Lunacy Solutions has decided to move forward with a modular design with a detachable habitat or Hab for use in crewed missions, a Landing Module (LM), and an Orbital Module (OM) that remains in a polar Low Lunar Orbit (LLO) for the duration of the operations on the lunar surface. During cargo missions, the Hab will remain attached to the Deep Space Gateway while the LM and OM ferry the cargo payload to the lunar surface and back. Alternatively, during crewed missions all three modules will travel to the lunar surface and back with the astronauts located in and supported by the Hab. These configurations are described in greater detail in Figure 4-1

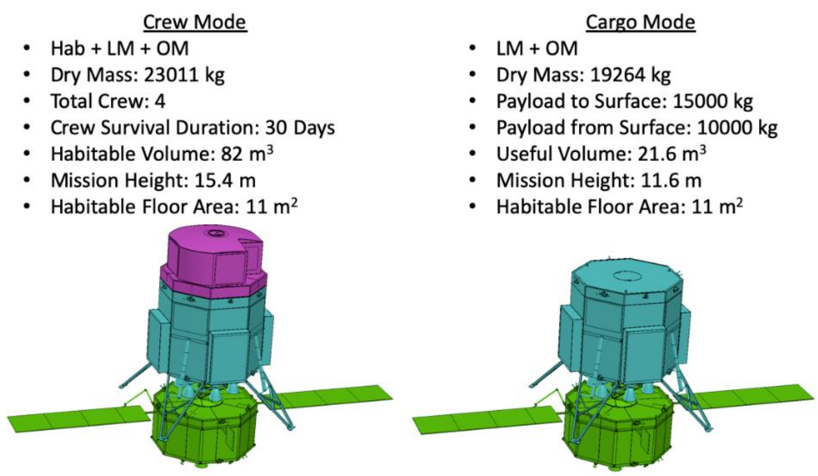


Figure 4-1: Mission Modes for Crewed or Un-Crewed Vehicle Configurations

The structure of the mission design for both the crewed and cargo missions is derived to satisfy the high-level requirements presented in the RFP. These explicit and derived requirements are presented in Table 4-1. These requirements present the framework, from which the mission can be designed.

Table 4-1: Mission Design Derived Requirements

REQ #	REQ Statement
2.1-1	Mission design shall enable a 24-hour lunar surface operation
2.1-2	Polar Low Lunar orbit will be utilized to maximize lunar surface access
2.1-3	Crewed missions shall employ a plane change to maximize surface access while reducing mission duration
2.1-4	Optical sensors will be employed to determine viable landing sites and avoid obstacles
2.1-5	Both crew and cargo total mission duration shall be synodically resonant with DSG
2.1-6	Maneuver ΔVs shall be minimized to reduce propellant mass

1.0 Cargo Configuration Mission Design

The decision was made to develop separate mission designs for our crewed and cargo missions. This decision was driven by the fact that the crewed mission timelines should be minimized as much as possible to decrease the risk the crew would be exposed to in terms of environmental hazards, mechanical failures, and over-burdening the Environmental Control and Life Support Systems (ECLSS). To that effect it was determined that missions conducted in the cargo configuration had no such time restrictions and therefore the mission could be designed purely to minimize the maneuver ΔVs, partly at the expense of mission time. Shown graphically in Figure 4.5.0-1, a nominal cargo mission concept of operations is as follows. Initially the CAVEMAN begins the mission docked with the Deep Space Gateway (DSG) in a 11:3 southern L2 Near Rectilinear Halo Orbit (NRHO). At the apoapse of the NRHO the CAVEMAN separates from the DSG and performs a ΔV, supplied by the Orbital Module (OM) to initiate a transfer to a polar Low Lunar Orbit (LLO). At the periapse of the transfer trajectory the OM section of the CAVEMAN supplies another ΔV to circularize its orbit around the moon at a height of roughly 100 km. The polar LLO takes advantage of the fact that the moon is tidally locked because over the course of a lunar period, the moon effectively rotates beneath the polar LLO, allowing the CAVEMAN to access all of the lunar surface. Once the polar orbit aligns with the landing site, the OM and the Landing Module (LM) separate and the LM begins its landing ΔV, leaving the OM in its polar LLO. Once the LM lands it spends at least 24 hours on the surface, primarily robotically unloading cargo. When the surface operations are complete, the LM takes off from the lunar surface and maneuvers into a polar orbit. By this point the OM will have performed a plane change to align with the orbit of the LM, allowing the OM and LM to rendezvous and dock. At this point the complete CAVEMAN will transfer from the polar LLO back to the NRHO to rendezvous and dock with the DSG, completing a nominal mission in the cargo configuration. The discrete ΔVs and time of flights are described in greater detail in Table 4.1.0-1.

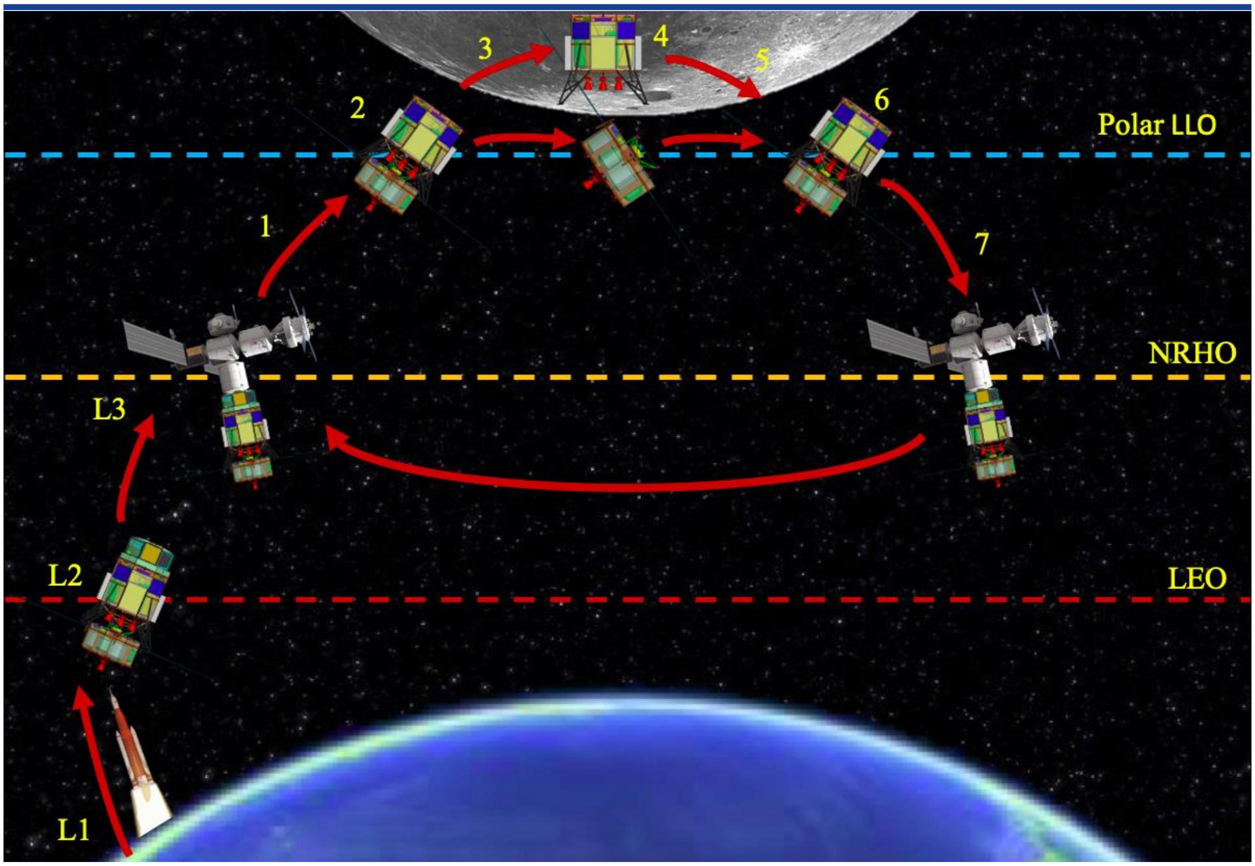


Figure 4.1.0-1: Cargo Configuration Concept of Operations

The ΔVs and times of flight depicted in Table 4.1.0-1 inform the design of the propulsion system along with the vehicle mass and payload mass requirement.

Table 4.1.0-1: Tabulated Cargo Concept of Operations

OP	Description	ΔV (m/s)	Time (days)	Thrusting Module
L1	Launch from KSC into LEO	9400	3 - 4	SLS
L2	TLI from LEO to southern 11:3 L2 NRHO to dock w/ DSG	3200		SLS
L3	Capture at 11:3 southern L2 NRHO	450		OM
1	Separate from DSG and transfer from NRHO to LLO	723	3.73	OM
2	Wait in LLO until landing site is accessible	----------------	14 (Max)	--------------------
3	Module separation and landing	2067	Negligible	LM
4	Lunar surface operations	----------------	1	--------------------
5	Launch from lunar surface	2067	0.1	LM
6	Plane change and docking	391		OM
7	Transfer from LLO to NRHO and dock with DSG	723	3.73	OM
--------	Repeat steps 1-7 as required	----------------	------------------	
Total:	--	5971 (Ops)	22.56	

2.0 Cargo Configuration Trajectory Simulation

Due to that fact that an NRHO is a product of 3 body Astrodynamics, it was necessary to construct a 3-dimensional computational simulation of the desired orbits and transfers in order to calculate accurate values for the ∆Vs and times of flight. These simulations were developed with a custom trajectory optimization tool built in MATLAB and relying heavily on another custom circular restricted 3-body problem function. As shown in Figure 4.2.0-1, the simulation begins in a 11:3 southern L2 NRHO by integrating the circular restricted 3-body function using the ode45 MATLAB function and initial conditions corresponding to the desired NRHO.

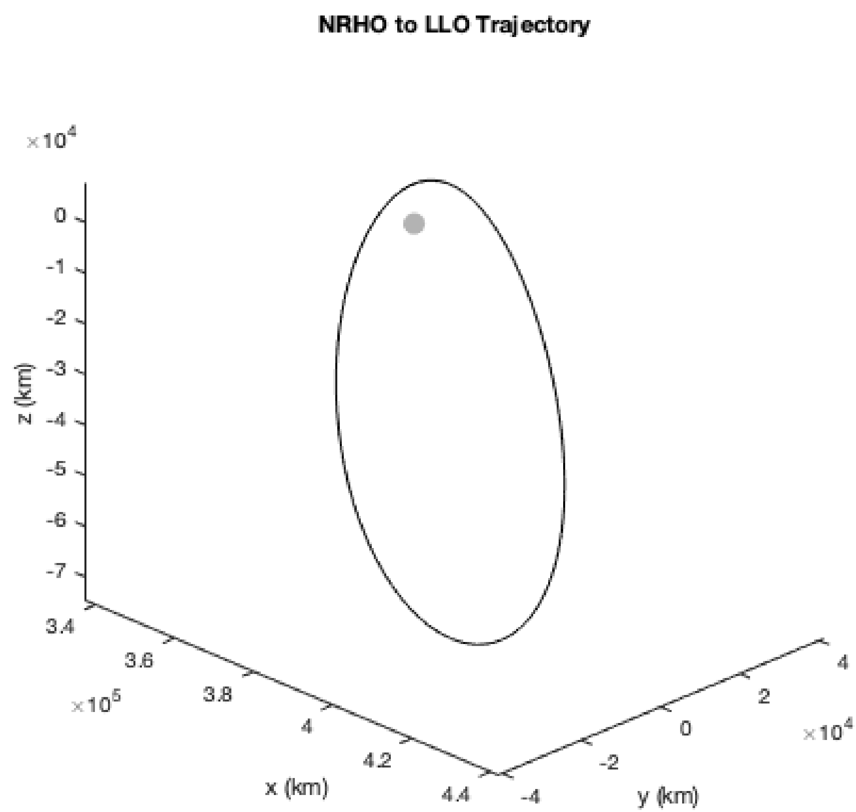


Figure 4.2.0-1: 11:3 Southern L2 NRHO

The bulk of the computational power required by the simulation went into determining the transfer orbit depicted in Figure 4.2.0-2. The determination of this transfer was accomplished by finely varying the cartesian components of the velocity vector used as inputs in the ode45 function until the desired lunar altitude conditions were met (roughly 100 km above the surface) at the periapse of the transfer orbit. Ultimately the determination of this transfer orbit required running ode45 10,000 times.

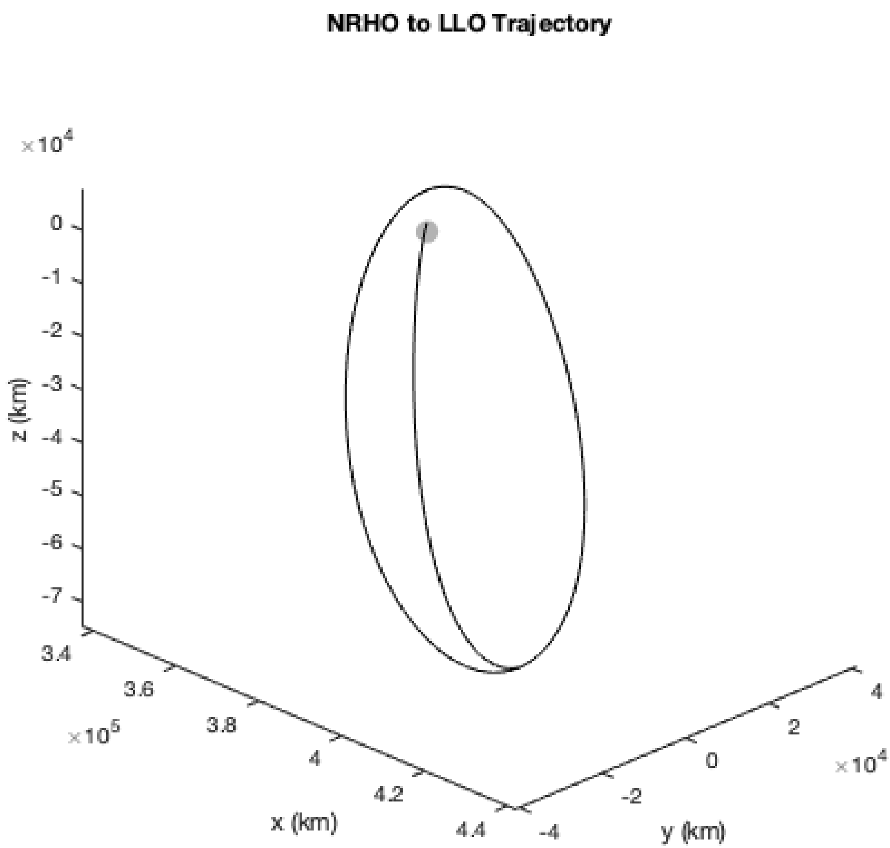


Figure 4.2.0-2: Transfer from NRHO to Polar LLO

Once the desired periapse altitude conditions were met, 2-body dynamics were applied to approximately calculate the necessary ΔV required to insert into a circular polar orbit around the moon at an altitude of roughly 100 km. This is depicted in Figure 4.2.0-3. The moon appears to be completed shaded by the polar LLO in the figure because the circular orbit allows for complete coverage of the lunar surface over the course of a full lunar period (28 days).

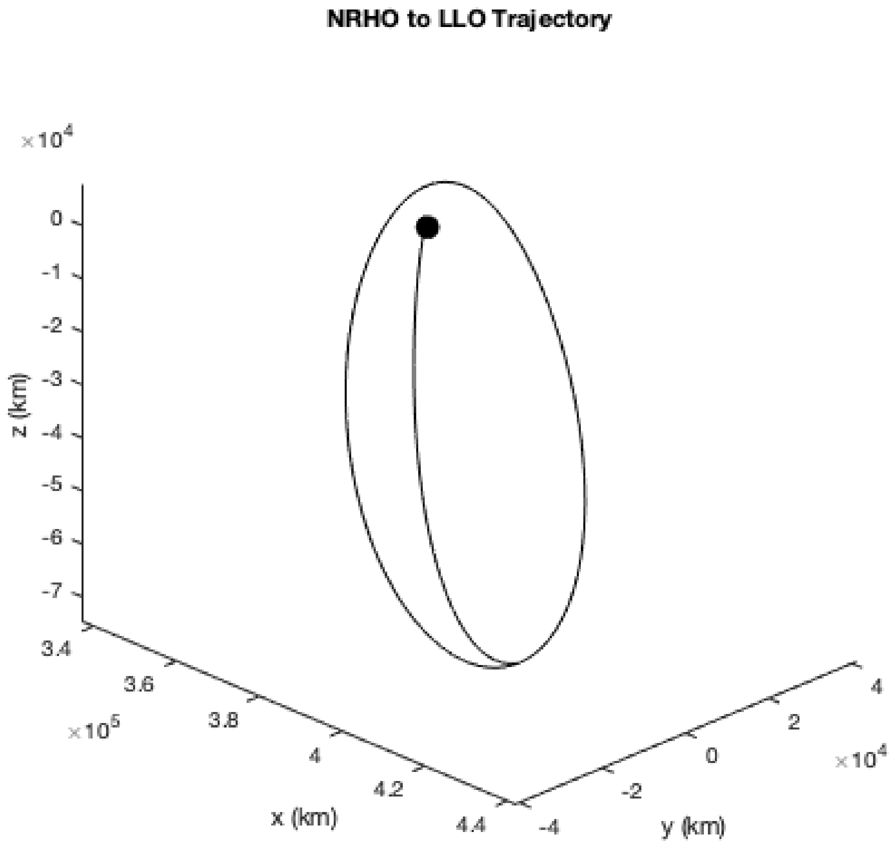


Figure 4.2.0-3: Circular Orbit Rotating Around Lunar Inertial Frame

Figures 4.2.0-4 through 4.2.0-6 depict the circular polar LLO at approximately 1 day, 7 days, and 14 days after circular orbit insertions respectively. These figures conclusively show that, as the tidally locked moon orbits around the earth, the polar LLO effectively rotates around the moon over the course of the lunar period granting complete access to the lunar surface by day 14. In the cargo mission configuration, the CAVEMAN will need to orbit the moon for a maximum of 14 days to satisfy the requirement that 100% of the lunar surface be accessible.

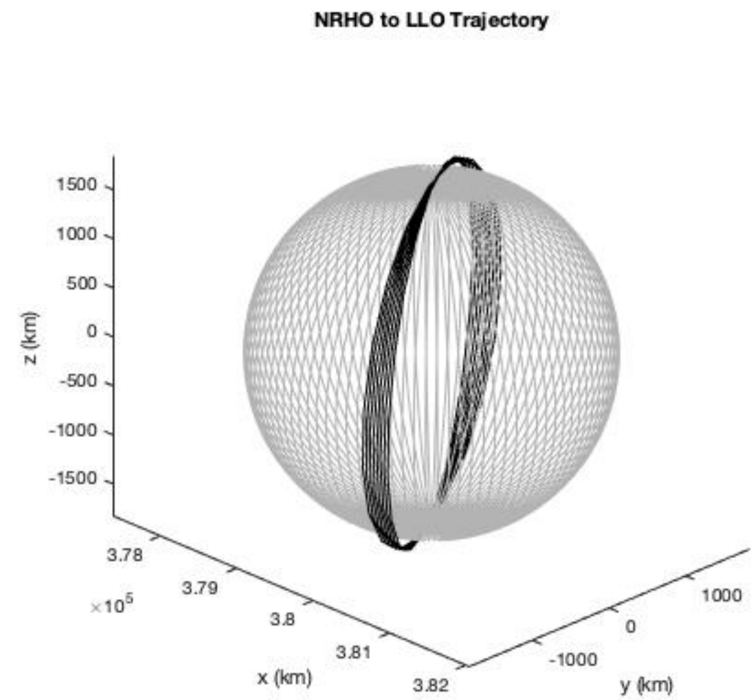


Figure 4.2.0-4: Polar LLO T+ 1 Day Since Circularization

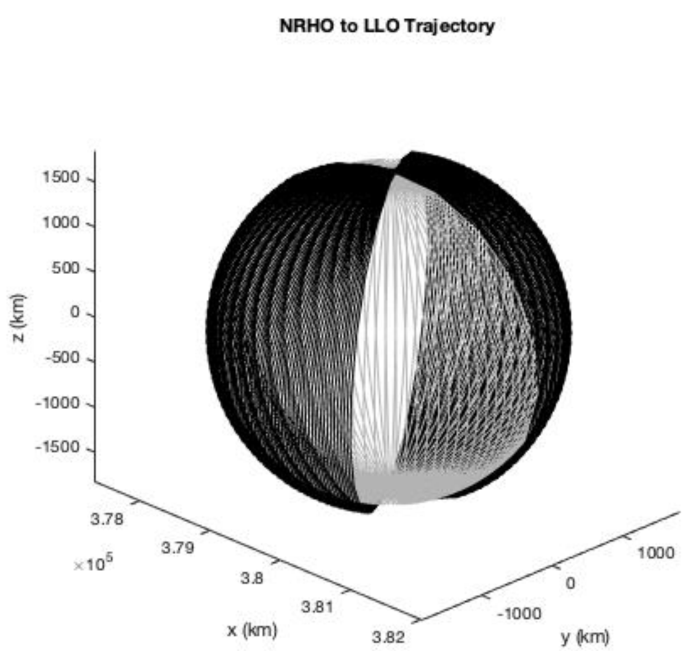


Figure 4.2.0-5: Polar LLO T+ 7 Day Since Circularization

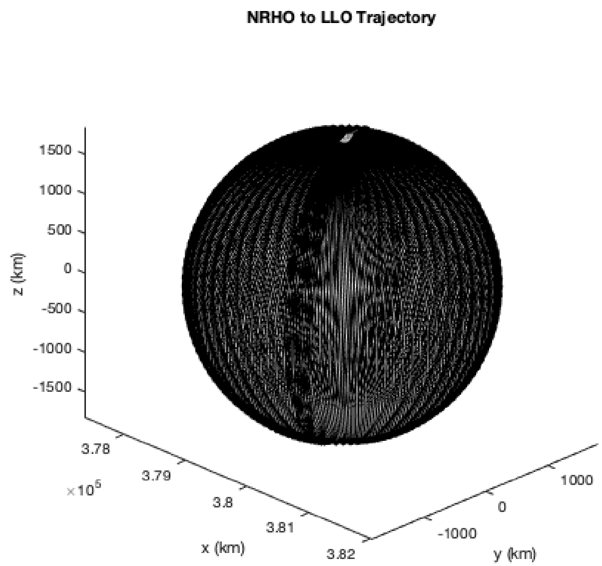


Figure 4.2.0-6: Polar LLO T+ 14 Days Since Circularization

3.0 Crew Configuration Mission Design

The crew mission was designed to first minimize the required ΔVs and second minimize the total time of flight in order to reduce the crew's exposure to environmental hazards associated with cis-lunar space and to reduce the load on the ECLSS system. With these constraints in mind, the crew configuration mission design is as follows. Initially the CAVEMAN is docked with the DSG in a 11:3 southern L2 NRHO. At the apoapse of the NRHO, the CAVEMAN separates from the DSG and burns using the OM to enter into a transfer orbit to a polar LLO. Once at the periapse of the transfer orbit, the OM burns again such that the CAVEMAN enters a polar elliptical orbit. This ellipse allows for a plane change so that the landing site can be accessed without revolving around the moon for 14 days. At the periapse of the elliptical orbit, the OM burns again to circularize the orbit of the CAVEMAN above the landing site. At this point the OM and the LM separate, allowing the LM to begin its landing burn while OM resides in a polar LLO for the duration of the lunar surface operations. Once on the surface, the crew is able to conduct their surface operations for 24 hours. After these 24 hours, the OM performs a plane change to align its orbit with the LM as the LM takes off from the surface. Then the OM and LM rendezvous and dock in a polar LLO. At this point the complete CAVEMAN transfers from a polar LLO to the NRHO such that it can rendezvous and dock with the DSG, completing its mission. This concept of operations is shown graphically in Figure 4.3.0-1.

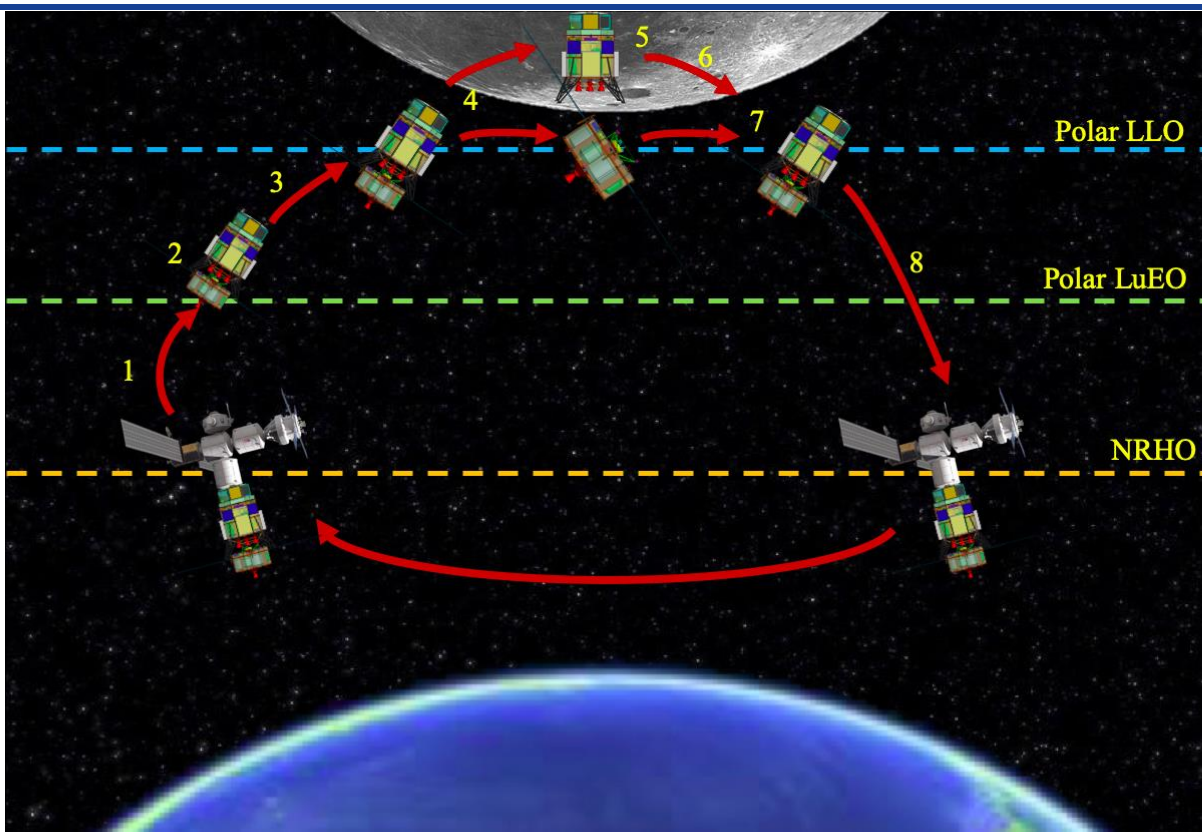


Figure 4.3.0-1: Crew Configuration Concept of Operations

Table 4.3.0-1 shows the discretized ΔVs and times of flight for the crew configuration mission design. The total mission duration of ~16 days represent a baseline used for the design of the ECLSS system.

Table 4.3.0-1: Tabulated Crew Configuration Concept of Operations

OP	Description	ΔV (m/s)	Time (days)	Thrusting Module
1	Separate from DSG and transfer from NRHO to polar Lunar Elliptical Orbit	125	3.73	OM
2	Plane change at apolune to access landing site	181	7.42	OM
3	Transfer from LuEO to LLO	598		OM
4	Module separation and landing	2067	Negligible	LM
5	Lunar surface operations	0	1	------------------
6	Launch from lunar surface	2067	Negligible	LM
7	Plane change and docking	391	0.1	OM
8	Transfer from LLO to NRHO and dock with DSG	723	3.73	OM
-------	Repeat steps 1-7 as required	-----------	------------------	------------------
Total:	--	6152	15.98	------------------

4.0 Crew Configuration Trajectory Simulation

Using a similar version of the simulation used for the cargo configuration, the crew configuration mission design was accurately modeled using a circular restricted 3-body problem function, the ode45 MATLAB function, and a custom script to model the transfers. Figure 4.4-1 again shows the 11:3 southern L2 NRHO in which the DSG resides.

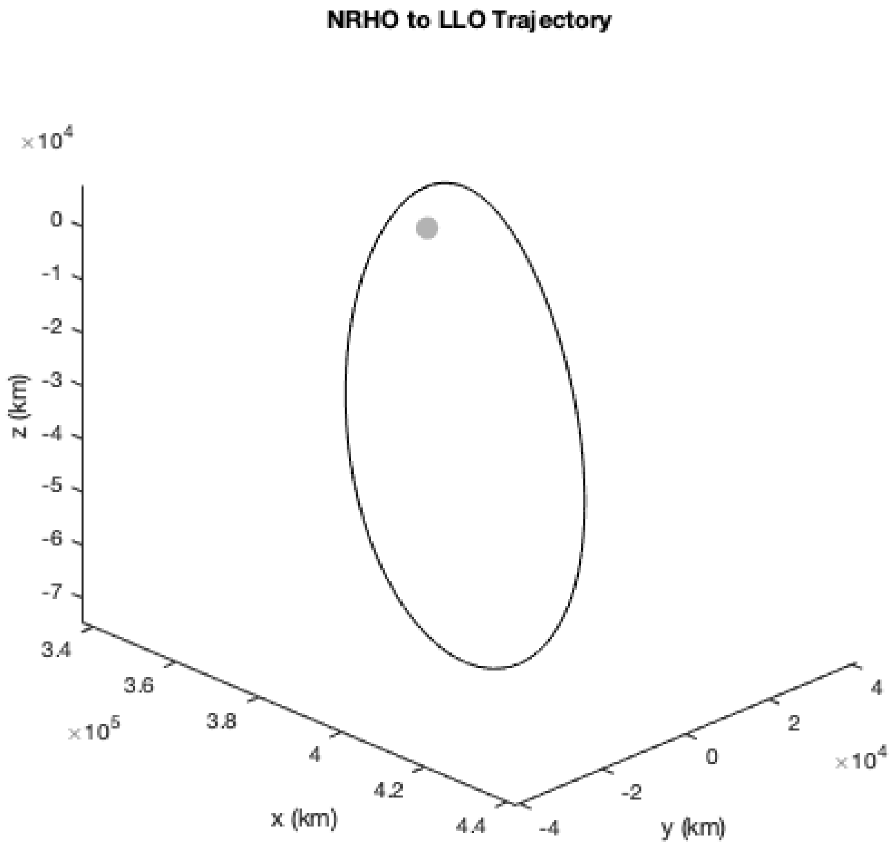


Figure 4.4.0-1: 11:3 Southern L2 NRHO

Figure 4.4.0-2 again displays the same simulated transfer from the NRHO to a periapse roughly 100 km above the surface of the moon. The primary difference is that at the periapse the CAVEMAN burns to ellipticize its orbit rather than circularize it like in the cargo configuration simulation.

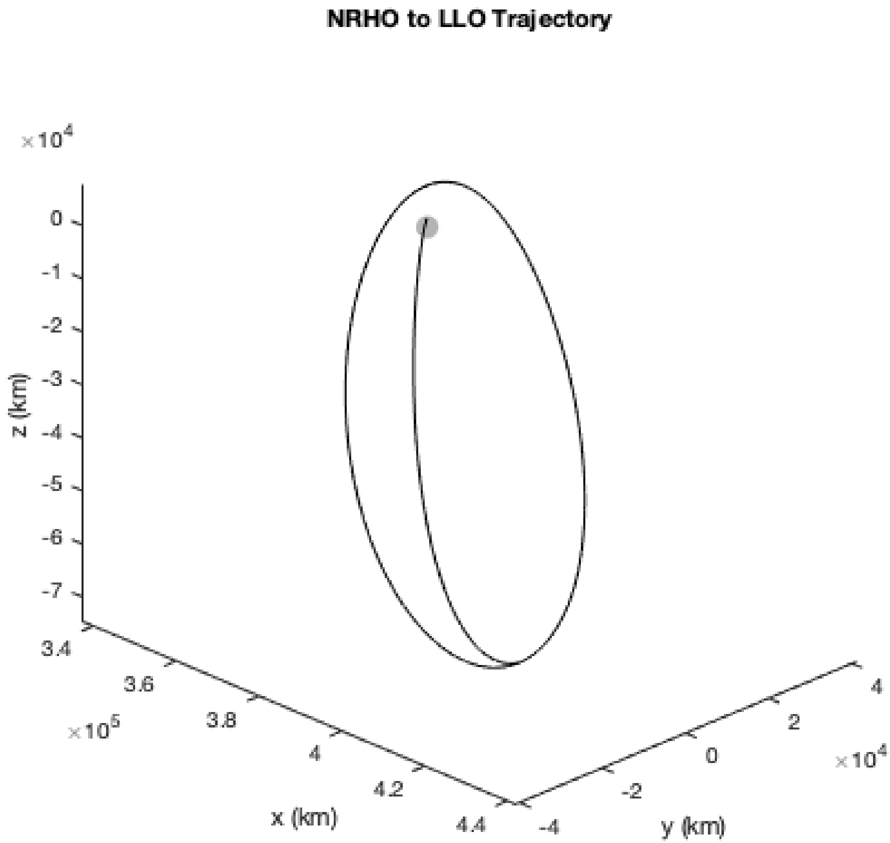


Figure 4.8.0-2: Transfer from NRHO to Polar Elliptical Orbit

Figure 4.4.0-3 then shows this elliptical orbit. At the periapse of the ellipse a plane change can be conducted at a relatively low cost such that the desired landing site can be access following the circularization of the orbit at the periapse of the ellipse.

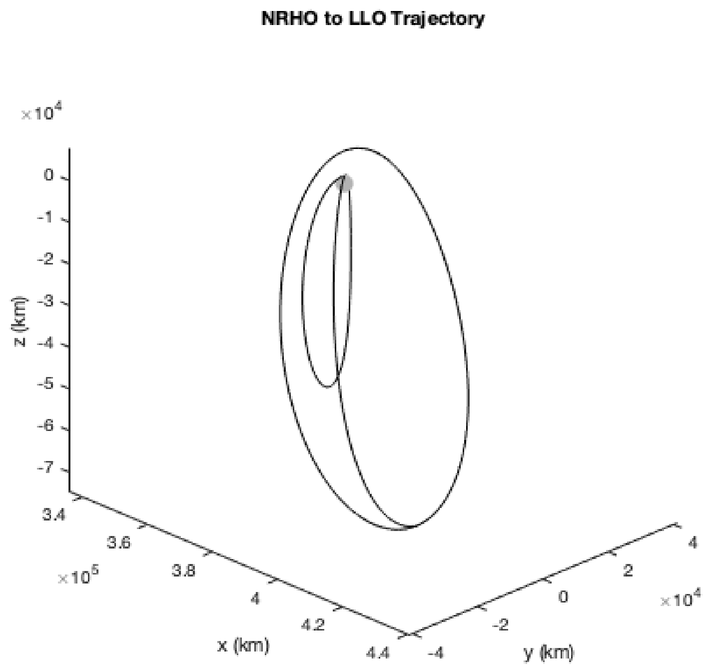


Figure 4.4.0-3: Polar Elliptical Orbit

Finally Figure 4.4.0-4 shows the circularization of the orbit into a polar LLO which is aligned with the landing site. This alignment allows for the separation of the two modules and the landing of the LM followed by surface operations.

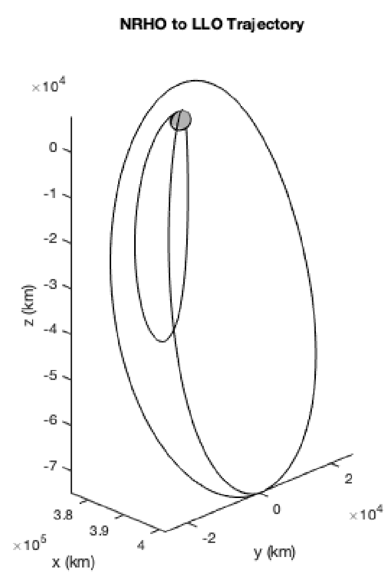


Figure 4.4.0-4: Insertion into Polar LLO

5.0 Launch Vehicle

The current selected launch vehicle for deploying our spacecraft to the Deep Space Gateway is the Space Launch System Block 2. This vehicle was selected due to its great payload weight capacity along with its large payload fairing volume. It has a usable diameter of 9.1 m and a total payload area of 905 m^3 which is more than enough room for our current spacecraft design. It is also about to deliver a maximum of 45 tons to Trans-Lunar Injection which is our design point and since we don't need to bring any cargo, crew, or significant amount of propellant this limit is more than achievable for our current design. There are a few issues with whether it will be available by the promised 2028 date since it is mission critical to have the spacecraft at the DSG by 11/20/28. The estimated cost ranges from 5 M-USD - 1.5 B-USD which is between 5 to 15 percent of our total budget.

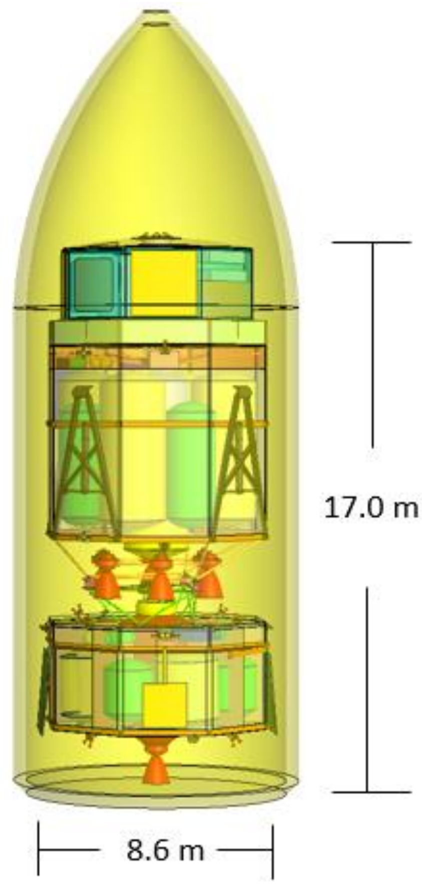


Figure 4.5.0-1: Launch Vehicle Configuration

V. Spacecraft Design

1.0 System Overview

The down selected architecture is displayed below. From top to bottom, the purple module is the Hab, the blue module is the Lander Module and the green module is the Orbital Module.

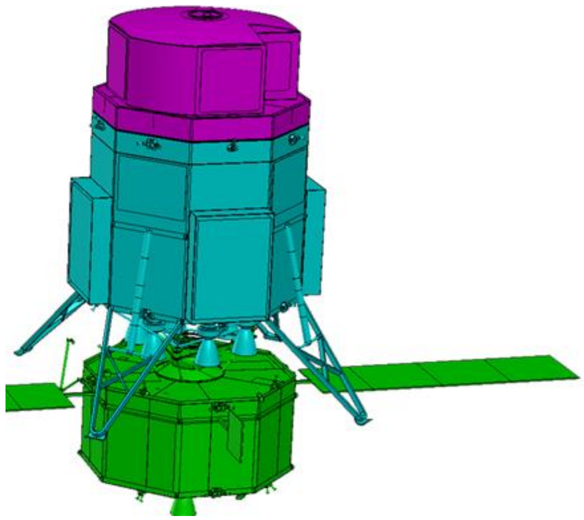


Figure 5.1.0-1: Three Separate Vehicle Modules

Below, the general layout of the vehicle is described for quick reference and intuition. These are not strict but rather give an idea of vehicle layout.

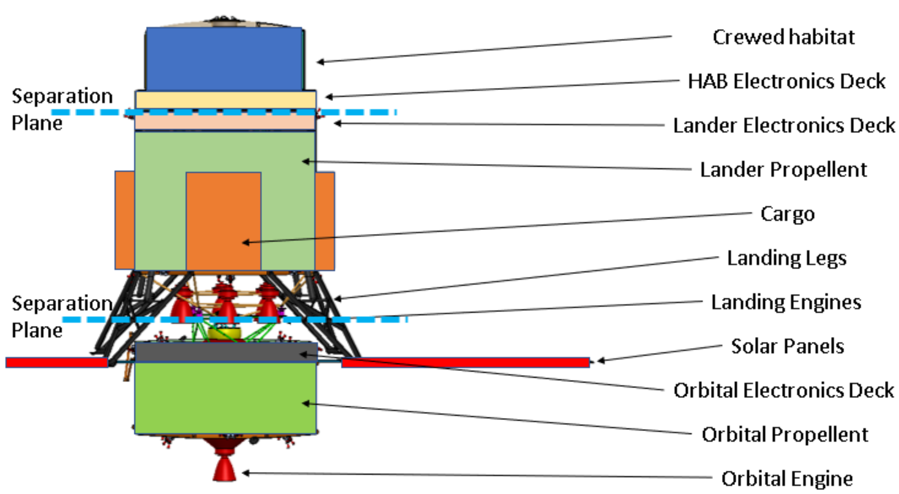


Figure 5.1.0-2: Major Subsystem Breakdowns

The mass breakdown for the various modules and configurations, both wet and dry, can be seen below. The wet mass displayed is maximum mass for the mission profile, fully loaded cargo, and full propellant tanks.

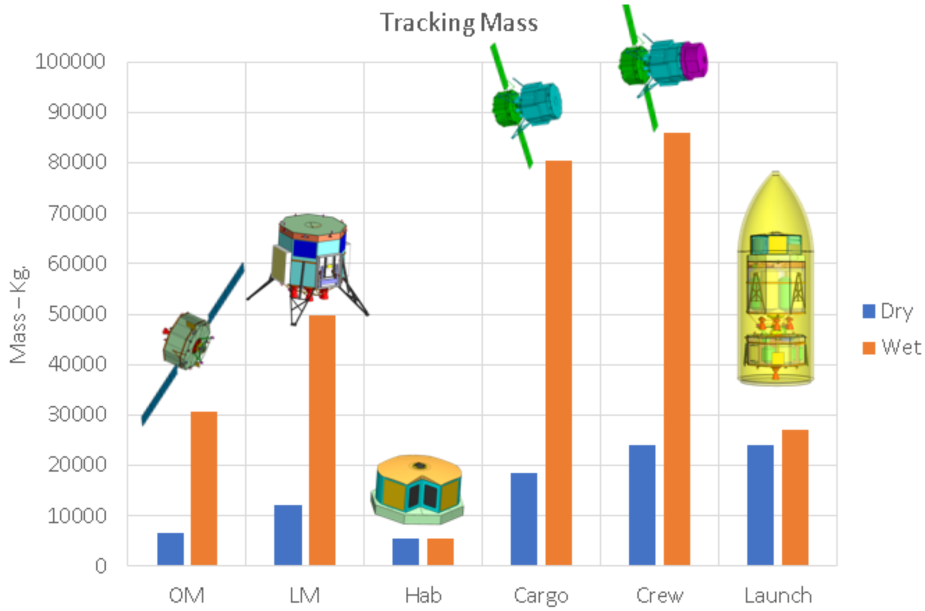


Figure 5.1.0-3: Visual Mass Breakdown for Various Modules

The vehicle's chosen propellent is LOX / Hydrogen, with the RL10 as the main thruster on all modules. This fuel-oxidizer combination was useful due to efficiency, throttle-ability and the possibility of using fuel cells for power and heat generation in polar landings. The legs are single time deployable and the cargo is mounted onto the side of the landing module. All three modules are launched at the same time within the SLS Block 2 10m fairing. While landing, the vehicle has 1.5 meters of engine bell clearance and utilizes an electronics package that allows the vehicle to avoid large obstructions such as rocks or craters.

When designing for cargo capacity, a worst-case cargo density of 800 m^3 was assumed. Giving us a derived requirement for 22 cubic meters of cargo. Cargo is stored in 3 radial boxes on the lander which are accessed by a robotic lunar forklift, stored in a fourth cargo box, that is deployed during surface operations. Each box has 4m x 2.9m x 0.75m dimensions and has mounting features to allow for a variety of sizes depending on what the customer requires. Cargo boxes can be loaded at the DSG by robotic arm and can, ideally, meet specific environmental requirements as needed by the payloads themselves.

1.1 Power System Overview

The CAVEMAN's power system is designed such that each module can share power between themselves. This is done using docking ring electrical adapters on each module. These adapters will connect during docking with other modules or birthing with the DSG and allow the spacecraft to recharge or transfer power to each of its modules through the docking rings. Each module is equipped with its own set of batteries but the main power system selection by module was determined by considering mission profiles as a primary criterion and subsequently, based on minimum weight, volume and maximum capacity. This is shown in the following tables.

Table 5.1.1-1: Mission Profile Drivers for Power System Selection

Time Eclipsed per Module				
Module	**Time in shade**		**Time in sun**	
CM				
Low Lunar Orbit	45.24	min	77.26	min
Surface Ops. (Polar)	1440	min	0	min
PM				
Low Lunar Orbit	45.24	min	77.26	min

Table 5.1.1-2: Power System Trade Study

Architecture	**Mass**		**Volume**	
	Value - kg	*Rank*	*Value - m^3*	*Rank*
Batteries	24.864	1	0.045	1
Solar Panels	823.816	3	1.628	3
Fuel Cells	358.356	2	0.502	2
	*Lower is better		*Lower is better	

Architecture	**Energy Capacity**		**Weighted Rank**	
	Value - kWh	*Rank*	*Raw*	*Ranked*
Batteries	7.246	2	21	**1**
Solar Panels	5.482	3	54	**3**
Fuel Cells	38.102	1	33	**2**
	*Higher is better		*Lower is better	

Weight Factors	
Mass	6
Volume	9
Energy Capacity	3
***Higher is more important**	

Table 5.1.1-3: CAVEMAN Power System Selection

Power System Selection		
Module	**Power Source 1**	**Power Source 2**
CM	Batteries	Fuel Cells
PM	Batteries	Solar Panels
HAB	Batteries	N/A

Because it is possible for the LM to experience a minimum of 24 hours in the shade during the event of a polar surface operation, solar arrays could not be used as a primary power source for this system. Also, given that the LM will be responsible for powering itself as well as the Hab, it needed a power system with a high capacity. Thus, fuel cells were selected for the LM because of their high capacity and relatively low weight. They also operate on liquid oxygen and liquid hydrogen which is the same propellant as the main propulsion system. This enables the propellant required for the fuel cells to be stored in the main propellant tanks since the tanks are sized to account for fuel cell requirements. To increase redundancy and to deliver power for both the LM and Hab a total of 3 fuel cell units will be used. However, each fuel cell will not run at maximum capacity for the entire duration of a mission as power output will be shared across each of them.

For the OM, it was more beneficial to use solar panels as a primary power system because it will be parked in an LLO orbit or in transit to the lunar surface for the entirety of its life. Therefore, the amount of shaded time that it will experience is far less than that of the LM and Hab and will benefit from charging its batteries via solar arrays over fuel cells because it will not have to use any propellant to fuel its power system.

Batteries are the only power supply on the Hab because of weight constraints. Because there is a plethora of other systems that must be integrated into the Hab to ensure the safety and feasibility of manned missions, mass and volume requirements were minimized in order to maximize the amount of allocated mass towards other systems. As a result, the Hab utilizes the LM and DSG for charging these batteries.

A diagram of each module's power system is found in Figures 5.1.1-1 & 5.1.1-2. Figure 5.1.1-1 displays the power system's cross-feeding capability and Figure 5.1.1-2 shows a more detailed component-wise representation of the power system.

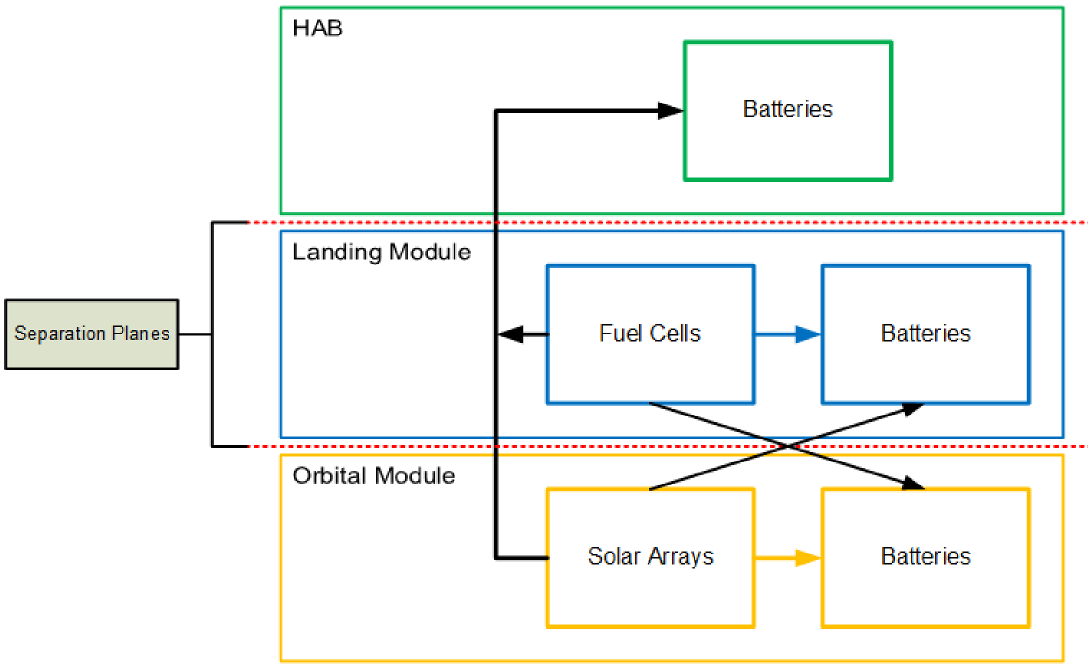


Figure 5.1.1-1: CAVEMAN Power System Diagram by Module

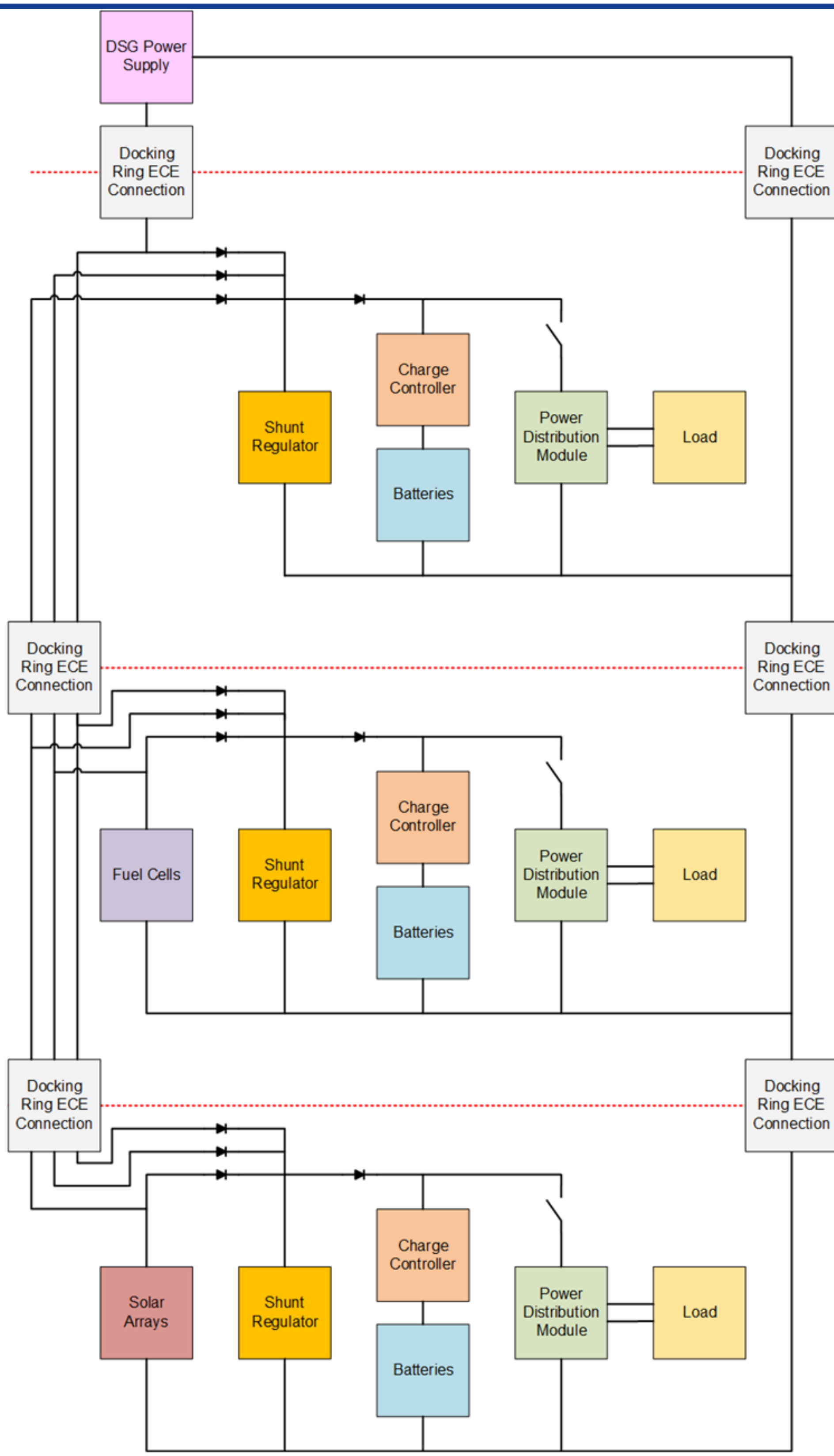


Figure 5.1.1-2: Detailed CAVEMAN Power System Schematic by Module

Because the CAVEMAN is capable of splitting into multiple modules, a power system mission profile has been created to display each power system configuration that the CAVEMAN will experience throughout the duration of any given mission type. Table 5.*1.1*-4 displays which power systems are active and inactive for each mission case.

Table 5.1.1-4: Power System Mission Profile

Power Profiles						
Configuration	**Dock @ DSG**	**Crew Mission**	**Cargo Mission**	**Crew Landing**	**Cargo Landing**	**Orbital Module**
DSG Supply	x					
Batteries - HAB	x	x		x		
Batteries - LM	x	x	x	x	x	
Batteries - OM	x	x	x	x	x	x
Solar Arrays	x	x	x			x
Fuel Cells		x	x	x	x	

1.2 ACS System Overview

The CAVEMAN's ACS system is designed such that it will enable the spacecraft to perform the maneuvers that are necessary for mission success. The spacecraft's designed called for a high pointing accuracy, high maneuver rate, and minimal propellant cost solution. Thrusters were then sourced to these parameters and placed accordingly. The ACS system's control feedback loop is driven by various sensors and onboard equipment that help point, guide and stabilize the spacecraft.

1.2.1 ACS Thruster Clusters

The thruster that was selected is the Marquardt R1-E Hypergolic Thruster which provides an I_{sp} of 280s and a thrust of 111N which, when clustered, will provide adequate torque and thrust to complete any necessary ACS maneuvers.

Given the spacecrafts large size, the ACS thrusters had to be able to overcome the inertial forces to position and point the spacecraft. Thus, carefully placed thrusters were rotated to aggressive attack angles that ensure ample roll, pitch, and yaw control authority. These thrusters were also located in such a way that their field of view with a 45º half angle do not interfere with each other or the spacecraft and its components. The thrusters are arranged into thruster clusters with various configurations that enable complete 6-DoF control of the spacecraft.

Figure 5.1.2-1: Marquardt R1-E Hypergolic Thruster

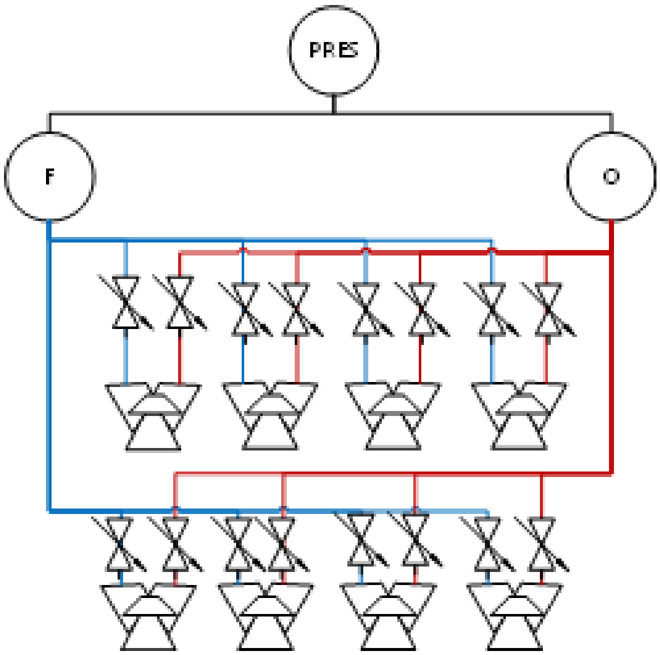


Figure 5.1.2-2: ACS Thruster Part & Instrument Diagram

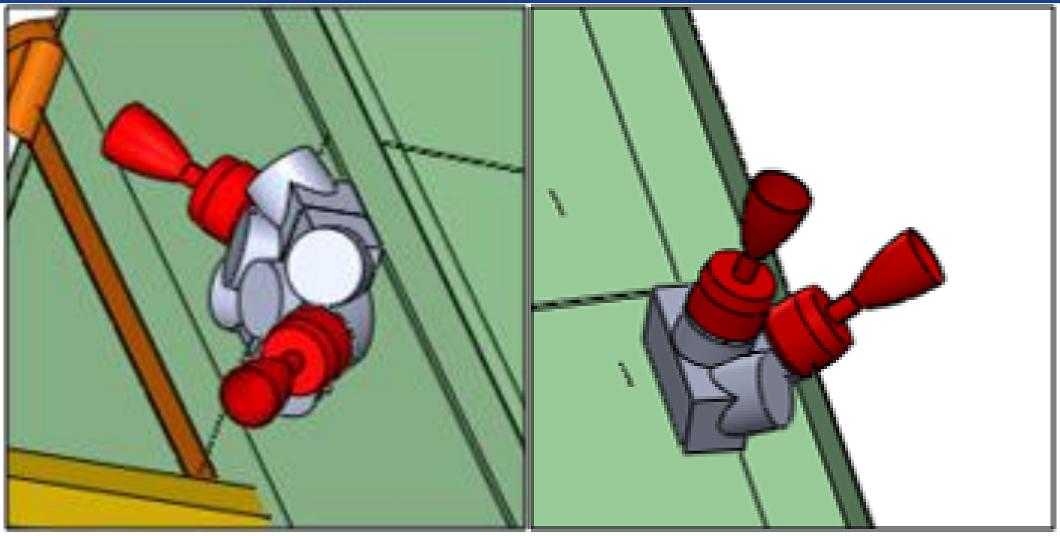


Figure 5.1.2-3: ACS Thruster Cluster Arrangements

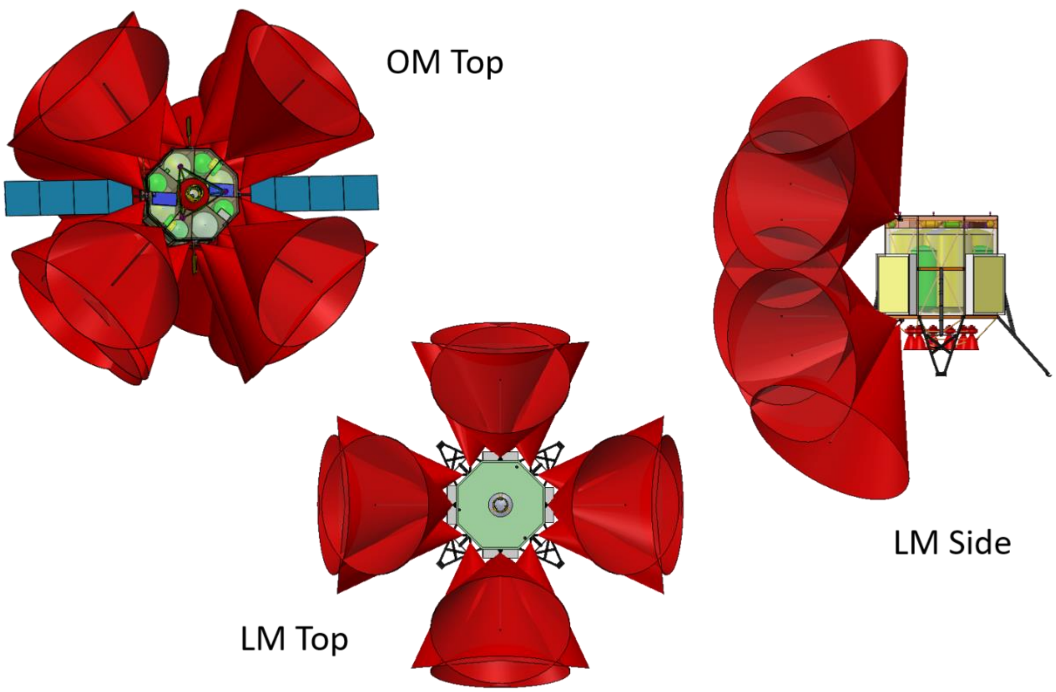


Figure 5.1.2-4: ACS Thruster Cluster Field of Views

1.2.2 ACS Sensors

In order to perform such maneuvers, the spacecraft needs to know where it is located and pointed. By using a wide range of various sensors, the ACS system's control feedback loop will be able to propagate its location and rotation in 3-space and determine the appropriate maneuvers to take for course correction if necessary. For general

ACS maneuvers, sensors such as star trackers, ring laser gyros, magnetometers, and IMUs will be used to gather information about the spacecraft's orientation and location.

Because the CAVEMAN can dock and undock with its modules, berth with DSG, and land on the lunar surface, there are more sensors that are required to accurately execute these maneuvers. The CAVEMAN will integrate laser rangefinders as well as LIDAR doppler velocimeters and infrared cameras for both docking and landing. A system like the Autonomous Landing Hazard Avoidance Technology will also be integrated, enabling the LM to evaluate possible safe landing sites on the lunar surface. The infrared cameras will ensure that the lunar dust that is kicked up upon approach of the lunar surface will not affect the system's ability to accurately evaluate landing zones. A full list of sensors that will be utilized can be found in Table 5.1.1.2-1.

Sensors will be mainly located in the equipment deck of the LM. Those sensors such as star trackers that need a clear field of view of the spacecraft's surroundings will be radially distributed about the LM's circumference. Sensors related to docking and landing will be placed near the LM's docking rings to allow a direct view of incoming modules and the lunar surface.

Table 5.1.1.2.2-1: Sensor Listing

Sensor	Mass (kg)	Power (W)
Star Tracker	3.3	18
Ring Laser Gyro	0.454	1.6
IMU	2.75	10
Magnetometer	0.1	0.45
Star Tracker	3.3	18
Space Integrated GPS/INS	9.5	34
Laser Range Finder	5	16.5
NGAVGS	8	30
LIDAR Doppler Velocimeter	Based on ALHAT	
Laser Altimeter		
Infrared Camera		

2.0 Hab

2.1 Module Summary

The Hab Module was designed to accommodate a crew of four for extended periods of time. Designed to this end, it was also continuously designed as an addition to the Deep Space Gateway in between mission operations and end of life. Much of the inspiration was taken from the Orion capsule and the Apollo Lunar Excursion Module. The entire system was on a strict budget of under 6 tons which allowed it to be considered "cargo" when bringing astronauts to the surface and allows for added ΔV to implement the extra maneuver as outlined in section 2.3.0. The design of radiation protection followed ALARA (As Low As Reasonably Allowable), allocating extra mass margin to shielding. Most of the Hab internals are essentially cartoons which could be elaborated on later in the design process, but the volumes and masses are representative.

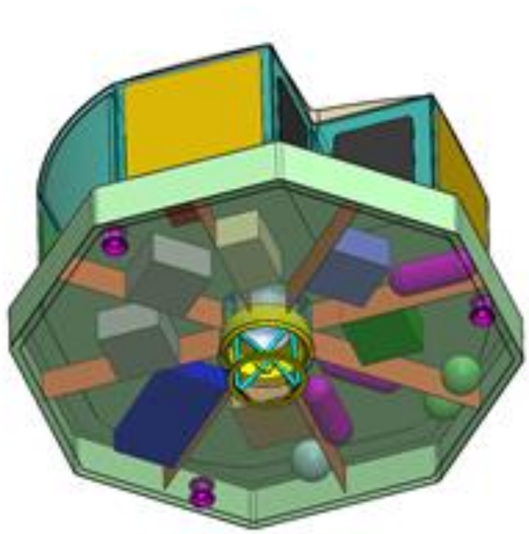


Figure 5.2.0-1: Underside of Hab

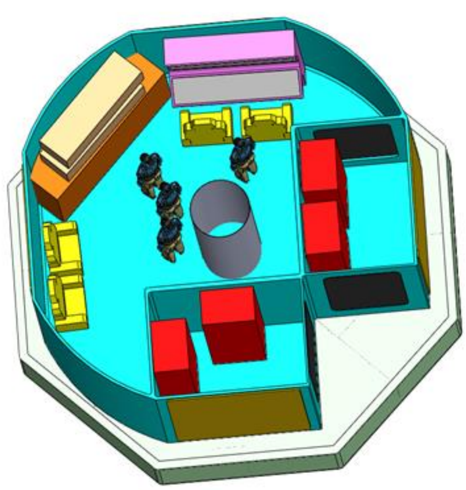


Figure 5.2.0-2: Topside of Hab

2.2 Environmental Control and Life Support System (ECLSS)

Any crewed space mission requires the development of an Environmental Control and Life Support System (ECLSS) and the requirements that drive the ECLSS mission for this mission are presented below in Table 5.2.2-1. These requirements essentially describe what is required for the crew to be comfortable and healthy for the duration of the mission. These requirements primarily reflect the conditions currently found on the International Space Station, which can be considered the standard for crewed missions. Additionally, the ECLSS systems also derives its design from the total crewed mission duration presented in the mission design section. The nominal crewed mission duration lasts 16 days, so the ECLSS system has been designed for a 30-day mission duration to add considerable margin.

Table 5.2.2-1: ECLSS Derived Requirements

D REQ	REQ Statement
1.1.1-1	Total Hab pressure shall be kept between 99.9 and 102.7 kPa
1.1.1-2	Total partial pressure of oxygen shall be maintained between 19.5 and 23.1 kPa
1.1.1-3	Partial pressure of atmosphere not made up by oxygen shall nominally be comprised of nitrogen
1.1.1-4	Partial pressure of carbon dioxide shall not exceed 0.4 kPa
1.1.1-5	Relative humidity shall rest between 30%-70%
1.1.1-7	Ventilation shall provide air circulation velocities between 0.08 to 0.2 m/s
1.1.2-1	Hab shall provide 3.5 kg/p-d of potable water
1.1.2-2	Hab shall provide 25.3 kg/p-d of hygienic water
1.1.2-3	Hab shall provide 7.3 kg of potable water per EVA
1.1.4-1	Biologically decomposable liquids shall be recycled for reuse
1.1.4-2	Biologically decomposable solids shall be stabilized, compacted, and stored until disposal
1.1.3-1	Hab shall provide between 2400 - 4800 kCal of equivalent food per day of mission
1.1.5-1	Hab oxygen partial pressure shall not exceed 30% of total atmosphere composition to reduce fire risk
1.1.5-2	carbon dioxide fire extinguishers shall be provided to extinguish and fires that may occur
1.1.5-3	Pure oxygen assemblies shall be provided to astronauts to reduce inhalation of carbon dioxide and trace contaminants in the instance of a fire

The requirements presented in Table 5.2-1 present a framework for the design of the ECLSS. To satisfy these requirements, Lunacy Solutions has chosen to utilize a design which recycles the Hab atmosphere and water supply, while disposing of solid non-recoverable waste and relying on the food and consumable supply of the Deep Space Gateway in between sorties to the surface. A block diagram presenting the flow of inputs and outputs of the ECLSS System is presented in Figure5.2.2-1. The primary inputs to the system are an initial supply of water, O_2, N_2, food, other consumables, and power. The primary outputs of the system are waste heat, non-recoverable solid waste, trace contaminants, and CO_2. To reiterate an initial supply of water, O_2, and N_2 are supplied by the DSG to the CAVEMAN prior to its mission to the lunar surface but over the course of the mission these resources will be recycled using the technologies described in Table 5.2.2-2. The non-recycled resources like food and consumables must be supplied by the DSG prior to each sortie. Additionally, waste will be disposed of at the DSG at the end of each mission.

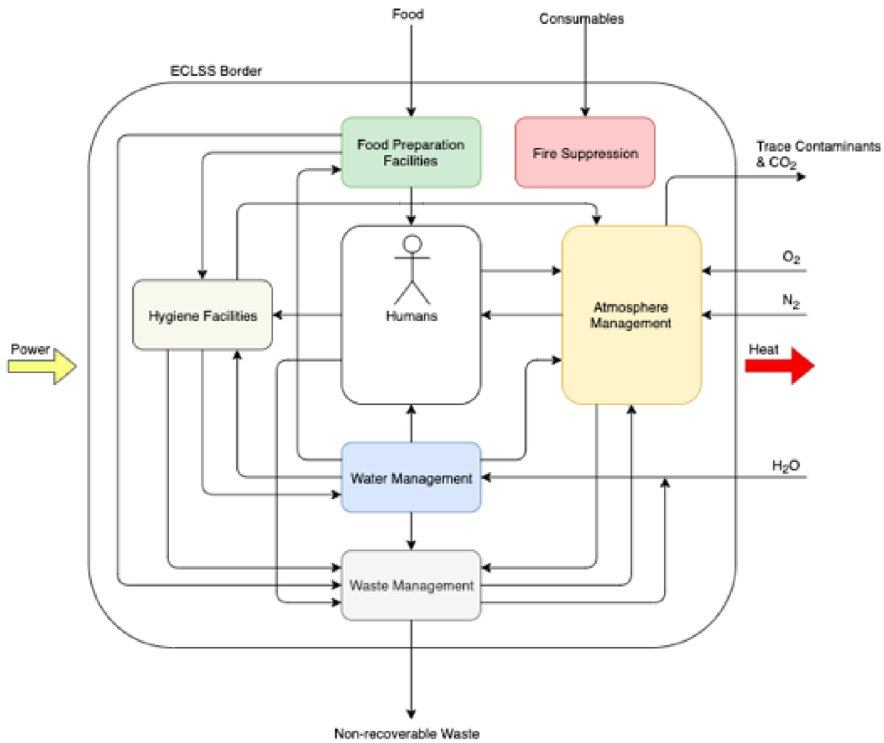


Figure 5.2.2-1: ECLSS Block Diagram

The primary components that comprise the ECLSS system are described in Table 5.2.2-2. The mass, volume, and power requirements for each of these components are also tracked to ensure accurate mass and power budgets and to make sure that these components all fit in the provided volumetric envelope.

Table 5.2.2-2: ECLSS Master Equipment List

Component	Function	Mass (kg)	Volume (m^3)	Power (kW)
4BMS	Removes CO_2	120	0.6	1.2
TCCS	Removes trace gases & contaminants	80	0.6	0.2
Stored O_2	Supplies O_2 make up gases	228 (108 tank)	0.75	--
Stored N_2	Supplies N_2 make up gases	215 (138 tank)	0.25	--
CHX	Removes water vapor from air and regulates temperature	8	0.1	0.4
MF	Purifies waste water (nonurine)	40	0.6	1.2
VCD	Purifies Urine	100	0.4	0.12
Stored H_2O	Provides water	122.4 (2.4 tank)	0.125	--
CO_2 Fire Suppressant	Manually Extinguishes fires in Hab	48	0.036	--
Aux. O_2 Supply	Personal O_2 in instance of fire	83.6	0.72	--
LiOH Canisters	Emergency excess CO_2 removal	28	0.02	0.012
Total	--	**1073**	**4.133**	**3.132**

The components listed in the master equipment list then need to be physically integrated into the Hab, which is shown in Figure 5.2.2-2. The components were integrated into a deck on the underside of the Hab so as not to remove any critical floor area from the crew during their lengthy missions. Ultimately the hardware easily fits where it needs to go.

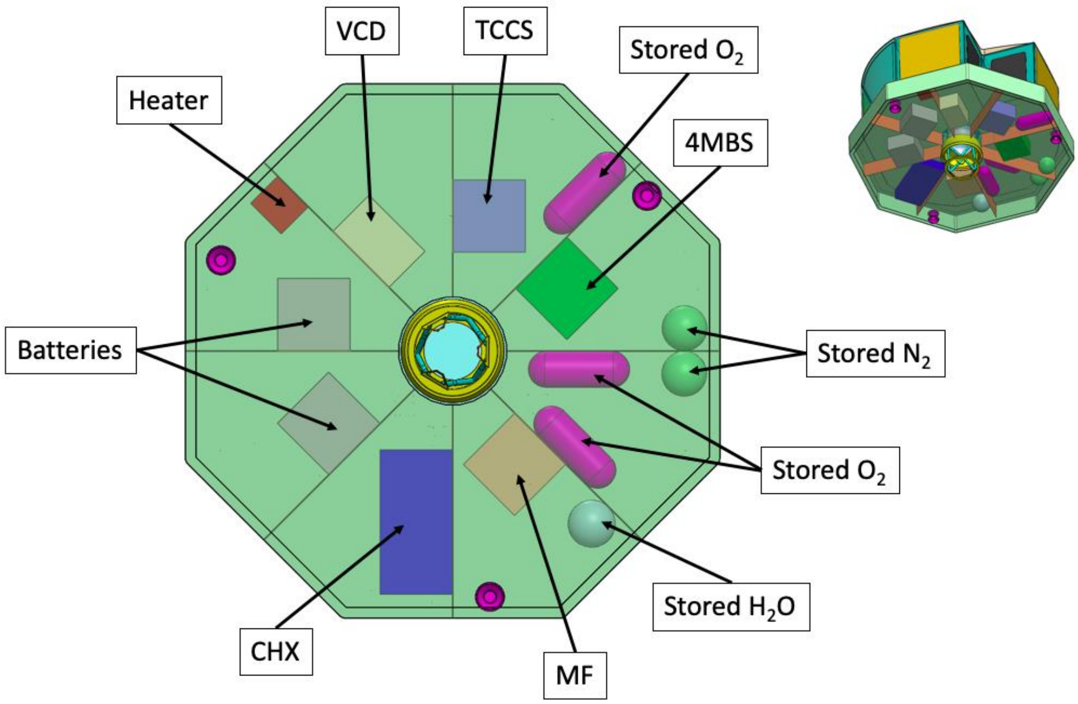


Figure 5.2.2-2: Integrated ECLSS Hardware

2.3 Structures

The main requirements for the Hab's structure subsystem are as follows:

Table 2.3.0-1: Hab Structures Subsystem Requirements

REQ #	REQ Statement
1.2a-1	Shall maintain structural integrity during all launch, thrust, docking, and landing load cases
1.2a-2	Shall withstand the difference of pressure between the vacuum of space and internal pressure necessary for the survival of the astronauts
1.2a-3	Shall dock safely with both the Deep Space Gateway and Cargo Module
1.2a-4	Shall allow entry and exit for all 4 crew members and one incapacitated crew member
1.2a-5	Shall minimize dust contamination of crew chambers and Deep Space Gateway

A mass and power statement for the Hab's structures can be seen below.

Table 2.3.0-2: Hab Structures Mass and Power

Item	Mass (ea)	Power (ea)	Qty	Mass Total	Power Total
	kg	W		kg	W
Walls	1200	0	1	1200	0
IDSS passive	60	0	1	60	0
IDSS active	85	500	1	85	500
Entry port	75	125	2	150	250
Suitlock	593	75	2	1186	150
			sum	2681	900

2.3.1 Static Structure

The Hab structures serve two main purposes, which are to maintain their integrity during all launch, thrust, docking, and landing load cases, and to withstand the difference of pressure between the vacuum of space and internal pressures necessary for the survival of the astronauts. The entirety of the Hab's walls are composed of aluminum 7075-T6 for its high yield and buckling strength along with low density. The walls were designed with uniform 2 mm thickness for manufacturability. The module is also compartmentalized into two sections: the crew compartment and the ECLSS deck. The primary structure for the crew compartment includes the outer shell of the Hab, the walls of the suitlocks, and the inner cylinder used to support the module during docking. The ECLSS deck serves as the floor for the crew compartment and provides a bay for all ECLSS and electronics hardware kept with the Hab. In total, the structures of the Hab have an estimated mass of 1,200 kg.

2.3.2 Mechanisms

Mechanisms on the Hab include the docking ports, suitlocks, and ingress/egress hatches. Docking mechanisms on the upper side of the Hab are passive, assuming the DSG interface utilizes the International Docking System Standard (IDSS). The lower half of the Hab uses the same IDSS alongside three ball-lock mechanisms distributed at 120° around the ECLSS deck's bottom side. These docking mechanisms and ball-locks can be seen in *Figure 5.2.3.2-1* below.

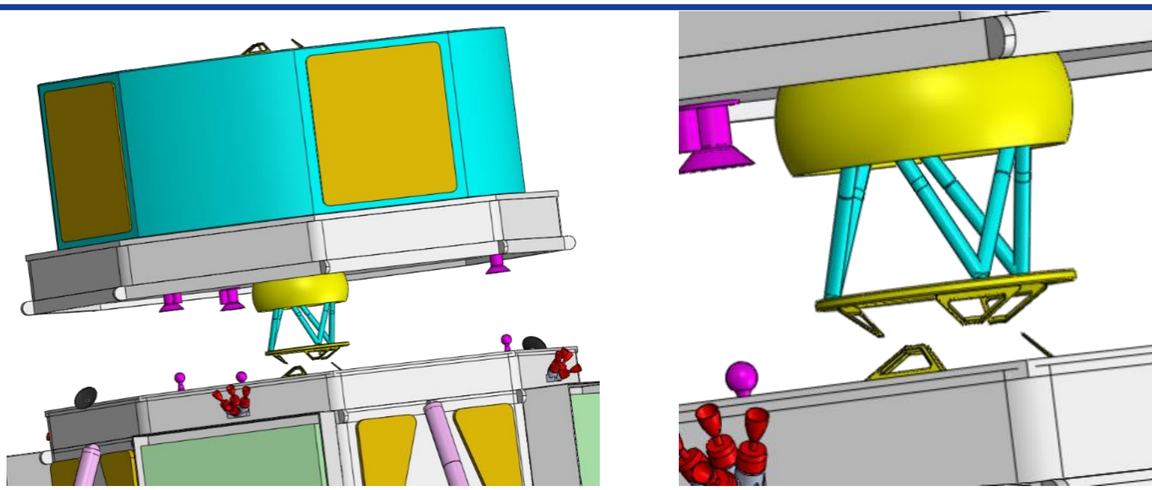


Figure 5.2.3.2-1: Docking mechanisms and close up view of IDSS

The entry door mechanisms save space on open and close. They swing open upwards with minimal extrusion and provide ample passageway for the astronaut's entry and exit through the hatches. Images of how the mechanisms open can be seen in *Figure 5.2.3.2-2* below.

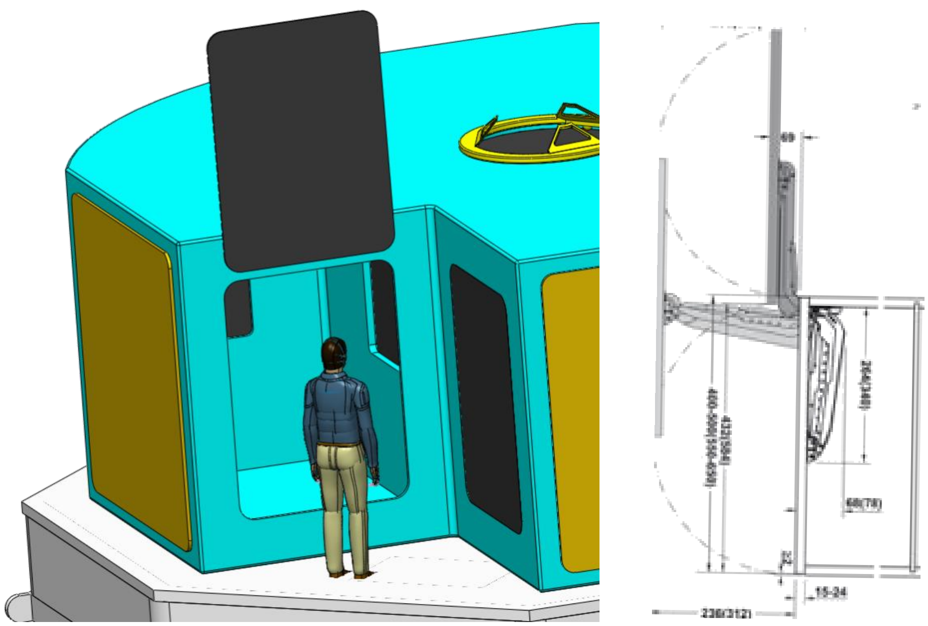


Figure 5.2.3.2-2: Entry door mechanism in action

Finally, the choice for the suitlock rather than an airlock came from the need to protect the crew chamber as well as the DSG from dust contamination. The selection of a suitlock was highly dependent upon the component's ability to isolate the living chambers. Interior design of the Hab and suitlocks account for a dustibule, the space needed for spacesuit ingress/egress, a comfortable seating arrangement, and food storage. Symmetrically placed suitlocks provide the capability for the entirety of the CAVEMAN's crew to perform EVAs on the lunar surface. A diagram of the floor layout can be seen in *Figure 5.2.3.2-3* below.

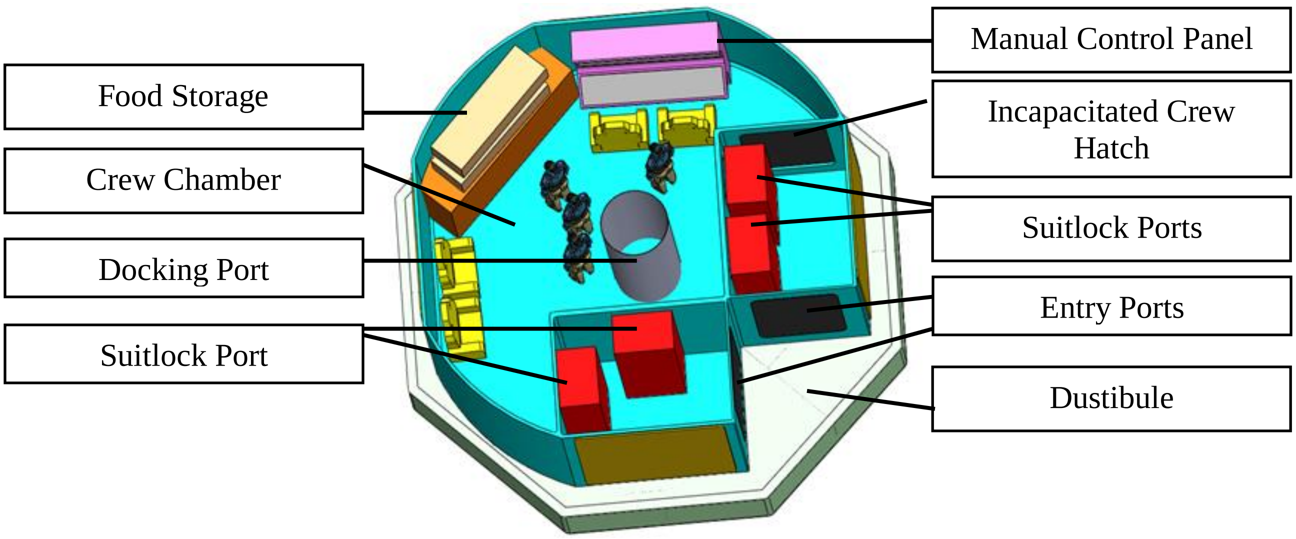


Figure 5.2.3.2-3: Hab Interior Layout

2.4 Electrical Power

The power system for the Hab module was designed such that it would be able to support the crew for the entire duration of a crewed mission. The primary power source on the Hab module are LiPo batteries which receive power via docking ring electrical connections that allow the Hab to be charged from the LM and/or the OM. This also enables the Hab to draw power from the batteries located on the LM and OM in the case that there is a battery failure in the Hab. The batteries for the Hab module are sized for the worst-case power requirements of a crewed mission which is the polar surface operations scenario. In this case, solar arrays cannot be used as there will not be direct line of sight with the sun. To meet the minimum surface operation time requirement on the lunar surface, battery capacities were determined from the Hab's subsystem power requirements for a 24-hour duration. This allows the Hab to continuously discharge for the entire duration of a surface operation. In the event there is either a battery outage on the Hab or more power is required for the duration of a surface operation, the Hab can draw power from the LM's power supply. The LM's fuel cell units are sized such that they can supply power for simultaneous charging and power draw for both the LM and the Hab's batteries.

Table 5.2.4-1: Hab Power System Mass & Power Statement

Hab Mass & Power Statement		
	Mass (kg)	***Power (W)***
Batteries	*1437.072*	*13322.0*
Solar Arrays	*0.000*	*0.0*
Fuel Cells	*0.000*	*0.0*
Total	*1437.072*	*13322.0*

2.5 Environmental Protection System

The Lunar environment is a very harsh environment, which has radiation levels and temperatures ranges drastically different from that experienced on Earth. These risks will have to be mitigated to acceptable levels using passive radiation shielding along with a robust thermal control system capable of keep the crew and cargo within their acceptable temperature ranges. In order to save both mass, power, and reliability our system will be initially passive and then have some active systems to allow our spacecraft to handle any situations that could occur during a mission.

2.5.1 Thermal Control System

The main design point for the Hab is keeping the crew within the design temperature which is between 5 °C and 25 °C, this design point has a margin of 5 °C. The limits can be seen in Figure 5.2.5.1-1 with the black bars are the limits that the thermal control system must maintain the spacecraft at using a variety of components. All other spacecraft components operational temperatures are well within this range except for the cryogenic propellants which will be address in a later section.

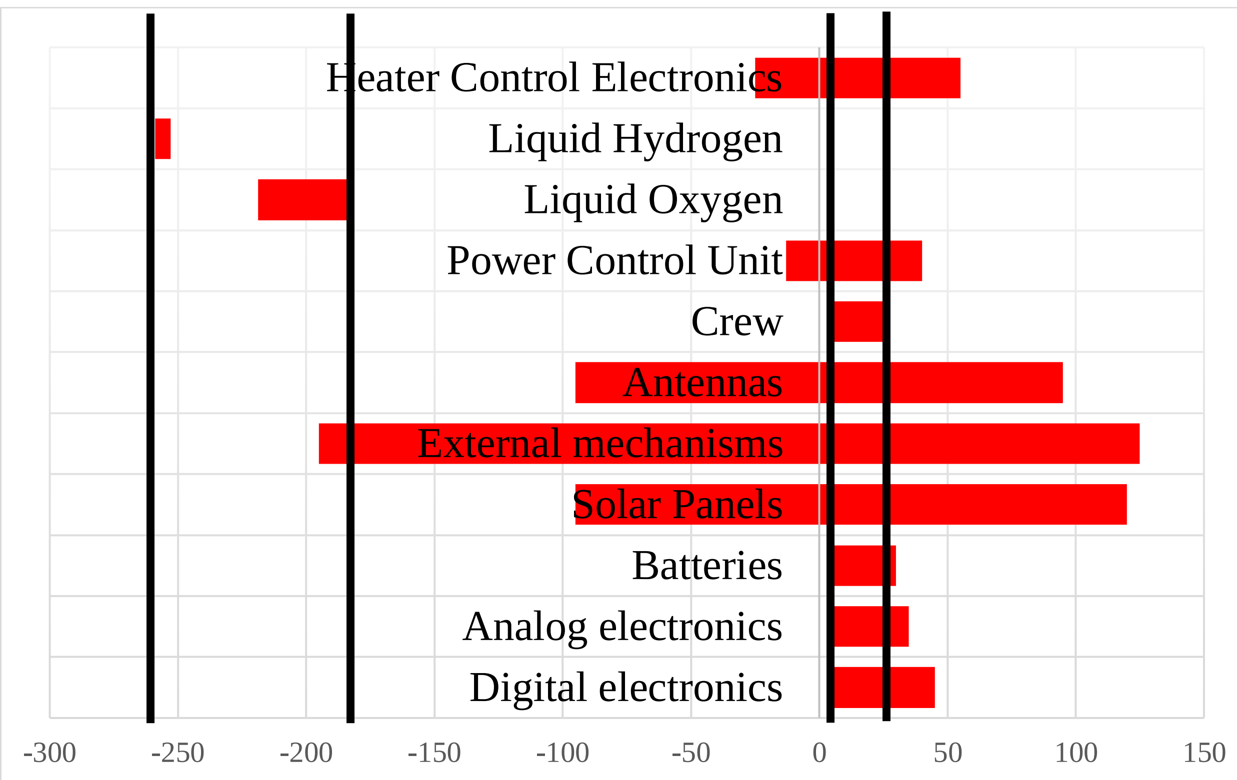


Figure 5.2.5.1-1: Temperature Range Requirements

2.5.1.1 Thermal Coatings

In order to reduce weight and power required to keep the crew alive during missions, the following thermal coatings have been selected to reduce the amount of solar energy absorbed to a minimum while also allowing for heat to dissipate through the outer layer of MLI. These finishes were selected so the spacecraft operates hot being that it requires significantly less energy to radiate excess heat then it is to generate heat inside of a cold spacecraft.

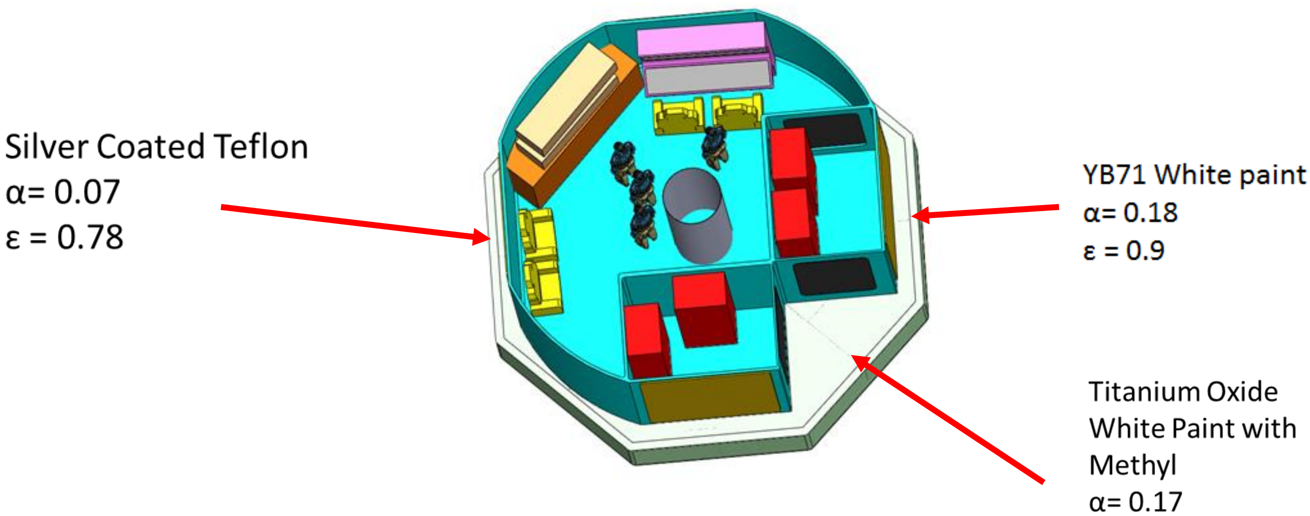
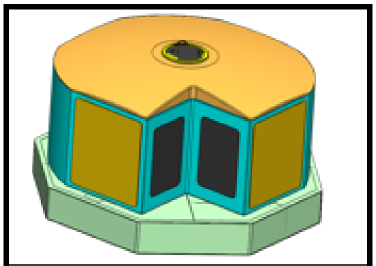


Figure 5.2.5.1.1-1: Coatings and Finishes of the Hab

2.5.1.2 Temperature Cases

The coatings and finishes were used to calculate the worst temperature cases as seen in Figure 5.2.5.1.2-1. The hot case is when the spacecraft is still in LLO and in full view of the sun, so it is experiencing the maximum amount of solar energy entering the system. The cold case for this is if both the Hab and lander module are parked in a deep crater and have zero solar energy to heat up the system. This case is very cold and would require the most amount of energy to maintain the spacecraft at the design temperature. There is also extra insulation not accounted for when you add the polyethylene. This was done by separating each module and assuming a spherical black body to make the math simpler for an initial design of the thermal system.



Thermal Case	Temperature (°C)
Hot Case	38.6
Cold Case	-134

Figure 5.2.5.1.2-1: Hab Thermal Cases

2.5.1.3 Radiator Design and Sizing

In order to prevent the spacecraft from overheating four radially placed louvers will be placed on the Hab. The purpose of placing them radially is that even if one of them is facing the sun there is one on the other side that can be radiating excess heat. The use of louvers over just conventional radiators is to increase their effectiveness when we are in orbit or parked on the lunar surface in direct sunlight. They allow us the option to maintain our upper limit even in our worst condition and require a less total area. The thermal system will be using standard heat pipes to wick away all the heat from the electronics, components and Hab to the radiators to disperse into space.

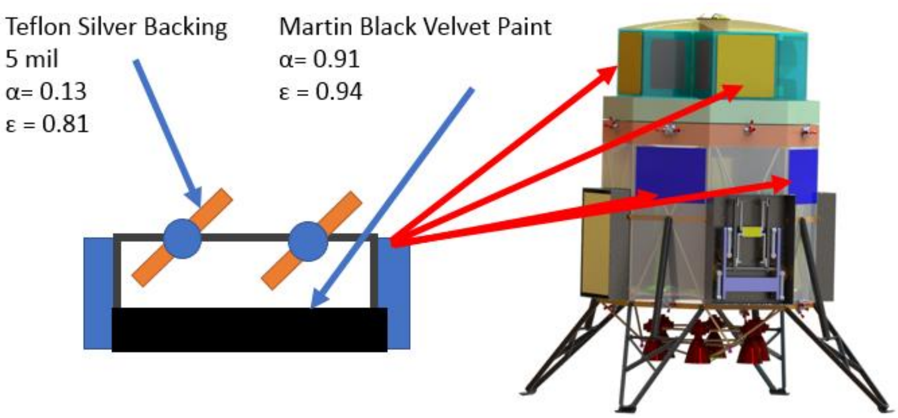


Figure 5.2.5.1.3-1: Radiating Louver Design and Location

Each of the louvers have an effective area of five m^2 with a maximum heat rejection of 171 Watts. Along with these radiators we will be using electronic resister heaters to maintain the spacecraft within the lower limit even in the worst cold case. These heaters will be placed all along the inside of the Hab to generate all the necessary heat.

Table 5.2.5.1.3-1 Mass, Power, and Volume of the Hab

Thermal Control System Crew Module			
Thermal Components	Mass (kg)	Power (W)	V (m^3)
Multi-layer insulation	5.52	0.00	1.47
Louvers	87.79	N/A	8.77
Heat Pipes	4.07	0.00	0.0028
Heaters	18.28	2611.38	negligible
Totals	**115.66**	**2611.38**	**10.25**

2.5.2 Ionizing Radiation Protection

Since our spacecraft will be transporting crew through an environment with high doses of radiation the spacecraft will have to have adequate shielding. NASA has a certain limit for the exposure limits depending on the mission duration as seen in Figure 5.2.5.2-1 where a rough estimate for the limit of a 20-day mission to be around 0.2 sieverts. We took the design approach of allocated a fixed amount of mass to the shield as an initial guess and then computing the effective dosage for the mission and checking to see if it was below NASA's limit.

Table 5.2.5.2-1 NASA's Astronaut Exposure Limit

Depth of Radiation Penetration and Exposure Limits for Astronauts and the General Public (in Sv)				
	Exposure Interval	Blood Forming Organs (5 cm depth)	Eyes (0.3 cm depth)	Skin (0.01 cm depth)
Astronauts	30 Days	0.25	1.0	1.5
	Annual	0.50	2.0	3.0
	Career	1-4	4.0	6.0
General Public	Annual	0.001	0.015	0.05

To meet this limit, we took an average exposure dosage per day of a lunar mission to be roughly 0.0566 Sv in a 30-day mission. This number was assuming that the spacecraft used a shielding of 5 g/cm2 of aluminum, but instead assumed that polyethylene was 60% more effective than aluminum based on Figure 5.2.5.2-1. Then used that value to compute an effective shield thickness and compared it to the initial effective dosage per day.

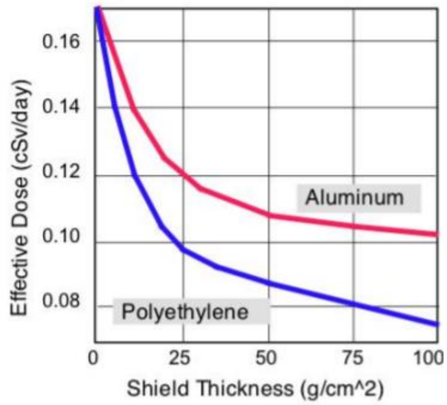


Figure 5.2.5.2-1: Comparison between Polyethylene and Aluminum

Then these effective dosages were calculated to be roughly 0.11 sieverts which allows for a big margin in case there is significant solar activity during our mission. We tried to keep the dosage “As Low As Reasonably Achievable” while staying with the designed allocated mass limit. Figure 5.2.5.1.1-2 illustrates how the radiation shield will be installed relative to the Hab outer structure wall.

Aluminum Shielding (g/cm^2)	Polyethylene Shielding (g/cm^2)	Total Shielding Thickness (cm)	Effective Dose Limit 30 day (Sv)	Calculated Effective Dose 30 day (Sv)	Margin of Safety
0.5	0.7	.92	0.25	0.11	2.3

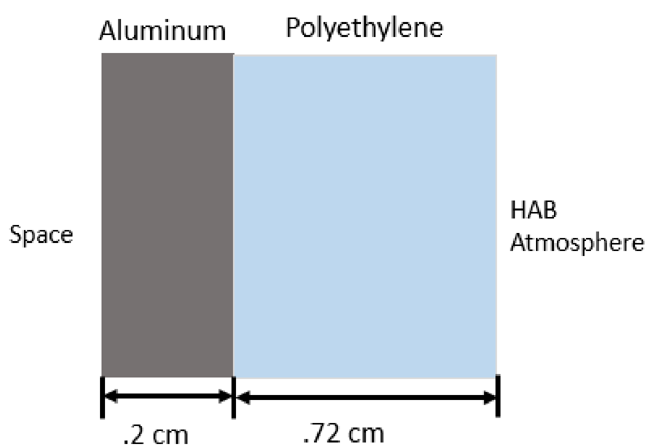


Figure 5.2.5.1.1-2: Radiation Protection Thickness

In case of an unexpected high dose of radiation the mission will have to be aborted while the astronauts wait in the DSG which should have more radiation protection and the mission can continue a later date. If some large wave a radiation does occur during the mission our margin of safety of 2.3 should be more than enough to keep any major health issues to the crew.

2.6 Mass and Power Breakdown for Hab

Table 5.2.6.1: Mass Breakdown Hab Module

HAB SUBSYSTEM	% Total Mass Estimated	Subsystem Mass Estimate (kg)	Subsystem Mass Calculated (kg)	% Total Mass Calc	E/C?
ECLSS	18%	900	1073	25%	C
Structural	28%	1400	1900	44%	E
Power	14%	700	718.7	17%	C
Thermal	30%	1500	115.66	3%	C
Cabling	10%	500	500	12%	E
Budget Total	100%	5000	4307.36	86%	
Margin	10%	500	1192.64		
On-Orbit Dry (No		5500	5500		
Propellant		0	0		
Pressurant		0	0		
On-Orbit		5500	5500		
Adapter		0	0		
Launch Prop Mass		0	0		
Launch Mass		5500	5500		

Table 5.2.6.2: Power Breakdown Hab Module

HAB SUBSYSTEM	% Total Power Estimated	Subsystem Power Estimate (W)	Subsystem Power Calculated (W)	% Total Power Calc	E/C?
ECLSS	68%	3400	3132	46%	E
Structural	10%	500	900	13%	E
Power	3%	150	150	2%	E
Thermal	19%	950	2611	38%	C
Cabling	0%	0	0	0%	E
Budget Total	100%	5000	6793	136%	
Margin	15%	750	867.15		
On-Orbit Power		5750	7660.15		

3.0 Landing Module

3.1 Module Summary

The Landing Module was designed from the design philosophies of, “low center of mass”, “sporty & stiff structure”, “single deployment legs with wide footprint” and “be sensibly modular”. To this end we also emphasized cargo placement from day one, allowing us to access our cargo reasonably close to the surface for low CG, low potential energy in the case of loading mishaps and quick surface operations.

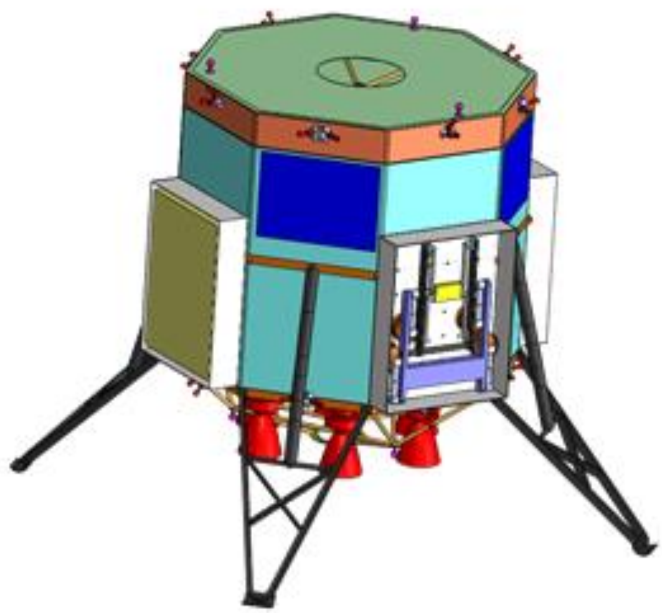


Figure 5.3.1-1: Lander CAD

3.2 Main Propulsion System

3.2.1 Propulsion system requirements

The propulsion system for this design begins with defining requirements that are needed to be met and fulfilled for every operating mission. The requirements are listed below in Table 5.3.2-1

Table 5.3.2-1: Propulsion System Requirements

Req. No.	Requirement text
1.1b-1	Propulsion system shall deliver at least 2,070 m/s total Δv per descent and ascent of lander
1.1b-2	Propulsion system shall be capable of restarting and operating within their lifetime capacity
1.1b-3	Propulsion system shall keep optimal thrust to weight ratio for descent and ascent off the lunar surface
1.1b-4	Propulsion system shall maintain optimal pressure of fuel to ensure max efficiency of engines

3.2.2 *Propulsion System Trade Study*

To meet these requirements, several propulsion system options were explored, including electric thrusters, solid motors, hybrid motors, and bipropellant engines (both cryogenic and non-cryogenic propellants). Each are evaluated by examining mission critical features and deemed feasible if these features are met.

Electric thrusters, specifically ion thrusters offer high efficiency with Isp values around 7000 seconds. Due to their low thrust capabilities and high-power consumption, limit the phases of the mission during which it can be used since a habitual mission is required. Additionally, to be able to compensate for the high-power consumption, an RTG would need to be incorporated to the design. If an Ion thruster were to be used, a liquid engine and ACS thrusters would still need to be incorporated for the landing phase due to the high thrust requirements and low burn times.

Solid motors are another commonly used propellant due to their simpler design and moderate Isp values of about 230-300 seconds. These motors could be used as kick motors and can be staged on the landers so that after each burn is completed, the spent kick motor can be jettisoned, reducing excessive mass for storage and tanks. The downfall to these kick motors is that after every mission phase the cores need to be replaced and casings need to be evaluated for possible modes of failure, reducing the turnaround time and adding complexity to maintenance. If a solid motor was to be used for the descent phase, a liquid engine and ACS thrusters would still be needed since solid motors cannot be throttled nor can they be restarted.

Hybrid motors are an up and coming propellant type due to their simpler design and moderate Isp values of about 250-400 seconds and ability to be turned off and restarted. These motors keep oxidizer and fuel separate, thus avoiding manufacturing and shipping hazards but increase design complexity over solids. The downfall to hybrid motors is that after every mission phase just like solids the cores need to be replaced and casings need to be evaluated for possible modes of failure, reducing the turnaround time and adding complexity to maintenance. If a hybrid motor was to be used for both descent and ascent phases, extra cores would need to be stored and brought adding to the overall total mass and decreasing efficiency at the same time with the added mass.

Propellant selection for liquid engines is divided into two categories, storable (non-cryogenic), and cryogenic. Cryogenic propellants tend to offer higher specific impulse and thrust values Non-cryogenic fuels offer great efficiency, about 300 sec and 400+ sec for cryogenic, but their temperature restrictions typically limit their usage to launch vehicles or added weight for active thermal control. The spacecraft propulsion systems we considered for our design were liquid engines. Both types can be restarted, are throttleable and no restrictions on availability. Below in

Table 5.3.2-2 each considered fuel type was scrutinized with the rocket equation to determine the amount of mass of propellant needed to complete a single cargo mission also considering the boil off created by the cryogenic fuels.

Table 5.3.2-2: Propellant Trade Study

Module	LOX/LH_2 (kg)	LOX/CH_4 (kg)	LOX/RP-1 (kg)	N_2H_4/N_2O_4 (kg)
LM	37703.8	49180.4	69665.2	61394.5
OM	24197.1	30793.8	45127.2	16744.7
SUM	61900.7	79974.2	114792.3	100639.1

As seen in the table above, the chosen propellant type is HydroLox due to having the lowest propellant mass for this mission. Since choosing this propellant type the need for another form of fuel for ACS thrusters has to be implemented into the spacecraft.

3.2.3 Engine selection

The main engine requirements are to maximize the thrust capability so that burn times are minimized and can be treated as instantaneous burns. The engines shall be capable of being throttled, restarting and have a sufficient lifespan to complete the designed mission. Each engine will be capable of restarting and providing adequate thrust. A trade study was done to review the engines being considered for the architecture, shown below in Table 5.3.2-3.

Table 5.3.2-3: Engine Selection

Engine	Manufacturer	Isp (s)	Thrust (kN)	Propellants
RL10C-1	Pratt & Whitney	449.7	101.8	LOX/LH2
Lighting 1	Firefly Aerospace	322	70.1	LOX/RP-1
RD-263	PA Yuzhmash	318	1040	N2H4/N2O4
Raptor	SpaceX	380	1900	LOX/CH_4

While each of these engines offered high Isp values for liquid engines, only the Pratt & Whiney RL-10-1 can produce the thrust necessary to minimize the burn time for the spacecraft, utilize the selected propellant type and is capable of being deep throttled down to 6% of its original thrust. The landing module will utilize 6 RL10C-1 engines operating from 100% - 6% thrust, allowing for mission completion.

During the descent and ascent phase of the mission the desired thrust to weight ratio is 6.1g (lunar) and is the equivalent of 1 earth g. When the landing module is fully fueled for the mission the thrust to weight max on descent is 5.8g but as propellant is used the mass of the spacecraft will lower and thus allow us to achieve the 6.1g needed for 1 earth g. The table below shows the thrust to weight for each mission phase and both lunar g and earth g that will be experienced.

Table 5.3.2-4: Thrust-to-Weight Ratios by Mission Phase

Mission Phase	**Mass at Phase (kg)**	**T/W_{max}**		**T/W_{min} (6%)**		**Desired T/W**	
		Moon g	Earth g	Moon g	Earth g	Moon g	Earth g
Descent	65046	5.8	1.0	0.3	0.1	6.1	1.0
Hovering	35704	10.5	1.7	0.6	0.1	1.1	0.2
Ascent	30704	12.2	2.0	0.7	0.1	1.4	0.2

3.2.4 Tank size and shape

The sizing of the tanks begins with the calculation of the propellant masses. Beginning with the landing modules dry mass and the ΔV required for the descent and ascent phase, the Tsiolkovsky rocket equation is used to determine the amount of propellant needed. A 0.25% per day for boil off, 10% margin for emergency maneuvers along with margins for residual propellant (2%), outage (1%) and loading error (0.05%) are added to the propellant mass. Using an O/F ratio of 5.50, the mass for the oxidizer and fuel can each be calculated, followed by the volume. The tanks caps were chosen to be in parabolic shape to maximize tank height and volume with keeping an optimum packing factor for placement geometry. The table and figure below show the breakdown of mass and volume of the tanks and optimal placement of tanks.

Table 5.3.2-5: Propellant Tank Sizing

Main Propellant Tank Sizing					
Module	Propellant	Volume (m^3)	Volume per Tank (m^3)	No. of Tanks	Mass per tank (kg)
LM	LH_2	107.7	21.5	5	459.4
	LOX	36.9	9.22	4	252.9

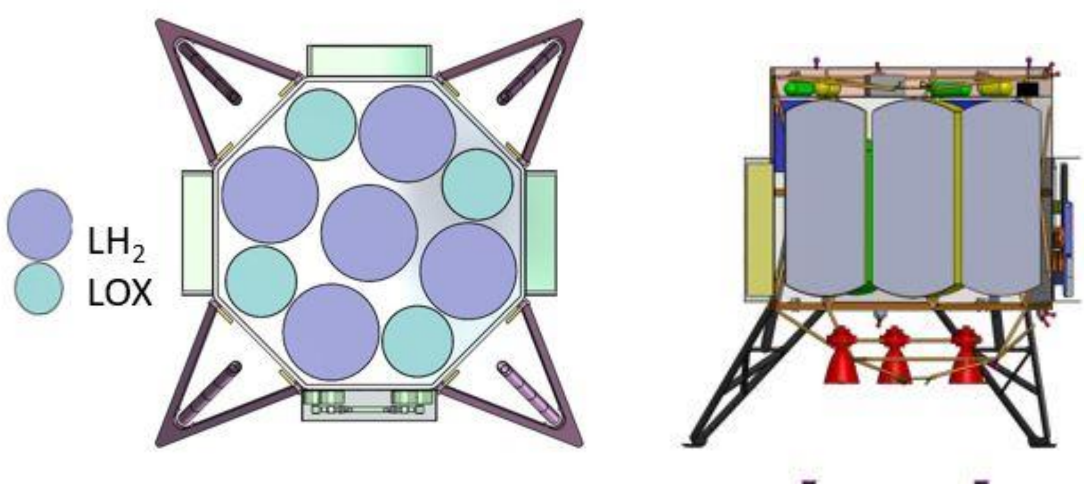


Figure 5.3.2-1: Tank size and orientation

For filling the landing module, a propellant transfer arm provided from the DSG allows the module to be filled. The landing module also has both fill and drain ports, the orbital module also has a propellant transfer arm on it allowing it to be filled from the landing module. The figure below shows how the landing module will be filled from the DSG.

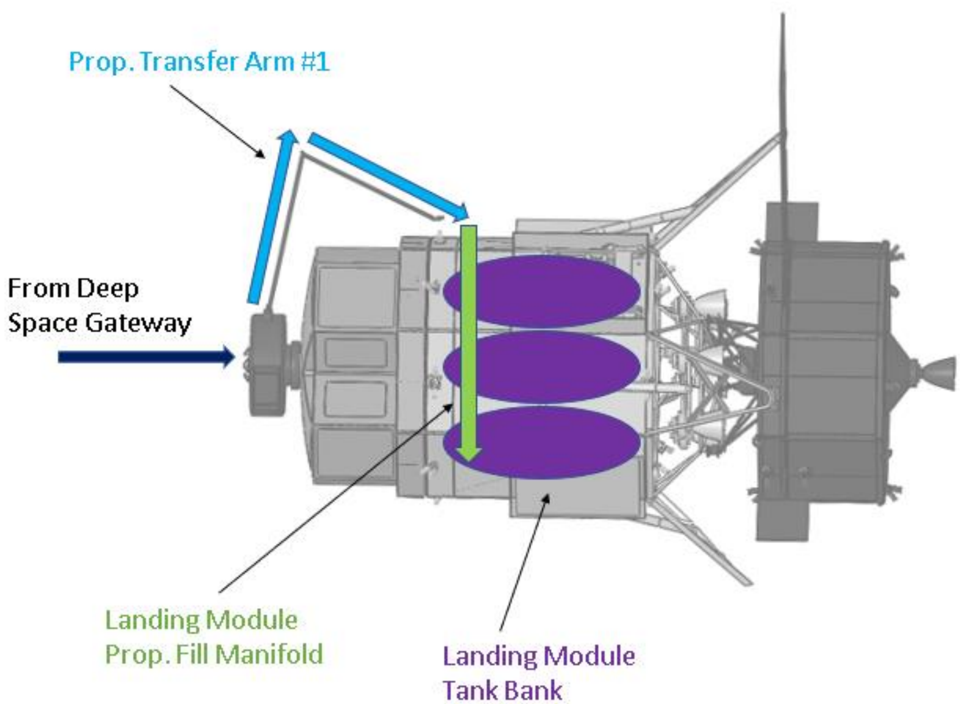


Figure 5.3.2-2: Filling of Landing Module Tanks with Cryo

3.2.5 Propulsion Schematic

In all, the propulsion subsystem consists of 6 main engines. The propellants, LOX and LH2, are pressure fed through a regulated system using helium while going through a series of check valves, filters and pressure regulators. The Figure below provides an overview of the entire subsystem, including the associated valves, regulators and sensors.

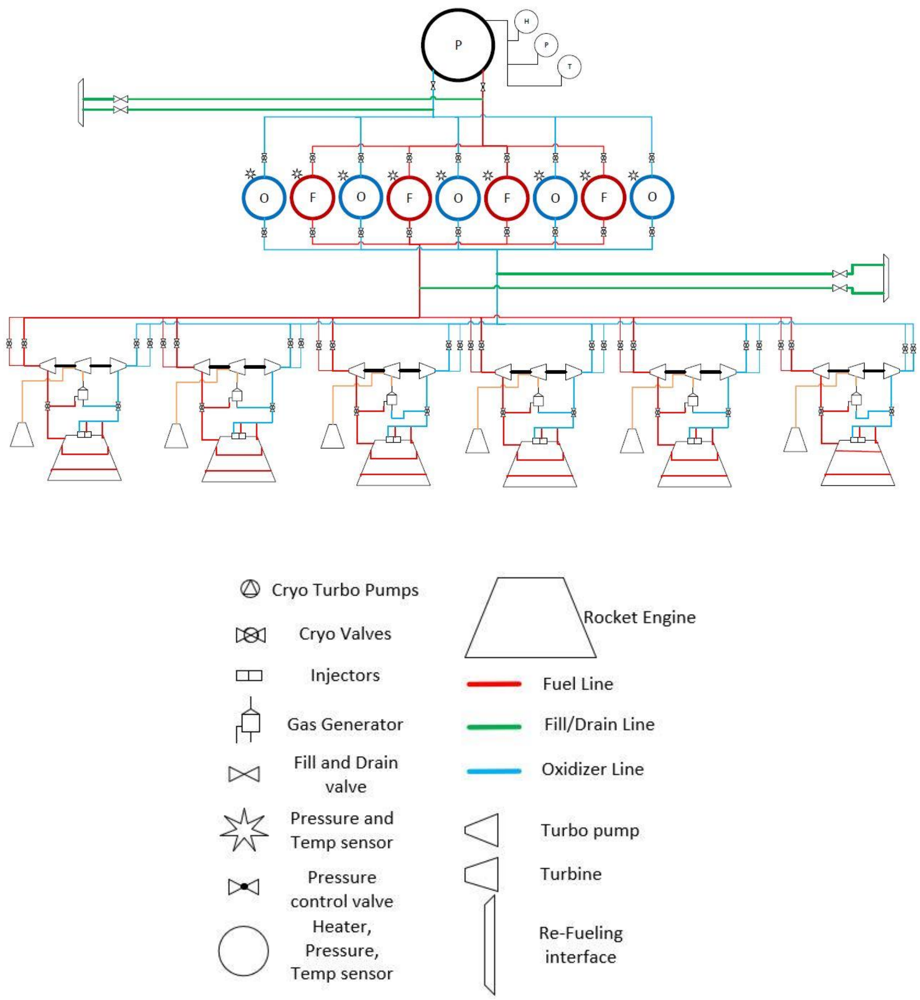


Figure 5.3.2-3: Propellant P&ID Schematic

3.2.6 Propulsion Mass and power Statement

The final mass of the propulsion system is 3769.5 kg when dry and roughly 43981.4 kg when wet. An itemized breakdown of the subsystem's components can be found on Table 5.3.2-6. The subsystem consumes 450 W when firing the main engines. Additional power consumption stems from valves and sensors for the propulsion system which approximately is 7 W.

Table 5.3.2-6: Propulsion System Mass & Power Statment

MEL LOX/LH2				
Landing Module				
Propellant		unit mass (kg)	number	Total mass (kg)
	useable	37703.729	1	37703.729
	unusable	1696.668	1	1696.668
	Total	39400.397	1	39400.397
Feed system				
	LOX Tank	252.868	4	1011.471245
	LH2 Tanks	459.398	5	2296.990
	LOX Tank Diaphragm	17.840	1	17.840
	LH2 Tank Diaphragm	32.491	1	32.491
	valves	2	36	72
	filters	0.3	2	0.6
	lines and fittings	5		5
	temp transducers	0.1	28	2.8
	pressure transducer	0.3	6	1.8
Engines				
	Rl10-C-1	190	6	1140
Mounting hardware				200
Total with prop				43981.390
Total with Mount Hardware				44181.390
Total w/o prop				3769.522

Module	Mass (kg)	Power (W)
LM	3769.5	450

3.3 Structures

The main requirements for the landing module's structures subsystem are summarized below.

Table 5.3.3-1: Landing Module Structure Subsystem Requirements

REQ #	Requirement Statement
1.2b-1	Shall maintain structural integrity during all launch, thrust, docking, and landing load cases
1.2b-2	Shall withstand the difference of pressure between the vacuum of space and internal pressure necessary for the survival of the astronauts
1.2b-3	Shall dock safely with both the Hab Module and Orbital Module
1.2b-4	Shall be capable of storing at least 22 m^3 of useful payload
1.2b-5	Shall be capable of landing at 2 m/s on the lunar surface

A mass and power statement for the LM's structures can be seen below.

Table 5.3.3-2: LM Structures Mass and Power

Item	Mass	Power	Qty	Mass Total	Power Total
	kg	W		kg	W
Engine Thrust Structure	80	0	1	80	0
Ball Lock Structure	131	0	1	131	0
Outer Skeleton	2930	0	1	2930	0
IDSS Passive	60	0	1	60	0
Landing Legs	100	300	4	400	300
Comm Dish Tilt Pan	4	25	4	16	100
			sum	3617	400

3.3.1 Static Structure

The SLS Block II's launch profile was assumed to induce, at worst case, 5 gs of axial stress and 3 gs of lateral stress. The structures and the materials they are composed of can be seen in the figure below.

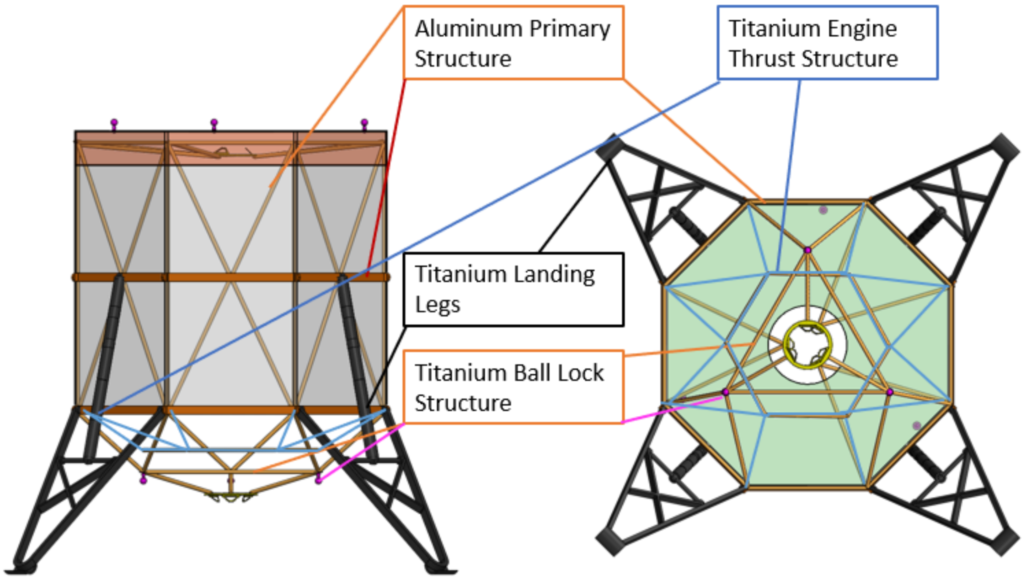


Figure 5.3.3.1-1: Structural Skeleton of Landing Module. Side View and Bottom View

The titanium ball lock structure, in its preliminary phase, is composed of round bars with a radius of 2.1 cm. These ball locks were assumed to distribute the weight of the spacecraft during launch evenly amongst themselves and the IDSS interface based off equivalent line loadings of the contact line for the ball locks and docking rings. These loads, combined with the moment reactions based off the lateral launch loads, produce a worst-case margin of 0.65 during launch. This analysis assumed the moment was reacted in the direction of the asymmetric corners of the triangular ball lock structure. An image of the finite element analysis done in FEMAP follows. The maximum stress occurs on the pink connecting bar towards the right side of the top view image.

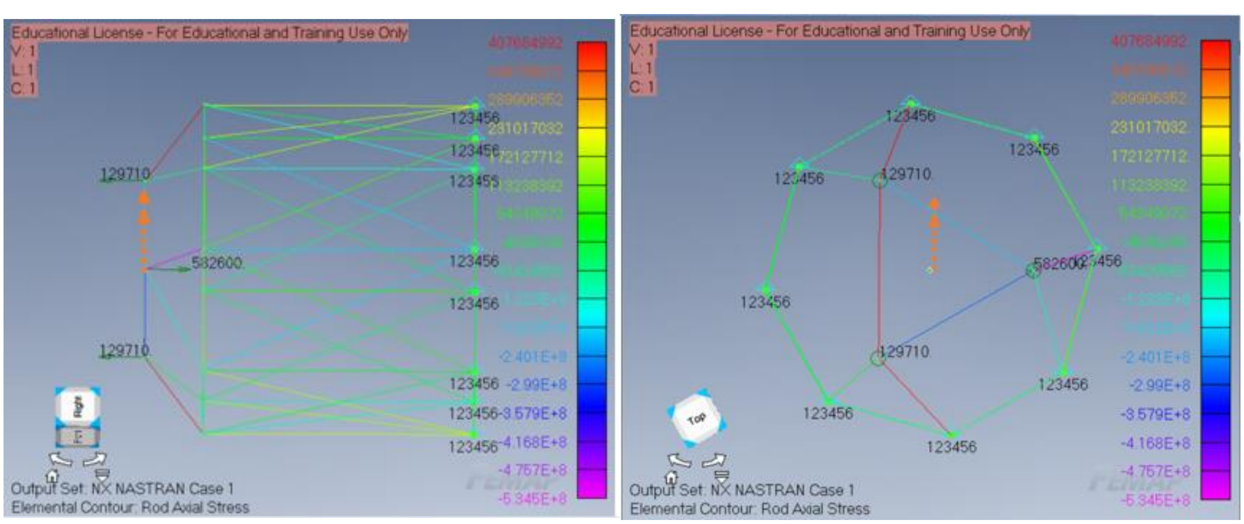


Figure 5.3.3.1-2: LM FEA: Worst Case Launch Load. Side View, Top View

The titanium engine thrust structure was designed as a bridge between the engines and the outer structure of the LM. The structure is composed of solid round titanium bars with a radius of 1.2 cm. This stucture was analyzed under maximum thrust coming from all engines. Under this load case, the system went through a margin of 0.11, with the maximum stress occuring in the bars connecting the two symmetrical halves of the bridge. An image of the finite element analysis can be seen below, with the bars which experience highest levels of stress boxed in white.

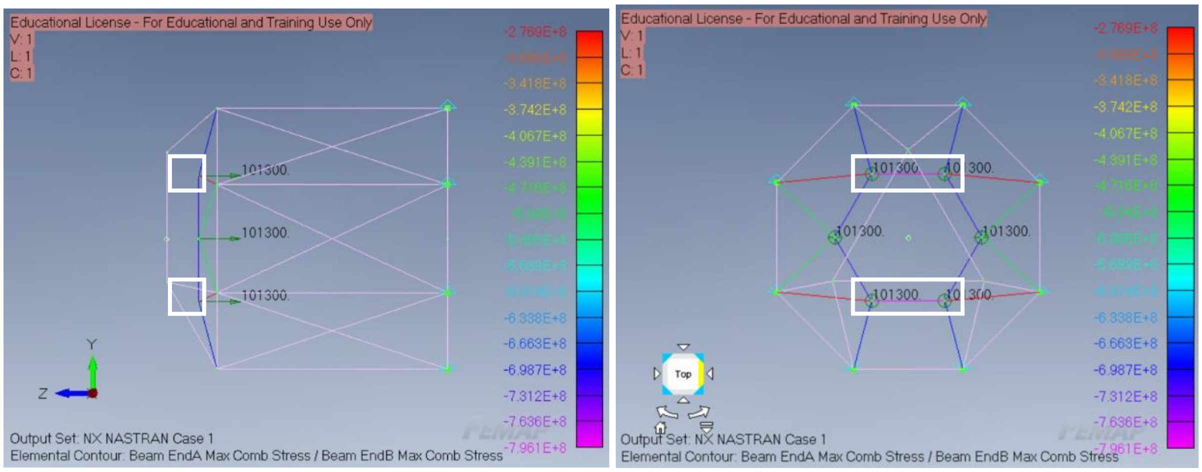


Figure 5.3.3.1-3: LM FEA: Max Thrust. Side View, Top View

3.3.2 Mechanisms

The main mechanisms located on the LM are the landing legs and IDSS docking rings, which were summarized in the Hab's mechanisms section, 5.2.3.2. The LM uses passive docking rings on both ends in order to minimize mass taken down to the lunar surface. The landing legs base their design off those of the Falcon 9, which have been trialed and proven to perform as necessary. Further analysis must be done to ensure the system maintains structural integrity while landing. The legs deploy once the vehicle is in space and remain deployed during the vehicle's lifetime. The landing module also utilizes pan tilt mechanisms in order to point the communications dishes wherever needed.

3.4 Thermal Control System (TCS)

The main purpose of the thermal control system is to passively reduce the amount of boiloff that occurs during the mission. This module has the largest tanks therefore has the potential for the largest amount of boiloff. The radiators used in this module is identical to the louvers used on the Hab, but their per louvers is 2.8 m^2 per louver.

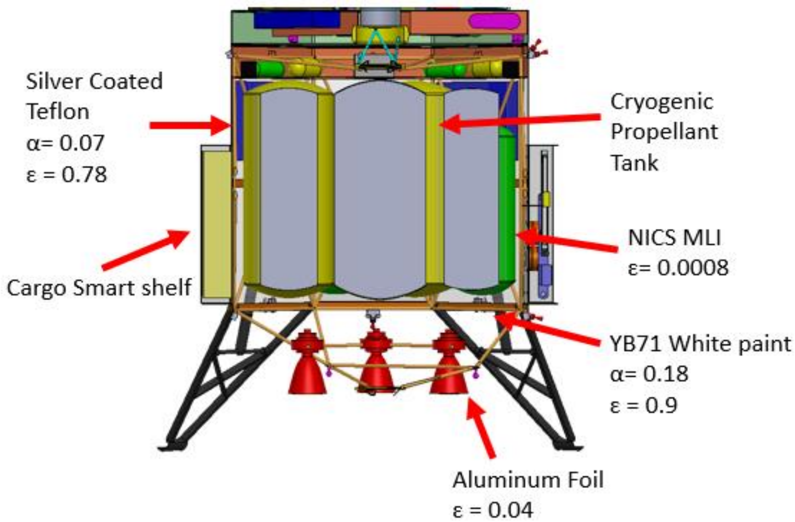
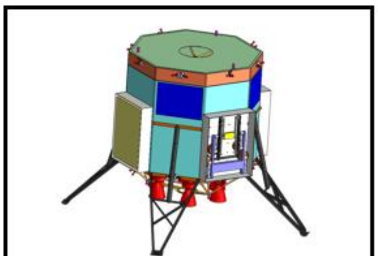


Figure 5.3.4-1: Landing Module Coatings & Finishes

3.4.1 Thermal Cases

The coatings and finishes were used to calculate the worst temperature cases as seen in Figure 5.3.4.1-1. The hot case and cold case for this module occur in the same situations as the Hab's cases. The Landing Module was assumed to be a different blackbody from the Hab and OM in order to calculate the necessary thermal control components.



Thermal Case	Temperature (°C)
Hot Case	36.4
Cold Case	-185.7

Figure 5.3.4.1-1: Landing Module Thermal Cases

3.4.2 Non-Interlayer-Contact Spacer MLI

To insulate the cryogenic propellant tanks from any source of heat trying to enter the system, a Japanese designed MLI will completely wrap the tanks. This new method of connecting the various layers reduces the heat transferred through radiation and conduction which in a vacuum like space are the only way for heat to be transferred between surfaces. Unlike the Hab the LM design temperature is significantly lower due to the low boiling point of the cryogenic propellants. Figure 5.3.4.2-1 illustrates the difference between conventional MLI and this new design, allowing the spacecraft to save more mass and power as it must reduce its total boiloff.

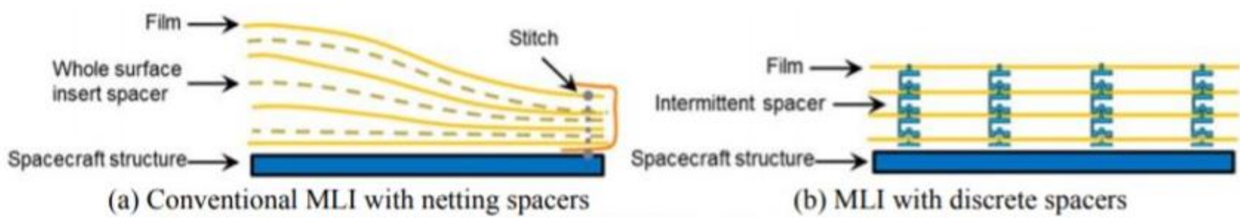


Figure 5.3.4.2-1: Schematic of Conventional MLI vs NICS MLI

Table 5.3.4-1: Mass, Power, and Volume of Lander Module

Thermal Control System Lander Module			
Thermal Components	mass (kg)	Power (W)	V (m^3)
Multi-layer insulation	49.68	0.00	1.26
Louvers	47.88	N/A	4.78
Heat Pipes	4.07	0.00	0.003
Heaters	9.97	1424.4	negligible
Totals	**111.61**	**1424.4**	**6.05**

3.5 Attitude Control System (ACS)

3.5.1 Thruster Locations

The ACS thruster locations must be positioned such that they provide ample maneuverability for docking, berthing, and other en-route adjustments or unexpected corrections. The thrusters should be able to produce both rotational and translational movement which will be essential in ensuring safe and efficient docking procedures. In the event of an emergency, the ACS thrusters should be capable of rotating or translating the spacecraft in any or multiple degrees of freedom. The thrusters must also be located such that they form coupled reactions that result in minimal movement due to center of gravity offsets. This becomes increasingly important because the LM is largely responsible for not only its own maneuverability, but also the *Hab*'s. Thus, the thrusters need to be able to be distributed such that the reactions about the center of gravity of both modules is minimal as well.

3.5.2 Sensor Locations

The LM's ACS system's sensors will be located as shown in Figure 5.3.5.2-1. Because the LM is capable of both docking and landing, it is necessary for the LM to have sensors related to landing hazard avoidance placed appropriately. These sensors will need to be located near the lower docking ring and will have to have line of sight directly below it which means that the main thrusters cannot obstruct the view of the sensors when they are firing upon landing. This will not be an issue for the infrared cameras, but the laser rangefinder and LIDAR doppler velocimeter will need to have a clear view of the landing site.

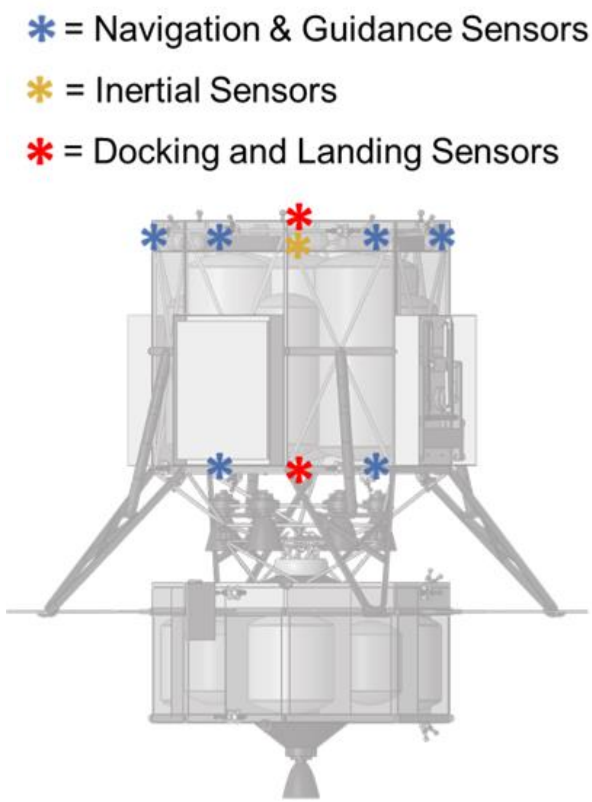


Figure 5.3.5.2-1: LM Sensor Locations

3.5.3 Mass & Power Statement

The mass and power statement for the ACS system can be found below.

Table 5.3.5.3-1: ACS Mass & Power Statement

Module	Mass (kg)	Power (W)
LM	558.3	372.8

3.6 Telecommunications

With the upcoming Orion missions, the Near-Earth-Network is the prime candidate capable of maintaining communications up to the moon. At the current rate that technology is improving and the birth of laser communications, currently ground stations across the nation are being upgraded to perform with new demands from technology. While current ground stations are being upgraded so comes a new ground station, KSC uplink station, currently supporting uplink Launch Operations. This project will expand the NEN capabilities and provide a new downrange station in Bermuda (NASA article). This insight offers candidate ground station locations for communications with our spacecraft and what to expect for the future of these missions.

Given the challenge for this subsystem is driven by the ability of our spacecraft to land anywhere on the Lunar Surface and still maintain Earth-comms. The DSG, as mentioned previously, with constant direct line of sight to the Earth may be used as a Comms relay during Lunar Surface Operations. Doing this will eliminate black-out periods, especially on far-side missions, on top of that, ease the budget and complexity of our telecom system. This telecom relay is best described below in Figure 5.3.6.1-1, demonstrating the communications links our spacecraft need to maintain, most critical of these of course being Spacecraft – Earth link.

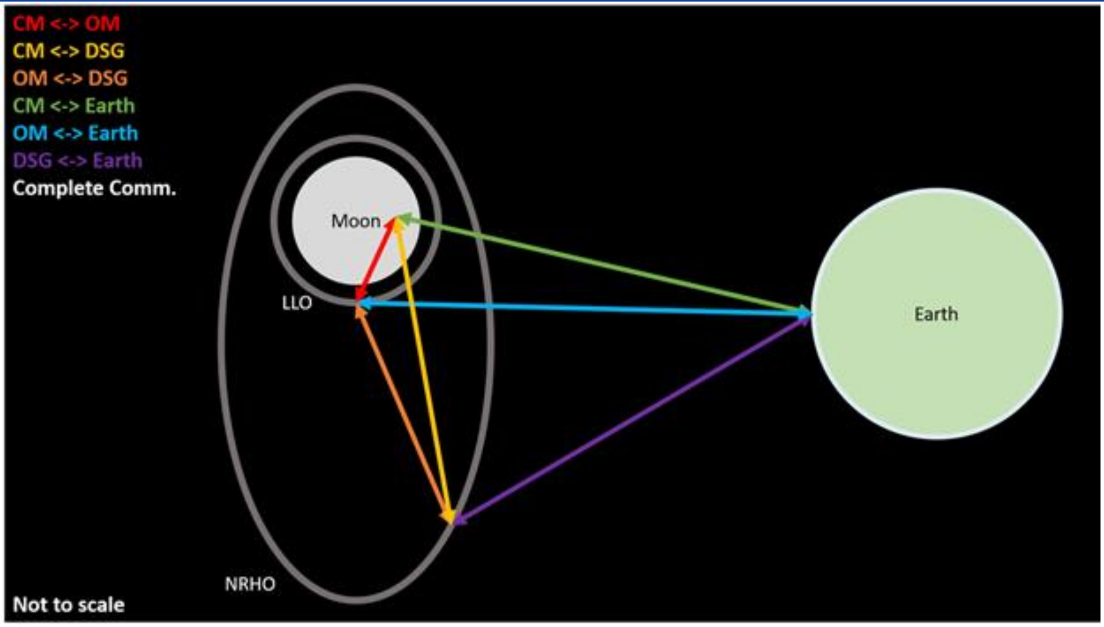


Figure 5.3.6.1-1 Complete telecom link diagram

CAVEMAN will be able to communicate over an S-Band frequency on the Near-Earth Network through its High Gain Antenna attached to the Landing Module and its omnidirectional mid-gain antennas found on both modules. The landing module, as mentioned, contains a dedicated high gain antenna when in direct line of sight, and three omni-directional mid-gain antennas that link with the orbital module, DSG or Earth. This creates a dual transmission over the S-band at 2300 MHz (HGA) and 2200 MHz (MGA) supported by our primary ground station located at Wallops, VA and secondary located at White Sands, NM, according to the Near-Earth Network (NEN) User's Guide.

3.6.1 Ground Stations – Wallops 11.3 and Backup – White Sands 18.3 m

Wallops, VA currently handles the data from the new KSC Uplink station and is intended to remotely handle launch support from ground stations at and near KSC. This Makes it the prime candidate to handle and consolidate the launch and communications network consisting of KSC, Goddard, JSC and Huntsville. As for White Sands, it currently handles the large volumes of data transmitting from the ISS routed through TDRSS. The Lunar Reconnaissance Orbiter used this antenna to downlink data, which in our case would make it another candidate to handle CAVEMAN's downlink data as well.

Table 5.3.6.2-1: Ground Station Link Budgets Important Results

System Budget Requirements	Wallops Downlink		White Sands Downlink	
Operating Frequency	2.30	GHz	2.30	GHz
Earth-Moon Distance	384000.00	km	384000.00	km
SNR required Eb/N0 Required	9.60	dB	9.60	dB
Margin	6.00	dB	6.00	dB
Data rate, Rd	84.77	dB-bps	84.77	dB-bps
Transmit Power, Pt	12.17	dBW	12.17	dBW
SNR Available, Eb/N0 Available	15.60	dB	15.60	dB
Required Transmit Gain, Gt	50.04	dB	45.86	dB
EIRP	62.22	dBW	58.03	dBW
Transmit Antenna Required Diameter, D	0.40	m	0.40	m

3.7 Command and Data Handling (C&DH)

The CAVEMAN's Command & Data Handling system is responsible for relaying, monitoring, and saving spacecraft state information, executing commands, performing orbit propagation calculations, determining necessary reactions, and error handling.

3.7.1 Hardware

Commands and data are handled by 4 onboard computers that cross-reference themselves for error checking. The computers process and/or store the data that they received from any of the other spacecraft subsystems. This includes monitoring the ECLSS equipment in the Hab to ensure that the interior atmosphere is suitable for life, performing attitude determination calculations using sensor input, and ensuring that the appropriate data regarding the physical state of the spacecraft is either stored or sent to the desired destination. Data will be stored on solid-state data recorders. Remote Terminal Units will be utilized as bridges between sensors and other electronic equipment and the onboard computers and will handle the incoming and outgoing data over all required protocols. Multiplexers and Demultiplexers will be used to increase the amount of possible data connections by lumping protocols that handle similar data together. Processing all this data and performing intermediate calculations will require a separate engineering data processor which will off-load the main flight computer's processors and allow them to execute commands and handle data more efficiently. Because the CAVEMAN is heavily reliant on communication between modules, each module will also contain a telecommunications interface that will allow the module or the crew to communicate with the other modules and the DSG and ground station. A full list of C&DH hardware is found in Table 3.7.1-1.

Table 5.3.7.1-1: C&DH Hardware Listing

Description	Mass (kg)	Power (W)	QTY
On-board Computer	26	80	4
Remote Terminal Units	10	34	2
TM/TC units	0.35	1.1	4
Solid State Data Recorder	4	12	4
Engineering Data Processor	10	5	1
Telecommunication Interface: **Data Flow Controls** **Signal Conditioning** **Command Processing** **Spacecraft Clock**	30	20	1

3.7.2 Software

The main purpose of the C&DH software is to safely, reliantly, and efficiently handle incoming data, process this data, and execute commands either remotely if in the cargo configuration or manually if in the crew configuration. A lines of code estimate was determined by considering both the Orion crew module which houses a similar number of crew and combining it with the Falcon Heavy to estimate lines of code for complex remote landing procedures.

Table 5.3.7.2-1: Lines of Code Estimate

Parameter	Estimate		Source	
HAB/ECLSS	2.00E+06	lines	Orion Capsule	Ref [1]
Landing/Docking	3.00E+05	lines	Falcon Heavy	Ref [2]
Complexity Factor	1.50	-		
Total	**3.45E+06**	**lines**		

Data rates for the C&DH system were determined by assuming the bit rate of the Space Network and by applying the CCSDS recommended coding protocols.

Table 5.3.7.2-2: Data Rate Determination

Function	CCSDS recommended coding	Symbols/bit	Symbol rate (ksps)
Commands	BCH(63,56)	1.125	337500.00
Data	R-S(255,223)	1.143497758	343049.33
Data, very low error	R-S(255,223)+Conv. Rate 1/2, K = 7	2.286995516	686098.65

Because the introduction of human interfaces can lead to accidental inputs that may not be desirable, the software will also be responsible for implementing human-error lock-outs. This will ensure that the commands that are entered are not accidental and will not result in unexpected spacecraft behavior. Another reason to include these lock-outs is the event that a crew member or teleoperator attempts to execute a command that would put the spacecraft in a dangerous state. The software should be able to know what the proper state of the spacecraft is and determine whether a command differs from this state.

The software is also able to self-diagnose itself in that when an internally generated error occurs, it can point directly to the source of the error allowing proper debugging procedures to both determine the severity of the error and implement a solution to fix it. A complete software function diagram can be found in Figure 5.3.7.2-1.

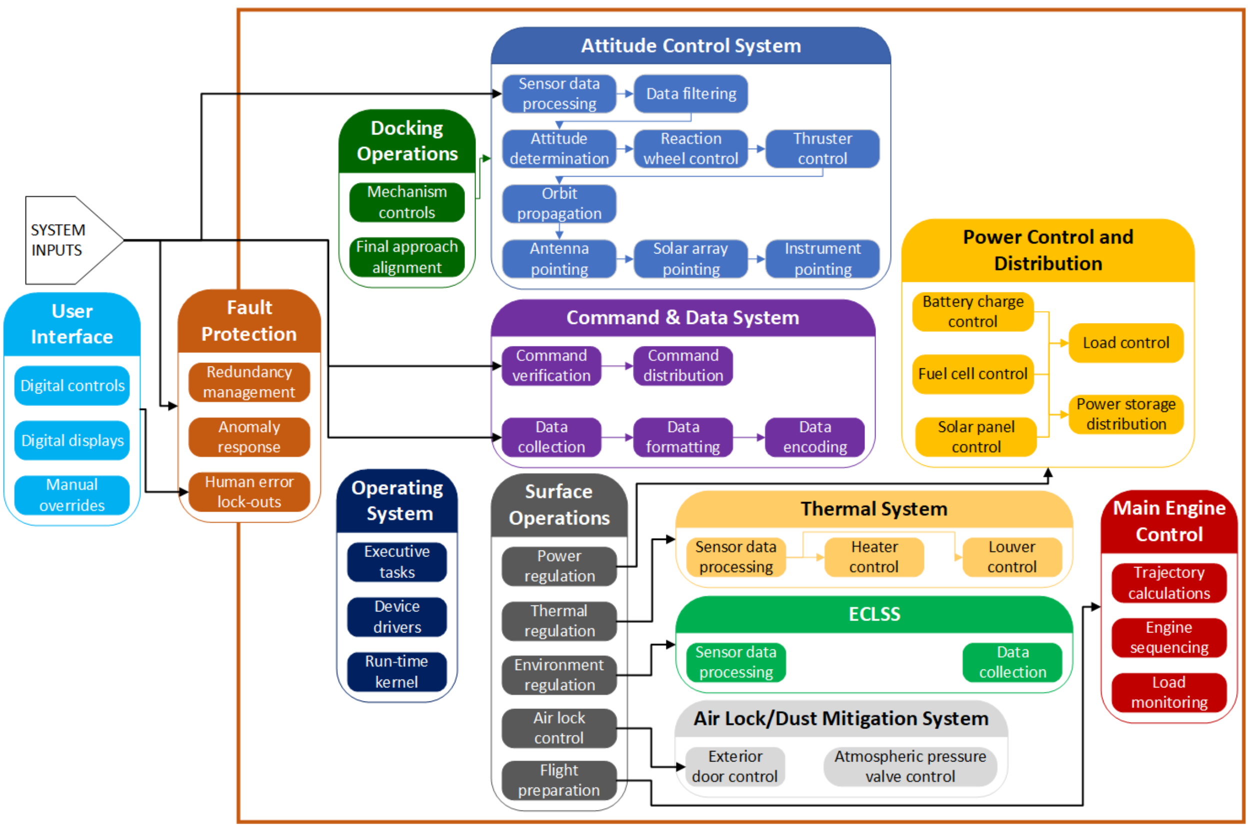


Figure 5.3.7.2-1: C&DH Software Function Diagram.

3.8 Electrical Power

The LM's power system is sized so that it can provide ample power for itself and the Hab simultaneously. This is done by use of 3 fuel cell units each with a nominal power output of 7kW. These fuel cell units will not operate at maximum capacity unless needed as the power requirements can be distributed amongst the three units. However, the LM will only need to supply power for the Hab in the event of crewed missions as outlined in Table 5.1.1-4. Because the fuel cells utilize the same propellant types as the LM's main propulsion system, they will receive their catalyzing propellants from the LM's propellant tanks. The propellant tanks are sized to account for the fuel cell's operational requirements.

Table 5.3.8-1: LM Power System Mass & Power Statement

LM Mass & Power Statement		
	Mass (kg)	**Power (W)**
Batteries	784.224	7270.0
Solar Arrays	0.000	0.0
Fuel Cells	1075.067	21000.0
Total	1859.291	28270.0

3.9 Cargo Loading and Unloading Bot (C.L.U.B.)

The final trade of how to take on and off cargo, came down to robotic arm(s) vs a "forklift style robot". The thought process here actually came down to landing proximity to cargo on the lunar surface. Because it was not specified within the RFP, there was no inkling on our development team as to what, where and why we would be loading cargo making us lean on a more versatile cargo loading system. Retrieving or unloading cargo on the lunar surface more than a few meters away from the lander is not a problem with said robotic forklift system. This robot is tethered to the lander for power and data and could be severed and left on the surface if necessary. For operation it can be remotely operated from the surface, inside or outside the Hab, the DSG or Earth. It travels to and from the lunar surface every mission and is stored in one of the cargo boxes which are radially placed along the craft. The wheels were sized with lunar soil pressure from the Apollo missions and the main lift is actuated to control center of mass.

Figure 5.3.9-1: Deployed Configuration CLUB

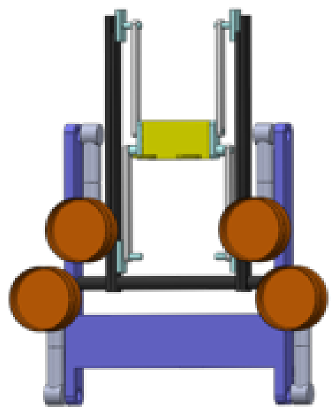


Figure 5.3.9-2: Folded Configuration CLUB

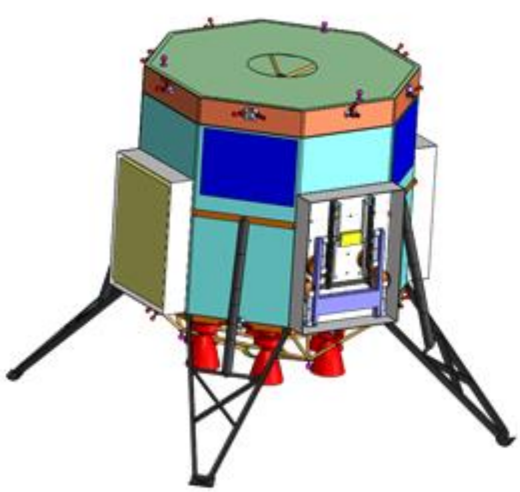


Figure 5.3.9-4: CLUB stored on side of Lander Module

Table 5.3.9-1: Requirements for CLUB

Requirements		
Packing Volume	< 8.85	m
Mass	< 100	kg
Vertical Reach	> 8000	mm
Static Stability	Reasonable	1
Must lift	> 1250	kg
FS	2	1

3.10 Mass and Power Breakdown for Landing Module

Table 5.3.10-1: Mass Breakdown Landing Module

LM SUBSYSTEM	% Total Mass Estimated	Subsystem Mass Estimate (kg)	Subsystem Mass Calculated (kg)	% Total Mass Calc	E/C?
Propulsion	40%	5040	3769.5	40%	C
Mechanical	30%	3780	3780	40%	E
ACS	9%	1134	558.3	6%	C
Telecomm	4%	504	29.58	0%	C
CDS	5%	630	213.4	2%	C
Power	5%	630	511.5	5%	E
Thermal	3%	378	112	1%	C
Cabling	4%	504	504	5%	E
Budget Total	100%	12600	9478.28	75%	
Margin	10%	1260	2590.46		
On-Orbit Dry (No		13860	12068.74		
Propellant		0	37703.8		
Pressurant		0	0.207		
On-Orbit		13860	49772.75		
Launch Adapter					
Launch Prop Mass			0		
Launch Mass		13860	12068.74		

Table 5.3.10-2: Power Breakdown Landing Module

LM SUBSYSTEM	% Total Power Estimated	Subsystem Power Estimate (W)	Subsystem Power Calculated (W)	% Total Power Calc	E/C?
Propulsion	4%	240.8	450	12%	C
Mechanical	5%	301	301	8%	E
ACS	11%	662.2	372.8	10%	C
Telecomm	30%	1806	146	4%	C
CDS	15%	903	1045.4	27%	C
Power	2%	120.4	120.4	3%	E
Thermal	33%	1986.6	1424.4	37%	C
Cabling	0%	0	0	0%	C
Budget Total	100%	6020	3860	64%	
Margin	15%	903	320.25		
On-Orbit Power		6923	4180.25		

4.0 Orbital Module

4.1 Module Summary

The Orbital Module was designed to be as simple as possible, although it does have several features that makes it a distinct spacecraft. These features include a robust passive thermal system to prevent boiloff of cryogenic propellants, a unique ball lock structure which is able to rigidize the lander to the tug during transportation, a propellent transfer arm and large actuated solar arrays.

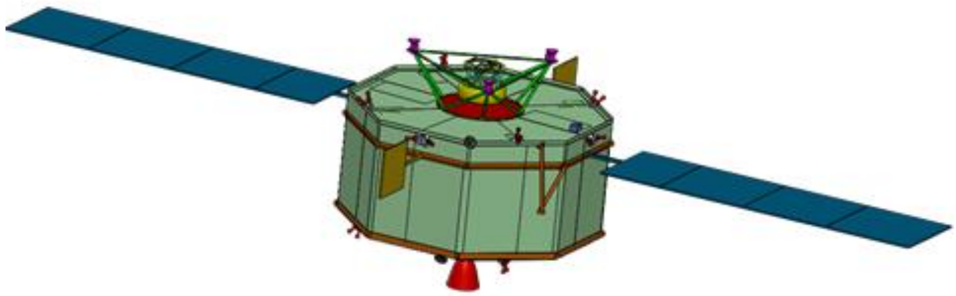


Figure 5.4.1-1: Orbital Module, Deployed Configuration

4.2 Main Propulsion System

4.2.1 Propulsion system requirements

The propulsion system for this design begins with defining requirements that are needed to be met and fulfilled for every operating mission. The requirements are listed below in Table 4.2.1-1.

Table 4.2.1-1: Propulsion System Requirments

Req. No.	Requirement text
1.1c-1	Propulsion system shall deliver at least 1,852 m/s total Δv
1.1c-2	Propulsion system shall be capable of restarting and operating within their lifetime capacity
1.1c-3	Propulsion system shall provide enough thrust to tug landing module from DSG to lunar orbit and back
1.1c-4	Propulsion system shall maintain optimal pressure of fuel to ensure max efficiency of engines

With these requirements the propulsion section for the orbital module went through the same propellant selection, engine selection, and tank sizing as seen in the landing module propulsion system section 3.2. The figures below show the tank orientation, filling flow and schematic.

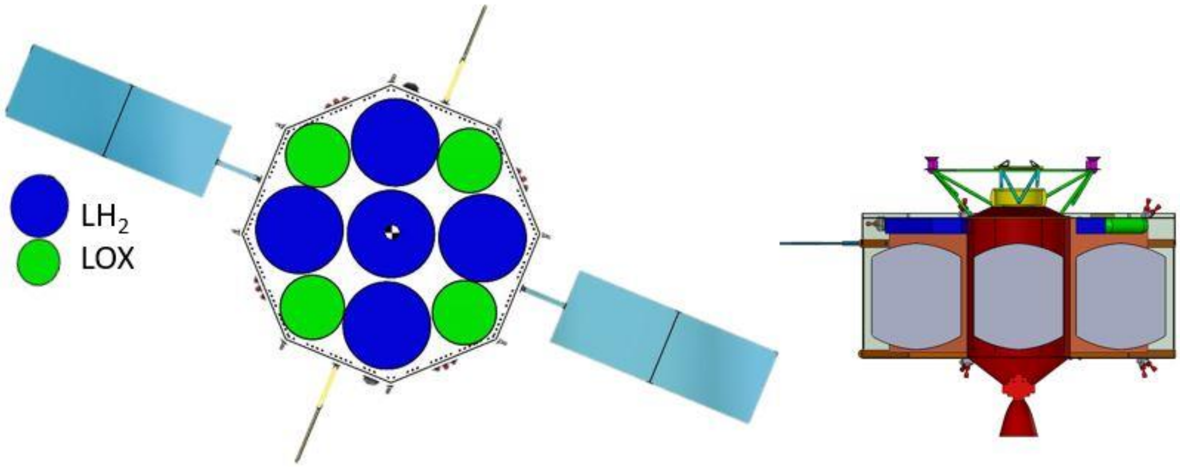


Figure 5.4.2.1-1: Tank orientation and cross section

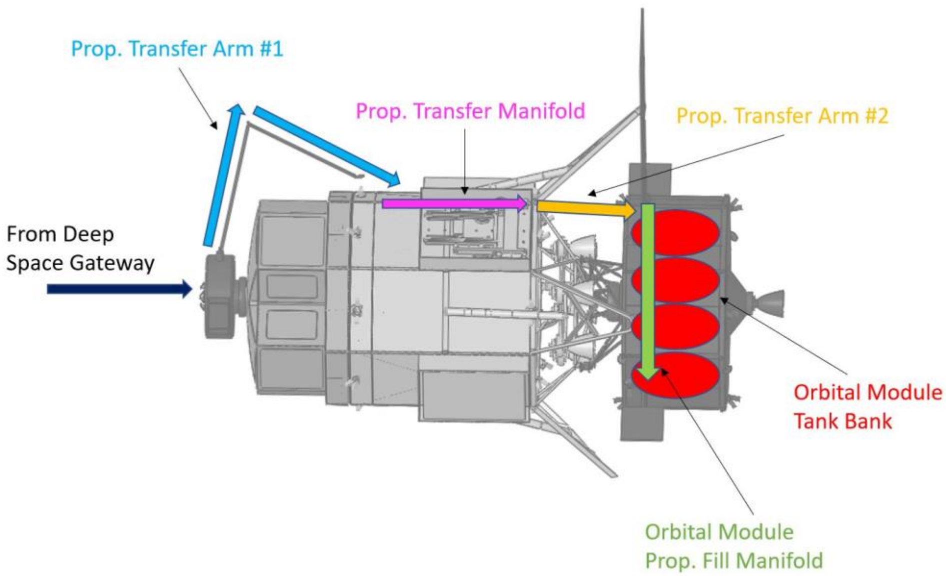


Figure 5.4.2.1-2: Fuel flow diagram

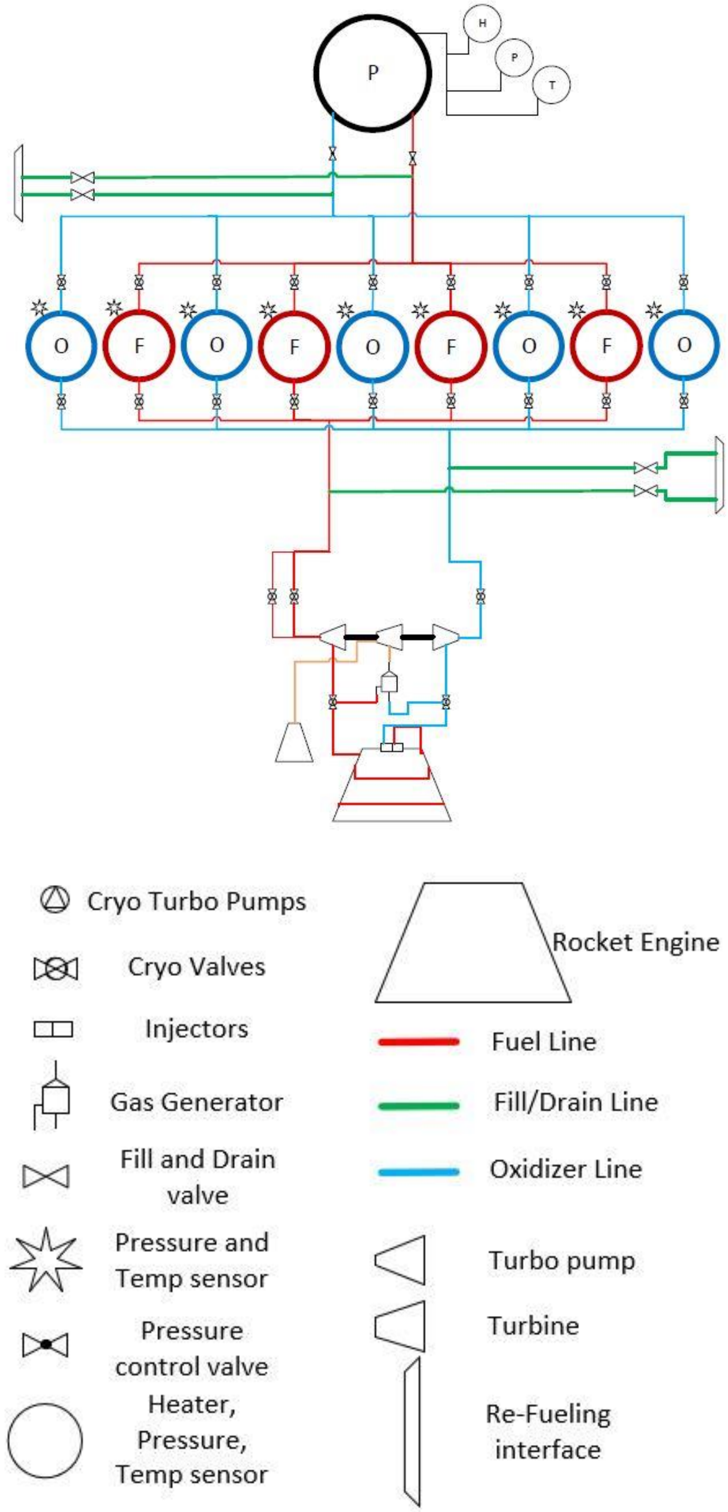


Figure 5.4.2.1-3 P&ID Schematic

4.3 Structures

Subsystem requirements for the Orbital Module's structures are summarized in the table below.

REQ #	Requirement Statement
1.2c-1	Shall maintain structural integrity during all launch, thrust, docking, and landing load cases
1.2c-2	Shall withstand the difference of pressure between the vacuum of space and internal pressure necessary for the survival of the astronauts
1.2c-3	Shall dock safely with the Landing Module
1.2c-4	Shall orient solar arrays towards the sun whenever necessary, as well as prevent solar panels from damage during docking maneuvers
1.2c-5	Shall orient the radiators as necessary for maximum heat dissipation when necessary

A mass and power statement for the LM's structures can be seen below.

Table 5.4.3-2: OM Structures Mass and Power

Item	Mass	Power	Qty	Mass Total	Power Total
	kg	W		kg	W
Inner Skeleton	1055	0	1	1055	0
Ball lock structure	38.3	0	1	38.3	0
Comm Dish Tilt Pan	4	25	4	16	100
J1 Deployment Solar Panel Spring System	1	12	2	2	24
J1 Damper Solar Pannel	2	20	2	4	40
Solar Panel Latch	1	15	2	2	30
Solar Panel Pan-Tilt	4	13	2	8	26
Solar Panel Twist	2	20	2	4	40
Radiator Tilt Motor	3	15	2	6	30
			sum	1135.3	290

4.3.1 Static Structure

The design approach for the Orbital Module was to take the loads from the payload attach fairing and the engine and allow them to travel through an internal skeleton towards the docking interface between the OM and the LM. The design for the OM structure can be seen below.

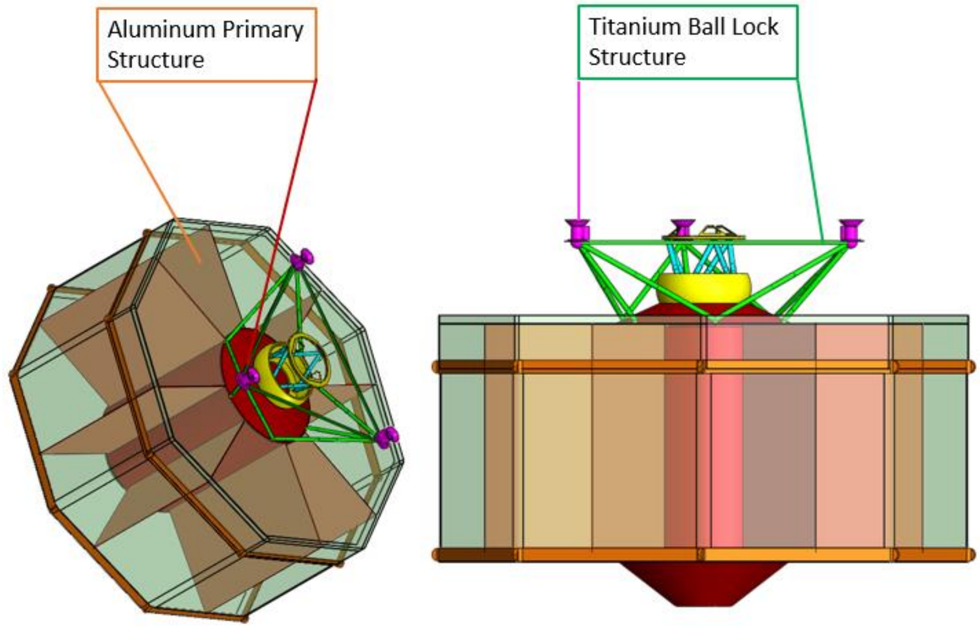


Figure 5.4.3.1-1: Structural Skeleton of Orbital Module

The worst-case launch loads were derived from the same scenario as that in section 5.3.3.1. These produced a bending moment as well as an axial load at the base of the cylinder. The axial stress on the structure is also carried by the panels connecting the inner cylinder to the walls of the OM. Atop the module is the docking structure which interfaces with the LM. The ball lock structure here was sized to a radius of 2.6 m. This structure was analyzed under the same loading as the LM's docking interface, and a minimum margin of 1.0 was found through FEA. The maximum stress occurred on the bar bisecting the angle of the triangle where the maximum load was placed. An image of this can be seen below, where the bar referenced is highlighted in a white box.

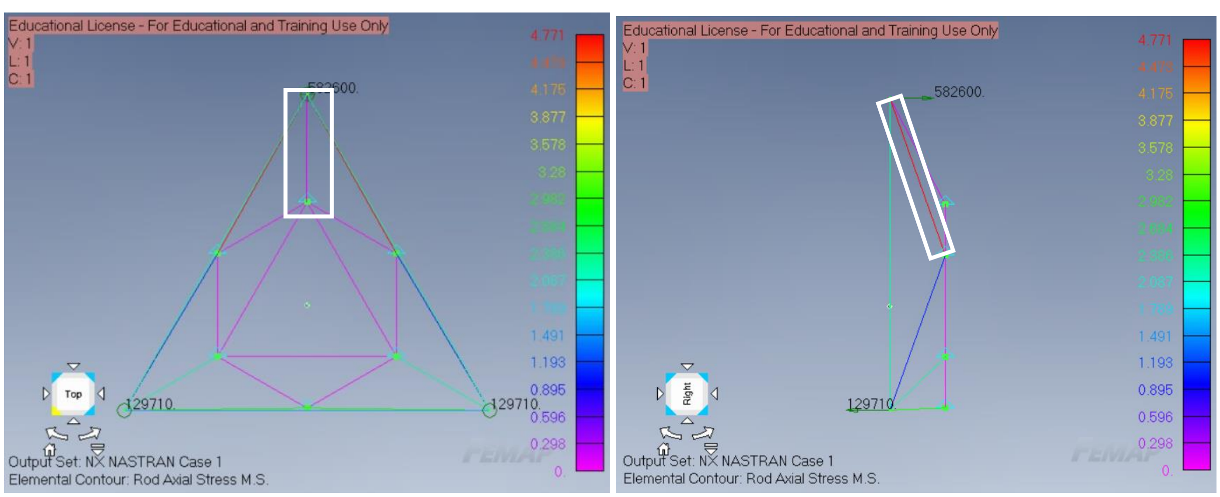


Figure 5.4.3.1-2: Orbital Module Finite Element Analysis: Launch Load

4.4 Thermal Control System (TCS)

4.4.1 Thermal Coatings

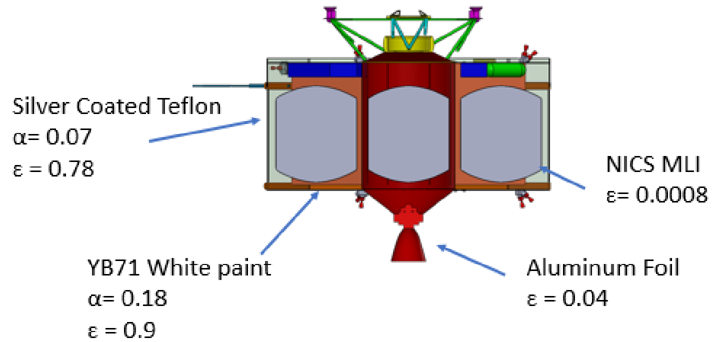
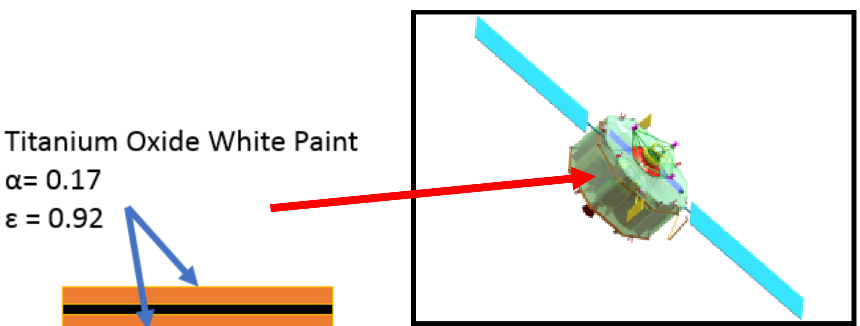


Figure 5.4.4.1-1: OM Finishes and Coatings

The coatings for this module need to reduce the amount of boiloff that occurs during the duration of the trip and it must do so passively in order to save mass and power.

4.4.2 Thermal Cases

Unlike the LM and Hab the OM never lands on the lunar surface, so its cold case only occurs when the module is in the complete shadow of the Moon and the hot case occurs when it is idling in a halo orbit in complete view of the sun. These were then used to size our radiators which are mounted on bars with heat pipes to transport the heat to the radiator and they have a tilting joint, so the radiator can always stay perpendicular to the sun and reduce the amount of solar energy entering the system while maximizing the amount of heat they can reject.



Thermal Case	Temperature (°C)
Hot Case	30.7
Cold Case	-179

Figure 5.4.4.2-1: OM Thermal Cases

4.4.3 *Sun Shield Design*

Based off the design of the James Webb Space Telescope the best way to keep a system cool passively is using a Sun shield. This reflective layer will be wrapped around the exterior structure of the spacecraft to reduce the initial amount of heat added to the system. Then it is connected to the structure through very small thermal isolators to reduce the amount of heat conducted through the joints. The Figure below illustrates the design of the sun shield and the various layers and material that will be used to minimize the total amount of energy entering the system. With the inside layers of MLI are used to reduce the amount of heat radiation that will reach the cryogenic tanks. Using 1-dimentional heat transfer an expected boil off to lower than the designed limit of 0.25 % per day. In table 4.4.3-1 each of the modules with a sun shield have a margin of safety allow less than a fourth of a percent per day allow for less total propellant need for each mission. This also allow that extra propellant to be used in emergency maneuvers and reduce the extra power normally required to prevent cryogenics from boiling off a significant amount.

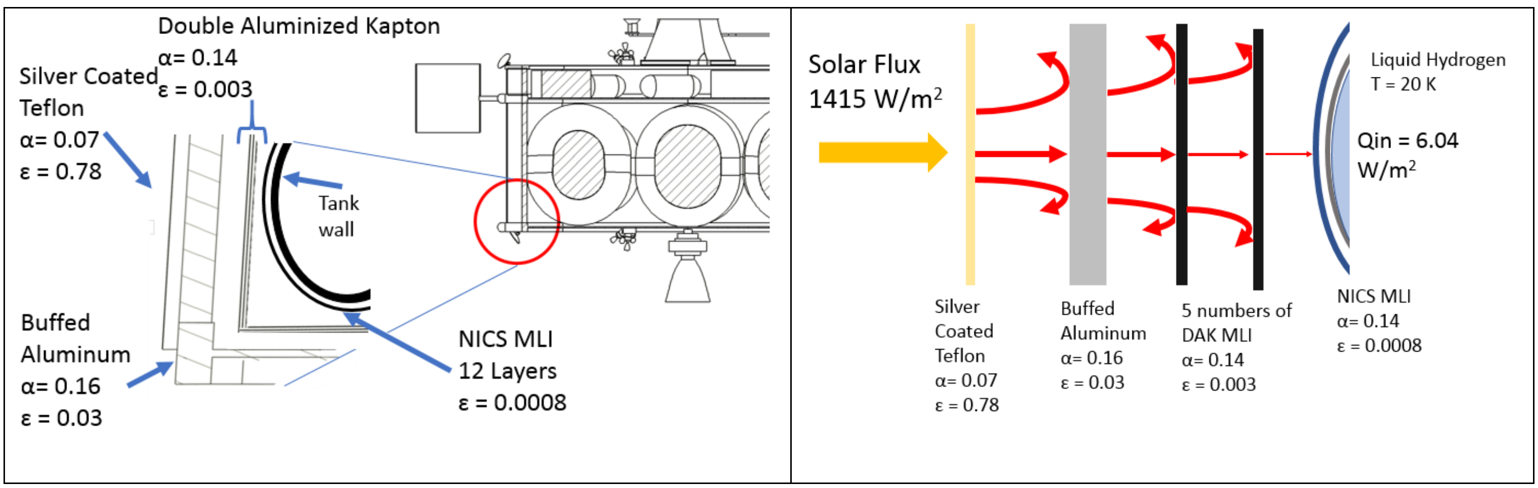


Figure 5.4.4.3-1: Sun Shield Design

Table 5.4.4.3-1: Sun Shield Calculated boiloff

Module	Design Limit (tons) (0.25 % per day)	Number of Layers	Calculated Boiloff	Margin of Safety
Cargo	2.81	3	1.66	1.7
Orbital	0.57	5	0.47	1.2

4.5 *Attitude Control System (ACS)*

4.5.1 *Thruster Placement*

The thruster placement on the OM is relatively straightforward. The main goal is to minimize the amount of unintentional rotation and translation due to C.G. offset. The primary purpose of the thrusters on the OM is docking alignment with the LM and station-keeping while in a parking LLO orbit. When docked with the LM it will also provide support thrust to aid with maneuvering the entire spacecraft.

4.5.2 Sensor Placement

Sensor Placement on the OM is very similar to that of the LM. A near identical set of sensors will be used for both attitude determination and docking alignments. The main difference between the OM and LM's set of sensors is that the OM does not have to perform any landing site evaluations and thus does not need a hazard detection system like the ALHAT.

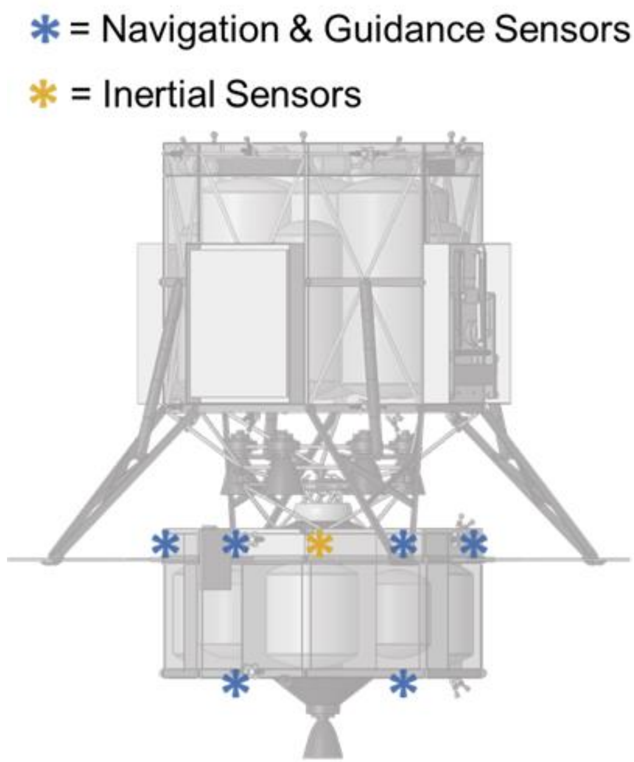


Figure 5.4.5.2-1: OM ACS Sensor Placement

4.6 Telecomm

4.6.1 OM Telecomm System Overview

The CAVEMAN Orbital Module comes equipped with three omni-directional mid-gain antennas (MGA) for linking with Landing Module to Earth or DSG. This set-up eases ACS maneuvers when orbiting, no matter which plane the module is facing one will always be in line of sight to the LM. The OM in other words would serve to complete the communications relay during Lunar Surface Operations.

4.6.2 Mass and Power Statement

The mass, power and data tables of the telecommunication subsystem were calculated to accommodate both S-band ground stations utilizing the values recommended by *Human Spaceflight: Mission Analysis and Design*[#]. Tables 4.6.1 and 4.6.2 demonstrate the total power needs for both the LM and OM respectively. Both modules come equipped with a set of MGAs for both redundancy and dual transmission with the HGA on the LM.

Table 5.4.6.2-1 LM Telecommunications Mass and Power Statement

Telecomm Instrumentation LM			
Telecomm Components	**Qty**	**Mass (kg)**	**Power (W)**
High Gain Antenna	1	3.62	0.00
Intermediate Gain Antenna	3	6.30	0.00
Receiver	1	3.60	33.00
Transponder	2	9.80	36.00
Transmitter	1	2.30	15.00
SS Power Amplifier	2	3.96	62.00
Total		29.58	146.00

Table 5.4.6-2 OM Telecommunications Mass and Power Statement

Telecomm Instrumentation OM			
Telecomm Components	Qty	Mass (kg)	Power (W)
Intermediate Gain Antenna	3	6.30	0.00
Receiver	1	3.60	33.00
Transponder	1	4.90	18.00
Transmitter	1	2.30	15.00
SS Power Amplifier	2	3.96	62.00
Total		21.06	128.00

4.7 Electrical Power

The electrical system for the OM consists only of batteries and solar arrays. The reasoning for this decision is discussed in section 1.1.1. The solar arrays are sized such that they are large enough to both charge the onboard batteries as well as provide power for all subsystems simultaneously. The solar arrays are also sized to the amount of time in the sun that they experience. Because LLO does not guarantee direct line of sight with the sun for the duration of the mission, it is necessary to examine a worst-case situation in which the time in the sun is minimum. The solar array sizing calculations also consider various solar panel efficiency parameters and solar cell packing factor.

Table 5.4.7-1: OM Power System Mass & Power Statement

OM Mass & Power Statement		
	Mass (kg)	**Power (W)**
Batteries	13.104	3854.4
Solar Arrays	451.570	8709.8
Fuel Cells	0.000	0.0
Total	464.674	12564.2

4.8 Mass and Power Breakdown Orbital Module

Table 5.4.8.1: Mass Breakdown Orbital Module

OM SUBSYSTEM	% Total Mass Estimated	Subsystem Mass Estimate (kg)	Subsystem Mass Calculated (kg)	% Total Mass Calc	E/C?
Propulsion	40%	2160	1508.7	41%	C
Mechanical	13%	702	962	26%	C
ACS	11%	594	305.7	8%	C
Telecomm	6%	324	21.06	1%	C
CDS	7%	378	213.4	6%	C
Power	13%	702	237	7%	C
Thermal	4%	216	64.05	2%	C
Cabling	6%	324	324	9%	E
Budget Total	100%	5400	3635.61	67%	
Margin	10%	540	2922.49		
On-Orbit Dry (No		5940	6558.1		
Propellant			24197.1		
Pressurant			0.323		
On-Orbit		5940	30755.52		
Launch Adapter					
Launch Prop Mass			3000		
Launch Mass		5940	9558.1		

Table 5.4.8.2: Power Breakdown Orbital Module

OM SUBSYSTEM	% Total Power Estimated	Subsystem Power Estimate (W)	Subsystem Power Calculated (W)	% Total Power Calc	E/C?
Propulsion	4%	39.2	75	3%	C
Mechanical	5%	49	300	14%	E
ACS	11%	107.8	327.8	15%	C
Telecomm	30%	294	128	6%	C
CDS	15%	147	1045.4	48%	C
Power	2%	19.6	75	3%	E
Thermal	33%	323.4	227	10%	C
Cabling	0%	0	0	0%	C
Budget Total	100%	980	2178.2	222%	
Margin	15%	147	38.1		
On-Orbit Power		1127	2216.3		

VI. Systems Engineering

1.0 System Summary

The formal systems engineering discipline focused on the Lunar Surface Access Vehicle's ability to become operational in the next decade. This included looking at subsystem manufacturers and locations, transportation, budgeting and scheduling. The final Assembly, Test and Launch Operations (ATLO) happen at the cape due to the ability to receive oversight from NASA and reduced transportation risk, assuming that V&V and integration happens both at the vendor level and the ATLO level. All of Pre-A, A, B and C phases were budgeted the maximum recommended time as per NASA advising and cost was attempted to be frontloaded so that the project could uncover large, "bombshell" issues before they resulted in a significant amount of schedule slip due to tight deadlines.

2.0 Manufacturing Concept

2.1 Candidates

An image of potential candidates and their locations who will carry out the manufacturing of our design, can be seen in figure 2.1.1 below. Specifically, we are seeking companies that are not only certified but are technologically capable and skilled with proven flight heritage. This ensures delivery of a quality product up to standards ISO9001/AS9100 Rev D. Preferably in continental US for ease of assembly integration and logistics. From the figure below Assembly, Test and Launch Operations will happen at Kennedy Space Center. As far as assembly goes, only launch integration, not the full assembly.

Figure 6.2.1-1 Contractor, HQ and ATLO locations

Electronics/Telecomm Components: General Dynamics located in San Jose, CA

Propulsion System: Aerojet Rocketdyne located in El Segundo, CA

HQ: Lunacy Solutions located in Pomona, CA

Mechanical Systems and Components: Honeywell located in Torrance, CA

Robotic Arm: NASA Goddard located in Greenbelt, MD

ATLO: Kennedy Space Center located in Merritt, FL

2.2 Manufacturing Design and Validation Methods

Inspection, Performance Qualification, Tooling and testing of the different sub-components with benchmark forms to ensure they meet specifications, clean room maybe necessary for some parts. Both modules of the spacecraft have been designed to be similar, emphasizing ease in manufacturability and assembly of CAVEMAN. Both modules of CAVEMAN are to be manufactured in sections, utilizing its octagonal design this is best demonstrated in Figures 6.2.2-1 and 6.2.2-2. The sections of both modules come together at the central frame, which as mentioned earlier will be the section of the frame experiencing most of the loads.

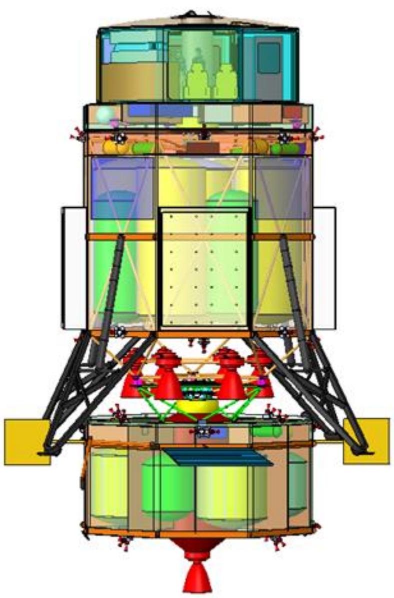


Figure 6.2.2-1: Stowed CAVEMAN configuration with internals shown

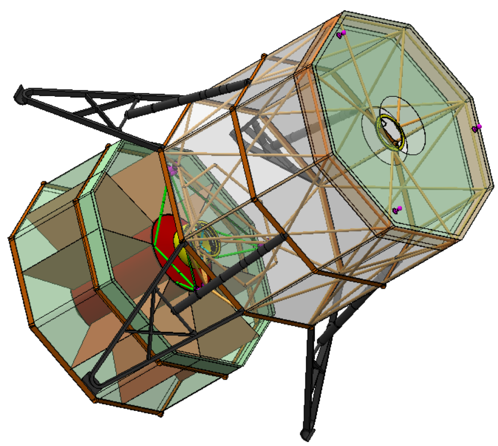


Figure 6.2.2-2: Stowed CAVEMAN configuration without internals shown

3.0 Maintenance Concept

The routine maintenance for this spacecraft is to check valves, communication check, propellant and engine inspection and suit lock leak check. Each of these maintenance implementations can be categorized into three modes of maintenance organizational, intermediate, and critical. Organizational concept requires minimal skills and a simple IVA will suffice. Intermediate concept requires high skill level and requires a EVA and EVR to complete. Critical is a concept where the manufacturer need to redesign and fix an example of this would be a complete engine failure. The figure below shows the progression of the maintenance during the mission life cycle.

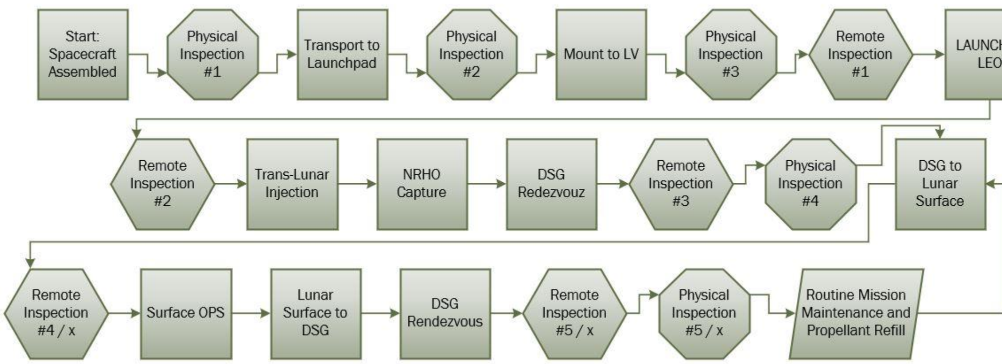


Figure 6.3.0-1: Maintenance flow chart

4.0 Disposal Concept

For this spacecraft design three types of disposal concepts were chosen to be analyzed. One; a lunar impact where at the end of life of the spacecraft it crashes down into the lunar surface. Two; the spacecraft is parked on a lunar with no reentry to the surface. Third; park the spacecraft at the DSG to be a permeant module. The selected concept is to keep permanent residency at the DSG to provide electrical power, habitual human space, propellant and cargo storage space.

5.0 Risk Analyses

The space environment, as demonstrated earlier, introduces harsh scenarios the CAVEMAN must endure. From radiation exposure to variance in temperature. These risks are not only representative of environmental effects, but also mechanical and electronic anomalies CAVEMAN may endure and must mitigate throughout the course of its mission. The following table, 6.5.0, demonstrates risks our system will most likely experience throughout its missions. Following the Risk table, Figure 6.5.0-1, incorporates these risks into a risk cube demonstrating the severity and probability of each of those risks.

Table 6.5.0 Summary of Risk Analysis

Program Risks		
Risk #	**REQ #**	**Risk Description**
1	1.1.1-1	Spacecraft engine ignition failure
2	1.1.2-1	Propulsion system valve malfunction
3	1.2.1-1	Docking mechanism deployment failures
4	P0.2-1	Contractor delays delivery of a product
5	P0.2-2	Loss of staff

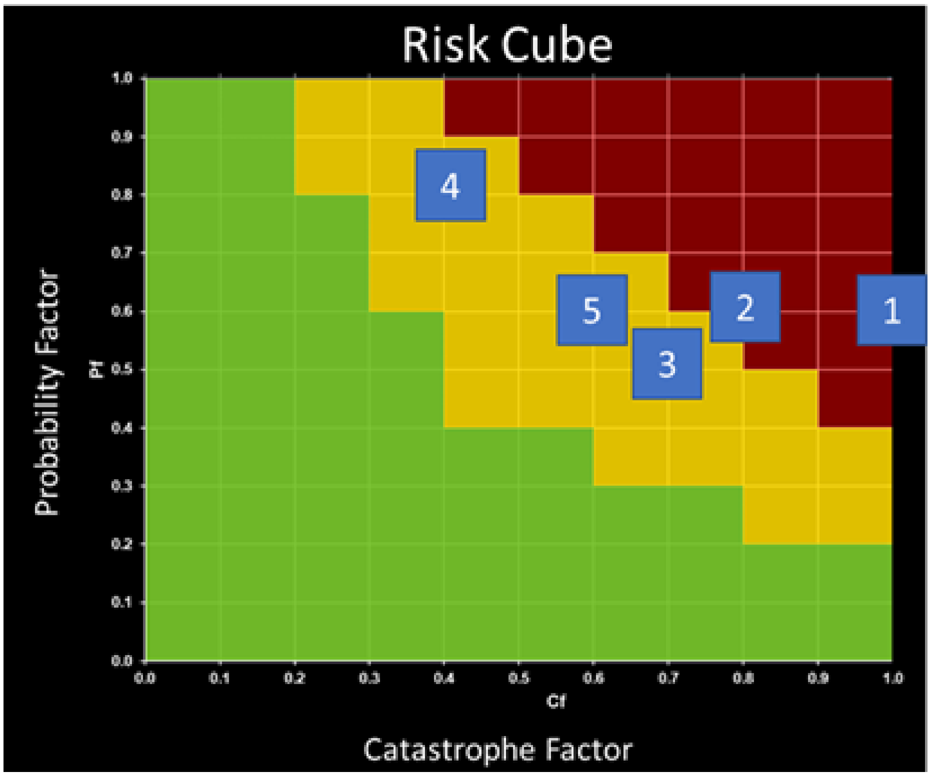


Figure 6.5.0-1 Fault Analysis Risk Cube

5.1 Risk Mitigation

An important note, the risks listed above are not the entirety of the risks associated with CAVEMAN, instead they summarize some of the programmatic risks we may encounter. The risk summary is a growing document that encompasses all the subsystems in CAVEMAN and may be encountered at any step of the Systems Engineering Process. Following the risks comes the mitigation process and how either management or a subsystem will respond to these risks, the following figures demonstrate an example of the mitigation activities responding to one of the risks found above. The mitigation process as follows presents the affected requirement followed by a risk statement and the mitigation activity in response to the risk.

REQ 1.1.1-1: Spacecraft must deliver enough thrust to safely transport crew and cargo.

Risk Statement 1.1.1-1: If an engine were to fail to ignite, the spacecraft fails to safely deliver crew and/or cargo.

Mitigation: Through rigorous testing of the engines before flight and redundancy capabilities at Critical Mission Phases.

Mitigation Activities	Date Complete	Evaluation		
		Cf	Pf	Rf
#0 Initial	2026	1.0	0.6	0.60
#1 Ignition system testing	2027	0.9	0.6	0.54
#2 Add engine redundancy capability at Critical Mission	2027	0.8	0.5	0.40
#3 Robust Startup Procedure with redundant thrust capability	2028	0.7	0.3	0.21

Figure 6.5.1-1 Mitigation Activity Schedule

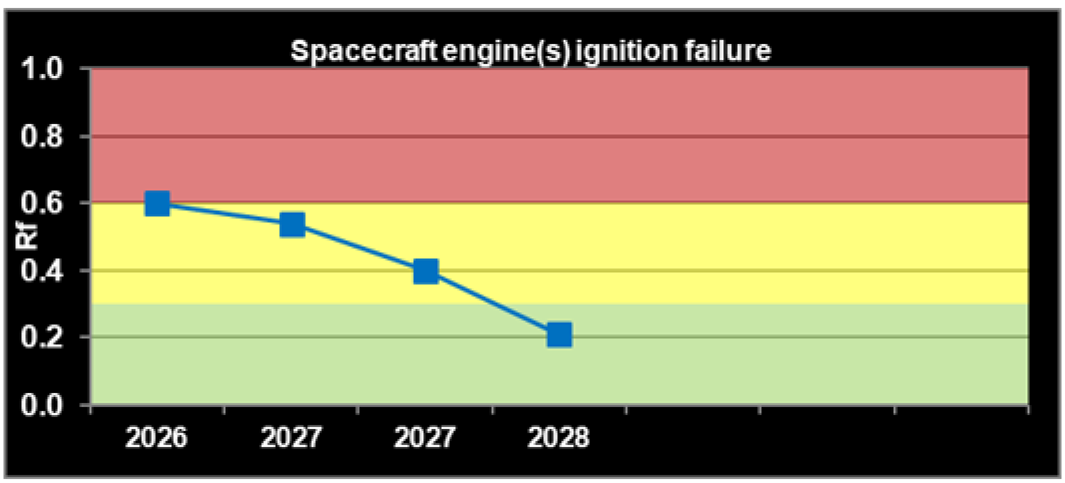


Figure 6.5.1-2 Mitigation Chart Timeline of Events

6.0 Schedule

The RFP issued the requirement of departing from the DSG towards the lunar surface on the system's first mission by the year 2029. In order to accomplish that, all stages of design, manufacturing, integration, and testing must be completed with results that prove the system will function as desired. The preliminary schedule is as follows.

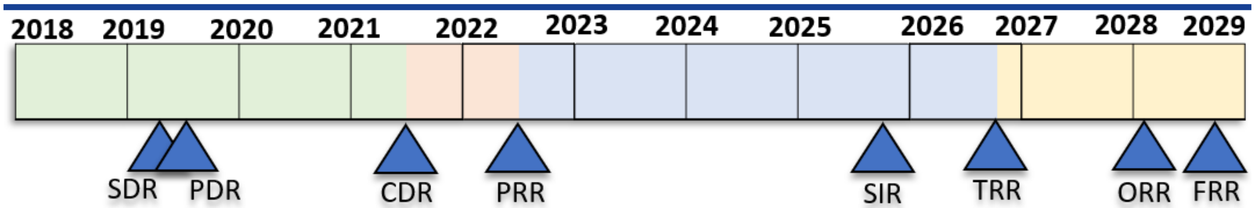


Figure 6.6.0-1: System Review Schedule Summary

The system definition review (SDR) and preliminary design review (PDR) have already concluded. The next step in development of the CAVEMAN is the critical design review (CDR) which would occur in two years on 05/12/21. This marks the end of design phase and the transition into the production and manufacturing phase. As CDR ends, a year is allocated for any comments and changes to be implemented into the design. On 5/16/22, production readiness review (PRR) occurs, which grants permission for the system to move into manufacturing, integration, and subsystem/component level testing. Towards the end of this phase, on 11/21/25, system integration review (SIR) occurs. This is where all components have been manufactured to spec and tested (if necessary) with positive outcomes, and the full system can be integrated together. Once the system is fully integrated, the test readiness review (TRR) occurs on 11/20/26. Then the vehicle undergoes full system testing to ensure it works as designed. These tests lead to operational readiness review (ORR) and flight readiness review (FRR) on 10/27/28 and 11/17/28 respectively. These final reviews ensure the system is fully operational, properly integrated into the SLS Block II Cargo, and ready for its mission. Operations-wise, the schedule of initial milestones can be seen below. This table picks up where the previous schedule of reviews leaves off.

Table 6.6.0-1: Operational Milestone Schedule

Date	Operational Milestone
11/20/28	Launch to Deep Space Gateway from Cape Canaveral, Florida
11/22/28	Arrival at Deep Space Gateway at Apolune. Begins system inspection
12/01/28	Cargo Mission I Departure
12/24/28	Cargo Mission I Return. System Inspection & Maintenance (SIM) begins.
04/12/29	SIM complete. Crew Mission I Launch
04/28/29	Crew Mission I Return.

As long as the FRR allows, the CAVEMAN will begin its journey via the SLS, arrive at the moon, and begin inspections to ensure no damage occurred to the system prior to its maiden voyage. From there, the system departs on its first cargo mission towards the moon and back. This mission returns cargo to the DSG and then undergoes system inspection and maintenance (SIM). If all goes well, the first cargo mission would serve as a baseline for human-rating of the spacecraft. Once SIM completes, the vehicle takes its first passengers to the lunar surface and back. The missions that follow would then follow the same cycle of SIM prior to missions as needed.

7.0 Cost Estimations

Cost of the system was estimated using NASA's Project Cost Estimating Capability (PCEC) and the Human Spaceflight: Mission Analysis and Design (SMAD) model. The two cost models estimated $8.7B and $9.3B respectively. Results from NASA's PCEC are broken down in the pie chart below.

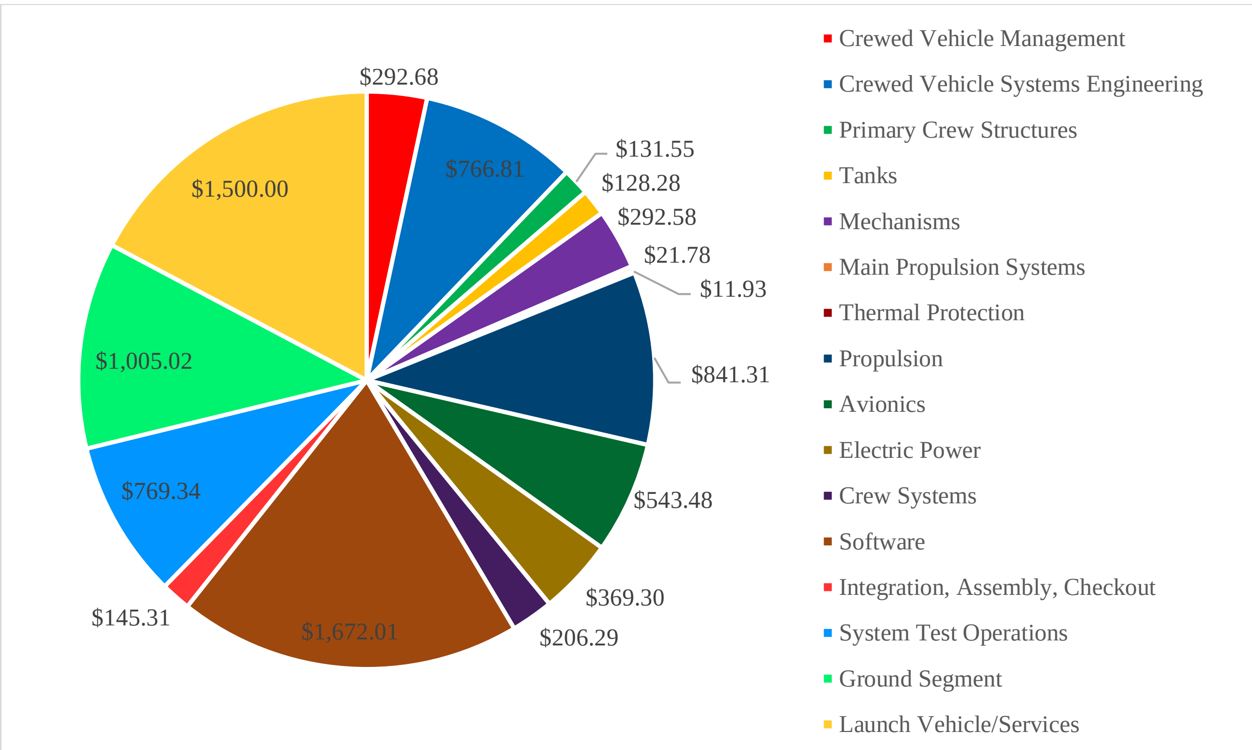


Figure 6.7.0-1: NASA PCEC Estimate for CAVEMAN Transport

The PCEC cost model can also be seen distributed in the work breakdown structure that follows. Note that this cost model includes launch cost, design development test and evaluation (DDT&E), and theoretical first unit (TFU), but not disposal or any missions after the primary cargo launch.

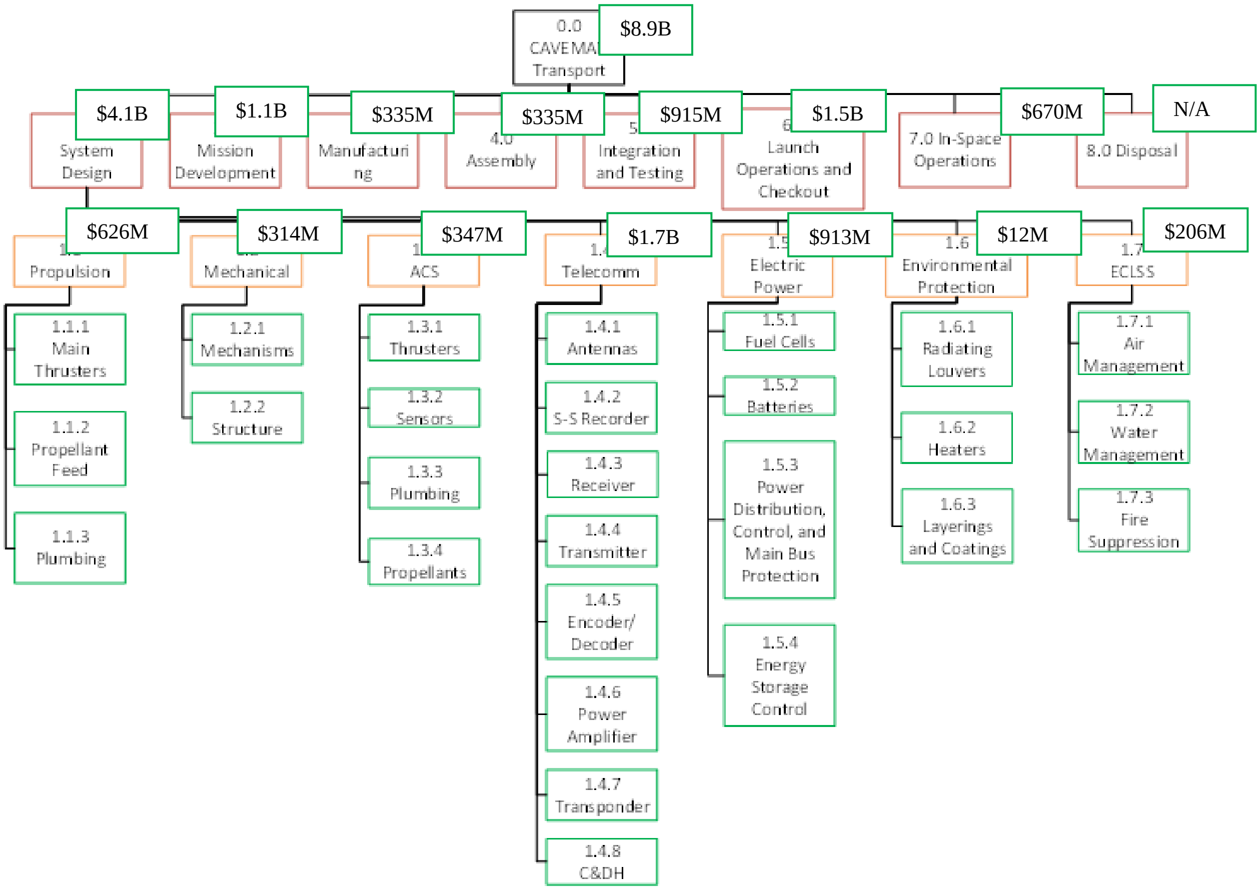


Figure 6.7.0-2: WBS with Cost Breakdown

8.0 Compliance Matrix

REQ #	REQ Statement	Addressed?
T 0.1	Deliver crew and/or cargo to anywhere on the surface of the Moon from the Deep Space Gateway	Trajectories in both modes enable us to land anywhere on the lunar surface
T 0.2	Operate in crew or cargo delivery mode Crew Mode: • 4 astronauts, 24 hour life support on surface • Cargo Mode: 15,000 kg to surface, 10,000 kg from surface	Modular Hab design allows for inhabited missions when needed. Propulsion system was designed with the payload requirements and sized accordingly. ECLSS system has healthy margin for crewed mission duration.
T 0.3	Vehicle shall make multiple trips to and from the DSG utilizing propellant refill.	Maintenance concepts call for system checkout of valves, seals, and other critical components prior to launch from DSG.
P 0.1	Cost shall not exceed $10B (FY17) Includes launch cost	Human SMAD and PCEC cost estimation show we are within budget allowable.
P 0.2	The vehicle shall make its first trip from the Deep Space Gateway to the lunar surface no later than December 31, 2028	Proposed schedule places Cargo Mission I launch from DSG on December 1, 2028 as long as SLS Block II keeps schedule.

VII. Conclusion

The first principles used when designing this vehicle attempted to mitigate risk while also giving a sporty flight system capable of being competitive with emerging technologies. One of the largest risks associated with our current baselined design is the large quantity of cryogenic propellants required to do a full 15-ton cargo sortie mission. Working under the assumption that the DSG was a fuel depot was convenient from an undergrad perspective, but the legitimacy of this assumption in the real world has yet to be proven. Our system, although highly modular, is designed to increase mission reliability by making systems completely independent and redundant. Our lunar forklift system ensures that cargo can be quickly loaded and unloaded even with a wide landing ellipse and is done robotically with a natural, “joystick from Earth”, control system. To recap, our system comes in on budget, on schedule, uses almost entirely TRL 9 level technologies, draws upon industry players to tackle subsystems for us and some would argue most importantly, we had fun designing it. Cheers!

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