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# System Architecture Design and Development for a Reusable Lunar Lander

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# System Architecture Design and Development of a Reusable Lunar Lander, *aVOLlo*

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**To support NASA's current goals for the Lunar Orbiting Platform – Gateway, a design for a reusable lunar lander was developed by a senior design team at the University of Tennessee Knoxville. Affectionately named *aVOLlo*, this lunar lander will operate between the Moon's surface and the Gateway, which will be parked in a Near Rectilinear Halo Orbit about the Lagrange Point L2. The lander will be able to carry a minimum of 15,000 kg of payload to the Moon's surface and 10,000 kg of payload back to the Gateway, in accordance with common NASA and AIAA reusable lunar lander design requirements. Furthermore, the lander will be capable of supporting a crew of up to four astronauts for a maximum of seven days on the surface of the Moon, with extended stays made possible by the future development of a lunar habitat. Throughout every stage of the study, efforts were made to optimize component choices, including their ability to be used in multiple subsystems across the lander. The technological readiness requirements were also considered to ensure that the components and overall design will be available for a launch time frame of 2030 atop the Space Launch System Block 1B configuration. Finally, in-situ resource utilization will be an integral part of the propellant resupply in order to promote sustainable lunar, and deep-space, exploration.**

## I. Nomenclature

| | | |
|---|---|---|
| $A$ | = | Cell area |
| $\Delta V$ | = | Velocity change |
| $F$ | = | Faraday's constant |
| $I$ | = | Current |
| $i$ | = | Current density |
| $I_{sp}$ | = | Specific impulse |
| $\lambda$ | = | Stoichiometry number |
| $MW$ | = | Molecular weight |
| $\dot{m}$ | = | Substance production |
| $n$ | = | Equivalent electrons per mole of reactant |
| $\#_{cells}$ | = | Number of cells in fuel cell stack |
| $\#_{days}$ | = | Number of days per mission |
| $P$ | = | Power |
| $T/W$ | = | Thrust-to-weight ratio |
| $V$ | = | Voltage |

## II. Introduction

### A. Background

Fifty years ago, this July, humans first stepped foot on the Moon. In the half-century that has followed, human presence past Low Earth Orbit (LEO) has been significantly neglected. Figure 1, below, details the history of significant space missions since the 1950's. As can be seen, several notable accomplishments have occurred in the last 50 years, but humans are not mentioned onwards of 1969 [1]. In fact, the last time humans left LEO was December 11, 1972 on Apollo 17, which also happened to be the last Moon landing [2]. In recent years, however, NASA has again set eyes on establishing a consistent human presence beyond LEO, specifically on and around the Moon. By utilizing the Moon as a stepping stone to train astronauts, test new technologies, and increase space-mission

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experience, strides can be made towards the ultimate goal of sending a human to Mars. To facilitate these goals, NASA is working with industry partners to develop components for a crewed lunar-orbiting station called the Lunar Orbiting Platform – Gateway [3].

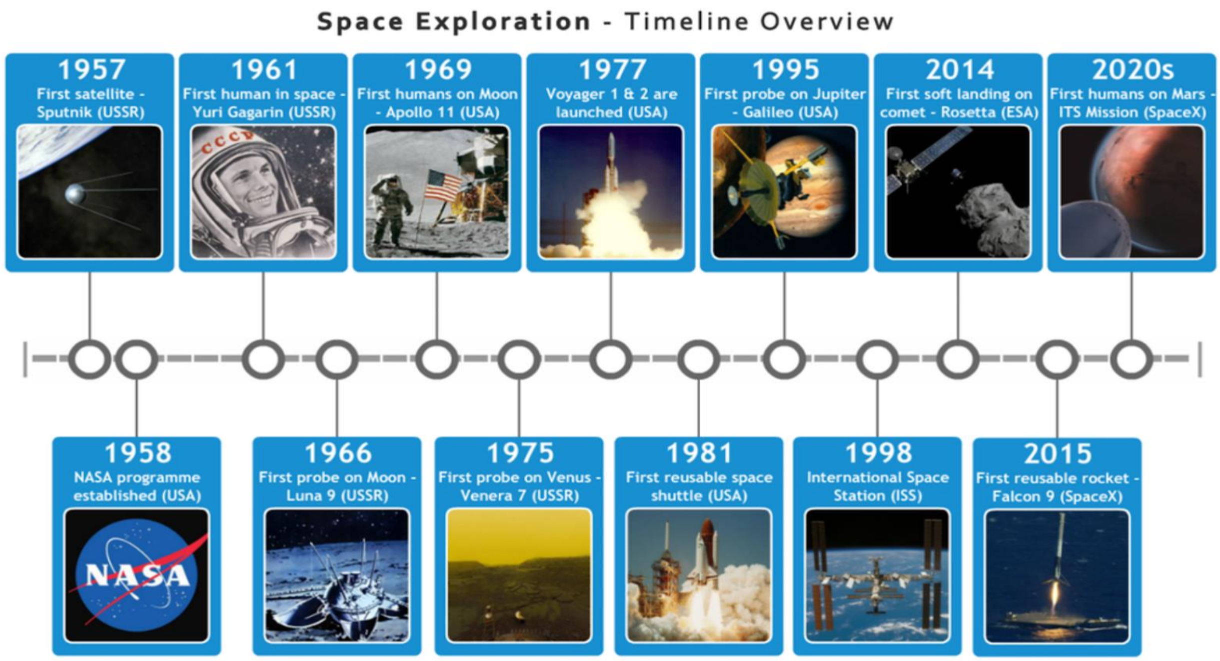


**Fig. 1 History of space exploration [1]**

### B. The Gateway

The Lunar Orbiting Platform - Gateway will serve as a docking station for Moon missions, complete with power and propulsion elements as well as habitation, logistics, and airlock capabilities. It will also provide a consistent presence in cislunar space in the coming years as NASA strives to explore the Moon and understand how to better utilize its resources. Figure 2 shows a conceptual design of the Gateway's configuration plan and the industry partners that are working with NASA to make this vision a reality.

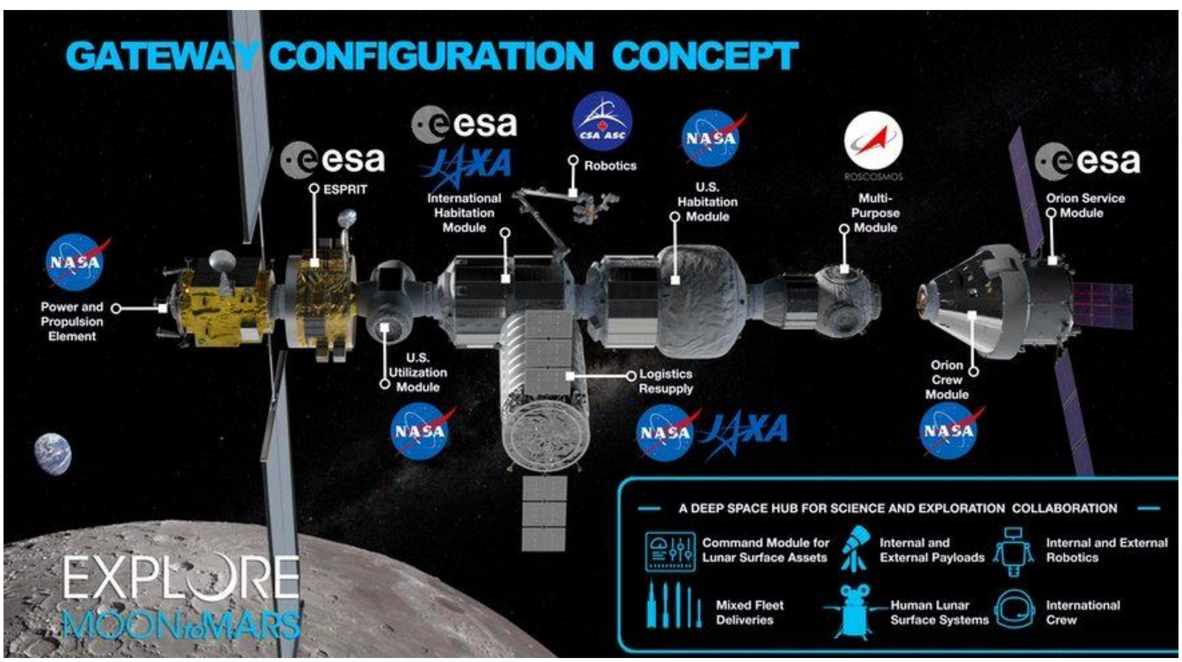


**Fig. 2 Gateway configuration conceptual design [4]**

The Orion Multi-Purpose Crew Vehicle, currently being developed by Lockheed Martin Space, will serve as the primary transit vehicle from Earth to the Gateway. Orion, and all of the Gateway components, will launch on the newest and most powerful rocket, the Space Launch System (SLS), which is currently being developed by Boeing. Hopefully, Orion will fly its first un-crewed mission by 2020 and its first crewed mission by 2023. Construction of the Gateway is planned to begin in 2022 with the launch of the power and propulsion element, and habitation capabilities are planned to be added in 2024 [5].

The Gateway will operate in a Near-Rectilinear Halo Orbit (NRHO) about the Earth-Moon Lagrange Point L2, which can be seen in Fig. 3 below. This orbiting point is ideal as it provides convenient access to the surface of the Moon, specifically the poles, where a majority of the Moon's water resources are located. Additionally, this orbit allows for constant communication with Earth as the Gateway never travels behind the Moon. The Gateway's orbit will also allow scientists to better understand the harsh environment of deep space, but from the platform of a safer, and closer to Earth, location [3].

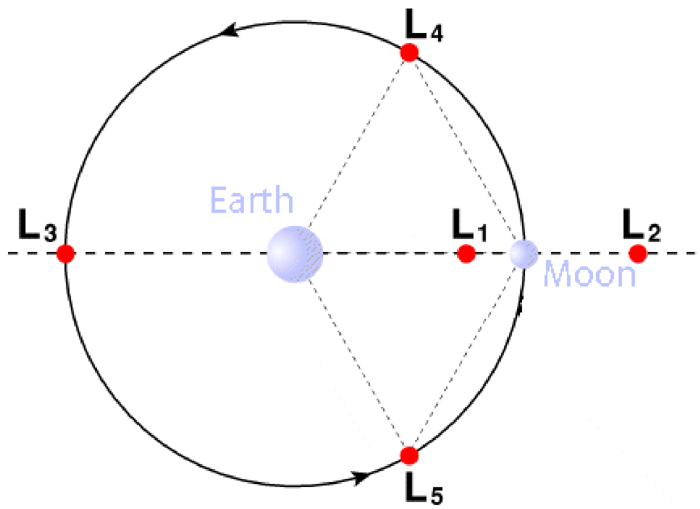


**Fig. 3 Earth-Moon system Lagrange points [6]**

### C. Reusable Lunar Lander

A component of the Gateway's design is a reusable lunar lander that will provide continuous access to the lunar surface for payload delivery and return, as well as lunar exploration [3]. Though there are well-known differences between the lunar and Martian environments, returning to the surface of the Moon, and establishing a long-term presence there, will allow scientists and engineers to better understand the demands of the environment. This will ultimately pave the way for the innovations required to send humans to Mars. As can be seen in Fig. 1, sending humans to Mars is an active goal for the 2020-2030 timeframe.

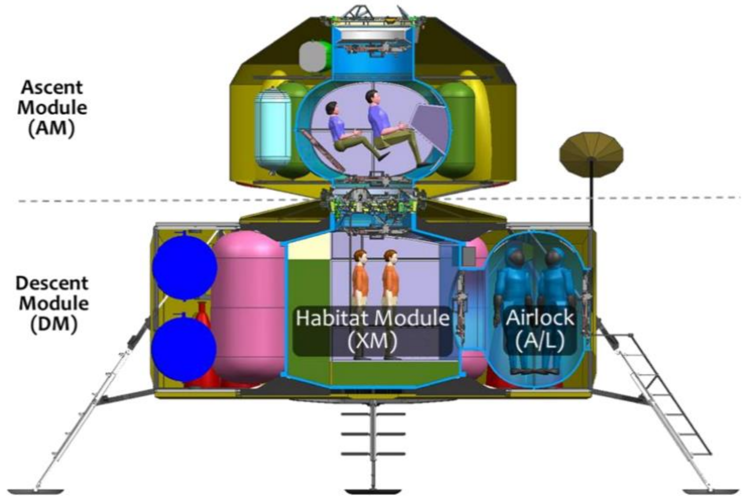


**Figure 4. Boeing's lunar lander concept [7].**

There are many current lunar lander studies as this is an engineering goal gaining much attention lately. Boeing's concept, as one example, utilizes a two-stage configuration seen in Fig. 4, above. Like the Apollo missions, Boeing's descent module will remain on the surface of the Moon while the ascent module is claimed to be fully reusable. Boeing's propulsion design uses dinitrogen tetroxide ($N_2O_4$) and monomethylhydrazine (MMH), a fuel pairing used on many space missions due to its storability and high-performance characteristics [7].

While this is only one company's design, many other proposals incorporate a lot of the same design choices. However, there are several concerns with these choices. The first is that if the descent module is left on the surface of the Moon, more descent modules must be made and attached to the reusable ascent module. This closely resembles the Apollo-era practices of leaving the descent module on the surface of the Moon, and this renders the design less reusable than it potentially could be. The second concern is the choice of propellant. MMH/$N_2O_4$ cannot be made anywhere on the Moon, thus decreasing the sustainability of the overall mission. Though MMH/$N_2O_4$ is a highly desirable propellant for many orbiting satellites near Earth, if the primary goal of these missions is to prove reusability and sustainability for an eventual trip to Mars, the lander needs to incorporate propellants that can be made from the Moon's resources. It would also be ideal for the fuel and oxidizer to be gases that are needed elsewhere in the lander as well for full system integration.

### D. Design Requirements

This paper aims to develop a lunar lander, A*VOL*LO, that will have technologies capable of landing on the Moon in the 2030 timeframe, a decade that has been repeatedly mentioned by NASA and other agencies in discussions about returning humans to the Moon [8]. Though it would be ideal to develop a lander capable of landing on both the Moon and Mars sustainably with no design differences, there are considerable implications when designing for two drastically different (and far apart) celestial bodies. If a lunar lander was developed for eventual use on Mars, a lunar landing would take significantly longer to achieve, as the technologies to take humans to Mars safely don't currently exist. Furthermore, waiting to land on the Moon until these technologies are available would arguably render a lunar landing unnecessary and a waste of time, resources, and money. Thus, this design will only consider what resources are available on the Moon, and what engineering already exists or will exist in the next decade. It will not include components that draw upon resources that are proposed to be available on the Martian surface, such as methane, nor will it use technologies that have been speculated to exist, but that will not have an acceptable technology readiness level in the required timeframe.

Ultimately, sustainability and reusability will be key for a timely and cost-efficient mission back to the Moon. Launching a coupled transit vehicle and lunar lander from Earth for each lunar mission, and leaving the one-use lander parked on the Moon afterwards, as was the practice during the Apollo era, is extremely costly. A much more cost- and time-effective method includes launching from and returning to Earth in the transit vehicle, Orion, utilizing the Gateway as a resupply platform and orbiting base, and operating the lander continuously between the Gateway and the Moon's surface. This, in addition to in-situ resource utilization (ISRU), which is the harnessing of resources at the exploration site to decrease dependence on Earth-based resupplies, will allow astronauts, engineers, and other mission stakeholders more time to study the effects of long-term space travel and prepare for a trip to Mars [9].

This paper will discuss the major components and operations associated with the development of a reusable lunar lander. The following will be discussed: orbital mechanics, in-situ resource utilization, power, propulsion, materials and structural analysis, and environmental control and life support. Primary design constraints include an ability to repeatedly descend from the Gateway in its NRHO to the surface of the Moon and back without having to return to Earth; the requirement to deliver 15,000 kg of payload to the surface and return 10,000 kg of payload back to the Gateway; and the necessity to land four astronauts on the surface and sustain them for the duration of the Gateway's orbit, which is approximately six and a half days [10].

The motivation and thus basic requirements for this project came from a combination of NASA and AIAA design competitions found here:

- *NASA:* http://rascal.nianet.org/themes
- *AIAA:* https://www.aiaa.org/docs/default-source/uploadedfiles/education-and-careers/university-students/design-competitions/undergraduate-team-space-systems-2018-2019.pdf?sfvrsn=9b9f6ea6_0

A summary of NASA's criteria is shown here:

- *Crew is delivered from Earth to the Gateway via NASA's Space Launch System (SLS) and Orion*
- *Crew returns to Earth from Gateway via Orion*
- *A reusable ascent/descent cabin/vehicle is based at the Gateway, where it is resupplied and refueled between lunar missions*

- *The lander must accommodate two mission modes (near polar location at a minimum):*
    - *6 days on the surface with 2 crew and 500 kg of cargo – no pre-deployed support infrastructure*
    - *2 days on the surface with 4 crew and 100 kg of cargo – longer stays enabled by pre-deployed infrastructure (rovers, habitats, etc.)*
    - *Both above mission modes must also accommodate the crew during their transit to and from the lunar surface*
- *Initial architecture and program model that is not "dead-ended" and facilitates evolution from initial capability to a model that leverages commercial services in order to reduce costs for sustained crew access to the lunar surface.*
- *Open for trade:*
    - *Distribution, number, location, and staging of propulsive elements (single stage, braking stages, descent modules, ascent modules, etc.)*
    - *Types of propellant and propulsion systems*
    - *Propellant resupply strategy (anticipated launch vehicles, depots, etc.)*
    - *System procurement mechanisms*
    - *Evolution path of NASA and commercial capabilities*
- *Considerations:*
    - *Impact of elements on the Gateway (controllability, power, thermal, etc.)*
    - *Number of SLS launches (ideal would be one per human lunar mission once the reusable ascent/descent vehicle is delivered to the Gateway with the remaining launches provided by commercial or international partners)*
    - *Technology readiness and cost to support a crewed lunar mission from the Gateway in 2028*

It should be noted that originally the lander was designed based on the 2-day mission with 4 crew. Upon analysis of the Gateway's orbit and the necessary trajectory to and from the Moon, it was decided that a 2-day mission was not possible. In fact, no less than a 6.5-day mission is possible, as that is the time it takes the Gateway to complete one orbit. Any other time before that is simply not feasible.

A summary of AIAA's criteria is shown here:

***Design a reusable Lunar surface access vehicle***

- *The vehicle should enable delivery of payload and/or crew to anywhere on the surface of the Moon from the Deep Space Gateway and back*
- *The vehicle should be able to operate in either a crew delivery mode or cargo delivery mode. Design of the vehicle should allow for capability of switching between the two modes*
- *In crew mode, the vehicle should support a crew of four (4) astronauts for the duration of the transit from the Deep Space Gateway to the lunar surface and should support the crew without resupply from surface assets for a minimum of 24 hours on the surface*
- *In cargo mode, the vehicle should have a payload capacity of at least fifteen metric tons (15,000 kg) to the surface of the Moon and can return at least ten metric tons (10,000 kg) of payload from the surface back to the Deep Space Gateway*

    *- Payload to the surface could include components for surface habitat, rovers, or crew supplies*

    *- Payload back to the Deep Space Gateway could include materials or resources that was mined from the lunar surface*
- *The vehicle should be able to make multiple trips to the lunar surface and back to the Deep Space Gateway utilizing propellant resupply from the Deep Space Gateway*

***Design and define the mission operations, including orbit transfer, station keeping, and other maneuvers necessary to deliver payload to/from the lunar surface***

- *Assume the Deep Space Gateway is in a stable Near Rectilinear Halo Orbit*
- *Determine the propulsive cost and time of flight of a range of landing sites from equatorial to polar*

- *Define the design requirements for trajectory and orbital maneuvers for both the crew and cargo mission modes and determine and describe the transit strategies for the both and discuss how each mode drives the design of the vehicle*
- *Determine the station keeping, orbital phasing and rendezvous, and proximity operation for the vehicle, assuming the Deep Space Gateway is the passive vehicle in all docking sequence*

Partially due to the time and manpower limitations of our group, some of the AIAA criteria were removed. The lander will not be capable of landing anywhere on the Moon, as it was deemed infeasible during the calculation of the trajectory given that the Gateway's orbit provides direct access to the South Pole. The lander will also not switch between crew and cargo mode - it will be able to deliver both simultaneously.

## III. In-Situ Resource Utilization

The ability to harness resources at the landing site, ultimately decreasing dependence on additional resupply missions from Earth, will become extremely important on Mars for producing water, oxygen, propellants, and eventually food. Therefore, testing and verifying performance capabilities of reliable ISRU technology on the Moon is crucial. Despite the launch cost from Earth drastically diminishing for these Moon missions due to the reusability of the lander in conjunction with the Gateway and Orion, the launch cost will still be steep if all the propellant, water, and oxygen needed for each mission is always launched from Earth. Propellants comprise a significant amount of the weight, and therefore launch cost, on all spacecrafts, often accounting for 70% of the total mass, as on the Apollo spacecraft [11]. In addition, consumables such as water, oxygen, and hydrogen for any power supply and crew needs will also need to be resupplied for each mission, further adding to the cost if resupply is necessary using Earth's resources. Therefore, ISRU will serve as the foundation of this lander design in order to promote sustainability and independence from Earth-based resource resupply. A comparison of the propulsion, life support, and power subsystems with and without ISRU integration is detailed in Table 1 to emphasize the importance of this foundational design consideration [12].

**Table 1. Comparison of subsystems with and without ISRU integration [12]**

| | Without ISRU | With ISRU |
|---|---|---|
| **Propulsion** | Propellant selection based on development cost and performance | Propellant selection based on ISRU products from available resources |
| | Propulsion cycle (pressure vs pump) based on development cost and performance | Propulsion cycle based on influence of ISRU on Delta-V and reusability |
| | Non-reusable or limited reusability with Earth supplied propellants and depots | Reusability with single stage landers possible |
| **Life Support** | Air and Water recycling technologies and systems based on maximizing closure of oxygen and water loops | ISRU products can reduce the level of closure required, thereby reducing development cost and system complexity |
| | Trash/waste processing aimed at maximizing water extraction and minimizing oxygen usage | Trash/waste transferred to ISRU to maximize fuel production and minimize residuals. Trash processing hardware can be minimized to some level of drying |
| **Power** | Self-contained units. Solar array and batteries | Distributed power generation and storage, esp. fuel cell reactant storage and regeneration |
| | Fuel cell reactant based on regeneration technique alone | Fuel cell reactant based on in-situ resources available |
| | Increase in power generation is a function of delivery from Earth | In-situ growth of power thru fuel cell consumable, in-situ thermal, and in-situ manufacturing |

There are two studied approaches for obtaining the necessary resources for this mission via ISRU: mining iron-oxide rich regolith, or mining water from the ice found at the poles of the Moon. Thanks to samples brought back from the Apollo missions, it is known that all lunar regolith contains approximately 45 wt% oxygen in the form of oxides [13]. It is possible to reduce these oxides to extract the oxygen through the hydrogen reduction process. The most promising oxide is iron oxide (FeO), and as seen in Fig. 5, it is found in abundance near the Moon's equator [14].

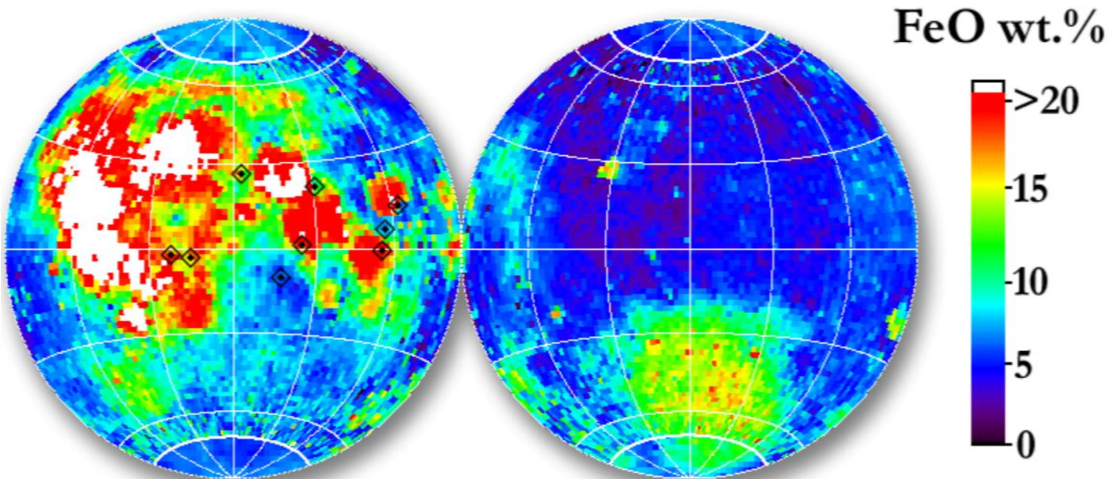

**Figure 5. Iron-oxide concentration on the Moon (black markers indicate past landings) [14]**

The chemical equation for the hydrogen reduction process is:

$$FeO + H_2 \rightarrow Fe + H_2O$$

Therefore, by reacting hydrogen with the iron-oxide, the oxygen can be separated from the iron to form pure iron and water vapor. The water vapor can then be separated from any excess hydrogen and electrolyzed into oxygen and hydrogen. Ultimately, the oxygen and hydrogen can be condensed to liquid, cryogenically stored, and then used in various subsystems across the lander [13]. There are four different lunar samples that have been studied for this hydrogen reduction process ISRU technique: ilmenite, basalt, unprocessed soil, and volcanic glass. A summary of their respective oxygen yields can be seen in Table 2. Volcanic glass yields the highest mass of oxygen per mass of sample and is argued to be the optimum lunar rock material from which to extract oxygen [13].

**Table 2. Oxygen yields for each class of lunar sample studied**

| Rock Type | Oxygen Yields |
|---|---|
| **Ilmenite** | 2-2.5 wt% |
| **Basalt** | 2.93 wt% |
| **Unprocessed Soil** | 0.5 wt% - 4.5 wt% depending on the soil composition |
| **Volcanic glass** | 5 wt% |

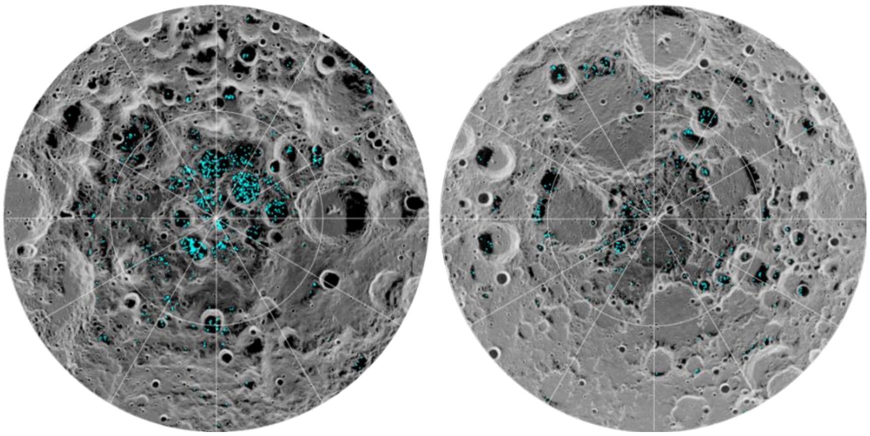

**Figure 6. Distribution of water ice found on the lunar South Pole (left) and North Pole (right) [15]**

The other mining technique is mining the water ice from the poles of the Moon. It is known, thanks to NASA's Mineralogy Mapper that flew aboard India's Chandrayaan-1 lunar probe, that there are large quantities of ice near the lunar poles, specifically in the shaded craters [15]. The total ice water between 87.5°S to 90°S (South Pole latitudes) is estimated to be 135 – 240 million MT compared to an estimated 62 million MT from 87.5°N to 90°N (North Pole latitudes) [16]. Figure 6, above, shows the distribution of water ice on the lunar South Pole (left) and North Pole (right).

Though mining iron-oxide rich volcanic glass regolith has been more heavily studied and proposed than mining water ice to yield oxygen and hydrogen, the Gateway provides direct access to the lunar South Pole. Because the South Pole contains an abundant amount of ice water to electrolyze into oxygen and hydrogen, this lander will depend on this ISRU technique. If the Gateway provided access to the equatorial region, the regolith mining would have been a better choice. With this in mind, it was then necessary to determine how A*VOL*LO would acquire the necessary resources. During the preliminary design phase, the lander was originally planned to have its own excavation, electrolysis, and storage capabilities. However, it was decided that the power and mass requirements of such a process far exceeded the capabilities of a single lander. Fortunately though, several companies have preliminary plans for establishing ISRU manufacturing plants on the lunar poles, and propellant depots in Low Lunar Orbit (LLO) (Bigelow, Shackleton Energy Company, and United Launch Alliance to name a few) [17,18]. A*VOL*LO was designed based on the assumption that at least one of these companies will have ISRU manufacturing operations underway in the desired timeframe, based on their current claims [18]. By utilizing South Pole manufacturing sites and LLO depots, A*VOL*LO's propellant, water, hydrogen, and oxygen needs will be resupplied in the vicinity of the Moon without the dependency on Earth's resources, including time and money.

One of the most important benefits of ISRU for the A*VOL*LO missions revolves around the choice of liquid hydrogen ($LH_2$) as the fuel, which will be discussed more in-depth in the propulsion section below. $LH_2$ experiences a high boil-off rate, rendering its use as a propellant almost impossible without the ability to transfer it to the lander very soon before ascent or descent. Though work is being done to mitigate the boil-off effects through the use of newer materials and thicker insulations, those technologies are not currently at the desired technology readiness level [19]. Relying on the manufacturing proposals of other companies will enable A*VOL*LO to use $LH_2$ as a fuel with less of a concern for the fuel boiling off before being used. A*VOL*LO will obtain the necessary $LH_2$ and liquid oxygen (LOX) while docked to the Gateway from a commercial propellant depot, assuming that the depot will have compatibility with the Gateway. After the surface mission is completed, A*VOL*LO will resupply $LH_2$/LOX from the commercial manufacturing site and return to the Gateway. Further analysis regarding the propellant trade study is detailed in the following section.

## IV. Propulsion

### A. Propellant Selection

The primary concern for the propulsion system was choosing the type of propellants that would be used throughout the course of the mission. Liquid propellants are better for restart capabilities but have challenges when it comes to storing them. Solid propellants can only be ignited once, and once ignited, there is no way to stop the initial burn. This renders solid propellants unsuitable for outer space excursions such as this where there are prolonged coasting periods when the engine is not being used at all. Hybrid propellants have the start and stop capabilities of liquid propellants but cannot be refueled due to the depletion of the solid propellant portion of the rocket [20]. Ion thrusters are an unreasonable option due to the fact that they have high specific impulses but very low thrust applications, which is a critical factor in this mission [20]. Considering all of the objectives and goals required of the lander, liquid propellants were the clear choice.

After selecting a viable propellant type, further specifications had to be made. The most commonly used propellant combinations for similar missions, and the ones researched in this analysis, include liquid oxygen and liquid hydrogen (LOX/$LH_2$), liquid oxygen and liquid methane (LOX/$LCH_4$), a liquid fluorine and liquid oxygen mixture (FLOX), and nitrogen tetroxide and monomethyl hydrazine (NTO/MMH). There were many different parameters considered in the analysis process, including toxicity, technology readiness, storability, and boil-off rates. The outlined parameters are discussed below in a more detailed propellant trade study analysis.

Liquid oxygen and liquid methane (LOX/$LCH_4$) is a viable combination for a lander on Mars due to the immense amount of methane that resides on and below the Martian surface. It performs better than most storable propellants and does not require a high-volume storage tank like liquid hydrogen. It has a clean burn and is non-toxic, which is important for crew member safety. It also has a lower mass than common hypergolic propellants. However, of the fuels listed, it is the least advanced in terms of technology readiness. Because Mars is the ultimate goal for future crewed space exploration, there are still some developments that need to occur before technology that can operate on LOX/$LCH_4$ can be successfully implemented. There is no history or background on using this propellant in previous flights and there is limited ground test history as well. Further testing and prolonged use analysis would be necessary before this combination could be used on a crewed flight mission [21].

A liquid fluorine and liquid oxygen mixture (FLOX) is a reasonable choice if considering only engine performance. The addition of fluorine results in a 5% increase in rotor motor efficiency than LOX alone, and produces a 15% better specific impulse in comparison to LOX. However, this combination is extremely toxic and can harm the crew members if leaked into the cabin. This super-oxidizer reacts violently with nearly everything, except nitrogen, lighter noble gases, and substances that have already been fluorinated. Taking this into consideration, this combination is not a valid option due to the nature of a long duration mission such as this one [21, 22].

Nitrogen tetroxide and monomethylhydrazine (NTO/MMH) have been used consistently throughout spaceflight history and, therefore, have a strong research and testing background for comparison. This hypergolic propellant's technology readiness is the highest in comparison to the other considered propellants, but has problematic characteristics that outweigh the benefits associated with its lengthy history. It has easy start and restart capabilities which is very important for the mission objectives. However, it is highly toxic and can harm the crew members if contact occurs. The primary issue with this propellant combination is that it cannot be used with a common bulkhead as the risk of cracking is substantial and could cause an explosion large enough to destroy the entire lander [21, 22].

A liquid oxygen and liquid hydrogen (LOX/$LH_2$) combination is a reasonable choice for a lunar lander due to the fact that is contains the highest specific impulse, $I_{sp}$, out of the considered propellants. It has a 30-40% higher $I_{sp}$ than most other fuels which is great for storage because rockets that have high specific impulses produce more force without using as much fuel as rockets that have lower specific impulses [23]. However, $LH_2$ is a cryogenic liquid, which means it requires a greater effort to keep it from boiling off. This is due to the fact that cryogenic fluids need to be stored at extremely low temperatures, meaning that boil-off is highly likely as the required storage temperatures are well below that of the ambient environment. Along with the boil-off challenge, liquid hydrogen is difficult to store due to its very low density of 70.8 kg/$m^3$ (which is small in comparison to water at 997 kg/$m^3$). With such a low density, the propellant tanks would have to be enlarged in order to accommodate a large volume of the liquid propellants [21, 22]. Despite the difficulties associated with these propellants, LOX and $LH_2$ were chosen based on evidence confirming the presence of lunar water under the surface of the Moon. This water, if extracted, could be converted into liquid hydrogen and liquid oxygen, allowing for consistent refueling capabilities throughout the duration of the mission, as discussed in the ISRU development section [24]. Figure 7, below, shows a miniature radio frequency (Mini-RF) image of the lunar South Pole, further confirming the presence of ice-rich craters which could be harnessed for this mission. Scientists have analyzed the Mini-RF images for areas of varying reflectivity and discovered that there is ice on the floor of the Cabeus crater and Shackleton crater (not shown on image). The formations of ice appear as radar-transparent material, or the very white areas viewed on the image.

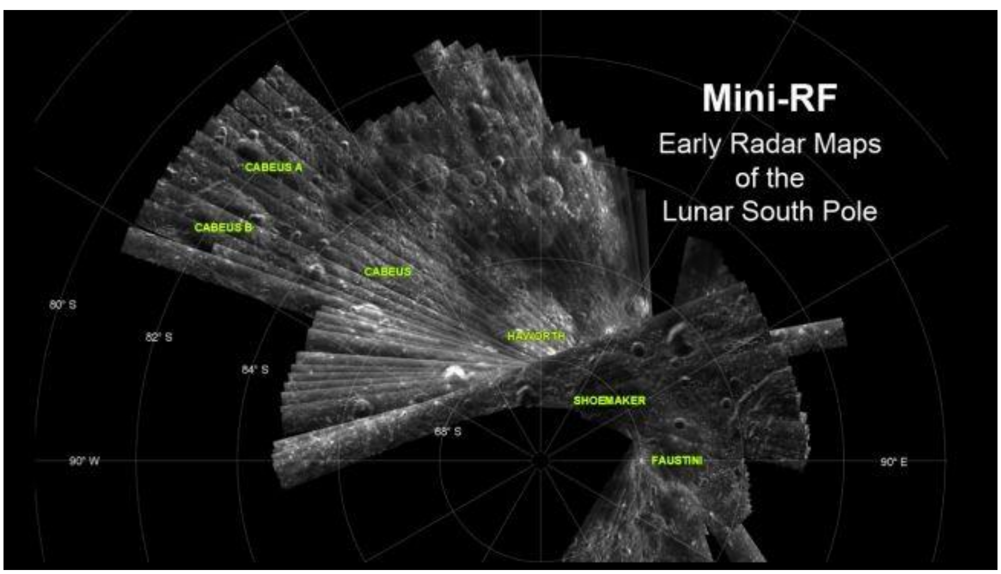


**Figure 7. A Mini-RF image of the Lunar South Pole [24]**

## B. Propellant Feed System

With the propellant chosen, the next step was to develop the propellant feeding method. Two main methods were analyzed: a pressure-fed system and a pump-fed system. A pressure feeding system uses the pressure induced by helium or nitrogen to push the propellant down to the combustion chamber, where it is ignited and exhausted. These types of systems are very reliable for small changes in velocity (ΔV's) due to the simplicity of the design; however, they require thick walls to accommodate for the extremely high pressures which would, in turn, cause the lander to be more massive [25, 26]. Additionally, the lander would require refueling of both the propellant *and* the pressurized gases, which adds to the overall complexity of the system. A pump-fed system, on the other hand, operates at lower pressures, allows for thin-walled designs, and has a lighter mass. They are preferred when considering high-thrust and long-burn maneuvers [26]. Figure 8 shows a simplified version of a pressure-fed system and a type of pump-fed system called an expander cycle. The pump-fed system was chosen for the lander as it satisfies the design requirements and is less massive and costly.

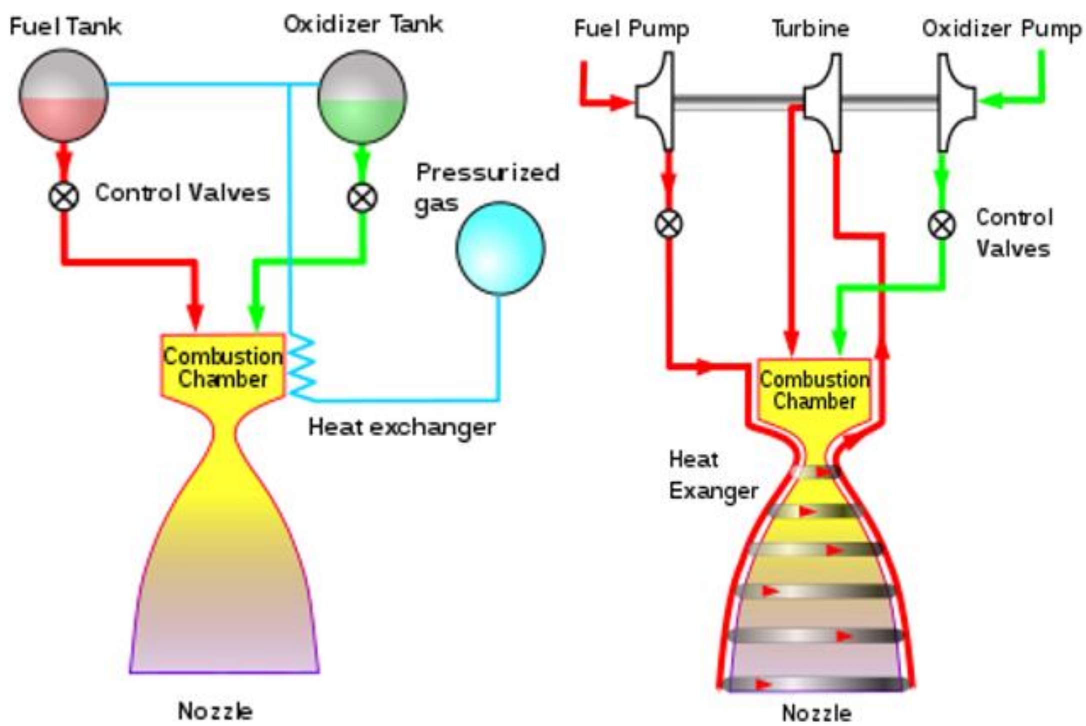


**Figure 8. Pressure-fed system (left) and an expander cycle pump-fed system (right) [27, 28]**

## C. Engine Selection

After selecting the propellants and feeding system, an engine had to be chosen that operated under such conditions, offered long-term durability, and provided the necessary thrust determined by the overall weight of the lander. Based on the engine selections of similar past missions, as well as related lunar lander proposals, engine considerations were narrowed to two options: Pratt & Whitney's RL10 CECE (Common Extensible Cryogenic Engine) and Northrop Grumman's TR202. Both of these engines are expander cycle engines that operate on liquid oxygen and liquid hydrogen, they are both used for ascent and descent stages for current or proposed landers, and they both have wide throttling ranges. However, the RL10 CECE has much more available data and a broader range of testing than the TR202. Table 3 details an engine trade study performed to compare the primary characteristics of both engines.

**Table 3. RL10 CECE and TR202 engine trade study [29]**

| | RL10 CECE | TR 202 |
|---|---|---|
| **Designer/ Manufacturer** | Pratt & Whitney | Northrop Grumman |
| **Current Status** | In production | Cancelled program |
| **Propellant** | $LOX/LH_2$ | $LOX/LH_2$ |
| **Thrust** | 67 kN | 38.8 kN |
| $I_{sp}$ | 445 seconds | 436-453 seconds |
| **Thrust-to-weight (T/W) ratio** | 0.39 – 0.26 | 0.31 |
| **Length** | 1.53m | 2.24m |
| **Used on** | Centaur, S-IV, DCSS | --- |
| **Weight** | ~160 kg | ~127 kg |

As can be seen in the table, the TR202, seen in Fig. 9, is a Northrop Grumman rocket engine that has a maximum thrust of 38.8 kN and a mass of 127 kg. It was designed for a lunar lander as part of the Constellation Project before its cancellation in 2009. For this reason, much less data exists for this engine in comparison to the RL10 CECE. The throttling range for this engine is 18.8:1, which is excellent as it allows for increased maneuverability. Additionally, the TR202 showed high consistency in testing when throttling down to 10% due to the variable area pintle injector which allows for more control of the propellant injection. This injector was also noted to prevent chugging issues throughout the entire range of its throttling capabilities [29]. Chugging is best described as an unsteady flow of propellant to the combustion chamber. It causes oscillations in the mass flow rate which causes pulsations in the chamber pressure, making the chamber pressure inconsistent. There have been a few tests that aimed to validate the thermal integrity of the hardware, but as it has not been used in missions before, the long-term integrity cannot be verified [30].

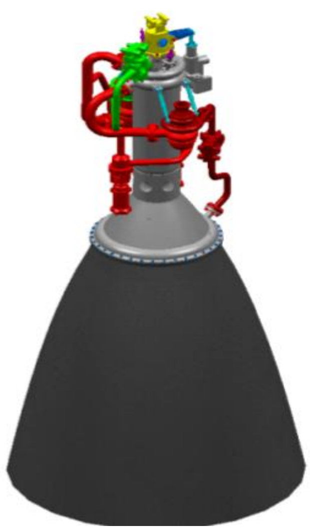

**Figure 9. TR202 engine mock-up [30]**

The RL10 CECE, in comparison, is a Pratt & Whitney rocket engine that has a thrust of 67 kN and a mass of 160 kg. The throttling range for this engine is 10:1, but it lacks consistency in the 16-20% range. Previous testing has shown that lower power chugging oscillations have occurred when throttling in this range due to vapor formations in the injector oxygen manifold [31]. However, later tests and developments were able to fix these occurrences. The gimbal capability of this engine is 4°. This is significant as it helps with thrust vectoring in the case of an engine-out, in which case the remaining engines have to take over the added weight. Additionally, it is rated as having 50 in-space starts and a service life of 10,000 seconds [32]. Trajectory analysis iterations for this mission, which will be further detailed in the Orbital Mechanics section, determined that the engine will burn for a total of 49 seconds during descent and approximately 25 seconds during ascent before rendezvous begins with the Gateway. Assuming that the service life of the RL10 CECE is truly limited to 10,000 seconds, this mission could be repeated 135 times, which would be more than adequate for the overall goals of this lunar lander mission. Figure 10 shows an image of the RL10 CECE engine.

**Figure 10. RL10 CECE rocket engine throttling from 10:1 [33]**

A significant factor in the engine selection process was determining the overall mass of the lander and the required thrust-to-weight (T/W) ratio necessary to support the required mass. The total system mass was calculated, gravity losses were incorporated, and an appropriate T/W ratio was chosen. Apollo's Lunar Module (LM) T/W ratios during the initial takeoff for both the descent and ascent trajectories were 1.63 and 1.78, respectively [34]. These values were chosen as a baseline for this mission. Table 4 tabulates these T/W values as well as the various mass values necessary for the thrust calculations. The initial mass values include estimates for all subsystems and the required payloads. The propellant masses were determined using ascent and descent $\Delta V$ requirements, which will be discussed further in the Orbital Mechanics section.

**Table 4. Mass and T/W approximations**

| | **Descent** | **Ascent** |
|---|---|---|
| **Required Payload (kg)** | 15,000 | 10,000 |
| **Departure Mass (kg)** | 56,329 | 46,886 |
| **Arrival Mass (kg)** | 29,308 | 24,308 |
| **Propellant Mass (kg)** | 27,019 | 22,578 |
| **Takeoff T/W** | 1.63 | 1.78 |
| **Max Thrust (kN)** | 148.7 | 135.2 |

Using these mass estimates and T/W ratios, the maximum thrust required for A*VOL*LO was determined to be approximately 148.7 kN. Based on an analysis of similar projects and landers, a four-engine configuration was chosen to provide the necessary thrust as well as engine-out survival capabilities. As such, the RL10 was chosen for the A*VOL*LO lunar lander. With the lower thrust of the TR202, a fifth engine would be required were an engine-out to occur, consequently increasing the weight of the lander. Overall, the RL10 CECE has all the characteristics necessary to meet mission requirements including high thrust capabilities, a large gimbaling range, excellent throttle control, long-term durability, and proven reusability [27].

## V. Orbital Mechanics

The Gateway's orbit is one of the family of L2 halo orbits in the Earth-Moon system as seen in Fig. 11 [10]. These orbits utilize the mathematical model of the Lagrange point to achieve unique orbit characteristics. At the L2 point, the gravitational pull of both the Earth and the Moon are equal to each other which allows an object at this point to orbit the Earth with the same orbital period as the Moon. This location, however, is unstable and any perturbations with the orbit will cause the object to fall out of the L2 orbit [35]. The family of orbits in Fig. 11 build off the base L2 point to create the family of orbits that all have the similar characteristics [10].

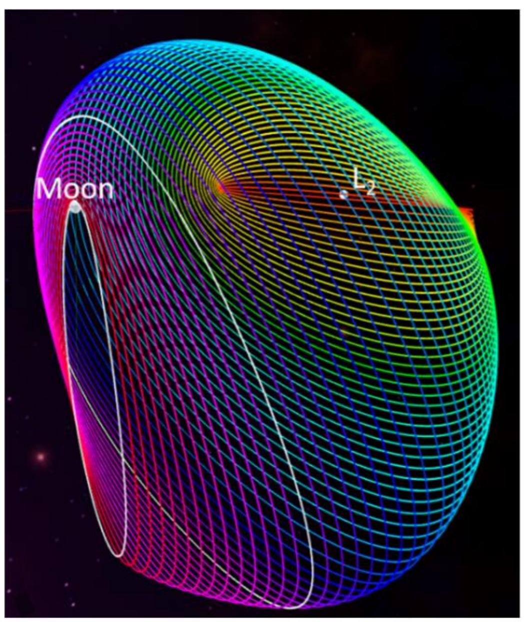


**Figure 11. Family of L2 Halo Orbits with NRHO on the far left [10]**

The NRHO chosen by NASA engineers as the parking orbit for the Gateway can be seen in Fig. 12 from a Moon-centered/Moon-fixed reference frame. The NRHO was chosen for many reasons including the ease of access to the lunar surface due to its periapsis proximity to the South Pole and constant communication with Earth since the orbit never travels behind the far side of the Moon [3]. Through an analysis of the most stable and cost-effective orbit, the Gateway's NRHO will have a periapsis and apoapsis of 3,233 km and 71,733 km, respectively [10]. This analysis compared the required $\Delta V$ to maintain a stable orbit with the required $\Delta V$ for descent and ascent to and from the Moon. More emphasis was placed on the $\Delta V$ of the ascent and descent because these values changed more drastically with different orbits [3]. One prominent downside of the chosen orbit is the instability caused by the high apoapsis that ventures outside of the Moon's sphere of influence. Without minor adjustments on a weekly basis, the Gateway would eventually fall out of orbit due to the stronger pull of Earth's gravity near the apoapsis. These minor velocity adjustments would amount to a mean annual $\Delta V$ of 7.79 m/s [10].

**Figure 12. NRHO depicted in a Moon-centered, Moon-fixed reference frame with Earth in the foreground**

The lunar lander will utilize a two-burn maneuver for both its descent and ascent trajectories, which can be seen in Fig. 13. A two-burn maneuver was chosen in order to eliminate the abundance of variables associated with multi-burn maneuvers, such as multiple engine starts and stops. To land on the Moon, the lander will execute a short duration burn near the Gateway to take the lander out of the parking orbit. This burn puts the lander on a path towards the desired landing location with the burn solely in the direction opposite of the current velocity to minimize fuel expenditure while efficiently slowing it down. After the initial burn, the lander will coast for an extended period and then fire its engines a second time for the landing burn.

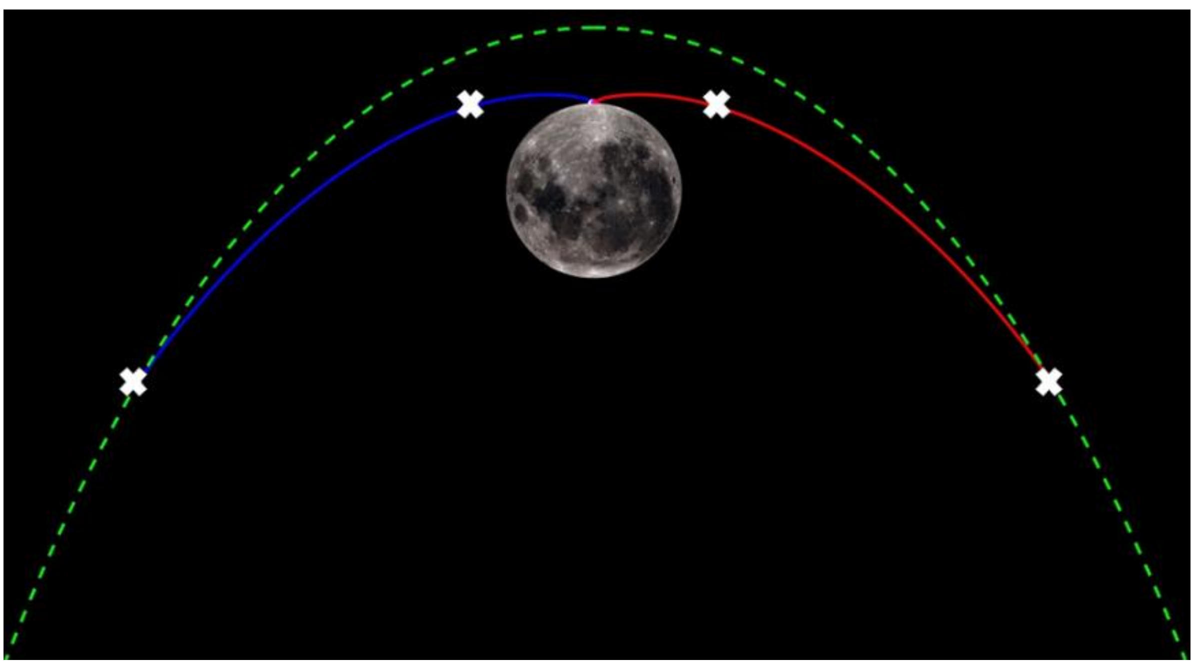

**Figure 13. Trajectory visualization from the NRHO (green) to the Moon for both descent (red) and ascent (blue), with engine firings (white) noted**

After successive iterations to find the necessary altitude for the start of the landing burn, the landing burn was determined to fire at an altitude of 1,845 km. This will ensure the lander reaches a velocity of 1 m/s at an altitude of 50 meters [36]. This is done to account for variations in the topography of the lunar surface, so the lander will not crash on the moon. The lander will then throttle down its thrust to allow for a constant velocity burn to the surface of the moon. Figure 14 shows these velocity changes in relation to increasing/decreasing altitude for both the ascent and descent of the lander, respectively.

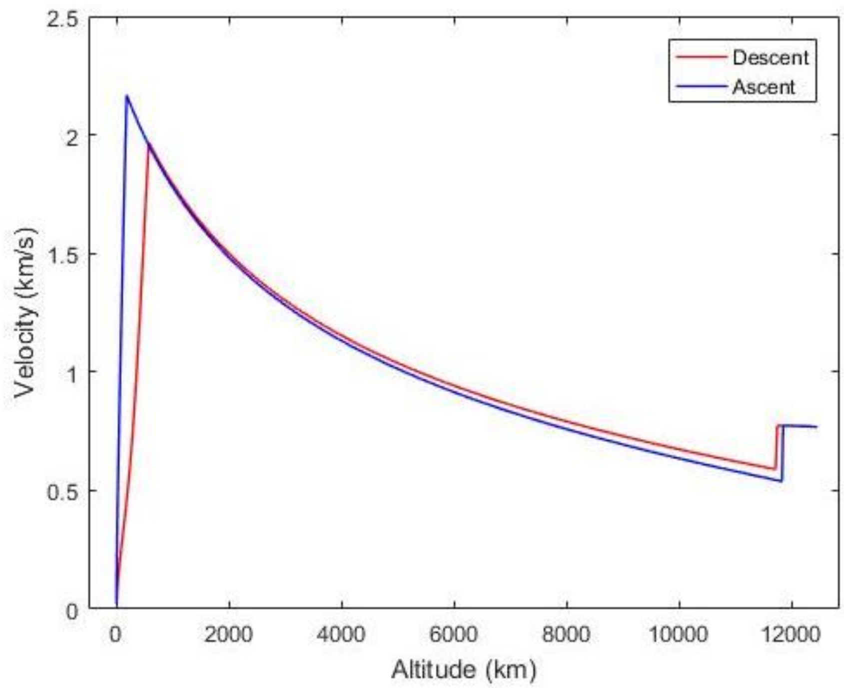


**Figure 14. Velocity versus altitude for the descent and ascent**

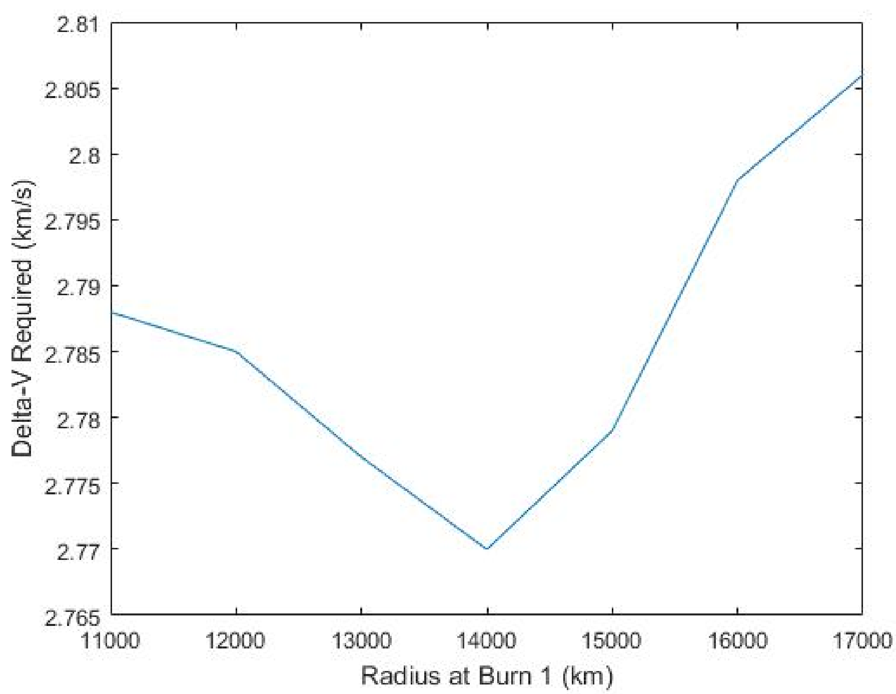


**Figure 15. Required Delta-V vs Starting Radius for Burn 1**

To optimize the descent and ascent trajectories there were four different variables that affected the required trajectory ΔV's. These four variables are the starting altitude of both burns and burn duration of both burns. The starting altitude and burn duration of the second burn was constrained as the lander needed to reach the surface with no velocity. This meant that only the starting altitude and burn duration of the first burn could affect the required ΔV. With a landing spot on the South Pole, the simulation was designed to automatically target this area. This was done by testing a range of different starting altitudes and having the simulation determine the required burn duration for both burns and the starting altitude for the second burn. With the initial input of the starting burn altitude, the simulation would iteratively increase the duration of the burn until the lander successfully landed at the South Pole. The results can be seen in Fig. 15, above, for the total ΔV for each required iteration and these values were compared to determine the optimal starting radius for the first burn. After the successive iterations, an optimal starting radius of 14,000 km and a burn duration of 25 seconds was found to optimally reduce the ΔV for the descent to 2.77 km/s. Similar processes were conducted for the ascent burns.

After successful touchdown, the lander will sit idle for approximately six and a half days while the Gateway completes one orbit around the Moon and the astronauts perform the mission requirements. Once the Gateway comes back in the vicinity of the South Pole, the lander will launch and complete a two-burn maneuver to rendezvous back with the Gateway. Due to a lower payload mass for the ascent, the lander will be able to rendezvous with the Gateway more efficiently by reducing gravity drag, requiring a ΔV of only 2.68 km/s for the ascent. The mission timeline can be seen in Fig. 16.

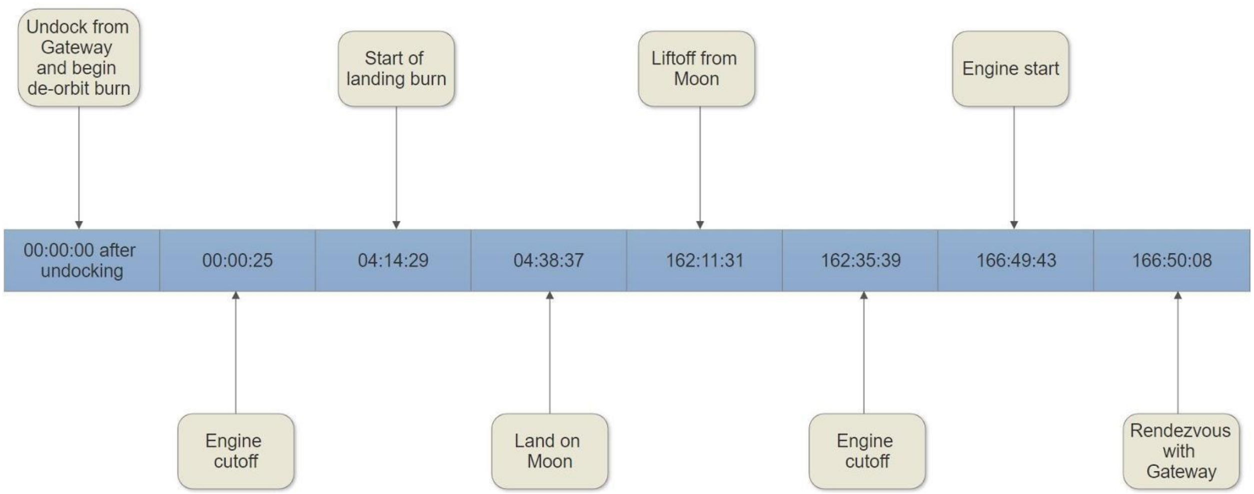


**Figure 16. Timeline of descent and ascent beginning with undocking from Gateway**

## VI. Power

Conventionally, space vehicles are known to use either radioisotope thermoelectric generators (RTG), batteries, solar panels, or fuel cells to provide adequate electric power to the system. Considerations when choosing the power source include physical characteristics of the landing site including sun exposure and topography, complexity of the power system, safety for crew members, lifespan of the components, and overall mass penalties that stem from the source's energy and power density. Table 5, below, shows a brief qualitative analysis of each power source considered during this design.

**Table 5. Trade study of each power source**

| Power Source | Advantages | Disadvantages |
|---|---|---|
| **RTGs** | No moving parts that can fail or wear, highly reliable, extensively used, can operate with or without an atmosphere, provides heat to warm the rest of the spacecraft's electronics, extensive lifespan | Radiation exposure possible for the crew, expensive, inefficient heat conversion |
| **Batteries** | Not very complex, can be lightweight for shorter missions not requiring a lot of power, many different kinds of batteries depending on mission requirements, area of high developmental research attention | Can't be recharged if they are used as primary power source, become heavier and volumetrically larger for longer missions and more power demands |
| **Batteries + Solar Panels** | Battery can be recharged if used in conjunction with photovoltaic energy | Lack of sun exposure and difficult topography could hinder the expansion of the solar panels |
| **Fuel Cells** | No need to recharge, only produce water and heat due to their high efficiency operation, no mass penalties associated for longer missions because they are energy conversion (not storage) devices, no moving parts, operation is silent, can operate even when the ambient temperature is less than the fuel cell operating temperature | Regenerative fuel cells desired for Mars missions but technical readiness is still low, electrolyte membrane will degrade over time |

### A. Radioisotope Thermoelectric Generators (RTGs)

RTGs operate based on the fact that radioactive materials, specifically plutonium-238 in this case, generate heat as they decay. Thermocouples then convert this heat into electricity for the spacecraft [37]. Figure 17 shows an expanded view of the current RTG model called the Multi-Mission Radioisotope Thermoelectric Generator (MMRTG), which was used on the two Viking landers as well as the Pioneer 10 and 11 spacecraft. This RTG can be used either on a planet with an atmosphere or in the vacuum of space, hence the Multi-Mission naming convention. Furthermore, the excess heat produced is beneficial in maintaining a proper operating temperature for the entire spacecraft [38]. The MMRTG is designed to provide 120 W of electrical power for at least 14 years [39].

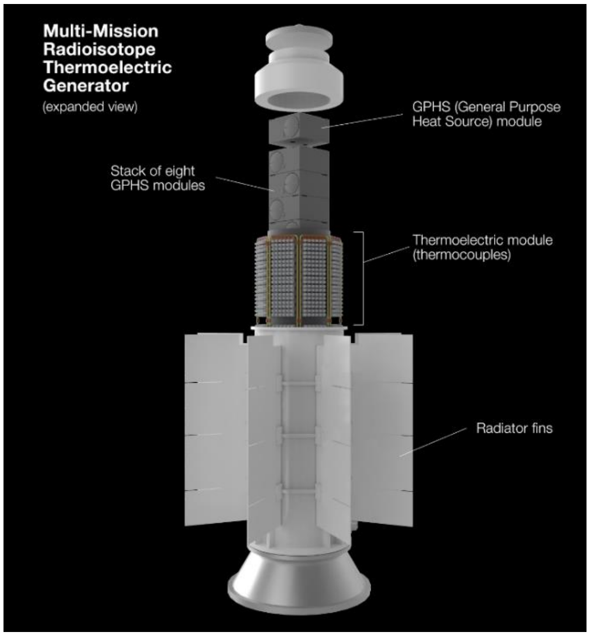


**Figure 17. Expanded view of the current Multi-Mission RTG [38]**

RTGs have been used on spacecraft for a combined time of over 300 years, and because they have no moving parts that can fail or wear, the thermocouples used to convert the heat into electricity have never once stopped providing power. As such, RTGs are considered a highly reliable and attractive option for many space missions, especially long-term, deep-space probes that have one chance of performing their mission and no chance of being repaired [38]. An added benefit to long-term missions is that because plutonium-238 has a half-life of 87 years, even after five years, the heat resulting from the decay is still outputting at 96% of the original heat output. The power output curve for plutonium-328 is seen in Fig. 18 [40]. Additionally, many deep-space probes, such as Voyager and Cassini, use RTGs for power also because their power generation is independent of solar flux, as will be a factor for the solar panel option.

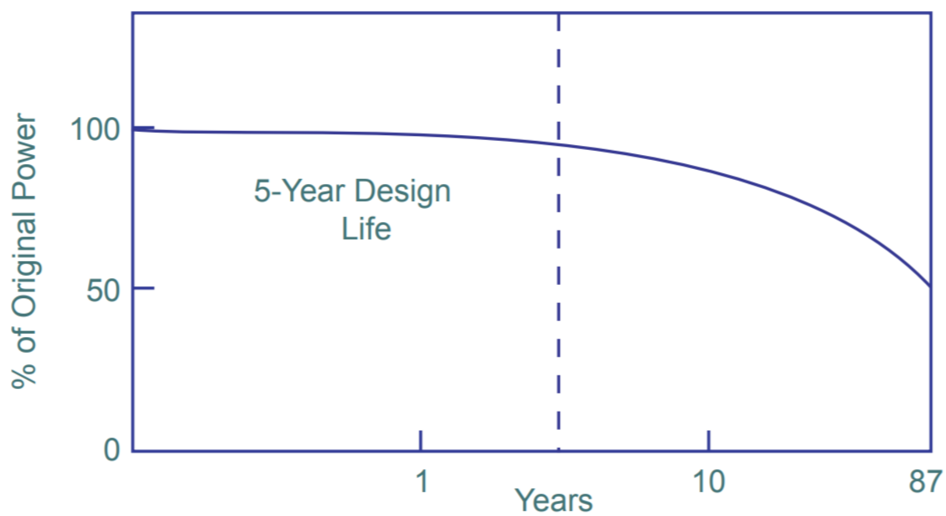


**Figure 18. Power output over time for plutonium-238 [40]**

Despite these benefits, RTGs are currently expensive and inefficient [41]. The thermocouples have a relatively low energy conversion efficiency, typically in the range of only 5-9% conversion of the heat generated by the radioactive decay [42]. Furthermore, because they generate power from radioactive isotopes, the safety of the crew becomes an issue as shielding them from extra radiation becomes an imperative design consideration. Though RTGs have a long list of advantages, the disadvantages associated with their use are substantial enough concerns, and therefore RTGs were not chosen.

### B. Batteries and Solar Panels

Batteries offer a minimal-complexity, inexpensive, and safe option for spacecraft power. There are two classifications of battery usage - primary or secondary usage. Primary batteries are used instead of any solar power, but without the harnessing of solar energy, these batteries cannot be recharged [43]. Typically, batteries as a primary source of power are seen on short-duration, minimal-power missions. Because this mission requires large amounts of power and will need to support humans on the Moon for up to seven days at a time, batteries as a primary and sole source of power were not an ideal choice, and were therefore not analyzed any further.

On the other hand, secondary batteries are used in conjunction with a photovoltaic system. The solar array will provide the main source of power, but during the times that the sun's energy is not available to the spacecraft, the battery will take over. When the solar cells are in use, they are also used to recharge any energy depletion in the battery [43]. Though this combination is more suited for longer-duration missions such as this one, there are a few complications associated with it. Because batteries are energy storage devices, the more energy the spacecraft needs, the larger the battery must be to accommodate the storage demands. While batteries used on short-duration and lower-power missions are relatively light, for a mission such as this one, the battery storage demands would add considerable mass to the lander [44]. Furthermore, the topography of the landing site adds concerns to the choice of expansive solar panels. First, though the South Pole landing site experiences the greatest amount of direct sunlight, there are areas of the South Pole that are potentially void of direct sunlight, specifically in the craterous areas that have the most water ice for the ISRU benefits [45]. Second, if the lander were to land in a crater, the solar panels could potentially have issues expanding. As such, the battery and solar panel option was eliminated from consideration.

## C. Fuel Cells

Fuel cells, unlike batteries, are not energy storage devices, but rather energy conversion devices. Polymer electrolyte membrane (PEM) fuel cells draw upon oxygen and hydrogen (either pure or from another hydrogen source like liquid methanol) to produce an electrical current and a pure water byproduct, as seen in Fig. 19 [40]. The oxygen is supplied to the cathode and the hydrogen is supplied to the anode. The hydrogen protons are separated from the electrons at the anode, and the protons continue towards the electrolyte while the electrons travel through a circuit to produce a current and heat. At the cathode, the hydrogen protons combine with the oxygen molecules and the electrons to form pure water [46].

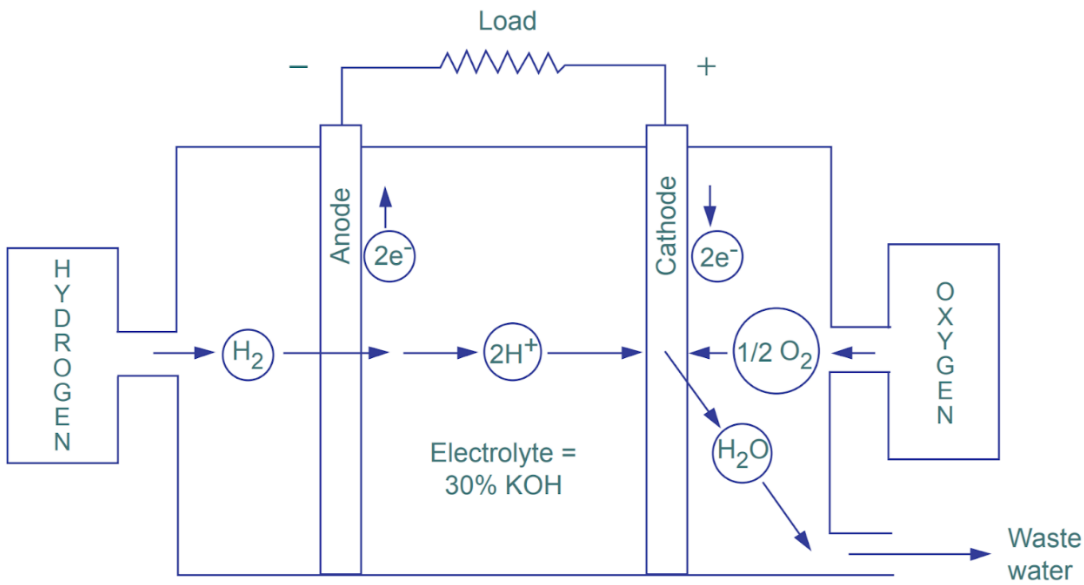


**Figure 19. Fuel cell operational diagram [40]**

Fuel cells have an extensive list of advantages. First and foremost, the fuel and oxidizer used in a hydrogen fuel cell (hydrogen and oxygen, respectively) can be obtained from the Moon's resources via the ISRU integration serving as the foundation of this mission. That means that even though fuel cells require constant resupply of these gaseous consumables, using a hydrogen fuel cell will allow resupply to come directly from the Moon. In addition, because fuel cells are simply converting chemical energy into thermal-electric energy, they do not require recharge. Hence, they can operate without additional sources of power, and longer-duration missions don't require additional storage requirements and thus mass penalties. Next, fuel cells have no moving parts. Even though they are complex pieces of engineering, the absence of moving parts means quiet operation and reliable, long-term use. Fuel cells can also operate under ambient conditions where the temperature is less than the operating temperature, meaning they can even operate in the harshly cold vacuum of space [46]. Finally, as the only byproduct of a fuel cell's operation is pure water, the fuel cells can serve as the primary source of water for the crew's needs. In contrast to the wide array of advantages, fuel cells arguably only have one major disadvantage - the electrolyte membrane will degrade over time and will eventually need replaced [47]. However, not much information is known currently about specifically how long the membrane will last, and this topic will need to be researched more in the future to obtain a full understanding of the lifespan of this lander.

Recently, regenerative fuel cells (RFC) have gained attention for use in long-duration space missions due to their sustainability. RFCs operate in both forward-operation and reverse-operation. During forward-operation these fuel cells perform exactly as normal fuel cells do. However, the sustainability comes into play during reverse-operation when a current is supplied from a solar cell to electrolyze the water byproduct back into hydrogen and oxygen for future forward-operation [45]. RFCs are gaining a lot of attention recently because they offer a recyclable alternative to a standard fuel cell that will become important for missions to Mars. However, due to this lack of technology readiness by the 2030 timeframe, and because these require an additional source of power from solar panels, which have already been removed from consideration, a standard PEM fuel cell will power A*VOL*LO without the regenerative capabilities. This also allows A*VOL*LO to draw all of the necessary water from the fuel cells without the need for external water supplies.

## D. Fuel Cell Design

A simplified schematic of the power system is shown in Fig. 20. The hydrogen and oxygen will be drawn from the common hydrogen and oxygen storage tanks that will also feed the propulsion and life support systems. The water byproduct will be directly fed into the life support system for all human water needs.

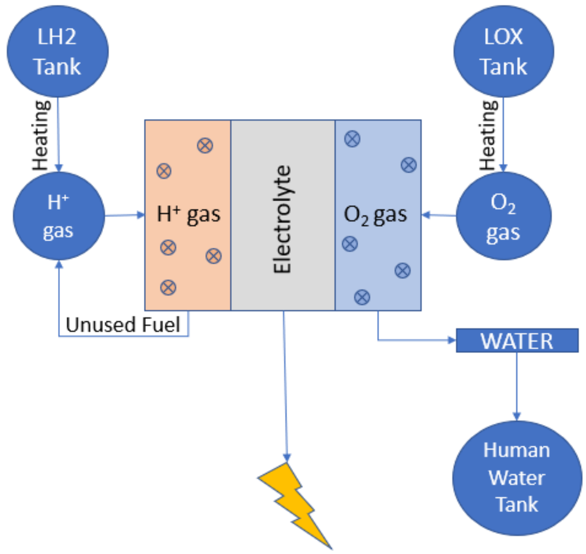


**Figure 20. Simplified power system schematic**

In order to design the fuel cell system, the lunar lander from the Constellation Project, Altair, was studied. Altair was proposed to require a maximum of 6 kW of power, as seen in Fig. 21 as the lunar descent power requirement [48]. Since Altair was roughly on the same scale as A*VOL*LO with the same mission requirements, A*VOL*LO's power requirement was modeled after this one.

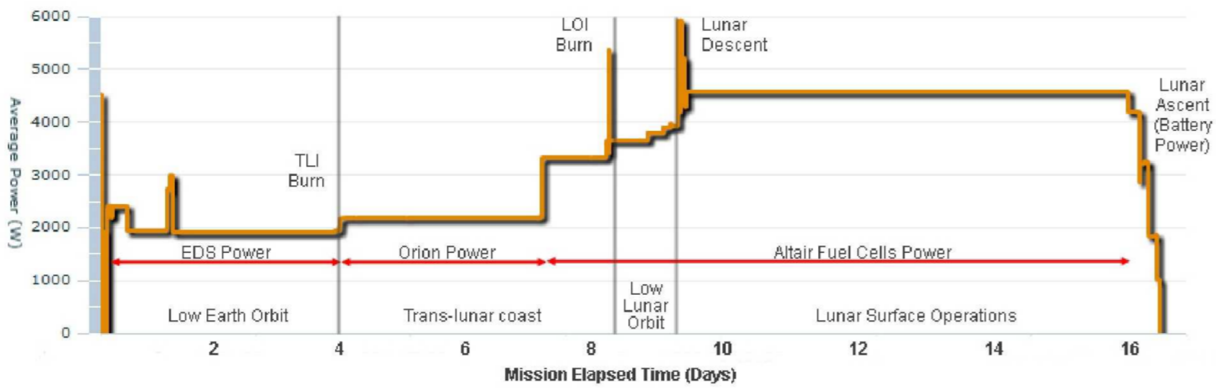


**Figure 21. Altair power requirements for different phases of the mission [48]**

Based on the Department of Energy's goals for fuel cell operations by 2020, the specific power of this fuel cell will be 2,000 W/kg and the power density will be 2,250 W/L [49]. As such, the fuel cell will weigh 3 kg (~6.5 lbs) and its volume will be 0.0026 m^3 (2.6 L). Because fuel cells are scalable, multiple cells can be stacked together in series or parallel to achieve the required power. Our design was based on the assumption that the stack would be arranged in series. Hence, each cell would experience the same current flow and the stack would experience the summation of all the voltage drops across each cell, where it is assumed each cell experiences the same voltage drog. With this in mind, Eq. (1) was used to determine how many cells needed to be stacked. The total power required, $P$, is equal to the total current, $I$, multiplied by the total voltage, $V$. Fuel cell calculations normally involve a current density, $i$, given in units of amperes per cubic centimeter (A/cm$^2$). Therefore, the total power is equivalent to the current density multiplied by the cell's area, $A$, the voltage per cell, $V$, and the number of cells aligned in series, $\#_{cells}$. Table 6 details the design parameters, including common values for the current density, voltage output, and cell area [50].

$$P = IV = iAV\#_{cells} \tag{1}$$

**Table 6. Fuel cell design parameters**

| | |
|---|---|
| **Power required (*P*)** | 6 kW |
| **Current density (*i*)** | 0.7 A/cm^2 |
| **Cell area (*A*)** | 100 cm^2 |
| **Voltage per cell (*V*)** | 0.75 V |
| **Number of cells required ($\#_{cells}$)** | 115 |

As the fuel cell will be the primary source of water throughout the mission, it was necessary to verify how much water this fuel cell would produce per day. To do this, Eq. (2) was used where $\dot{m}_{H_2O}$ is the amount of water produced per day, $n$ is the equivalent electrons per mole of reactant, $F$ is Faraday's constant, and $MW_{H_2O}$ is the molecular weight of water. Using the values tabulated in Table 7, the water production per day will be approximately 65 kg. It should be noted that fuel cell water management is a concern, as this is more than enough water per day (specific water needs will be discussed further in the Environmental Control and Life Support System section). There are some excess water mitigation techniques available including raising the operating temperature to allow some of the water to evaporate, or it could be collected in backup water storage.

$$\dot{m}_{H_2O} = \#_{cells}\frac{iA}{nF}MW_{H_2O} \tag{2}$$

**Table 7. Variables affecting water production rate**

| | |
|---|---|
| **Equivalent electrons (*n*)** | 2 eq e-/mol |
| **Faraday's constant (*F*)** | 96,485 C/eq e- |
| **Molecular weight ($MW_{H2O}$)** | 18 g/mol |
| **Water production ($\dot{m}_{H_2O}$)** | 65 kg/day |

Finally, Eqs. (3) and (4) were used to determine the overall oxygen and hydrogen requirements for this mission. The anode and cathode stoichiometry numbers, $\lambda_a$ and $\lambda_c$, respectively, are based on efficiency and are defined as the actual rate of fuel/oxidizer delivered to the anode/cathode divided by the theoretical rate of fuel/oxidizer delivered. An optimistic efficiency of 60% was used, so that the stoichiometry numbers were both equal to 1.67 [50]. Ultimately, the hydrogen and oxygen requirements for a seven-day mission will be approximately 84 kg and 674 kg, respectively. Table 8 tabulates the required values used in these equations.

$$m_{H_2} = \#_{cells}\#_{days}\lambda_a\frac{iA}{nF}MW_{H_2} \tag{3}$$

$$m_{O_2} = \#_{cells}\#_{days}\lambda_c\frac{iA}{nF}MW_{O_2} \tag{4}$$

**Table 8. Variables affecting total hydrogen and oxygen consumption**

| Variable | Hydrogen | Oxygen |
|---|---|---|
| **Stoichiometry number (λ)** | 1.67 | 1.67 |
| **Molecular weight (MW)** | 2 g/mol | 32 g/mol |
| **Equivalent electrons (n)** | 2 eq e-/mol | 4 eq e-/mol |
| **Total mass produced (m)** | 84 kg | 674 kg |

Finally, after knowing the power requirements of the mission, the graph in Fig. 22 was considered to verify the choice of fuel cells for a 6 kW and 7-day mission. As is indicated by the red marker, a fuel cell is a sufficient choice to meet the requirements.

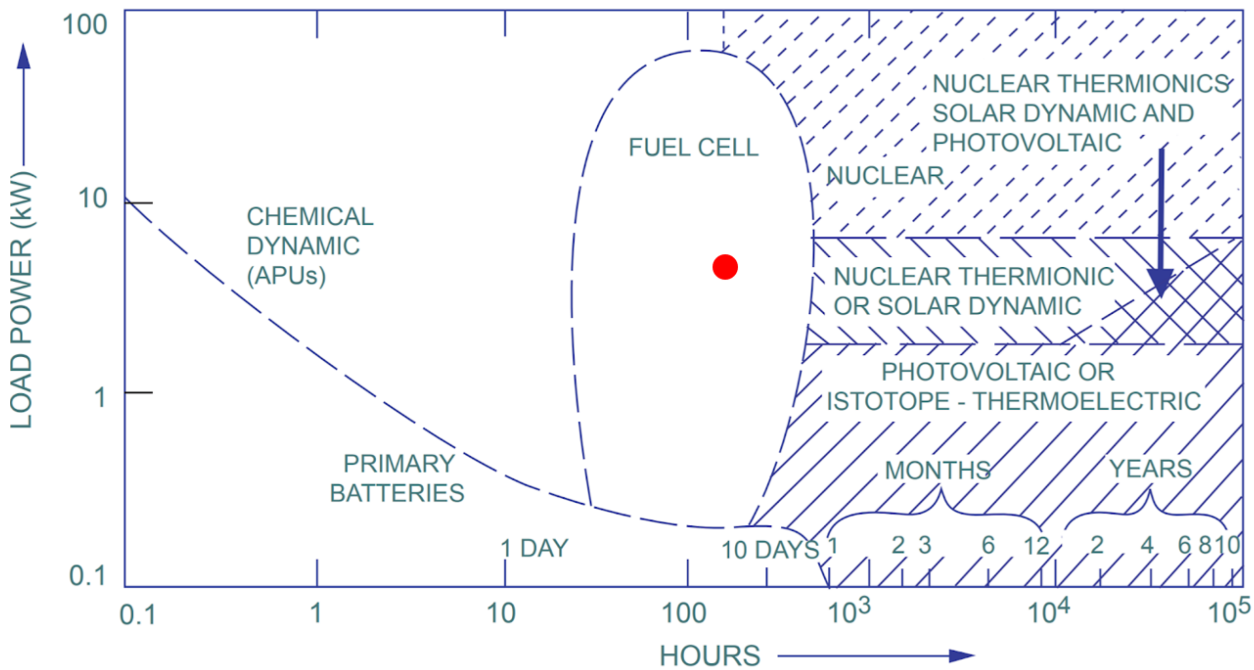


**Figure 22. Applicability of various power sources depending mission length and power requirements [40]**

## VII. Structure and Material Selection

Some common problems associated with space travel include thermal cycling (repeatedly moving into and out of the sun's path), outgassing (the removal of trapped gases from a material due to the extreme vacuum of space), and life limitations due to ultraviolet radiation and exposure to a wide temperature gradient [51]. The material and structural decisions made for this lunar lander were based on these issues as well as the load bearing capabilities dictated by the propulsion system. Table 9 details some of the material options that were analyzed for use on the lunar lander.

**Table 9. Material choice trade study**

| Component | Material | Advantages | Disadvantages |
|---|---|---|---|
| **Main body + Crew compartment** | Aluminum alloy composite | Lightweight, durable, pressure capable | Expensive thermal processing required |
| **Main body + Crew compartment** | Titanium | High strength, durable, long-lasting | Very heavy compared to other options |
| **Footings** | Aluminum honeycomb (sandwiched between aluminum alloy composite of main body) | Lightweight, high stiffness, fire resistant, compression and shear resistant | Historically only used for one descent journey |
| **Footings** | Aluminum alloy composite | Lightweight, durable, pressure capable | Will require reinforcements at joints to support long term durability and reusability |
| **Propellant tanks** | Fluoropolymer composite material | Oxygen compatible, usable at cryogenic temperatures, lightweight, low CTE | Newly developed (2005). initially prone to microcracking under thermal cycling |
| **Propellant tanks** | Kevlar and epoxy | Commonly used in composite overwrapped pressure vessels (COPVs) for space missions | Prone to buckling in the inner liner; new methodologies working to combat this |

For the mainframe of the structure, it was decided that a 10mm-thick aluminum alloy would be the best option as it is relatively lightweight, but still provides the structural rigidity necessary for long term use. Basic aluminum alloys have been used on similar spacecraft, as it provides a reasonable balance between performance, cost, and availability [11]. The benefit of using a metal like aluminum is that it is extremely malleable, but when alloyed, has a much greater strength and is therefore commonly used in the construction and transportation industries. The only drawback to using metal alloys is that the production process can require a great deal of thermal processing, but because aluminum is inexpensive and weak on its own, alloying has multifaceted benefits [52]. Additionally, while aluminum is an excellent compromise between weight and strength when alloyed, it is sometimes insufficient for areas of extremely high loads. In these areas (mainly joints and local load concentrations), titanium will be used. Using titanium only at the essential points will minimize weight additions and allow for wider margins of the structure's weight bearing capabilities.

The first non-mainframe, but equally essential, components that were studied were the propellant tanks. Propellant tanks are typically either spherical, cylindrical, or a combination thereof. Spherical tanks were chosen for this mission because they occupy a lesser volume, can bear greater force as stress is evenly distributed across the surface, and weigh less overall [53]. Figure 23 shows the stress forces across a cylindrical and a spherical pressure vessel. As can be seen from the stress diagrams, the cylindrical pressure vessel has both tangential and axial stresses, while the spherical vessel has only a singular membrane stress that is uniform across the entire body [54]. For systems that will require high pressurization of propellants, as well as the ability to withstand the extreme vacuum of space, spherical pressure vessels offer a stronger, more reliable storage option for a long-duration mission.

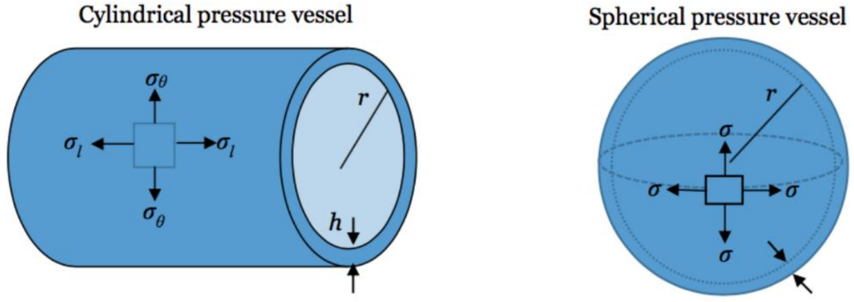


**Figure 23. Stress comparison for cylindrical versus spherical pressure vessels [54]**

These propellant tanks, which need to be capable of carrying $LH_2$ and LOX, require specialized materials developed for handling high pressure gases in the vacuum of space. In a majority of recent space missions, propellant tanks have been composite overwrapped pressure vessels (COPVs) [55]. COPVs consist of two main parts: a composite overwrap and a metallic inner liner. The purpose of this dual-material tank is to contain substances under high pressure and prevent corrosion should the substance be highly toxic [56]. The material typically used in COPVs, as can be seen in Table 9, is Kevlar 49/epoxy, a heat resistant, extremely strong synthetic fiber. The total thickness of these tanks is consistently less than 20mm [57]; while this decreased thickness can increase the risk of crack formation, the pressure capabilities of the tank provide minimal risk [58]. Currently, there is a risk of the inner lining buckling due to weak adhesion or lack of proper bonding, which can increase the risk of cracking in the liner; however, new methodologies are being developed to prevent this that should make this problem almost entirely avoidable by 2030 [58].

Next, buckling and stress analysis was performed on the struts attached to the landing gear. There are multiple parameters to take into consideration when designing struts, such as the type of material (modulus of elasticity), length of the strut, and the cross-section geometry. Knowing the approximate maximum mass of 56,000 kg (noted in Table 4 as the descent mass), simulations were performed in SolidWorks to ensure failure did not occur while maintaining a lightweight structure. Originally the legs were to be machined out of a titanium alloy because of the higher modulus of elasticity compared to an aluminum alloy. After running the simulations based on a lunar environment (lunar gravity and maximum assumed temperatures), the aluminum alloy was proven to sufficiently handle the estimated load and stresses while maintaining a factor of safety of 8. Once aluminum was verified to be a strong enough material to support the lander, another consideration involved creating an aluminum honeycomb structure for the legs, as detailed in Table 9. This type of material has been utilized in several space missions including Apollo 11 [59]. Figure 24 presents a basic mock-up of the internal structure of the landing strut used on Apollo 11.

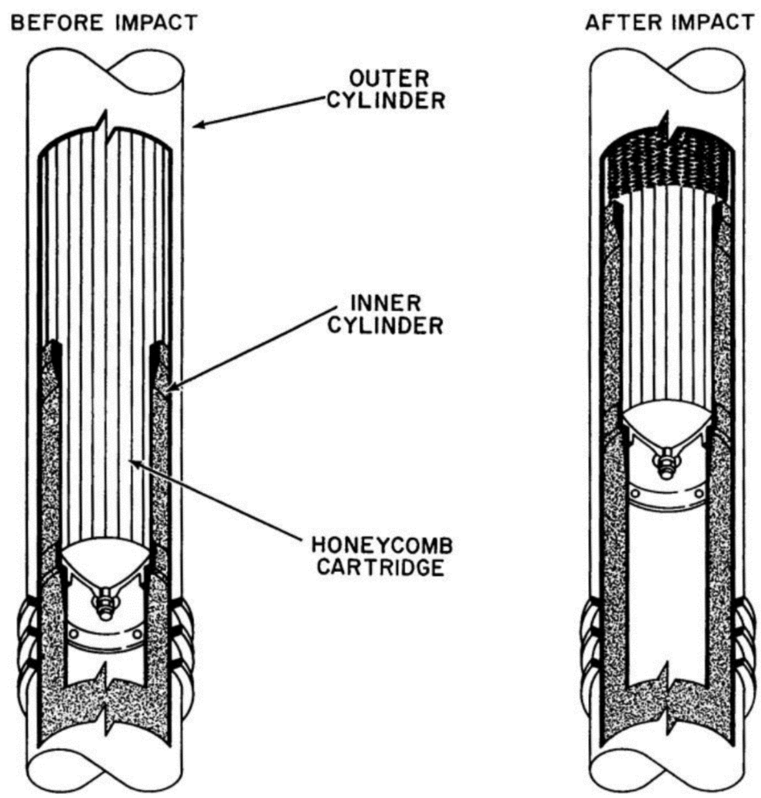


**Figure 24. Landing gear primary strut for Apollo 11, with aluminum honeycomb design [59]**

The benefit of this honeycomb design is that it allows for compressibility in the landing strut to add ease to the landing maneuver and decrease the resulting forces on the struts. While this design was met with great success on the Apollo 11 mission, it was found unsuitable for this mission as it does not allow for reusability; once the honeycomb is compressed, it has essentially lost all structural integrity and wouldn't serve its purpose a second time. The final decision regarding struts was determining how many legs should be used for optimal support. Though three legs are sufficient to keep a structure balanced, A*VOL*LO will have four legs to further reduce the force and stress on each leg.

The housing structure was designed last to ensure all equipment and components could fit inside, while still being mindful of protecting the crew from the harsh lunar environment. The primary constraints for the housing structure were ensuring that it physically fit within the launch vehicle's payload bay, and that its weight was within the launch vehicle's capabilities. As mentioned before, this lander will need to launch atop an SLS like the rest of the Gateway components including Orion. Specifically, A*VOL*LO was designed to launch on an SLS Block 1B configuration that is proposed to be available by the mid-2020s [60]. Though an SLS Block 2 is proposed by the 2030's and will be even more powerful than the Block 1B, it is unclear when in the 2030's it will be available, and as such was not a viable candidate [60]. The SLS Block 1B is the required launch vehicle because it is the only launch vehicle capable of sending Orion and the Gateway components to the Moon. Figure 25 shows the SLS Block 1B payload capacity compared to other vehicles, emphasizing the importance of its development.

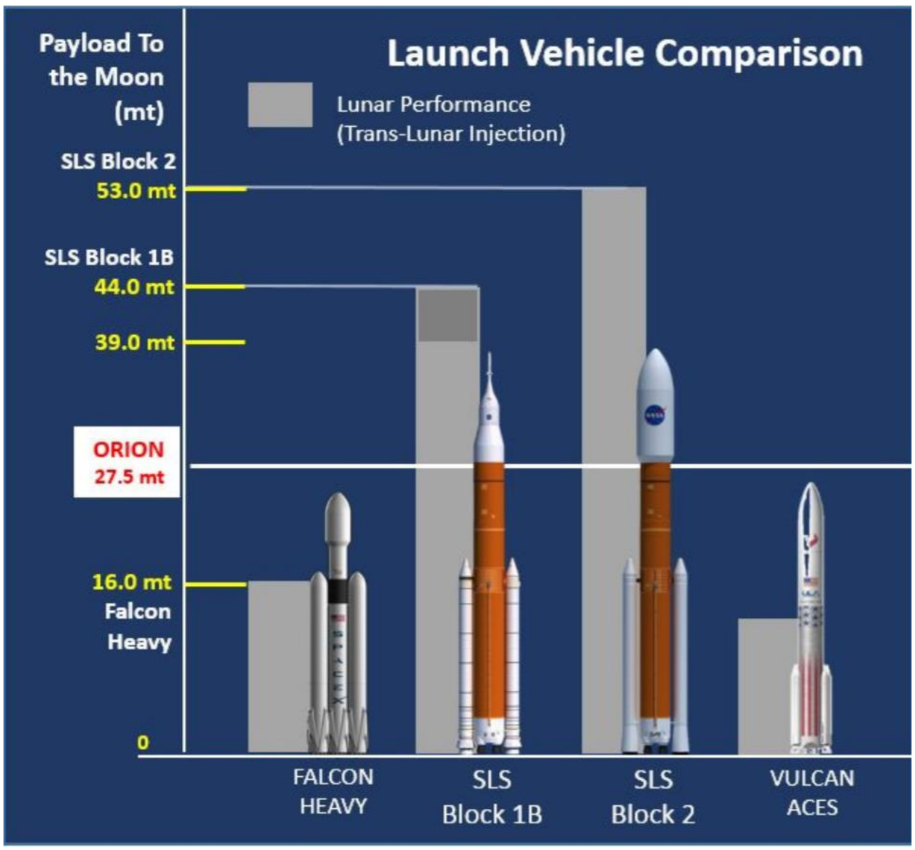


**Figure 25. Launch vehicle comparison for lunar payload delivery [60]**

As can be seen from Fig. 25, the SLS Block 1B can lift up to 44 mt (44,000 kg) to the Moon [60]. A detailed mass breakdown of A*VOL*LO is shown in Table 10, which shows that after all of the subsystems were incorporated (except for an accurate mass estimate for the life support system due to complications in accurately quantifying that mass), the total maximum mass is greater than the SLS launch capability, even without accounting for any margins. However, if the lander is launched from Earth to the Gateway with no oxygen or hydrogen for the propulsion or life support systems on board, it would only weigh approximately 29,000 kg, which is plenty within the SLS's launch capability. The lander will be launched with the power system's necessary hydrogen and oxygen so that once the lander reaches the Gateway, it will be capable of docking. It will then receive the necessary consumables from the on-orbit propellant depots before being home to the four crew members for their seven-day journey to the surface.

**Table 10. Mass breakdown of AVOLLO's components and subsystems**

| Component | Mass (kg) |
|---|---|
| **Engines (x4)** | 640 |
| **Oxygen - Propulsion** | 23,159 |
| **Oxygen - Power** | 671 |
| **Oxygen - Life Support** | 24 |
| **Total Oxygen** | 23,853 |
| **Hydrogen - Propulsion** | 3,860 |
| **Hydrogen - Power** | 84 |
| **Total Hydrogen** | 3,944 |
| **Nitrogen - Life Support** | 30 |
| **Fuel Cell** | 3 |
| **Navigation** | 10 |
| **Structure** | |
| **Kevlar LH2 Tank (x4)** | 2,348 |
| **Kevlar LOX Tank (x2)** | 972 |
| **Landing Gear (x4)** | 2,238 |
| **Outer Structure** | 3,077 |
| **Reaction Control System (x4)** | 136 |
| **Ladder** | 99 |
| **Chairs (x4)** | 307 |
| **Control Panels** | 2,144 |
| **Windows (x5)** | 34 |
| **Miscellaneous** | 2,271 |
| **Total Structure** | 13,627 |
| **Max Payload (Descent)** | 15,000 |
| **Overall** | **56,329** |

Furthermore, the physical dimensions of the lander must fit within the payload bay of the SLS Block 1B, visualized in Fig. 26. The largest version of the Block 1B payload fairing has an internal diameter of 7.5 meters, and a height of approximately 27 meters [60]. A*VOL*LO, which is seen in Fig. 27, has a diameter of 7.4 meters with the struts in the stored position, and a height of 8 meters. Future iterations of the CAD design should take into account the necessary safety margins such that the diameter of 7.4 meters is slightly reduced to allow adequate room on all sides for vibrations during launch.

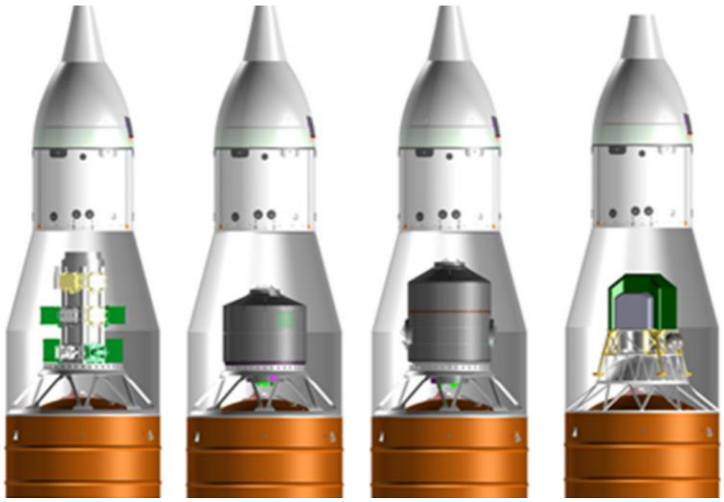

**Figure 26. SLS Block 1B payload size constraint visualization [60]**

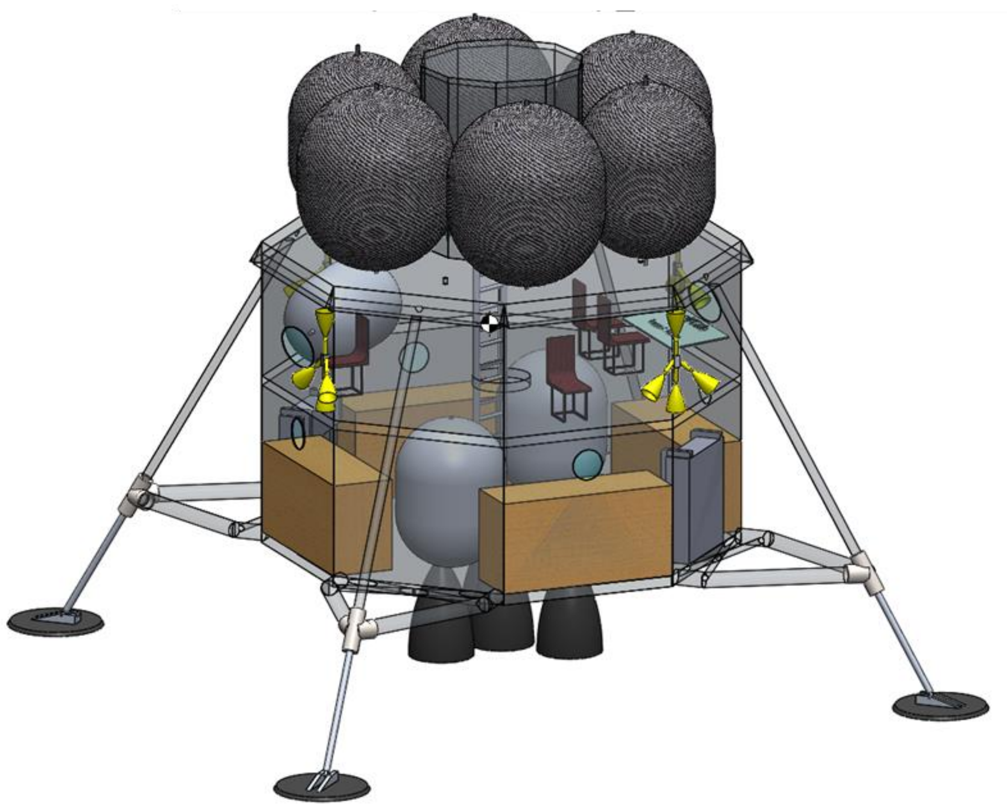

**Figure 27. CAD design of A*VOL*LO**

## VIII. Environmental Control and Life Support System

The environmental control and life support system (ECLSS) for A*VOL*LO was designed with the same sustainability goals as the rest of the subsystems. By depending on ISRU for the oxygen and hydrogen resupply and the fuel cell for the primary source of water, the only consumable that will need replenished from Earth is nitrogen, which for the entire mission is only approximately 30 kg. Basic requirements for ECLSS designs include atmosphere supply, control, and revitalization; carbon dioxide and trace contaminant removal; a potable water supply; waste management; temperature and humidity control; fire detection and suppression; and extra-vehicular activity (EVA) capabilities including suit connections and an airlock. The three primary systems will be discussed further in this paper: atmosphere supply, control, revitalization, contaminant control, and water supply.

### A. Atmosphere Supply, Control, and Revitalization

In compliance with recommendations from NASA's Exploration Atmospheres Working Group (EAWG) for a lunar lander, the main cabin will have the capability to pressurize to either 8.0 psia / 32% O2 or 10.2 psi / 26.5% O2. These requirements are based on human physiology needs as well as the compatibility with the Crew Exploration Vehicle (CEV), Orion. Furthermore, these atmospheres were determined to be the most desirable for this type of mission to promote efficient EVA preparation. EVA preparation includes the need to pre-breathe oxygen, where pre-breathing is defined as the time in the EVA suit after purge and leak-check until the absolute pressure on the body is 4.3 psia. Pre-breathing oxygen decreases the risk of decompression sickness, and is partially dependent on what the cabin atmosphere is. Certain atmospheres lend themselves to having shorter pre-breathe time requirements, and these two atmospheres were determined by the EAWG to be adequate and ideal for shorter duration lunar landing missions with high-frequency EVA excursions [61].

The remaining gas percentage will be comprised of nitrogen. A nitrogen/oxygen simplified model of Earth's atmosphere will provide a comfortable environment for the astronauts while also simplifying design parameters and minimizing the possibility of hypoxia, decompression sickness, or cabin fires. These two gases, however, will need to be resupplied into the atmosphere at different rates due to a variety of factors, including human consumption and spacecraft leakages. This was taken into consideration when determining how to store the oxygen and nitrogen before they are fed into the cabin. The gases could be stored separately or as a mixture. Storing them as a mixture not only decreases the required tanks and therefore volume and mass, but it also yields a less complicated design with less parts because there is no need for a mixing device. However, if stored as a mixture, the mixture would have to be resupplied from Earth. On the other hand, if stored separately, and if the oxygen is retrieved from the common LOX tanks used to supply the propulsion and power systems, the life support system could also benefit from the ISRU integration. Furthermore, because oxygen is consumed by humans at the rate of 0.84 kg per day, and because nitrogen often leaks through the walls of spacecraft structures, storing the gases separately is beneficial for needing to resupply these at different rates [62, 63]. The nitrogen and oxygen will flow from their respective tanks into a mixing chamber, where it will then be released into the cabin atmosphere. Portable fans will be stored onboard at various places throughout the cabin to ensure that the air is properly mixed at all locations; this will eliminate pure oxygen pockets that could cause fires and pure nitrogen pockets that could cause asphyxiation.

### B. Contaminant Control

Unwanted contaminants, including carbon dioxide ($CO_2$), microbes, and dust, will inevitably build up if not carefully monitored and removed due to the lack of interaction with an outside environment. As such, there are three primary components to a contaminant management system: trace contaminant removal, $CO_2$ removal, and high-efficiency particulate air (HEPA) filters. The trace contaminant control system (TCCS) for A*VOL*LO will mimic that of the ISS, consisting of a charcoal bed, a catalytic oxidizer, and a lithium hydroxide post-sorbent bed, as seen in Fig. 28 [64]. Most contaminants will physically adsorb onto the charcoal bed, but those that aren't easily adsorbed will pass through the catalytic oxidizer. Through thermal catalytic oxidation, the contaminants will be oxidized to carbon dioxide and water. The oxidation products will then pass through a lithium hydroxide post-sorbent bed where they will be adsorbed along with any unwanted acids produced during oxidation [64]. Outside of the TCCS, dust particles, aerosols, and allergens will be removed by HEPA filters in the return air ducts.

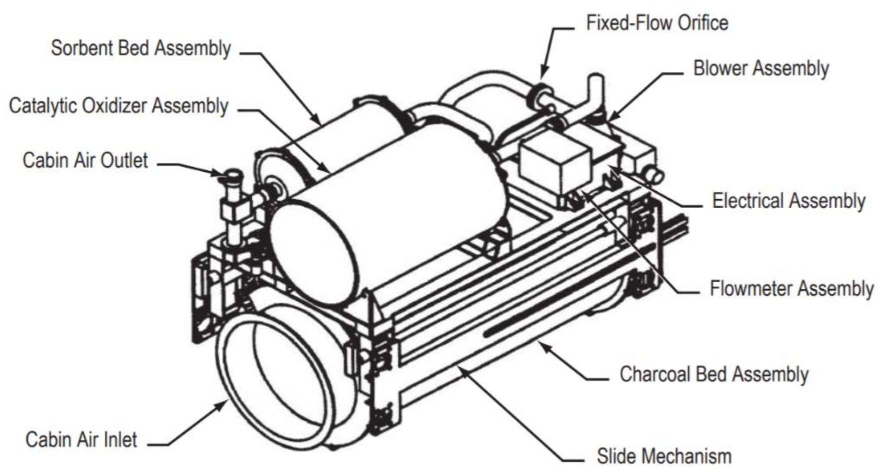


**Figure 28. Trace Contaminant Control System used on the ISS [64]**

The third contaminant management system is the carbon-dioxide removal system. As with any engineering system, especially those used in space applications, no perfect $CO_2$ removal system exists. Ideally, future Mars spacecrafts will solely rely on the Sabatier process where $CO_2$ is reacted with hydrogen ($H_2$) to produce methane ($CH_4$) and water ($H_2O$). The methane could then be used as a propellant (especially because methane is obtainable through Martian ISRU processes), and the water could be split into oxygen and hydrogen atoms for other uses [65]. Unless the methane is also decomposed, this process results in a net loss of hydrogen, therefore requiring resupply [66]. In consideration of the reusability goals for this lander, as well as the technical readiness level desired by 2030, the Sabatier reaction was not chosen as the $CO_2$ removal system. Instead, and though it comes with flaws of its own, A*VOL*LO will use an appropriately sized version of the regenerable Carbon Dioxide Removal Assembly (CDRA) that is currently onboard the ISS. The CDRA on the ISS is a four-bed molecular-sieve system that not only adsorbs $CO_2$ and vents it overboard, but also collects excess water molecules from the air to serve as humidity control, which can be fed into the water supply system [67].

### C. Water Supply

As noted above, one of the main reasons for choosing fuel cells as A*VOL*LO's power source was the added benefit that their only byproduct is water. Thus, the primary source of water will come from the fuel cell productions, with backup water supplies held in contingency water containers resupplied from the ISRU mining sites or the excess fuel cell water production. Table 11 details the minimum water needs per person per day for a space mission. These will obviously vary based on the mission requirements, but they can serve as a baseline. The fuel cell will output approximately 65 kg of water per day, which is an incredible amount more than is needed. As was mentioned in the Power section, the water output can be decreased by changing operating parameters of the fuel cell itself, or from a life support standpoint, the astronauts could increase their daily water usage to include showering, flushing, etc.

**Table 11. Minimum water needs per person per day [68]**

| | |
|---|---|
| **Drinking** | 1.62 kg |
| **Food prep** | 0.76 kg |
| **Shower (min)** | 1.82 kg |
| **Handwash** | 4.09 kg |
| **Assume no urine flush** | 0 kg |
| **Total** | **8.29 kg** |

The water will pass through a microbial control system before feeding into either the EVA suit connectors, the drinking water storage, or the hygiene water access. The drinking and hygiene water sources begin the waste management control system. The drinking water will be used to rehydrate food, and any food waste will be collected in a combined wet trash/fecal storage bladder. The industry standard waste collection system (WCS), otherwise known as a toilet, will be integrated into the waste management system. The WCS separates fecal matter from urine and disposes both to their corresponding bladders [69]. The wet trash/fecal and waste water storage bladders will be returned to the Gateway and returned to Earth on Orion. Urine is typically vented overboard when in orbit, but if a permanent lunar base is the end goal for these lunar missions, that is not a sustainable practice [70].

A simplified schematic of the three major systems noted above can be seen in Fig. 29. The life support system unfortunately makes the lunar lander less reusable than will ultimately be required on Mars. Food is not grown on the lander, human waste is returned to Earth, and $CO_2$ is not recycled back into the system, thus making this an open-loop life support system. Though much progress is being made toward a fully-closed life support system, there is not one readily available yet. Given the relatively short duration of the landings, this amount of purging and resupply was deemed acceptable.

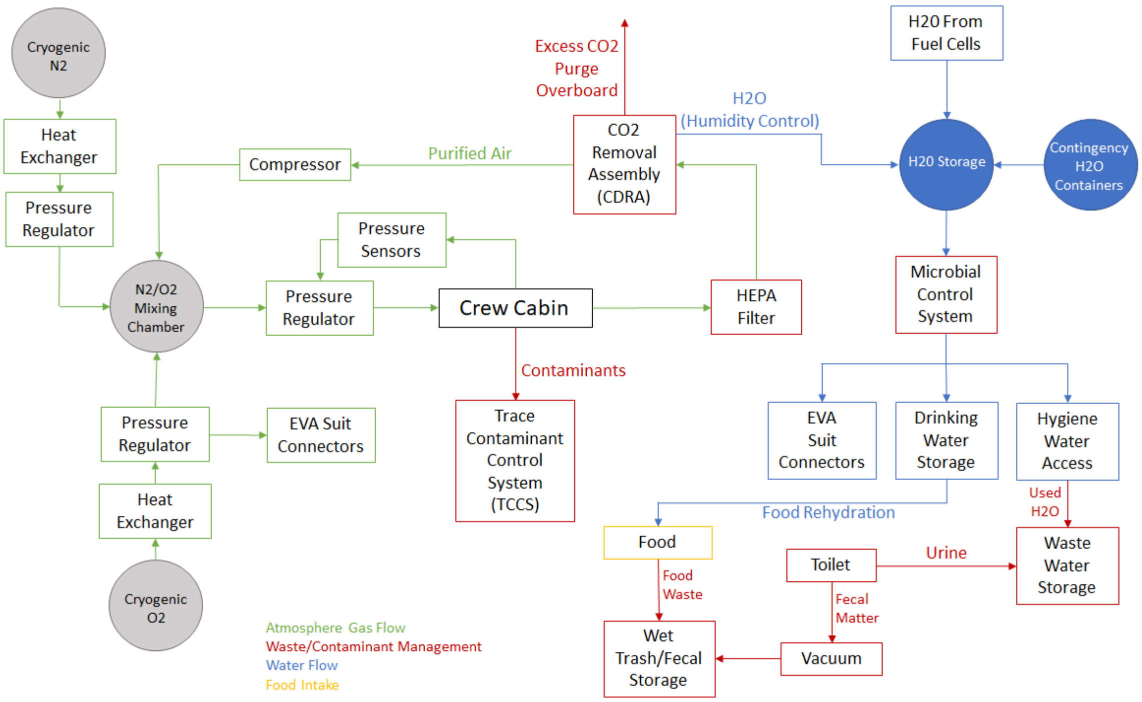


**Figure 29. Simplified ECLSS schematic**

# IX. Conclusion

This year, the National Aeronautics and Space Administration, in conjunction with AIAA, created a design competition to develop a reusable lunar lander capable of docking with the Lunar Orbiting Platform - Gateway and performing a variety of tasks on and around the Moon. While this project was outlined with this competition in mind, some parameters and requirements were adjusted, as detailed in the Introduction section of this paper. As such, a lunar lander, A*VOL*LO, was designed with these capabilities: to support a crew of four for up to seven days, to deliver a payload of 15,000 kg to the surface of the moon, to deliver a payload of 10,000 kg from the Moon back to the Gateway, to fit inside the payload bay of NASA's SLS Block 1B, and to sustainably operate back and forth between the Gateway and the South Pole of the Moon. In developing this lander, all critical subsystems and structures were analyzed, including the in-situ resource utilization techniques that would be implemented on the Moon. The final components of each system were selected after careful consideration of technology readiness levels, mass and size constraints, durability of materials, and reusability/sustainability goals. NASA's motivation in establishing this project was to ultimately develop a lander that could be used for a long-duration mission to Mars. A*VOL*LO provides a cost-effective, reliable, and sustainable permanent presence beyond low earth orbit that will serve as a stepping stone towards this larger goal by allowing astronauts to be trained and technologies to be tested in a much safer proving ground, the Moon.

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