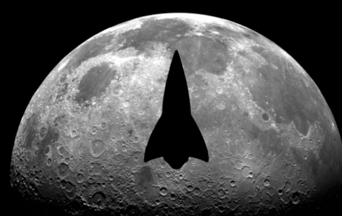
Team Voyager



LUNARPORT

A launch and supply station for deep space missions

Caltech SPACE CHALLENGE

March 26-31, 2017

sponsored by AIRBUS DEFENCE & SPACE

Preface

This document is the final report for Team Voyager of the 2017 Caltech Space Challenge. All worked enclosed herein was performed over five days (March 26-31, 2017) at the California Institute of Technology by a team of graduate and undergraduate students from universities around the world.

The Caltech Space Challenge is a 5-day international student space mission design competition. The topic of the 2017 Space Challenge was 'Lunarport;' a station which extracts resources from the polar regions of the Moon, converts the resources to usable rocket propellants, and refuels space vehicles in an effort to greatly improve deep-space accessibility for future missions. Students from a wide range of backgrounds were invited to Caltech, formed into two teams, and given the mission design problem. The teams attended lectures related to mission planning, were given the necessary development tools, and were challenged to produce a viable mission design. This confluence of people and resources was a unique opportunity for young and enthusiastic students to work with experienced professionals in academia, industry, and national laboratories.

"Ice Rush"

Just over 70 years ago, spurred by the discovery of gold in the Sacramento Valley of California, a flood of prospective gold miners flocked to the state of California, expanding its population by a hundredfold in under three years. History has shown us that the discovery of valuable resources can be a powerful driver of human expansion, technological advancement, and societal prosperity.

In just the last decade, water has been discovered at the surface of the Moon near and inside permanently shadowed regions of its south pole in quantities far larger than ever previously known. These unique polar regions are amongst the coldest known areas in the solar system, due to eternally receiving little or no sunlight, allowing for water to exist in stable ice forms. Such ice can be mined and processed into propellants for rockets traveling to destinations around the solar system. Because the most expensive part of space travel is leaving Earth's atmosphere and gravitational pull, the ability to produce large quantities of rocket propellants in an accessible location away from the Earth gives this ice tremendous value to humankind.

It is team Voyager's hope that this study can pave the way for a new kind of rush, an "Ice Rush" in which humankind first begins to leverage resources from beyond Earth, fostering international collaboration in cis-lunar space, and empowering us to make the next great leap - to the red planet and beyond.

Acknowledgements

Team Voyager would like to thank the California Institute of Technology (Caltech), the Graduate Aerospace Laboratories of the California Institute of Technology (GALCIT), and the NASA Jet Propulsion Laboratory (JPL) for hosting the 2017 Caltech Space Challenge. We would also like to thank the sponsors of the event for allowing such an incredible event to take place (Airbus Defence and Space, Microsoft, Keck Institute for Space Studies, Orbital ATK, Northrop Grumman, Moore-Hufstedler Fund, Blue Origin, Boeing, Lockheed Martin, Schlumberger, and Honeybee Robotics). We are also extremely grateful to Ilana Gat and Thibaud Talon, the student organizers, as well as the several other student experts who helped make the event a reality. Finally, we thank the many technical experts who gave relevant lectures and graciously committed their time to provide their technical advice on the many challenges faced this week.

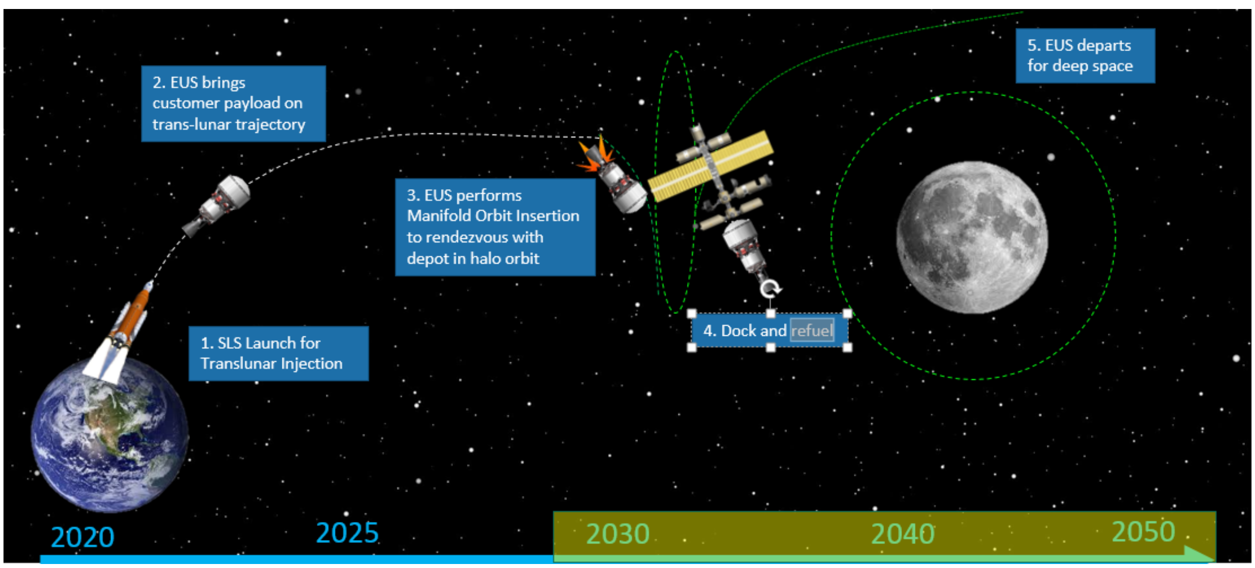
Team Voyager

Team Member	University	Major
Bryan Sinkovec	California Institute of Technology	Space Engineering
Daniel Pastor	California Institute of Technology	Space Engineering
Donal O'Sullivan	California Institute of Technology	Astrophysics
Gary Li	University of California, Los Angeles	Aerospace Engineering
Jack Henry de Frahan	ISAE-SUPAERO	Applied Math, Aerospace Engineering
Joseph Sparta	Georgia Institute of Technology	Aerospace Engineering
Matt Vernacchia	Massachusetts Institute of Technology	Aeronautics and Astronautics Engineering
Mercedes Herreras Martinez	Arizona State University	Aerospace Engineering
Nathan Sharifrazi	University of California, Irvine	Aerospace Engineering, Mechanical Engineering
Nick Jamieson	University of Cambridge	Aerospace Engineering
Sophia Casanova	University of New South Wales	Mining Engineering
Sumudu Herath	University of Cambridge	Civil Engineering
Sung Wha Kang	Rhode Island School of Design	Industrial Design
Sydney Katz	Washington University in St. Louis	Electrical Engineering, Systems Engineering
Therese Jones	Pardee RAND	Policy Analysis
Vinicius Guimaraes Goecks	Texas A&M University	Aerospace Engineering

Executive Summary

Since first starting to traverse and populate the Earth, curiosity and the desire to explore have been intrinsic qualities of all humankind. The next phase in human exploration takes us to deep space; to Mars, the outer solar system, and beyond. We are already taking our first tentative steps in this direction. Mars 2020 will build on the incredible success of Curiosity, helping us to understand the history and ultimate fate of our closest solar system analog. NASA's strategic roadmap highlights "boots on Mars" as a goal for the 2030s (Northon, 2015). Just this week, NASA announced the Deep Space Gateway, which will put a temporary human habitat in a cis-Lunar orbit for crewed missions to Mars (Smith, 2017).

In this rapidly evolving context, we present a preliminary design of a Lunarport – a refueling station for deep space missions. With a construction budget of $1 billion per year, we aim to provide 650 metric tons of propellant to refuel a total of five Exploration Upper Stages (EUS) of NASA's Space Launch System (SLS) by 2032. Current design reference architectures estimate that this is the number of SLS upper stages that would be required for a complete cargo and crew mission to Mars (two for cargo, one for crew; see: Drake, 2009).



Illustrated above, our design consists of an orbital space depot at the Earth-Moon L1 Lagrange point with a large (~811 metric tons of LH_2/LO_2 propellant) storage capacity which is fueled by a Lunar Resupply Shuttle (LRS) cycling between the depot and Lunar surface. On the surface we have a launch pad, power generation system, a fleet of ice mining rovers, H_2O storage, and an electrolysis plant to provide propellant for the LRS to return to the depot. Our three main design principles were technical feasibility, autonomy and scalability. These principles were chosen to push Lunarport to be operational as early as possible so that it can begin to capitalize on its propellant production at a lower level and scale up to full capacity in an economically efficient way, rather than building immediately to full capacity on our prescribed budget.

Our system architecture highlights scalability and modularity in a number of ways. Rather than relying on a central mining station – we build on existing technologies to develop a custom self-contained mining robot which we estimate to extract ~10 kg of ice/water per day from the lunar regolith. Six of these miners can fit inside the payload of a Falcon Heavy rocket, allowing us to scale up our fuel production to approximately 150 metric tons per year by 2031 with one launch every two years until 2027, and one per year until 2031. The LRS can transport 5.5 metric tons of H_2O per trip to the depot, and has a lifetime of 10 trips. With two LRS deliveries per year, we can keep the depot full enough to completely fill a full Mars mission every six years. With three LRS deliveries per year, we can completely service the referenced Mars missions every four years.

While our design focuses on technological and economic feasibility, the long timeline encourages us to push the technological limits and drive development where possible. Our fleet of mining robots utilizes emerging technology for drilling and power. For drilling, we utilize Planetary Volatile Extractor Corer (Zacny, 2015) developed by Honeybee Robotics (Technology Readiness Level, TRL 5), which has shown breakthroughs in water extraction efficiency from lunar regolith. For lunar surface power, we take advantage of proven solar array technology for power generation, but introduce wireless power transmission to beam power from the solar arrays to the rovers in the dark shadowed regions. Wireless power transmission has been accomplished high efficiency on the ground and is currently TRL 5. In-space refueling is also a low TRL area in which significant development will be required. The only tested technology to date transmitted 8 kg of liquid in a number of weeks, however we will need on the order of tons in a matter of days by the 2030s, requiring a significant push for faster automated docking and transfer.

With our design of Lunarport – our capacity for both Mars and other deep space missions will be greatly expanded, with key lessons about refueling in space, reusable rocketry on extraterrestrial bodies, in-space wireless power transmission, autonomous resource utilization and the economics of extraterrestrial solar system resources.

1 Introduction 8

2 'Ice Rush' – Mission Overview 9

2.1 Mission Statement 9

2.2 Objectives and Constraints 9

2.3 Top-Level Requirements 9

2.4 Mission Justification 11

3 Lunar Site Selection 15

4 System Architecture 20

4.1 Lunar Surface Systems 20

4.1.1 Sintering Robot 21

4.1.2 Lunar Prospector 22

4.1.3 Mining Robots 23

4.1.5 Power 26

4.1.6 Supporting Infrastructure 28

4.1.8 Environmental Protection and Control 31

4.2 Space Systems 32

4.2.1 Lunar Resupply Shuttle (LRS) 32

4.2.2 Orbital Fuel Depot 42

4.2.3 Solar Electric Propulsion (SEP) Tug 56

4.2.4 Lunar Transfer Vehicle 59

4.2.5 Lunar Lander 61

5 Mission Design 62

5.1 Launch Vehicle Selection 62

5.2 Orbit Selection 64

5.3 Payload Optimization 66

6 Operations 69

6.1 Base Operations 69

6.2 Construction Timeline 70

6.3 Operational Timeline 71

7 Programmatic Considerations 74

7.1 Developmental and Operational Cost 74

7.2 Long Term Roadmap 76

7.3 Political Considerations 77

7.4 Planetary Protection 78

7.5 Public Relations and Outreach 78

References 80

1 Introduction

The desire for deep space missions that advance our current state of knowledge regarding our solar system and beyond has created a demand to launch heavy payloads on high energy trajectories. In particular, the drive to send humans to Mars has maintained a strong presence in space planning over the past few decades. In order to achieve these momentous tasks, it is important to consider architectures that break down deep space mission design into smaller stepping stones that both demonstrate new technology and lay down a framework for a sustainable launch architecture to locations in deep space. While attempting to identify such a stepping stone, the Moon surfaces as a prime candidate due to both its proximity and abundance of resources that can be utilized in-situ.

The gravity well of the moon is significantly smaller than that of the Earth. The energy required to escape Earth's large gravity well is a roadblock in our ability to send heavy payloads into deep space. Every extra unit of mass launched from Earth is costly, and efforts should be made to minimize initial launch mass. For this reason, an architecture that utilizes resources outside of Earth's gravity well in a location such as the moon proves an attractive option for missions that require large launch payloads and high delta-v's. Specifically, utilizing lunar resources to create propellant allows for the launch of greater payload masses to deep space by eliminating the need to launch all of the required propellant for a deep space trajectory out of Earth's gravity well.

Recent lunar observation missions have revealed the potential to make this concept a reality, having identified water ice, methane, ammonia, and other exploitable materials near the Moon's poles (Colaprete et al., 2010). These materials suffice to produce the propellant and infrastructure on the moon required to create a refueling station. With this knowledge in mind, this study outlines a design and architecture for a launch and supply station on the lunar surface called Ice Rush. A complete lunar base and refueling architecture is laid out and considered in the context of a human Mars mission. In addition to the development and demonstration of technology, Ice Rush creates a framework for refueling deep space missions that intends to push the limits of human exploration capability further than ever before.

2 'Ice Rush' – Mission Overview

2.1 Mission Statement

Construct a depot in space which will supply vehicles with propellants created with resources extracted from the lunar south pole, greatly reducing the cost of deep-space missions and enabling humanity's reach to extend further than ever before.

2.2 Objectives and Constraints

The Lunarport mission objectives and constraints are listed in **Table 2.2a**. Congress has allocated $1 billion per year to construct an autonomously-functioning Lunarport for deep space travel, which will be able to refuel a 2032 Mars mission of two cargo vehicles and one crew vehicle, based on the requirements of the Mars Design Reference architecture (Drake, 2009).

Table 2.2a. *Lunarport Mission Objectives and Constraints.*

Mission Objectives
● Lunar refueling port for deep space travel
● Autonomous construction and operation
● Scalable design to 600T depot propellant capacity by 2032
Mission Constraints
● Surface station required near polar ice caps
● $1bn/year construction budget
● Self-sustaining resource model

2.3 Top-Level Requirements

For the Lunarport mission, Team Voyager was asked to design a launch and supply station using lunar resources, with funding of $1B per year for the duration of the mission. The Level 1 and Level 2 requirements given are provided in **Table 2.3a.**

Table 2.3a. *Top-level requirements.*

1.0	**Level 1 Requirements**
1.1	Lunarport shall provide propellant to a vehicle or set of vehicles traveling to a destination within the solar system
1.2	Lunarport shall provide cost savings for refueling vehicles in comparison to direct trips between Earth and the vehicle's final destination
1.3	Lunarport shall provide propellant produced using resources from the Moon
1.4	Lunarport shall be a robotically operated system
1.5	Costs shall be limited to $1 billion/year, with unused funds available to roll over to future years
2.0	**Level 2 Requirements**
2.1	Lunarport shall be able to accommodate eventual human visitors
2.2	Once operating at full capacity, Lunarport shall provide sufficient propellant to support a crewed mission to Mars which includes cargo and crew
2.3	At full operating capacity, Lunarport shall be capable of supporting the intended rate of crewed Mars missions
2.4	Lunarport shall ultimately be a financially self-sufficient refueling facility
2.5	Lunarport shall be built and tested incrementally in order to reduce technical and financial risk
2.6	The components of Lunarport shall have sufficiently long lifetimes to make the maintenance of the base economically feasible
2.7	NASA is the initial funding source for the mission, with transition to the commercialization of the facility possible for long-term use

2.4 Mission Justification

As an Executive Branch agency, the strategic goals of NASA are subject to both changes in presidential administrations and Congressional budget constraints. The 2017 NASA Transition Authorization Act (S.442) prioritizes cis-lunar exploration as a first step to a crewed Mars mission, stating, "the United States should have continuity of purpose for the Space Launch System and Orion in deep space exploration missions, using them beginning with the uncrewed mission, EM–1, planned for 2018, followed by the crewed mission, EM–2, in cis-lunar space planned for 2021, and for subsequent missions beginning with EM–3 extending into cis-lunar space and eventually to Mars [in the 2030s]." While the ability to fulfill this plan is dependent on the passing of a NASA budget which reflects these priorities, these objectives are consistent with NASA's Evolvable Mars Campaign (EMC; **Figure 2.4a**), first announced in 2014 as a path to utilize near-Earth space assets to create Earth-independent crewed missions to Mars of 2-3 years in length (NASA Exploration Forum, 2014).

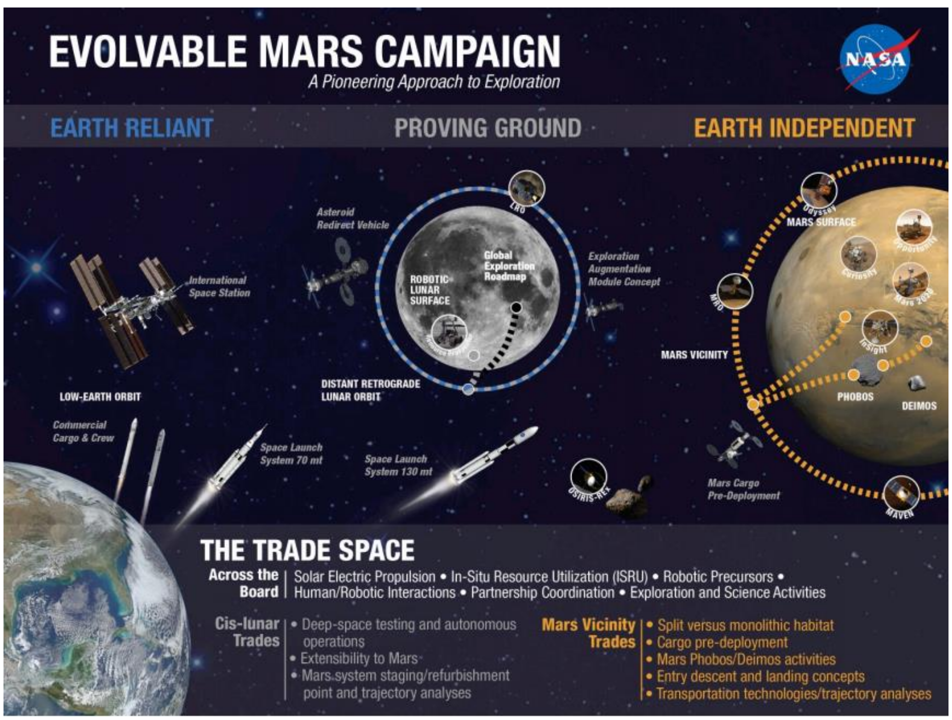


Figure 2.4a. *NASA's Evolvable Mars Campaign (Crusan, 2014).*

Representative John Culberson (R-TX) has asserted that just like Eisenhower was remembered for the creation of the interstate highway system, Trump would be remembered for the creation of an interplanetary highway system, suggesting that this Earth-Moon-Mars transport system is of high priority for the current presidential administration.

A more detailed plan to reach Mars through a Deep Space Gateway (**Figure 2.4b**) was announced by NASA Associate Administrator for Human Space Exploration and Operations William Gerstenmaier on March 28, 2017.

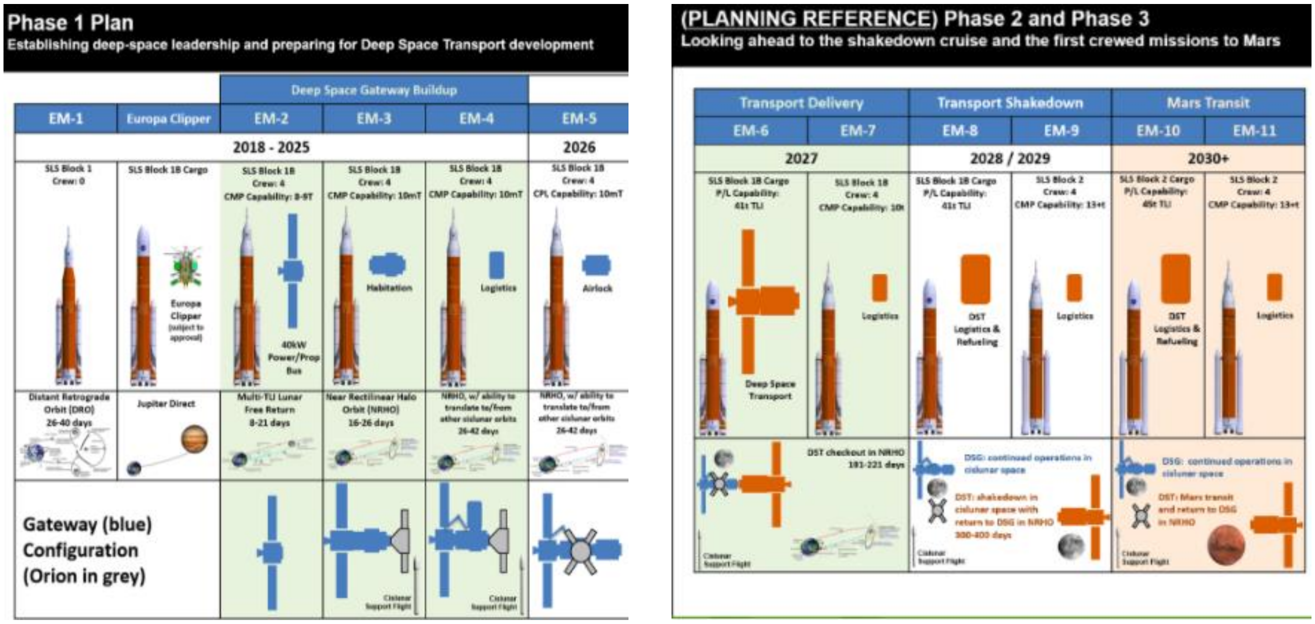


Figure 2.4b. *NASA's plan to reach a deep space gateway announced in March 2017.*

This plan establishes a human path to Mars by 2033 via three phases of development, supported by crewed SLS missions. Phase 1 establishes a Deep Space Gateway (DSG) in cis-lunar space, from approximately 2022-2026, gradually building up the gateway in four stages delivered by SLS: 1. 40kW Power/Propellant Bus, 2. Habitation module, 3. Logistics, 4. Airlock. The DSG is intended to be human-tended, rather than a consistently inhabited station. From 2027-2033, six SLS missions would establish the Deep Space Transport (DST), with an initial delivery of the transport vehicle followed by alternating logistics and logistics and refueling payloads (delivered by SLS crewed and cargo missions, respectively). The entire architecture assumes one crewed SLS/Orion launch per year beginning in 2023 plus one cargo SLS launch per year beginning in 2027 (Smith, 2017).

Lunarport as a Critical Component of NASA's Evolvable Mars Campaign

The in-situ resource utilization (ISRU) technologies used in the Lunarport mission are a follow-on to the Lunar Resource Prospector (LRP), acting as a bridge to future crewed missions in cis-lunar space or on the lunar surface. The present roadmap to the DSG and DST do not presently include a fuel source; a robotic ISRU mission is a logical and cost-effective mechanism to enable Mars transportation. The modular structure of the DSG and DST allow for the Lunarport to gradually ramp up propellant production as the Mars-bound vehicle is completed. In the event of a Lunarport failure, it may be possible to service the Lunarport using the Deep Space Gateway as a staging ground.

Lunarport additionally fulfills a number of Lunar Human Exploration Strategic Knowledge Gaps (SKGs) related to ISRU, lunar surface exploration, and power, as detailed in **Table 2.4a** (NASA, 2016). These SKGs are also integral to further Mars exploration. NASA proving ground objectives for Mars include utilizing Lunar Distant Retrograde Orbit as a staging ground to Mars, utilizing ISRU in micro-gravity, and operations with reduced logistics capabilities (Crusan, 2014).

Alternative uses of Lunarport

Lunarport may also be used for other solar system probes. Additionally, a number of stakeholders have expressed interest in both crewed and uncrewed lunar missions. The European Space Agency's Moon Village concept or Bigelow's proposed lunar habitat could be good candidates for future customers of the Lunarport. The United Launch Alliance (ULA), has also been promoting an architecture, CisLunar-1000, which seeks to have 1000 people living and working in cis-lunar space within the next 30 years (Kutter, 2016), suggesting ULA may develop commercial technologies that may be used by NASA in Lunarport construction and be a potential long-term stakeholder in a cis-lunar propellant depot. Section 8.12 details the potential long-term commercialization of the Lunarport and stakeholders whose launch timelines and mission profiles align with the Lunarport construction and operation timeline.

Table 2.4a. Lunar Human Exploration Strategic Knowledge Gaps Addressed by Lunarport

Strategic Knowledge Gap		LP Relevance
I. Understanding the Lunar Resource Potential		
D-3	Physical characteristics of entrained volatiles	VH
D-4	Understand slopes, elevations, block fields, cohesiveness of soils, trafficability	VH
D-5	Landed missions to understand the charge reservoirs (plasma or ground) in the low conductivity environment	VH
D-6	Determine the form, concentration and distribution of volatiles, how they vary from depths 0-3 m over distances of 10-100m scales.	VH
E	Understand the volatile contents of RDMDs, as well as their depth and distribution	LM
G	Measure the actual efficiency of ISRU processes in the lunar environment.	M
III. Understand How to Work and Live on the Lunar Surface		
A-1	Collect raw materials; create trenches, roads, berms, etc.; enables ISRU, surface trafficability, and ejecta plume mitigation.	VH

A-2	Load, excavate, transport, process, and dispose of regolith; enables ISRU, surface trafficability, and ejecta plume mitigation.	VH
A-3	Crush, grind regolith; understand effects of comminution; enhances ISRU process efficiency.	VH
B3	Ability to remotely traverse over long distances enables a) prepositioning of assets, and b) robust robotic precursor missions.	H
B4	Autonomous landing capability for robotic missions similar to that demonstrated by Chang'e-3 lander.	VH
C2	Characterization of geotechnical properties and hardware performance during regolith interactions on the lunar surface.	H
D4	Multiple landings at the same location on the lunar surface may scour or damage systems and equipment already emplaced at that location. Ejected regolith velocity, departure angles, and energy in engine plume exhaust need to be measured in situ to better understand mitigation strategies	M
F2	Polar missions may be in areas with extended solar availability; blackouts may extend to 3-5 days requiring 100s of kW-hours; batteries will be prohibitively expensive.	VH

3 Lunar Site Selection

The approach taken for site selection and the development of full scale mining operations on The Moon has been adapted from the traditional terrestrial mining life cycle model (**Figure 3a**). By taking this phased approach, the hope is to reduce geological uncertainty through detailed prospecting and exploration, which may increase initial cost and take additional time in order to complete a thorough exploration program, but will aim to reduce financial and engineering risk in the longer term.

Figure 3a. *Typical Mining Life Cycle.*

The exploration and prospecting stage would seek to identify potential prospects of a chosen site and consider the following key factors:

Geological Factors:

- What is the likelihood that a water ice deposit exists in a region undergoing investigation? - This analysis would be conducted prior to initial prospect site selection and would form the basis of that selection in conjunction with an engineering feasibility analysis. A preliminary analysis of this kind is outlined below.
- Does water ice exist in the area of interest, if so in what quantities and of what quality? - The answer to question would need to be addressed by a detailed exploration program conducted by a rover.
- What is the likelihood that the quantity and quality of the deposit will differ after mining from what was expected at the time the mine was initially developed? - This question considers the confidence in exploration data in order to estimate resources and reserves. If there is not sufficient confidence that the estimates will not change dramatically after mining, then further exploration, appraisal, and geological modelling of the site is required.

Technical Factors:

- Can the resource be extracted and processed with existing or likely future technologies? - This question relates to the engineering constraints and capabilities of the proposed mining equipment. Factors that may be considered under this question include: the depth of the ice, the temperature of the surrounding environment, terrain and topography, rock strength and the geomechanical properties of any overburden/regolith etc. The prospecting rover would be required to assess these factors in conjunction with the geological data.

Economic Factors:

- Can the water be extracted economically? - Given both the geological and technical factors of the site does it make economic sense to extract water from the moon to supply propellant to Mars at this particular location.

Recent discoveries by the Lunar Reconnaissance Orbiter and the LCROSS impactor suggest the possibility of significant water ice deposits on the surface and the upper subsurface of the shadowed regions of the south Lunar poles where temperatures may be as low as >40K. Modelling suggests that subsurface water is likely present in temperature conditions of <100K and less than <70K for other volatiles such as carbon dioxide and hydrogen sulfide (**Figure 3b**).

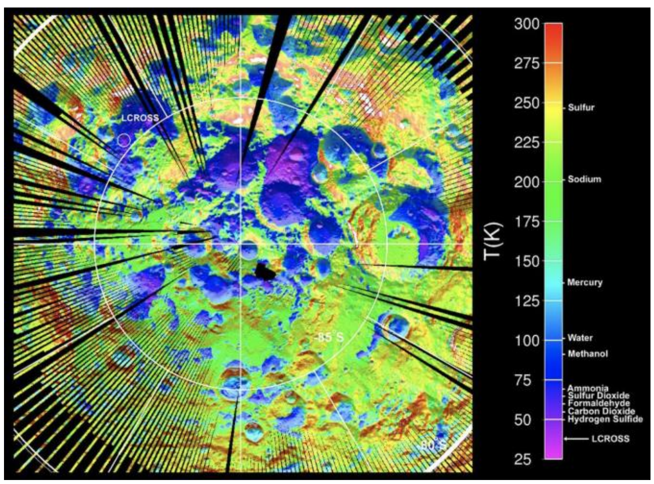


Figure 3b. *LRO Diviner Lunar Radiometer Experiment surface temperature map of the south polar region of the Moon (NASA, 2010).*

The selection of the proposed mining site is based on a number of factors. The primary target is the extraction of water, so the initial analysis focused on locations where water ice is most likely to be present based on orbital data. Site selection is restricted to where water ice could be stable within the top 10-20cm (red; **Figure 3c**) with regions of possible stable water on the surface (white) representing the most prospective regions. Stable water ice on the surface coincides with the permanently shadowed regions of the lunar surface. Thus far there has only been one in-situ measurement of water in the south lunar pole which was derived from the LCROSS impactor in the Cabeus Crater.

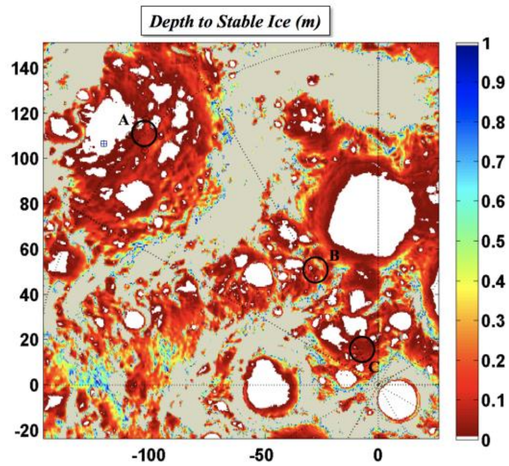


Figure 3c. *Depth to stable ice (m) and proposed Lunar Resource Prospector Landing Sites (Image credit: Sanders 2016,* https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20110014548.pdf*). Chosen Landing Site located in the white regions in proximity to site C. Cabeus Crater is located near site A.*

In conjunction with geological considerations prospects also need to be chosen based on engineering constraints. These may include number of days of light visibility (**Figure 3d**) which has implications for power requirements, days of direct to Earth communication (**Figure 3e**) which has impact on communication capability and slope due to engineering constraints for rover mobility and a landing site for the resupply shuttle. Thus it is imperative that a site be selected based on meeting the engineering requirements for landing and operation as well as be co-located with prospective water ice resources.

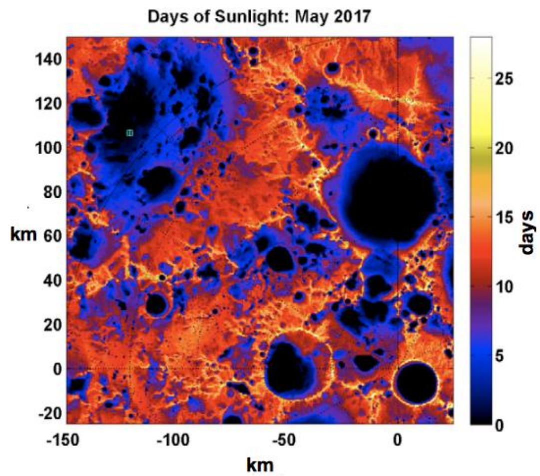


Figure 3d. *Days of sunlight (May 2017). The color scale runs from 0-28 days. The landing site has > 20 days. Landing site in white box (Image credit: Sanders 2016).*

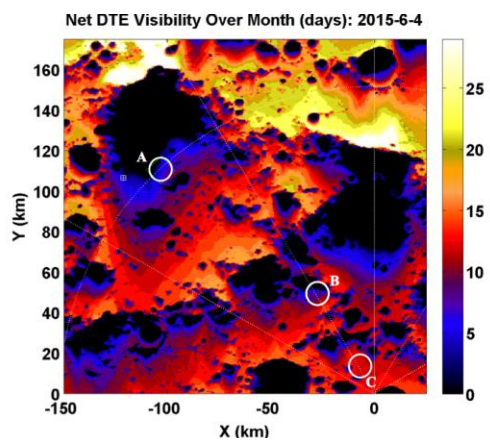


Figure 3e. *Net Direct to Earth (DTE) visibility. Landing site white box (Image credit: Sanders 2016). Chosen site located in Proximity to site C.*

The team has identified the following requirements for a suitable base site (landing site/launch pad facilities, storage, and power facilities), transportation route, and mining site. The criteria are summarized in **Table 3a**.

Table 3a. *Base site selection, transportation route, and mining site criteria.*

Base Site Selection Criteria	Transportation Route Criteria	Mining Site Criteria
● Slope <5 degrees ● Light availability >20 days ● Landing ellipse > 500m ● Absence of identifiable hazardous terrain	● Slope <20 degrees ● Transversable terrain ● Minimal distance to mining site	● Slope <10 degrees ● Depth to stable water <10 cm ● Temperature <100 K ● Limited identifiable hazardous

Based on these criteria, the team located a number of potential sites. The final selection was chosen on the basis of being the most favorable in meeting both selection criteria for geological factors as well as engineering design and operational constraints.

The site is located in the lunar south pole (approximately 50km north-west of Shackleton Crater (**Figure 3f**). Cabeus Crater was considered as a potential target location as it is favored from a geological uncertainty standpoint, being the only location where water has been definitively proven thus far as shown by the LCROSS impactor results. However; the current site is preferable over Cabeus Crater on the basis of landing and operational constraints which made Cabeus unsuitable due to long travel distances required between sunlit areas and the PSRs and the extreme temperatures in the central part of the crater (<40K) which would make operations, given current technology, extremely difficult and expensive. Prospecting missions will sure up resource estimates and ensure that they are appropriate for long term mining operations. Therefore, the mining site locations and prospects within may change as more scientific information becomes available.

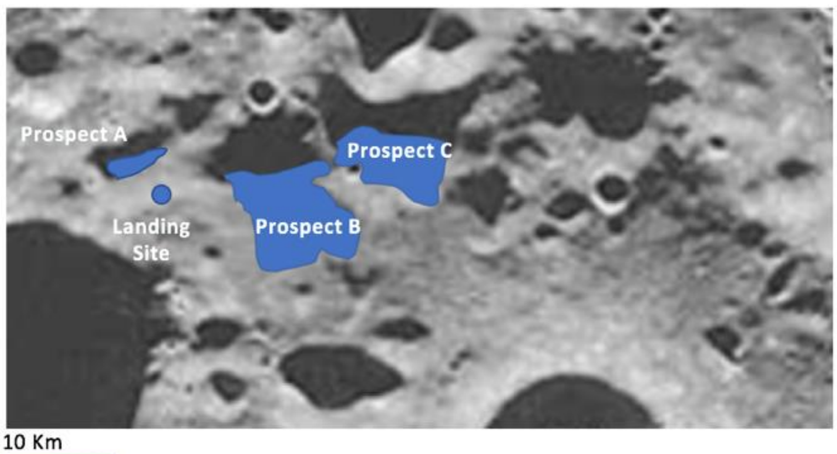


Figure 3f. *Prospect Site Selection locations.*

Three potential prospects have been selected all of which are located within permanently shadowed regions. Prospect A is located in close proximity to the landing site (~500 m) however the approximate areal extent of the prospect is relatively small. This would be a good first prospect to test operations. Prospect B is located within less than 2 km of the landing site at it's closest point and 15 km at it's furthest point. Prospect C is located much further away from the landing site. To mine this site would either require moving the base operations or redesigning the mining rover power design. If the resource prospector identified resources in this area but current technology constrained its extraction then this prospect may be considered a contingent resource until new technologies to operate in this environment are developed. **Table 3b** outlines the key parameters of the landing site location and the three prospect areas.

Table 3b. *Geological and Engineering Parameters for site selection.*

	Distance from Landing Site	Depth to Stable Water	Temperature	Slope (degrees)	Percentage of Lunar Day Receiving Sunlight	Percentage of Lunar Day with Direct to Earth (DTE) Communication
Landing Site	0 km	< 20 cm	~ 180 K max ~ 90 K min	< 5	~ 70%	60 %
Prospect Site A	0.5 km	Surface - 10 cm	~ 100 K	10	0 - 20%	< 20%
Prospect Site B	~ 2 km	Surface	~ 40-100 K	< 10	0	0
Prospect Site C	~ 12 km	Surface	~ 40-100 K	< 10	0	0

For the purposes of this report, it is presumed that the resource prospector / scouting mission was successful and an in-situ discovery of an economic deposit was made at the proposed site. The exploration stage would then be followed by a detailed feasibility analysis and mine planning phase in which an assessment of the costs and equipment requirements will be made and a mine design plan will be developed. Assuming the site is found to feasible and a go-ahead decision is made, the mine construction stage would then be entered in which the power, communications, storage, electrolysis facilities will be delivered to the site and the construction/sintering rover will begin preparing the site for future mining operations. The mine development and operation stage will begin when the mining rovers are sent to the site and begin their mining / water extraction operations. As the site is further developed attention will turn to locating new prospects. The lunar prospecting rovers will be sent to new locations and the mining life cycle will begin again.

Although beyond the scope of this study it is also important to consider reclamation and mine end-of-life plans which should be considered early in the mine design process. One possible suggestion would be to re-purpose the remaining structures after mining operations have ceased for human habitation and use.

4 System Architecture

4.1 Lunar Surface Systems

The infrastructure of the proposed surface system is fully scalable. Additionally, the surface systems are designed to minimize cost and maximize efficiency while paying special attention to safety and redundancy within each subsystem. A fully functional system is achieved with the shipment of four landers to the lunar surface. Additional shipments of mining rovers are required to scale the system to meet Mars mission propellant demand rates.

The first launch consists of four Cubesats to enable communication with the Earth for the entire lunar day as well as a lunar prospector to scout the proposed site and finalize site selection. The second shipment will arrive two years later, bringing robotic systems and cargo so construction of the surface base may begin. A sintering robot capable of building roads and protective berms will start the infrastructure to prepare for the arrival of the mining rovers. This sintering robot is powered by the lander via beamed microwave energy. The third lander will bring mining rovers so water extraction can begin. All landers are powered by solar panels and contain batteries as well as a thermally protective space for their robots to be stored during the lunar night. Within two years the electrolysis unit, liquid hydrogen, liquid oxygen and the water tanks arrive with the final shipment to enable fuel and water delivery to the depot.

4.1.1 Sintering Robot

A sintering robot is used to construct the road network that will facilitate rovers to move among different components of the Lunarport. It will use a robot similar to ATHLETE, which was developed and tested by NASA and with improved automated docking systems to attach and detach construction equipment (NASA JPL, n.d.). The sintering robot has to work in rapidly changing terrains, temperature conditions and to carry heavy loads while excavation, filling and transporting goods. Hence a robot with higher carrying capacity and a relatively higher speed would contribute greatly towards the efficiency and productivity of the construction operations.

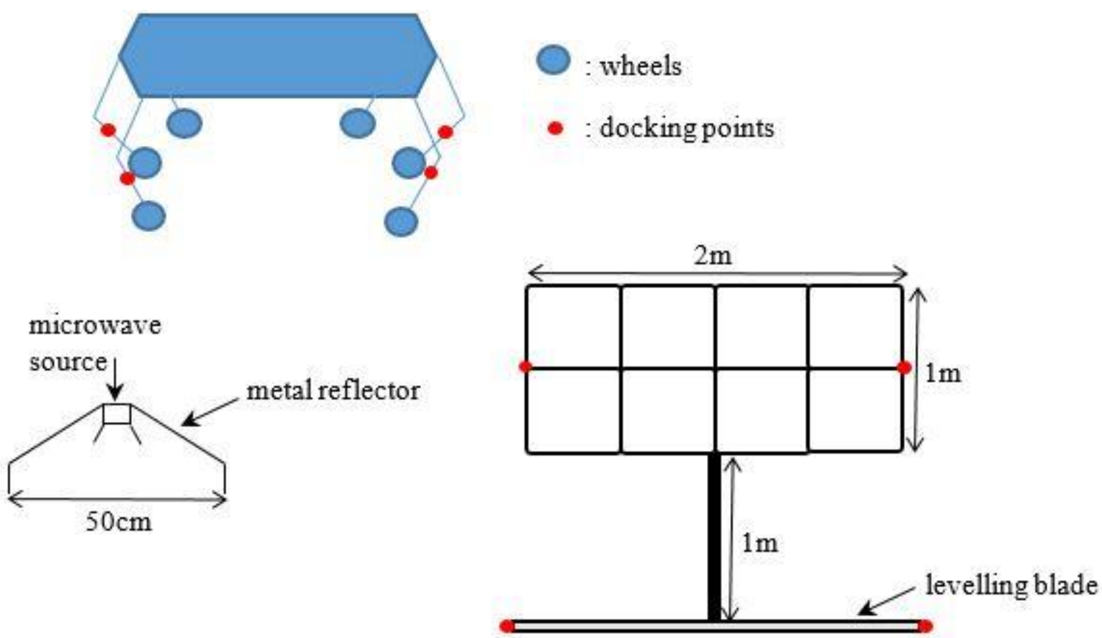


Figure 4.1.1a. *Sintering robot diagram.*

It is expected to have the sintering robot with the ability to detach and replace its components and perform several tasks simultaneously. It will make the construction process much more efficient and productive. And also it is expected to manufacture a third generation ATHLETE robot hence achieving an estimated learning curve of 70%. The mass of the sintering robot is estimated to be 600 kg.

Experimental work has been carried out on microwave sintered soil on earth soil and soil brought from moon and the higher iron (Fe) percentage would lead to better results for roads on the moon. Moreover, microwave heating has the advantage over conventional heating in many aspects as energy savings, efficiency and productivity.

4.1.2 Lunar Prospector

The purpose of the 'Casanova' Lunar Prospecting Scout is to understand the distribution, concentration and extent of lunar volatiles in the polar regions of the Moon. This allows for feasibility studies, generating geological models, and estimating resources. Additionally the scout will provide valuable information for mine planning and construction. It will assist with the base architecture and launch pad construction design planning, assess accessibility to the mine sites, assist road building construction design and identify hazardous terrain. The following outlines the primary and secondary requirements for the "Casanova" Lunar Prospecting Scout.

Primary Requirement:

- Identify water-rich deposits (>4% H_2O by weight) for future mining missions

Secondary Requirements:

- Define composition, concentration and extent of the of the water-rich deposits
- Characterize terrain and environment (i.e. slope, identify geo-hazards, trafficability, temperature)
- Define accessibility / extractability of the resource (geo-mechanical properties of regolith, depth to resources)

The design of the scout is based primarily on the Lunar Resource Prospector (Picard et al 2014). **Figure 4.1.2a** shows the main subsystems aboard the prospector which will be utilized for resource prospecting.

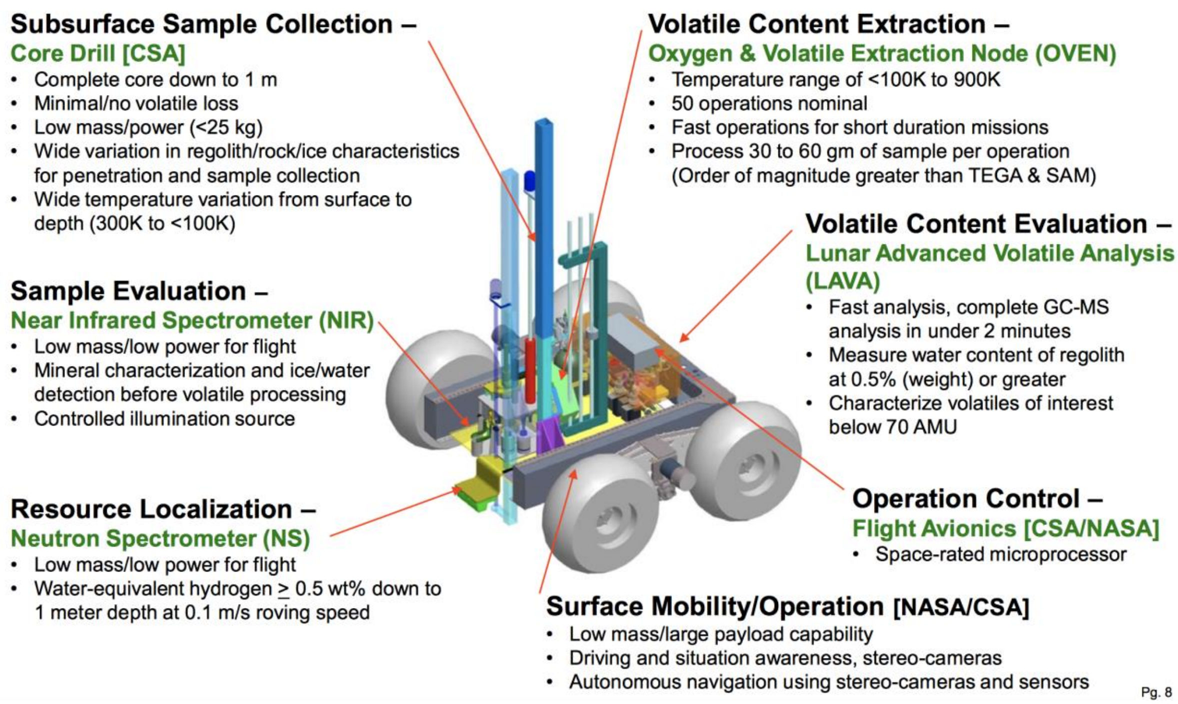


Figure 4.1.2a. *Principal subsystems of the Lunar Resource Prospector (Sanders, 2011).*

Modifications to the design of the lunar prospector include the addition of an RTG in order to allow operation in the permanently shadowed regions (PSRs). In the original design the Lunar Resource Prospector uses solar panels to cut down costs however it will not be able to operate for an extended period within the (PSRs) which is a mission requirement for the Scout. It is also hoped that the addition of an RTG will allow the rover to operate over a multi-year period. Additionally a LIDAR system will be added to the design in order to localize and guide the Scout while it explores the surface of the Moon.

A prototype has been developed for the Lunar Resource Prospector referred to as "RP15". A number of the subsystems have undergone environmental testing including the OVEN subsystem, honeybee drill, LAVA mass spectrometer subsystem and the rover OVEN and Drill prototype (Colaprete, 2016). In addition wheel grouser studies have been conducted including obstacle climbing @ 1/6g in the ARGOS gravity offload facility. The prospector wheels and steering have undergone TVAC testing. Further testing to include an RTG and LIDAR system is required in order to enable extended operations within the permanently shadowed regions.

4.1.3 Mining Robots

The lunar 'Spartan' miners will extract and process volatiles from the regolith. The key production requirements for the combined system of miners are outlined below:

Primary Requirements:

- Produce 19.5 tons of water by 2028 (LRS propellant transfer technical demonstration)
- Produce 33.4 tons of water by 2031 (technical demonstration of propellant transfer between LRS and orbiting propellant depot)

- Produce 703.8 tons of water by 2036 (1st two cargo missions to Mars)
- Produce 175.9 tons of water by 2038 (1st crewed mission to Mars)
-

Secondary Requirements:

- Continue refueling two cargo and one crewed mission to Mars every four years post

The graph below (**Figure 4.1.3a**) shows the proposed total water production rate (tons/year). This assumes an additional delivery of new miners every two years. The slight drops in production rate in the mid-2030s are a result of older mining rovers reaching their end of life before being replaced. The decrease in production past 2040 is a result of the cessation of new miners being delivered as operations begin to wind down for the site. The drop in production is also indicative of the best water deposits having been mined out early on in the operation. As lesser quality deposits are mined towards the end of a field's life, it is expected that production will also decrease as lower quality deposits start being mined.

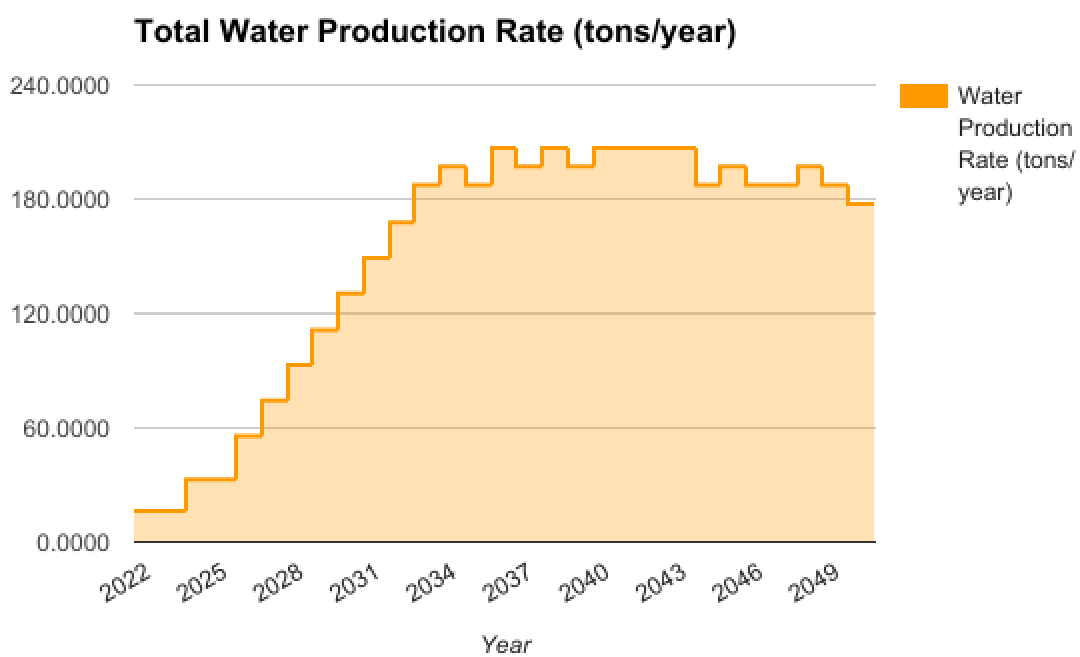


Figure 4.1.3a. *Total water production rate (tons / year).*

The proposed structure of the lunar miner is based Apollo lunar roving vehicle (TRL 9). This design decision was made on the basis of its load carrying capacity and that it is proven technology in a lunar environment. This design will allow the mining rover to store approximately 200 kg of ice before transporting it back to the storage facility.

For the coring and processing system the Honeybee Robotics Planetary Volatiles Extraction (PVEx) Corer (Zacny et al., 2015) will be used. Each mining rover will carry four core drills each capable of drilling 1 core per hour. Each mining rover will have a self-contained regolith processing unit aboard based on the Honeybee Robotics design. This will heat the icy regolith enough to extract the water and store the vapor in a cold trap. Although this will require greater power it reduces the need to transport large amounts of regolith waste rock back to a central processing unit. This has a significant impact on production rate and is necessary to meet the production rate targets.

As reported by Zacny et al. 2015 the drill and extraction subsystem have undergone laboratory lunar analog testing. All tests were conducted using JSC-1A lunar simulant mixed with water to achieve 6 wt% or 12 wt% saturation level. Further analysis on the extraction capabilities in the <100 K and 40 K environment is required as well operation of the equipment powered by microwave beaming technology as the current design uses RTG's. It will also be necessary for a prototype using the larger storage tanks and four-core system to be developed.

Figure 4.1.3b outlines the risks for the both the Lunar Prospector and Lunar Resource Miner.

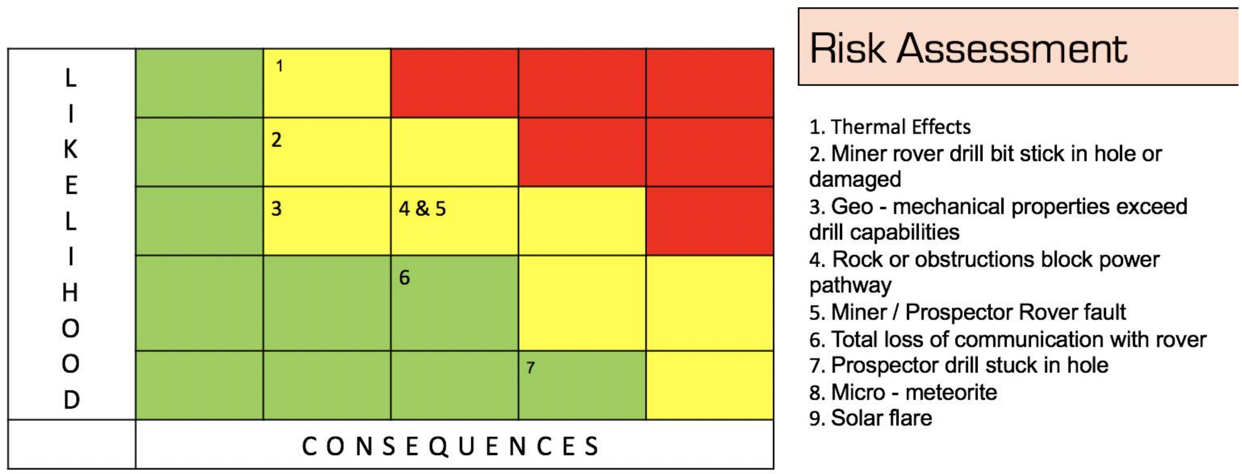


Figure 4.1.3b. *Risk assessment for the Lunar Prospector and Lunar Resource Miner.*

Mitigation Strategies:

1. Thermal effects: Rovers will be designed to operate in cold environments (40-100 K). During lunar nights when beamed power is unavailable, rovers will return to insulated and thermally controlled environment within their landers in order to keep warm.
2. Damage to drill bits and stuck in hole can be mitigated with mechanism to eject faulty drill bits and replace with spares.
3. Geo-mechanical issues can be mitigated by monitoring drill conditions and entering fault mode to await instruction from earth.
4. Obstruction and loss of power connection beamer can be mitigated with back up batteries and a return to base override.
5. Rover fault or drill error can be mitigated by entering fault mode and returning to base or awaiting instruction from earth.
6. Loss of communication can be mitigated by entering fault mode and returning to base.
7. Prospector stuck in hole cannot be mitigated as spare drill bits will not be available for the prospector. Prospectivity will rely solely on the neutron mass spectrometer.
8. Micro-meteorites are mitigated by impact shields.
9. Solar flare activity can be mitigated by returning back to base for protection.

4.1.5 Power

The Sun is the main power source. The Lunarport site selection was designed to receive sun for at least 20 out of the 28 Earth days of the lunar day. The main power consumption at the Lunarport is due to the mining activities in PSRs. Lower temperatures (~50 K, as seen in **Figure 3.1.1B**; NASA, 2010) at the PSR and the volatile extraction from the cold regolith requires around 71 kW-h per day of energy for continuous operation, per mining robot (Zacny et al., 2015). Since direct sunlight is only present at the border of the crater, energy was required to be transferred in some manner to the miner robots. Four main subsystems need to be powered for the fully operation of Lunarport: resource prospector robot, sintering robot, miner robot, and a small electrolysis plant on the ground to fuel the LRS.

To achieve a modular architecture, each subsystem was designed to have its own power source. The resource prospector robot requires continuous supply of 300 W for its nominal operation. The sintering robot demands 500 W of power for mobility and additional 500 W for each of its eight microwave generators to solidify the lunar regolith, totalizing 4500 W of power. The miner robots require 71 kW-h per day of continuous power for drilling and heating during lunar days.

Solar panels power most of the robots, including miners and sintering robot responsible for the construction of launch pads and infrastructure. The assumptions made for the solar panels were established by J. E. Freeh (2009), and are summarized below:

- The photovoltaic cell data is based on EmcoreTM BTJ cell. This cell area is 26.6 cm^2 with a specified efficiency of 28.5%.
- Solar array is assumed to only operate when in full Sun (Sun illumination fraction = 1) to simplify power requirements calculation, which should add a safety margin to the power budget.
- Dust accumulation on the solar array cause 1% loss in power generation per year.
- Cosmic background radiation decreases the performance of the solar panel by 1.2% loss per year.
- Other loss assumptions include the cell mismatch loss, array flatness loss, coverglass/interconnect/cell (CIC) loss, misalignment loss, and a voltage drop due to the blocking diode.
- Packing factor for the solar panels is considered to be 0.85.

Different solutions were considered for transferring the power from the top of the crater (illuminated by the Sun) to the bottom (PSR), including nuclear reactors, batteries, and direct cable connection. Since it is expected to have more than 50 miner robots during Lunarport's peak operation (consuming more than 3.5 MWh of energy per day when mining), the main power decisions were centered on powering the miner robots.

Nuclear reactors would be ideal in technical terms. They provide heat (useful to keep electronics warm and extract ice from the lunar regolith) and electricity required to power the drills, electronics, and mobility systems. This option was immediately discarded when some initial calculations pointed to the necessity of acquiring dozens of small Radioisotope Thermoelectric Generators (RTGs), which are both difficult to acquire and very expensive). Long cables were also discarded due the risk of installation, lack of mobility, and generally high long-term operational risk.

From the options analyzed, it was chosen to transfer the power from the solar panels to the rover using wireless power transmission.

The energy collected by the solar panels on the lander at the top of the crater is converted and transferred as 5.8 GHz microwaves to the bottom of the crater. According to Jaffe and McSpadden (2013), efficiencies of 17 to 19% can be achieved over the total sun energy available. The system is analogous to the illustrated at the figure below (dimensions not accurate), but with solar panels attached to the lander on the ground, instead of orbiting in space.

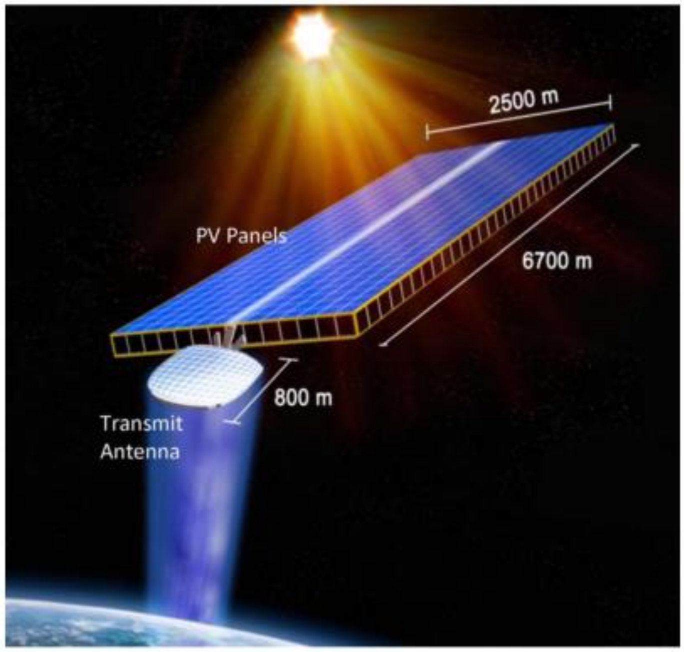


Figure 4.1.5a. *Illustration of microwave energy transmission.*

Due to infrastructure restrictions, the first rover to land - the Lunar Prospector Scout - is powered using one Radioisotope Thermoelectric Generator (RTG). Although nuclear generators lead to increase in cost and mission complexity, this power is ideal for a single low-power rover that will mostly prospect in a PSR. One reactor is enough to power its basic functions and science instruments while generating heat to keep electronics warm and extract volatiles from the analyzed regolith. It is assumed the reactors deteriorate by 10% over 14 years based on Zacny et al. (2015).

The use of solar panels and nuclear reactors in space are proven technologies and can be considered TRL 9 on the scale defined by NASA. The wireless power transmission, according to Jaffe and McSpadden (2013), most of the proposed concepts is well understood, and techniques for the safe retro-directive control of the microwave beam have been developed and demonstrated. However, the system still has yet to undergo testing in space. This would classify the microwave power transfer as TRL 5. The use of solar panels and nuclear does not present significant technology development risks. According to Sasaki et al. (2013), the microwave power transfer concept was proposed in 1968. This technology was already demonstrated in controlled environments but still need to be proven is space.

Operational risks include the deployment of the solar arrays and microwave transmitter from the lander. Transmission of energy can be decreased in case of block of line of sight, but rovers are equipped with emergency batteries to handle this risk (better discussed on “Operation Plan” section).

Table 4.2.5a. *Power subsystems risk analysis matrix.*

LIKELIHOOD					
			Microwave power transfer		
			Deployment of the solar arrays and microwave transmitter	Solar panels and nuclear power	
	CONSEQUENCES				

4.1.6 Supporting Infrastructure

Supporting infrastructure is required to successfully extract resources from the lunar surface. A set of high level requirements for the supporting infrastructure are defined as:

- Infrastructure shall be provided at the lunar surface station to allow for autonomous surface operations
- Infrastructure shall also allow for tele-operated control of surface operations when desired
- Surface infrastructure shall be constructed prior to beginning mining operations
- Surface infrastructure shall include a permanent reusable launch and landing pad
- Surface infrastructure shall allow mining rovers to access the ice storage facility as well as the cargo landers for thermal shelter during the lunar night

Launch Pad

Using the sintering robot and its attachments, a permanent reusable launch and landing pad for the LRS will be constructed in a selected and cleared site. **Figure 4.1.6a** shows the potential temporary landing site's geographical features and dimensions.

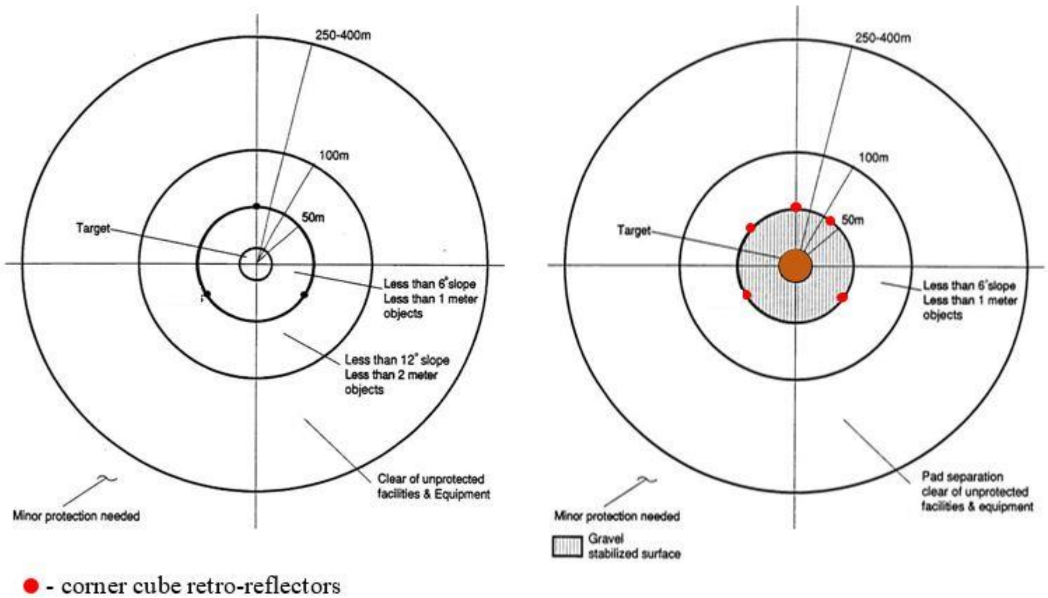


Figure 4.1.6a. *Landing Site Diagram.*

A permanent launch pad will be constructed using gravel deposition where gravel is produced using the microwave sintering robot and covering it with a sintered regolith layer on top as shown in **Figure 4.1.6b**. It will be constructed 500 m away from any lunar infrastructure to keep the infrastructure safe from debris thrown away by shuttle landings. Top sintered layer will improve the durability, dust protection, protection from debris ejection, stability and hence the overall performance.

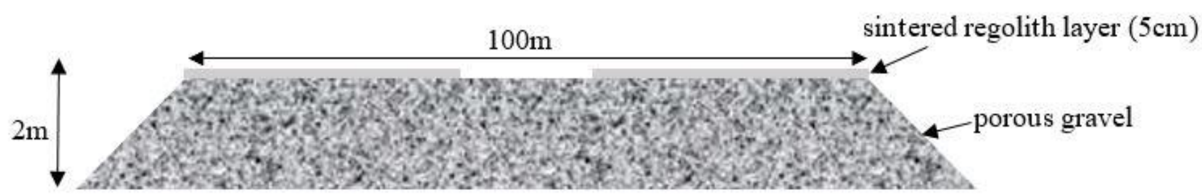


Figure 4.1.6b. *Launch Pad Construction.*

Road Networks

The unique properties of lunar regolith make for the extreme coupling of the soil to microwave radiation. It is possible to sinter lunar soil at 1,200–1,500°C in minutes in a normal kitchen-type 2.45-GHz microwave. Doing so would lead to a relatively fast construction of a road with good quality for rover movements. A 4-m wide road network is proposed for the transportation in the lunar base and beacons are located on the roads to navigate the rovers. It is found that use of sintered soil roads are well within the industry specifications, and hence durability, performance, stability and safety can be assured.

Covering berm

The covering berm is proposed to be constructed to keep the lunar infrastructure safe from the dust and debris which are ejected away by shuttle landings and launches (**Figure 4.1.6c)**. Moreover, the covering berm will act as a shield against radiation and asteroid impacts on lunar structures and rovers. A 5-m tall and 50-m long berm will be constructed using the excavation and filling attachments of the sintering robot. Construction process will allow propellant transfer pipes to be embedded in the berm.

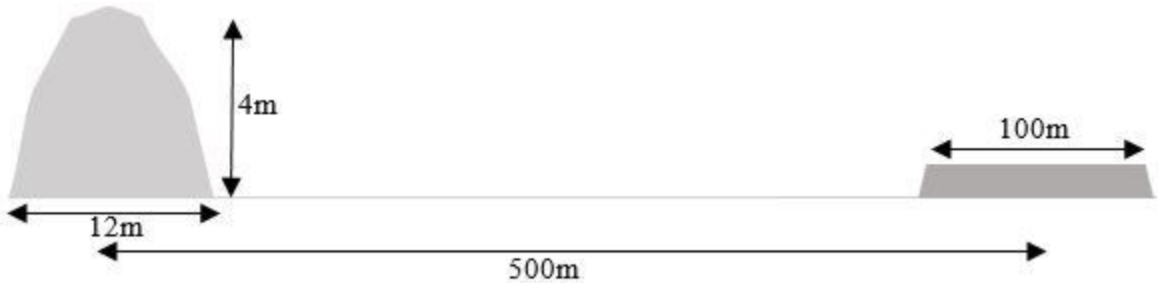


Figure 4.1.6c. *Covering berm Diagram.*

Deployable Structures

Once the landers land on the temporary landing pads on the moon, they will be used as solar energy harvesters and communication centers. Hence it is important to keep these structures safe from radiation and asteroid impacts. As a remedy, landers will initially be covered by a carbon fiber deployable structure and its openings for rover movements will be controlled by Z-type origami components. Subsequently it will be covered by a regolith layer for protection against asteroid impacts, temperature fatigue and radiation.

Table 4.1.6a. *Supporting infrastructure risk analysis matrix.*

LIKELIHOOD					
		Overestimate Vehicle Lifetime			
		Cannot develop technology in time	Berm slope failure		
			Road failure	docking failure	Sintering robot failure
	CONSEQUENCES				

4.1.8 Environmental Protection and Control

It is very important to take into consideration the environmental risks of an autonomous lunar port as it is highly prone to uncertain and extreme environmental conditions at the surface of the lunar south pole. There are three main impacts, namely the regolith, radiation, and asteroid impacts for which protection designs are necessary to be implemented. Furthermore, it is essential to control the extreme thermal environment so that all operations will function well as expected.

Regolith Protection

Influence of regolith can be identified in two main aspects, abrasive lunar dust and high speed ejection of regolith while landing. Both of these impacts on permanent structures can be mitigated by having first, a deployable structure to cover from dust, and second, a regolith layer covering the structure which can absorb external regolith impacts. The lower gravity on the Moon would increase the bearing capacities of structural components approximately by 6 times and hence thin shell structural components will perform better against the external loading from outer regolith layer. The structure will allow access to rovers via openings controlled by z-fold origami structures.

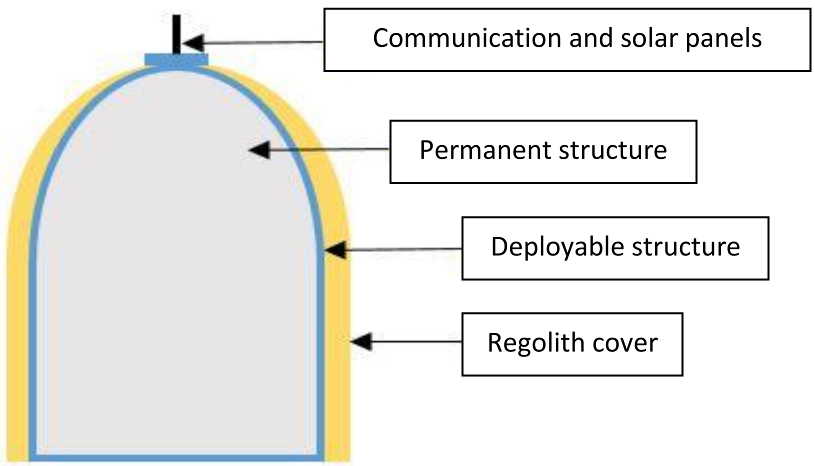


Figure 4.1.8a. *Regolith protection structure schematic.*

Dust affecting the rovers, solar panels, and communication systems would be protected by electromagnetic vibration systems attached to each of these components. Furthermore, the proposed berm will reduce the extent of regolith ejected during launch and landing at the lunar launch/landing pad, thus reducing the thickness of the required regolith layer on permanent structures.

Radiation Protection

Radiation onto the lunar surface takes two main forms, electromagnetic and ionizing radiation. Ionizing radiation can penetrate up to a few centimeters in depth with severe magnitude. Ionizing radiation takes three forms, solar wind, solar cosmic rays, and galactic cosmic radiation, all of which can be avoided by the use of regolith layers up to several meters of thickness. Previous research work has estimated that a roughly 2.5m thick regolith layer is sufficient to limit the annual radiation dosage of five rem into the structure, which is the limit for radiation workers (Ruess, 2006). A structure designed for non-human operation would require a significantly thinner regolith layer, given the significantly lower radiation requirements on non-human system components.

Asteroid Protection

Asteroid and meteoroids are naturally occurring solid bodies traveling through space at very high speeds. Most likely, a layer of compacted regolith will be placed atop the structure for protection against all of those hazards. It provides shielding against most micrometeoroid impacts because the relatively dense and heavy regolith absorbs the kinetic energy. Furthermore, a relatively tall, covering berm will also protect the surrounding structures by shielding against asteroids and meteoroids.

Thermal Control

Temperatures at the south pole at the Moon are among the lowest temperatures ever recorded in the Solar System due to areas being permanently shadowed from the Sun. Data from the 2009 Lunar Reconnaissance Orbiter indicate that south-pole temperatures range as low as 25 K (-250 °C) in permanently shadowed regions to as high as 300 K (27 °C) in areas receiving sunlight greater than 70% of the lunar day. As such, constructing, operating, and maintaining a mining base on the south pole represents a great and unique thermal engineering challenge.

Most systems are strategically placed out of the permanently shadowed regions to avoid the extreme cold. These systems will still require active heating, especially during the lunar night (approximately seven Earth days at the chosen site), so power has been budgeted to keep these systems warm. However, some systems will still have to travel into permanently shadowed regions, including the prospecting robot and mining robots. These robots will have a cold-biased design and will include active heating elements to keep crucial components such as electronics and actuators above minimum functioning temperatures. The prospecting robot will receive power from a nuclear power source which is sufficient to keep itself warm in the cold. The mining robots receive beamed power generated from solar array panels, so they will require additional thermal control strategy during times when solar power is not available. Just before the sun sets, the mining robots will drive back to the site, drop off their mined ice loads, and drive into a thermally controlled environment. This thermally controlled environment will be the lunar lander that each respective mining robot was originally delivered to the surface in. The mining robots will spend the lunar night in this insulated and actively heated environment in a low power state until the sun rises approximately seven Earth days later. Once the sun has risen, the mining robots will drive back to the cold regions and restart their mining operations.

4.2 Space Systems

4.2.1 Lunar Resupply Shuttle (LRS)

The Lunar Resupply Shuttle (LRS) is an unmanned spacecraft which flies between a base on the lunar surface and a propellant depot at L1 to transport resources to the depot. The vehicle is reusable, with an expected life of 10-15 base-depot cycles. We choose to produce many small vehicles: this allows launch on cheaper, frequently-launching rockets, and will reduce production costs through learning curve effects. The components of the LRS are technically mature: all are currently under active development with a TRL of at least 4, and many have spaceflight heritage. The LRS design deposits 0.13 tons of propellant in the depot for every ton of water mined on the Moon.

High Level Requirements

1. The LRS shall transport propellant resources from the surface of the Moon to a propellant depot in cis-lunar space.
2. The LRS shall finish development within 7 years, and undergo a test flight at the Moon in 2026.
3. The design, production and operation of the LRS shall minimize the cost of delivering a unit mass of propellant to the Depot.
4. The design and operation of the LRS shall minimize the risk a failure of the LRS would pose to the overall Lunarport.

Concept of Operations

The role of the LRS is to fly cycles between the base and the depot in order to deliver water to the depot. At the depot, the water is split into H_2/O_2 propellant. Some of the propellant is taken by the LRS for its return trip to the Moon, while the rest is added to the depot's stockpile. The key performance metric for LRS operations is the mass leverage:

Mass leverage = mass of propellant deposited at depot / mass of water mined

The mass leverage is driven by the amount of propellant the LRS consumes in transporting resources to the depot. The mass leverage of the current architecture is 0.13.

To reduce costs and launch requirements, each LRS is re-used for several cycles. To meet the required delivery rate and system uptime, there will be one active LRS and two in-space spares deployed at any time (once Lunarport is operating at full capacity). When first deployed, the LRS lands at the base on the moon surface. The LRS loads 15 tons of water and 20 tons of propellant into its tanks. The base needs to mine 42 tons of water to generate this much water and propellant. The LRS then burns these 20 tons of propellant to launch to the depot. At the base, water is pumped into the LRS bladder as a liquid, but it will likely freeze during the three-day flight to the depot.

Once docked to the depot, the LRS uses electrical power from the depot to melt the water ice in its bladder, and pump liquid water into the depot's holding tank. This process takes approximately 1.5 days. The 15 tons of water from the LRS is electrolyzed by the depot to produce 11 tons of propellant over several weeks. The LRS waits at the depot until it needs to fly another cycle. Just before departure, the LRS withdraws 5.5 tons of propellant from the Depot. The LRS burns this propellant to make fly to the Base. After many LRS cycles, a large amount of propellant is deposited at the depot. This propellant is then transferred to a customer vehicle.

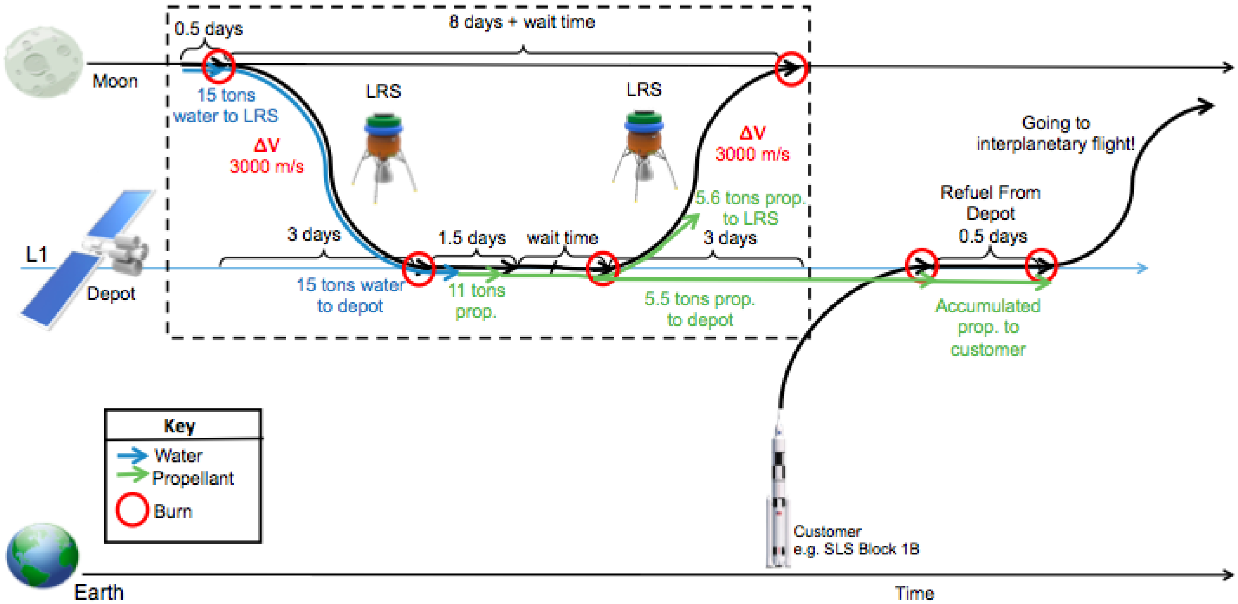


Figure 4.2.1a. *The LRS flies cycles between the Depot and the Base to transport water. After the cycle in the dashed box is repeated several times, enough propellant is deposited at the Depot to support a customer mission.*

Table 4.2.1a. *Delta-V budget for the LRS carrying water. Assumes 97% expulsion efficiency for water and propellant tanks. Delta-V figures include 500 m/s margin for maneuvering, attitude control, and boil-off of propellant. The mass leverage is 0.13. A net mass of 5.5 tons of propellant is deposited in the Depot by each LRS cycle.*

Event	Delta-V (m/s)	I_{sp} (s)	Total mass before event (Mg)	Total mass after event (Mg)	Dry mass (Mg)	Change water mass during event (Mg)	Change propellant mass during event (Mg)	Water payload mass after event (Mg)	Propellant mass after event (Mg)	Burn time (s)
Depot to Base	3000	460	11.5	5.9	5	0	-5.6	0.45	0.45	229
On Base			5.9	40.7	5	15	19.8	15.45	20.2	
Base to Depot	3000	460	40.7	20.9	5	0	-19.8	15.45	0.45	811
At depot			20.9	11.5	5	-15.0	5.6	0.46	6.0	

Concept Trades

The propellant resource can be carried to the depot as liquid cryogens or as water. The comparison is two-fold: both the conversion efficiency of water to propellant and the volume of the propellant in either state must be considered. A low conversion efficiency of water to propellant through electrolysis would favor conversion of propellant on the lunar surface in order to avoid the efficiency loss after launch. On the other hand, liquid cryogens have a lower density than water and therefore require larger tanks to transport them. Assuming an electrolysis efficiency of 0.95 (coupled with stoichiometric losses of 0.78), it was ultimately determined that the conversion efficiency was high enough for the smaller tank volumes to be worth the loss in efficiency of water launched to orbit.

An alternative delta-V budget is shown in **Table 4.2.1a** for an LRS concept which carries cryogens, not water. This LRS is sized to fit into the same launch fairing as the water-carrying LRS. However, it has a lower total fluids capacity, because cryogens must be stored in rigid tanks, not bladders.

The cryogen-carrying LRS has a 30% higher mass leverage but a 30% lower delivered propellant mass per cycle than the water-carrying LRS. A lower delivered propellant mass per cycle means that more cycles will need to be flown, and more production and launches will be needed to maintain the LRS fleet. Therefore, the water-carrying LRS is chosen, as this concept reduces maintenance requirements.

If later efforts determine that the water/cryogen trade was made incorrectly, the LRS can be easily switched to a cryogen-carrying concept by simply removing the water bladder.

Table 4.2.1b *Delta-V budget for the LRS carrying cryogens. Assumes 97% expulsion efficiency for water and propellant tanks. Delta-V figures include 500 m/s margin for maneuvering, attitude control, and boil-off of propellant. The mass leverage is 0.17. A net mass of 4.3 tons of propellant is deposited in the Depot by each LRS cycle.*

Event	Delta-V (m/s)	I_{sp} (s)	Mass before event (Mg)	Mass after event (Mg)	Dry mass (Mg)	Change water mass during event (Mg)	Change propellant mass during event (Mg)	Water payload mass after event (Mg)	Propellant mass after event (Mg)
depot to surface	3000	460	11.5	5.9	5	0	-5.6	0	0.9
On surface			5.9	25.1	5	0	19.2	0	20.1
Surface to depot	3000	460	30.7	15.8	5	0	-14.9	0	10.8
At depot			15.8	11.5	5	0.0	-4.3	0.00	6.5

Descent and Landing

The descent trajectory is designed to minimize the risk to base infrastructure in the event of an LRS failure. It is important to minimize this risk because LRS spacecraft will be aggressively re-used, so failures near the end-of-life are likely. The LRS uses precision landing algorithms while performing a landing sequence that prevents a collision between the LRS and base should the engine fail. The landing sequence is shown in **Figure 4.2.1b**. The initial braking burn places the LRS on a landing trajectory that targets points outside of the zone of the lunar base. This ensures that engine failure at any point during the initial burn does not put the LRS on a trajectory for collision with the base. Next, the LRS uses radar to acquire the location of reflective fiducials on the landing pad next to the base. Terrain-relative navigation from a LiDAR sensor provides a redundant pose estimate. Once establishing a base-relative pose estimate, the LRS converges on a guidance solution and performs a divert burn to redirect its trajectory to the landing pad. A final landing burn is performed followed by a controlled vertical descent onto the pad.

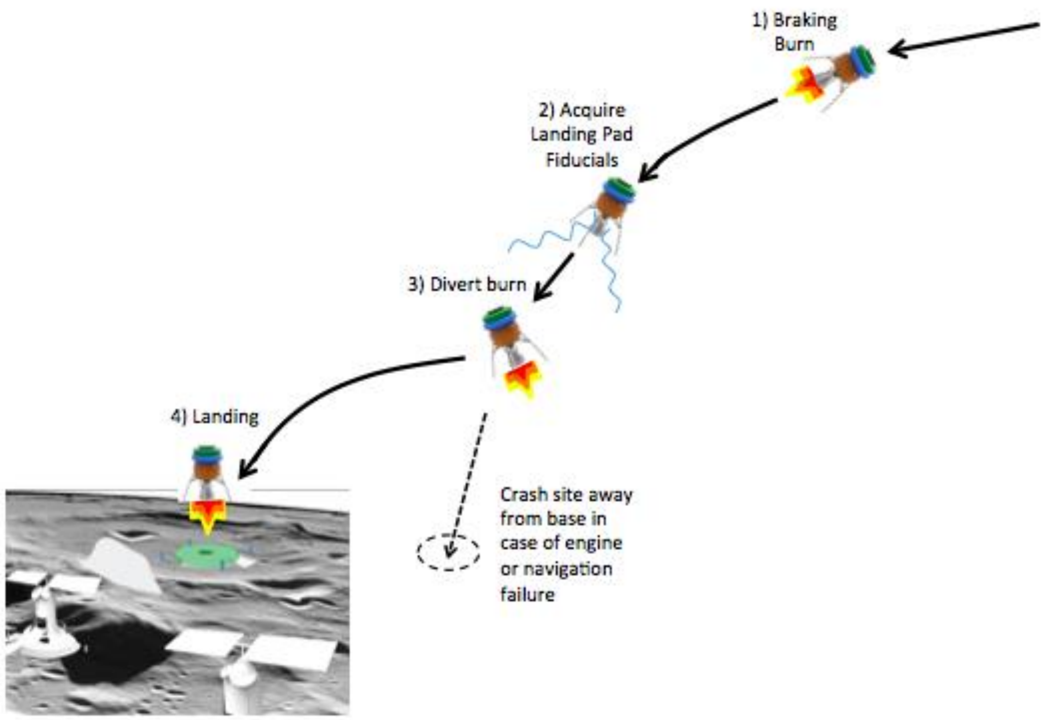


Figure 4.2.1b. *The descent trajectory of the LRS is designed to reduce the risk to Base infrastructure in the event of an LRS failure.*

Design

Figure 4.2.1c shows a mockup of the LRS. A 17-ton oxygen tank rests on top of a three-ton hydrogen tank with a 15-ton inflatable bladder for water storage wrapped around their junction. The main engine is an RL10, which is the current engine on the Atlas Centaur Upper Stage. A docking port to fit the propellant depot is on top of the oxygen tank and contains interfaces for propellant and water exchange. A communications antenna is located on the upper portion of the hydrogen tank and landing radars are located around the lower portion (providing full coverage). The entire vehicle rests on four landing legs. The overall height of the LRS is 15 m with the legs unfolded.

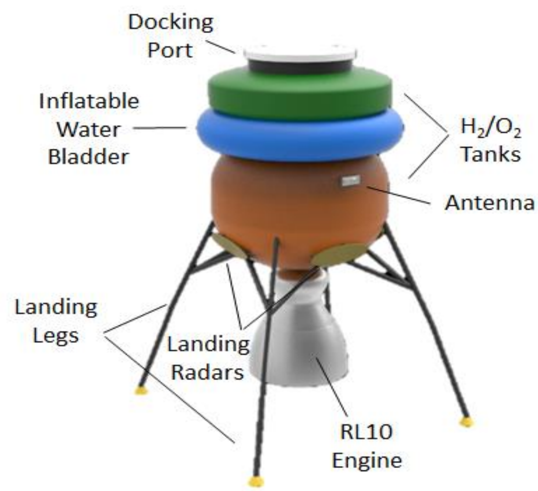


Figure 4.2.1c. *Configuration of the LRS with key parts labeled.*

Figure 4.2.1d shows the stowed configuration of the LRS for launch. The landing legs fold up and the RL10 nozzle extension telescopes for stowing. During launch, the water bladder is deflated to fit within the launch vehicle fairing. The docking adapter doubles as a hardpoint to attach the LRS to the launch vehicle.

Figure 4.2.1d. *Stowed configuration of the LRS.*

Mass and Volume Constraints

The LRS will be launched directly to the Moon from Earth by a commercial heavy-lift launch vehicle such as SpaceX's Falcon Heavy or Blue Origin's New Glenn. Using a commercial launch vehicle reduces costs and improves available launch frequency when compared to the SLS (by about an order of magnitude on both counts). In this proposal, the Falcon Heavy is baselined because its fairing dimensions are currently available (SpaceX, 2015). The launch vehicle will place the LRS into a Trans-Lunar Insertion (TLI) orbit at which point the LRS will separate from the launch vehicle. Upon reaching the Moon, the LRS will brake and land under its own power, therefore requiring that the LRS will need to carry propellant for braking and landing during launch - this proposal assumes that braking and landing requires 4.5 km/s delta-V. The need to carry this propellant constrains the dry mass of the LRS to five tons. Assuming that the launch vehicle can place 15 tons onto TLI, both the dry mass of the LRS and its propellant for braking and landing cannot exceed this number. Given the 460-s I_{sp} of the RL10 engine and a plausible expulsion efficiency (97%), the dry mass of the LRS can be at most five tons.

Table 4.2.1e. *Mass of propellant capacity, water capacity, and subsystems.*

Category	Subsystem	Mass (Mg)
Propellant capacity		**20.5**
Water capacity		**15.5**
Dry mass	**Total**	**5**
	Propulsion	0.3
	Structures and tanks	3.1
	Power	0.2
	Docking and propellant transfer	0.3
	Miscellaneous	0.6
	Margin	0.5
Overall dry mass fraction		**13%**

LRS Lifetime, Fleet Size, and Fleet Maintenance

It is assumed that the early LRS versions will have an operational lifetime of ten base-depot cycles (A.C.Charania, personal correspondence). It is further assumed that lessons learned from the operation of the vehicle will allow engineers to improve its lifetime to 15 cycles after a decade of operation. The LRS takes a minimum of eight days to complete a base-depot cycle. If the LRS fleet is operating at its busiest possible rate (eight days/cycle), and the LRS has a lifetime of ten cycles, each LRS will last for 80 days. Only one active LRS is required to supply the depot with propellant resources. At full capacity, Lunarport will provide 600 tons of propellant to a Mars mission every four years. The depot will need to be stocked with propellant at a rate of 0.4 Mg/day. A LRS can deliver (net) 5.5 Mg of propellant to the depot per eight-day cycle, or 0.68 Mg/day. Therefore, a single active LRS is sufficient. To improve system uptime, two spare LRS spacecraft will be kept and parked at the Depot.

At Lunarport's full capacity, the LRS will need to be replaced at a rate of two per year. This assumes a propellant demand of (600 tons per four years), an LRS lifetime of 15 cycles, and a delivery capacity of 5.5 tons of propellant per LRS cycle. LRS production will need to be maintained at a rate of about two per year. This is in-line with the typical production rate of high-production-volume space systems (see

Table 4.2.1f). A rate of two units per year is also compatible with the current production rate of the RL10 engine.

Table 4.2.1f. *Production rates of recent, high-production-volume space systems. Data for spacecraft busses from Krebs, 2017. Data from RL10 from Aerojet Rocketdyne (2017).*

System	Number produced	Years of production	Mean units / year
SSL 1300 bus	100	1984-2015	3.2
HS-376 bus	58	1978-2003	2.3
Lockheed Martin A2100 bus	37	1996-2013	2.2
SpaceX Dragon spacecraft	11	2012-2016	2.8
Aerojet Rocketdyne RL10 engine	500	1959-2003	11

Propulsion

The Aerojet-Rocketdyne RL10 was selected as the main engine for the LRS because of its low mass, high specific impulse, deep throttling and spaceflight heritage. THe RL10 also can use a telescoping nozzle extension, which is convenient for fairing packaging.

The RL10 maximum thrust (110 kN vacuum) gives a lift-off (full vehicle) thrust/weight (T/W) ratio of 1.7 under lunar gravity, sufficient for liftoff with low gravity losses. The minimum thrust (8 kN; Aerojet Rocketdyne, 2017) gives a landing (dry vehicle) T/W ratio of 0.9. A minimum T/W ratio below one enables a wider variety of landing trajectories, and avoids a difficult hover-slam maneuver.

The RL10 will require slight modifications to meet the LRS reusability target of ten cycles. Ten cycles will put three hours of runtime on the engine, but the current RL10 is only rated to one hour (Aerojet Rocketdyne, 2017).

The LRS reaction control system (RCS) thrusters will burn gaseous H_2/O_2. The gases will be partially sourced from tank boil-off, which makes boil-off losses less detrimental to the overall system performance.

Power

The LRS will generate electrical power from a H_2/O_2 fuel cell. It is expected that the LRS may need to operate in dark polar conditions or fly trajectories over the night side of the Moon, so fuel cells provide an advantage over solar panels. Fuel cells are preferred over batteries because of their superior low temperature performance.

Guidance, Navigation, and Control

The LRS will determine its attitude using star trackers and a tactical-grade IMU (e.g. Northrop Grumman’s LN200; Northrop Grumman, 2013). For pose determination during landing, the LRS will use two redundant sensor technologies: terrain-mapping LIDAR (e.g. JPL’s Autonomous Landing Hazard Avoidance Technology, ALHAT; Harbaugh, 2016) and radar fiducials. A phased-array radar on the LRS will detect passive retro-reflective fiducials placed around the base landing pad.

Tanks and structures

The main propellant tanks of the LRS will be based on the thin walled "steel balloon" design used on Centaur. However, the LRS needs longer endurance than Centaur (three days vs. half a day), so the tank insulation will need to be improved to reduce boil-off. To this end, the H_2 and O_2 tanks are separated (no common bulkhead). This incurs a slight mass penalty, but eases the design and installation of tank insulation. The tanks may also incorporate zero-boil-off technology which ULA is developing for their Advanced Cryogenic Evolved Stage (ACES; Kutter, 2016).

The LRS stores water in a flexible, toroidal bladder mounted around the gap between the hydrogen and oxygen tanks. A flexible bladder is required because there is not room for a rigid tank of sufficient volume within the launch vehicle fairing. The water in the bladder may freeze, so bladder will also need to contain electrical heating elements to melt the ice so it can be pumped out of the bladder. TransAstra is currently in the early stages of developing a similar bladder. The LRS lands on four legs. After launch, the legs unfold and permanently lock into their extended configuration.

Technology Development

The technology development schedule of the LRS must support a test flight in cis-lunar space by 2026, and a production rate of two per year by 2030. The technology development process will proceed in the following phases:

2017-2022: Mature mid-TRL technologies
2021-2023: Design LRS and issue contracts
2023-2025: Integrate test LRSs
2026: Launch two LRS spacecraft for cis-lunar test flight
2026-2028: Fix lessons learned in test flight
2028 onwards: Streamline production process

*Note that some of the phases can overlap.

To reduce the costs of Lunarport, it will be important to learn and improve the LRS production process during the final phase of development. Many components of the LRS have spaceflight heritage, and all are under active development with a TRL of at least 4. The status of major components is listed in **Table 4.2.1d** below.

Table 4.2.1g. *TRL of LRS Components.*

Component	Off-the-shelf solution or analogous systems	Modifications from analogous solution	TRL
Main engine, ~100 kN, liq. H_2/O_2	RL10	Increase lifetime from 1 to 3 hours	9
H_2/O_2 tanks	e.g. Centaur	Change dimensions, reduce boil-off	9
Landing legs	e.g. Apollo LM	Design to support loads on impact for our particular vehicle	9
Power	H_2/O_2 fuel cell	Determine configuration on vehicle	9
Communications	S-band radio	Size Antenna	9
RCS thrusters, ~100 N, gas. H_2/O_2	ULA ACES thrusters Various prototypes for Space Station Freedom	Reconfigure for LRS	6
GNC for precision landing	e.g. JPL's ALHAT, Mighty Eagle Lunar Lander	Refine algorithms for particular mission	6
Flexible bladder for water storage	e.g. TransAstra's APIS	Needs further lab testing Modify shape to fit LRS	4

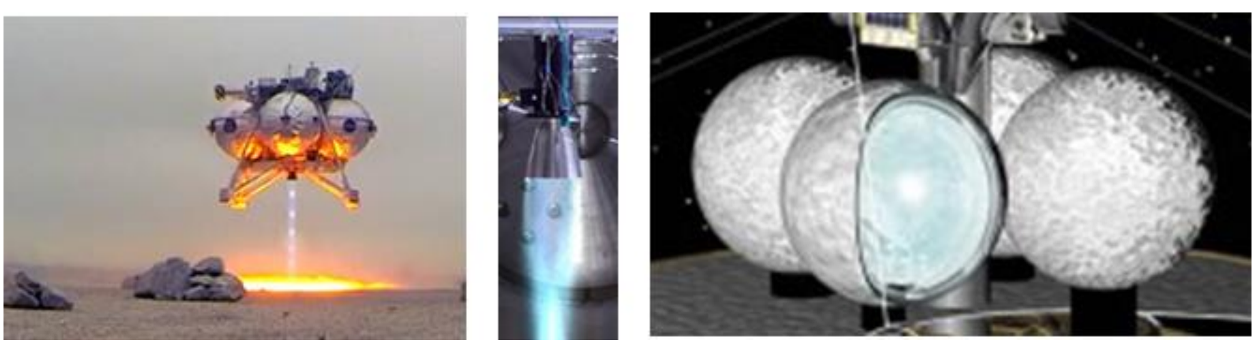


Figure 4.2.1e. *Technologies needed for the LRS are already under active development. JPL's ALHAT sensor demonstrates using terrain-relative navigation for precision landing (shown on left, in a test flight on the Morpheus lander). ULA is test-firing H_2/O_2 reaction control thrusters for their Advanced Cryogenic Evolved Stage (middle). TransAstra is developing flexible water bladders under a SBIR contract (right).*

Risk Plan

The LRS risk assessment is presented in **Table 4.2.1f**. The highest-consequence risk is an LRS crash which destroys base equipment; the likelihood of this risk is reduced through a clever choice of landing trajectory. The most probable risk is that the estimate vehicle lifetime estimate cannot be achieved; however the consequences of this risk are only a marginal increase in the LRS production and launch rate required to maintain the fleet.

Table 4.2.1f. *LRS risk analysis matrix.*

L I K E L I H O O D					
		Overestimate Vehicle Lifetime			
		Cannot develop technology in time	Refueling Failure		
			LRS Crashes on Landing	LRS Crashes into Base	
	CONSEQUENCES				

4.2.2 Orbital Fuel Depot

High Level Requirements

- The refueling depot shall receive water (H_2O) from the LRS and convert it to liquid oxygen (LO_2) and liquid hydrogen (LH_2) through electrolysis.
- The refueling depot shall store water and propellants at necessary conditions (warm for H_2O and cryogenic for LO_2 and LH_2).
- The refueling depot will be designed in such a way to minimize costs.
- The refueling depot will be designed in such a way to minimize risks.
- The development time of the initial module of the refueling depot shall be five years or less, with two years of development time for each sequential module to follow.
- The refueling depot shall provide a docking capability for use by commonly used spacecraft.
- The refueling depot shall have a modular design.

The metric of performance for the depot design is the efficiency of converting the received water into LO_2 and LH_2 and the amount of LO_2 and LH_2 that can be stored.

Transfer of Propellant

An important consideration is also the propellant transfer rates that are achieved by the refueling depot to the customer. It is assumed that the total 600 tons of propellant can be transferred to the customer in a 48-hour period. This estimate is conservative, and assumes that similar flow rates can be achieved as were achieved in the refueling of the space shuttle external tanks. The space shuttle was able to have its external tank (containing LO2 and LH2 propellants with a mass of approximately 735 tons) completely refuelled in the 24 hours leading up to launch. A distinction between the two is that the refueling depot will transfer fuel in space whereas the space shuttle was fueled under Earth's gravity. This is accounted for by allowing an additional 24-hour period for fuel transfer. There is also a consideration for the fuel transfer between the LRS and the refueling depot. Compared to the mass that is required to be transferred to the customer, the amount of mass transferred between the LRS and the refueling depot is on the order of a couple of hours, and thus not a significant hindrance to the LRS schedule.

Concept of Operations

The overall role of the on-orbit depot is used to convert and store water ice that is delivered from the LRS into useful propellants for future space missions. The propellants stored during the envisioning of this Lunarport are solely LH2 and LO2. As the water ice is transported from the LRS to the depot, the first step is to heat up the ice to a point to convert the water into liquid form using an electrical interface and transferring it into a H2O storage tank. The water is stored within its own tank, and gets transferred into a device using electrolysis to break down the water into its parts of hydrogen and oxygen. Hydrogen and oxygen are then transferred into their own respective tanks with intermediate liquefiers in order to store the spacecraft-usable propellants of LH2 and LO2. The depot stores the propellants until such a time that they can be transferred into a customer vehicle that docks with the depot. The depot is a modular design that starts out as a small standalone operations base that includes the major components of a water tank, electrolysis propellant converter, LH2 and LO2 storage tanks, a power system with solar panels, and an attitude control system for station keeping. The concept of operations of the mission is depicted in **Figure 4.2.2a**.

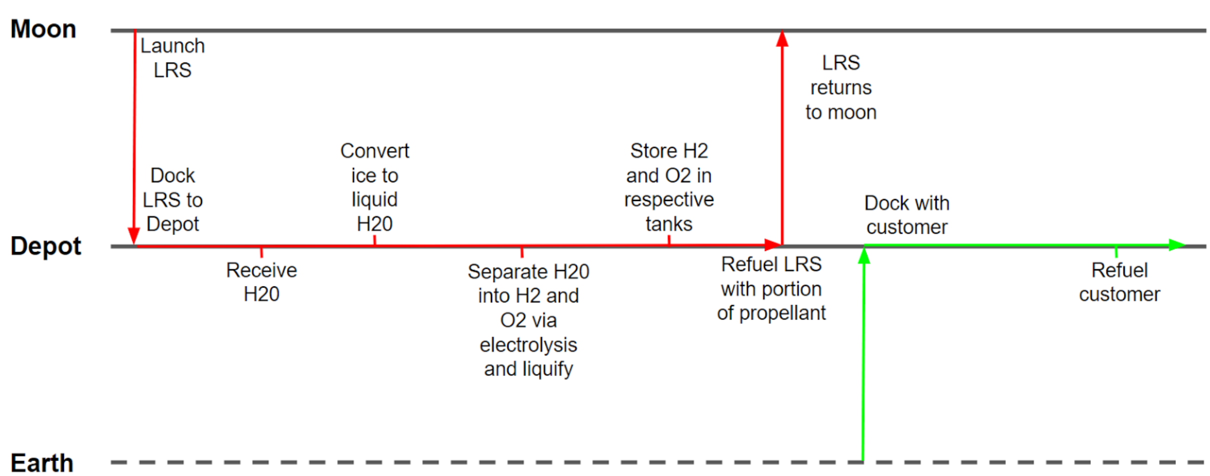


Figure 4.2.2a. *Concept of Operations for the refueling depot.*

System Modules

This section describes the individual system modules that make up the refueling depot system.

Module 1: Demonstrator Module

The demonstrator module of the refueling system is the first module to be launched. It will be launched on a SLS EUS. It serves the purpose of providing an initial proof of concept whilst also acting as the first step towards a fully functioning refueling depot. The module comprises of a main strut structure; a water tank; a LO2 tank; a LH2 tank; an electrolysis system for converting the H20 into H2 and O2; a liquefier for converting gaseous O2 and H2 to LO2 and LH2; four Moog ISP DST-11H thrusters for attitude control and station keeping; and two solar panels including a power system. It is assumed that all tanks have a wall thickness of 0.008 m and that the LO2 and LH2 tanks weigh 25 % less than their aluminium counterparts (Knapschaefer, 2016). The demonstrator module is shown below in **Figure 4.2.2b** and the technical specifications are summarized in **Table 4.2.2a**.

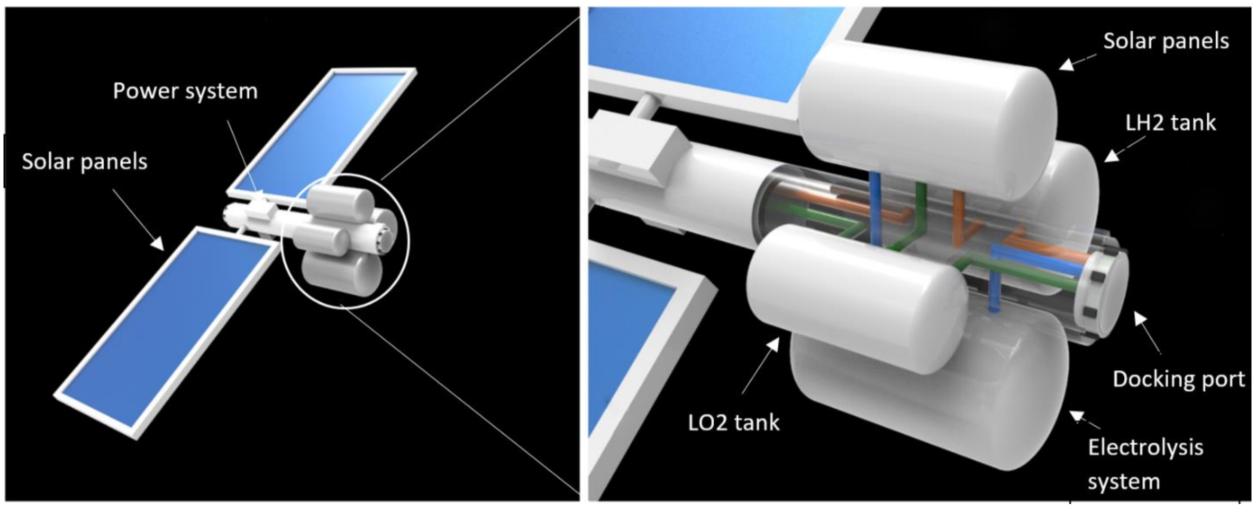


Figure 4.2.2b *Module 1 overall and zoomed-in schematic.*

Table 4.2.2a. *Technical specifications for Module 1 of refueling depot.*

Specification	Value
H20 tank dry mass (kg)	746
H20 liquid mass (kg)	15450
H2O tank length (m)	3.147
H2O tank diameter (m)	2.5
H2 tank dry mass (kg)	705
H2 tank length (m)	3.12
H2 tank diameter (m)	3
H2 liquid mass (kg)	1586

O2 tank dry mass (kg)	370
O2 liquid mass (kg)	9519
O2 tank length (m)	2.63
O2 tank diameter (m)	2
ACS thruster mass (kg) (4)	3.08
Propellant mass for ACS thrusters (kg)	Negligible compared with propellants onboard
Avionics and power mass (kg) (assumed)	300
Strutt mass (kg) (assumed)	200
Docker mass (kg) (assumed)	640
Electrolysis system mass (kg)	4800 (assuming each unit weighs 200 kg - 24 units considered)
Solar array mass (kg)	3801
Total mass without propellant (kg)	**11,561**
Total mass with propellant (kg)	**22,666**
Solar panel area (for two) (m^2)	280.3
Solar panel efficiency (kW/m^2)	0.267
Solar panel surface density (kg/m^2)	2.315
Solar panel battery density (kg/m^2)	11.244
Solar panel power output (kW)	74.84
Electrolysis length (1 unit) (m)	5.09
Electrolysis diameter (1 unit) (m)	3
Number of electrolysis units	24
Power required for electrolysis (per month) (kW)	74.84

Module 2: Propellant Tank Extension Module

The propellant tank extension module is a self-contained module that can be launched on the SLS EUS to L1 to dock with the existing demonstrator module. It serves the purpose of providing larger storage facilities for propellant so that a customer mission to Mars can be refueled up to 600 tons. In the entire refueling depot system there are two identical propellant tank extension modules used. They are launched as the second and fifth modules in the depot system. As they are identical, this module will only be described here. The module consists of a large O2 tank and a large H2 tank. The tanks are equipped with the appropriate thermal control systems to ensure that the H2O is sufficiently warm such that it remains in liquid phase and that the LO2 and the LH2 are sufficiently cooled such they remain in liquid phase (discussed in following section). Both tanks are connected to a central strut. It is assumed that both tanks have a wall thickness of 0.008 m and that the propellant tanks weigh 25% less than their aluminium counterparts (Knapschafer, 2016). The propellant tank extension module is shown in **Figure 4.2.2c** and the specifications summarized in **Table 4.2.2b**.

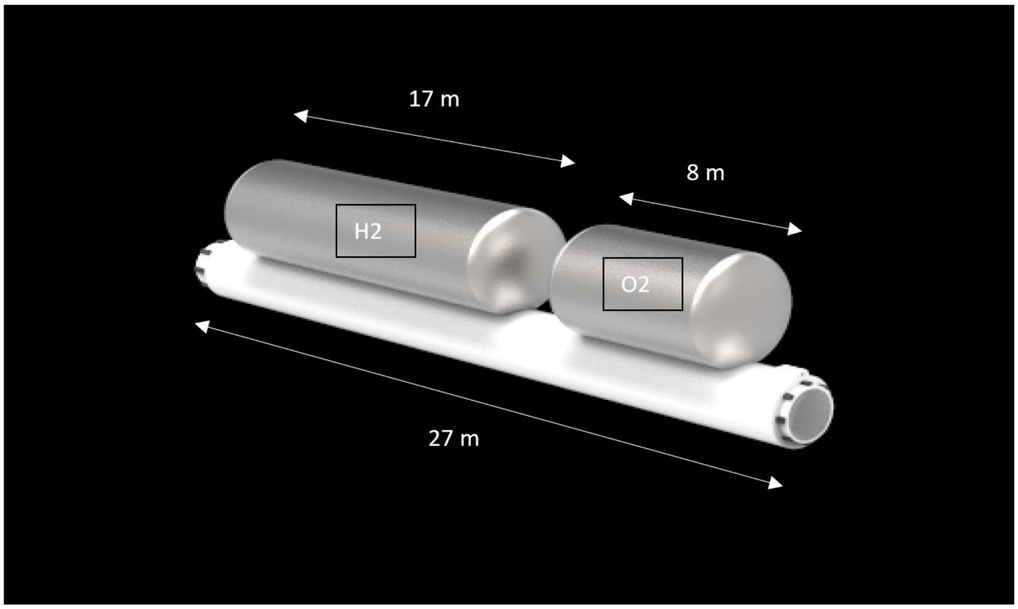


Figure 4.2.2c. *Module 2 (oxygen and hydrogen tanks) of refueling depot.*

Table 4.2.2b. *Technical specifications for Module 2 of refueling depot.*

Specification	Value
H2 tank dry mass (kg)	8550
H2 liquid mass (kg)	50,000
H2 tank length (m)	17
H2 tank diameter (m)	8
O2 tank dry mass (kg)	4885.8
O2 liquid mass (kg)	350000
O2 tank length (m)	8
O2 tank diameter (m)	8
Tank thickness (m)	0.008

Docker mass (kg) (assumed)	640
Strut mass (kg) (assumed)	400
Avionics mass (kg) (assumed)	300
Total mass without propellant (kg)	**14,776**
Total mass with propellant (kg)	**414,776**

Module 3: Solar Panel Extension Module and Additional Docking

The solar panel extension module is a self-contained module that can be launched on the SLS EUS from Earth to L1 to dock with Module 2. The solar panel extension module serves the purpose of providing enough power to the scaled up electrolysis module (module 4) to separate water to produce 811 tons of propellant over a 24 month period. The module consists of a central strut, to which two large Miura foldable solar panels are attached. The Miura folding technique of rigid membranes (Miura, 1985) has been demonstrated for solar panels by researchers at NASA JPL and Brigham Young University. Researchers are currently developing a deployable solar panel array that can expand from a folded diameter of 2.7 m to a deployed diameter of 25 m. In this report it is assumed that by 2030 this technology will be mature enough to be a viable option for solar panel storage and deployment. The solar panel extension module is depicted in **Figure 4.2.2d** and the specifications are summarized in **Table 4.2.2c**.

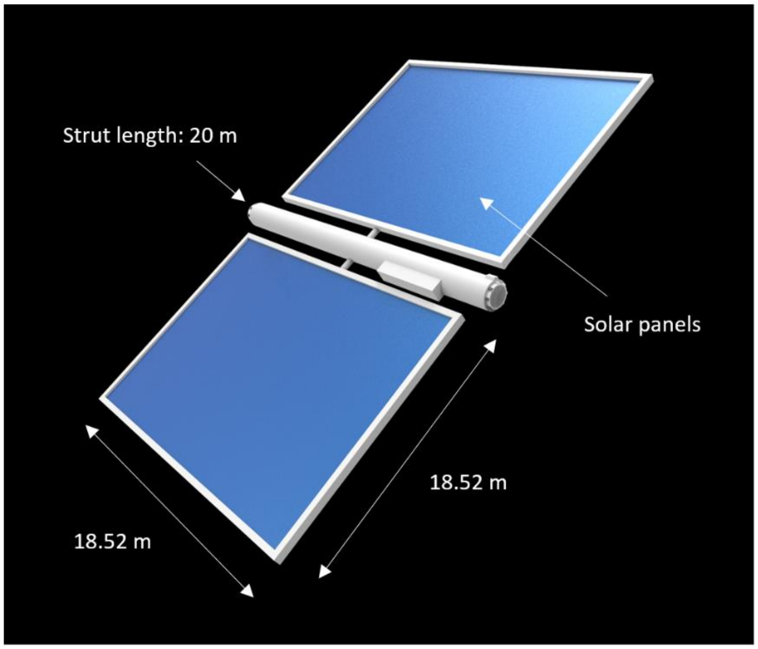


Figure 4.2.2d. *Module 3: solar panel extension module.*

Table 4.2.2c. *Technical specifications for Module 3 of refueling depot.*

Specification	Value
Folded diameter (m)	2
Unfolded diameter (m)	18.52
Fold ratio (assumed from NASA development)	9.26
Area (for two solar panels) (m^2)	686
Efficiency (kW/m^2)	0.267
Surface density (kg/m^2)	2.315
Battery density (kg/m^2)	11.244
Power output (for two solar panels) (kW/month)	183.13
Mass of solar panel (for two solar panels) (kg)	9030
Avionics mass (kg) (assumed)	500
Strut mass (kg) (assumed)	400
Docker mass (kg) (assumed)	640
Total mass (kg)	**10,840**

Because the depot will need to accommodate multiple LRS spacecraft, additional docking ports need to also be taken into consideration. For this reason, an extension port totaling 25 m in length is included in this shipment on the SLS. This would connect to the original docking port on the first module of the depot and open up two additional ports for future use. This is shown in **Figure 4.2.2e**.

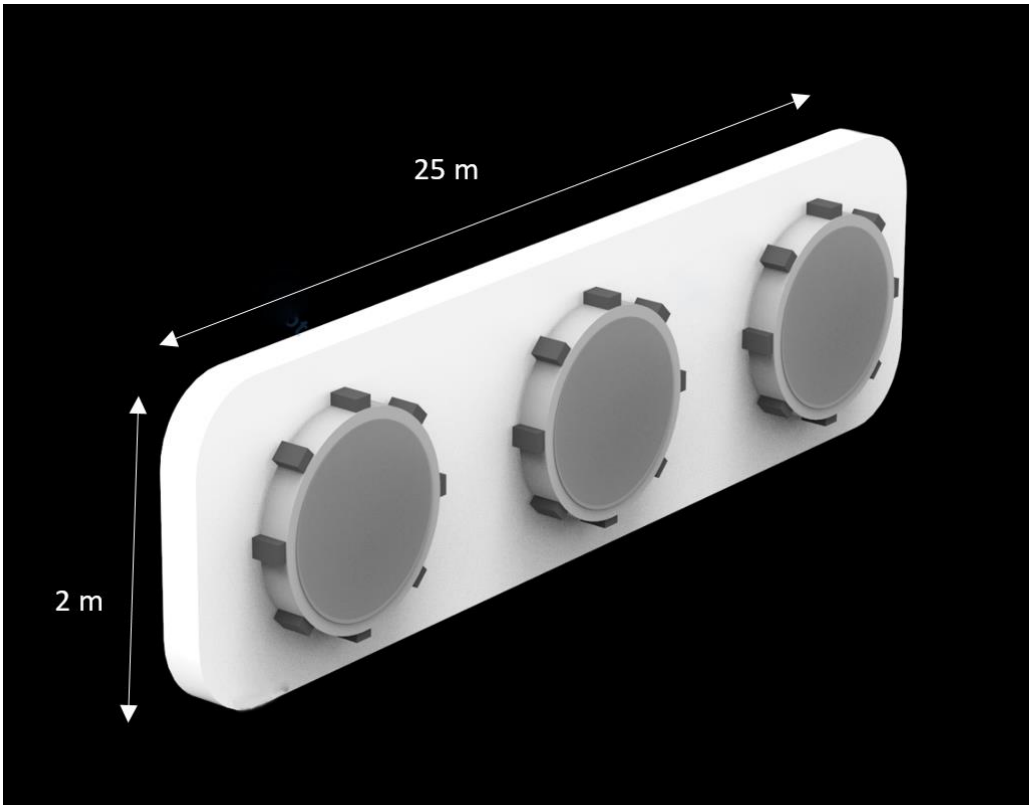


Figure 4.2.2e. *Additional docking port.*

Module 4: Electrolysis extension module

The electrolysis extension module is a self-contained module that can be launched on the SLS EUS from Earth to L1 and dock with the existing refueling depot system. The primary purpose of the electrolysis extension module is to convert the water stored to 800 tons of propellant over a 24-month period. The module consists of two major components: (i) a scaled up electrolysis system, and (ii) a scaled up water storage tank for increased water capacity and functionality. The main assumption made in the design of this module was that we could linearly scale the demonstration modules electrolysis system to meet the requirement of producing at least 600 tons of propellant over a 24-month period. The electrolysis extension module is depicted in **Figure 4.2.2f** and the specifications are summarized in **Table 4.2.2d.**

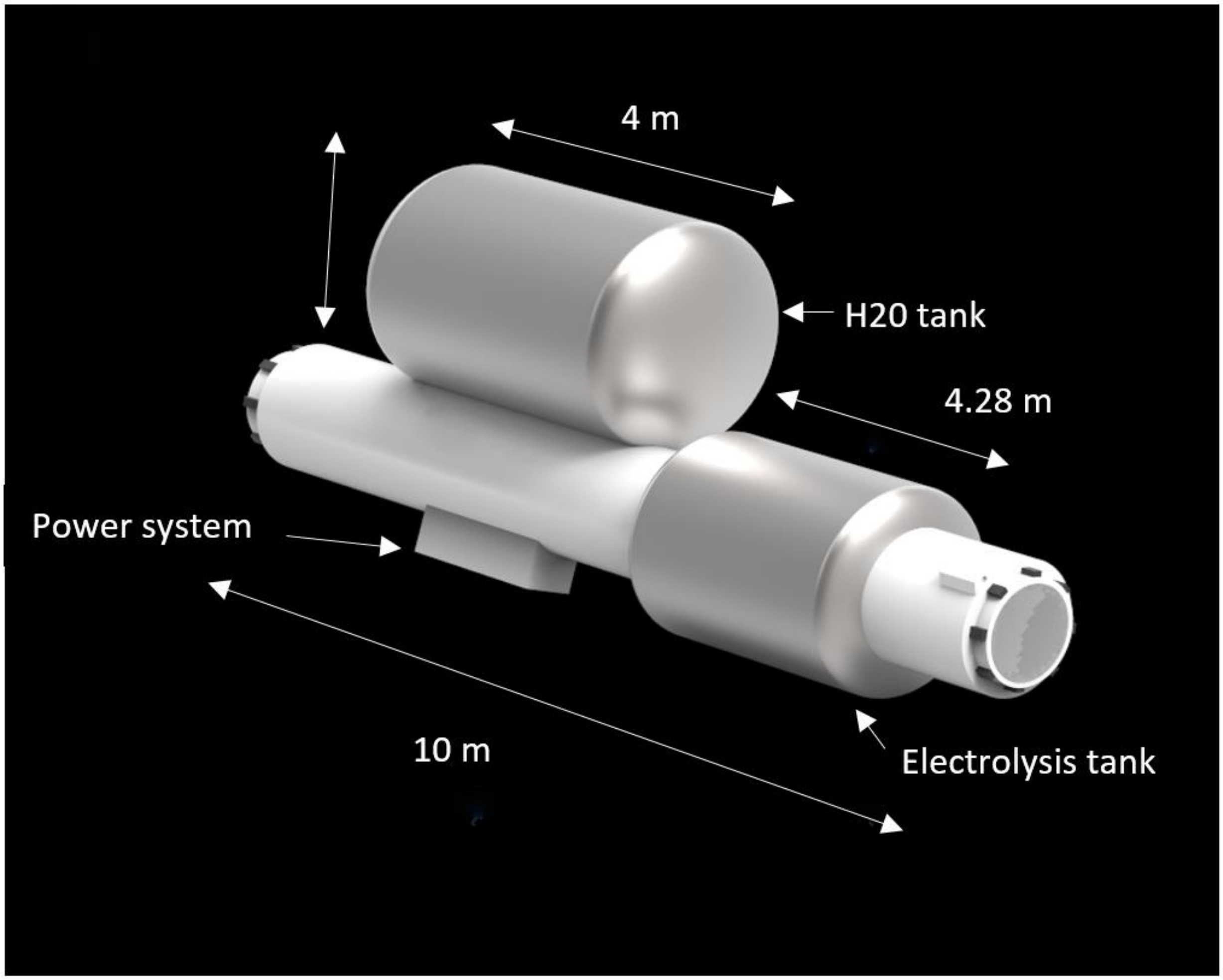


Figure 4.2.2f. *Module 4: electrolysis extension module.*

Table 4.2.2d. *Electrolysis extension module specifications.*

Specification	Value
Power required to convert H2O over 24 months (kg)	168.75
Length of electrolysis unit (m)	4.28
Diameter of electrolysis unit (m)	5
Total number of electrolysis units for 24 month conversion	56
Amount of propellant produced per month (kg)	33792

Mass of electrolysis system (kg)	11,200 (assuming each module weighs 200 kg)
Mass of H2O tank (kg)	3563
Length of H2O tank (m)	4
Diameter of H2O tank (m)	7
Liquid mass of H2O (kg)	150,000
Avionics mass (kg) (assumed)	500
Strut mass (kg) (assumed)	400
Docker mass (kg) (assumed)	640
Total dry mass of module (kg)	**16303**

The total dry mass of the fully assembled refueling depot is about 68.3 tons. The total mass of the fully assembled refueling depot assuming the maximum water and propellant capacity is utilized is 1,044 tons.

System Components

Table 4.2.2e shows the components that make up the refueling depot system. Each component is presented with any modifications or assumptions that have been made about it. The Technology Readiness Level (TRL) is also given.

Table 4.2.2e. *Component TRL specifications.*

Component	COTS or Analogy	Modifications from Analogy (e.g. size, weight, power difference)	TRL
Propellant generation system (Electrolysis)	Scaled-up version of the Oxygen Generation System onboard the ISS	Modified version of the ISS OGS specifically designed to produce propellants	7
Storage tanks	LO2 / LH2 storage in NASA Composite Cryogenic tank	A 5.5 m diameter version of the tank was demonstrated in 2015-16	6
Propellant transfer interface	COTS	Assuming expedited transfer method over next 5-10 years	9
Water-to-LH2 and Water-to-LO2 Liquefiers	COTS	Assuming efficiency increases over next 5-10 years	9

Docking / modular connector	COTS	Assuming improvements in computer vision docking technology over next 5-10 years	9
Solar Array and Power System	Foldable solar panels based on the Miura Fold	Assume current development of a 2.7-m side length to 25-m length foldable solar panel is successful	4
ADCS	Analogy - Moog ISP DST-11H Bi-Propellant thruster	Assume we can modify the system to utilize LO2 and LH2 as the propellants	7
Strut	COTS	Standard satellite strut design	9

Technology Development Plan

This section details the technology development plan for the components used in the refueling depot system that have not yet reached a TRL of 9. The components that are addressed in this section are:

- Solar array and power system (TRL 4)
- Attitude dynamics and control system (TRL 7)
- Storage tanks (TRL 6)
- Propellant generation system (electrolysis system) (TRL 7)

Solar Array and Power System (TRL 4)

The solar panel deployment mechanism utilises the principle described by Miura (1985) whereby, using principles from origami, a large rigid membrane surface can be folded to a significantly smaller geometry. The technology has been demonstrated on the Japanese Space Flight Unit as a "2D Array", launched in 1995 (Japan Space Systems, 2013). Whilst demonstrated on a small scale, a larger scale development is being researched by scientists at JPL and Brigham Young University (Landau, 2014). They are currently developing a fold-up solar panel that can be deployed from an original 2.7-m diameter to unfold to a 25 m diameter. In the design of this component it has been assumed that the deployable design can be linearly scaled to the dimensions detailed above. It has also been assumed that at the time of launch of Module 3: solar panel extension module that the technology for foldable satellites is matured to an operational level.

Attitude Determination and Control System (ADCS) (TRL 7)

The refueling depot will utilize four modified MOOG ISP DST-11H bi-propellant thrusters for attitude control and station keeping. The thrusters are a mature technology and has or is scheduled to be used in Intelsat, BepiColombo, Wild Geese, Tenacious and GOES-R missions (MOOG, 2014). The engine was originally configured for use with Monomethlyhydrazine (MMH) and Mixed Oxides of Nitrogen (MON). For use in the ADCS of the refueling depot, the engine will be modified for operation with LO2 and LH2 so that it can used in parallel with the current propellant stores of the refueling depot. It has been assumed that this modification to the propellant selection will be developed and implemented in time for the initial launch of the demonstrator refueling depot in 2022. The specifications of a single MOOG ISP DST-11H thruster are given in **Table 4.2.2f** (Moog, 2014).

Table 4.2.2f: *MOOG ISP DST-11H specifications.*

Specification	Value
Thrust (N)	22
Nozzle expansion	300:1
Mass (kg)	0.77
Specific impulse (s)	310

Storage Tanks (TRL 6)

Tank Structures

Tank structures are required to store the liquid water, LO2, and LO2. For the liquid water tank, a simple thin aluminium tank will be used; for the LO2 and LH2 tanks, the under-development NASA Composite Cryogenic tank will be used (Knapschafer, 2016). The specifications of the tanks used in each module of the refueling depot are summarized in the previous section. The assumptions made in the design and selection of the under-development NASA Composite Cryogenic tank are that: (i) it will be developed in time for launch on Module 1: demonstrator module in 2022; (ii) The tank provides weight reductions of 25 % compared with its aluminium tank counterpart; (iii) The tank sizing can be scaled linearly to size specified in the tables of the previous section.

Thermal Control of Tank Systems

The requirement being addressed here is primarily concerned with the storage of liquid H2O, LO2 and LH2. This section details the thermal system that is in place for the temperature control and details assumptions that have been made.

Heating of the H2O Tanks

The requirement that this subsystem addresses is the proper storage of the liquid H2O in its respective tank. In the harsh conditions of space, it must be ensured that water remains in liquid phase for the electrolysis process. To ensure this, two common heating elements will be employed. Firstly, a Kapton patch heating element (Durex Industries, n.d.) will be attached to the outer wall of the tank, followed by a multi-layer insulation (MLI) package (NOAA, 2016). It has been assumed that by employing both of these heating methods that the H2O will be able to be maintained in a liquid phase.

LO2 and LH2 Tank Selection

The requirement that this subsystem addresses is the proper storage of LO2 and LH2 in their respective tanks. Cryogenic technology is required to keep oxygen and hydrogen in liquid form. For this subsystem, the under development NASA Composite Cryogenic Tank will be used (Knopschafer, 2016). A 5.5-m diameter version of the tank was demonstrated in 2015-16 and assuming it scales up to current space launch vehicle dimensions should offer a 30% weight reduction and a 25% cost reduction. It is assumed that by the first launch associated with the refueling depot in 2022, this technology will be mature enough to be operational and utilized with the specified sizings.

Propellant Generation System (Electrolysis) (TRL 7)

The electrolysis system is based on the current oxygen generation system (OGS) used onboard the international space station. Conversion efficiencies and power outputs have been calculated based on numbers obtained on the current OGS (Tietronix, n.d.). The amount of energy required to convert one kg of water to O2 and H2 is calculated to be 0.278 kW-h. This assumes a mass efficiency of 0.95, which includes a chemical efficiency of 0.78 and a water delivery efficiency of 0.97. The OGS onboard the ISS can convert 5.63 kg of water to 5.0 kg of O2 in a 24-hour period (Tietronix, n.d.), so this is taken as a baseline in the design calculations. The amount of power required to convert all the water in each module is calculated for a 24-hour period and then scaled to yield the amount of power required to convert the water to O2 and H2 over a longer time period (specified in tables in previous sections). The number of electrolysis system units is calculated based on comparing the amount of power output by the baseline OGS and the power output required by the scaled electrolysis system. Assumptions that have been made in the design of this section include: (i) The propellant generation system will be fully developed and functional by the launch of Module 1: demonstrator module in 2022; and (ii) Through the research and development of the propellant generation system, it has been assumed that the system used on the refueling depot is 30 times more efficient than the OGS currently used by the ISS.

Risk Plan

There are a few mostly minor or rare risk factors that could affect the development or operations of the refueling depot. One of the insignificant risks that will almost certainly happen to a very limited extent is the process of propellant boil-off. The rate of boil-off with current technology is on the order of about 0.1 kg of loss of propellant for each passing hour. With the expected composite NASA cryogenic tanks that can be used, the boil-off can be reduced to essentially zero for the intents and purposes of the mission at hand. The total masses of the propellants in this mission are large and the margins built into the calculation easily overcome the boil-off loss factor. A minor but slightly more significant risk is the possibility of a missed docking of one module to the next when initially delivered. The SLS upper stage has useable thrusters for alignment purposes to overcome this and can be remotely done if necessary. An even more significant risk related to the docking would be the alignment of pipes and / or electrical interfaces between the docking modules. If there is a major misalignment, fuel could leak or connection issues could arise. The risk of this becoming a problem is unlikely, but would be a relatively major issue if it arises. A separate maintenance mission might have to be launched in order to alleviate this. A further rare risk is the potential of an inaccurate thermal control event occurring on any one of the storage tanks. On the largest scale, this risk could increase evaporation of the water or boil-off of the propellants. Insulation for the water tank is well-developed and has been successful in past missions. As mentioned, the boil-off factor is very insignificant on such a large scale, and even a small increase in that would be minor. Finally, a major risk that could happen but is again rare would be the failure of the the deployment of the solar panels, especially in the third module. The Miura Fold concept is in late stages of development and can confidently be considered to be a sound architectural and operational design. Two large solar panels are included for this module so there is redundancy in that sense and the amounts of power that can be generated with each solar panel is high. The final working design requires large amounts of power, so any issue with a major solar panel would be major. However, the option for additional redundancy (or maintenance of a malfunctioning array) is always present and can be incorporated with the modularity that is built into the design. A risk analysis matrix outlining these specific risks is shown in **Table 4.2.2g**.

Table 4.2.2g. *Depot risk analysis matrix.*

LIKELIHOOD	Propellant boil-off				
		Missed docking		Pipe misalignment	
			Inaccurate thermal control	Failure of solar panel deployment	
	CONSEQUENCES				

Schedule

As specified, the refueling depot is a modular design that starts as a small standalone spacecraft and increases in modules over time to increase its capability. The small encompassing Module 1 is launched first, with a planned launch and arrival year of 2022 on the SLS launch vehicle. Because the vast majority of the components have been demonstrated or are at a far-along readiness level, the development time is reasonable over a period of five years. The second module with large LO2 and LH2 tanks is launched second, with a planned launch and arrival year of 2024 on the SLS launch vehicle. As described above, this module connects with the first module, and all subsequent sections connect at the universal ports of the previously launched module. The third module with larger solar panels is then planned to be launched and delivered in 2026 on the SLS launch vehicle. The fourth module with a large electrolysis converter and water tank follows and is planned with a launch and arrival year of 2028 on the SLS. The fifth and final module to bring the depot to full capability for current envisioned Mars missions is a duplicate of the large tanks as in Module 2. This final module fits with a planned launch and arrival in 2032, once again using the SLS launch vehicle, ideally with six years of potential improvement over time to be included. Beyond the first five modules, the full depot will remain modular and can be added to with desired modules for redundancy and end-of-life processes. Time for maintenance of the depot can be scheduled with launches of either robotic maintainers or human astronauts if necessary. **Table 4.2.2h** summarizes the launch schedule.

Table 4.2.2h. *Estimated launch departure and arrival of modules.*

Module	Purpose and Description	Launch Departure and Arrival
1	Small Standalone Module	2022
2	Propellant Tank Module (1)	2024
3	Solar Panel Module	2026
4	Electrolysis and Water Tank Module	2028
5	Propellant Tank Module (2)	2032

Cost Estimation

Rough estimates of the development and construction of the components and parts of the individual modules have been summarized in **Table 4.2.2i**. This covers each module with an order of magnitude estimation and has a complete estimation cost of $1.05 billion. The cost of the launches from the SLS is not shown here but is taken into account later in the report.

Table 4.2.2i. *Cost estimation of individual modules and total depot.*

Module	Purpose and Description	Cost Estimation
1	Small Standalone Module	$300 M
2	Propellant Tank Module (1)	$150 M
3	Solar Panel Module	$250 M
4	Electrolysis and Water Tank Module	$200 M
5	Propellant Tank Module (2)	$150 M
TOTAL	Complete Depot	$1.05 B

4.2.3 Solar Electric Propulsion (SEP) Tug

A Solar Electric Propulsion (SEP) tug is an advanced concept that enables mass savings through high Isp but low thrust. As of 2017, no demonstration missions have been performed but many future deep space mission architectures (i.e. Mars/NEOs) have utilized tugs both for higher payload capability and cheaper launch costs. Several architectures including the Mars DRA 5.0 (Drake, 2009) require upwards of ten SLS launches which may be prohibitively expensive for a government agency. The pros and cons of using an SEP tug system are described in **Table 4.2.3a.**

Table 4.2.3a. *Pros and cons of a SEP tug system.*

Pros	Cons
Reusability	Solar arrays required > 100 kW
Redundancy (clustered EP)	Solar array design

Advancing technology (solar arrays, EP thrusters)	Cost of ground testing and evaluation (massive vacuum chambers)
Faster transfers (deep space)	TRL 9 only demonstrated for < 20 kW EP thrusters (prior to 2017)

During the tug-assisted operational phase described in Section 7.2, the SEP tug delivers the customer's spacecraft via the SLS Block 1b EUS from LEO to the L1 depot. The requirements for the SEP tug are:

- Less than 1.5 year delivery time
- At least one round trip per propellant tank
- Less than 1 MW solar array requirement

The tug will have a solar array providing 700-800 kW BOL and use three 200 kW magnetically shielded concentric channel Hall thrusters, providing long lifetimes and high thrust densities. Magnetic shielding was recently demonstrated on laboratory thrusters, and concentric channel Hall thrusters with projected power levels of 200 kW have been designed and tested. The solar panels will use state-of-the-art solar cells in the 2030s, with projected efficiencies upwards of 33%, and have the capability to fold and deploy within the SLS fairing (10 m x 31 m).

Table 4.2.3b. *Projected magnetically shielded concentric channel Hall thruster properties.*

Parameter	Value
HET Power	200 kW
Isp	5000 s
Thrust	48 N
Total Efficiency	60%
Lifetime	> 50,000 hrs

a) b)

Figure 4.2.3a. *The images shown above are a) the X2 Hall thruster (University of Michigan) and b) a SEP tug CAD model from Donahue et al. (2011).*

Table 4.2.3c. *The 600 kW SEP tug used in this study is based on the 600 kW SEP tug designed by Myers, R. et al (2011).*

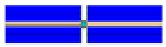
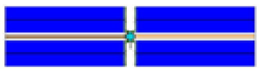
Element Concept	Element Name	Dry Mass, kg	Maximum Wet Mass, kg	Dimensions
	300kW SEP Tug	6,000	36,000	5m dia., 8m overall length (OAL), Stowed
	600kW SEP Tug	8,000	76,000	5m dia., 10m overall length (OAL), Stowed

Table 4.2.3d. *The design parameters were based on a scaled projection of concentric channel Hall thruster properties from Brown, D. et al. (2010).*

	Concentric Channel HET (3 channels)
Input Power	200 kW
Specific Impulse	1300 – 5000 s
Thrust	5 – 14 N (25 – 70 mN/kW)
Mass Flow Rate	100 – 1100 mg/s (Xe)
Efficiency	45% – 64%
Specific Mass	0.5 kg/kW (thruster) 1.4 kg/kW (thruster, PPU)
Major Thruster Dimensions	0.65-m diameter 0.10-m length

To satisfy the requirements, a 600 kW SEP tug system with 200 kW Hall thrusters was chosen. The Hall thruster parameters were based on the values provided by Brown et al. (2010) for next generation high powered electric propulsion. The parameters chosen were a specific impulse of 4000 s, thrust density of 60 mN/kW, total efficiency of 60%, and the masses and dimensions corresponding to a 600 kW tug design from Myers et al. (2011).

Table 4.2.3e. *A trade study of Isp, power, and thruster type was performed to maximize payload delivery capability within the required transit time.*

Assuming Falcon Heavy (53 mt IMLEO), 300 kW SEP-tug - GIT (high-ISP) (Myers 2011), LEO to LLO (8 km/s)

Isp	m1	m_prop1	dM_prop	# trips/tank	m_tank (mt)	m_refuel (mt)	Price Refuel	mdot (kg/s)	t (days)	t (mo)
1000	88.39218	-24.6078	89.60782	0.7253832	5.81152	70.81152002	$ 150,120,250.00	0.0036735	282.33	9.41
2000	125.4345	12.43448	52.56552	1.236552				0.0018367	331.24	11.04
3000	140.9563	27.95634	37.04366	1.7546864				0.0012245	350.14	11.67
4000	149.4233	36.42335	28.57665	2.2745843				0.0009184	360.15	12.00
5000	154.7458	41.74577	23.25423	2.7951899				0.0007347	366.34	12.21
6000	158.399	45.39896	19.60104	3.3161498				0.0006122	370.54	12.35

The final transit time was determined to be one year with a total of 2.27 trips/tank of propellant, resulting in a round trip time of two years and one round trip/tank with some margin. The tank is refueled by Falcon Heavy launches with a tank delivery cap of 54 tons. Replacement of the thrusters needs to be performed approximately every six years (3x round trips). The cost is estimated to be approximately $2B based on an Asteroid Redirect Mission cost of ~$1.5B as of 2017. Maintenance and refueling is projected to cost $1B over a ten-year lifetime for the tug corresponding to five round trip deliveries for a 100 ton EUS payload. The cost benefit for the SEP tug is demonstrated in the cost analysis provided in Section 7.3.

4.2.4 Lunar Transfer Vehicle

High Level Requirements

- The Lunar Transfer Vehicle (LTV) - *Compact* shall be able to host a maximum Payload Mass ($M_{payload}$) of 1.9 tons, the LTV – 5 tons Plus, a maximum $M_{payload}$ of 11 tons and the LTV – 20 tons Plus, a Maximum $M_{payload}$ of 22.2 tons.
- The LTV - Compact shall be able to include an Astrobotic lander of 4.5 m diameter and 1.6 m height and a P-POD deployer with four 6U cubeSats, the LTV – 5 tons Plus, a payload of five Astrobotic landers, and the LTV- 20T Plus, a depot carrier of 20 m height and 9 m diameter.
- The LTV – Compact and LTV – 5 tons Plus should be able to provide enough delta-V to insert into LLO and provide a Midcourse Inclination Correction to their correspondent payload masses.
- The LTV – 20 tons should be able to provide enough delta-V for insertion into L1 Halo orbit from LTI.

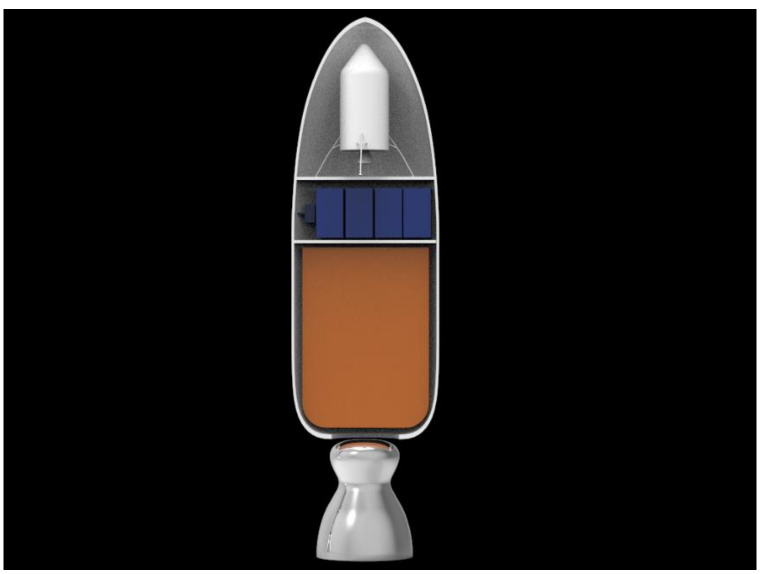


Figure 4.2.4a *LTV-Compact before and cubesat and Lunar Lander deployment.*

Table 4.2.3f. *LTV specifications.*

	LTV – Compact	LTV – 5T Plus	LTV – 20T Plus
Payload	1 Astrobotic Lander + P-POD deployer with 6U CubeSats	5 Astrobotic Landers + Ground Deployment Mechanism	Depot Carrier with Full 20T Depot.
Objective	Transfer from LTI to LLO + Lander and CubeSats Deployment into LLO	Transfer from LTI to LLO + 5 Landers Deployment into LLO	Insertion into Halo L1 orbit and full 20T Full Depot Deployment.
ΔV_{TOT} (m/s)	1480	1480	500
$M_{payload}$ (T)	$M_{TOTAL\ PAYLOAD}$ = 2 T - M_{Astrob_LL} =1.9T (*) - $M_{P\text{-}POD + CubeSats}$ = 0.0383 T	11	20
M_{fuel} (T)	0.8	4.5	0
M_{struct} (T)	0.1	0.5	2.22
$M_{TOT\ LTV}$ (T)	3	16	22.2
Launch Opportunity	Shared Falcon Heavy Expandable (4T share)	Falcon Heavy Expendable (16T Full Payload capacity)	SLS B1b

*Astrobotic Landers are loaded with fuel for powered descent.

The basic technologies for lunar landing have been already proven. The latest example of Lunar Lander is the Chinese Lander Chang'e 3. For high precision landing: the current proven technology (TRL 9) has 100m ellipse error. For these purposes it should be reduced to 10 m with the help of fiducials. This technology has a current TRL of 6.

Table 4.2.3g. *LTV risk analysis matrix.*

LIKELIHOOD					
		P-POD deployment failure		Failure during lunar lander deployment	Engines fail
					Lunar Transfer Vehicle not developed in time
	CONSEQUENCES				

4.2.5 Lunar Lander

The ‘Mercedes’ Lunar Lander is a custom lander able to deploy miner robots to the surface of Moon, deploy solar arrays, and protect the assets during lunar night time. The lander is required to to deploy 40 kW of solar panels and keep its internal temperature above 250 K during the lunar night. It is also required to land five tons of payload to the surface of the Moon.

The Lunar Lander is designed to be the main transportation of the miner robots. It allows power generation on the illuminated part of the crater and power transmission to permanently shadowed regions. Miner robots are stored inside the lander during lunar night time (maximum capacity: six miner robots), while keep the temperature established on the requirements. The technology required has been already developed and proven is space, except for the microwave transmitter. Operational risks include the deceleration of the lander during payload delivery and mechanical failure on solar panel and microwave transmitter deployment.

Table 4.2.3h. *Lunar lander risk analysis matrix.*

LIKELIHOOD					
				Mechanical failure of solar panels; Microwave transmitter deployment failure	Crash of the lunar lander
	CONSEQUENCES				

5 Mission Design

5.1 Launch Vehicle Selection

The launch vehicle selection depends on the payload mass, dimension and timing needs of each phase of the mission. For this reason, the selection process is carried out individually for each payload. **Table 5.1a** shows the main parameters for each launch vehicle, with many of the parameters estimated based on existing technologies or company projections. The main criteria to choose the launcher are the lowest cost per kilogram for the launchers that satisfy our fairing size and mass requirement. For smaller payloads, there was an investigation into combining them into one launcher if the overall cost was smaller. **Table 5.1b** shows the characteristics for each payload along with the selected launcher. More details about each subsystem will be given in their respective sections.

Table 5.1a. *Launcher Comparison.*

Launch Vehicle	Mass to take to LEO (t)	Payload mass to take to TLI (t)	Payload Fairing Diameter (m)	Payload Fairing Length (m)	Volume (m^3)	Cost of Booster ($)	Cost per ton ($)
SLS 1B	105	39	10	31	2435	$1 B*	$26 M
Falcon Heavy (Reuseable)	20	6	4.6	11	183	$90 M*	$15 M
Falcon Heavy (Expendable)	54	16	4.6	11	183	$150 M*	$9.375 M
New Glenn 2-Stage	45	15	4.5	14	223	$150 M*	$10 M
New Glenn 3-Stage	62	20	6	18	509	$200 M*	$10 M
Delta IV Heavy	28	9	5	19	375	$440 M	$49 M

***estimated based on existing launch vehicles or company projections**

Table 5.1b. *Payload Launcher Selection.*

Payload	Mass (tons)	Maximum orbit time	Selected launcher
LRS	20	Short, 1 month	Falcon Heavy
Lunar Lander	15	Long, several months	Falcon Heavy
Small Depot	30	Short, 1 month	Falcon Heavy
Solar Panels for Depot	25	Short, 1 month	SLS 1B
Scale Up Package	15	Short, 1 month	Falcon Heavy

5.2 Orbit Selection

The orbit for the following required trajectories is defined:

- Earth to Lunarport: the considered options are: electric propulsion spiral orbit, low energy orbit with chemical propulsion, and direct transfer with chemical propulsion. The trade study for the different trajectories in shown in **Table 5.2a**. The selected orbit for cargo delivery is direct transfer with chemical propulsion based on the simplicity and low operational risk. The EP spiral orbit was implemented for the advanced operational phase in which an SEP tug tows the customer from LEO to L1. Low-energy chemical maneuvers showed promise for reduced delta-V, but required complex simulation to discern the necessary delta-V and trajectories; future architectures may implement low-energy transfers for optimized performance.

Table 5.2a. *Trajectory options.*

Trajectory Selection	**Criteria**					
	Transit Time	**Complexity**	**Operational Risk**	**TRL**	**Total Cost**	**Propellant Mass Ratio**
EP Spiral	V. Slow	High	Moderate	5	Low	Very Low
LE Chem	V. Slow	Medium	Low	9	Medium	Medium
D Chem	Fast	Low	Low	9	High	Medium

Rendezvous Orbit Selection

The next design decision was the location of the rendezvous orbit with the customer. The main orbits considered are:

1. High Earth Orbit (HEO): a highly elliptical orbit with its perigee on the original customer circular orbit, and a large apogee
2. L1: the Earth-Moon Lagrange Point 1.

Initially, HEO was considered because of its lower delta-V requirement, but later it was realized that, given our limitations in the design of the LRS, using HEO would require the customer to wait more than one month per LRS delivery. That limitation led to the use a cis-lunar depot at L1 for fuel storage and rendezvous. The phasing of a the LRS rendezvous with the customer is also complicated so an LRS would need the precise timing to exactly coincide a translunar trajectory with the HEO location.

Without the depot, refueling of the customer would require 30 dockings with LRS which increases complexity, risk and time (assumption: max capacity of LRS of 20 tons and a Mars cargo mission with 600 tons of fuel). With a depot, the need to have a cryogenic system on the Moon and on the LRS is avoided to ensure zero boil-off which reduces both cost and mass. With a depot, a cryogenic system is only needed on the depot and can produce fuel on-demand for the customer thus ensuring zero boil-off.

The selection of L1 as the rendezvous point also determines the orbits for the flights of the LRS from the Moon to the depot and back. The customer will also traverse to L1, but in this case, from Earth. Each transfer maneuver is listed in **Table 5.2b** with its vehicle, location, initial orbit, destination orbit and delta-V.

Table 5.2b. *Delta-V budget.*

Vehicle	Location	Initial Orbit	Destination Orbit	Delta-V
Lunar Lander	Mid-course	TLI	TLI-Corrected	30 m/s
Lunar Lander	Moon	TLI-corrected	100km Polar Lunar Orbit	4 km/s
Lunar Lander	Moon	100 Polar Lunar Orbit	None	1.9 km/s
LRS	Moon	None	Insertion to L1	2.4 km/s
LRS	L1	Insertion to L1 from the Moon	L1 Halo	20 m/s
Small Depot	L1	Insertion from Earth	L1 Halo	3.8 km/s
Small Depot	L1	Insertion from Earth	L1 Halo	3.8 km/s
Customer	L1	Insertion from Earth	L1 Halo	3.8 km/s
SEP Tug	L1	LEO	L1 Halo	7 km/s

SEP Tug Orbit Selection

The SEP tug described in the following section utilizes a spiral orbit from LEO to L1. Many trajectories have been calculated using codes that account for gravity loss, low-thrust, and perturbations generally yielding the same delta-V to within approximately 20%. For calculating the propellant requirements, a rough value of 7 km/s is assumed as shown in **Table 5.2b**. This level of accuracy was suitable for the early stage concept design in this report. Future studies may incorporate a more detailed calculation of spiral orbits from LEO to a halo L1 orbit with precise delta-V requirements.

5.3 Payload Optimization

The payload delivered to deep space destinations must be optimized to ensure that the propellant depot allows for cheaper missions rather than adding cost and complexity.

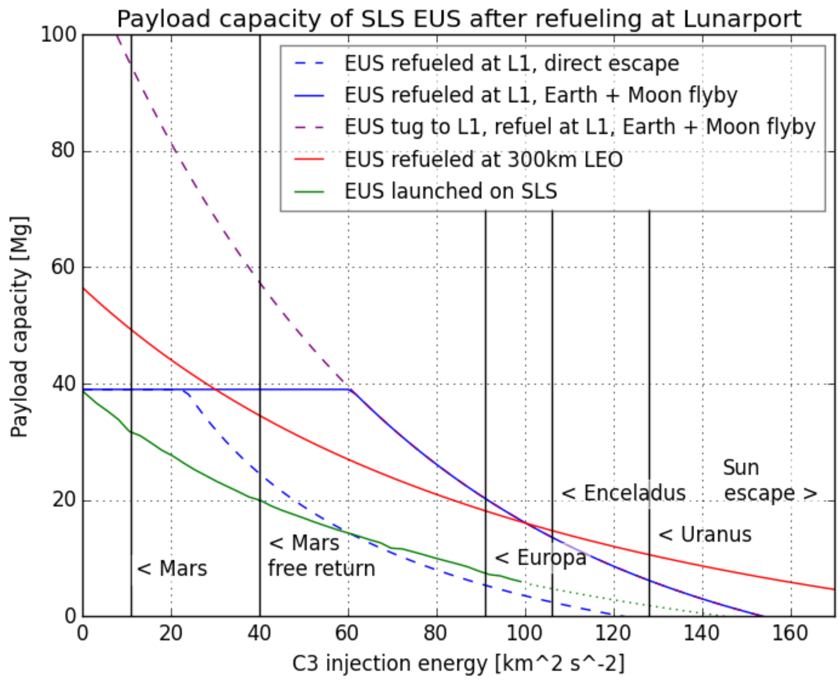


Figure 5.3a. *Refueling at Lunarport's L1 depot increases the payload capacity of the Space Launch System's (SLS) Exploration Upper Stage (EUS).*

In **Figure 5.3a**, the green curve shows the payload capacity of the EUS if launched directly onto an Earth-escape trajectory. The blue curves show the payload capacity of EUS after refueling at L1. Leaving L1 via Earth + Moon Oberth maneuvers is favorable. The blue curves are capped at 39 tons because the SLS/EUS can only bring 39 tons of payload mass to L1. If an SEP tug pulls the EUS and payload to L1, larger payloads are possible (purple curve). Payload capacity for an EUS refueled in LEO is shown for reference (red line); however our architecture does not allow this option because of the high energy cost of transporting propellant from the Moon to LEO. (Data for direct launch of EUS on SLS from Donahue and Sigmon (2013). L1 departure Oberth Trajectories from Schaffer et al. (2012))

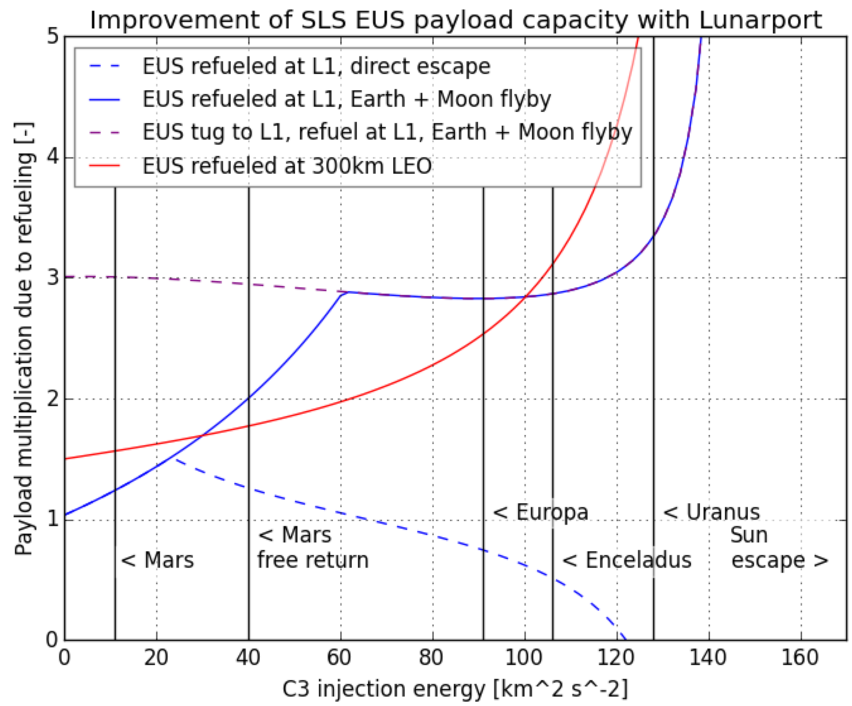


Figure 5.3b. *A threefold increase in the payload capacity of the Space Launch System's (SLS) Exploration Upper Stage (EUS) to many destinations can be achieved by using Lunarport.*

For the icy moons (Europa and Enceladus), launching the EUS to L1, refueling at Lunarport's depot, then departing Earth via Oberth maneuvers triples the payload capacity compared to a direct launch of the EUS. For Mars missions, very heavy payloads (~90 tons) can be injected by an EUS from Lunarport. However, EUS can only lift 39 tons to Lunarport, so a SEP tug is required to realize the full benefit of Lunarport for Mars missions. A first order cost savings can be performed with the following estimations:

1. SEP tug cost ~$2B (including materials cost, development, testing and evaluation)
2. SLS Launch ~ $500M (76 kg to LEO, assuming cost decrease by 2040 and selling ~58% of the remaining payload mass)
3. Maintenance ~$1B (Hall thruster replacement, propellant resupply)
4. Operation costs ~$1B / year

Figure 5.3b shows that the payload to Mars can be increased by 200% resulting in an effective savings of two SLS launches or $2B. The total cost is reduced by performing multiple trips, specifically five round trips of the 102-ton wet mass EUS with 90-ton payload over the course of ten years before replacement of the entire SEP tug. The net income generation will be equal to the $2B cost savings minus the yearly operating cost minus the SEP tug cost divided by five deliveries:

```
$2B - $1B - ($3.5B / 5) = $300M saved per year
```

Therefore, the above calculation demonstrates that net positive revenue can be produced with the assumptions stated above. The above cost assumptions are highly dependent on the decreasing cost of SEP tug technology including > 100 kW solar arrays, deployment, high powered EP with long lifetime, and overall cost estimation methodology. Overall, this result is very promising, showing that future deep space missions, particularly to Mars, can be strongly affected by an ISRU-supplied fuel depot with an SEP tug.

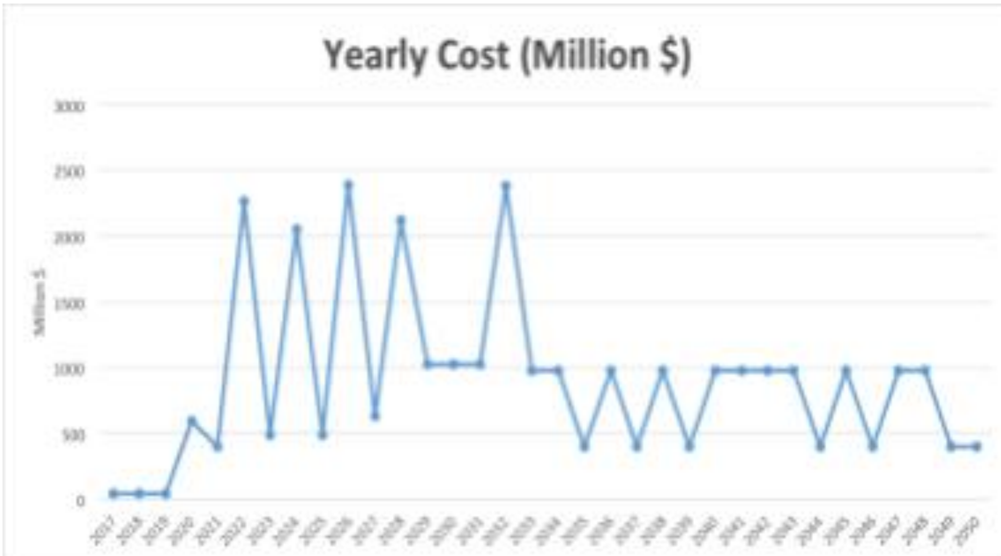


Figure 5.3c. *Lunarport yearly cost estimate in one-year increments from 2017 to 2050 spaced in $500 M increments ranging from $0 to $3000 M.*

6 Operations

6.1 Base Operations

As previously discussed, the Lunarport surface operations will begin with a Lunar Prospecting Scout mission. It is anticipated that this activity will be conducted over a two year period to adequately assess one to two potential mineable prospects in close proximity to the landing site. Prospecting will continue after mining operations begin to continue assessing new mining locations which will be developed at a later date.

The design of the Scout, which will be discussed in greater detail later in the report, is based on the Lunar Resource Prospector and its operations will similarly resemble the planned operations of the Resource Prospector but over a more extended period. The Scout will use the Neutron Spectrometer System (NSS) to detect hydrogen in the subsurface down to a concentration of 5wt% and a depth of one meter. When hydrogen is detected in large enough concentrations near the surface (i.e. <10 cm) a drill sample will be taken to test for the presence of water. The Oxygen and Volatile Content Extractor (OVEN) will heat the sample to high enough temperatures to evolve the volatile gases which are then transferred to the Lunar Advanced Volatile Analyzer (LAVA) for analysis. LAVA has the capability to measure water at concentrations above 0.5 wt%. To be considered a discovery the sample must contain water at a concentration greater than 4 wt%. Deposits on the order of 6-12 wt% will be considered high graded targets and will be the focus of initial mining operations. Once an area of interest is identified by the prospecting instruments the option to map the area in greater detail to delineate the deposits continuity, areal extent, quantity and quality will be decided upon by the science team. This will form the basis of a preliminary resource estimate. Once feasibility studies of the resource are undertaken incorporating engineering, mine planning and cost estimate studies it may be possible to convert the resource to a reserve signaling intent to mine in the very near future.

Lunarport surface mining operations will begin with the robotic mining rovers being delivered to the base site located in relatively well sunlit location. They will leave the base and navigate to the identified resource locations within the permanently shadowed regions (PSRs). Upon arrival at the mining sites, the mining rovers will use their four Honeybee Robotics Planetary Volatile Extractor Corer systems to drill directly down into the icy regolith. The drill system will heat the regolith inside the captured cores, sublimating the ice out of the soil into water vapor. The vapor is then transferred into a cold-trap where it is then deposited into an ice storage tank on the vehicle. The mining rover will drill cores at a rate of 1 core per hour for approximately 20 hours per Earth day, giving it a total ice collection rate of about 10 kg of ice per day per rover. After approximately 20 Earth days of mining and just before the end of the lunar day, the mining rover's ice tank will be full with approximately 200 kg of ice. At this time, the mining rovers will collectively head back to the base, arriving no later than five hours before lunar sunset. The mining rovers will unload their mined ice at the ice storage facility and then head inside the landers that originally brought them down to the surface. Here they will be take shelter for the lunar night in the insulated and actively heated interior of the lander. The mining process restarts at lunar sunrise approximately eight Earth days later.

The mining rovers are equipped with approximately 24 hours of contingency battery power in case they lose access to the beamed power. In this circumstance the rover will enter an emergency mode, mining operations will cease and the rover will attempt restore access to the beamed power source. If unsuccessful, the mining rover can be commanded to head back to the base.

After the ice has been delivered successfully to the storage tanks and has undergone a final filtration some of the water is transferred to the electrolysis unit to produce propellant for the LRS. The electrolysis unit is powered by the solar panels at the site and produces the 20 tons of liquid hydrogen and liquid oxygen propellants. This is required to deliver a 15 ton payload of water/ice up to the L1 orbital depot. An additional LRS will return to the base approximately two weeks later to reload for another trip back to the depot.

6.2 Construction Timeline

The first mission prior to construction is the prospector mission. This exploring phase will determine with greater detail the availability of the resources needed for Lunarport. Alongside the prospector robot a constellation of CubeSats will be launched. They will provide communication with Earth for the days when there is not line of sight between Lunarport and Earth.

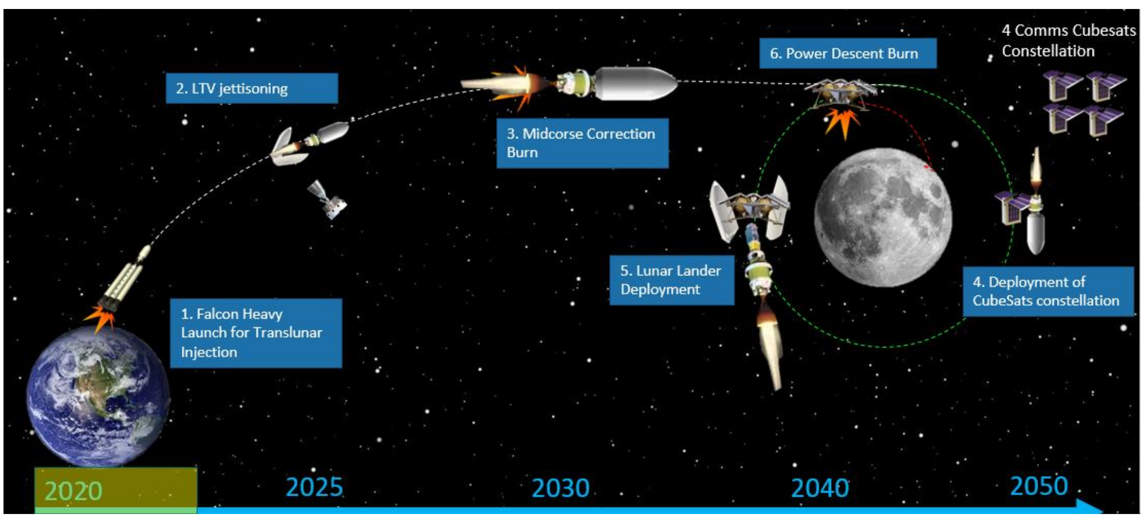


Figure 6.2a. *The first launch includes the prospector and communication CubeSats.*

The next phase is the construction of the base. A 'Mercedes' Lunar Lander is launched with a sintering robot to build the necessary equipment. It will build the roads, the launching pad and the covering berm. This completion of this phase is expected by 2026. The launching pad will allow the landing of the first two LRS spacecraft. They will perform a simulated refueling operation between them to test the refueling operations.

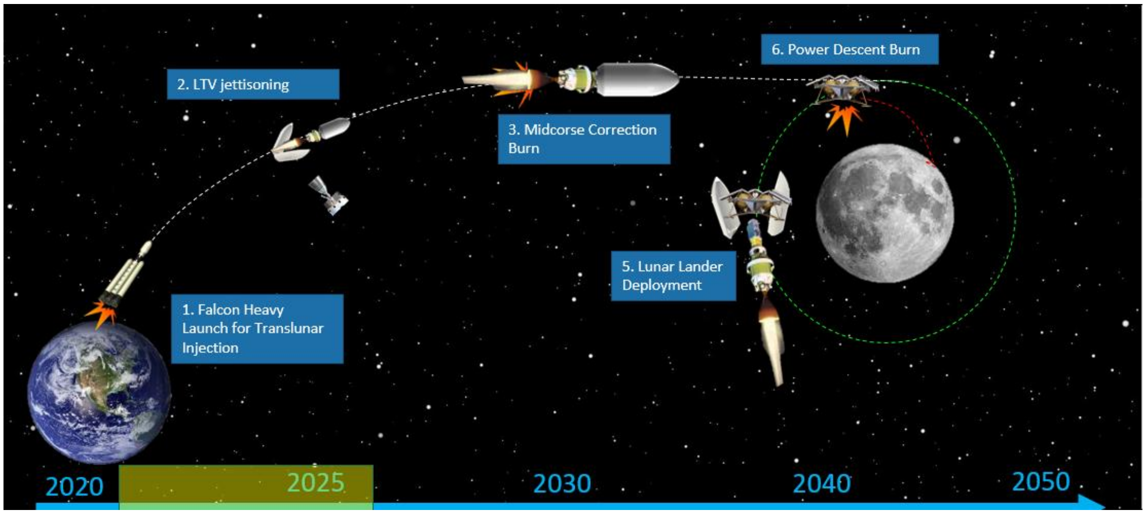


Figure 6.2b. *The general construction phase involves Falcon Heavy deliveries with various Lunar Landers providing the final descent burn to the surface.*

6.3 Operational Timeline

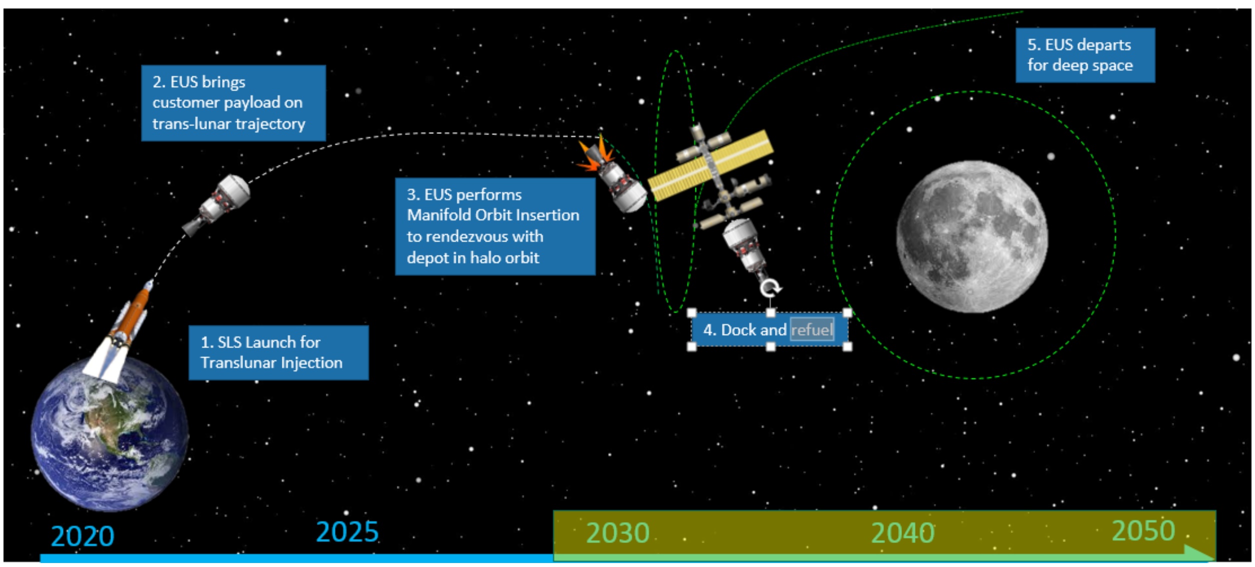


Figure 6.3a. *The operational phase above is reached when the depot begins reaching 120 t capacity with the required delivery cadence from the LRS.*

After direct LRS-to-LRS refueling has been demonstrated and the depot has been fully completed with a capacity of 411 tons (11-ton depot + 400 tons tank module), the operational phase begins. In this phase, the customer can purchase up to 120 tons of propellant at the L1 depot initially in 2023, and up to 400 tons around 2032. The production and delivery rate will continue to scale as an increasing amount of rovers and depot modules are delivered. The SLS Block 1b delivers 39 tons to TLI via the EUS which constrains the maximum payload possible during this phase. This constraint is removed in the tug-assisted operational phase described below.

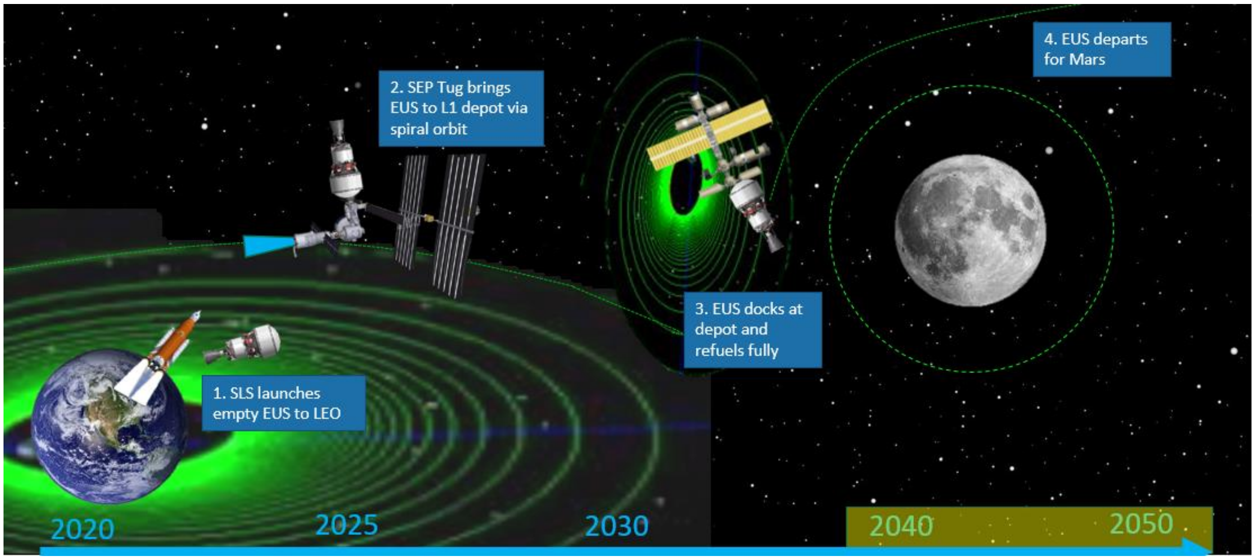


Figure 6.3b. *The schematic above describes the tug-assisted timeline operations once the fully operational phase is reached in the 2040s.*

The final operational phase is reached once the production rate and LRS delivery capability can yield a total of 600 tons of propellant to the depot every four years. The SEP tug is utilized to tow the customer spacecraft (no crew) from LEO to L1 as shown in Figure 7.2b. This savings in delta-V allows the customer to increase the payload mass delivered to Trans-Mars Injection (TMI) according to the tug-architecture shown in **Figure 6.3b**.

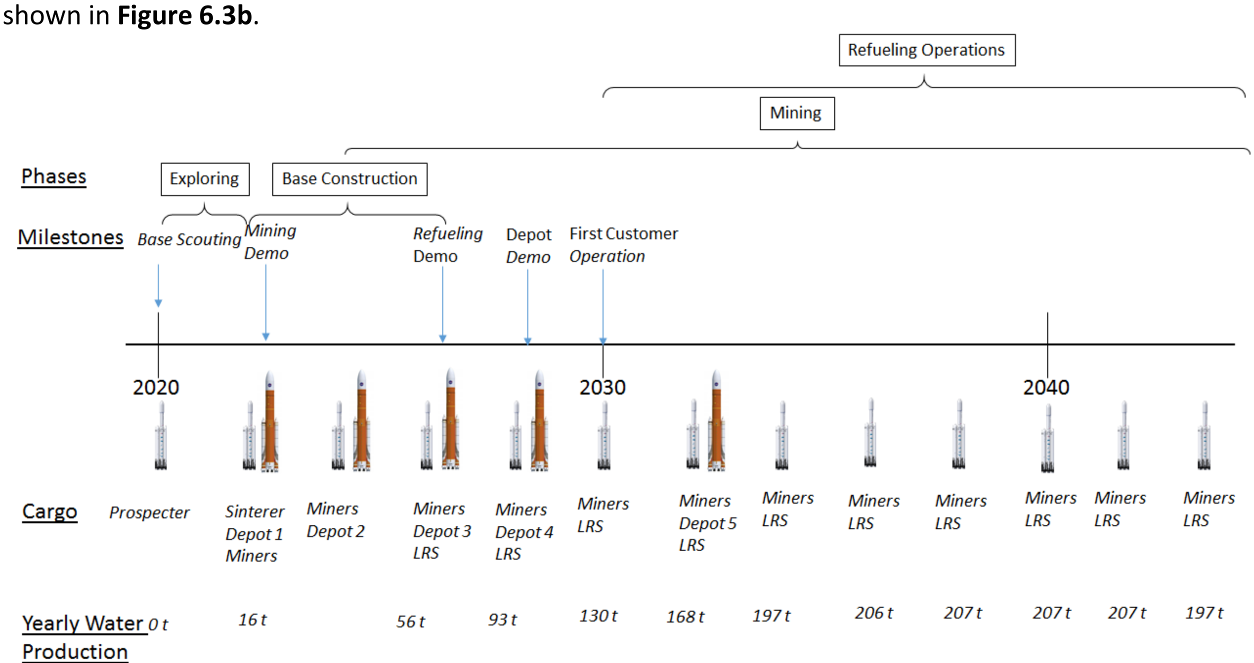


Figure 6.3c. *A detailed timeline highlighting the cargo delivery and yearly water production in their respective phases.*

The propellant production, refueling, and depot propellant capacity are shown in **Figures 6.3d and 6.3e**.

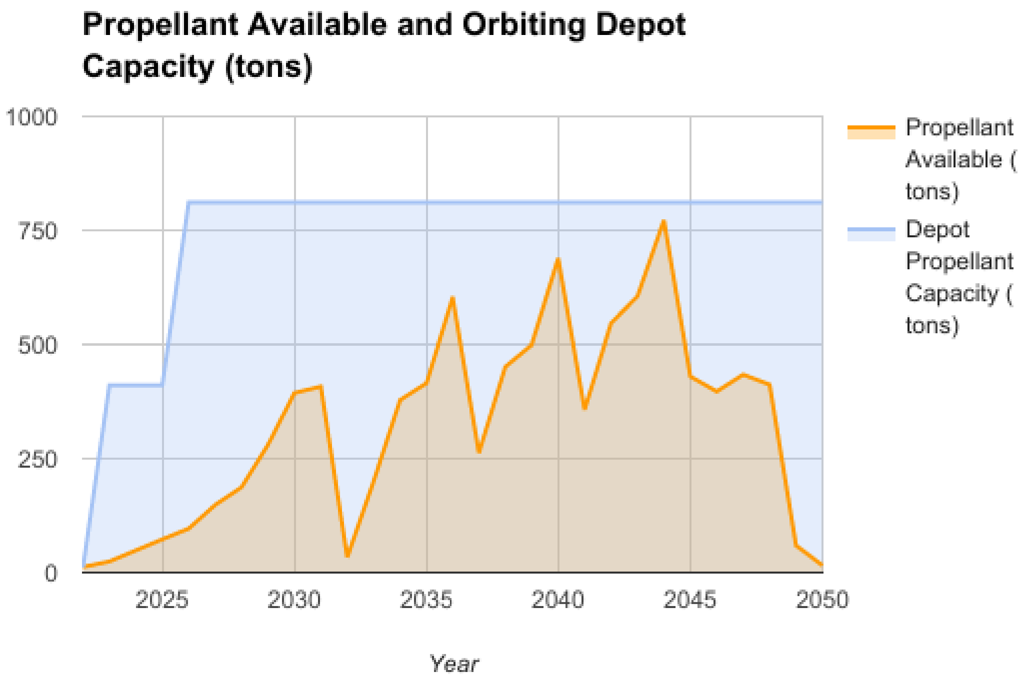


Figure 6.3d. *Propellant available for customer refuel and propellant capacity at orbital depot.*

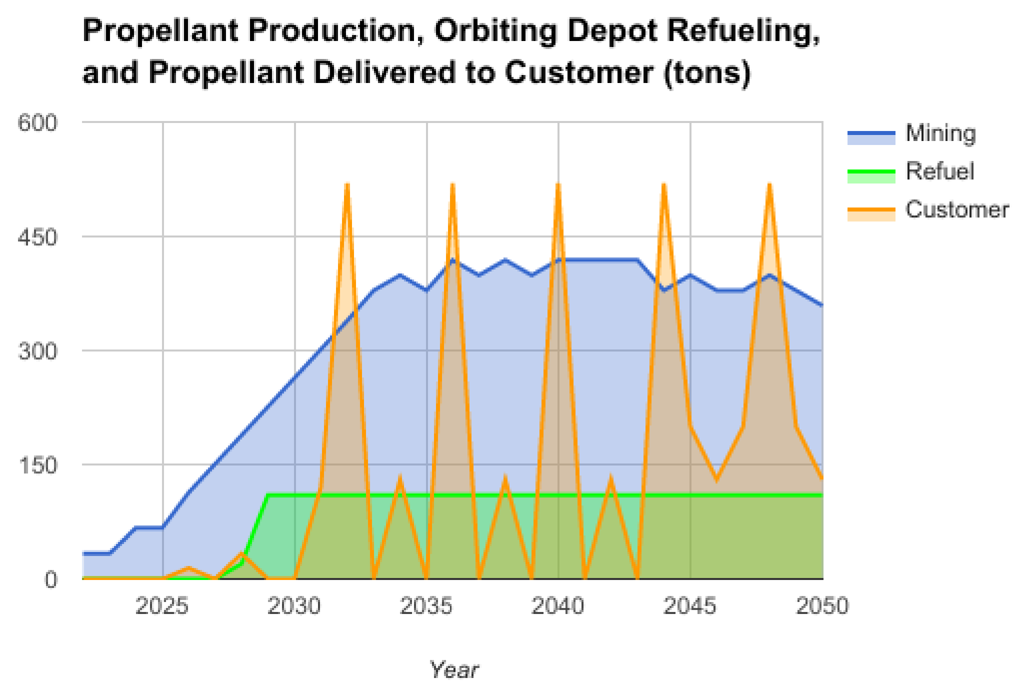
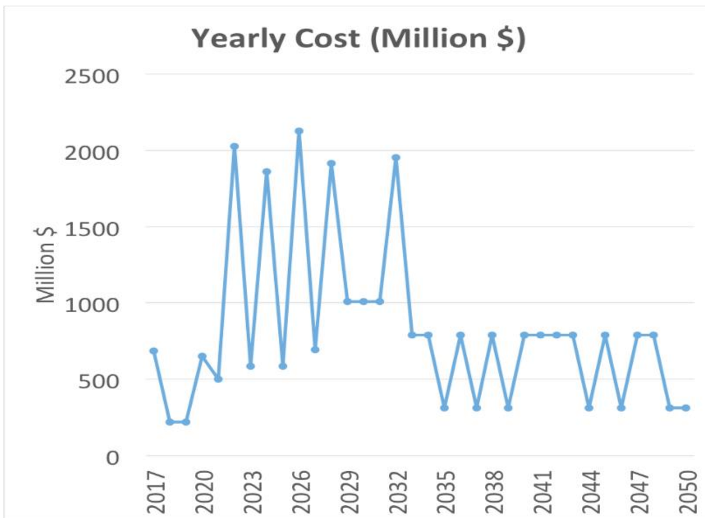


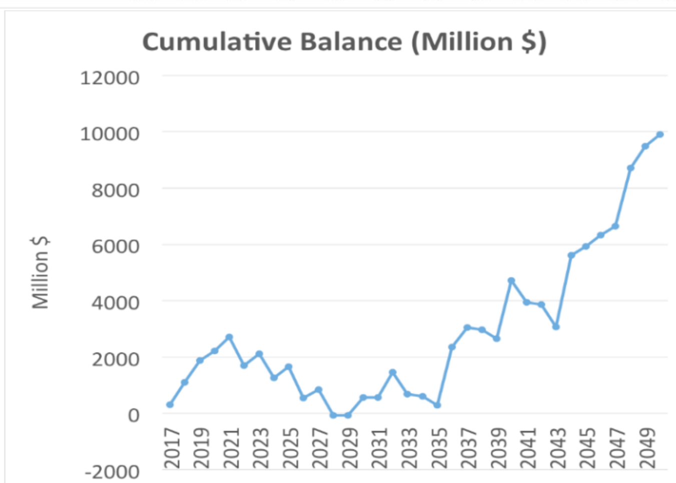
Figure 6.3e. *Propellant capacity, depot capacity, and delivery to customer.*

7 Programmatic Considerations

7.1 Developmental and Operational Cost

The development and construction costs of the project are shown in **Table 7.1a**, and represent the total costs for the initial project development starting in 2017 and ending when the first set of rovers are operational in 2022. It is found that total development costs for this initial mission amount to $4.6B. Analogous technologies were used when available, and scaled if necessary to accommodate creation of new technologies. Approximate scaling factors were utilized for Project Management, Systems Engineering, Safety and Mission Assurance, Science and Technology, Mission Operations, and Ground Control that were suggested by the JPL A-team.





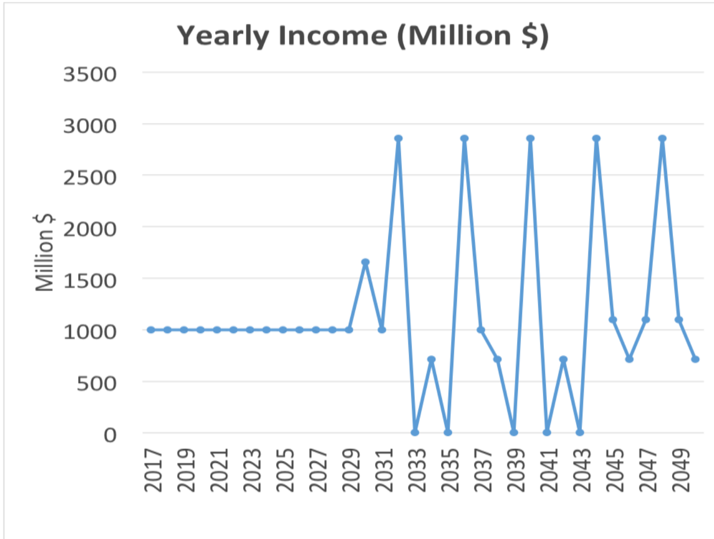


Figure 7.2a. *Yearly cost(top), cumulative balance (middle) assuming $1billion per year in funding through to 2031, and yearly income (bottom) given the cost of operations and nominal Mars refueling missions every four years beginning in 2032 and smaller refueling every two years beginning in 2030.*

Figure 7.2a shows the overall operating costs for the mission in the top panel, while the middle panel shows the total money available to the mission at any given time (annual budget plus rollover minus operating costs). Total costs through 2032 equal $17B. The bottom panel shows the yearly income from refueling missions and the $1B yearly budget. The initial operating cost is low, due to a lack of launches at the beginning of the mission. The $1B / year subsidy of the project ends in 2032, as the project then is making enough profit to be self-sustaining. It is assumed that there are small refueling missions of 120 tons per year every four years as well as Mars-scale missions of 520 tons every four years at full operational capability. The breakdown of the total operating costs by year and assumptions is found in Appendix B.

As shown in **Figures 5.3a** and **5.3b**, Ice Rush is capable of extending the amount of propellant available to a variety of deep space missions, including scientific missions to Jovian moons, Mars missions and Mars free returns; missions to asteroids and station-keeping satellites.

Table 7.1a. *Development costs. Based on initial 2022 mission, with analog technologies listed. Recurring costs are listed in Appendix C.*

System	Source	Development and Construction Cost ($M)
Project Management, Systems Engineering, Safety and Mission Assurance	9% of all non-launch costs	$102
Mission Operations and Ground Control	10% of all non-launch costs	$114

Payload and Spacecraft	Sintering Rover— based on mass Prospector— Lunar Resource Prospector with RTG and LIDAR Miner Rover—Structure: Apollo Lunar Roving Vehicle, Drill from Honeybee Robotics Electrolysis unit—ISS Small Lander—Astrobotic lander Big Lander— Apollo lander Depot modules—NASA protype request Communications—Cubesat estimates LRS—Cargo Dragon, Cygnus, and Centaur	$2405
Systems I&T	Double Curiosity for non-recurring; same as Curiosity for recurring	$200
Launch/Vehicle Services	SLS plus Falcon Heavy	$1270
Science/Technology	5% of total cost	$57
Reserves	Fixed cost, 20% of total	$481
Total		$4629

7.2 Long Term Roadmap

Ice Rush has a present budget allocation of $1B per year for the lifetime of its mission. As NASA's mission has transformed from one responsible for all U.S. space activity to one that primarily enables the development of non-commercially viable scientific and technological missions, unless NASA's cost structure changes, the Lunarport will have to transition to another business model once profitable. Several options could exist for the long-term future of Ice Rush:

<u>No-cost NASA refueling station</u>
NASA could aim to keep operations of Ice Rush at a level that only meet NASA or US government requirements for space missions. The refueling station may also be of interest to NOAA and DoD missions that require station keeping, particularly for expensive GEO satellites. The propellant depot reserve could also provide risk-mitigation for end-of-life satellite operations/refueling scenarios.

<u>No-cost international collaboration refueling station</u>
NASA may exchange free refueling to other space agencies for utilization of other resources of said space agencies; e.g., in exchange for using the Moon Village

No-cost commercial and NASA refueling station for solar system exploration

In order to promote the U.S. space industry (currently a mission of FAA AST), NASA could provide propellant at cost to U.S. commercial ventures if in excess of U.S. government need. This would effectively subsidize the commercial industry, and encourage space companies to incorporate within the U.S. If demand were too high for the Lunarport production rate, some form of cost-sharing structure between the private industry and NASA could be utilized.

Sale of Ice Rush to net $0 cost

NASA, once Ice Rush enters full operational phase and is deemed a “proven” technology/business, could offer the sale of Lunarport to a commercial entity to amortize the total cost to NASA. It could be continuously offered on the free market for the total outstanding “debt” NASA has for the project, until purchased by a commercial entity, which would be able to develop it as they saw fit. The sale would likely have to be to a U.S. company due to International Traffic in Arms Restrictions (ITAR).

Sale of Ice Rush in exchange for future NASA utilization

NASA could also contract with a commercial company for the sale of Ice Rush in exchange for refueling use for further NASA missions. ITAR would again likely restrict the sale to a U.S. company.

7.3 Political Considerations

The United Nations Treaty on *Principles Governing the Activities of States in the Exploration and Use of Space, Including the Moon and Other Bodies*, or Outer Space Treaty, ratified by 105 nations, including all major space powers, was enacted in 1967 after nearly a decade of negotiations on space law post Sputnik launch. This treaty, in addition to three other space treaties, forms the basis of international space law.

Article II of the United Nations Outer Space Treaty of 1967, asserts that “Outer space, including the moon and other celestial bodies, is not subject to national appropriation by claim of sovereignty, by means of use or occupation, or by any other means.” While there was an effort to create a follow-on Moon Treaty, the *Agreement Governing the Activities of States on the Moon and Other Celestial Bodies,* in 1979 to more carefully define lunar activities, this effort failed, with only 17 countries party to the treaty (including no spacefaring countries). The Moon Treaty likely failed for several reasons, including Article XI, which required that “neither the surface nor the subsurface of the moon, nor any part thereof or natural resources in place, shall become property of any State, international intergovernmental or non-governmental organization, national organization or non-governmental entity or of any natural person”. Because of the failure of the United Nations Committee on the Peaceful Uses of Outer Space to coalesce on a narrower version of “claim of sovereignty”, the utilization of extraterrestrial resources is at present a legal gray area.

Both the United States and Luxembourg have chosen to adopt the perspective that resource extraction does not violate the Outer Space Treaty, with Luxembourg partnering with Planetary Resources for asteroid mining (Planetary Resources, 2016), and approvals by the Federal Aviation Administration Office of Commercial Space Transportation (FAA AST) for both the Moon Express probe as well as the Bigelow Aerospace lunar base. Though Bigelow has no immediate plans for a lunar base, the AST payload review was seen as a step to measure the regulatory uncertainty for lunar property rights (Foust, 2015). The approval was viewed by many as a U.S. government endorsement for commercial activities on other celestial bodies, suggesting that permanent lunar fixtures, not just short-term probes such as Moon Express, are likely viable for commercialization.

While it may be possible to receive AST authorization for commercial lunar activities, the process is presently ad hoc for missions that are not standard (e.g., launches of communications satellites), and may involve approval from other agencies, including the State department, National Oceanic and Atmospheric Administration (NOAA), and the Federal Communications Commission (FCC) depending on the mission payloads. Though this creates complexity in the mission approval process for commercial activities, AST is required by the Commercial Space Launch Act (HR. 3942) to make licensing decisions within 180 days.

7.4 Planetary Protection

Article IX of the Outer Space Treaty requires that states party to the treaty conduct operations on the moon and other celestial bodies to avoid harmful contamination. Under NASA's Planetary Protection guidelines, non-returning lunar missions fall under Category II (NASA, NPR 8020.12D). This project will request a preliminary Planetary Protection Office categorization, and will provide the Planetary Protection Plan at the end of Phase B (the Conceptual Study). In addition, the Pre-Launch Planetary Protection Report, Post-Launch Planetary Protection report, and End of Mission Report will be provided in coincidence with our mission timeline.

7.5 Public Relations and Outreach

The Lunarport will have several Public Relations and Outreach components.

Adopt-a-Rover Competition

In the spirit of "Boaty McBoatface", which highlighted a U.K. Antarctic research vessel which otherwise would not have received world renown, the Lunarport team will solicit naming ideas from K-12 students through an annual process. The names will then be voted on in an online poll, with the top contenders chosen for future rover launches. Winning classrooms will be able to "adopt" their rover, and participate in education and public outreach activities over the course of the mission, which include a launch-watching party with NASA scientists (locally or virtually, depending on location), weekly updates on their rover, an opportunity to plan one day of science research with NASA scientists, and NASA swag sent in installments as the rover hits certain production landmarks.

Prospector Robotics Tutorial

Grade 3-12 classrooms will also be able to request a loan of Sphero robots to participate in a mock lunar prospecting activity. The basis for this activity has already been developed by the Australian Victorian Space Science Education Center, and has been tested with students already. Students will learn how to program the robot in stages, including sending students through an obstacle course as robots given written directions from their partners, and then translating this knowledge to program and navigate the Sphero robots through a mock lunar surface.

Citizen Science: Mars Trail

As site selection for rover prospecting is not yet certain, the science team for operation Ice Rush will establish a citizen science program, entitled Mars Trail, utilizing already reduced data and training sets to help identify further sites to prospect. Participants will be able to earn resources to be able to construct their own virtual architecture to Mars (e.g., examine 20 images, gain a lunar prospector. Examine 10000 images; gain a cis-lunar habitat module). The virtual architecture will be placed in a Sim City-esque environment, where continuous operations of the participant's program will be continuously modeled, and decisions about mission design can be made by the participant, such that they are effectively constructing their own gateway to Mars.

Lunar Virtual Reality (VR)

A further concept is to additionally build upon the Mars 2030 virtual reality (VR) platform, and make the imaging data available in VR. This data may also be integrated into the citizen science project.

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Appendix – Work Breakdown

Breakdown			Team Member
Project Management	Schedule Management		Donal O'Sullivan
	Resources Management		Donal O'Sullivan
Systems Engineering	Concept Development		Sydney Katz
	Design		Sung Wha Kang
	Analysis		Sydney Katz
Safety and Mission Assurance			Therese Jones
Science and Technology			Gary Li
Payload	Management		Mercedes Herreras Martinez
	Product Assurance		Mercedes Herreras Martinez
	System Engineering		Jack Henry de Frahan
	Integration, Assembly, Test, and Check		Jack Henry de Frahan
Flight System and Spacecraft	Project Management		Donal O'Sullivan
	Systems Engineering		Sydney Katz
	Product Assurance		Therese Jones
	Spacecraft	Structures and Mechanisms	Nariman Sharifrazi
		Thermal Control	Vinicius Guimaraes Goecks
		Electrical Power	Vinicius Guimaraes Goecks
		GN&C	Matt Vernacchia
		Propulsion	Matt Vernacchia
		Communications	Daniel Pastor Moreno
		C&DH	Daniel Pastor Moreno
		Software	Daniel Pastor Moreno
		Integration and Test	Nicholas Jamieson
	Lunar Lander Systems		Mercedes Herreras Martinez
	Base	Manager	Nariman Sharifrazi
		Site Selection	Sophia Casanova

		Power	Vinicius Guimaraes Goecks
		Mining	Sophia Casanova
		Roads/Launchpad	Sumudu Herath Mudiyanselage
		Communication	Nariman Sharifrazi
	Lunar Resupply Shuttle	Manager	Matthew Vernacchia
		GNC	Sydney Katz
		Propulsion	Matthew Vernacchia
		Structures	Matthew Vernacchia
		Docking and Propellant Transfer	Bryan Sinkovec
	Refueling Depot	Manager	Bryan Sinkovec
		Power	Bryan Sinkovec
		Configuration	Nicholas Jamieson
		Transfer Mechanisms	Nicholas Jamieson
Mission Operations	Mission Operations Center		Joseph Sparta
	Science/Data Operations Center		Joseph Sparta
	Data Distribution and Archival		Therese Jones
	Ground Stations		Jack Henry de Frahan
	Communication		Daniel Pastor Moreno
Launch Vehicle Services			Gary Li
Ground Operations			Joseph Sparta
Systems Integration and Testing			Sydney Katz
Education and Public Outreach			Therese Jones
Artistic Renderings			Sung Wha Kang