LEEP

Lunar **E**xtraction for **E**xtra-terrestrial **P**rospecting

CALTECH SPACE CHALLENGE - 2017

LEEP

CALTECH | GALCIT | JPL | AIRBUS

The Caltech Space Challenge

The Caltech Space Challenge is a 5-day international student space mission design competition.

The Caltech Space Challenge was started in 2011 by Caltech graduate students Prakhar Mehrotra and Jonathan Mihaly, hosted by the Keck Institute for Space Studies (KISS) and the Graduate Aerospace Laboratories of Caltech (GALCIT). Participants of the 2011 challenge designed a crewed mission to a Near-Earth Object (NEO). The second edition of the Caltech Space Challenge, held in 2013, dealt with developing a crewed mission to a Martian moon. In 2015, the third Caltech Space Challenge was held, during which participants were challenged to design a mission that would land humans on an asteroid brought into Lunar orbit, extract the asteroid's resources and demonstrate their use.

For the Caltech Space Challenge, 32 participants are selected from a large pool of applicants and invited to Caltech during Caltech's Spring break. They are divided into two teams of 16 and given the mission statement during the first day of the competition. They have 5 days to design the best mission plan, which they present on the final day to a jury of industry experts. Jurors then select the winning team.

Lectures from engineers and scientists from prestigious space companies and agencies (Airbus, SpaceX, JPL, NASA, etc.) are given to the students to help them solve the different issues of the proposed mission. This confluence of people and resources is a unique opportunity for young and enthusiastic students to work with experienced professionals in academia, industry, and national laboratories. The 2017 challenge subject and the team members are presented in the following sections.

The whole Team Explorer would like to thank the organizers, **Ilana Gat** and **Thibaud Talon** as well as all the mentors and sponsors for making this event possible.

Organizers, Guest Lecturers, Mentors, and Sponsors

The co-chairs of the 2017 Caltech Space Challenge are Ilana Gat and Thibaud Talon. Ilana is a Ph.D. Candidate in Space Engineering at Caltech working on Sub-grid Scale Modelling of Non-uniform Density and Pressure Flows under Professor Paul Dimotakis. Thibaud is a Ph.D. student in Space Engineering at Caltech working on the Space Solar Power Initiative and the "Autonomous Assembly of a Reconfigurable Space Telescope" project under Professor Sergio Pellegrino. The guest lecturers are Steve Matousek, NASA JPL; Damon Landau, NASA JPL; A.C. Charania, Blue Origin; Kris Zacny, Honeybee Robotics; Brian Roberts, NASA Goddard; Jay Trimble, NASA Ames and Antonio Elias, Orbital ATK. The list of sponsors of the challenge is presented here:

Fig. 1: The 2017 Caltech Space Challenge Sponsors

The mentors are Ashley Karp, Ph.D., Propulsion Engineer; Andreas Frick, Systems Engineer; Frank E Laipert, Mission Design Engineer, Heather Duckworth, Systems Engineer, Farah Alibay, Systems Engineer; Jonathan M Mihaly, Technologist (Co-Chair of 2011 Caltech Space Challenge), Jason Rabinovitch, Mechanical Engineer (Co-Chair of 2013 Caltech Space Challenge); Hayden Burgoyne, VP, Spacecraft Systems at Analytical Space, Inc. (Co-Chair of 2015 Caltech Space Challenge); Niccolo Cymbalist, Associate in Thermal Sciences at Exponent (Co-Chair of 2015 Caltech Space Challenge), Jennifer R Miller, Systems Engineer; Sydney Do, Systems Engineer; Emily A Howard, Mechanical Engineer; John B Steeves, Optical Engineer; Manan Arya, Technologist; Kristina Hogstrom, Systems Engineer; Aline K Zimmer, Systems Engineer; Daniel M Coatta, Systems Engineer; Alan Didion, Systems Engineer; Carl Seubert, Guidance and Control Engineer; Adrian Stoica, Senior Research Scientist and Group Supervisor; Jared Atkinson, Sr. Geophysical Engineer at Honeybee Robotics Spacecraft Mechanisms Corporation and Jessie Kawata, Creative Strategist + Industrial Design Lead.

Team Members

Fig. 2: Team Explorer and MAGGIE, the Mars Science Laboratory's (Curiosity) Engineering Model at JPL

The 2017 Caltech Space Challenge Team Explorer members are:

Abhishek Anand	: Harvard University
Alexander Reeves	: California Institute of Technology
Andreas Marquis	: University of Toronto
Andrew Kurzrok	: Yale University
Daniel Crews	: University of Washington
Danielle DeLatte	: University of Tokyo
Flora Mechentel	: Stanford University
Isabel Torron	: Rhode Island School of Design
Jerome Gilleron	: ISAE-SUPAERO
Manuel Diaz Ramos	: University of Colorado Boulder
Nikhil More	: Technische Universität Berlin
Padraig Lysandrou	: Cornell University
Peter Buhler	: California Institute of Technology
Samuel Wald	: Massachusetts Institute of Technology
Shane Carberry Mogan	: New York University
Sonia Ben Hamida	: Ecole Centrale Supélec

Executive Summary

The development of space and human progress beyond our world is largely limited in this day and age by the cost per kilogram to deliver a payload to orbit. Furthermore, the current most powerful launch vehicle in the world has a maximum deliverable payload to Low Earth Orbit (LEO) of about 29 metric tons (mT). For comparison the Saturn V which carried astronauts to the moon could deliver a payload of 140 mT to LEO. NASA is currently developing the next generation of heavy launch vehicle to rival the performance of the Saturn V, but access beyond the Earth will still be limited by existing launch vehicles. What happens if a mission requires more performance and is it achievable without the exorbitant cost of developing ever larger launchers?

Lunarport seeks to answer this question by going back to the moon. Working within a proposed budget of 1 billion dollars a year, a mining base is to be established on the south pole of the moon to extract water frozen just beneath the surface of a permanently shadowed crater. The ultimate goal of Lunarport is to explore the economic feasibility of refueling a NASA SLS upper stage with propellant harvested from the moon.

LEEP is a proposal for a Lunarport that incorporates high TRL systems and a highly robust, modular, fault-tolerant design to produce propellant for deep space missions at the lunar South Pole on a short time scale and with a low risk of mission failure. Every attempt has been made to make LEEP both realistic and feasible, and to design a mission that provides direct and indirect benefits in the most cost-effective way possible.

Operating at nominal capacity expected in the late 2020s, LEEP can resupply one mission to Mars per year, enabling a 27.6% increase in payload delivered to Trans-Mars Injection (39.5 mT vs. 31.7 mT). The modular architecture could be expanded in the future to enable multiple missions per year, and its modular nature means that LEEP's expansion can be completed for a fraction of the cost of the initial system.

One particularly interesting application of LEEP's architecture is in support of refueling missions to high-energy destinations. Early numbers indicate a 250% increase in payload delivered directly to a Trans-Saturn Injection compared to a mission that is not refueled, for example, and the more energetic the destination, the greater the benefit. This has direct applications for robotic exploration of the outer Solar System and for vastly expanded mission capabilities at very little additional cost.

1. System Architecture

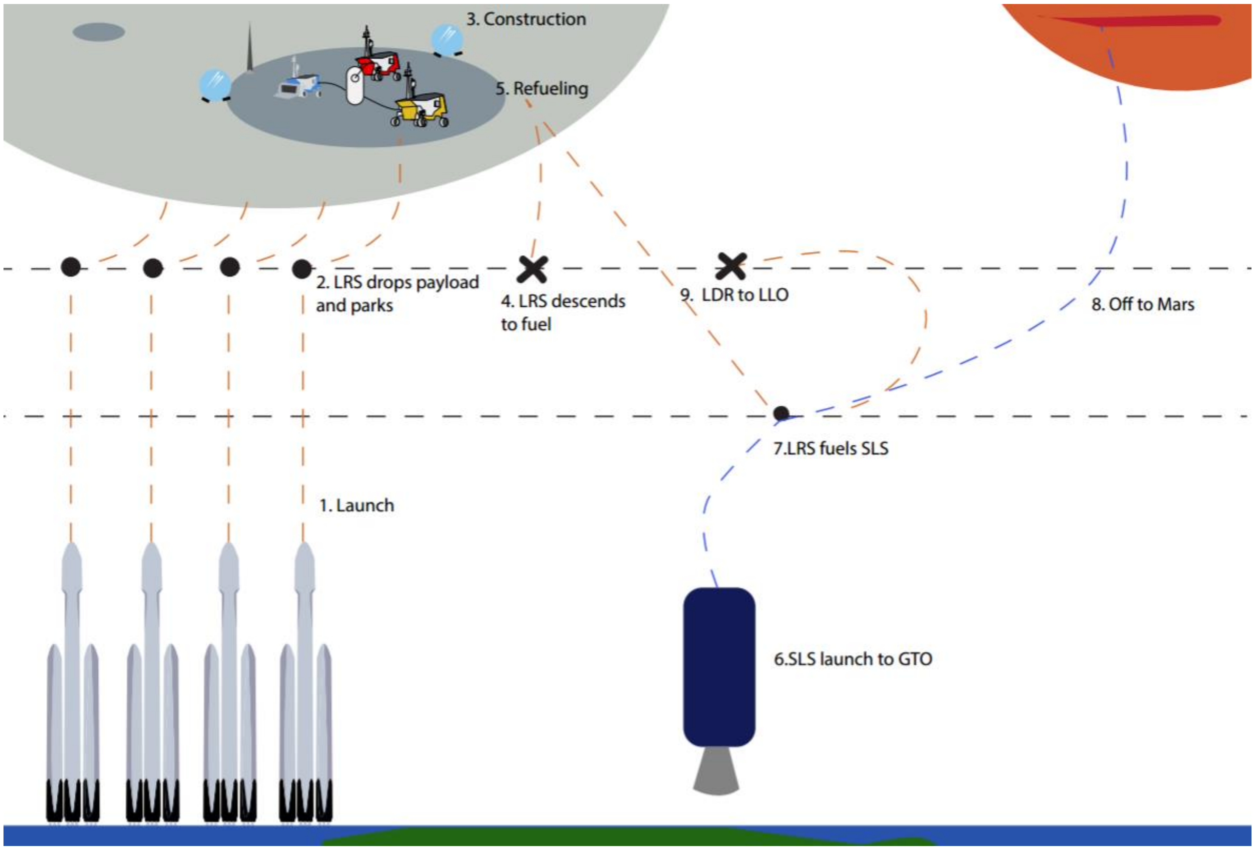


Fig. 3: CONOPS for LEEP

Launch Year	Deployment
2024	Deployment of two solar focusing stations in separate locations along the rim of Cabeus crater.
2026	Delivery of equipment into a permanently shadowed crater region to prepare for ISRU.
2027	Landing of H2O extraction rovers and electrolytic processing equipment.
2028	Delivery of additional extraction rovers.

Tab. 1: CONOPS details for LEEP

2. Ground Operations

Ground operations are conducted to extract water from the icy lunar regolith and process it into cryogenic LOX/LH2 fuel for the refueling tankers (the modified Centaurs). Ground operation deployment consists of four launches:

1. Power System for H_2O Electrolysis (2024)
    - Station on rim to beam power into the dark crater for extractor units.
    - Electrolyzer unit must operate continuously at 70 kW to meet fueling requirements.
2. Prospector and Multipurpose Constructor Rovers (2026)
3. Electrolyzer Unit and Extractor Rovers (2027)
4. Remaining Extractors for Full Capacity (2028)

The first payload is launched in 2024 and deploys solar focusing equipment along the crater rim to illuminate the landing site and provide available power. The second payload delivers four rovers in 2026 into the permanently shadowed crater region, two of which are for construction and maintenance, and two for ice deposit prospecting. The construction/maintenance rover then deploys a solar farm within the crater region to power the in-coming Electrolyzer unit. In 2027 the third ground payload delivers the ISRU electrolysis unit and a first batch of extraction rovers. Water extraction and processing begins. Lessons learned are incorporated into the builds of the second batch of extraction rovers, which are delivered into the crater as the fourth lunar surface payload in 2028, bringing the total number of extraction rovers to twelve and the base to full propellant production capacity.

The delivery sequence of lunar surface equipment requires delivering multiple robotic rovers at once and in the same location. This is done with a larger version of a typical retrorocket descent rover deployment shell called the Lunar Landing System (LLS). The LLS consists of a platform, capable of receiving a modular payload that has an integrated hypergolic bipropellant propulsion system intended for one-time use and designed to be as versatile as possible when it comes to delivering our equipment to the lunar surface. The propulsion system is an Aerozine 50/N2O4 hypergolic system. Three kinds of equipment are delivered. On the crater rim, two LLS's carrying 5 folded solar focusing mirrors each land in typically lit regions. These deploy to their determined locations and focus solar light into the crater. The used landing system then deploys a parabolic dish for direct-to-Earth communications.

An LLS with two prospecting rovers and two construction rovers land within the volatile-rich darkened region of Cabeus crater. The constructors prepare crater base for the Lunar Resupply Vehicles (LRS) to land by clearing loose regolith with a bulldozer. The ISRU H_2O processing unit lands with retrorockets on a modified LLS without any rovers, and a total of twelve extractor rovers are deployed in two LLS runs. It's estimated that each extractor rover can mine and deliver to the Electrolyzer unit 40 kg/day of H_2O when equipped with four Honeybee Robotics PVEx coring devices. Once the base is fully deployed in 2028 as described, it can extract and process 90 mT of H_2O per year with an Electrolyzer unit operating at 70 kW (assuming 35 kW of water splitting power). This meets the 60 mT of propellant required for an EUS refuel mission with ample margin for problems with extractors and for LH2 boil-off problems.

3. Space Operations

LEEP's primary mission is to refuel spacecraft in cis-lunar space. To do so, it uses modified Centaur upper stages as Lunar Resupply Shuttles (LRSs). These Centaurs are modified with composite landing legs, enhanced GNC systems, ULA's IVF system for reducing boil-off and vehicle complexity, and other modifications (e.g. solar panels) as necessary depending on the performance of the IVF system. These Centaurs are refueled on the lunar surface by an ISRU, then launch into LLO, transfer to a low-periapsis elliptical orbit around the Earth, rendezvous with a craft to be refueled, transfer their excess fuel, and then return to the lunar surface.

The use of Centaurs leverages a mature and proven technology to decrease development costs and increase reliability of the process, and the use of multiple smaller refueling vehicles adds redundancy and fault tolerance to LEEP's ability to conduct refueling operations, reducing the risk associated with putting a vehicle in orbit and trusting that LEEP will be able to resupply it. Using Centaurs and refueling the Large Upper Stage (LUS) of the Space Launch System (SLS), our team developed computational analysis tools to determine the ideal rendezvous orbit.

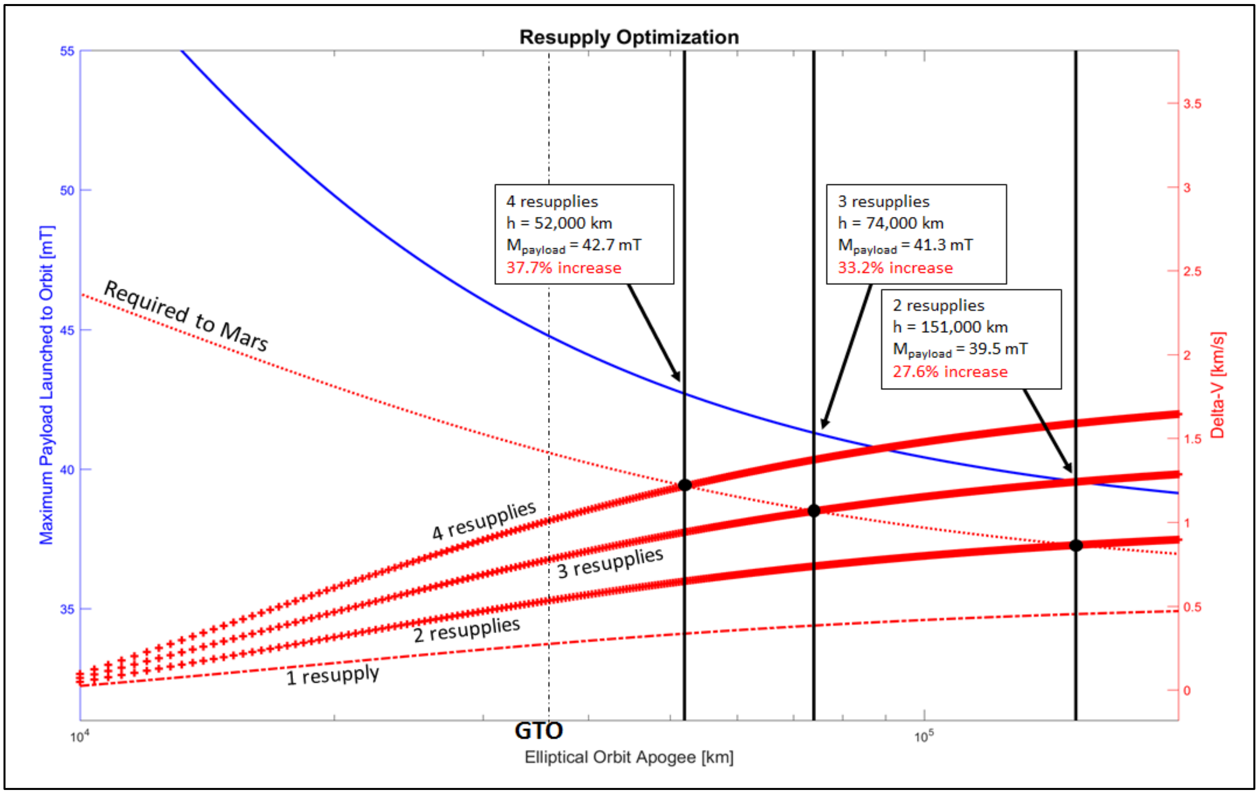


Fig. 4: Trajectory Optimization Analysis

Figure 4 shows the results of optimizing rendezvous orbits for refueling a Large Upper Stage using various numbers of Centaur LRSs. We believe that the optimal solution is to send two refueling vehicles, because sending more LRSs represents a very large investment in propellant production operations but does not result in a comparably large increase in payload. This suggests that our increase in mass sent to TMI is approximately 28%. Keeping in mind that each payload mass includes the empty mass of the LUS, the increase in usable payload is over 45%. As mentioned in the overview, sending smaller payloads to more energetic orbits more fully utilizes LEEP's capabilities than sending large payloads to less energetic orbits.

4. Economics & Schedule

The total non-recurring cost for LEEP is approximately $10.2B and the estimated average recurring annual cost is $80M per year. The development of the system is spread over 12 years. The break-even point when only considering single-launch SLS missions to TMI is 37 launches, or 1200 mT. However, the benefit to missions to the outer planets could see significantly larger increases in payload capacity and increased value.

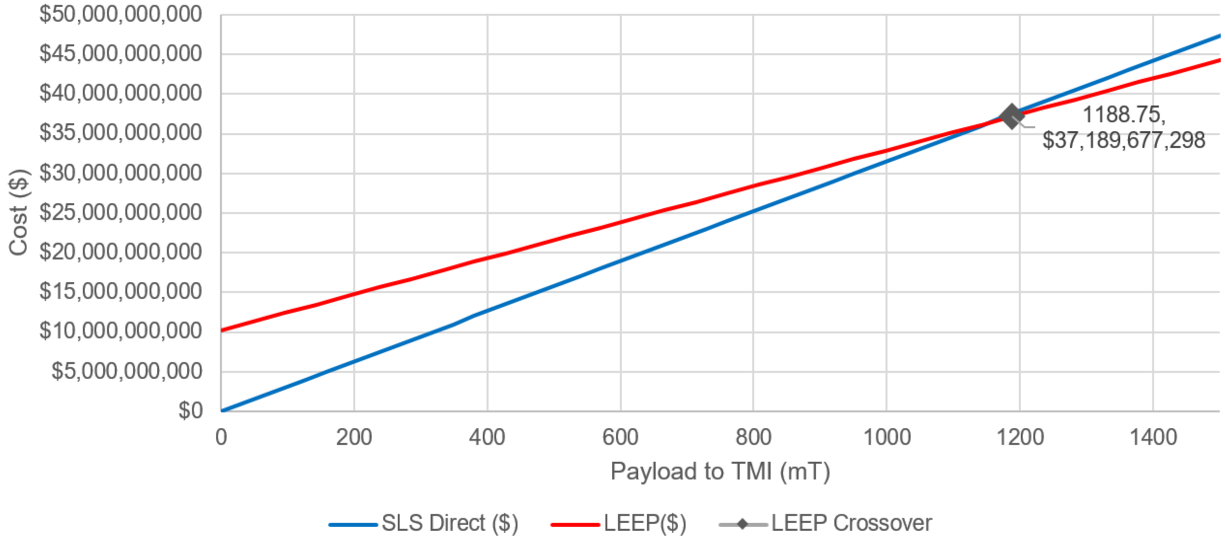


Fig. 5: Cost vs Payload TMI

The development of technologies and hardware takes place over 12 years. The cost has been spread over this period to meet budget constraints and realistic development times. System reviews have been scheduled for this period. The project schedule has been shown on the next page.

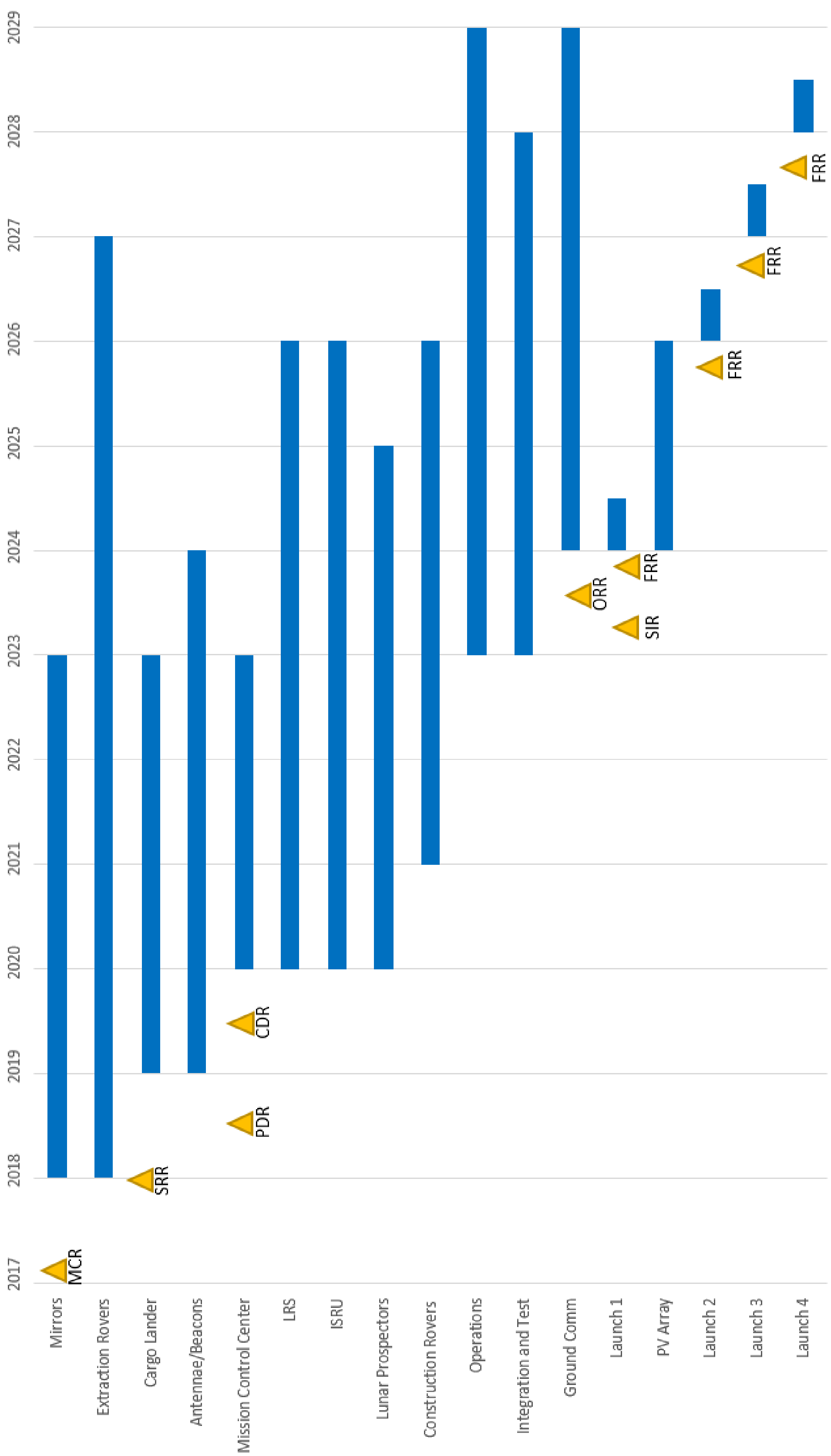


Fig. 6: Project Schedule

5. Future Expansion

LEEP was conceptualized in the context of a tight schedule (boots on Mars by the end of the 2030s) and a small budget ($1 billion per year). Because of these constraints, we were not able to take advantage of innovations like electric propulsion, small modular nuclear reactors, nuclear thermal rockets, and similar technologies. However, LEEP could be upgraded with these technologies as they become available and costs decrease. Its modular architecture makes LEEP an excellent platform for continual improvement as new technologies become available, and provides an already-in-place infrastructure that allows for easy deployment and utilization of new technologies.

Contents

Acronyms

AOS : Acquisition of Signal
CFM : Cryogenic Fuel Management
CML : Concept Maturity Level
CRYOTE : Cryogenic Orbital Test
DPS : Descent Propulsion System
EUS : Exploration Upper Stage
EIRP : Estimated Isotropic Radiative Power
FSPL : Free space path loss
GNC : Guidance, Navigation and Control
GPS : Global Positioning System
HEO : High Earth Orbit
ISECG : International Space Exploration Coordination Group
ISRU : In-Situ Resources Utilization
IVF : Integrated Vehicle Fluids
JPL : Jet Propulsion Laboratory
LCROSS : Lunar Crater Observation and Sensing Satellite
LEEP : Lunar Extraction for Extra-terrestrial Prospecting
LEO : Low Earth Orbit
LLO : Low Lunar Orbit
LLV : Lunar Landing Vehicle
LRS : Lunar Resupply Shuttle
LUS : Large Upper Stage
LV : Launch Vehicle
MLI : Multi-Layer Insulation
NASA : National Aeronautics and Space Administration
R&D : Research and Development
TEI : Trans Earth Injection
TLI : Trans Lunar Injection
TRL : Technology Readiness Level
ULA : United Launch Alliance
ZBO : Zero Boil-Off

1. Introduction

1.1 Problem Statement

Human space exploration on the moon and in Earth orbit is the defining endeavor of the 20th and 21st century. Now, with humans planning to travel to Mars, the notion of becoming a multi-planet species is a quickly arriving reality. Physics, however, offers no short cuts, and the historical challenge of requiring fuel just to power the movement of stored fuel remains. In-space refueling may offer an alternative solution to this problem.

Problem Statement: Develop a full mission concept to refuel deep space bound vehicles with fuel extracted from the Moon.

1.2 Inspiration

A five-year-old, who watched the first footprints on the moon in July, 1969 is 54 years old today. We believe in the dream of 54 year olds across the globe seeing a second set of historic footprints within one lifetime: on Mars. Stretching from the moon to Mars in one generation is the grand challenge facing today and tomorrow's workforce of scientists, engineers, mission planners, designers, fabricators, and educators. And plans are coming together. Twelve spacefaring nations have a coordinated Mars strategy, and, in the United States, NASA is rocketing forward with the development of the Space Launch System (SLS) and the Orion crew capsule by the mid-2030s. If we get started today we can reach our goal, on the Martian surface and among a generation following from earth.

1.3 Context

Mars is approaching. U.S. National Space Policy declares that NASA "will send humans to orbit mars and return them safely," a goal echoed in NASA's strategic plan. The funding follows: today, nearly 19% of the agency's budget supports SLS or Orion, the two most prominent elements of the journey to Mars architecture.[1] The United States is not alone in the goal of Mars. European and Indian satellites currently orbit Mars alongside American counterparts, and 2020 may see the first private departure to Mars in the form of SpaceX's Red Dragon.

[1] "National Aeronautics and Space Administration FY 2016 Spending Plan for Appropriations Provided By P.L. 114-113," NASA, September 2016. Available online at https://www.nasa.gov/sites/default/files/atoms/files/fy16_operating_plan_4sept_update_0.pdf.

To develop the technology and techniques necessary to get to Mars, NASA, in cooperation with international partners, has constructed a roadmap of three phases to prepare for Mars: 1) Earth Reliant missions, 2) Proving Ground missions, and 3) Earth Independent missions. Of these three, the phase of greatest relevance to lunar refueling is Proving Ground. Figure 7 excerpts NASA's objectives for this phase.

Proving Ground Objectives		
Category	**Title**	**Objective**
Transportation	Crew Transportation	Provide ability to transport at least four crew to cislunar space.
Transportation	Heavy Launch Capability	Provide beyond LEO launch capabilities to include crew, co-manifested payloads, and large cargo.
Transportation	In-Space Propulsion	Provide in-space propulsion capabilities to send crew and cargo on Mars-class mission durations and distances.
Transportation	Deep Space Navigation and Communication	Provide and validate cislunar and Mars system navigation and communication.
Working in Space	Science	Enable science community objectives.
Working in Space	Deep Space Operations	Provide deep-space operations capabilities: • Extravehicular activity (EVA) • Staging • Logistics • Human-robotic integration • Autonomous operations
Working in Space	*In-Situ* Resource Utilization	Understand the nature and distribution of volatiles and extraction techniques, and decide on their potential use in the human exploration architecture.
Staying Healthy	Deep Space Habitation	Provide beyond LEO habitation systems sufficient to support at least four crew on Mars-class mission durations and dormancy.
Staying Healthy	Crew Health	Validate crew health, performance, and mitigation protocols for Mars-class missions.

Fig. 1.3.1: Extract from NASA Journey to Mars proving ground objectives[2]

In late March 2017, NASA announced the Deep Space Gateway to support Mars mission learning objectives.[2] However, the cancellation of the Asteroid Return Mission (ARM) in NASA's FY18 Proposed Budget removes a substantial pillar of the "Proving Ground." At the same time, there is a tremendous opportunity in the commercial space sector by providing the infrastructure that is needed to support the businesses and ventures that drive the global economy. Interest in cis-lunar economy is demonstrated by the interest in the Google Lunar X-Prize, the many private start-ups and proposals, and the tremendous opportunities and wealth of resources found on the moon. Doing a sustained mission on the moon over

[2] "Deep Space Gateways to Open Opportunities for Distant Destinations," NASA, March 28, 2017. Available online at https://www.nasa.gov/feature/deep-space-gateway-to-open-opportunities-for-distant-destinations.

decades provides an incredible wealth of information about how to operate in a harsh environment not only for a two week mission, but for a duration and sustained presence. Numerous ideas have been proposed, but what is missing is the real, in-situ experience and increased TRL levels.

The Lunar Extraction for Extra-planetary Prospecting (LEEP) mission is the key to unlocking deep space missions, beginning with Mars. LEEP will help NASA, partner agencies, and the private sectors develop critical deep space technologies, starting with ISRU and robotics. For NASA, LEEP would provide a "lifeboat" for the first long-duration Orion mission and could enable a 30% increase in payload to Mars for the first human mission.

LEEP is also the first power plant for the solar system. While the costs today are prohibitively high, it is likely the forerunner for a new industry of providing fuel as a service on orbit. This is the exact same model we see in cloud computing. Physics remains cruel; it takes fuel to lift fuel. Why not outsource? As more entities move into orbit, offering flexible energy and logistics services will be big business, just like it is here on earth. Now is the time and place to learn those skills.

This project's focus on heritage hardware and increasing TRL-6 level projects to TRL-8 and TRL-9 opens up the options for groups who have made various proposals. From the table below and the entrants to competitions such as the Google Lunar X-Prize, it is clear that no one nation owns interest in going back to the moon. As ESA has suggested with Moon Village, it will take all of humanity to go back and set up permanent off-Earth habitation.

In addition to the mining capabilities that are demonstrated and developed in the LEEP project, capabilities are enabled for other nations or missions to take part in. There has been tremendous interest in the South Pole as a place for radio astronomy, infrared missions, a test bed for teleoperation, and sustained instrumentation[3]. This project would set up the infrastructure and raise TRL levels for a wide variety of technologies both on the lunar surface and in orbit. Once assets like communications infrastructure and launch pads start to develop, other missions have an easier time with their early stages and benefit from the lessons learned.

Mars is coming. The research accomplished by the LEEP mission will move humans on the Red Planet from science fiction to science.

[3] Davis, G.W. et al. "The Lunar Split Mission: Concepts for Robotically Constructed Lunar Bases." International Lunar Conference, 2005.

The primary mission benefits of LEEP to the main stakeholders can be summarized as below:

Humans	**Space Agency**	**Private Company**
• Open up universe to humanity • Search for life outside Earth	• Explore deep space and minimize launched mass from Earth • Create international partnership with private companies and space agencies • Gain knowledge and competencies on deep space exploration	• Test mining systems • Develop new markets

Fig. 1.3.2: Primary LEEP Mission Benefits

2. Mission Overview

2.1 Mission Statement

Mission Statement: LEEP delivers an in-space refueling service to enable deep-space exploration and commercial missions. Fuel is produced from lunar resources. The Lunar port also affords to gain knowledge and experience, as well as fosters international partnerships with institutions and private companies.

2.2 Mission Objectives

The mission addresses the following objectives for the different phases of the mission.

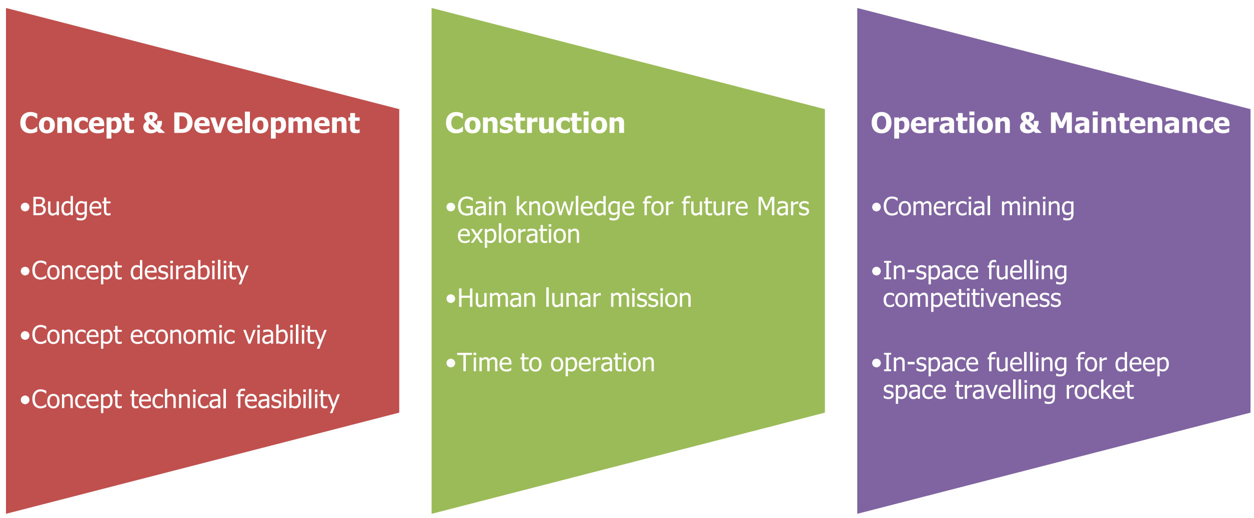


Fig. 2.2.1: Mission Objectives

The systems relationships have been depicted on the next page.

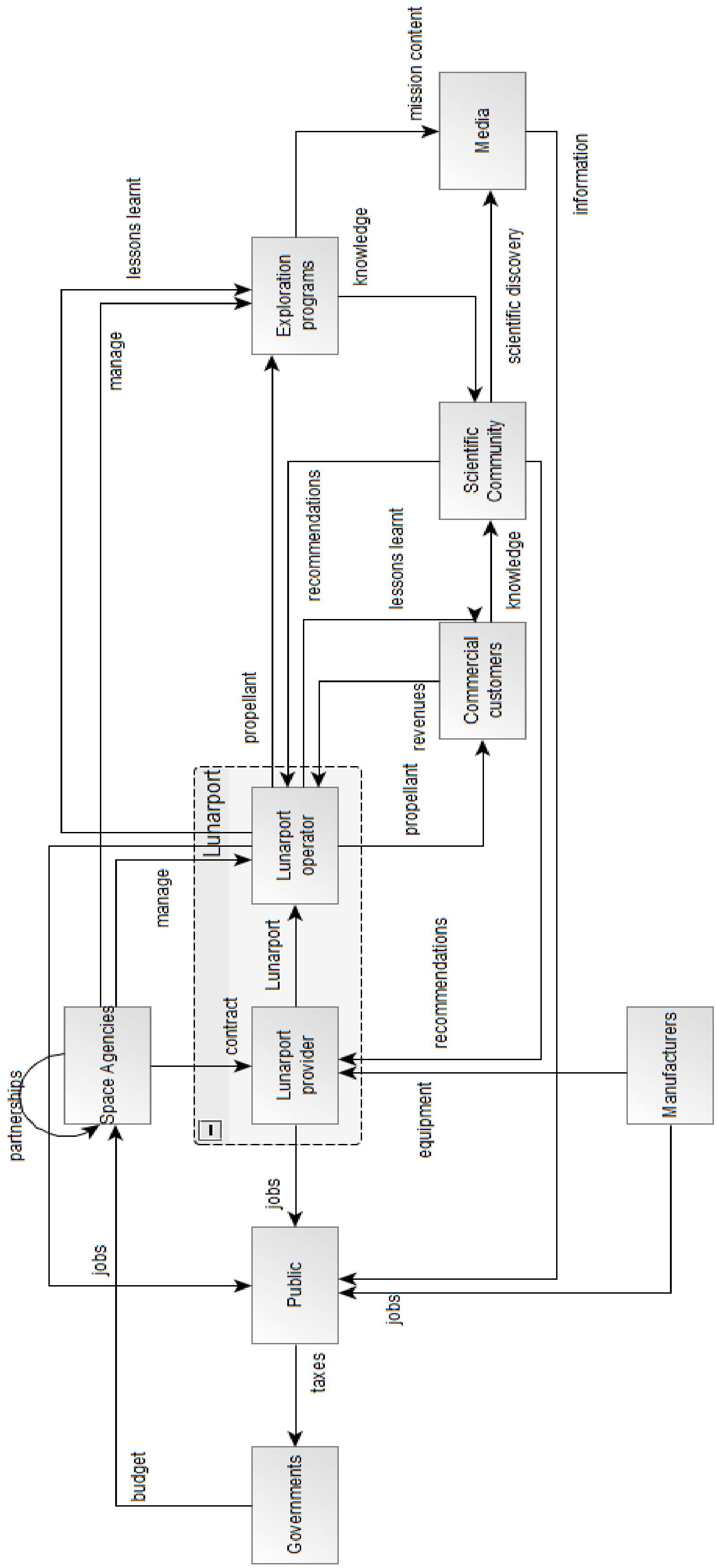


Fig. 2.2.2: System Relationships

2.3 Mission Requirements

The following table lists the requirements and limitations considered for the mission.

Tab. 2.3.1: Mission Requirements

Id.	Objective	Requirement	Type	Origin	Weight	Rationale
1.1 Concept & Development						
1.1.1	Budget	The design, construction and maintenance of the LEEP shall be under $1billion per year (with unused funds of one year available the next)	Constraint	Originating	100	Statement of Work "The design should include a detailed construction and operation/maintenance plan for the ISRU station, main hub, and refueling subsystems, under the constraint of an annual budget of $1billion (with unused funds of one year available the next)"
1.1.2	Concept desirability	The LEEP shall deliver value to the identified beneficiaries	Programmatic	Originating	100	Statement of Work
1.1.3	Concept economic viability	The LEEP shall be economically viable	Constraint	Originating	100	Statement of Work
1.1.4	Concept technical feasibility	The LEEP shall be technically feasible	Constraint	Originating	100	Statement of Work
1.2 Construction						
1.2.1	Gain knowledge for future Mars exploration	The LEEP shall help to gain knowledge and experience for future Mars exploration	Programmatic	Originating	50	Statement of Work "Technologies and operation experiences for accessing and utilizing

						lunar resources are relevant to future Mars exploration"
1.2.2	Human lunar mission	The LEEP could allow a human mission to the Moon	Incentive Award Fee Criterion	Derived	30	
1.2.3	Time to operation	The LEEP shall be operational no later than 2039	Constraint	Originating	100	Statement of Work
1.3 Operation & Maintenance						
1.3.1	Commercial mining	The Lunarport could double as a commercial mining base to allow the moon's resources to be exploited	Incentive Award Fee Criterion	Derived	30	Source: (MailOnline, 2016)
1.3.2	In space fueling competitiveness	The LEEP shall fuel the deep-space traveling rocket at a lower cost than a direct mission	Constraint	Originating	100	Statement of Work
1.3.3	In space fueling for deep-space traveling rocket	The LEEP shall fuel deep-space traveling rocket in cis-lunar orbit	Programmatic	Originating	100	Statement of Work

2.4 Mission Architecture

2.4.1 Functional Architecture

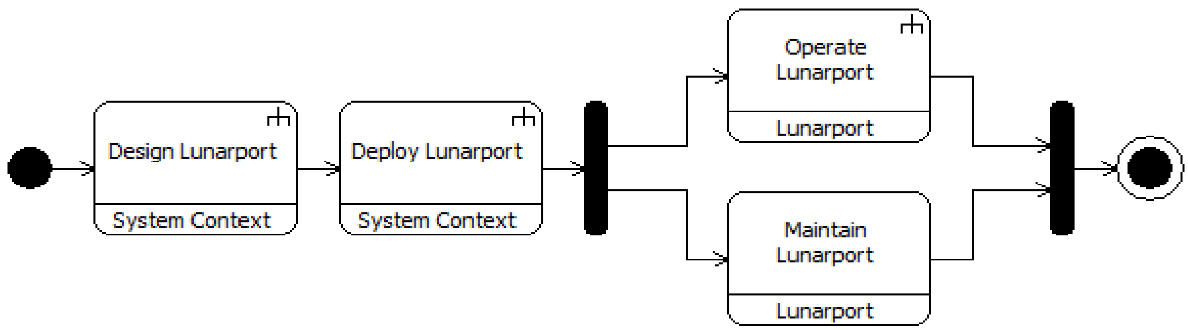


Fig. 2.4.1.1: LEEP Life Cycle (Activity Diagram)

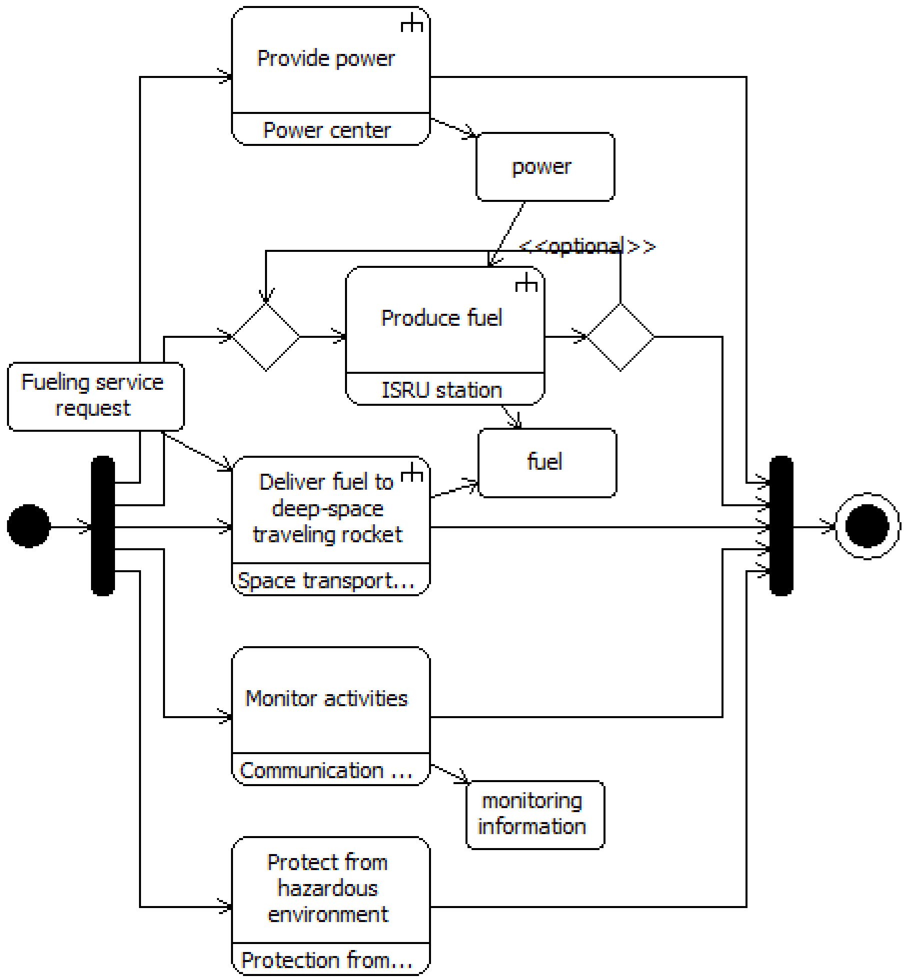


Fig. 2.4.1.2: LEEP Concept of Operations (Activity Diagram)

2.4.2 Physical Architecture

The following picture describes the systems elements of the Lunarport:

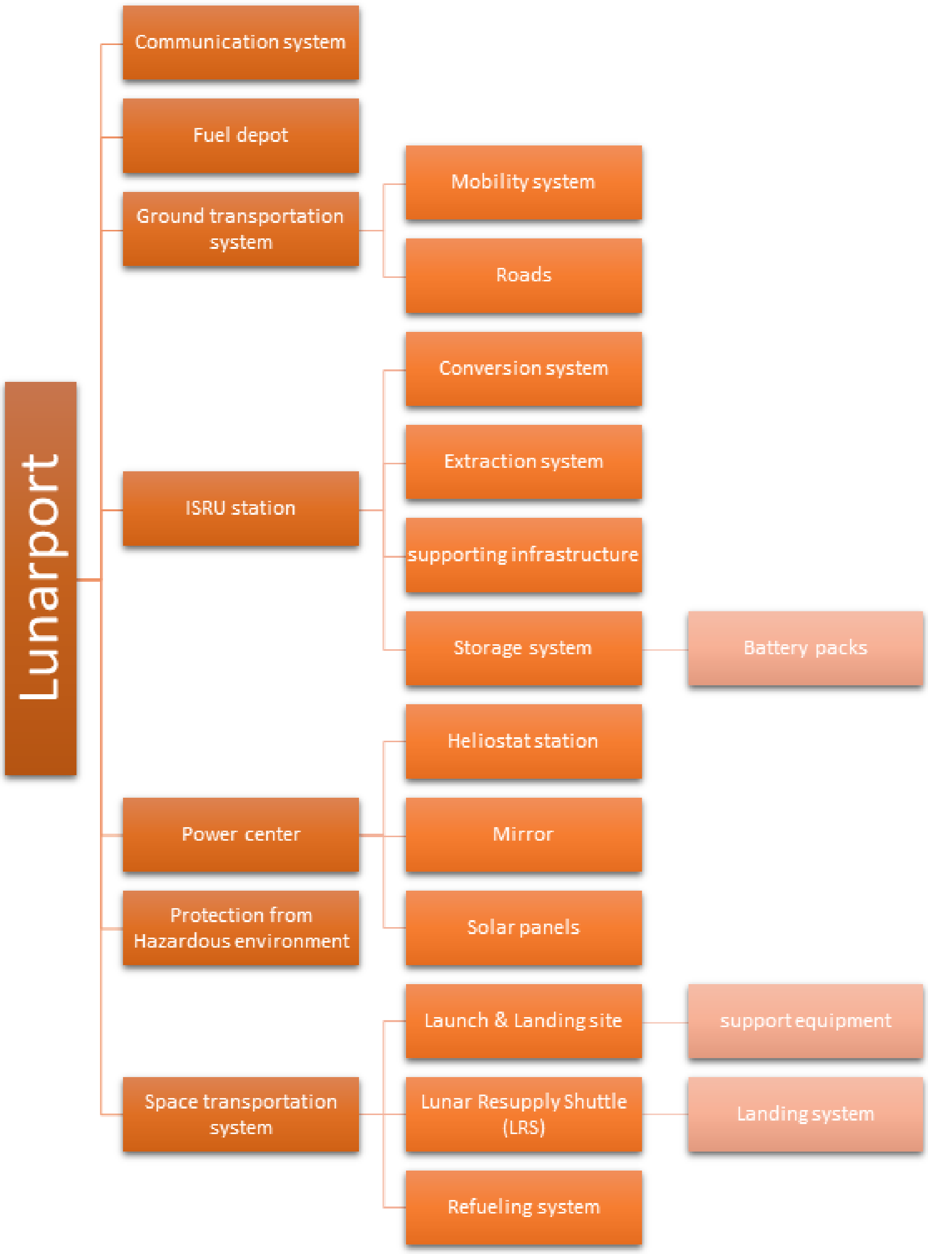


Fig. 2.4.2.1: LEEP System Elements

The following picture depicts the elements part of the enabling systems, for construction and maintenance activities:

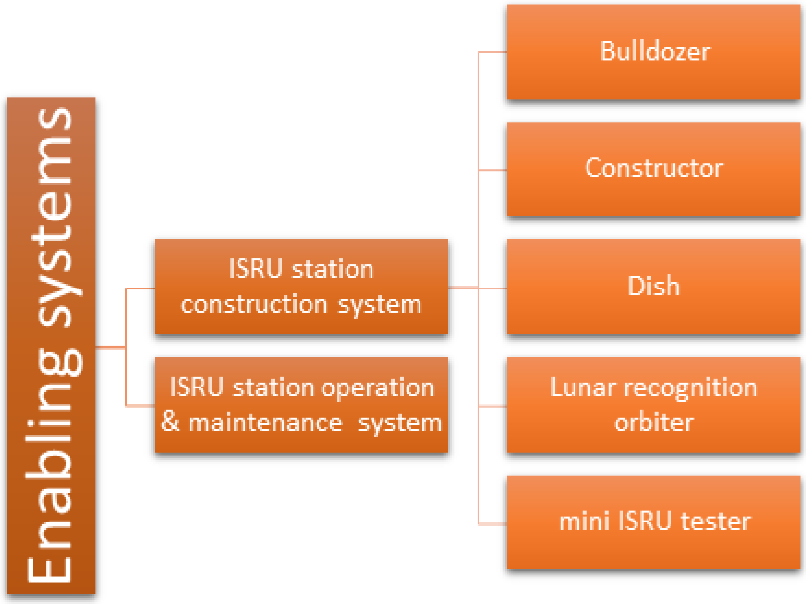


Fig. 2.4.2.2: Elements of Construction and Maintenance Systems

The following table details the descriptions of main systems elements:

Tab. 2.4.2.1: Description of Main System Elements

Id.	System Element	Description
0	System Context	The system context identifies the physical context (the environment and external systems your system interacts with) enabling to specify the system boundary
1	Crew	
2	Deep-space traveling rocket	
3	Enabling systems	
3.1	ISRU station construction system	
3.1.1	Bulldozer	

3.1.2	Constructor	
3.1.3	Dish	
3.1.4	Lunar recognition orbiter	
3.1.5	Mini ISRU tester	
3.2	ISRU station operation & maintenance system	Maintenance, repair, and operations [1] (MRO) involves fixing any sort of mechanical, plumbing, or electrical device should it become out of order or broken (known as repair, unscheduled, casualty or corrective maintenance)
4	Lunarport	A launch and supply station using lunar resources, dubbed Lunarport
4.1	Communication system	In order to maintain constant high definition communication between Earth and the lunar base, a communication system is put in place composed of communication satellites. All communication satellites are in different polar orbits around the Moon
4.1.1	Antenna	
4.1.2	Repeaters	
4.2	Fuel depot	
4.3	Ground transportation system	Road facilities and equipment, including the network, parking spaces
4.3.1	Mobility system	
4.3.2	Roads	
4.4	ISRU station	Self-sustainable ISRU station on the lunar surface
4.4.1	Conversion system	
4.4.2	Extraction system	

4.4.3	supporting infrastructure	
4.4.4	Storage system	Central thermally insulated electrolysis unit. Storage of H2 and O2 in cryogenic form
4.4.4.1	Battery packs	
4.5	Power center	Facility for the generation of electric power
4.5.1	Heliostat station	5 large mirrors installed at an optimal location on the rim of Cabeus crater
4.5.2	Mirror	
4.5.3	Solar panels	
4.6	Protection from Hazardous environment	System to Protect Against Environmental Hazards; structures or other hardware for protection from thermal, radiation, micrometeoroid, environment
4.7	Space transportation system	
4.7.1	Docking system	
4.7.2	Launch & Landing site	Site for launching or landing; a vertical takeoff-vertical landing requires a landing pad
4.7.2.1	support equipment	
4.7.3	Lunar Resupply Shuttle (LRS)	
4.7.3.1	Landing system	
4.7.3.2	Refueling system	
5	Space Agency	

2.5 Mission Design Choices

The table below lists the main trade-offs considered:

Tab. 2.5.1: LEEP Main Trade-Offs

Design Decision	Description	Choice rationale	Option A	Option B	Option C	Option D	Option E	Option F
Resource Transfer to Orbit	What resource to transport from LEEP to Space?	Fuel for RL-10 engines used on EUS and Centaur water ferry vehicles	**H_2 / O_2**	H_2O	Other Volatiles	Regolith	Metals	N_2
Rendezvous	Where should the LEEP LRS intersect with the deep-space craft?	Can rendezvous in multiple locations, fits the customer, and allows flexibility	LEO	**High Earth Elliptical Orbit**	LLO	L1	L2	LRO
Transfer	What to transfer in orbit?	Second type of payload required for orbit; off the shelf LRS; simplified operations	**Propellant**	Tanks	Propulsion Stage			
LEEP location	Where to locate the LEEP?	LCROSS experiment guarantees presence of water in Cabeus crater	North Pole	**South Pole**	Equatorial			
Conversion Location	Where to convert H2O to propellant?	Cryogenic surface temperatures allows zero LOX boil-off, so less water needs to	Orbit	**Surface**	Orbit/ Surface	LRS		

		be mined; less mass must be lifted from lunar surface as the RL-10 fuel-ox mixture ratio may be launched						
Storage Location	Where to store propellant?	Surface shades	Orbit	**Surface**	Orbit/ Surface	LRS		
Maintenance Strategy	How to maintain the facility?	Solar panels limit lifetime to 15 years, must be replaced	Dedicated	**Replacement**	Permanent			
Contractual Arrangement	Which entity will bear the risk and costs of designing and operating LEEP?	Insufficient expected demand for lunar resource on a reasonable timescale to entice private investment	**Public**	Public-Private	Private			
Power Production Location	Where to produce power?	Solar concentrators focus light into the dark crater for power and operations	**Ground**	Orbit				

The following table lists the criteria used for the trade-offs:

Tab. 2.5.2: LEEP Trade-Off Criteria

Criterion	Definition
Construction timeline	How fast can it be built and deployed?
Energy/Propellant output	How many yearly missions can LEEP support?
Fueling capacity	How much additional mass per mission can be sent to Mars with LEEP?
Operation & Maintenance complexity	What are the hardware maintenance and refueling operation costs?
TRL maturity	Does it help gain knowledge and competencies for future Mars exploration?
Technical risks	Does it bring high risks?
Cost	How costly is the development and production?
Partnership	Does it foster partnerships with space agencies and private companies?

The major mission trade-off involves the selection of the location where to convert H2O to fuel and where to store fuel. The following 5 options were identified, cf. Pugh matrix in appendix:

1. ISRU is located on the moon and the LRS is a EUS on the moon. The refuel happens in space. This configuration +30% increase in payload mass.

2. Similar configuration to option 1 but instead of having one EUS, we use multiple (2 to 4) centaur vehicles on the Moon. This configuration has a benefit of 45% of propellant.

3. In this option, the ISRU is in orbit. The Centaurs constitute the LRS system. They bring brings water into space. Electrolysis and fuel creation happen in orbit. This configuration has a negative balance.

4. This configuration is a mix of option 1 and option 2. Centaurs are on the surface and are launching to refuel an EUS which stays in orbit. The EUS tank is being refueled by those Centaur LRS. The EUS can be seen as a PRS - propellant refueling system. PRS is going to its rendezvous orbit to refuel the mission we want to refuel. +70% fuel but needs to extract 2 to 2.5 times faster.

5. This fifth option is mix of option 1, 2 and 3. The ISRU are located in the LRS (Centaurs). The rovers fill the LRS tanks and it prepares just enough propellant to launch. It brings water in orbit to a power station full of solar panels. Then, it starts creating the propellant for the refueling as well as for its return on the Moon. The benefit of this +70% of more fuel but triple the extraction rate. This solution also uses an EUS in orbit as well. Disadvantages: if a Centaur LRS breaks apart, you lose two systems. The benefits is having no need of an ISRU on the Moon.

Tab. 2.5.3: Final Comparison of Shortlisted Options

Options	Limitations / Challenges	Advantages	Fuel
1 - ISRU on the Moon; EUS as LRS	EUS expensive, no redundancy, 120 tons prop/shuttle, maintaining ISRU on the moon	One docking only	+30%
2 - ISRU on the Moon; multiple centaur as LRS.	Need more centaur, 20 tons per shuttle, maintaining ISRU on the moon, multiple dockings	Centaur cheaper, redundancy	+45%
3 - ISRU in orbit; multiple centaur as LRS	Complex system	Microgravity helps	<0%
4 - One centaur on the Moon and one EUS in orbit	Multiple dockings, IRSU on the moon, complex system, needs a faster extraction rate	Redundancy, service availability	+70%
5 - ISRU inside the LRS	New technology, complex system, needs a faster extraction rate	No ISRU on the Moon, flexible and agile, multiple facet rocket	+70%

3. Engineering and Project Components

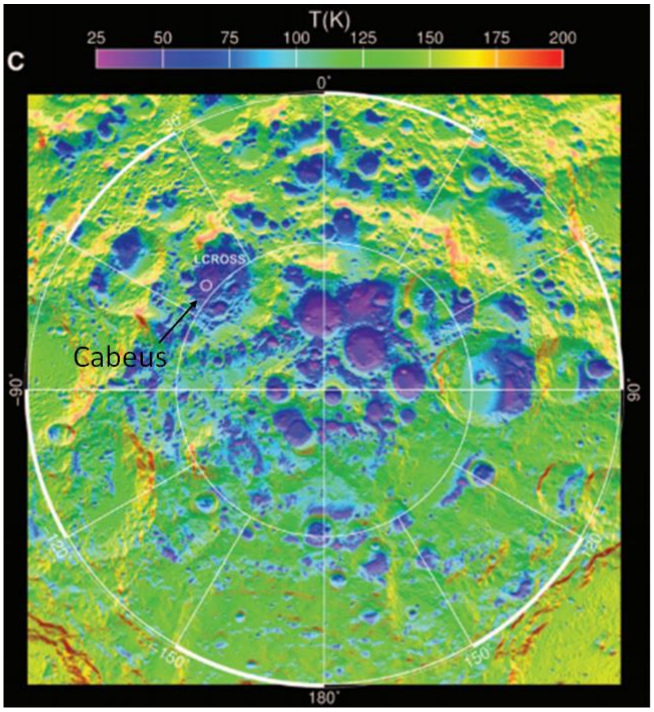
The lunar base is comprised of two main parts: a small crater rim station and the larger main base inside the crater.

1. **Rim Station:** Two rim stations consists of mirrors that will direct sunlight into the crater to power the equipment and rovers, as well as communications equipment for telecom with Earth.
2. **Crater Main Base:** The larger crater main base is the site of the majority of the lunar operations. The key operations are prospecting, extractions, ISRU conversion, and construction of the launch pad and roads. The first payload to land inside the crater will demonstrate the technology and capabilities of the full system on a small scale.

This section details the overview of the lunar base and Lunar Resupply Spacecraft (LRS). Further details can be found on the Site Selection in Section 3.1, Structural & Operations Design in Section 3.2, Electrolysis Unit in 3.3, Power in 3.4, Transportation and Launch Pad in 3.5, LRS in 3.6, and Communications in 3.7.

3.1 Analysis of LEEP Site Selection

The most critical aspect of the mining site selection is the availability of resources to mine. The only location with a proven and quantified measurement of H_2O ice is in Cabeus crater on the south pole of the Moon (Fig 3.1.1), which was verified by the Lunar Crater Observation and Sensing Satellite (LCROSS) when a planned impact launched a cloud of debris that was measured to have 4% H_2O content by mass (Paige et al., 2010). The reason water ice is found in Cabeus crater is the fact that its eastern region is permanently shadowed from sunlight (Fig. 3.1.1). This creates an extremely cold environment that acts as a cold trap for volatiles (such as water), which build up to quantities useful for mining over geologic timescales. The supportive figures have been shown on the next page.



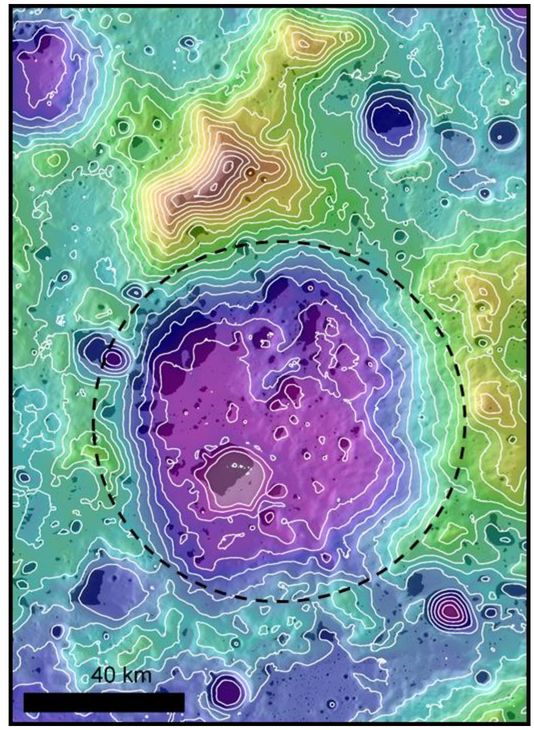


Fig. 3.1.1: Left: Temperature map of the Lunar South Pole. Outer white circle marks 80 Deg. latitude (Paige et al., 2010). Right: A topographic map of Cabeus crater (dashed outline included to aid the eye) constructed from gridded LOLA data. Cool colors are low (down to -4 km global elevation) and hot colors are high (up to +4 km global elevation). Dark patches are permanently shadowed.

While there are other permanently shadowed regions on the moon, the existence of H_2O in these locations (such as Shackleton crater) is controversial (Mitrofanov et al., 2010; Zuber et al., 2012). Therefore, in order to maximize the potential for mission success, we selected the **only** site on the Moon where H_2O has been definitively detected in amounts suitable for large-scale extraction. We select a mining location 10 km from the LCROSS impact site in order to be close to the known H2O ice location, but also far enough from the impact such that the ice resources have not been destroyed by thermal activity from the impact (Hayne et al., 2010). We also optimize for selecting a flat location within the crater in order to ease mining operations.

We also examined schemes for microwave extraction of hydrogen adsorbed to the regolith near the lunar equator due to implantation from solar wind (Thompson, 2009) or directly collecting hydrogen from the solar wind near the equator. However, the quantity of hydrogen implanted in the regolith is unknown and technology for its extraction remains unproven. In terms of direct collection of hydrogen from the solar wind, the hydrogen flux is too low (~1 g km^{-1} day^{-1}) to be viable.

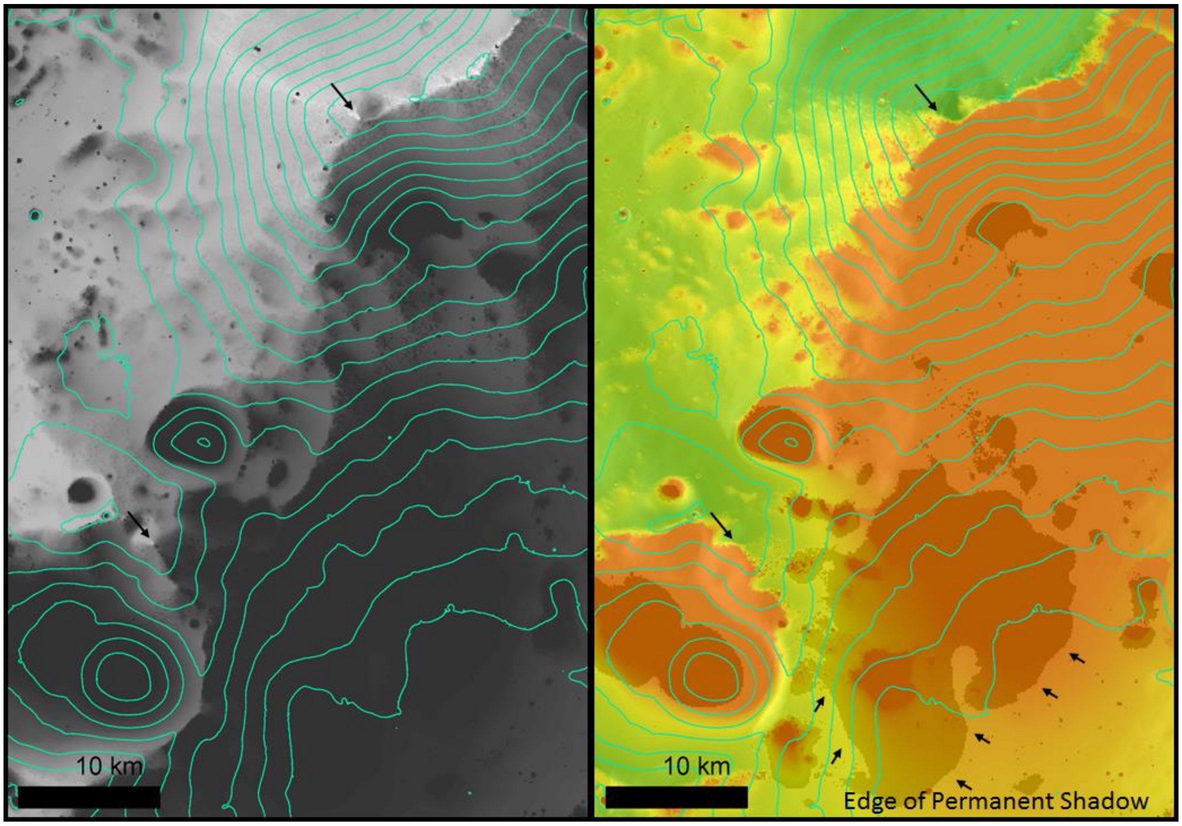


Fig. 3.1.2: Both: Longer black arrows indicate main base and auxiliary base locations (see Figure 3.1.3). Cyan lines indicate 500 m elevation contours. Left: A map of average sunlight received over a lunar year derived from LOLA topography. Range 0% (black) to 96% (white). Right: A map of average line-of-sight over a lunar year derived from LOLA topography. Range 0% (red) to 95% (green). The edge of the permanently shadowed region is indicated by small black arrows.

The LCROSS impact indicated 4% water ice by mass (Paige et al., 2010) in the large eastern shadowed region in Cabeus crater. We therefore select a site close to the LCROSS impact site (Figure 3.1.2). Since the region is permanently shadowed, however, directing communications and power to the mining site are of concern. The most efficient way to conduct mining operations is to route power and communications into the mining base from a location with solar power and line-of-sight to the mining site (See Sec. 3.4) because transporting mined H_2O resources to a sunlit location is slow (since rovers travel at ~100 m/hr, maximum). Additionally, transporting the ice into sunlit areas carries the danger of sublimating the ice and losing the mined resources.

We therefore select a mining location in the permanently shadowed region that is close to locations on the rim that (i) receive consistent sunlight, (ii) have consistent line-of-sight to Earth, and (iii) are flat enough to safely land assets. Fig. 3.1.2 shows maps characterizing these parameters. We select two rim stations, as shown in Fig. 3.1.3: one main rim station and one auxiliary rim station. The main rim station receives sunlight 78% of the time and has line-of-sight to Earth 79% of the time and has a region approximately 300 m in diameter with slopes <5%. The auxiliary station receives sunlight 93% of the time and has line-of-sight to Earth for communications 95% of the time.[4] The communication and sunlight blackout times are cyclic throughout the 27-day lunar sidereal rotation period, such that power and communications are not available to the base for only ~1 day every 27 days. Phasing of the blackout times between the two rim stations also means that power delivery to the mining operations is actually greater than 93%. The main station is 10 km from the proposed mining site, and so is able to provide more power than the auxiliary station, which is 30 km away. However, the auxiliary site provides more continuous coverage than the main site and allows low (~10%) power operations; this is enough to protect the mining equipment from large temperature swings and long periods of blackout (see Sec. 3.4).

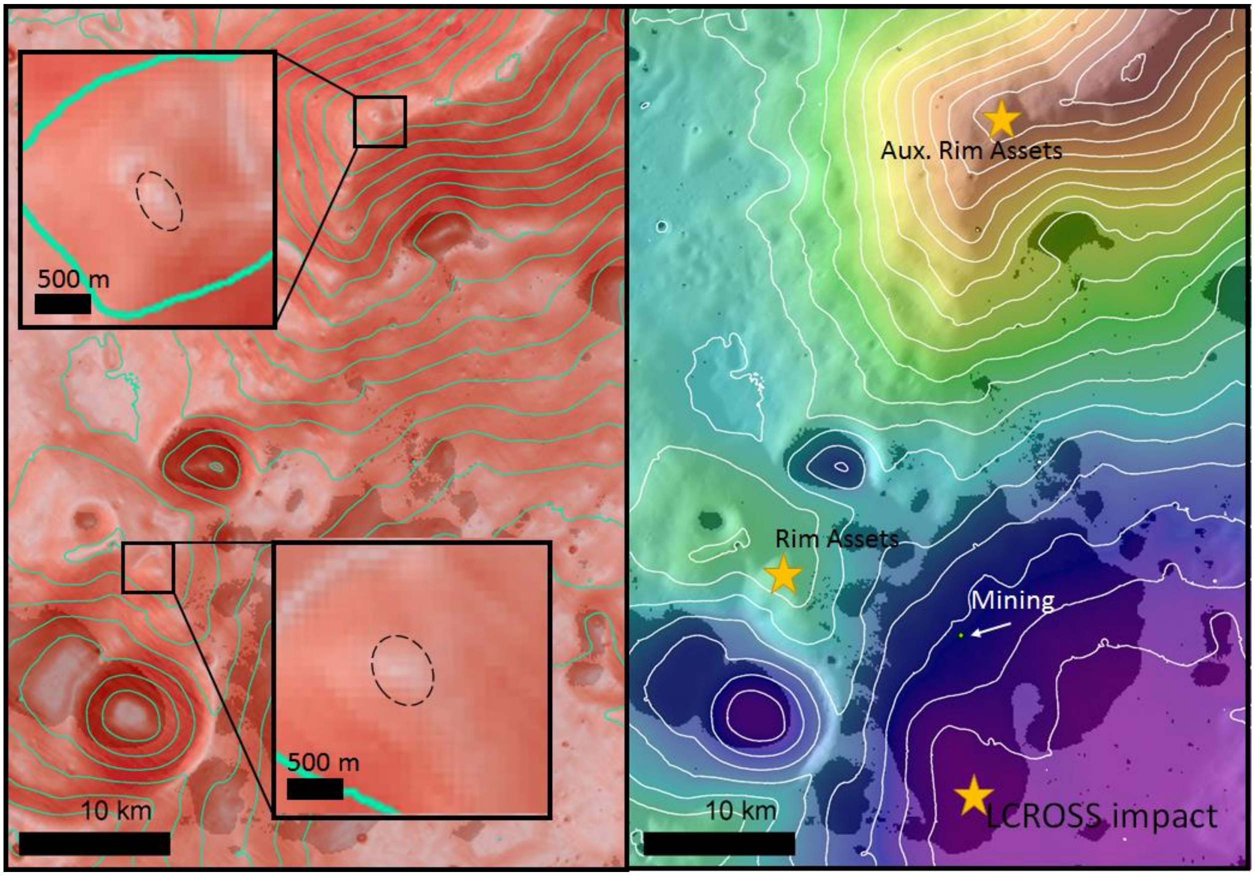


Fig. 3.1.3: Left: Map of slope; range: white (0% slope) to dark red (30% slope). Dark boxes are zoomed-in regions on the rim station locations. Dashed ovals indicated landing regions with low slope. Right: A topographic map of the mining site and rim stations, marked with stars. The LCROSS impact is also marked. The mining station is marked as a green dot. The dark regions are regions of permanent shadow.

[4] Note: Values quoted for line-of-sight and illumination for both rim locations have ~10% uncertainty.

3.2 Lunar Base Structural Design

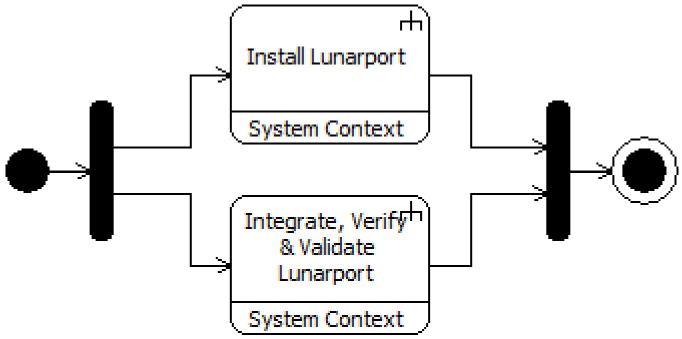


Fig. 3.2.1: Deploy LEEP

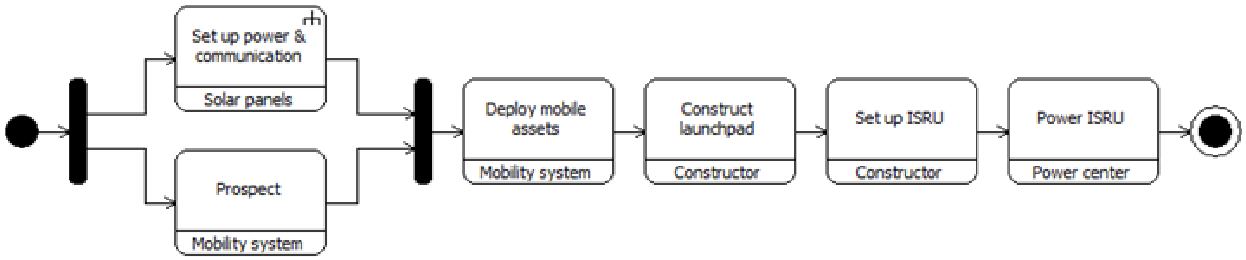


Fig. 3.2.2: Install LEEP (Activity Diagram)

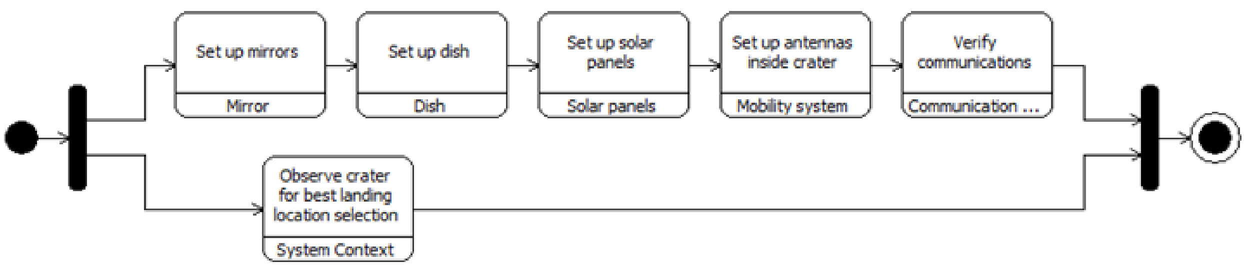


Fig. 3.2.3: Set up Power (Activity Diagram)

3.2.1 Lunar Base Components

For all of the components, an effort was made to make them as modular as possible and to use as much heritage as possible. Where possible, technologies at TRL-6 are preferred. Modularity is employed throughout the designs to increase engineer familiarity with subsystems and to allow for upgrades in future missions.

The following table describes the major components of the base station and provides insight into design considerations.

Tab. 3.2.1.1: Description of Major Base Station Components

Component	Number	Mass (Per Unit)	Description	Design Considerations
Mirrors (rim)	5 mirrors per station in 2 locations	250kg	Focuses light into shadowed crater regions to provide power for extractors and illuminate landing area. Two locations used to keep illumination time >93% of time	Provide the necessary solar power to all crater rovers and electrolysis ISRU unit: 1. Mirrors are rover-mounted. Brings mirror from landing location to deployment 2. Deploys 10 kW solar panels for mirror control; see below
Solar panels (crater)	1 **500m²**	167kg	Solar panels used to power the ISRU electrolysis	Large PV farm. Doesn't need to move because mirror incidence angle changes very little
Rim Communication Solar Panels	1 **10m²**	27kg	Small solar panels used to power the infrastructure at the rim, including moving the mirrors and powering communications	Deploy from the mirror rovers
Electrolysis Unit (In-Situ Resource Utilization unit)	1	4000kg Max	Electrolyzes pure water: 1. Pressure-fed system to PEM Electrolyzer with no moving parts except solenoid valves 2. Vapor feed to the Electrolyzer membrane prevents two-phase flow considerations	Operates at 70 kW. Assuming 50% conversion efficiency, electrolysis occurs at 35 kW and generates 35 kW which heats the unit (a good thing in frigid environment) Huge mass margins for expected engineering difficulties including: 1. Thermal 2. Pressure 3. Liquefaction Process

Lunar Landing System	2	1000kg	Lander for 2-5 rovers, spare parts	4. Rover protection Details: see section 4.1.2
Prospecting rover	2	500kg	Rover includes a drill payload and scientific instruments that can determine the constituents of samples taken	Design is based largely on the Lunar Resource Prospector with modifications for a LIDAR advanced sensory system All energy from solar panels from light beamed from rim
Construction rover	2	500kg	Multi-function, modular rover. Prepares base and does work. Swappable tractor trailer attachments allow different work to be done	<u>Attachment 1</u>: Bulldozer - Clears loose regolith for take-off vehicles <u>Attachment 2</u>: Robotic Arm - Attaches hoses and power cabling. Tends to damaged vehicles Modular rover architecture takes inspiration and heritage from the NASA Goddard Robotic Refueling Mission, which gives the ISS robotic arm a "toolbox" and the NASA/Caltech JPL ATHLETE robot, which can switch out tools and reconfigure to complete various tasks
Extractor rover	12	500kg	Rover for removing regolith and transporting it to the Electrolysis Unit	Rover is based on the Honeybee technology and uses four of their TRL-6 drills for extracting regolith
Refueling Assistant Tool	1	100kg	System for connecting the Electrolysis Unit to the launcher for the refueling process	Must carefully insulate to prevent boil off or line freezing; takes lessons learned from on-orbit cryogenic refueling; has electromagnetic dust mitigation strategy

3.2.2 Lunar Base Set-Up

The lunar base is set up over the course of four payload missions that land in two places: around the rim (1) and in the crater (3). The table below details the arrival sequence, contents, and key operations of each mission.

Tab. 3.2.2.1: Arrival Sequence and Key Operations for Construction

Id.	Payload Arrival	Location	Contents	Key Operations
1	Power Set Up	Rim	Four mirrors, Communications (including beacons), Small solar panels (for local power, moving mirrors, and communications)	Set up mirrors Verify communications Observe crater for best landing location of next payload arrival
2	Shakedown	Crater	Lunar Landing System (LLS) with: 1. Prospector (2) 2. Constructor (2) 3. ISRU Test Unit	Deploy all mobile assets Use mini ISRU tester to verify communication and rim power Mini ISRU, send water sample, check out all the technology If fails, go elsewhere; tech fails, know
3	Production	Crater	Lunar Landing System (LLS) with ISRU main unit and solar farm Lunar Landing System (LLS) with Extractor units (6) Refueling Assistant Tool	Extractor rovers begin operations and verify water extraction Solar panels for the ISRU deploy
4	Expansion to Full Capacity	Crater	Lunar Landing System (LLS) with Extractor units (6)	Bulldozer builds launch pad near the location of maximum ice found by the prospector

				Refueling Assistant robot deploys between the ISRU (in preparation for LRS arrival) and the launch pad site Bulldozer is available to repack the landing pad for subsequent landings

3.2.3 Trade in Lunar Base Design

In considering designs for the lunar base, many trades were conducted. The tables below summarize the major decision points and considerations. Some aspects could be modified based on the success of earlier phases of the mission.

Tab. 3.2.3.1: Trade Space for Lunar Base Design Decisions

Trade	Options	Considerations	Decision
Source of rover power	1. RTG / (nuclear) 2. Solar using concentrating mirrors 3. Solar (on rim) and beam using microwave/laser	Desire to be human-friendly Laser is not proven/not high TRL Optimal microwave frequency couples with the lunar regolith which excites the particles and changes phase Laser and microwave suffer 2x inefficiency from photon-electron conversion as compared to reflecting mirrors	Solar using concentrating mirrors
Electrolysis (ISRU) location	1. Ground 2. In-orbit	See Table 3.2.3-2	Ground
Communications	1. Deep Space Network 2. Commercial provider	Desire for regular rover communications Concern about not getting priority on DSN for duration of mission Commercial is also ~200x cheaper than DSN	Commercial provider

Rover processing location	1. On Earth 2. In lunar orbit 3. On lunar surface	1. Processing power and speed is faster on Earth Lunar surface processing would support increasing the level of autonomy	On Earth * but with extra mass on the Lunar Landing System, SpaceCube could be tested to augment future on-moon processing
Number of and design of extraction rovers	1. 10 rovers, 4 drills per rover 2. 12 rovers, 4 drills per rover 3. 6 rovers, 8 drills per rover 4. 24 rovers, 2 drills per rover	Number of rovers/drill needed was calculated based on the needed propellant Configuration considered heritage, redundancy, cost, and replaceability See Section 4.2.2	12 rovers, 4 drills per rover
Crater location	1. North pole 2. South pole 3. Equator	Given 4% water assumption at two craters in problem statement Papers show equator does not have guaranteed water See Section 3.1	South pole, Cabeus Crater
Location of mirrors along rim	1. One location 2. Multiple locations	No single locations had complete solar coverage High number of rovers assumption and power needs necessitated option for additional power See Section 3.4	Two locations, separate

Due to the complexity of the Electrolysis (ISRU) location decision, a full trade is shown below.

Tab. 3.2.3.2: Trade Space for Electrolysis (ISRU) Location Decision

Trade-off Type	Orbit	Ground (Our decision)	Hopper
Power Available	High, Low cost	Low, high cost	Batteries only
Ground Infrastructures	Extractors, Landing pad	Extractors, Landing pad, IRSO	None
LH_2 boil off	Yes (or refrigerate)	Yes but less than in orbit	Yes (or refrigerate)
LOX boil off	Yes (or refrigerate)	No, must heat or will freeze	No, must heat or will freeze
Total Launch Mass from Lunar Surface	All water + unnecessary boil-off	Correct fuel-oxidizer mix means less launch mass	Several tons of battery dead weight
Ease of Electrolysis - Gravity	Harder	Easier	Easier
Ease of Electrolysis - Temperature	Easier	Harder	Harder
Telecoms	Easier	Harder	Harder
Number of launches from extraction point	4	5 or 4	20
Heat rejection	Radiator mass required	Conduction to regolith	Conduction to regolith
LRS cryogenic equipment	No	Yes	Yes

These trades helped inform the baseline mission design and provides inputs to other subsystems' design. Over the course of many iterations between sub-teams, these decisions were considered the least risky, most innovative, and/or most useful for decades of operations.

3.3 ISRU Drilling, Extraction, and Storage Technology

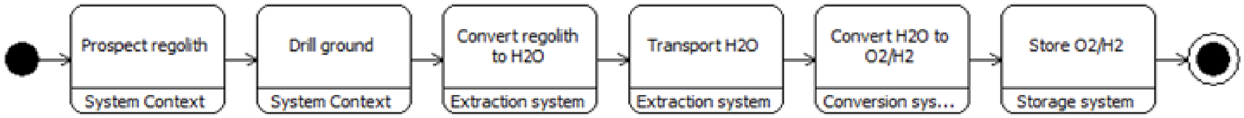


Fig. 3.3.1: Produce Fuel (Activity Diagram)

Our ISRU is done in three phases. It involves the extraction of pure water from the bake-out of icy lunar regolith containing multiple species of volatiles, its transport to a central thermally insulated electrolysis unit, and storage of the H_2 and O_2 products in cryogenic form. Primary difficulties from the environment include that of water extraction from the rock-hard frozen regolith and keeping units at the necessary operation temperature and pressure while in a cryogenic environment (40-80 K, depending on the illumination of the crater walls).

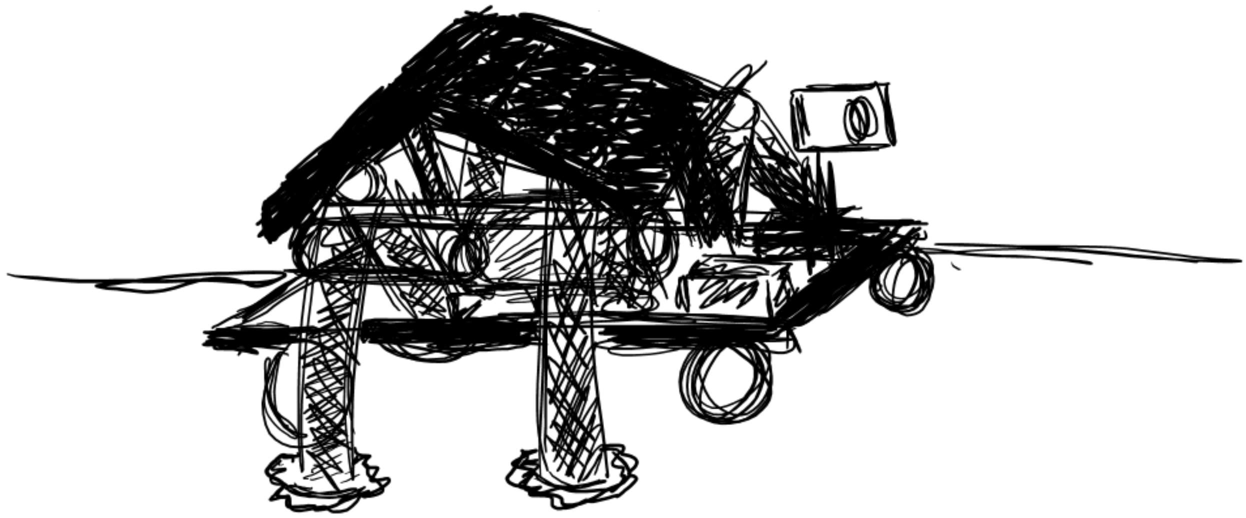


Fig. 3.3.2: Extractor Unit

Extractor unit depicted in Figure 3.3.2 is drilling into the icy regolith and extracting liquid H_2O using 4 Honeybee Robotics PVEx.[5][6] The extractor unit utilizes the Planetary Volatiles Extractor (PVEx) extraction core of Honeybee Robotics (Zacny et al). All storage of cryogenic propellant is conducted on the LRS vehicles. For a discussion of propellant storage and ZBO technology, see Section 3.6.

[5] Zacny, Kris. Lunar Prospecting and Mining (Presentation)

[6] Honeybee Robotics (Zacny et al), Planetary Volatiles Extractor (PVEx) for In Situ Resource Utilization (ISRU)

3.4 Lunar Power Station Design

Based on data from the LCROSS mission, and taking into account the assumption given in the problem statement that Cabeus crater contains in its permanently shadowed regions 4% water ice by mass, we quickly concluded that the lunar base station had to be located in a permanently shadowed region of Cabeus crater. Further, the aspect of H_2O extraction most likely to choke the extraction rate is rover speed, so that location of the processing facility as close as possible to the extraction site is imperative. This location, for obvious reasons, ruled out the direct use of solar panels to power the base. We considered four alternatives to provide power in the permanently shadowed regions of Cabeus crater: solar concentration (Heliostat), microwave and laser beaming, and a small nuclear reactor.

Tab. 3.4.1: Power Supply Trade-Off

System	TRL	Mass	Problems	Advantages	Rim Stations?	Feasibility
Heliostat	5-6	Medium	Expense of large mirrors Complex mirror control system Warming of regolith will cause outgassing unless heating is periodic	+ Illuminates operation area + Provides power to all operating rovers w/out secondary transmission	Yes	Medium
Microwave Beaming	3-4	Medium	Microwave frequencies strongly couple with regolith and may outgas volatiles Undemonstrated in space environment Secondary beaming to extraction rovers or they are RTG-powered	+ Phased arrays allow localized beaming	Yes	Low
Laser Beaming	4-5	High	Laser cooling requirements	+ PV farm in crater may be tuned to laser frequency for higher efficiency	Yes	Medium

			Thermal cycling of laser equipment Secondary beaming to extraction rovers or they are RTG-powered			
100kW Nuclear Reactor (e.g. SNAP-8)	9	Low	Must launch weapons-grade uranium Extreme political cost	+ No power beaming necessary	No	Low

The option of a small nuclear reactor was quickly discounted as it may cost as much as $500,000/W delivered [SMAD] (an estimate presumably including political costs), making a power available of 100 kW completely infeasible. Among the power beaming options, microwave beaming was quickly discounted because the microwave frequency will strongly couple with nanophase Fe0 in the regolith and quickly bake-out subsurface volatiles. Then between laser beaming and using a heliostat system, the narrow laser beam means that each extraction rover must be radioisotope powered or a complicated secondary beaming system must be used (with compounding losses).

The key advantage of a heliostat system is that all extraction rovers may be primarily solar powered (with batteries for load-levelling and high power tasks such as volatile extraction). In addition, the extraction area is well-lit for ease of operations. There are still volatile bake-out concerns from this strategy. If the heliostat remains focused in one area indefinitely, then all volatiles will bake-out from this region due to thermal diffusion. However, if the heliostat focuses are moved around to provide periodic heating, then the thermal penetration depth becomes constant (depending on the heating frequency) so that volatile bake-out below a certain depth can be avoided.

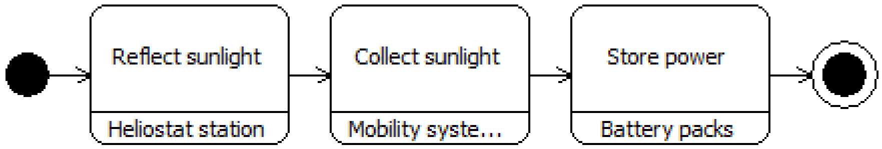


Fig. 3.4.1: Provide Power (Activity Diagram)

The H_2O ice is located in a permanently shadowed region on the floor of Cabeus crater, which makes powering mining operations difficult. In order to power to the mining operations, a power station consisting of five heliostat (large mirror) stations will be installed at an optimal location on the rim of Cabeus crater (detailed in Section 3.1). The mirrors reflect sunlight into the crater in order to provide power and are similar to terrestrial heliostat systems installed on Earth (Figure 3.4.1). Details of heliostat deployment are provided in Section 4.1.

Fig. 3.4.2: An example of a heliostat in Rjukan, Norway.[7] Left: Mirrors reflect light from mountaintop. Right: Close view of the heliostat station.[8]

We select the "TransFormers" heliostat design proposed by Stoica et al. (2014), which consists of an ultra-light ~100 nm-thick sheet that autonomously unfolds to form a 40 m diameter circular surface (~1200 m^2; Figure 3.4.2). The entire mirror has a mass of 100 kg, with an additional 150 kg support structure, base, and two 30 W motors that allow the mirror to rotate and pivot that are powered by a 1 m^2 solar panel. Each heliostat unit thus has a mass of 250 kg. The motors allow the heliostats to reflect light into different locations within the crater and are commanded through a link to the communications array that is also deployed on the crater rim (Section 4.1). The communications array on the rim is powered by a 10 m^2 solar panel, which provides the ~4 kW power draw of the communications antenna.

[7] Image source: http://www.dailymail.co.uk/news/article-2474800/Norwegian-town-Rjukan-enjoys-winter-sunlight-time-history-using-heliostats.html

[8] Image source: http://sourceable.net/century-old-engineering-idea-brings-sun-to-mountain-town

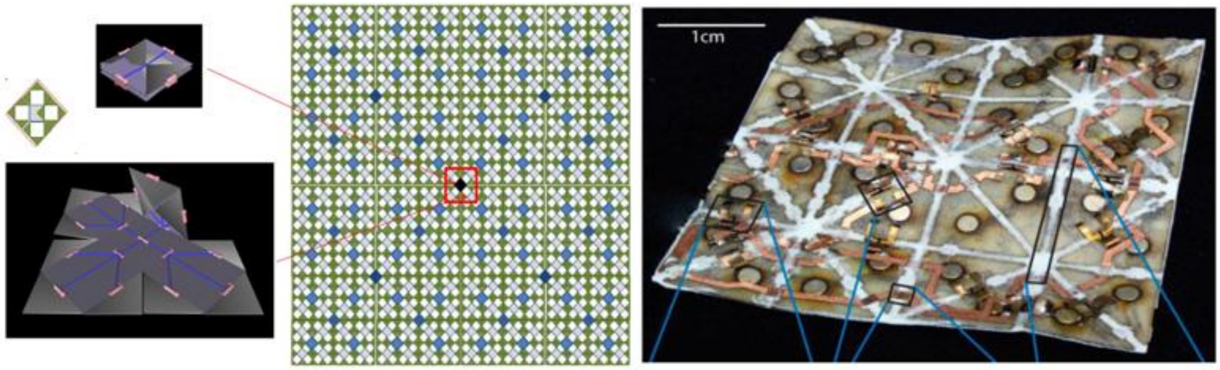


Fig. 3.4.3: Left: The programmable autonomous fractal origami unfolding system designed by Stoica et al. (2014). Right: an implementation of programmable folding from Hawkes (2009).

Each heliostat provides 300 W m^{-2} in a 34 m-diameter circular spotlight from a distance of 10 km to the crater floor (Stoica et al., 2014). This power is collected by a solar panel array that is used to power the electrolysis unit and communications from the floor to the rim. A 500 m^2 array of solar panels oriented toward the mirrors collecting with an efficiency of 30% collects 25 kW of power that is used to power the electrolysis unit and communications antenna on the crater floor. All rovers driving within the spotlighted region are equipped with 3.5 m-diameter (10 m^2) solar panel oriented toward the heliostats and collect 1 kW of power (assuming a collecting efficiency of 30%). This is sufficient to power a rover with Mars Science Laboratory-like capabilities (Stoica et al., 2014), such as the prospector rover and the construction rover, as well as the 1 kW draw from the mining rovers (Zacny et al., 2012).

The spotlights are oriented linearly and slowly rotated around the solar panel on the crater floor, such that the solar panels are always illuminated. With a conductivity of 1.5 $Wm^{-1}K^{-1}$ in the upper 2 cm n the lunar soil (Kring, 2006), the thermal skin depth can be kept to ~2 cm by rotating the spotlights such that no region is continuously illuminated for more than ~3 days. This protects the vast majority of the ice resources from sublimation.[9] The radius of the circle swept out by the line of spotlights is 157 m, which allows the rovers access to traverse a total area of 77,000 m^2 without losing access to continuous power. This provides illuminated (powered) access to 3600 metric tons of H_2O ice, assuming an average 4% ice content by mass as measured by the Lunar Crater Observation and Sensing Satellite (LCROSS; Colaprete et al., 2010). Mining rover traverses and nominal operations are powered by solar energy, with some energy transferred to 100 kg battery packs on board the mining rovers that allow the rovers to have 7 kW of pulsed power for drilling; this is sufficient power for the nominal, four drill, mining rover design (Zacny et al., 2012). The battery packs also provide power during times of darkness, when the heliostats lose line-of-sight to the sun.

[9] Kring (2006): Parameters of Lunar Soils. Presented at Lunar Exploration Initiative.

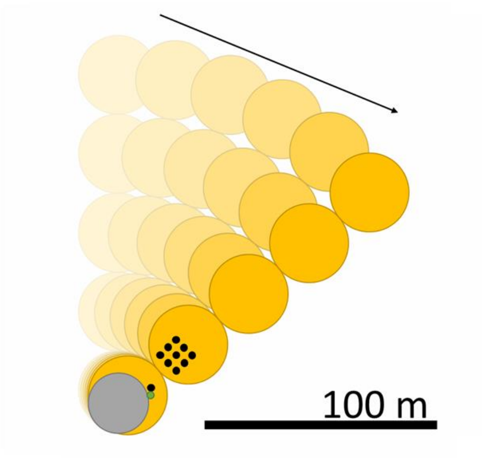


Fig. 3.4.4: The spotlights are aligned linearly and slowly rotated around the solar panel on the crater floor. Legend: yellow circles are spotlights, gray is solar array, and green is electrolysis unit, black are rovers. All drawn to scale.

The TransFormers heliostat system presented in Stoica et al. (2014) was at Technology Readiness Level (TRL) 3 and received funding from NASA Innovative Advanced Concepts (NIAC) to continue development. An alternative, high TRL option for unfolding smaller (10 m diameter) mirrors are also available from modified Ultraflex folding solar arrays shown in figure 3.4.5.

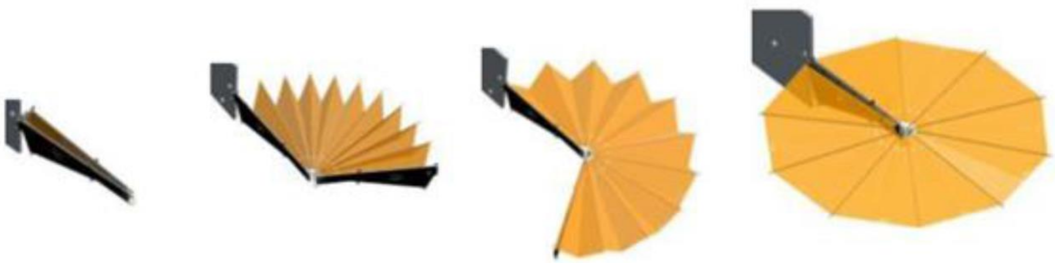


Fig. 3.4.5: The autonomous unfolding of the UltraFlex system (modified from Figure 4.4 of Stoica et al. (2014) and the UltraFlex System Fact Sheet from Orbital ATK).

3.5 Transportation and Launch Pad Design

The launch pad is created after the first payload arrives. The default configuration of the construction rover consists of a tractor with a bulldozer attachment, which flattens a piece of lunar land and removes loose lunar regolith from that area. This minimizes lunar soil flying around and clogging the seals and valves of pumping equipment when Centaur lands or takes off.

3.6 LRS Design

3.6.1 The Vehicle

The Lunar Refuel Shuttle (LRS) is the vehicle that will take fuel from the surface of the moon to a spacecraft mission in orbit around the Earth that requires fuel. Many options were considered when looking for practical vehicles to perform the transfer and refueling task. We considered building an LRS from scratch, but ruled it out due to cost and qualification requirements. This vehicle would have to be very large to make refueling efforts as simple as possible, and with as few trips as possible. Many options were immediately discounted given by initial requirement of the use of hydrogen and oxygen as propellant. The main three options in the trade space are the Exploration Upper Stage, the Advanced Cryogenic Evolved Upper Stage (ACES), and the Centaur Upper stage.

The Exploration Upper Stage is certainly the optimal vehicle for this effort, given that this mission is designed to the task of refueling another EUS on a path to Mars. However, due to the roughly $500M cost restraints it is not monetarily reasonable. The next option, the ACES vehicle, looked to be a good option via some preliminary reports of potential payload capability. However, there was not enough publically available information to characterize the launch vehicle as a possible LRS. The Centaur is an incredibly capable vehicle, which has been in service for over 50 years. This means that the vehicle is well understood and has a very high reliability rate. Given the entry, descent, and landing properties of a moon landing, the single engine version will suffice. This system has a dry mass of 2,247 kg and a maximum fuel mass of 20,830 kg and an I_{SP} of 450.5 seconds. The total ΔV capability of this system without additional mass is 10.304 km/s. This is more than enough to leave the surface of the moon, enter an elliptical orbit around Earth, and then transfer back to the moon and land. The Centaur upper stage has been shown on the next page.

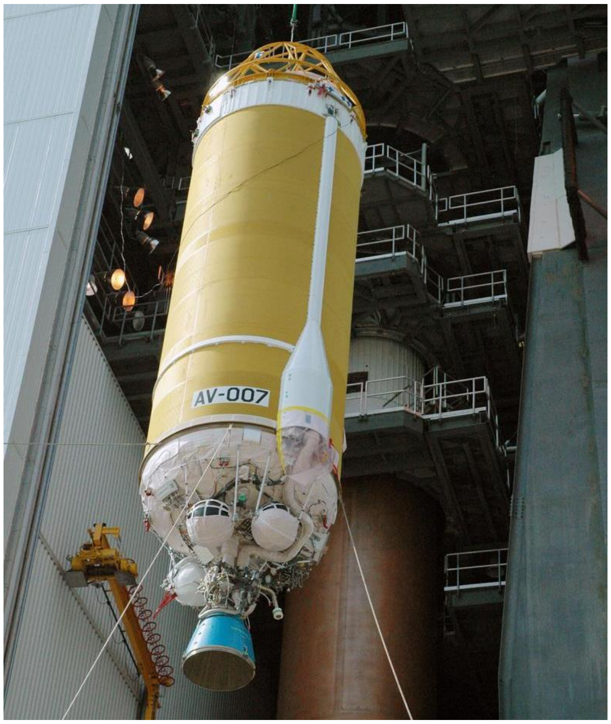


Fig. 3.6.1.1: Centaur Upper Stage

In order to use the Centaur vehicle, we must first make a number of modifications to the vehicle in order to land, land, and repeat. To qualify this vehicle for a mission of this architecture, the Centaur will need to have additional GNC capability, sufficient avionics modifications, thermal protection modifications, landing modifications, and ACDS modifications. The vehicle will require a new suite of sensors to be outfitted for the GNC/ACDS subsystems. The rocket will require, at minimum, an IMU, two GPS receivers (only to be used when in transfer orbit), a radar altimeter, and an advanced video guidance system (AVGS). The IMU is required for taking inertial measurements of rotational rate and acceleration. The differential GPS system is used for the docking/berthing procedure when refueling occurs. The differential aspect, where one receiver is at the top, and another is at the bottom, allows you to get a rough idea of your flight path angle in case of deficient observability. This also adds some level of robustness to the sensor system. The multi-patch radar altimeter is required for landing the rocket and for the GNC landing algorithms. The throttle of the engine will be directly related to the incoming speed and altitude off the lunar surface. Similar technology is used on the Falcon 9 reusable rocket. The AVGS system will be used for docking/berthing as well as for entry descent and landing. The camera system allows for the rocket to observe the landscape as a contour and maneuver around objects.

The thermal management system will require modification. Temperature on the moon can get down to 40K in some of the permanently shadowed regions. This means that the vehicle must now be able to withstand that harsh environment for the duration of its presence in it. The vehicle will require resistive heaters on the avionics and battery boxes. Other parts of the plumbing might require heating to minimize seizures.

The avionics modifications required for this system are in the CDH, power, and TT&C subsystems. The command and data handling system will now have to support the sensors required for GNC and attitude control. This means an upgrade on the flight computer, interface boards, and harnessing. The power system must be beefed up to drive all sensors, heaters, and actuators on the vehicle. Solar panels will must be mounted to the side of the spacecraft to charge the high-capacity battery system while in orbit. Recall that the LRS vehicle will be in orbit around the moon when it is not refueling, to mitigate the thermal issues. The telemetry and command subsystem must also be updated to include an X-band system so that it can communicate with the crater-rim relay station. This means that there should be continuous communication with the system when it is around the moon as well as around Earth. The flight software must almost entirely be rewritten. The spacecraft's flight computer should have software that is designed to be robust for long durations, just like a spacecraft. It must be able to read from all sensors, drive all actuators, and be in full control at all times. Landing legs and attitude control thrusters will also need to be driven at specific events in the operations of this spacecraft. This is an effectively entirely new avionics system. Development costs will be high on this system. However, it is possible for many COTS systems to be used.

As for landing technologies, the legs will permanently deploy once post-jettison from the initial launch vehicle. The legs will fold out outside the frame of the vehicle and have a large connection surface area to the surface. They will lock into place with a damper to damp out the initial impulse of hitting the ground. These legs will be made out of extensible carbon fiber and aluminum honeycomb, taking inspiration from the SpaceX Falcon 9 legs. This must be studied further to determine the number of landing cycles these legs would survive. There are most likely much better solutions.

Being able to land the spacecraft depends on a lot of elements. The landing profile is important, but we will not go over that for now. Object avoidance will be very important in this implementation of our LSR. Initial landings and those that follow will need to make sure to avoid landing in a rocky landscape as well as on the ISRU equipment. There exists a well-tested algorithm called the Guidance For Optimal Large Diverts (GFOLD), which can dynamically alter your trajectory as it observes obstacles within the flight path. It will do this using lossless convex optimization in real time and on board. This system has been testing on Masten's flight vehicles. The throttle ability of the version of the RL10 engine on our centaur goes from %104 to %5.9. It turns out that %5.9 is precisely the value required to land the vehicle softly on the moon. This means that no modifications need to be made to the engine. The engine thrust vector control system (TVC) moves +/- 4 degrees and will most likely be dynamic enough for our purposes and will not require modification.

Attitude control must also be added to the system to be able to spin the rocket over when landing and to spin the rocket during docking/birthing procedures. We plan on doing this by heating up excess oxygen and expelling it as a cold gas. We have yet to perform sizing calculations, but with a low angular rate the mass flow rate seems achievable.

3.6.2 Propellant Storage

Because propellant storage is a concern for LEEP no matter what, the team decided to let the solution for propellant storage be largely driven by our other trade space decisions. Long-term storage of cryogens, including LH2, is not impossible as long as one follows a few key cryogenic design principles:

- Minimize penetrations
- Minimize surface area
- Segregate the LO2, LH2 and warm mission module
- Use historic cryogenic lessons learned to the greatest practical extent
- Enable full system ground check out

As an existing upper stage, the Centaur we are modifying into the LRS already accomplishes several of these tasks, and the fact that we are modifying an upper stage limits the extent to which we can accomplish others. For example, the hydrogen and oxygen tanks are already segregated, but we can do little to minimize surface area without drastically altering the original design, which would contribute to compromising the efficiency, reliability, and high TRL advantages of using the Centaur as our template.

Despite the advantages conferred by the stock Centaur, we determined that because our proposal calls for storing propellant in the LRS tanks for long periods of time, a ZBO solution would be necessary on board the LRS. One advantage of using the LRS as storage for the cryogenic propellants throughout the production and transportation of the propellants is that only the LRS must be equipped with ZBO technology; this helps to save on mass and complexity of the overall system. A ZBO system of some sort is shown to be necessary by a paper submitted to IEEE, which estimates that a propellant depot consisting of Centaur tanks protected by a similar MLI scheme would suffer from boil-off of 0.1% per day[10], corresponding to boil-off of 3% per month. Due to the expected holding times of up to or even more than six months, the report's numbers indicate that we would lose up to 17% of our hydrogen propellant to boil-off. Because our resupply ability is already limited by deliverable hydrogen mass, not deliverable oxygen mass, and because as the paper notes, the boil off would be entirely or nearly entirely hydrogen, boil-off exacerbates issues already present.

[10] http://sciences.ucf.edu/class/wp-content/uploads/sites/58/2017/02/Propellant-Depots-IEEE-2011.pdf

On a more promising note, the paper does also note that the Centaur hydrogen tank is already quite robust against heat intrusion so long as there are not large bulkhead thermal gradients.

With this in mind, a ZBO system is necessary for LEEP to fulfill its mission effectively. Several systems were considered to provide ZBO capability to the LRS.

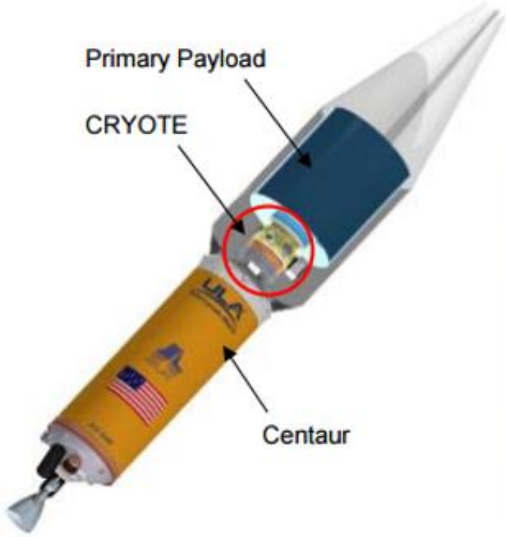
Tab. 3.6.2.1: Cryogenic Propellant Storage Technology

System	Benefits	Drawbacks
IVF (ULA)	Technology is already in development Designed to be integrated with the Centaur Simplifies Centaur subsystems substantially	Only reduces boil-off by 50-70% Requires systems for high-pressure GH2/GO2
Stirling Heat Engine	Higher Efficiency than a Pulse Tube engine	Lots of moving parts introduce the risk of failure
Pulse Tube Heat Engine	No moving parts on the hydrogen side Being developed for use with the Centaur Theoretically eliminates boil-off	Lower efficiency than a Sterling engine

For a variety of reasons, a system incorporating both IVF and the MLI used by the conceptual fuel depot was chosen. The IVF system, already being developed for use on the Centaur vehicle and therefore easily adaptable to the LRS, allows us to reduce boil-off to just 5% over the course of the six-month storage described above, and possibly lower with further development of the technology. Use of the IVF system also eliminates the need for extra pressurant (such as helium), which would have to be brought from Earth, and provides the LRS with constant milli-G ullage impulse. In comparison to the IVF system, both types of heat pumps left major problems unsolved while providing a comparatively small benefit as compared to IVF. While this approach does not completely eliminate boil-off and is therefore inappropriate for truly long-term propellant storage, our mission architecture does not call for such storage. IVF technology is also currently under development by ULA for integration into the Centaur vehicle, minimizing LEEP's investment into develop a ZBO system and maximizing the compatibility of our adopted ZBO architecture with the LRS. One final benefit is that the constant power produced by the IVF system at least partially eliminates the need for solar panels to provide electrical power for the LRS and keep the mission systems warm. The exact power requirements of the LRS and the amount of power that the IVF system can provide while minimizing boil-off will have to be evaluated in additional work. The IVF system is currently at TRL 6, but ULA has the goal of raising it to TRL 7 or TRL 8 by the end of 2019, substantially before we will need to integrate IVF into the LRS.

3.6.3 Fuel Transfer

The ability to refuel cryogenic propellant on-orbit is crucial to the success of Lunar Port. NASA is currently experimenting with a platform designed to demonstrate the major technologies required for this objective (Wall et al., 2017). The test bed will be capable of transferring residual liquid hydrogen (LH2) or liquid oxygen (LO2) from a centaur upper stage, and storage in a secondary vehicle for u to one year on-orbit. According to (Simple, Robust Cryogenic Propellant Depot for Near Term Applications IEEE 2011-1044), it could be feasible to deploy an affordable propellant depot into earth orbit this decade, and the technology could be scaled up to support more demanding missions and launch capabilities.



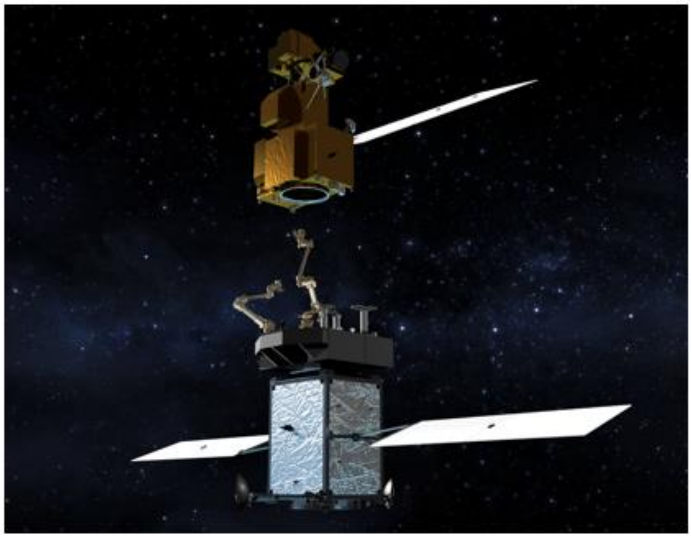


Fig. 3.6.3.1: Cryogenic Propellant Fuel Transfer Experiments

The CRYogenic Orbital TEst (CRYOTE) concept rides inside the Atlas V payload adapter and receives residual LH2 or LO2 from a Centaur upper stage. As can be seen in the following figure, once CRYOTE and its variants have been tested in orbit, the technology required to complete such cryogenic transfers in orbit shall be proven and ready for use. Results from CRYOTE can provide the critical in-space CFM demonstration to allow selection of mission architectures that utilize on-orbit fueling, long duration cryogenic storage and development of cryogenic propulsion stages truly designed for in-space use that have higher mass fractions and reduced boil-off compared to current designs.

Tab. 3.6.3.1: Technology Readiness Levels of Cryogenic Fuel Transfer Technology

Cryo Transfer Technology	Current TRL		TRL Post-CRYOTE Lite		TRL Post-CRYOTE Pup, Free Flier	
	0-g	Stld	0-g	Stld	0-g	10^{-4} g
Transfer System Operation	4	5	4	9	9	9
Pressure Control	4	9	6	9	9	9
Low Acceleration Settling	N/A	9	N/A	9	N/A	9
Tank fill operation	4	5	4	9	9	9
Thermodynamic Vent System	5	5	7	7	9	9
Multi-layer insulation (MLI)	9	9	9	9	9	9
Integrated MLI (MMOD)	6(2)	6(2)	9(7)	9(7)	9	9
Vapor Cooling (H_2 para-ortho)	9(4)	9(4)	9	9	9	9
Passive Broad Area Cooling (active)	9(4)	9(4)	9(4)	9(4)	9	9
Active cooling (20k)	4	4	4	4	9	9
Ullage and Liquid Stratification	3	9	9	9	9	9
Propellant acquisition	2	9	9	9	9	9
Mass Gauging	3	9	9	9	9	9
Propellant Expulsion Efficiency	3	9	9	9	9	9
System Chilldown	4	5	4	9	9	9
Subcooling P>1atm (P<1atm)	9(5)	9(5)	9(5)	9(5)	9(5)	9(5)
Fluid Coupling	3	3	3	3	9	9

Settled cryogenic propellant transfer can benefit from vast CFM (Cryogenic Fluid Management) experience used on Centaur and other cryogenic upper stages as well as near term flight demonstrations such as CRYOTE (Cryogenic Orbital Testbed). Autonomous docking is regularly performed by the Russian Soyuz and fuel transfer techniques being pioneered by NASA Robotic Refueling missions could be leveraged between the stages. Using Robotic arms to berth the massive spacecraft is unproven and likely unfeasible.

Rates of propellant transfer achieved in space:

The United Launch Alliance has been working on in-space cryogenic fuel transfer with the Centaur upper stage over the past 20 years. This is another reason to use the Centaur upper stage as our LRS as the technology is currently being developed for the very same vehicle. By rotating the stage at roughly 0.5 rpm, settling levels of propellant of up to 150 kg/hr can be achieved. This maneuver significantly simplifies the refueling process and maximizes the use of reliable and existing technology. The following key CFM technologies are all currently implemented by settling on both the Centaur and Delta IV upper stages: propellant acquisition, hardware chill-down, pressure control, and mass gauging.

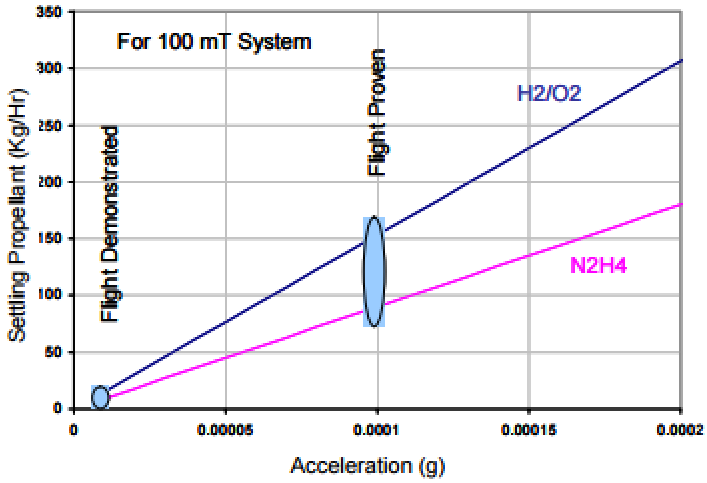


Fig. 3.6.3.2: Cryogenic Fuel Transfer Rates Achievable in Space with Centrifugal Acceleration

With low acceleration, propellant consumption for settled cryogenic propellant transfer becomes reasonable. At these rates, 5 mT of propellant could be transferred from at Centaur upper stage at a rate of 150 kg/hr in a few days. If higher rates of fuel transfer were desired, friction in the pipes, cavitation, and structural loads on the vehicle need to be taken into account.

Therefore, utilizing low level acceleration during cryogenic propellant transfer significantly simplifies the entire operation and maximizes techniques pioneered by ULA in cryogenic-fluid-management (CFM). Settled methodologies for propellant acquisition, hardware chill-down, pressure control and mass gauging are already in service on Atlas V Centaur and Delta IV upper stage.

3.6.4 Life Cycle

Each LRS is launched on a Falcon Heavy with a modified fairing to fit the Centaur upper stage and a payload. The LRS is not fully fueled for launch, but the tanks are pressurized to capacity with LH2 and LOX. Calculations based on the published capabilities of the Falcon Heavy indicate that this configuration will deliver the LRS to LLO, with enough fuel to descend to the surface safely and with 9.3 mT of usable payload. This 9.3 mT is utilized to construct the base, as detailed in Section 4.

After its delivery to LLO, the LRS is used for refueling missions. We estimate based on RL-10 performance data that the components of the LRS will last for substantially more than 100 firings of the engine, but we do not have data on how long the LRS will last in lunar orbit or on the surface of the Moon. By using the IVF infrastructure for ZBO outlined above, we are also eliminating many of the Centaur elements which would be difficult or impossible for us to maintain at the Moon. It therefore seems reasonable to assume that the LRS will survive for about as long as the various pieces of lunar surface infrastructure, which means that when an LRS must be replaced, we should also be supplying replacements for lunar surface infrastructure with the new LRS.

The question remains of what to do with an LRS at the end of its life cycle. When an LRS is no longer useful in a resupply capability, we still wish to gain the maximum utility from it. The three major disposal options are to use the LRS as a propellant storage tank on the Moon, to abandon the LRS somewhere on the Moon, or to perform one final refueling mission(at greater efficiency because we don't carry enough fuel to land the LRS again) and then crash the LRS. We believe that the best solution is the third option described, because (a) it allows an LRS that is nearly ready to be retired to provide one last useful refueling mission, does not waste the fuel required to land a retiring LRS on the surface of the Moon, and allows us to crash the LRS into an un-prospected crater and gain more information about which crater or craters it might be profitable to expand LEEP into, in the same way that the LCROSS impact provided information about Cabeus.

3.7 Communication Design

A surface relay at the rim of Cabeus crater (delivered along with the solar focusing equipment) enables a communications from lunar surface to Earth ground during periods of direct line of sight. We considered several communications system design options. We at a constellation of small satellite repeaters, and a rim stationed relay. The small satellite repeaters would require a lot of RF power, a large antenna, and would only allow a small window of operation. The context switching between satellites would also complicate operations. The fact that whole new spacecraft would require development also expands our mission scope and would expand development time and cost. The rim system will have near-constant sunlight and near-constant line-of-sight to the Earth (LOS). The rovers will communicate locally to the edge of the rim using a low power transmitter. We then looked at appropriate ground stations for the Earth. The Deep Space Network is optimal in terms of receive antenna gain, but would not be able to support our operations for long periods of time. It is also prohibitively expensive to use the DNS ($45 per minute, or $2M per Month). The commercial solution in study, Spaceflight Industries, can allow us to operate without limitations in bandwidth for $50k per month. With this in mind, we have chosen to move forward with utilizing commercial ground station operations. The Spaceflight Industries ground station network allows coverage around the Earth for S-band and X-band systems. Given that X-band RF operations can support large data rates, have flight heritage on the Curiosity rover, and the crater-rim station has large amounts of solar power opportunity, we have chosen to use X-band telemetry and command systems. Below you can see the link budget for the lunar communication system downlink. It takes roughly 4kW of repeater power to make sure the C/N is at least 3dB above the noise floor. The upgrades uplink of commands will be supported in the future when Spaceflight industries upgrades their X-band transmission equipment. The figure has been shown in the next page.

Xband Uplink Budget		
LINK PERFORMANCE ESTIMATION, VIOLET		output
Constants		intput
Temperature	270	
k (Boltzmann's Const. In J/K)	1.38E-23	
c (Speed of Light in Mm/s)	3.00E+08	
AOS=Acquisition of Signal		
	Option 1	
R (Data rate)	2,000,000	
Bandwidth(MHz)	30	
Modulation Scheme	GMSK	
Pwr (Tx Power in W)	4000.00	
Pwr (Tx Power in dBm)	66.02059991	
Ll (Line Loss in dB)	1	
Antenna variation factor (in dB)	0	
Gt (Tx Total Antenna Gain in dB)	16.23	
EIRP	81.25059991	
Raos (Dist to SAT @ AOS in km)	384400	
f (Desired Tx Frequency in GHz)	8	
FSPL (Space Loss @ AOS in dB)	222.31	
ISAB (Ionospheric Absorption Loss in dB)	1	
Latmo (H20 & O2 Atmo-/Ionospheric Losses i	0.5	
Gr (Rx Antenna Gain in dB)	45.9	
Power Received dBm	-93.65686741	
Figure of Merit(G/T)(dB/K)	25	
Noise Power (dBm)	-99.51635895	
C/N	4.36	

Fig. 3.7.1: Proposed Link Budget for Lunarport Communication to Earth

4. Construction

The LEEP construction phase is scheduled to take place over several years and multiple flights. The act of deploying the system will provide critical experience in multiple phased landings of materials (a skill NASA will need to master to support Mars crews) and robotic operations in extreme environments.

4.1 Construction and Deployment of Lunar Surface Structures

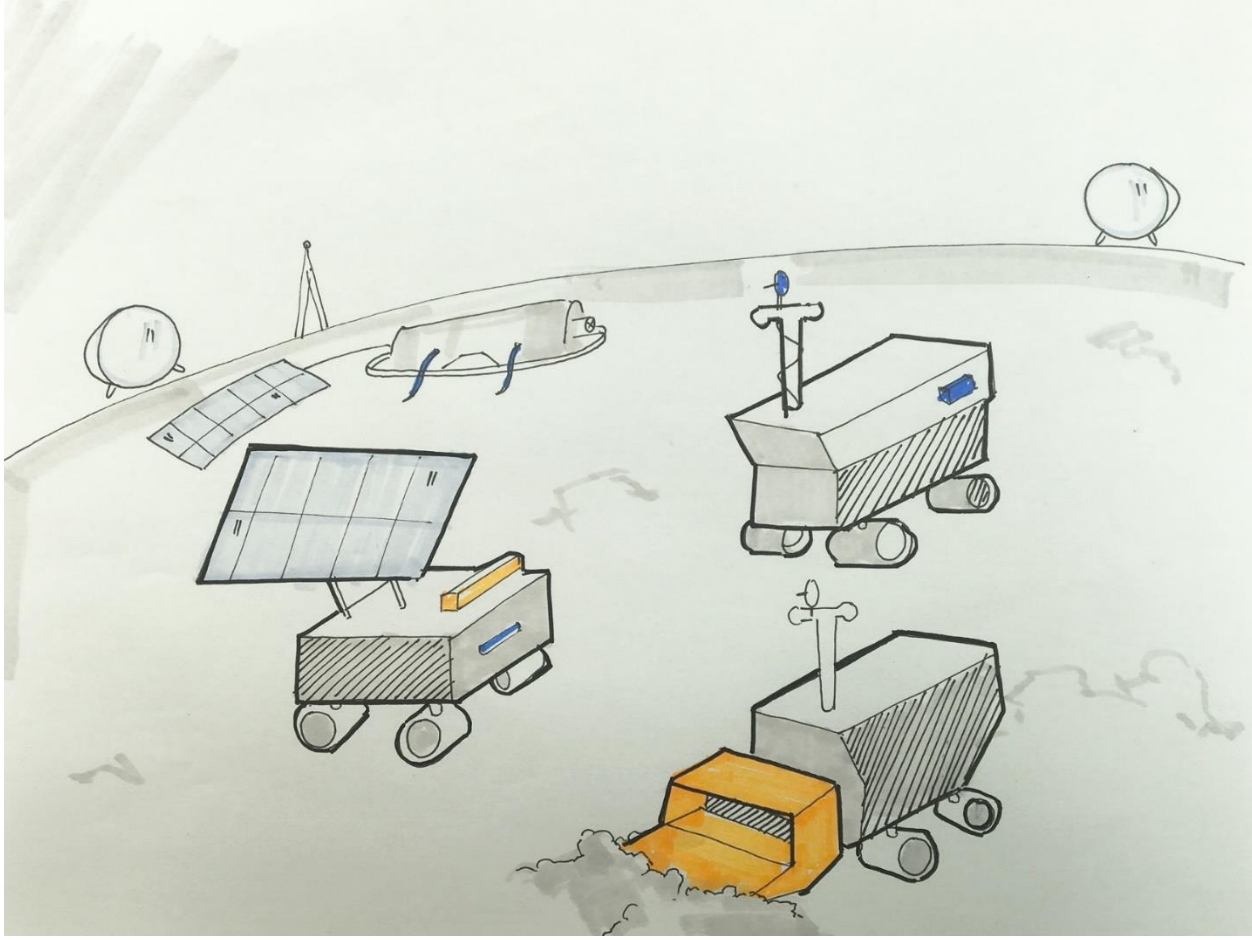


Fig. 4.1.1: Overview of some lunar surface infrastructure. Left to right: Solar concentrators, crater PV farm, constructor bots, ISRU unit, prospector

4.1.1 Base Deployment Sequence

In order to deliver the lunar infrastructure for LEEP to the lunar surface, we had to decide on a launch vehicle to get the infrastructure into orbit and on the way to the Moon. We wanted to launch our infrastructure using the LRS if possible, but clearly getting the payloads to the Moon is the priority. With that in mind, we considered four launch vehicles: SLS, Falcon Heavy, Atlas V, and Delta IV Heavy. We also considered Blue Origin's New Glenn vehicle, but too little information was available on the booster to make a decision and the design is still highly conceptual, so we excluded it from our main decision making process.

Tab. 4.1.1.1: Launch Vehicle Trade-Off

LV	Cost per launch	Payload to LEO	Payload to GTO	Compatibility with LRS
SLS	$500 million	105 mT	Est. 50 mT	Low
Falcon Heavy	$110 million	54 mT	22 mT	Medium
Atlas V 551	$153 million	19 mT	9 mT	High
Delta IV Heavy	$375 million	29 mT	14 mT	High

We chose the Falcon Heavy for lunar surface infrastructure deployment because of its combination of good payload to GTO, low cost, and reasonable integration with the LRS. While the SLS has more payload capability to any given orbit, on a budget of $1 billion a year a $500 million launch (before the cost of the payload) puts very tight constraints on the budget, and despite their good integration with the LRS due to their ability to use the Centaur as an upper stage, both the Atlas V 551 and the Delta IV Heavy were too expensive for the payload they allowed us to justify their use.

The biggest hurdle that needs to be overcome in using the Falcon Heavy is that the Centaur vehicle with a payload is too tall to fit inside SpaceX's fairing. However, SpaceX has indicated that they are willing to modify their vehicles for customers, and we would not need substantial modifications aside from lengthening the fairing slightly to accommodate the Centaur upper stage. We have budgeted an extra $15 million per launch to cover the initial cost of designing and validating the fairing modifications and the per-launch fabrication costs of the non-standard fairing. Considering the huge benefits of using the Falcon Heavy, from the much lower cost to the much greater payload, we believe that altering the fairing is a relatively small price to pay.

The use of the Falcon Heavy with the LRS and lunar base infrastructure as its payload allows us to deliver 15.3 mT to lunar orbit, of which 4 mT is the LRS, 2 mT is the fuel needed for the LRS to land itself on the moon, and 9.3 mT is the lunar infrastructure and its landing vehicle, detailed below in Section 4.1.2. LEEP ground infrastructure is laid over a series of four Falcon Heavy launches beginning in 2024, with full propellant extraction capability anticipated in 2028.

Tab. 4.1.1.2: Details of Consecutive Launches

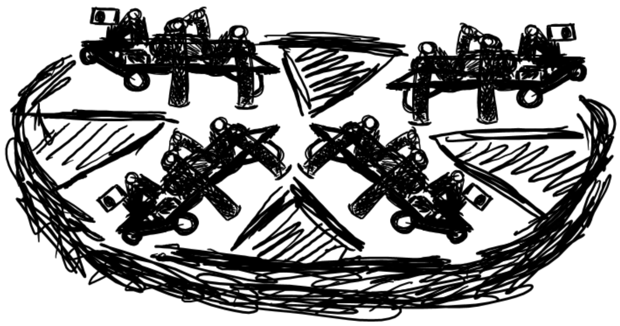
Launch Year	Utility
2024	Deployment of two solar focusing stations in separate locations along the rim of Cabeus crater.
2026	Delivery of equipment into a permanently shadowed crater region to prepare for ISRU.
2027	Landing of H_2O extraction rovers and electrolytic processing equipment.
2028	Delivery of additional extraction rovers.

The first payload is launched in 2024 and deploys solar focusing equipment along the crater rim to illuminate the landing site and provide available power, inspired by Stoica et al (2014). See the power section for details on solar focusing deployment. The second payload delivers four rovers in 2026, one for construction and maintenance, and one for ice deposit prospecting. These land in the permanently shadowed crater region and prepare the site for ISRU equipment delivery. The construction/maintenance rover then deploys a solar farm within the crater region to power the Electrolyzer unit.

In 2027 the third ground payload delivers the ISRU electrolysis unit and a first batch of extraction rovers. Water extraction and processing begins. Lessons learned are incorporated into the builds of the second batch of extraction rovers, which are delivered into the crater as the fourth lunar surface payload in 2028, bringing the total number of extraction rovers to ten and the base to full propellant production capacity.

4.1.2 Lunar Surface Equipment Delivery Design

The delivery sequence of lunar surface equipment requires delivering multiple robotic rovers at once and in the same location. This can be accomplished by a larger version of a typical retrorocket descent rover deployment shell (such as for the proposed Lunar Polar Volatiles Extractor mission), modified to deploy multiple rovers and carry additional modules. This system is called the Lunar Landing System (LLS). A cartoon of this delivery system, nicknamed the Pizza Delivery Truck, is shown on the next page.



Lunar Landing System (LLS)

- Nickname: The Pizza Delivery Truck
- Rovers Delivered: 6 Max
- Diameter: 4 m
- Deliverable Mass: 4 mT
- Launch Mass: 9 mT
- Additional Use:
  - Carries constructor modules instead of unplaced rovers.
  - Can deliver and house ISRU unit.

Fig. 4.1.2.1: Lunar Landing System (LLS) Description

The LLS consists of a platform, on which are mounted modular payloads, which has an integrated hypergolic bipropellant propulsion system. It is intended for one-time use and designed to be as versatile as possible when it comes to delivering our equipment to the lunar surface. The propulsion system is an Aerozine 50/N2O4 hypergolic system, chosen for its heritage of successful lunar use, easy-relight hypergolic characteristics, and the fact that the delivery system is out of necessity sized such that the DPS from Apollo can be directly lifted and installed into the LLS. This adoption of existing technology (TRL 9 as of 48 years ago) greatly simplifies this custom-built spacecraft and lowers development and validation costs. While the system does have a specific impulse of 311 seconds, substantially lower than the 465.2 seconds of the LRS's RL-10 engine, it avoids the problem of cryogen storage on a small, non-reusable spacecraft and reduces the complexity of a system we have to develop from scratch.

There are three kinds of equipment to deliver onto the lunar surface in and around Cabeus crater. On the crater rim, two LLS's carrying 5 folded solar focusing mirrors each land in a typically lit region with a near-constant line of sight to Earth. See section 3.4 for further details. These deploy to their determined locations and focus solar light into the crater. The Lunar Landing System deploys a folded parabolic dish for direct to Earth (DTE) communication, powered by a deployed rotating solar panel angled perpendicular to the lunar surface. This ensures that the landed LLS remains useful following mirror deployment.

The first landing in the lunar crater occurs within a permanently shadowed region now illuminated by the solar concentrators. An LLS with two rovers and two construction modules then lands. One rover is a solar powered water ice prospector based on the proposed Lunar Polar Volatiles Explorer (LPVE) system. Another is a construction rover (the constructor) with a tractor attachment to pick up constructor trailer modules stored in the vacant LLS positions. More vacant space in the LLS holds the deployable solar farm to collect mirrored energy. The constructor prepares crater base for the Lunar Resupply Vehicles (LRS) to land. To keep the landing vehicles from kicking up dust and interfering with operations (polluting solar panels and getting in the extractor H_2O interfacing valves), the constructor bulldozes the thin upper layer of loose regolith to clear a landing and H_2O operations region. Water extraction may occur outside of the bulldozed regions.

The ISRU H_2O processing unit lands with retrorockets on a modified LLS without any rovers. The constructor rover equipped with an arm module shepherds the deployment of this unit; the cryogenic fuel hoses and power cable are attached to its outlets. The section below discusses this in more detail. Six extractors are then delivered in a second LLS with this payload for full extraction capability. An additional shipment of six extractors (or fewer extractors replaced with maintenance equipment) may be sent on this Falcon Heavy launch if replacements are needed.

4.2 Robotic Construction Operations

Overall, the robots are designed for modularity, redundancy, and tool exchange. These concepts take heritage and inspiration from the NASA Goddard Robotic Refueling Mission (switching tools off a robot arm) and NASA/Caltech JPL's ATHLETE mission. Where possible, technologies at a TRL-6 or higher are used to improve mission reliability.

4.2.1 Operation Concept for Rovers

As a general philosophy, the rovers will follow a phased autonomy model:

- Teleoperation
- Teleoperation at critical phases only with notifications to ground if anomalous events occur
- Full autonomy with monitoring, anomaly monitoring, and notifications

Key technologies:

- Phased autonomy
- Hybrid processing: use SpaceCube to eventually process
- Intelligent notifications
- Fault protection

Over the years of operations as operators become more familiar and the modular rovers are upgraded, increasing autonomy could be provided. A few technologies such as decreasing cost and size of LIDAR and improved space processors enable this development over time.

For command and data handling aboard harvesting rovers on the lunar surface, NASA Goddard's spacecube could serve as an advanced data processor. SpaceCube ™ hybrid science data processing system that provides 10x to 100x improvements in computing power while lowering relative power consumption and cost.[11] This combined approach yields a significantly cheaper system.

Designed for in-space operations, the increased processing power available locally could allow for simultaneous location and mapping along with path planning and collision avoidance on the lunar surface. Such methods would alleviate the need for constant human interaction and increase general productivity of on-surface operations.

Advanced capabilities such as NASA Goddard's SpaceCube will be considered. Advanced data processors for in-space operations would help advance either the rover's on board processing capabilities or have that data processed more locally, possibly in the Lunar Landing System (see Section 4.1.2).

The operations plan will also include logic to prevent the rovers from going below 20% battery[12]

A tremendous amount of research has been conducted regarding rover capabilities, robot arms, and autonomy. The table below provides some of the context for rover research and space robotics.

Tab. 4.2.1.1: Current Space Robotics Missions and Proposals

Name	Group Researching	Comments & Significance	Link
RRM, RRM2, RRM3, Restore-L	NASA Goddard, Satellite Servicing Projects Division	In-orbit satellite servicing robotic technologies Both missions involve in-space robotic arms Fiducials being used to help locate important structures (ie, propulsion line)	https://sspd.gsfc.nasa.gov/missions.html
Archinaut	Made In Space	Autonomous manufacture and assembly in space	http://www.madeinspace.us/projects/archinaut/
Commercial In-space Robotic	Orbital ATK	Assembly of large space structures	https://www.orbitalatk.com/news-

11 "SpaceCube" <http://spacecube.nasa.gov>. Accessed: 30 March 2017.

12 Inspired from discussions with Jay Trimble (NASA Ames)

Assembly and Services (CIRAS)			room/release.asp?prid=204
Dragonfly	Space Systems Loral	Robotic on-orbit satellite assembly	http://sslmda.com/html/pressreleases/pr20151210.html
Robonaut	NASA JSC	ISS humanoid robot	https://robonaut.jsc.nasa.gov/R2/
Astrobees	NASA Ames	Follow up project to ISS based SPHERES Housekeeping around ISS	https://ti.arc.nasa.gov/tech/asr/intelligent-robotics/astrobee/
Regolith Advanced Surface Systems Operations Robot (RASSOR) Excavator	NASA Kennedy	Mining robots	https://technology.nasa.gov/patent/KSC-TOPS-7 https://www.nasa.gov/topics/technology/features/RASSOR.html
Planetary Deep Drill, Dust Removal Tool, Icy Soil Acquisition Device, etc	Honeybee Robotics	Autonomous robots, dust removal, drilling Several intended for Mars but could apply to moon	https://www.honeybeerobotics.com/technology/

4.2.2 Justification for Number of Extraction Rovers

Assumption: On average, 4% water in lunar regolith by weight.

To support missions to Mars, LEEP needs to produce 60 mT of water every 26 months. Each extraction rover is equipped with 4 HoneyBee core drills (refer to Section 3.3 on extractors) which produce 3 kg of water every day[13] (working 10 hours/day) based on the assumption above. Based on our analysis, 12 extraction rovers are required to have a maximum capability of producing 52.56 mT of water yearly. Moreover, all rovers are identical. Hence, multiple rovers increase the redundancy in our system and reduce

[13] Honeybee Robotics (Zacny et al), Planetary Volatiles Extractor (PVEx) for In Situ Resource Utilization (ISRU)

risk. Due to payload constraints, we send 6 of them in our third launch and the remaining 6 in the fourth launch.

This system offers modular flexibility as in the future we could add more rovers in order to expand our capabilities if the demand increases. We expect the lead time for this expansion to be 2 to 3.5 years. Given that the rover will be a proven design, we expect production to be approximately 2 years (based on analogy to the proven geostationary satellite system production.) However, given the unique nature of the mission, short notice launch manifesting on commercial systems may be an additional bottleneck. The following graph illustrates the total maximum amount of water that can be produced on the moon over the first 15 years from when we started extraction assuming we add 6 more extraction rovers after year 10 leading us to produce 78.84 mT of water every year.

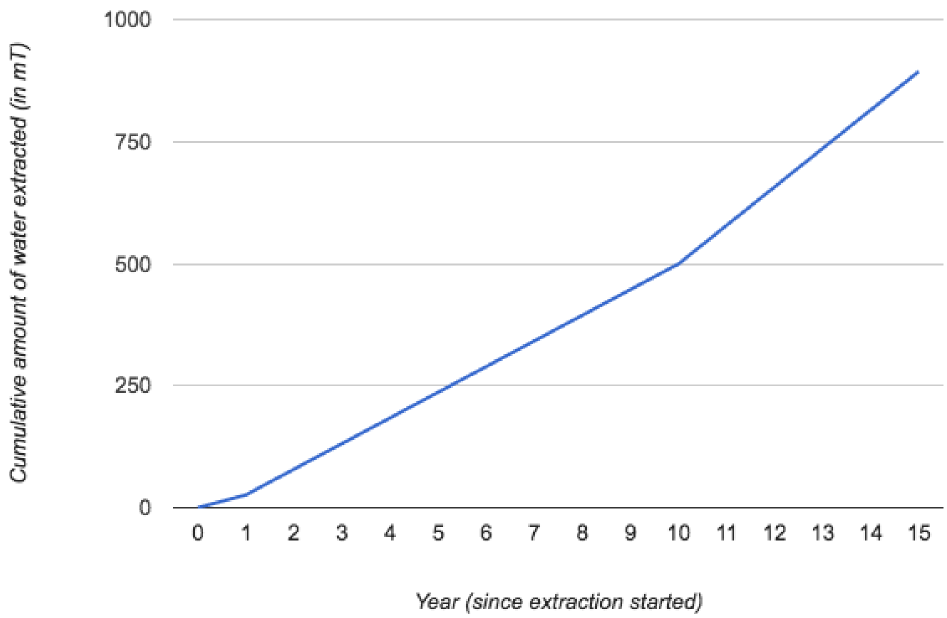


Fig. 4.2.2.1: Total Mass of Extracted H_2O by Year from Lunarport

Figure 4.2.2.1 shows the cumulative fuel production from starting year (2027) when six extraction rovers are delivered to the lunar surface. After one year, six additional rovers are delivered, increasing the production. After the second year, sufficient fuel has been produced for one SLS mission. From year 1 to year 10, fuel is produced at a rate twice that needed to fuel one SLS every 26-month window of opportunity to Mars (alternatively, at a rate needed to fuel 2 SLS missions at every 26-month window of opportunity to Mars). At 10 years (for example) after initial deployment, additional rovers can be deployed to further increase fuel production (note the kink in the plot in figure 4.2.2.1).

5. Operation

5.1 Refuel Trajectory Design

5.1.1 Orbital Considerations

The baseline of this study is to refuel the second stage of the Space Launch System S Block 1B, the Exploration Upper Stage (EUS) for a mission to Mars. The SLS program is designed to support a variety of missions, including crewed cis-lunar missions in the 2020s, and Mars exploration in the 2030s. Increasing the payload delivered to Mars with the EUS is chosen as a performance metric mainly because of data availability. The following figure[14] shows the usable payload delivery capability of the SLS to an elliptical apogee of a given altitude. This payload mass is taken as the maximum mass that could be further transported to Mars from that orbit altitude. As a baseline, the EUS can deliver 31.7 mT of payload to Mars. The EUS has a gross lift-off weight of 143.6 mT, and a burnout mass of 15.6 mT[15], and an Isp of 462.5 s[8].

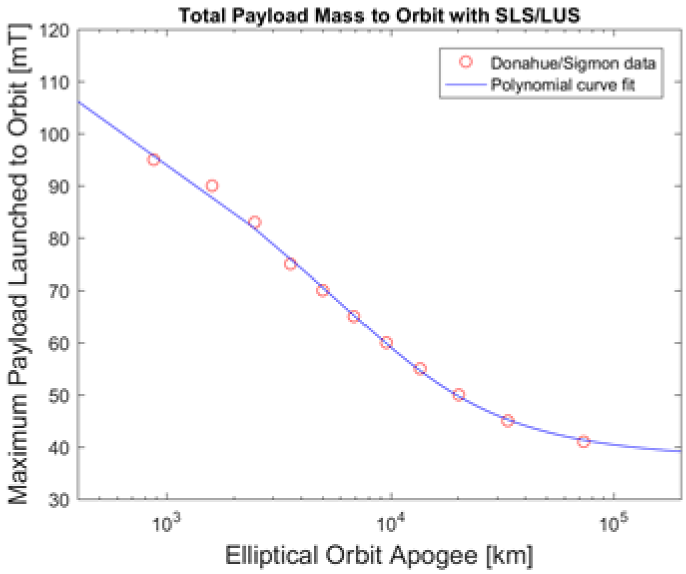


Fig. 5.1.1.1: Total Payload Mass to Orbit with SLS/LUS

The propellant capacity of the LRS is taken from the Centaur upper stage tank design, housing 20.8 mT of fuel and oxidizer. Including the modifications to the Centaur upper stage described in Section 3.6, the burnout mass is estimated to be 4 mT. The LRS uses the propellant to propel itself from the surface of the Moon to a rendezvous orbit, resupply the EUS, return to a low lunar orbit, and eventually land on the lunar

[14] Donahue B.,Sigmon S., The Space Launch System Capabilities with a New Large Upper Stage

[15] http://www.spacelaunchreport.com/sls0.html

surface to start a new cycle. This implies that the further the LRS has to travel to resupply the EUS, the less propellant it can transfer to the EUS. Therefore, the payload mass to Mars can be optimized by selecting the resupply orbit and transferring the exact amount of fuel to the EUS that is required to complete the journey to Mars. The first step is to determine the range of cis-lunar orbits to consider in order to determine the LRS Delta-V requirement.

The 2033 and 2035 launch opportunities to Mars span a range of different departure characteristics. For the following analysis, it is assumed that a payload is injected into an orbit to Mars during the 2033 time window. The SLS is launched at Cape Canaveral, at 28.5° latitude, into the Earth-Moon plane. Launch windows to the Moon can be considered daily and monthly[16]. It is assumed that a time launch window to the Moon exists such that the maneuver to Mars can be performed. This is a fair assumption given the fact that, for deep space missions, spacecraft are usually parked around Earth for a couple of months before injection.

- Low Earth Orbit (LEO)

Low Earth Orbits are an attractive solution for the customer since they consume very little propellant to reach the rendezvous orbit, allowing them to launch larger usable payloads. The round-trip Delta-V between the Moon and LEO is at least 11 km/s. Using an Isp of 465.2 s[17], a fully fueled LRS has a maximum Delta-V capability of 8.4 km/s. Therefore, reaching low Earth orbits will be very costly for the LRS, which will only be able to transfer insignificant amounts of propellant.

- Low Lunar Orbit (LLO)

On the other extremity of the range, if the LRS resupplies the EUS in low lunar orbit, it can deliver large amounts of propellant, but the EUS payload mass is now limited. Furthermore, the EUS has to be injected back into an orbit with lower perigee in order to have enough velocity to escape from the Earth, which can be done using a Moon flyby. It is known that the maximum payload the EUS can transport to Mars is 31.7 mT[18]. The payload mass that can be launched to a lunar orbit is on the scale of 38 mT using the SLS/LUS system[19]. This is the current upper limit of payload that LEEP would be able to support from low lunar orbit. Although this increases the propellant refueling capability significantly, the technology for autonomously docking around the Moon is well underdeveloped, and autonomous rendezvous would have to be performed

[16] Wheeler, R., Apollo Lunar Landing Launch Window: The Controlling Factors and Constraints (https://history.nasa.gov/afj/launchwindow/lw1.html)

[17] https://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/RL10%20data%20sheet%20Feb%202016.pdf

[18] Donahue, B. and Sigmon, S., The Space Launch System Capabilities with a New Large Upper Stage, AIAA Space Conference and Exposition, San Diego, CA, Spetember, 2013.

[19] Idem.

without the use of position radio navigation system such as GPS. This tradeoff encourages the development of a resupply system in Earth orbit, but allowing the capability of supporting deep-space missions when docking in lunar orbit has been proven.

- L1, L2, and stable/unstable manifolds

L1 and L2 have been proposed as docking points[20]. These options are not considered in this work because the design requires further analysis and the use of restricted-three-body dynamics, which usually takes longer. Furthermore, autonomous rendezvous and docking at L1/L2 has the same issues as rendezvous and docking at LLO.

- Circular and Elliptical Orbits

Circular and elliptical orbits around Earth reduce the operation complexity. In order to do a first order analysis, a patched conics approximation is utilized[21]. The goal of the analysis is twofold: get a ΔV budget for the LRS to go from the moon to a refueling orbit and back and obtain a ΔV budget for the EUS to be injected into a transfer orbit to Mars as a function of the apogee altitude of the refueling orbit. The apogee altitude is going to be selected later according to some optimization criteria. Two different refueling orbit families around the Earth and in the Earth-Moon plane are compared: circular orbits with altitudes from LEO (400 km) to 50% the radius of the sphere of influence of the Earth in the Earth-Moon system and elliptic orbits with perigee altitude at LEO (400 km) and apogee altitude spanning the same range as the circular orbit family altitudes.

The LRS is launched from the Moon's South Pole and injected into a polar circular LLO with an altitude of 100 km with a ΔV cost of 1.6 km/s[22]. A TEI maneuver is performed at a ΔV cost of approximately 0.8 km/sec (Hohmann transfer orbit in the Earth-Moon plane). The cost of injecting the LRS into the refueling orbit, which is variable, depends on the family and the apogee altitude of the orbit. In this orbit, the LRS and the EUS rendezvous and dock. After transferring the fuel, the LRS goes back to the moon using the same transfer orbits. The total ΔV budget for the LRS to go to the refueling orbit, rendezvous, dock, and come back to the Moon can be seen in the following figure, for both families of orbits as a function of the apogee altitude.

---

[20] Landau, D., Lunar Propellant for Interplanetary Missions, *JBIS*, Vol 69, 2016.

[21] Vallado, D., Fundamentals of Astrodynamics and Applications, 3rd Edition, Springer, 2007.

[22] Typical Delta V values for various space maneuvers (http://www.lr.tudelft.nl/?id=29271&L=1)

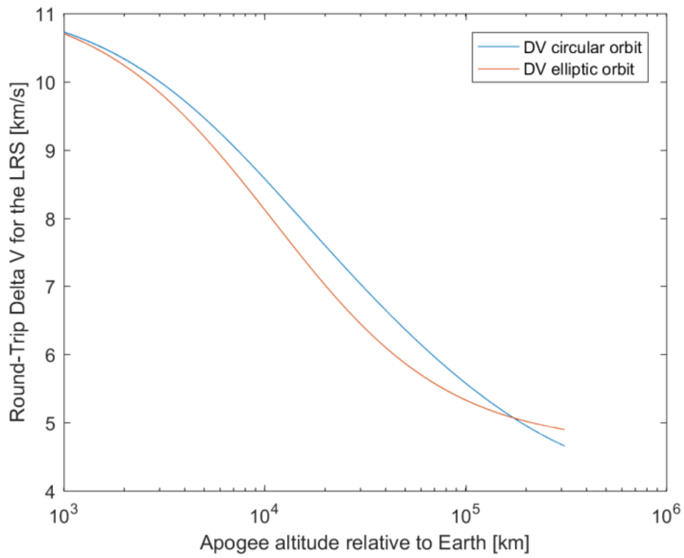


Fig. 5.1.1.2: Round-Trip Delta V Comparison for Circular and Elliptical Orbits

As expected, the elliptic orbit family is more ΔV-efficient. This is due to the fact that circularizing the orbit is more expensive than keeping the apogee of the refueling orbit larger. The benefit is more significant in orbits around GTO and gets smaller at higher orbits.

The SLS is launched from Earth, injecting the EUS into the refueling orbit. The ΔV needed to go to Mars has three components for an elliptic refueling orbit: an injection into a HEO orbit, an apo-twist[23], and a final TMI. The apo-twist maneuver may be needed in order to change the argument of periapse of the orbit such that the escaping-from-Earth maneuver is performed at perigee and V_∞ is aligned with the Earth's velocity vector. The argument of perigee change depends on the geometry of the Earth and Mars orbits (and the inclination of the equator relative to the ecliptic) at a given time window. For the 2033 window, the required change is around 45°[24]. Since doing an apo-twist maneuver is expensive when the velocity is high, an injection into a HEO orbit with perigee at the Moon's altitude is performed before TMI, such that the velocity at apogee is reduced. The cost of the apo-twist is computed and corroborated[25] and is around 1.5 km/s.

---

[23] Luidens, R. and Miller, B., Efficient Planetary Parking Orbits with Examples for Mars, Lewis Research Center, NASA, 1966.

[24] Landau, D., Comparison of Earth Departure Strategies for Human Missions to Mars, AIAA Space 2012 Conference & Exposition, Pasadena, CA, 2012. The actual angle is computed for a given apo-twist maneuver of 1.5 km/s at a HEO orbit.

[25] Idem.

For a circular orbit, the apo-twist maneuver is not necessary, since the burn can be performed at any point in the orbit. Therefore, the HEO injection is not performed and the TMI maneuver is done at the refueling orbit.

The ΔV budget for going to Mars in the 2033 time window is shown in the following figure. As expected, the elliptic orbit refueling family is more efficient.

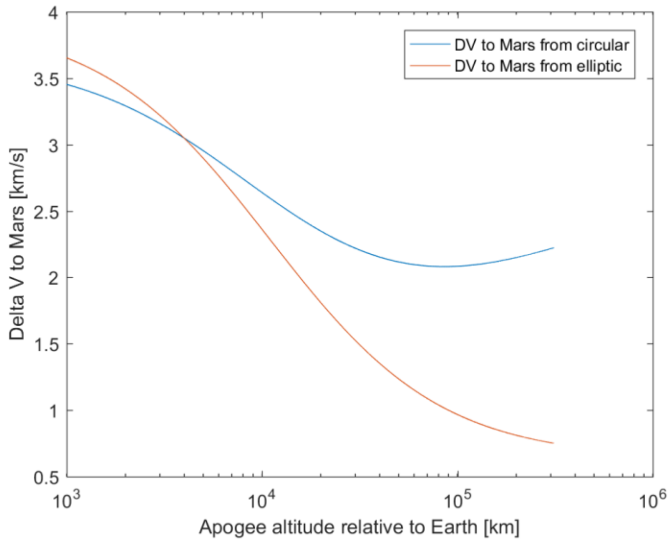


Fig. 5.1.1.3: Delta V to Mars Comparison for Circular and Elliptical Orbit

Using this analysis, circular refueling orbits are discarded since, in terms of ΔV performance, elliptic orbits outperform the circular family.

5.1.2 Resupply Optimization

The rendezvous orbit will be an elliptical orbit and the Delta-V requirement is calculated using the method described in the previous section. Given an elliptical orbit apogee, ΔV_{LRS} requirement for the LRS to take off from the lunar surface, reach the elliptical resupply orbit, and return to the Moon is calculated using the method described in the previous section. In the following equations, $M_{burnout}$ denotes the dry mass of the LRS after firing, $M_{resupply}$ refers to the mass of propellant for resupply, M_1 is the propellant for the trip from the Moon to the rendezvous orbit, and M_2 is the mass required to return to the Moon.

$$\begin{cases} M_2 &= M_{\text{burnout}} \left(e^{\frac{\Delta V}{2 I_{\text{SP}} g_0}} - 1 \right) \\ M_1 &= \left(M_{\text{burnout}} + M_{\text{resupply}} + M_2 \right) \left(e^{\frac{\Delta V}{2 I_{\text{SP}} g_0}} - 1 \right) \end{cases}$$

The following figure shows the capability range of the LRS design for refueling in Earth orbit. Between 10,000 and 200,000 km, the LRS can provide 1.6 to 8 mT of propellant.

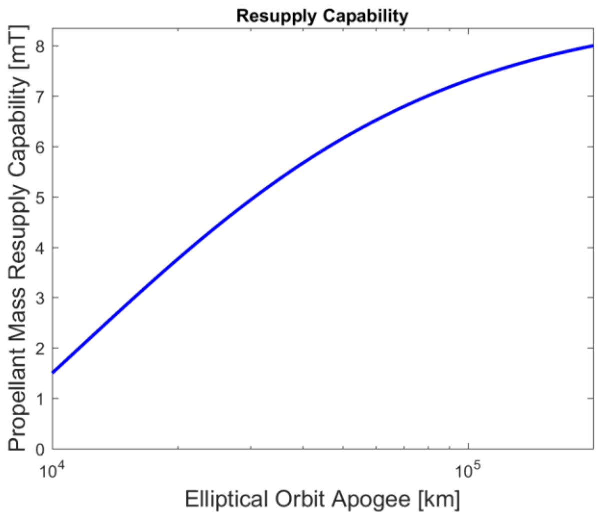


Fig. 5.1.2.1: Resupply Capability

Using the maximum amount of payload the SLS can launch in this elliptic orbit, and the amount of Delta-V required to reach Mars, the propellant resupply required can be computed. **The optimal rendezvous orbit corresponds to the altitude at which the propellant resupply capability is equal to the minimum amount of propellant required for the EUS**. The figure below shows the optimization results. The left blue vertical axis corresponds to the maximum amount of payload that can be launched with SLS/LUS at a given altitude. The right red vertical axis corresponds to the ΔV required to go to Mars and enabled by the LRS. The cross-section between the required and enabled ΔV corresponds to the altitude at which the LRS can deliver the exact amount of propellant the EUS needs to go to Mars. This point is the maximum payload mission to Mars LEEP can support. The resupply optimization analysis has been shown on the next page.

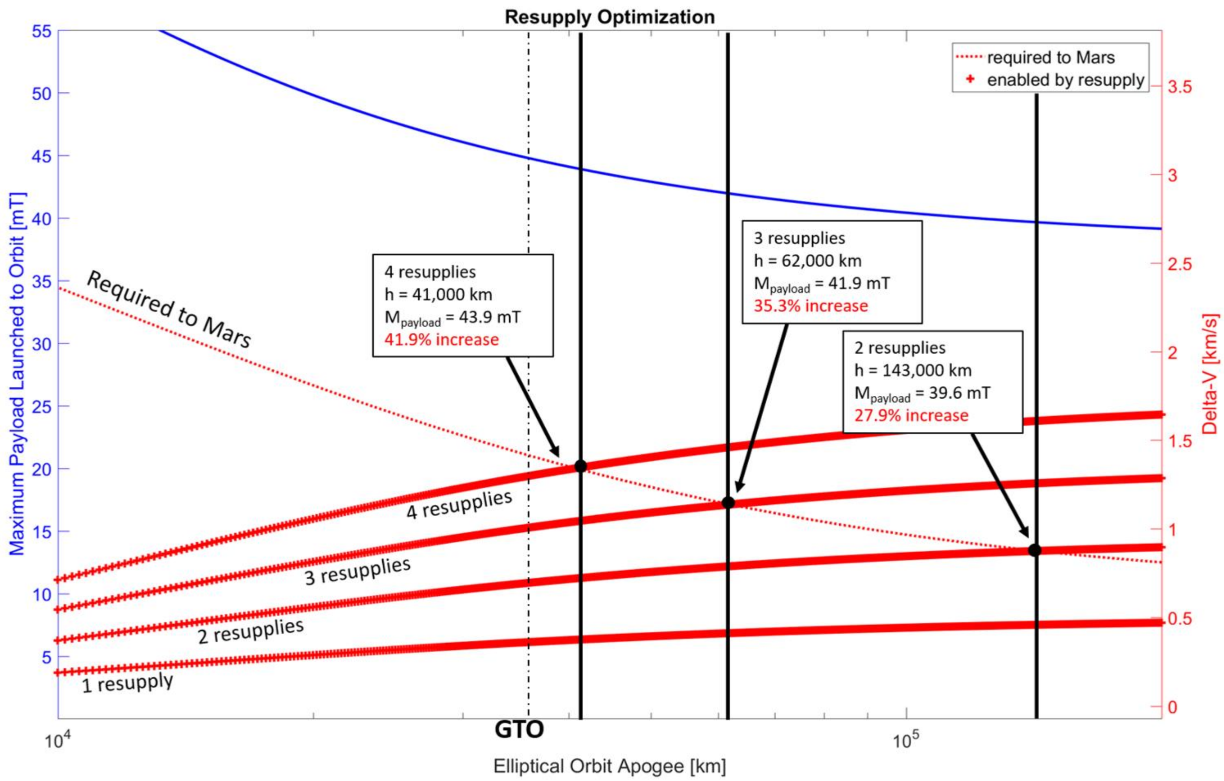


Fig. 5.1.2.2: Resupply Optimization

With two lunar resupply shuttles docking with the EUS at 151,000 km apogee orbit, the payload to Mars is increased by 27.6% (31.7 mT baseline), corresponding to a total propellant transfer of 14.4 mT (7.2 mT per LRS). With more resupplies, the payload can be further increased by decreasing the rendezvous orbit altitude. However, a tradeoff between risk, extraction time, and resupply time must be considered.

5.1.3 Rendezvous

The time of flight of the LRS from the Moon to the rendezvous point at perigee (400 km) in the TEI is 119.5 hours. The optimal elliptic orbit has a semi-period of 64.4 hours. The lead angle can be computed using basic orbital mechanics using this time of flight. No additional maneuvers have to be performed.

5.1.4 Docking

Docking maneuvers have been executed since Gemini 6 mission performed for the first time a docking maneuver at LEO and Apollo did it around the Moon.

The docking maneuvers (discrete maneuvers with coasting phases) are initiated after the rendezvous maneuver. A typical ΔV requirement is about 20 m/s[26], already considered in the total ΔV budget. No robot arm is required[27].

Autonomous and non-autonomous docking (for manned missions) can be performed using either the SLS as the chaser and the LSR as a target or vice versa. This decision can be made in a further iteration of the architecture and should not affect neither the cost nor the concept of the mission.

The navigation system for docking uses GPS (the docking occurs at low altitudes) and an IMU for propagating position and velocity of the chaser using a Kalman filter. The Advanced Video Guidance System (AVGS)[28] can be used for position and orientation estimation at close ranges.

5.2 LEEP Operation Schedule

The R&D phase dedicated to constructing and testing the devices, systems and vehicles necessary to create LEEP as well as operate and perform various tasks on the moon coupled with the actual launching of payload to the moon will be a ten-year endeavor. Once LEEP is completed and fully operational, however, it will be able to accelerate and enhance the efficiency of manned missions to Mars as well as assist with various other deep space missions for years to come.

Beginning with the R&D phase, roughly $6 billion will be directed towards advancing the CML and TRL of various technologies necessary for the construction and operation of LEEP from 2018 - 2023. This development will include continuing the progress of automated landing, reusable rockets meant for receiving fuel while on the surface of the moon then delivering that to various vehicles in orbit; modifying landing systems similar to those used in the Apollo missions meant to land the two stations, one on the rim of the crater and one at the base of the crater, and their encompassing devices; developing ISRU mechanisms meant for processing the regolith by electrolysis in order to convert it into hydrogen and oxygen meant to be used for fuel; testing prospector rovers meant for locating areas on the Moon ideal for extracting ice and extraction vehicles designed to drill into the regolith in order to extract ice then transport it to the ISRU; developing mirror-deflection technology meant for reflecting sunlight from the rim of the

[26] Ely, T. A., Sostaric, R., and Riedel, J., Preliminary design of the guidance, Navigation, and control system of the Altair Lunar Lander, AIAA Guidance, Navigation, and Control Conference, Toronto, Ontario, Canada, 2010.

[27] Brian Roberts, experienced Robotic Technologist from NASA's Goddard Space Flight Center, suggested not to use a robotic arm for large systems.

[28] Zimpfer, D., Kachmar, P., and Tuohy, S., Autonomous Rendezvous, Capture, and in-space assembly, 1st Space Exploration Conference, Orlando, Florida, 2005.

crater to the solar arrays and various vehicles operating within the permanently shadowed crater; designing bulldozer-type vehicles meant to flatten the lunar surface to create a landing pad for LRS's as well as create a road-like structure meant for a smooth passageway to connect various tubes and wires when the LRS is connected to the ISRU. Other technologies and applications such as the solar arrays, communications systems and LV's are at a high enough TRL that we can simply purchase them as they are and apply them towards the LEEP mission.

Once the R&D phase is complete, 4 launches to the moon will be spread out over 5 years from 2024 - 2028. This launch schedule is meant to maximize our $1 billion per year budget coupled with money rolled over from previous years of the mission. Each of the 4 launches will consist of an LRS and a landing system. The LRS will separate from the landing system in LLO and will remain there until it is refueled by the ISRU. The landing system, however, will continue to descend until it reaches the surface and delivers the payload onboard to their assigned destinations. The first launch in 2024 will consist of the components necessary for the station on the rim of the crater. This includes an antenna system meant to communicate with the orbiting LRS and satellites on Earth, a solar array used to power the communication system, and the heliostat systems needed to provide power to the future ISRU and extractors in the crater. The following launches in 2026, 2027 and 2028 will consist of components of the station in the crater. The second launch in 2026 will include two prospector rovers, two bulldozer-type vehicles and a scale model of the ISRU system that will arrive on a later launch which will test the feasibility of the electrolysis process. The third launch in 2027 will consist of the full-scale ISRU system, the solar array system that will receive energy from the mirrors at the rim of the crater and 4 extraction rovers. The fourth launch in 2028 will consist of 8 extraction rovers. After the 4 launches, there will be two stations on the lunar surface, one on the rim of the crater and one within the crater, as well as four LRS's, three of which will always be in LLO and one will always be on the surface getting refueled by the ISRU; thus allowing for a perpetual fueling process which will optimize potential launches.

Assembly of LEEP will begin as soon as the first set of devices land on the moon. The following construction will take place for up to 5 years and will be completed by 2028. Once operational, LEEP will require 26 months to extract regolith, convert it to fuel by electrolysis and deliver that fuel to the 4 LRS's. This 26 month window aligns with the transient Earth-Mars orbit that allows for the most efficient period of transportation to and from Mars. In 2031, missions can begin going to and from Mars by way of refueling SLS systems via LEEP and can continue in successive 26-month windows. Thus, this refueling system can allow for not one mission to the red planet by the end of the 2030's but five.

5.3 Mission Lifespan and Extended Mission Lifetime

We expect LEEP to maintain an operating capability until 2039, with core components designed to a 15 year operating life from the time of lunar insertion, scheduled for 2024. Table 5.3.1 describes the lifespans of each mission component.

Tab. 5.3.1: Lifespan of Mission Components

Component	Estimated Useful Life	Maintenance Strategy
Communications system	15 years	Design life of facility
Rim solar panels	15 years	Design life of facility
Fuel Depot	15 years	Design life of facility
Mobility System	15 years	Design life of facility
Roads	N/A	Roads continuously maintained by robot
ISRU: Conversion	15 years	Design life of facility
ISRU: Extraction	3 years for PVEX component	Robotic replacement from inventory
ISRU: Supporting Infrastructure	15 years	Design life of facility
ISRU: Storage	15 years	Design life of facility
ISRU: Battery Packs	10 years	Robotic replacement via cargo resupply
ISRU: Shelters	Long-lived	low risk of micrometeorite impact
Power Mirrors	20 years	Motors
Lunar Resupply Shuttle	15 years	Design life of facility
Launch and landing site	N/A	Refurbished via bulldozer as necessary
Support equipment	15 years	Design life of facility

The decision to use multiple cheaper vehicles also lends a form a redundancy in that total mission failure does not result from the potential loss of a single vehicle. This architecture also lends itself to incremental development should techniques for aero-braking large fueled upper stages or using advanced solar electric propulsion capable of pushing upwards of 20t between the Earth and moon reasonably. As time goes on, the architecture lends itself incrementally to many of the alternatives initially discarded due to financial or technological constraints.

For example, instead of performing all the Centaur rendezvous in the same elliptical orbit, the SLS upper stage could be boosted up incrementally after every refueling to get the most efficiency out of the fuel deliveries as the subsequent refueling Centaurs will not have to expend as much fuel to reach their target. This sort of maneuver improves the mass-to-Mars performance of LEEP, but requires a level of technical proficiency that must be acquired through experience with in-space refueling, elliptical rendezvous, and other immature technologies.

Additionally, because of the modular nature of the LRSs, if and when nuclear thermal rockets become available we can begin to phase out our chemical fleet and phase in a nuclear fleet. These nuclear thermal rockets may have a specific impulse as high as 1500 seconds, resulting in hugely improved performance characteristics of the LRSs. High-power electric propulsion could also bring about an incremental change in LEEP architecture if and when it becomes technically feasible and has reached technological maturity.

A final example is that as our modular components (e.g. rovers, ISRUs, etc.) wear out and become nonfunctional on the Moon, we can replace them with more advanced versions that improve upon them. Our individual pieces of hardware are small and inexpensive enough that when they have served their useful life, we can effectively abandon them instead of attempting repairs. This simplifies operations on the Moon immensely and allows for continuous upgrades to our lunar technology.

Because of it modular architecture, LEEP will for the foreseeable future be able to take advantage of improving technologies and growing capabilities, making the concept of a Lunarport more and more attractive as the years go on. An investment now leads to a hugely capable and technologically advanced infrastructure in the future.

6. Environmental Risk Mitigation

6.1 Regolith Protection Design

Continuous and prolonged exposure to the lunar regolith can be dangerous to robotic equipment. During the Apollo missions, the fine regolith particles caused jamming of the mechanical equipment and the seals of space suits, causing dangerous pressure losses. The dust impaired the proper operation of seals and lubricants used on various mechanisms and also accumulated heavily on exposed optical surfaces. "In addition, the atmosphere and internal surfaces of the lunar excursion module were contaminated by lunar dust which was brought in on articles passed through the airlock."[29] All the mechanical systems that will be placed on the lunar surface to accomplish the construction and sustained running of the Lunar Port, will be exposed to the lunar regolith continuously and it is essential to have a protective design parameter embedded into all the robotic machines to prolong their lifetime and ensure their smooth functioning during the entire project timeline.

Solution: The standard practices for dust mitigation; like implementing polished outer surfaces, designing and sizing after accounting for the loss of efficiency, etc.; that are already well established and used extensively in almost all the modern day space missions will be implemented. Additionally, Electrodynamic Dust Shield (EDS) technology will be used for all the equipment surfaces. The EDS technology involves thin wires made from conducting films which are embedded in surfaces and can be made transparent if necessary in case of using over optical surfaces. It creates an electric field through these wires which propagates outward in a transverse wave like motion carrying the dust particles along. EDS can be extended to space suits too. A brief list of specific implementation is as follows:

- For solar panels: Indium-tin-oxide wires
- For reflective films on rovers, landers, and ISRU unit: Aluminum or Silver wires
- For space suits: Carbon nanotube wires

The power requirement for EDS is just on the scale of milli-Watts. Also, dust-tolerant utility connectors and related mechanisms[30] that Honeybee Robotics Spacecraft Mechanisms Corporation has been developing with a focus on lunar surface system applications will be requested from Honeybee along with their extractors for this mission.

---

[29] Belden, Lacy, Kevin Cowan, Hank Kleespies, Ryan Ratliff, Oniell Shah, and Kevin Shelburne: Design of Equipment for Lunar Dust Removal, The NASA/USRA University Advanced Design Program, University of Texas at Austin, 1991

[30] Herman, Jason, Shazad Sadick, Michael Maksymuk, Philip Chu, and Lee Carlson: Dust-tolerant Mechanism Design for Lunar and NEO Surface Systems, Aerospace Conference, 2011 IEEE

6.2 Radiation Protection Design

The surface of the Moon is heavily exposed to cosmic rays and solar flares, and high energy radiation can penetrate through thick walls. Also, when cosmic rays hit the ground, they produce a splash of secondary particles. Radiation exposure can damage the electronics of a robotic system and that can render the entire system incapacitated. Most modern-day space robotic systems are defaulted to have higher wall thickness and aluminum shielding to be protected from radiation. And these standard measures can be implemented. However, an additional solution can be implemented to elongate the lifetime of the robotic equipment.

Solution: An additional solution could be polyethylene shielding. Polyethylene is great candidate for radiation shielding because Hydrogen is one of its major constituents and Hydrogen is capable of shielding against energetic particles. All the rovers and landers can be supported with inner polyethylene shell with a thickness roughly between 1 cm to 6 cm depending upon weight and cost optimization. NASA's ongoing research on Hydrogenated Boron Nitride Nanotubes (BNNTs) is currently in its development and testing phase. Hydrogenated BNNTs can strike a much-desired balance between effective radiation shielding and weight optimization, the only drawback being the nascency of the technology, which might be accounted for considering the number of development years available for the Lunar Port. Hydrogenated BNNTs can be incorporated for space suits too as they are flexible and can be woven into a fabric. This material has proven strong enough to be used at the structural material, but we can choose to incorporate it only for shielding owing to the time and cost constraints.

6.3 Thermal Control

Lunar surface temperatures vary drastically due to the lack of atmosphere. Furthermore, the temperatures in the permanently shadowed regions in the craters are even lower, reaching dangerous extremes. Currently and conventionally, rovers and landers rely on nuclear or electric heating systems to keep their electronics warm, but we will not be using that due to external constraints of cost and power. Without special heating systems the electronics and other systems of rovers, experiments, spacecraft, and habitats would eventually freeze into inoperability.

Solution: We intend to use heat patches and thermal insulation to protect the robotic equipment from extreme temperatures. The development years for the lunar port can be utilized to improve upon the materials used for these heat patches and insulations in order to extrapolate the existing technology to cater for the even lower temperatures in the permanently shadowed regions in the lunar craters. Apart from this, there are two new approaches currently being researched both by NASA and ESA for thermal protection that will turn out to be cheaper alternatives. The first approach involves using reflectors to heat patches of the lunar soil to form "thermal wadis," or hot spots, which can then be tapped at night to keep the devices warm enough to function. The second one involves using reflectors and heat pipes to collect solar energy to run the equipment directly during the day and to store excess heat to keep the larger installations warm. The two are quite similar and either can be considered as upgrades for the existing infrastructure that will be built in this project and can be incorporated into the development plan.

7. Human Factors

7.1 Interfaces for Human Operations

Although this mission does not include a human component, there are interfaces with humans in teleoperation, the refueling of the Mars crewed vehicle, and potential external missions.

Mission Philosophy:

Be human **safe,** but not human **dependent.**

Teleoperation & Extension Missions:

Humans on the ground interface deal with a several second time delay (4-25 seconds depending on connection and encryption). This system would also allow astronauts in lunar orbit to control and tele-operate the systems on the ground. Research done by Mark Lupisella[31] and others at NASA Goddard into low latency teleoperation for Phobos to Mars teleoperation can be tested in the Earth-Moon system.

Future Human Missions:

With the robotic and communications technology development completed over the course of the mission, future human missions benefit from increased TRL levels of landing technology, rovers, communications, and systems architecture.

Human Safe:

The mission architecture lends itself to a human safe environment. Future human missions that want to do science, set up a radio telescope, or set up a manned facility on the moon will benefit from the lack of concerns about nuclear power. Early trade studies would have used RTGs to power the rover, but the mirrors enabled a nuclear-free mission, which eliminates radiation concerns for astronauts working nearby.

7.2 Medicine and Radiation Considerations

Some of the medical considerations for humans come from the possible health hazards that might be the result of prolonged stay in the lunar environment. The major problems are: 1) Radiation exposure, 2) Regolith exposure, and 3) Thermal exposure. Radiation exposure is a serious problem to the human body because high energy radiation can pass right through the skin, depositing energy and damaging cells or DNA along the way. This damage can mean an increased risk for cancer later in life or, at its worst, acute

[31] Lupisella, Mark, et. al. "Low Latency Teleoperations for the Evolvable Mars Campaign." Future In Space Working Group. 7 September 2016. http://spirit.as.utexas.edu/~fiso/telecon/Lupisella-Bleacher-Wright_9-7-16/Lupisella-Bleacher-Wright_9-7-16.pdf

radiation sickness during the mission if the dose of energetic particles is large enough[32]. Being exposed to the lunar regolith is also a significant issue because the regolith attack causes problems to the space suits as well, apart from the mechanical systems on the lunar surface. During the Apollo missions, the abrasive and chemical reactive nature of regolith in the form of micron sized angular silica oxide particles, created problems for the space suits after only one EVA. Numerous procedures were put in place during Apollo to attempt to mitigate the problem but with limited success[33]. And if the lunar regolith dust get into an airlock, it can be toxic to the lungs. Extremely high lunar temperatures can cause hypotension, fatigue, breathing difficulties, confusion, and fainting. And the extremely low ones can cause frostbite and hypothermia with symptoms of initial pain, weakness, loss of coordination, slurred speech, little or no breathing, and gradual loss of consciousness[34].

Solution: The EDS technology can be extended to space suits as well as far as regolith shielding is concerned, and it has already been explained in section 6.1. Similarly, the radiation shielding solution implemented for the robotic equipment, i.e. the polyethylene shielding, can be extended to space suits as well. This has been explained in section 6.2. Apart from that, another possible solution, which is lighter and cheaper as well, could be the newer modifications to anthropomorphic protective covers for space suits. DuPont Inc. USA has been researching Space Suit cover configurations for their 9 mil thick DuPont Tyvek cover and it has been showing promising results. The DuPont Tyvek material basically is comprised of high-density polyethylene fibers and hence could essentially be used for radiation shielding as well after making necessary modifications in terms of thickness. As far as thermal protection goes, the astronauts can be protected by their space suits, just like the Apollo astronauts, for whom the space suits were designed with several layers of insulating materials. The radiation and regolith shields will be the additional layers over the space suits.

There are some more general problems arising due to microgravity and it is unclear as to how severe these might actually be on the lunar surface as we still have 1/6th of the earth's gravity there. Nevertheless, these problems can also be taken into consideration. They are: 1) Shift of intravascular and extravascular fluids in the human body and 2) Bone demineralization and Muscle atrophy.

Solution: We will have a Health Monitoring Computer with data collection and storage system with a self-contained medical expert scheme for performing treatment protocols. The expert system has 'data driven' and 'time driven' capabilities to facilitate automatic decision-making functions. It will hold an integrated

---

[32] https://www.nasa.gov/feature/goddard/real-martians-how-to-protect-astronauts-from-space-radiation-on-mars

[33] Cadogan, Dave, and Janet Ferl: Dust Mitigation Solutions for Lunar and Mars Surface Systems, ILC Dover LP, 2007 SAE International, 2007-01-3213, 2007

[34] Nancy J. Lindsey: Lunar Station Protection: Lunar Regolith Shielding, International Lunar Conference 2003, Hawaii Island, 2003

medical record and medical ‘reference’ information management component along with an inventory management system for medical supplies and pharmaceuticals. It will also have capabilities to facilitate video, audio, and data communications between the crew member on the lunar surface and ground-based medical personnel. While, monitoring the health parameters of the crew on the surface, we will simultaneously test the increased capabilities of the system after having incorporated the advanced algorithms. We can use the current Computerized Medical Decision Support System from NASA, but try to implement the advanced A.I. algorithms into it during the preparation phase and greatly improve its capabilities. These advanced A.I. algorithms can be developed and implemented by NASA in collaboration with Electrical & Computer Science Departments at Massachusetts Institute of Technology (MIT), Cambridge and Tufts University School of Medicine, Boston where the actual relevant research has been going on.

8. Programmatic Considerations

8.1 Activity Timeline and Cost Analysis

8.1.1 Activity Timeline

The mission will be developed, fabricated, tested, validated, and deployed over twelve years. The schedule for these activities are decomposed by each unique element that must be developed for the program. Each of these will need to include many sub-activities themselves. The highest level schedule is shown below. Note the SIR, ORR, and FRR technical reviews for LEEP before initial operations.

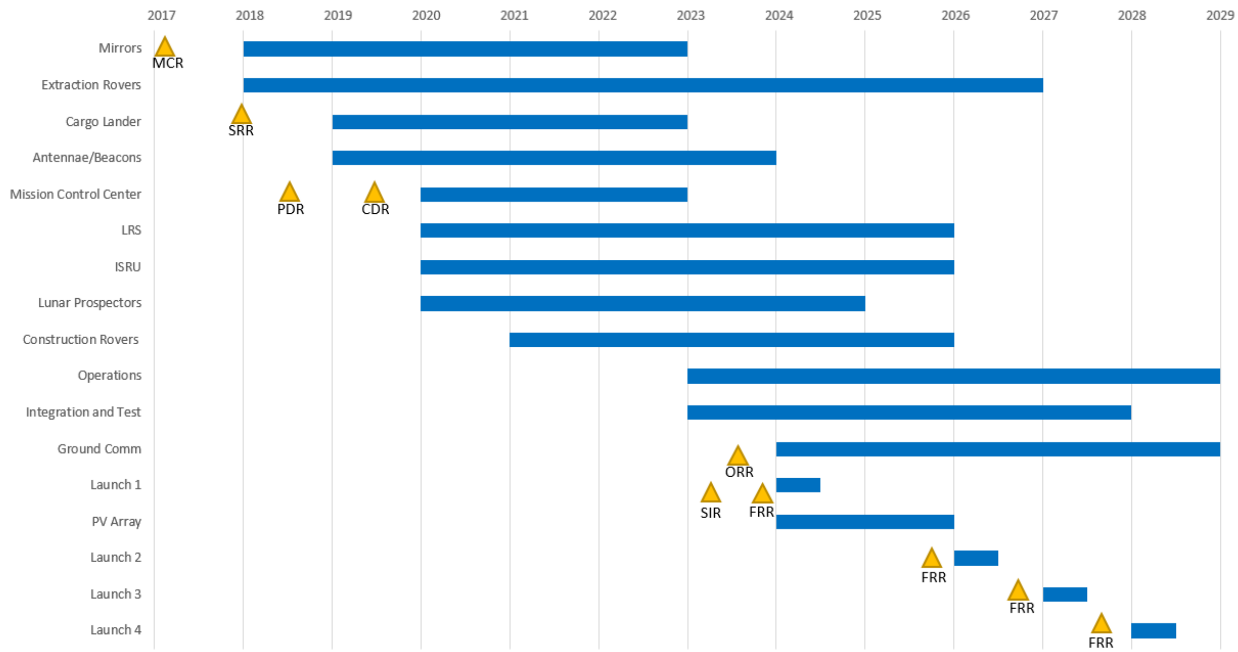


Fig. 8.1.1.1: Activity Timeline

8.1.2 Cost Analysis

8.1.2.1 LEEP Overall Costs

The performance of the LEEP system can, in part, be valued by the cost required to provide propellant to customers. The total mission cost is decomposed into non-recurring and recurring costs for each year of LEEP's development and operation. Each element of the system (e.g. LRS, extractor rover, power system, mission control center, personnel, etc.) is costed by analogy or parametric model and separated into development cost and per-unit cost. Margins are included for each element. Program management, system engineering, mission assurance, and integration and test are also necessary to account for. Finally, a large reserve is set aside for possible changes that may occur in such a complex development program. Additional details are given in the appendix of this report.

Tab. 8.1.2.1.1: Non-Recurring Cost Table

NON-RECURRING COST				
Mission Hardware		**Item**	**Cost**	
	Ground Seg			
		Construction Rovers	$	649,770,250
		Lunar Prospectors	$	397,052,475
		Extraction Rovers	$	2,801,001,935
		Antennae/Beacons	$	2,400,000
		Mirrors	$	220,000,000
		PV Array	$	120,000,000
		ISRU	$	1,000,000,000
		Cargo Lander	$	200,000,000
	Space Seg			
		LRS	$	355,000,000
	Earth Seg			
		Mission Control Center	$	420,000,000
Launch costs				
		Launch Vehicles	$	522,400,000
Integration and Test	0	Integration and Test	$	459,617,973
Program Management				
	Prog Level	Mgmt, SE, MA		$714,724,263
		Reserves (30%)	$	2,358,590,069
Total Non-Recuring Cost			**$**	**10,220,556,965**

The non-recurring cost of the system estimated to be approximately $10.84B to reach the capacity to supply a Mars mission every opportunity. The recurring annual cost for steady-state operations after the initial emplacement phase is completed is $0.103B. This recurring cost includes personnel for mission operations, communication time, and replacement of major failed or degraded elements like photovoltaic arrays. The largest development costs are the extraction rovers and ISRU system. These are relatively low TRL, high-complexity elements and will require resources to develop the fundamental technologies for use on the hostile lunar surface with high reliability. The specific cost of the program ($420,000/kg) is comparable to Mars Science Laboratory ($700,000/kg).[35][36]

[35] Mars Science Laboratory Spacecraft, https://mars.nasa.gov/msl/mission/spacecraft/

[36] Mars Science Laboratory Landing, https://www.jpl.nasa.gov/news/press_kits/MSLLanding.pdf

Tab. 8.1.2.1.2: Recurring Cost Table

RECURRING COST				
Operations costs	Earth Seg			
		Ground Comm	$	660,000
		Operations	$	41,704,813
	Space	Resupply Launch	$	23,400,000
		Resupply Rover	$	13,000,000
		Resupply LRS	$	18,460,000
Total Recurring Cost			**$**	**78,764,813**

The recurring costs for the system include ground communications service payments, operations for MCC and fabrication of replacement equipment. The budget also includes costs for planning a major replacement and launch of a rover and LRS every five years. For reference, the ISS has required a total cost of approximately $160B (15x LEEP) and has a $3B (29x LEEP) annual budget.[37]

8.1.2.2 LEEP Costs over Time

The LEEP annual cost is capped at $1B, but extra funds can be saved for future year's development. This is a significant benefit because it allows cost spreading without losing efficiency so that resources are allocated appropriately for early concept development through fabrication, testing, and assembly. It also serves to decrease to initial spike in funding for emplacement, deploy systems in a meaningful order, and to be able to gain knowledge on systems during early missions [3]. The LEEP team determined the total system lifecycle cost using engineering build-up phasing based on the lunar emplacement schedule and required development to meet it. The following figure includes all costs from 2018 to 2030 including reserves. The plot has been shown on the next page.

[37] http://www.space.com/24208-international-space-station-extension-2024.html

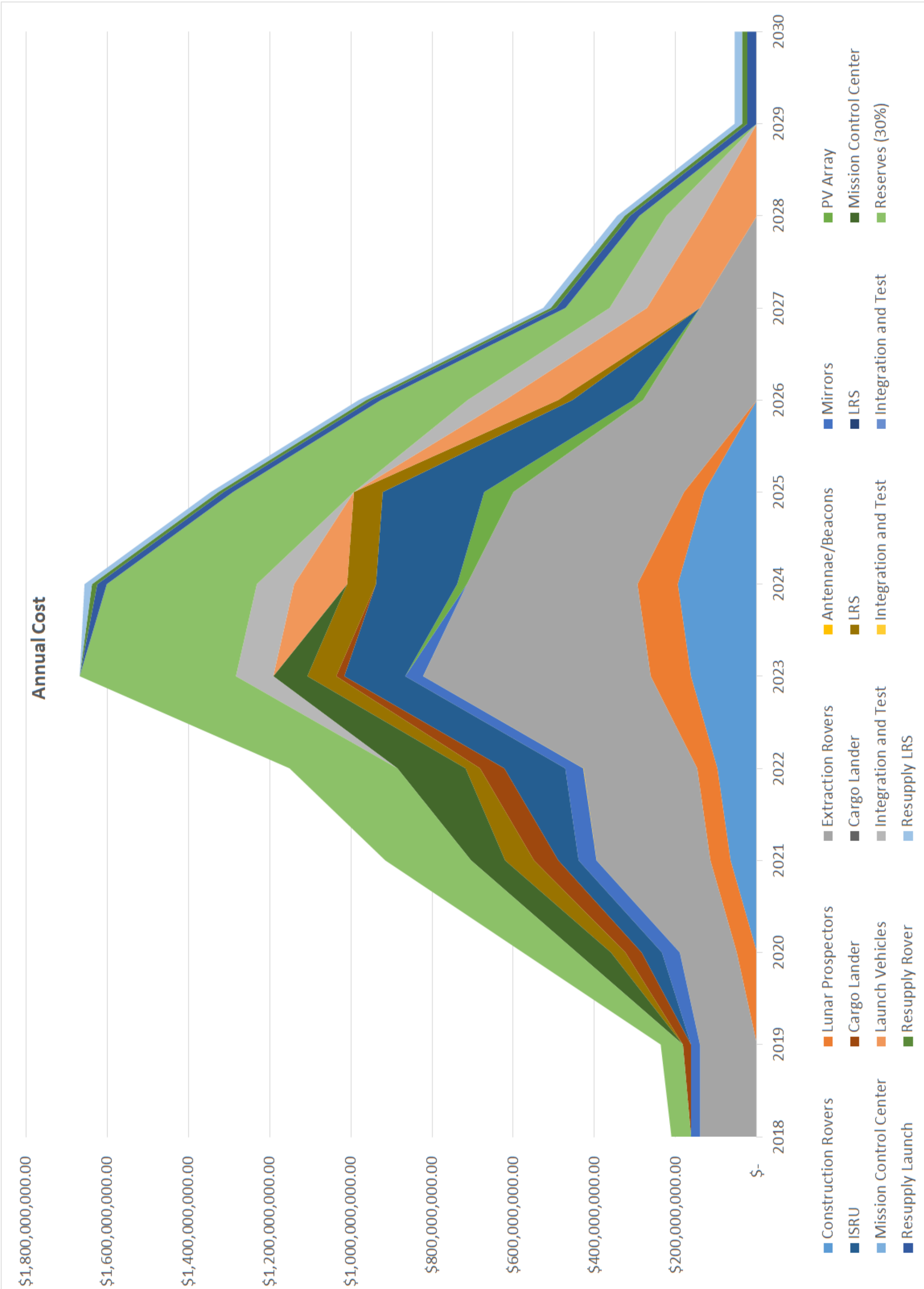


Fig. 8.1.2.2.1: Annual Cost for every year

The overall LEEP project cost follows an approximate 40:60 or 50:50 beta curve. This is typical for large-scale aerospace programs and has been seen historically in Apollo, Gemini, Mercury, and Skylab [Lafleur, 2010, Costs of US Piloted Programs, http://www.thespacereview.com/article/1579/1]. The program cost will peak in 2024 at $1.67B when the first deployment mission happens. By 2029 LEEP only requires continuing steady-state operations where the program will also prepare for resupply missions which may cause relatively small increases.

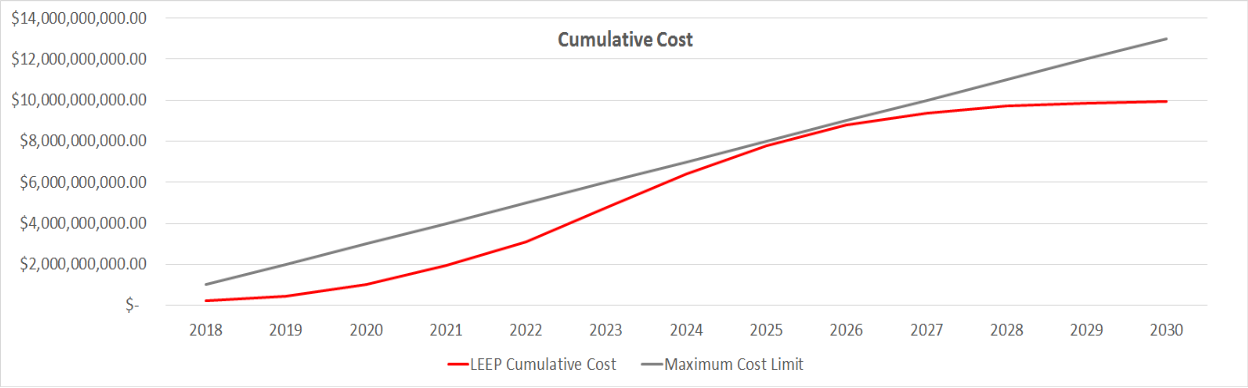


Fig. 8.1.2.2.2: Cumulative Cost over the years

The cumulative cost over time, starting in 2018, does not match the available budget due to the spending peak. The figure above shows both the LEEP cumulative cost and the maximum possible cost ($1B x years). While the annual budget is underutilized in the early years, by 2026, the banked resources have accounted for. The difference for future projects can then be used for alternative projects as the annual costs are only a fraction of the $1B.

8.1.3 Mars Mission without LEEP

In order to determine the break-even point for the LEEP system, the system's value is measured in payload mass through TMI (trans-Mars injection). The payload mass of the SLS Block II with EUS is 31.7 mT. We assume a cost of $1B dollars per SLS launch with a cadence of one per each 26-month Earth-Mars synodic period. The specific-cost of payload mass to TMI given these assumptions is $31,500/kg.

8.1.4 LEEP Cost Performance for Mars Mission

One of the primary purposes of LEEP is to allow for more payload for exploration mission or to reduce the costs of those missions in the future. If LEEP is successfully implemented, it will enable an increase in payload to Mars by 30% for a single SLS launch with EUS. The cumulative payload for a campaign of launches can be seen in the figure below. After ten launches mission planners could take advantage of 100 mT of extra payload by refueling at LEEP on the way. The specific cost using LEEP is only $22,700/kg to

TMI after accounting for operations and resupply costs, a significant reduction. This cost is equivalent to current cost of Soyuz to GTO[38].

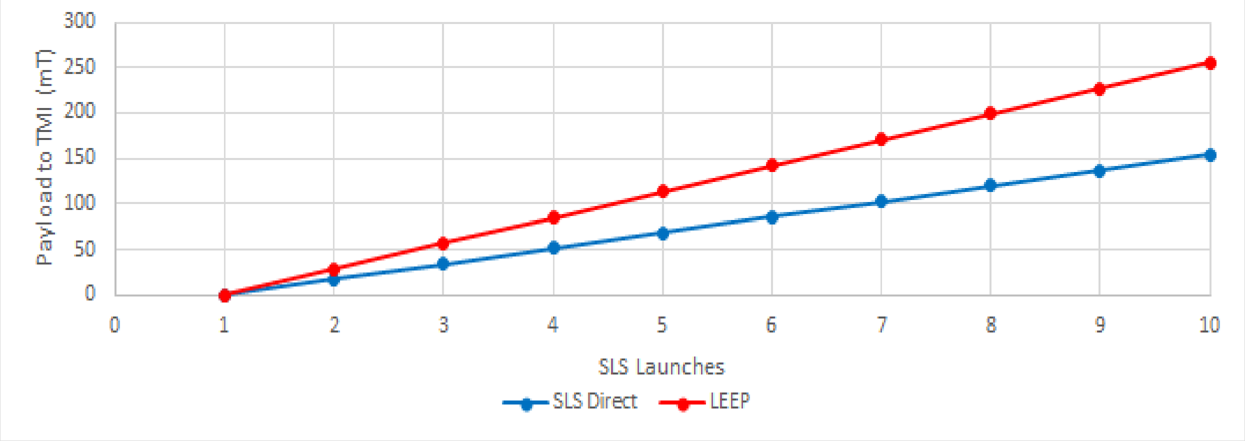


Fig. 8.1.4.1: Payload to TMI vs Number of SLS Launches

This additional capability does not come without cost; the capability to produce propellant refuel a deep space transportation stage first requires the large investments discussed above to build the lunar infrastructure. The total cost of an Mars mission using LEEP can be measured by the sum of initial non-recurring cost, the recurring annual cost, and the SLS launch cost.

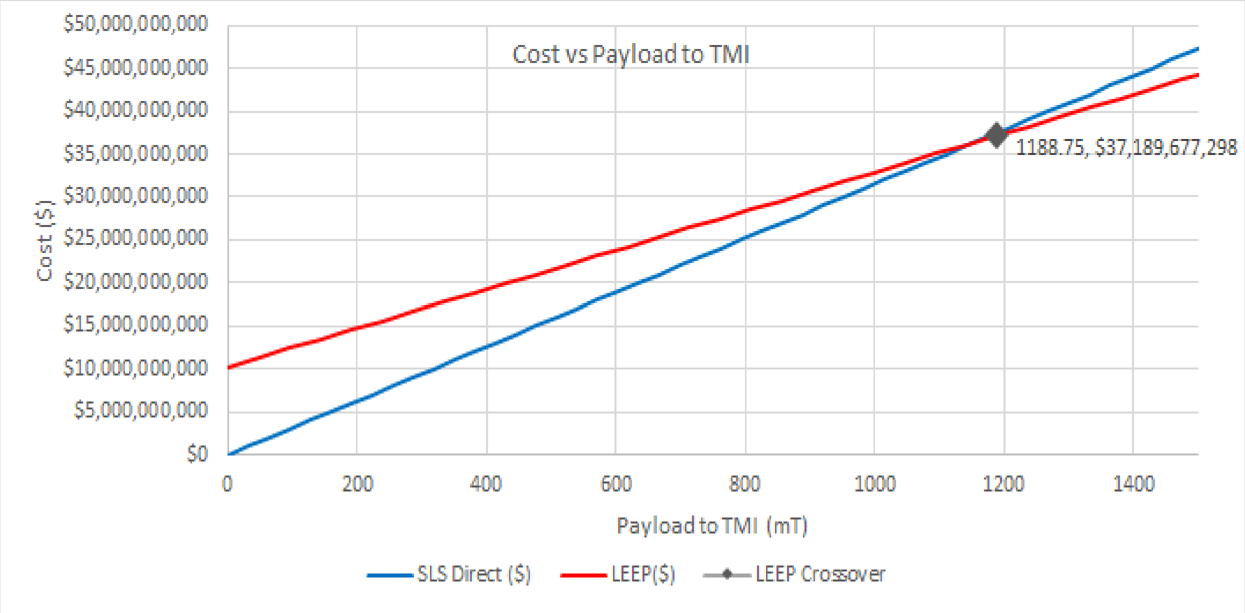


Fig. 8.1.4.2: Cost vs Payload to TMI

[38] Development and transportation costs of space launch systems, http://www.dglr.de/fileadmin/inhalte/dglr/fb/r1/r1_2/06-Raumtransportsysteme-Kosten.pdf

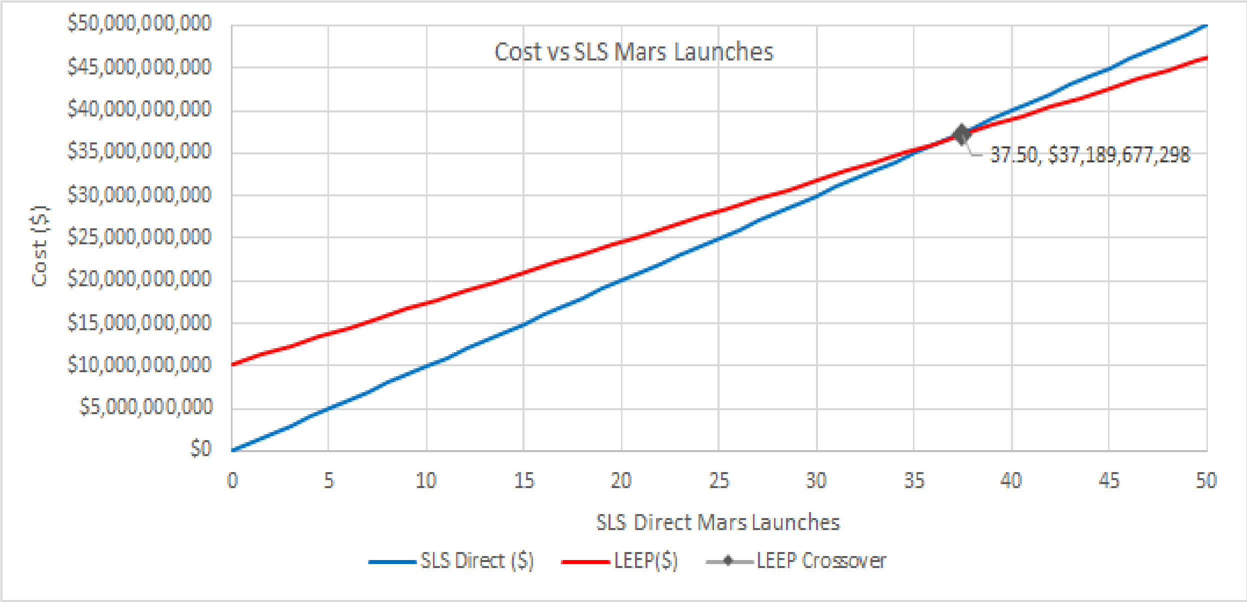


Fig. 8.1.4.3: Cost vs SLS Mars Launches

The break-even point for LEEP is approximately 1190 mT in TMI, equivalent to 38 SLS EUS missions. The total cost spent at this point would be $37.2B. This is a significant number of missions and payload mass to deep space. However, the LEEP project is a long-term investment which can be used for many years for other applications. Smaller launch vehicles than the SLS which would never be able to reach beyond the Moon or land on its surface will now have access to those locations. Other mission types and customers will also be able to benefit from the system including robotic and crewed missions. It is also expected that the modular, flexible nature of the design could allow for additional investment to expand capacity.

8.1.5 Funding Sources

Given the United States' strong, long-term support for Mars exploration, NASA is a clear funding target for the majority of LEEP's $1B annual funding profile. However, and cost-sharing is an important priority for the U.S. government as a whole. We therefore expect that foreign space agencies--most likely beginning with other partner members of the International Space Exploration Coordination Group (ISECG)[39]--would also provide in-kind or annual cash contributions. For context, $1B/year would represent approximately 5.2% of NASA's FY17 proposed budget of $19.1B.

Furthermore, as will be further discussed in section 8.x, we believe that the private sector has much to gain from participating in LEEP, including as a funding partner under appropriate circumstances. The private

[39] The 12 members are Canada's CSA, the European Space Agency, France's CNES, Germany's DLR, India's ISRO, Italy's ASI, Japan's JAXA, Russia's ROSCOSMOS, South Korea's KARI, Ukraine's SSA, the United Kingdom's UKSA, and the United States' NASA.

sector has significant funding they are willing to invest in deep space exploration: Amazon founder Jeff Bezos has invested over $500M in his Blue Origin, SpaceX head Elon Musk has said he has begun raising $10B to support his vision of a Mars colony, and SpaceX recently closed a $1B investment from Boeing. Private sector firms could contribute to LEEP via in-kind contributions or cash transfers; in return they could gain early access to relevant technology on a non-exclusive basis or perhaps free propellant on a non-priority basis. Gaining early financial buy-in from the private sector would further institutionalize NASA's policy of developing new economies that can be handed off to markets when sufficiently mature so that the Agency can continue pushing frontiers.

Between foreign space agencies and private partnerships, we assume that 20% of the $1B annual funding expense could be covered by a 3:1 ratio, respectively. This division is a reasonable first approximation because 11 agencies have joined the United States in the ISECG and many of these countries have made multi-billion dollar investments in the ISS. 20% is Further, the total U.S. contribution to date for the ISS represents approximately 50% of the overall expenditure, suggesting 20% is within the realm of possibility.[40] Figure 8.1.5.1 illustrates the cumulative costs borne by each partner during the development and construction phases, with operations costs estimated to be significantly cheaper and not worth analyzing more closely. Over the twelve-year construction phase of the mission, partnerships at this level of effort would save NASA $2.4B.

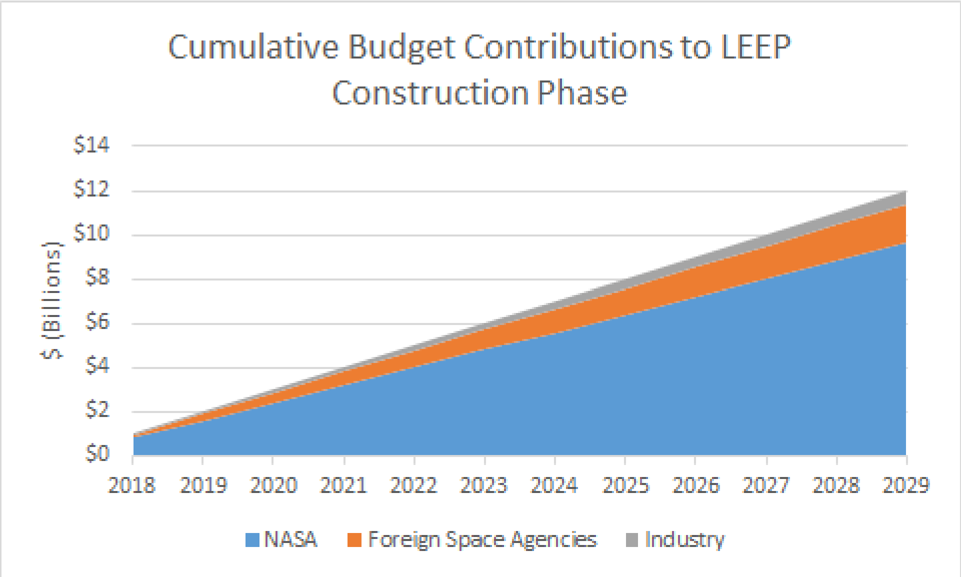


Fig. 8.1.5.1: Cumulative Budget Contributions in LEEP Construction Phase

40 Minkel, JR. "Is the International Space Station Worth $100 Billion?" Space.com, November 1, 2010. Available online at http://www.space.com/9435-international-space-station-worth-100-billion.html.

8.2 Risk Analysis

Our risk analysis is based on the methodology "New risk scoring guidance" developed at NASA JPL (see NASA Risk Analysis handbook in the References). The TeamX is the team who developed and uses it. Team X is a cross-functional multidisciplinary team of engineers that utilizes concurrent engineering methodologies to complete rapid design, analysis and evaluation of mission concept designs.

Our study is divided in three levels: system, space segment and ground segment. Those parts gather the most important risks faced by the LEEP concept.

8.2.1 Risks at the System Level

Budget

The first system-level risk we identify is related to the budget. As illustrated in the graph entitled "Risk Analysis Matrices - Costs", there is a high risk to be over budget with the LEEP project. The annual budget of $1 billion should not be exceeded. One way to mitigate this risk is to take important margins. In our project, we took an overall 30% reserves budget with some extra margin on elements with a low TRL. Another option is also to take a maximum of off-the-shelf elements when it is feasible. This decision comes with two benefits: these commercialized elements have a long heritage and they are usually cheaper.

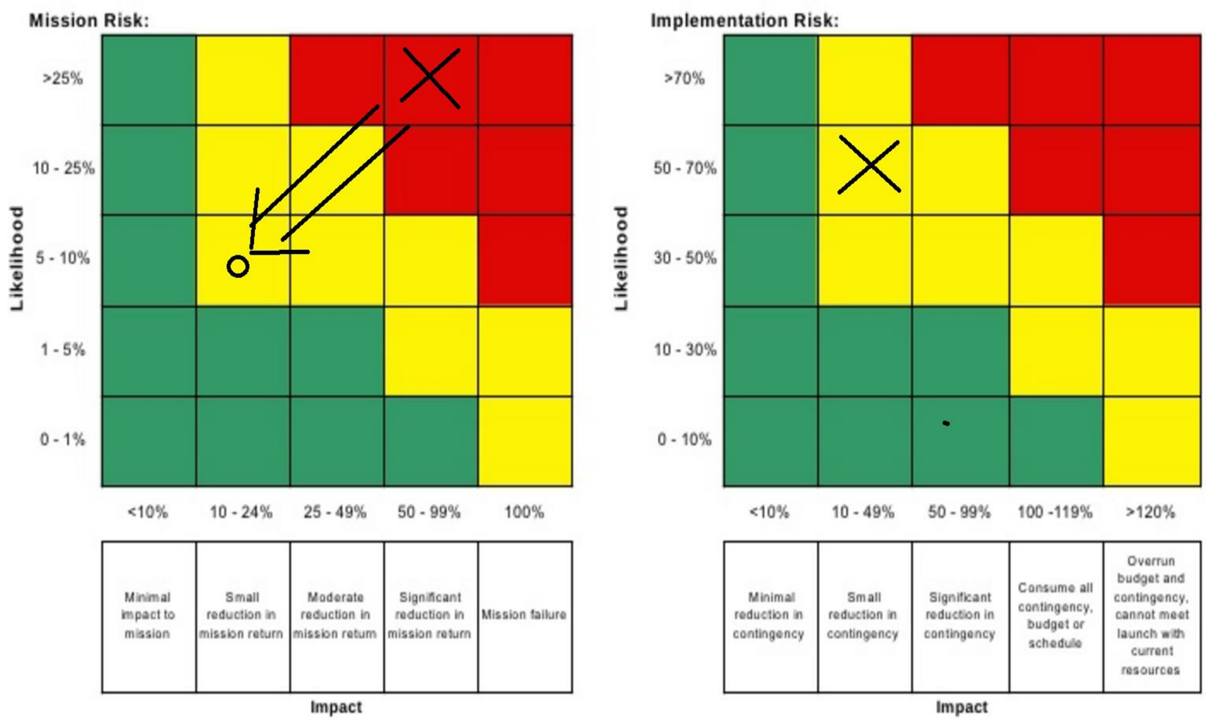


Fig. 8.2.1.1: Risk Analyzing Chart - Budget

Service Availability - Single Point of Failure

If a scientific mission to Mars is planned in advance with a refueling option with our LEEP and after its launch, we are not able to refuel our customer properly (lack of propellant, LRS explosion etc.…), the whole scientific mission is a failure. Indeed, the spacecraft is waiting on the rendezvous orbit with an insufficient amount of propellant to achieve the mission.

In that case, the LEEP was not able to fulfill its role. The matrices of the impact of this risk and the likelihood to occur are illustrated below.

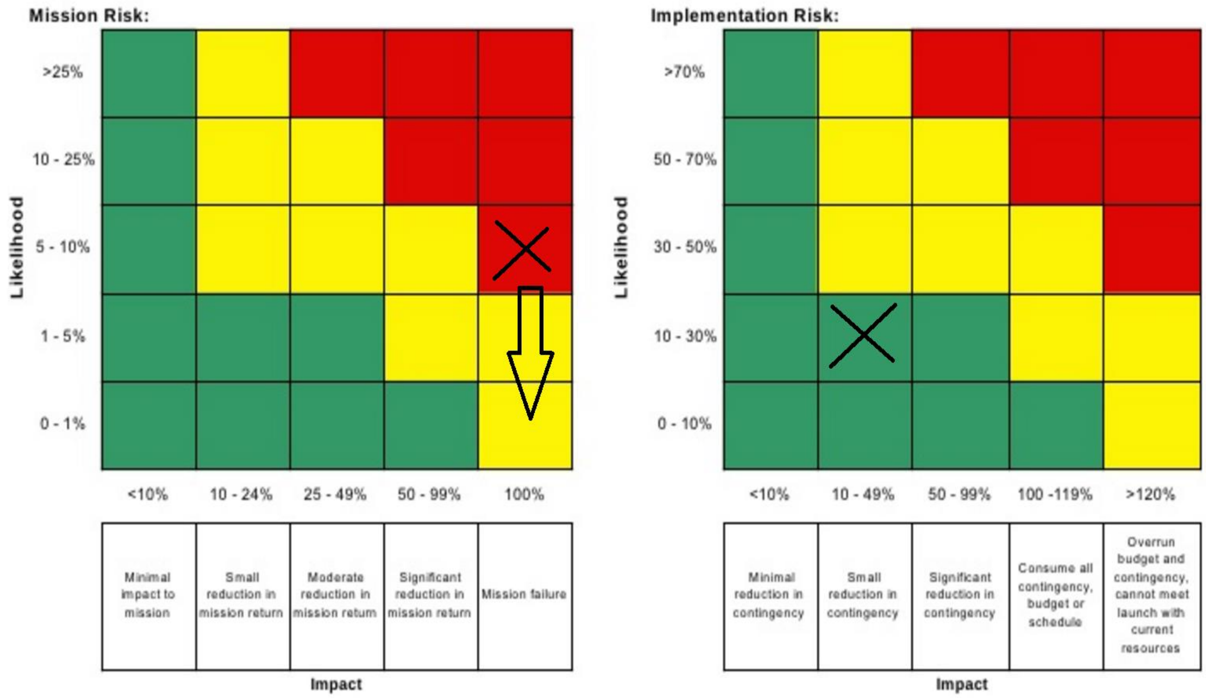


Fig. 8.2.1.2: Risk Analyzing Matrices - Service Availability - Single point of failure

One way to mitigate that is to install a reservoir station which can be loaded by multiple LRS. This reservoir station would then stay in the rendezvous orbit and be docking to all deep-space missions. We did not consider this option in our solution because it was out of budget but as our system is agile, it would be feasible to implement such a solution if the budget of the mission increases.

8.2.2 Risks at the Space Segment Level

On the space segment, we have listed the four highest risks our project is facing. They are listed in order of importance. Each risk has been covered on a separate page.

Crash during rendezvous

The highest risk concerning the space segment is the collision of the LRS with the deep-space missions. The mission risk is at 100% in that case. However, this event is not likely to happen because the rendezvous technologies have been studied, designed and tested for years now. As far as 2017, not a single docking failed.

The problem can be easy to tackle and the mitigation here is the implementation of a safe control mode in the software and the use of a robotic arm. In the case of a manned mission to Mars/an asteroid, the crew can deploy and the robotic arm to catch the LRS. The rendezvous matrices are presented below.

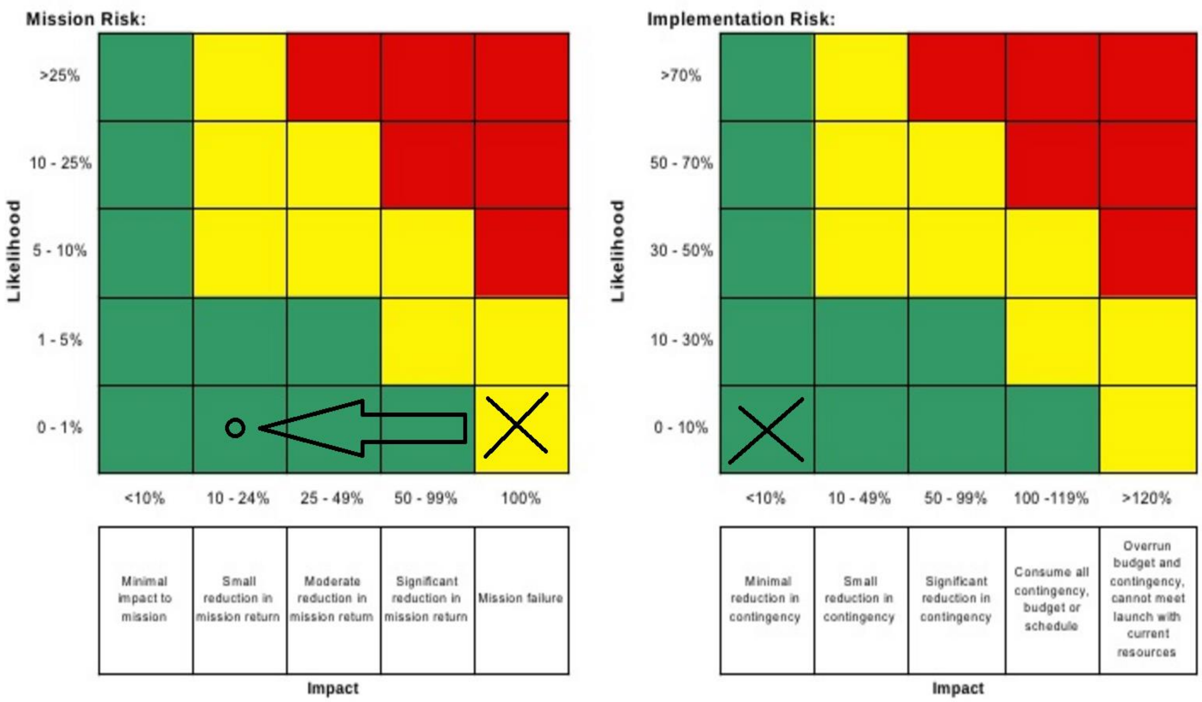


Fig. 8.2.2.1: Risk Analyzing Matrices - Crash during rendezvous

Landing

The landing risk is the risk of losing a LRS during the landing phase on the Moon ground. In 2017, reusable rocket is no longer pure theory and different rockets succeeded. However, the technology is not highly mature so we assessed a likelihood of 5-10% with an important impact on the mission (50-99%).

The possible mitigations are an improvement of the algorithms and sensors combined with smoother pads with a backup LRS, which can be cheap in our case as we are considering Centaur as LRS. The matrix has been shown on the next page.

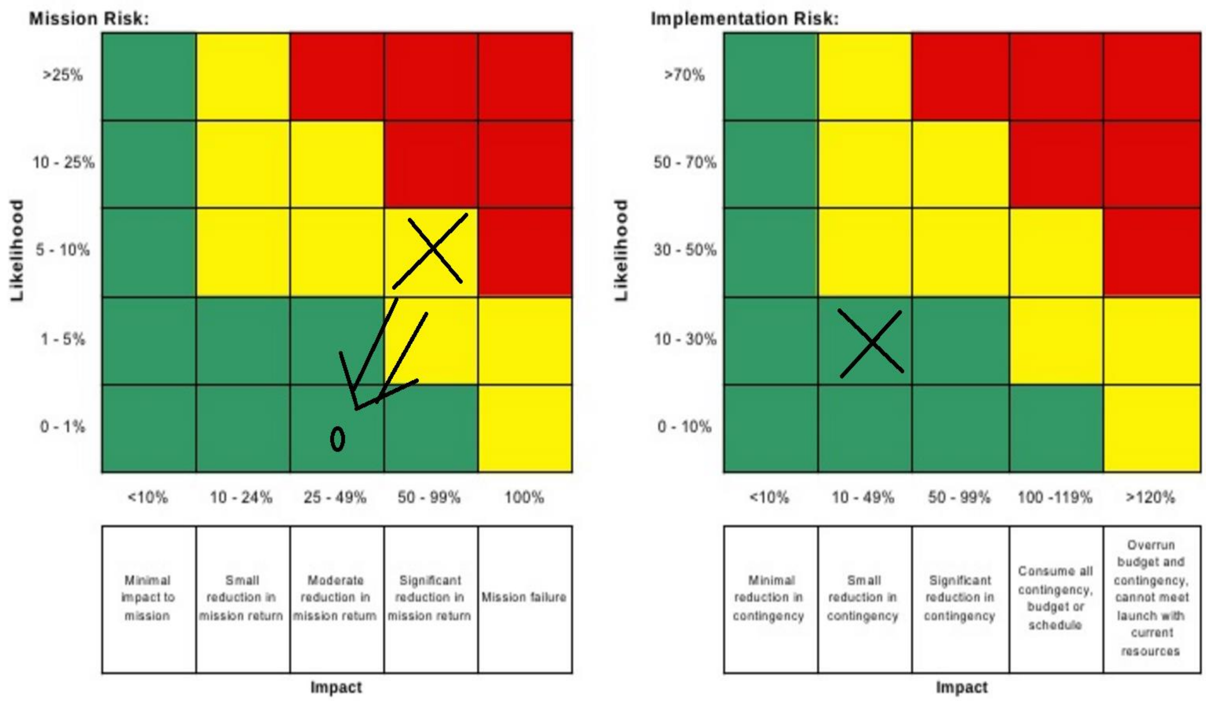


Fig. 8.2.2.2: Risk Analyzing Matrices - Landing

Leaking

One of the risk of transferring propellant from one spacecraft to another is partial or complete fluids losses. It will be hard to controlled this leak in real-time because of the existing communications delay. As the refueling technology is a new concept which had been tested on a few numbers of satellites, the likelihood is between 5 and 10%.

We can significantly reduce the impact of those by adding sensors and security checkpoints throughout the refueling process. Plus, when Landsat-7 has been refueled in space, the satellite was not designed and built to accept refueling. Procedures for refueling will be set in place allowing refueling to be easy, safe and fast.

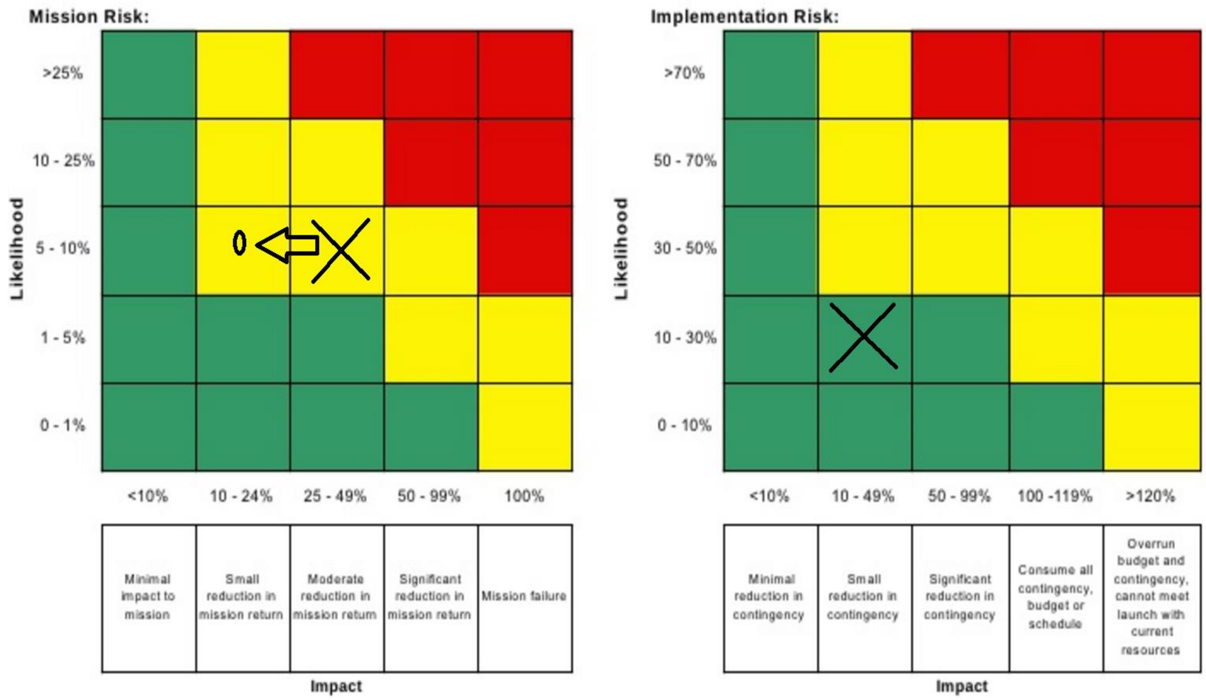


Fig. 8.2.2.3: Risk Analyzing Matrices - Leaking

Zero boil-off

One risk involved in transferring the fuel is that too much of it boils off during transport. This becomes more likely if there are unexpected delays, problems with weather, hitches in our supply flow, etc.

If this happens, we can still complete the mission but may need to bring in another LRS to pick up the slack, which cuts into contingency and means that we have less fuel available to resupply other missions and less stock available for further contingency situations.

To mitigate this risk, we implement zero boil-off technology involving 60-layer MLI on the hydrogen tank and ULA's IVF system for integrating vehicle fluids and limiting boil-off. This technology will very likely be mature without an investment from us by the time we are building our LRSs, and having the systems on board decreases the likelihood of having a boil-off event that necessitates depleting our contingency, reducing the likelihood of the failure but not the severity of the problem should it arrive. Additionally, because the modifications planned are already known to work or are being developed by third parties, there is little impact on our overall mission contingency.

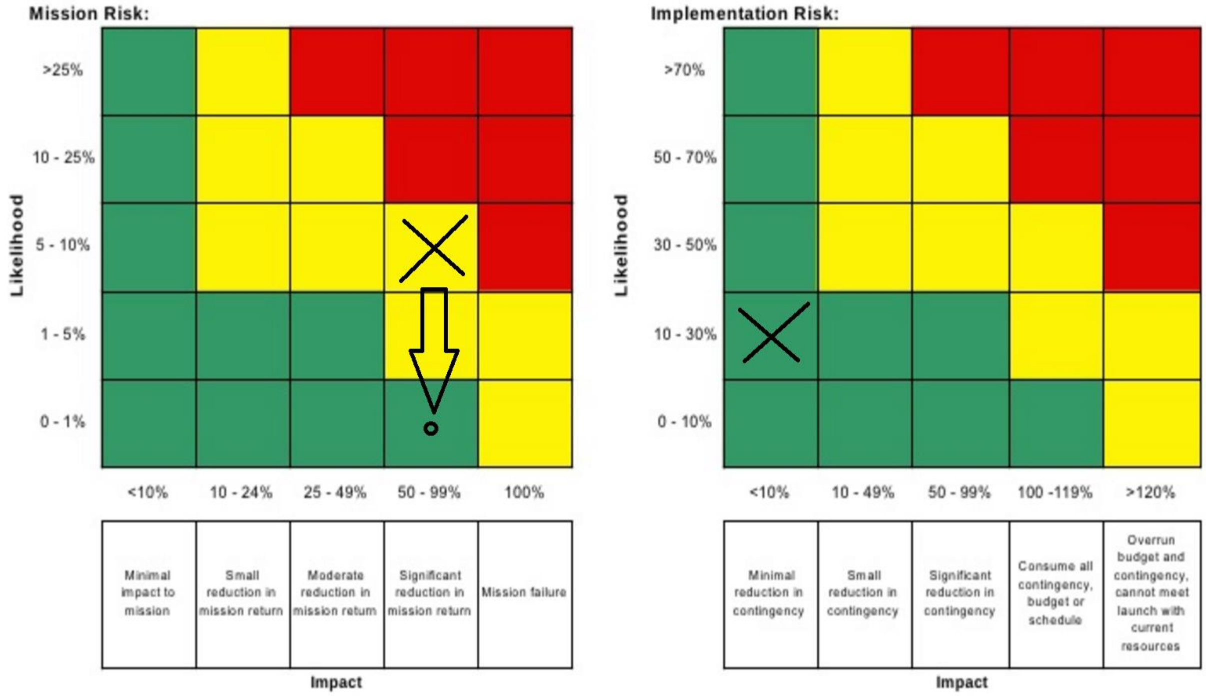


Fig. 8.2.2 4: Risk Analyzing Matrices - Zero boil-off

8.2.3 Risks at the Ground Segment Level

Lack of water

The most important risk on the Moon surface is installing a LEEP where the amount of water we can extract is lower than expected. Satellite images have proven that there might be water inside those Permanently Shadowed Regions (PSR). We cannot take the risk of lacking water after a weeks of operation. The impact on the mission is high even if the likelihood stays low.

One simple way to address that issue is to send a prospecting rover which will validate the region we want to explore. In our mission, we decided to increase the probability of finding a resource mine.
Another solution is to deploy to another crater. If we install another power station in a close crater where water has been detected, rovers can drive to that location.

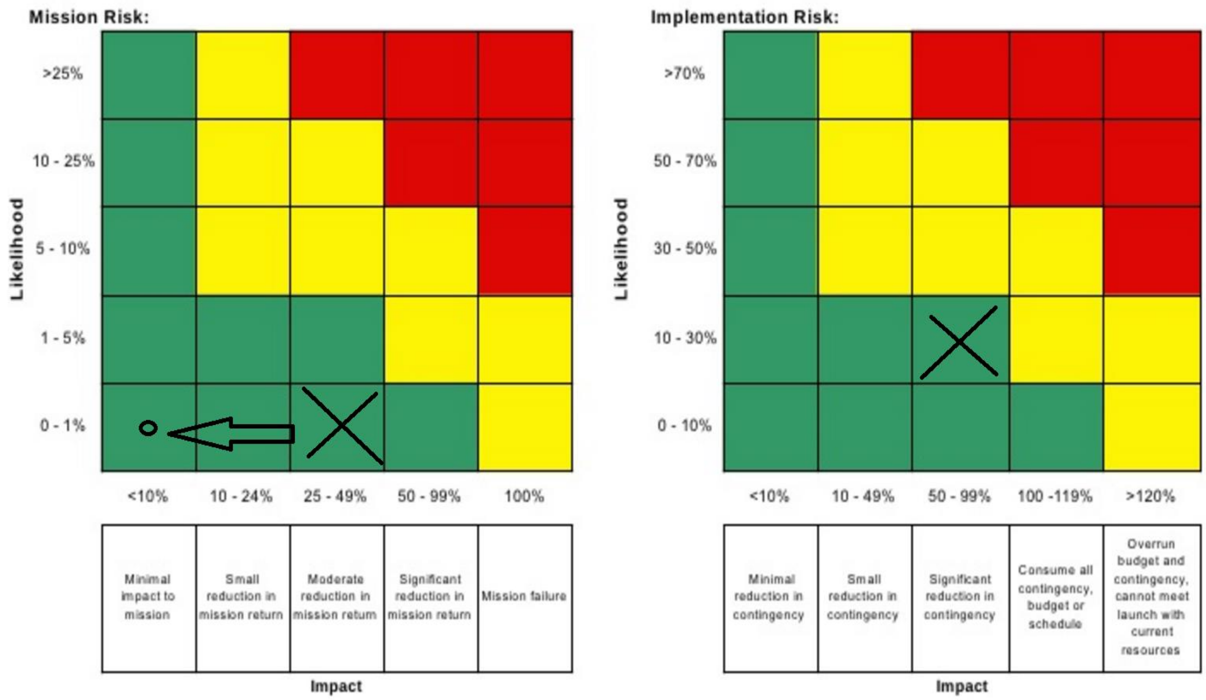


Fig. 8.2.3.1: Risk Analyzing Matrices - Lack of water

LRS crash into main base

The catastrophe we can expect with this project is a crash of the LRS on the lunar base. The impact is always 100% but it depends on the damages it will create.

The possible option to avoid that situation is to separate the landing pad from the ISRU. The multiple centaur we have with our solution will allow us to have safe LRS on the ground. The rovers will be reprogrammed to fix the ISRU or wait for additional maintenance or even an ISRU replacement. However, mitigations will be severely expensive and require much effort. This scenario stands as the worst-case scenario for our LEEP concept.

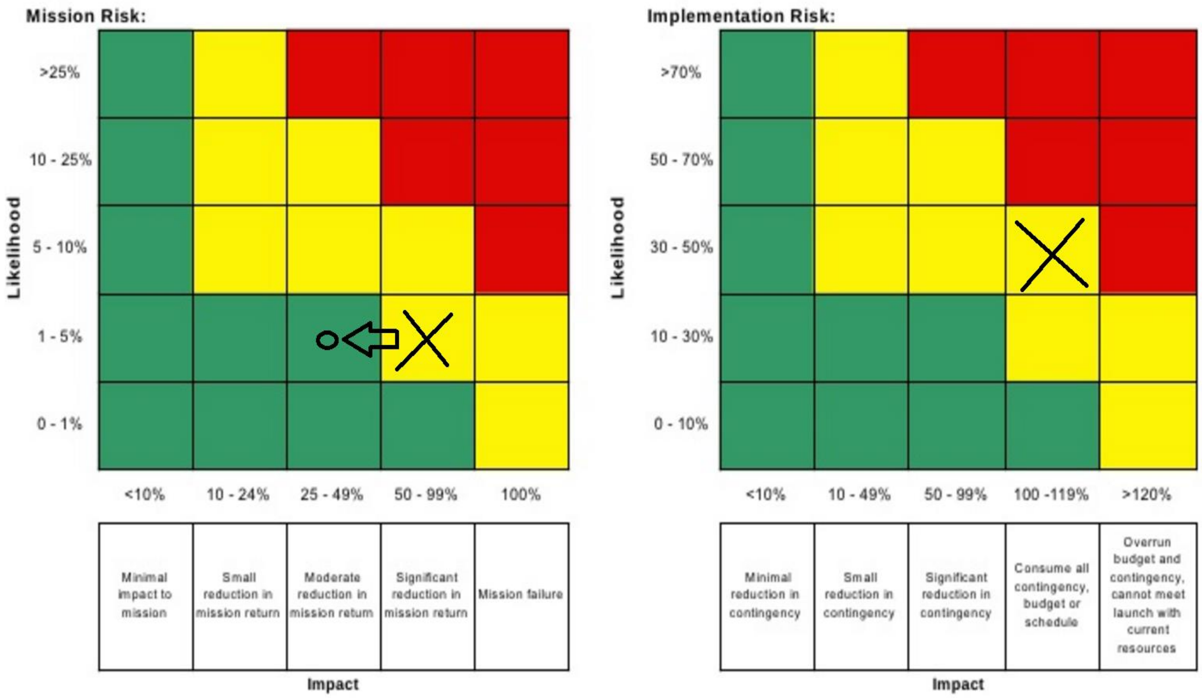


Fig. 8.2.3.2: Risk Analyzing Matrices - LRS crash into main base

Mirror technology

The power generation is our study is brought by the use of solar panels on the edge of the rim of the crater. One difficulty with those folding mirrors is the low level of maturity of the technology (TRL = 3). Thus, the associated likelihood is high.

One solution to solve that issue is to deploy smaller mirrors on the rim and pick a location with enough space on the rim. Smaller mirrors can be easily adapted from space-proven TRL 9 folding solar panel technologies (see section 3.4).

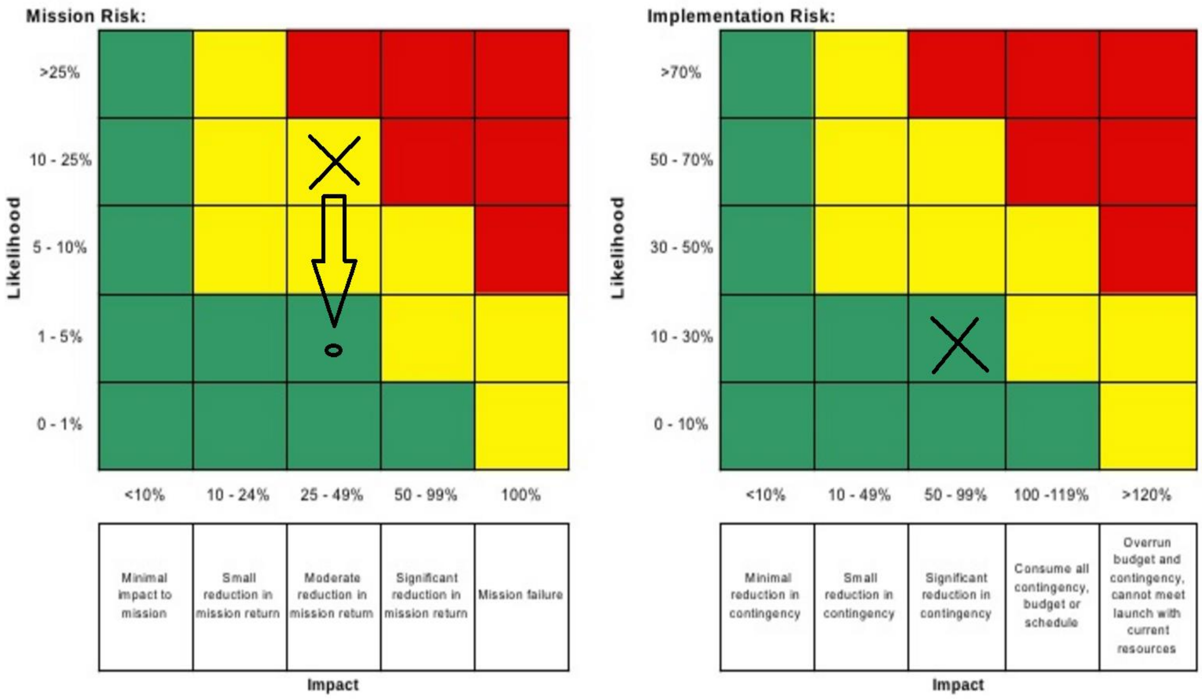


Fig. 8.2.3.3: Risk Analyzing Matrices - Mirror technology

Excavation, prospection or construction rover loss

The last of the biggest risks for the ground segment is a rover failure (or multiple failures). Especially, construction rovers can be critical for the whole discussion as they are in charge of building the ISRU. A simple solution is to have a rover redundancy but it does cost a lot more. Another solution is to have in-built capacity with rover able to fix other rovers. One final solution is to resupply the rovers with extra material to do the reparation and maintenance.

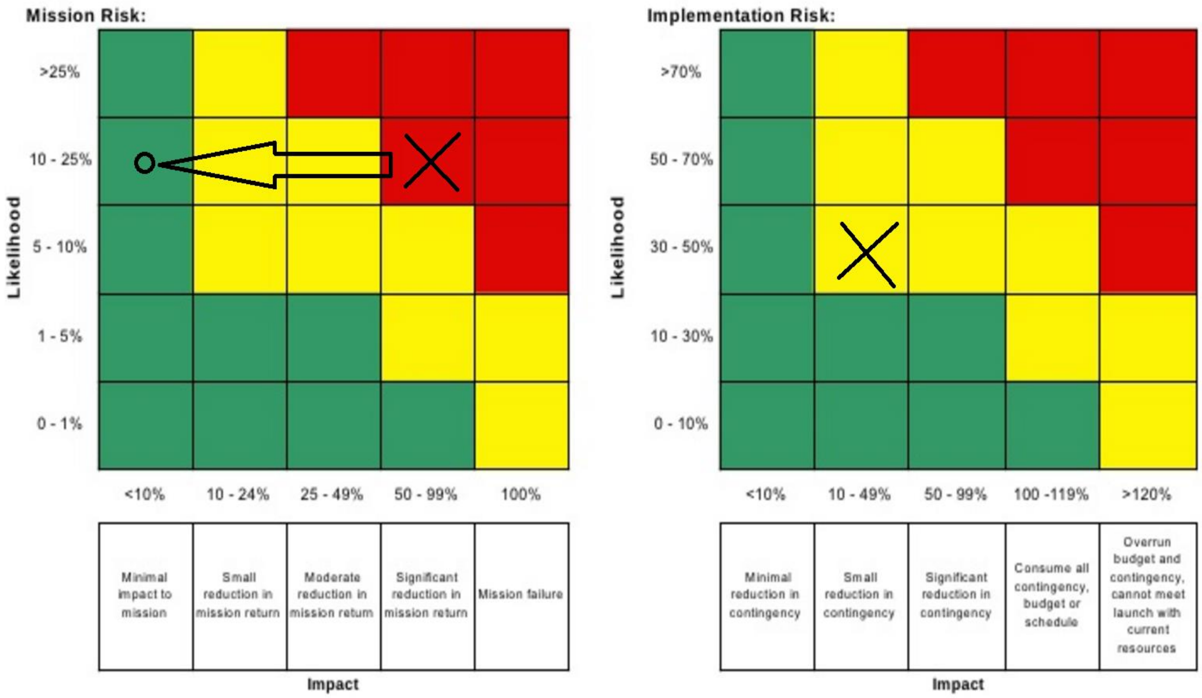


Fig. 8.2.3.4: Risk Analyzing Matrices - Excavation, prospection or construction rover loss

8.3 Political and Regulatory Considerations

The development and execution of the LEEP mission takes place in a complex political and regulatory environment. Success depends on satisfactorily coordinating the many stakeholders described in section 2.4. We assume that NASA will support the preponderance of LEEP's budget, as further described in section 8.1.5, and therefore primarily assess political and regulatory considerations through a United States-centric lens.

8.3.1 Domestic

8.3.1.1 State and Local

Primary state and local considerations are economic and environmental. Localities frequently compete to attract business. Though the U.S. Congress ended its practice of "earmarking" projects to support specific districts in 2010,[41] cities and states frequently use tax policy to entice firms to locate in their district. However, since LEEP draws heavily on existing technology, we expect that the likely implementing contractors will use existing facilities for the production of mission components. At the margins, subcontractors may seek to use LEEP-related contracting to seek favorable tax treatment, but we do not expect this to be a significant result.

In contrast, localities often worry about the environmental impacts of heavy industrial production involving hazardous materials. The LEEP architecture uses well-proven chemical rocket propulsion systems that are already produced commercially. The environmental, health, and safety risks regarding these chemicals are well understood and managed by industry, and we expect that any anomaly that did occur during production would be more closely linked to the manufacturer than the customer. Finally, LEEP also makes use of solar power rather than nuclear, further reducing political risk.

8.3.1.2 National

The LEEP mission is highly aligned with U.S. space policy and strategy, as described in detail in the mission introduction session. However, national politics remain relevant for LEEP. The support of the LEEP architecture among key Congressional constituencies would be critical to the program's viability over the nearly two decades that it would be under development and in operation. Such long-term support is never assured. Further, even with significant specific support, any large budgetary decreases to NASA (whether targeted to NASA or as part of broader reductions in government spending) would almost certainly

[41] "An end to earmarks," The Economist, November 18, 2010. Available online at http://www.economist.com/node/17525721.

adversely impact LEEP. Lastly, a combination of cost overruns and Congressional sensitivity to return on spending make any variance in the system cost particularly risky.

8.3.2 International

8.3.2.1 Export Control Compliance

The LEEP team envisions a range of international partnerships in the funding, development, and operation of the architecture. At minimum, we expect some level of participation from the 12 member agencies of the International Space Exploration Coordination Group. Any technical collaboration raises potential export sensitivities, though the particular technologies under consideration for LEEP are unlikely to trigger any novel export concerns. Mission managers will need to monitor compliance with all applicable regulations, which are already implemented via sophisticated programs at every major aerospace firm and laboratory.

8.3.2.2 Compliance with International Law

No treaty specifically addresses mining on the moon. Two foundational space treaties--the Outer Space Treaty (OST) and the Moon Treaty--address many relevant aspects but do not take on the issue directly. Nevertheless, the potential for a violation of international law, or at least a diplomatic row due to the treaties' content and interpretation (particularly OST, which has entered into force) makes this topic worthy of critical analysis.

Most centrally, Article II of OST prohibits ownership of celestial bodies, including the moon.[42] There is some debate whether mining in space is considered ownership.[43] In 2015, former U.S. President Obama signed the Commercial Space Launch Competitiveness Act (CSLCA), which provided a legal framework under which U.S. miners' property rights could be recognized under U.S. law. According to the International Space Law Institute, which issued a position paper acknowledging the viability of the United States' interpretation,

[42] Article II reads: "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."

[43] Pro: https://static1.squarespace.com/static/579fc2ad725e253a86230610/t/57ec6ac65016e1636a21e331/1475111622859/FletcherForum_Sum16_40-2_139-157_LINTNER.pdf, http://ncjolt.org/wp-content/uploads/2017/01/BlountRobison_Final.pdf; Unclear: Joanne Gabrynowicz, professor emerita of space and remote sensing law at the University of Mississippi; Frans von der Dunk, a law professor at the University Of Nebraska College Of Law; Against: Fabio Tronchetti, a professor at the Harbin Institute of Technology's School of Law in China.

Article I of the OST "specifies the right of the free exploration and use of outer space and celestial bodies."[44] In this light, the action of miners would reflect a valid action under Article I to explore space. However, the U.S. Department of State has acknowledged that at least one state has objected to the United States' actions under CSLCA. Kenneth Hodgkins, the State Department Director for Space and Advanced Technology, said that at a UN event held shortly after the law's passage "a number of delegations expressed their views concerning the actions that we took in Congress last year...In particular, the Russians made several interventions stating that what we did is inconsistent with our international obligations."[45]

Further, the Moon Treaty calls for the equitable distribution of any resources discovered on the moon. This principle led to widespread rejection of the treaty, which only has five ratifications, none of which are spacefaring nations. While it is possible that states would seek to promote broad ratification of the Moon Treaty to isolate the United States diplomatically on this issue, it seems unlikely. First, other spacefaring states clearly also have reservations regarding the treaty, as illustrated by their own non-ratification. Further, while the U.S. has received substantial diplomatic opprobrium for its failure to accede several widely-supported treaties such as the UN Convention on the Law of the Seas (UNCLOS) and the Comprehensive Test Ban Treaty (CTBT) such efforts have not changed U.S. policy, and such a result seems equally unlikely in this context. For states that truly have reservations regarding the U.S. policy and, by extension, LEEP's activities, we expect that a diplomatic compromise could be found.

8.4 Planetary Protection

As with any mission, it is imperative to avoid contamination of other worlds with microbes or other contaminants from Earth. According to NASA's Planetary Protection Office, our mission is Category II[46]. While the Moon is a body that is of interest in relation to chemical evolution and the origin of life, if only because it is so close to Earth and because the chemistry of the Moon can reveal a lot about the history of not just the Moon but also the Earth, but the knowledge resources on the Moon are generally isotropically spread across the entire surface and it is extremely unlikely that any particular source of knowledge would be destroyed by our mission, we are placed firmly into Category II.

---

44 "Position Paper on Space Resource Mining." International Institute of Space Law. December 20, 2015. Available online at http://www.iislweb.org/docs/SpaceResourceMining.pdf.

45 Hodgkins, Kenneth. Comments to the Secure World Foundation and Alliance for Space Development Seminar on Asteroids, Mining, and Policy. May 5, 2016. Available online at .https://swfound.org/events/2016/asteroids-mining-and-policy-practical-consideration-of-space-resource-rights.

46 https://planetaryprotection.nasa.gov/categories

According to the Committee on Space (COSPAR), for Category II missions

> "The requirements are for simple documentation only. Preparation of a short planetary protection plan is required for these flight projects primarily to outline intended or potential impact targets, brief Pre- and Post-launch analyses detailing impact strategies, and a Post-encounter and End-of-Mission Report which will provide the location of impact if such an event occurs."[47]

We should also consider the ENMOD convention, which applies to both Earth and space and was adopted by the United Nations to prohibit "military or any other hostile use of environmental modification techniques having widespread, long-lasting or severe effects as the means of destruction, damage or injury to any other State Party."[48] By the same reasoning as before, because our impact on the Moon is negligible compared to the Moon itself and because we are not causing any significant effects to other parties wishing to use the Moon, and because our modifications to the environment are for peaceful purposes and will not cause significant effects to the Moon, our mission should fall within the constraints placed on it by ENMOD.

8.5 Public Relations & Outreach

As an integral element of its execution, the LEEP mission is dedicated to inspiring and enabling other ventures and ideas. Public relations and outreach is therefore a critical element of the project's execution strategy. To most effectively engage the public, the LEEP mission has developed a four-part public outreach plan:

Audience: The LEEP mission would represent the most tangible step to date towards NASA's goal of reaching Mars. The mere existence robotic lunar base has the potential to enter the public's collective imagination in the same way as the space shuttle or International Space Station. We propose targeting our outreach to a broad audience, segmented primarily into the general adult population (non-scientist) and across the full age spectrum of STEAM education (early through post-graduate.)

Impacts: Among the general public, we will deliver two outcomes: 1) drive momentum towards SLS's upcoming launches, and 2) normalize the idea of extended human establishments off earth. For students, we will drive interest in science and exploration by using the experiment as a case study and laboratory. We will use internships as an opportunity to simultaneously vet future talent while providing foundational professional experience in spaceflight that would serve students well anywhere in the space community.

[47] https://cosparhq.cnes.fr/sites/default/files/ppp_article_linked_to_ppp_webpage.pdf

[48] https://ihl-databases.icrc.org/ihl/INTRO/460?OpenDocument

Schedule: We envision a progressively building public relations campaign. After mission approval and during the design phase, we will roll out an "early adopter" campaign to raise awareness among the science-interested general population. We will also target upper-level students to seed the mission's required near-term talent pipeline. As the concept moves towards tangible flight hardware, we expect to increase the pace of publicity to drive enthusiasm for the program. We will apply maximum resources during key mission operational milestones such as launch, landing, and "firsts." Our goal is to generate "I remember when…" moments for the public. As LEEP progresses through its operational life, we will sustain interest by reminding the public that LEEP is still in operation and producing useful results. Towards the later stages of LEEP's mission, we expect to coordinate our public outreach with the SLS program.

Media Mix: We plan to leverage traditional media techniques (press releases, TV interviews, educational outreach) with new media channels and tangible experience for members of the public to interface directly with the mission. In all cases, we plan to highlight the mission's unique architecture and staying power. To this end, we will seeks to leverage endorsements from SLS astronauts to humanize the relevance of LEEP's architecture and role in the roadmap towards Mars.

Concepts to implement this public communications strategy include:

- Virtual reality & games: drive one of the mining rovers on the moon! Look out over a lunar crater. Stargaze from a permanently shadowed region.
- Student project partnership with a university team: continuing from the example of missions like the REXIS sub-mission on OSIRIS-REx mission and IRIS on the proposed Moonrise mission, there is tremendous potential to inspire and train the next generation of scientists, engineers, and designers.
- Time capsules: this mission will create a lot of holes! Following excavation of a region, the holes created could be a destination for time capsules. This could spark a new business where a future mission would land a rover that would put a time capsule in one of the waterless holes. Customers could even drive a rover to place their time capsule in the hole[49].
- Mars Lunar webcam: streaming webcams have become a popular element of social media feeds. While the communications budget makes a high-resolution live stream infeasible, a short-duration livestream or a semi-frequently updated "earthrise" image could fundamentally change the public's relationship with their own planet by allowing anyone to view earth at a distance at any nearly time.

[49] Idea inspired from discussions with Dr. Kris Zacny

8.6 Partnerships with the Private Sector

At present, fuel prospecting on the moon is not sufficiently mature to represent a viable stand-alone business. Though a $12B capital raise to build a system identical to LEEP is well within the global financial market's wherewithal, it is challenging to identify sufficient paying demand for fuel to provide a return on the investment through the end of the 2030s. NASA has three SLS missions planned through 2039 and SpaceX has indicated it will land its first Red Dragon on Mars in 2020.[50] Even if NASA were to pay $1B to a Lunarport operator per fueling and SpaceX paid half that rate for the liquid oxygen necessary for its methalox propulsion system, the system could not complete sufficient missions to repay its initial investment for the given fuel extraction capacity. An evenly-split cost share with a space agency for the construction costs could make the system more viable, it would require taking significant entrepreneurial risk regarding demand for the fuel product. However, at those scales a space agency's participation in a Lunarport could be construed as a subsidy rather than a partnership, further inflaming the potential treaty tensions discussed in section 8.3.2.2.

LEEP represents a first step towards paving a new economy, furthering the 2010 National Space Policy goal of "encouraging and facilitating the growth of a U.S. commercial space sector."[51] Though the economics of space flight are in many ways different that most earth-bound businesses, logic of business remains the same. Companies worldwide have delivered greater value by focusing on core operations. Two significant recent examples of this trend are the outsourcing of back-office functions to service providers and the outsourcing of server management to cloud computing firms. These shifts have unlocked significant cost and operational efficiencies.

We expect a similar trend to unfold in the fuel market as businesses move into space, what we term Fuel as a Service (FAAS). Customers would be responsible only for getting themselves into orbit; beyond that fuel would be a temporary rest stop. A market-leading business would be one that believed in the vision of a world in which mission planners are constrained only by their imagination and not fuel. Such a firm could serve numerous customers, beginning in fuel operations and extending to other businesses:

- Fuel transfer for Martian or Jovian system human missions to increase payload mass.
- Fuel transfer for deep space robotic missions that can exit at significantly higher payload masses

[50] Javelosa, June. "Elon Musk has a new timeline for landing on Mars," February 19, 2017. Available online at https://futurism.com/elon-musk-has-a-new-timeline-for-humans-living-on-mars/.

[51] Executive Office of the President. "National Space Policy of the United States of America," June 28, 2010. Available online at https://obamawhitehouse.archives.gov/sites/default/files/national_space_policy_6-28-10.pdf.

or velocities.

- Waypoint destination for interplanetary tourists. Earth is a small dot from Mars, so conducting an EVA on an earth viewing platform during refueling would be a unique opportunity to see the full disk of the earth.
- Satellite removal tug. Operators today cease operations prior to propellant exhaustion (with safety margin) to ensure sufficient fuel to move to a parking orbit. Operators could extend service life by operating the satellite to exhaustion knowing that the tug would be able to drag the dead satellite to the parking orbit. As long as the revenue generated by extended operations exceeds the cost of the push there is a business logic for operators. The LEEP architecture intersects GEO so a similar system servicing satellites at low cost without interrupting other missions may be feasible.
- Transfer tug for lunar operations. Logistics and scientific payloads for experiments such as radio astronomy could ride aboard the LRS as it returns to the moon from a refueling. In the LEEP architecture, the LRS are already outfitted with payload platforms to meet their initial construction delivery requirements and payloads could use the same landing stage design. Some upgrading to the release mechanism may be required to support docking with a fresh payload on orbit.

Figure 8.6.1 is a Porter Forces diagram that suggests that the space refueling business is likely to be quite profitable unless firms engage in price competition. There may be opportunities for firms to reduce rivalry and increase profitability by differentiating themselves through servicing timelines or by securing long-term contracts with key customers.

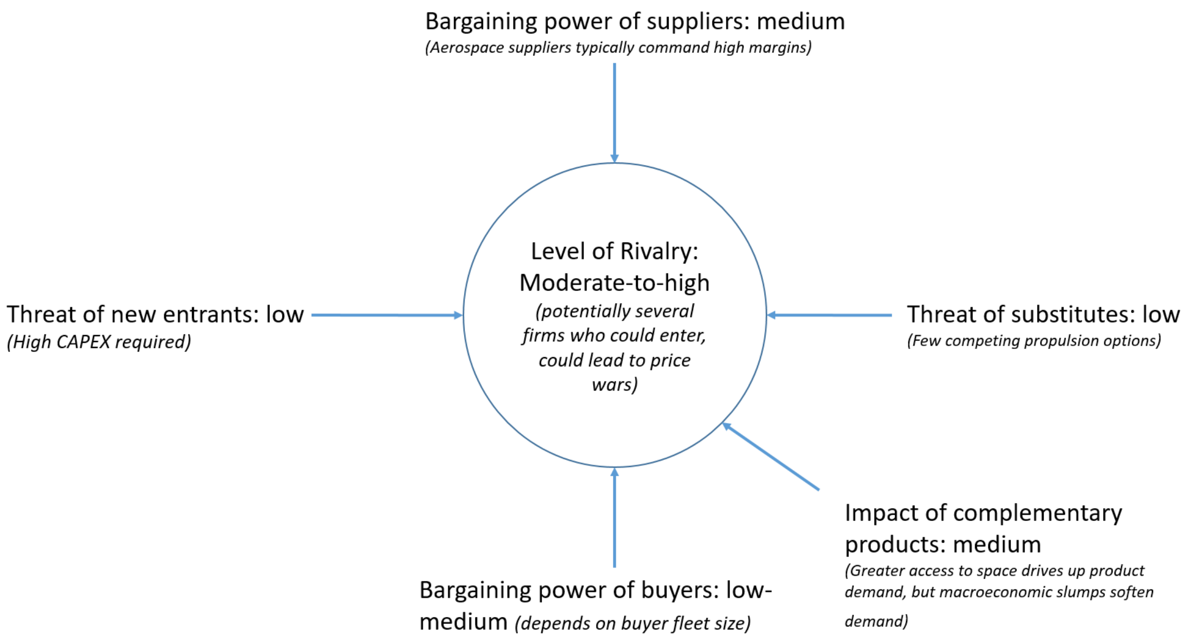


Fig. 8.6.1: Porter Forces Diagram

By partnering with the private sector in small but consequential ways through LEEP, NASA reduce its own technical and potentially financial risk by tapping into the private sector while beginning to transfer results and stimulate growth in this area. Potential areas for collaboration include:

- Subsystem design and operation (rover, extraction technology, power systems)
- Autonomous operations
- long-duration mission planning
- extreme condition engineering (Permanently shadowed regions)

References

Sensitivity Analysis of the Advanced Missions Cost Model, Andrew Owens and Olivier de Weck, Massachusetts Institute of Technology

Estimating the Life Cycle Cost of Space Systems, Harry W. Jones, NASA Ames Research Center, Moffett Field, CA, 94035-0001

Space Mission Engineering: The New SMAD, James Richard Wertz, David F. Everett, Jeffery John Puschell, Microcosm Press, Jan 1, 2011

NASA Risk Management Handbook, National Aeronautics and Space Administration, NASA/SP-2011-3422, Version 1.0, November 2011

Lunarport: A Proposed Lunar-Resource Station to Expand Deep-Space Travel Horizons, Ilana Gat, Thibaud Talon, IEEE Aerospace Conference Yellowstone Conference Center, Big Sky, Montana, Mar 4 - 11, 2017

Peter Eckart: The Lunar Base Handbook – An Introduction to Lunar Base Design, Development, and Operations, McGraw-Hill Primis Custom Publishing, 2006

Calle, C. I., C. D. Immer, J. Ferreira, M. D. Hogue, A. Chen, M. W. Csonka, N. Van Suetendael, and S. J. Snyder: Integration of the Electrodynamic Dust Shield on a Lunar Habitat Demonstration Unit, Proc. ESA Annual Meeting on Electrostatics 2010, Paper D1, 2010

Jordan Olive: Understanding and Improving Upon Electrodynamic Dust Shield Technology on a Lunar Rover, Hawaii NASA Space Grant Consortium, 2nd Summer 2009 Final Report, 2009

Calle, C. I., C. R. Buhler, J. L. McFall, S. J. Snyder: Particle Removal by Electrostatic and Dielectrophoretic Forces for Dust Control During Lunar Exploration Missions, Journal of Electrostatics 67 (2009) 89-92, 2009

Cadogan, Dave, and Janet Ferl: Dust Mitigation Solutions for Lunar and Mars Surface Systems, ILC Dover LP, 2007 SAE International, 2007-01-3213, 2007

Guetersloh, S., C. Zeitlin, L. Heilbronn, J. Miller, T. Komiyama, A. Fukumra, Y. Iwata, T. Murakami, and M. Bhattacharya: Polyethylene as a Radiation Shielding Standard in Simulated Cosmic-Ray Environments, Nuclear Instruments and Methods in Physics Research Section B: Beam Interactions with Materials and Atoms, Volume 252, Issue 2, November 2006

M. Bhattacharya: Polyethylene as a Radiation Shielding Standard in Simulated Cosmic-Ray Environments, Nuclear Instruments and Methods in Physics Research Section B: Beam Interactions with Materials and Atoms, Volume 252, Issue 2, November 2006

Nounu, Hatem N., Myung-Hee Y. Kim, Michael L. Gernhardt, Francis A. Cucinotta: A Study of The Shielding Effects of Lunar Topography and Rover Design on Humans in Future Lunar Exploration Missions Using NASA-Developed Radiation Analysis Software, Annual Report 2008, Human Research Program, NASA, 2008

Jones, Harry W. "Estimating the Life Cycle Cost of Space Systems." 45th International Conference on Environmental Systems, 2015.

Fabio Tronchetti: The Space Resource Exploration and Utilization Act: A move forward or a step back? Original Research Article. Space Policy, Volume 34, November 2015.

http://www.ulalaunch.com/uploads/docs/Published_Papers/Extended_Duration/CryogenicOrbitalTestbedCRYOTE2009.p

https://www.nasa.gov/feature/nasa-s-restore-l-mission-to-refuel-landsat-7-demonstrate-crosscutting-technologies

http://www.ulalaunch.com/uploads/docs/Published_Papers/Extended_Duration/SettledCryogenicPropellantTransfer.pdf

MailOnline, B. R. G. for. (2016, March 24). Could we be living on the moon by 2022? Retrieved March 31, 2017, from http://www.dailymail.co.uk/sciencetech/article-3507665/Could-living-moon-2022-Nasa-scientists-reveal-plans-build-lunar-base-six-years-10-billion.html

Wall, M., 30, S. com S. W. | M., & ET, 2017 06:30am. (2017). NASA Eyeing Mini Space Station in Lunar Orbit as Stepping-Stone to Mars. Retrieved March 30, 2017, from http://www.space.com/36270-nasa-deep-space-gateway-moon-orbit.html
https://www.nasa.gov/content/scientists-developing-ways-to-mitigate-dust-problem-for-explorers

https://www.nasa.gov/feature/goddard/real-martians-how-to-protect-astronauts-from-space-radiation-on-mars

https://www.nasa.gov/vision/space/travelinginspace/radiation_shielding.html

Mars Science Laboratory Landing, https://www.jpl.nasa.gov/news/press_kits/MSLLanding.pdf

Mars Science Laboratory Spacecraft, https://mars.nasa.gov/msl/mission/spacecraft/

NASA Cost Estimating Handbook, Appendix F,
https://www.nasa.gov/sites/default/files/files/CEH_AppF.pdf

Development and transportation costs of space launch systems,
http://www.dglr.de/fileadmin/inhalte/dglr/fb/r1/r1_2/06-Raumtransportsysteme-Kosten.pdf

Mission Operations Cost Estimation Tool (MOCET),
https://www.nasa.gov/sites/default/files/files/20_MOCET_NASA_Cost_Symposium_Briefing_2015-08-21.pdf

http://news.nationalpost.com/news/who-owns-the-moon-space-lawyers-increasingly-needed-for-legal-issues-beyond-earths-atmosphere http://www.popsci.com/it-could-soon-be-legal-to-mine-asteroids

http://www.telegraph.co.uk/news/science/space/10318077/Who-owns-the-moon-Time-to-call-in-the-space-lawyers.html

http://www.csmonitor.com/Technology/2015/0609/New-bill-in-Congress-could-help-launch-a-gold-rush-in-space

https://www.bloomberg.com/view/articles/2015-04-08/get-yourself-some-moon-rocks

http://spacenews.com/space-law-101-helping-fill-a-legal-vacuum/

http://spacenews.com/41825hearing-raises-questions-about-asteroid-mining-bill/

http://www.regblog.org/2015/12/31/rathz-space-commerce-regulation/

Appendices

Appendix A – Candidate Conceptual Designs

The Pugh-chart was used to select the concept of operations.

Pugh Matrix

Concept Selection Legend	
Better	+
Same	S
Worse	-

Key Criteria		*Importance Rating*	Alternatives: LRS = EUS	LRS = Centaur multiple no orbit change	LRS = Centaur multiple with orbit change	EUS/Centaur ISRU on Moon	EUS/Centaur ISRU on LRS
Construction timeline	How fast can it be built and deployed?	**10**		+	+2	-	S
Energy/Propellant output	How many yearly missions can Lunarport support?	**7**		S	S	-	-
Fueling capacity	How much additional mass per mission can be sent to Mars with Lunarport?	**10**		S	+	+	+
Operation & Maintenance complexity	What are the hardware maintenance and refueling operation costs?	**5**		S	-	-	S
TRL maturity	Does it help gain knowledge and competencies for future Mars exploration?	**8**		+	+	+	+
Technical risks	Does it bring high risks?	**7**		S	-	-	-
Cost	How coslty is the development and production?	**7**		+	+	-	-
Partnership	Does it foster partnerships with space agencies and private companies?	**8**		+	+	+	+
			Sum of Positives	4	4	3	3
			Sum of Negatives	0	2	5	3
			Sum of Sames	4	1	0	2
			Weighted Sum of Positives	33	33	26	26
			Weighted Sum of Negatives	0	12	36	21
			TOTALS	**33**	**21**	**-10**	**5**

Fig. 7: Pugh Matrix

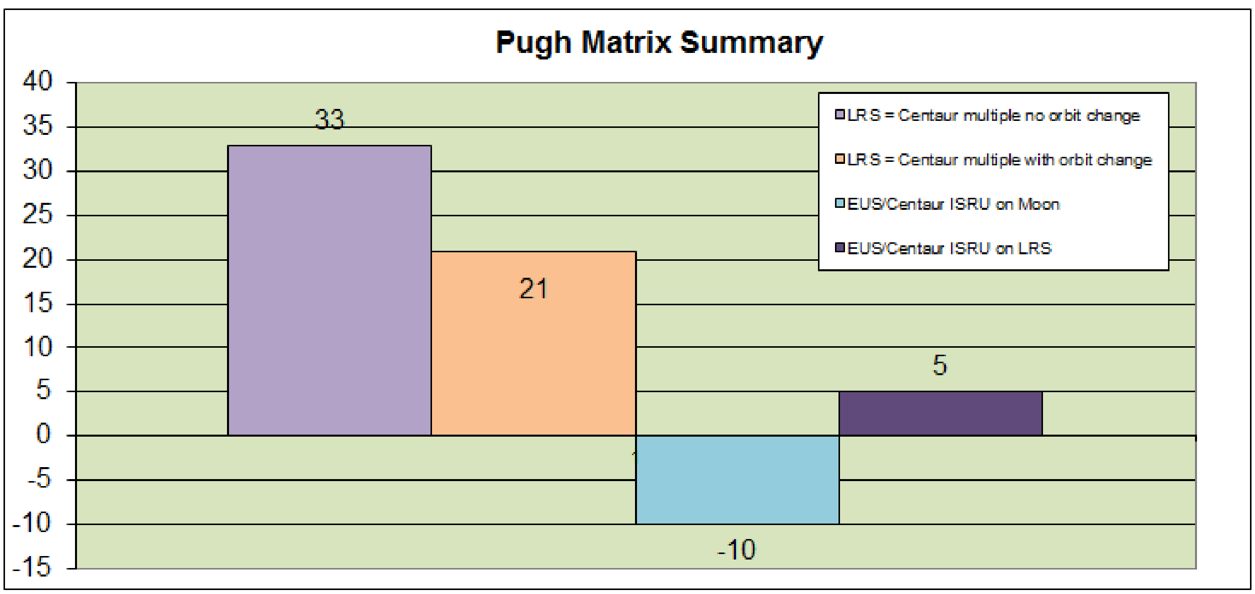


Fig. 8: Pugh Matrix Summary

Appendix B – Trajectory Optimization Matlab Code

```
clear all;
close all;

R_E = 6378.0; % [km] Radius of the earth
R_M = 1737.1; % [km] Radius of the Moon

mu_moon = 4.9e3;    % [km^3/s^2] Gravitational parameter Moon
mu_earth = 3.986e5;  % [km^3/s^2] Gravitational parameter Earth

R_EM = 384400.0; % [km] Distance between Earth and Moon

% Radius of influence of the moon
r_soi_moon = R_EM * (mu_moon/mu_earth)^(2/5);
r_soi_earth = R_EM - r_soi_moon;

% Period of an elliptic orbit with perigee at LEO and apoapse at the moon
a_TEI = (R_EM + R_E + 400)/2;
T_TEI = 2*pi*sqrt(a_TEI^3/mu_earth)/3600; % [hours]

% Period of an elliptic orbit with perigee at LEO and apogee at the
% refueling orbit
a_refueling = (400 + R_E + R_E + 150000)/2;
T_refueling = 2*pi*sqrt(a_refueling^3/mu_earth)/3600; % [hours]

time_of_flight = (T_TEI/2)

altitude=1000:100:(r_soi_earth-R_E);

DV_circular = zeros(length(altitude),1);
DV_elliptic = zeros(length(altitude),1);
DV_mars_circular = zeros(length(altitude),1);
DV_mars_elliptic = zeros(length(altitude),1);

delta_v_apotwist = zeros(length(altitude),1);

% Damon Landau uses
dv_apotwist = 0.15;
apoapse_earth_orb_moon = R_EM;
semimajor_axis_landau = (R_EM + 400 + R_E)/2;
v_apoapse_apotwist = sqrt(2*mu_earth/apoapse_earth_orb_moon -
mu_earth/semimajor_axis_landau);
delta_argPeriapsis = 2*asin(dv_apotwist/(2*v_apoapse_apotwist)) * 180/pi

for i =1:length(altitude)
  DV_circular(i) = patchConincs(altitude(i)); % [km/s] Total round-trip time DV for the LRS to a
circular refueling orbit
```

```
   DV_elliptic(i) = patchConincs_elliptical(altitude(i)); % [km/s] Total round-trip time DV for the LRS to a elliptic refueling orbit

   DV_mars_circular(i) = patchConincs_mars(altitude(i));
   DV_mars_elliptic(i) = patchConincs_mars_elliptic(altitude(i)); % [km/s] Total DV to go to Mars from an elliptical refueling orbit
end

% DV_circular = patchConincs(altitude); % [km/s] Total round-trip time DV for the LRS to a circular refueling orbit
% DV_elliptic = patchConincs_elliptical(altitude); % [km/s] Total round-trip time DV for the LRS to a elliptic refueling orbit
%
% DV_mars_circular = patchConincs_mars(altitude);
% DV_mars_elliptic = patchConincs_mars_elliptic(altitude); % [km/s] Total DV to go to Mars from an elliptical refueling orbit

figure()
semilogx(altitude, DV_circular)
hold all;
semilogx(altitude, DV_elliptic)
legend('DV circular orbit', 'DV elliptic orbit')
xlabel('Apogee altitude relative to Earth [km]')
ylabel('Round-Trip Delta V for the LRS [km/s]')

figure()
semilogx(altitude, DV_mars_circular)
hold all;
semilogx(altitude, DV_mars_elliptic)
%semilogx(altitude, delta_v_apotwist)
legend('DV to Mars from circular', 'DV to Mars from elliptic')
xlabel('Apogee altitude relative to Earth [km]')
ylabel('Delta V to Mars [km/s]')

function total_DV = patchConincs(altitude_E)

% Patch conics model to compute the total round trip DV to go from the moon
% and back from a circular refueling orbit

disp('circular')

R_E = 6378.0; % [km] Radius of the earth
R_M = 1737.1; % [km] Radius of the Moon

mu_moon = 4.9e3;     % [km^3/s^2] Gravitational parameter of the moon
mu_earth = 3.986e5;  % [km^3/s^2] Gravitational parameter of the moon

R_EM = 384400.0; % [km] Distance between Earth and Moon

a_LLO = 100.0 + R_M; % [km] radius of the LLO orbit
```

```
v_LLO = sqrt(mu_moon/a_LLO); % [km/s] Velocity at LLO

radius_earth_orb = R_E + altitude_E;
v_earth_orb = sqrt(mu_earth./radius_earth_orb);

% Velocity of the moon relative to earth
v_moon_earth = sqrt(mu_earth/R_EM);

% First Hohmann transfer
a_h_1 = (R_EM + radius_earth_orb)/2;
v_h_Max = sqrt(2*mu_earth/R_EM - mu_earth./a_h_1);
v_h_min = sqrt(2*mu_earth./radius_earth_orb - mu_earth./a_h_1);

% Delta v1: from LLO to TEI
v_inf_moon = v_h_Max - v_moon_earth; % Vinf relative to the moon
v_hyperbolic = sqrt(v_inf_moon.^2 + 2 * mu_moon/a_LLO);
energy_hyperbolic = 0.5*v_hyperbolic.^2 - mu_moon/a_LLO; % Check (>0)
delta_v1 = abs(v_LLO - v_hyperbolic);

% Delta v2: from TEI to LEO (or another circular orbit around earth)
delta_v2 = abs(v_h_min - v_earth_orb);

% Delta v3: From LEO to TLI
delta_v3 = delta_v2; % Same thing!

% Delta v4: From TLI to LLO
delta_v4 = delta_v1;

% Delta v5: From departing
delta_v5 = 1.6;
delta_v6 = 1.6;

total_DV = delta_v1 + delta_v2 + delta_v3 + delta_v4 + delta_v5 + delta_v6;

function total_DV = patchConincs_elliptical(apoapse_altitude)

% Patch conics model to compute the total round trip DV to go from the moon
% and back from an elliptic refueling orbit

disp('elliptic')

R_E = 6378.0; % [km] Radius of the earth
R_M = 1737.1; % [km] Radius of the Moon

mu_moon = 4.9e3;     % [km^3/s^2] Gravitational parameter of the moon
mu_earth = 3.986e5;  % [km^3/s^2] Gravitational parameter of the earth

R_EM = 384400.0; % [km] Distance between Earth and Moon

% Velocity of the moon relative to earth
```

```
v_moon_earth = sqrt(mu_earth/R_EM);

a_LLO = 100.0 + R_M; % [km] radius of the LLO orbit
v_LLO = sqrt(mu_moon/a_LLO); % [km/s] Velocity at LLO

periapse_earth_orb = R_E + 400; % [km] Perigee altitude (assumption!!)
apoapse_earth_orb = R_E + apoapse_altitude; %[km] apogee altitude -> Trying to optimize for this
semimajor_axis = (periapse_earth_orb + apoapse_earth_orb)/2;
v_earth_orb_periapse = sqrt(2*mu_earth/periapse_earth_orb - mu_earth./semimajor_axis);

% First Hohmann transfer
a_h_1 = (R_EM + periapse_earth_orb)/2;
v_h_Max = sqrt(2*mu_earth/R_EM - mu_earth/a_h_1);
v_h_min = sqrt(2*mu_earth/periapse_earth_orb - mu_earth/a_h_1);

% Delta v1: from LLO to TEI
v_inf_moon = v_h_Max - v_moon_earth; % Vinf relative to the moon
v_hyperbolic = sqrt(v_inf_moon^2 + 2 * mu_moon/a_LLO);
energy_hyperbolic = 0.5*v_hyperbolic^2 - mu_moon/a_LLO % Check (>0)
delta_v1 = abs(v_LLO - v_hyperbolic);

% Delta v2: from TEI to an elliptic orbit
delta_v2 = abs(v_h_min - v_earth_orb_periapse);

% Delta v3: From LEO to TLI
delta_v3 = delta_v2; % Same thing!

% Delta v4: From TLI to LLO
delta_v4 = delta_v1;

% Delta v5: From departing
delta_v5 = 1.6;
delta_v6 = 1.6;

% Delta v7: Rendezvous maneuver
delta_v7 = 0.02; % Preliminary design of the guidance, Navigation, and control system of the Altair
Lunar Lande

total_DV = delta_v1 + delta_v2 + delta_v3 + delta_v4 + delta_v5 + delta_v6 + delta_v7;

function total_DV_mars = patchConincs_mars(altitude_E)

% Patch conics model to compute the total DV to go from the circular
% refueling orbit to Mars

mu_sun = 1.327e11;   % [km^3/s^2] Gravitational parameter of the sun
mu_earth = 3.986e5;  % [km^3/s^2] Gravitational parameter of the earth
mu_mars = 4.282e4;   % [km^3/s^2] Gravitational parameter of the moon

R_E = 6378.0; % [km] Radius of the earth
```

```
R_mars = 3390.0; % [km] Radius of Mars

R_EarthSun = 149.6e6; % [km] Distance between the Earth and the Sun
R_SunMars = 227.9e6; %[km] Distance between the Sun and Mars

v_earth = sqrt(mu_sun/R_EarthSun); % [km/s] Velocity of the Earth

% Refueling circular orbit
a_Earth = R_E + altitude_E; % [km] radius
v_orbit_earth = sqrt(mu_earth./a_Earth); % [km] Velocity

% Hohmann transfer to Mars
a_h = (R_EarthSun + R_SunMars)/2;
v_h_Max = sqrt(2*mu_sun/R_EarthSun - mu_sun/a_h);
v_h_min = sqrt(2*mu_sun/R_SunMars - mu_sun/a_h);

% Escape from the Earth
v_inf_earth = v_h_Max - v_earth; % Vinf relative to the earth
v_hyperbolic_earth = sqrt(v_inf_earth^2 + 2 * mu_earth./a_Earth);
energy_hyperbolic = 0.5*v_hyperbolic_earth.^2 - mu_earth./a_Earth; % Check (>0)
delta_v1 = abs(v_orbit_earth - v_hyperbolic_earth);

total_DV_mars = delta_v1;

function total_DV_mars = patchConincs_mars_elliptic(altitude_E)

% Patch conics model to compute the total DV to go from the elliptic
% refueling orbit to Mars

mu_sun = 1.327e11;   % [km^3/s^2] Gravitational parameter of the sun
mu_earth = 3.986e5;  % [km^3/s^2] Gravitational parameter of the earth
mu_mars = 4.282e4;   % [km^3/s^2] Gravitational parameter of the moon

R_E = 6378.0; % [km] Radius of the earth
R_mars = 3390.0; % [km] Radius of Mars

R_EarthSun = 149.6e6; % [km] Distance between the Earth and the Sun
R_SunMars = 227.9e6; %[km] Distance between the Sun and Mars
R_EM = 384400.0; % [km] Distance between Earth and Moon

v_earth = sqrt(mu_sun/R_EarthSun); % [km/s] Velocity of the earth relative to the sun

% Refueling orbit
apoapse_earth_orb = R_E + altitude_E;  % [km] Apogee altitude -> Optimizing for this
periapse_earth_orb = R_E + 400; % [km] Perigee orbit (Assumption!!)
semimajor_axis = (periapse_earth_orb + apoapse_earth_orb)/2;
v_earth_orb_periapse = sqrt(2*mu_earth/periapse_earth_orb - mu_earth./semimajor_axis);
v_earth_orb_apoapse = sqrt(2*mu_earth./apoapse_earth_orb - mu_earth./semimajor_axis);

% Intermediate elliptic orbit -> at HEO
```

```
apoapse_intermediate_orb = R_EM;
periapse_intermediate_orb = R_E + 400; % [km] Perigee orbit (Assumption!!)
semimajor_axis_intermediate_orb = (periapse_intermediate_orb + apoapse_intermediate_orb)/2;
v_intermediate_orb_periapse = sqrt(2*mu_earth/periapse_intermediate_orb -
mu_earth/semimajor_axis_intermediate_orb);
v_intermediate_orb_apoapse = sqrt(2*mu_earth/apoapse_intermediate_orb -
mu_earth/semimajor_axis_intermediate_orb);

delta_v1 = abs(v_intermediate_orb_periapse - v_earth_orb_periapse);

% Hohmann transfer from Earth to Mars
a_h = (R_EarthSun + R_SunMars)/2;
v_h_Max = sqrt(2*mu_sun/R_EarthSun - mu_sun/a_h);
v_h_min = sqrt(2*mu_sun/R_SunMars - mu_sun/a_h);

periapse_transfer_orb_to_mars = periapse_intermediate_orb;
v_transfer_orb_periapse = v_intermediate_orb_periapse;
v_transfer_orb_apoapse = v_intermediate_orb_apoapse;

% Escape from the Earth
v_inf_earth = v_h_Max - v_earth; % Vinf relative to the earth
v_hyperbolic_earth = sqrt(v_inf_earth^2 + 2 * mu_earth/periapse_transfer_orb_to_mars);
%energy_hyperbolic = 0.5*v_hyperbolic_earth^2 - mu_earth/apoapse_Earth; % Check (>0)
delta_v2 = abs(v_transfer_orb_periapse - v_hyperbolic_earth);

% Apo-twist to change the line of apsis
delta_v_apotwist = 2* v_transfer_orb_apoapse * sin(pi/8);

total_DV_mars = delta_v1 + delta_v2 + delta_v_apotwist + 0.1; % -> 0.1: contingency
```

Appendix C – Cost Analysis

Tab. 2: Non-Recurring Cost Analysis

NON-RECURRING COST											
Mission Hardware											
	Ground Seg										
		Construction Rovers	7	$454,839,175	$194,931,075	2	0	500	20.00%	$649,770,250	
		Lunar Prospectors	8	$277,936,733	$119,115,743	2	0	500	20.00%	$397,052,475	
		Extraction Rovers	6	$163,391,780	$70,025,048	10	2	500	10.00%	$2,801,001,935	
		Antennae/Beacons	9	$1,000,000	$400,000	2	0	100	5.00%	$2,400,000	
		Mirrors	7	$50,000,000	$20,000,000	4	0	200	20.00%	$220,000,000	
		PV Array	9	$50,000,000	$20,000,000	2	0	200	5.00%	$120,000,000	
		ISRU	6	$714,285,714	$285,714,286	1	0	5000	20.00%	$1,000,000,000	
		Cargo Lander		$142,857,143	$57,142,857	4	0	1000	20.00%	$200,000,000	
	Space Seg										
		LRS	6	$59,166,666.67	$29,583,333.33	4	0	3500	20.00%	$355,000,000	
		LRS - Centaur (atlas upper stage)	9		$40,000,000						
		LRS - landing legs	6		$5,000,000						
		LRS - GNC/control	6		$20,000,000						
		LRS - Avionics	6		$6,000,000						
	Earth Seg										
		Mission Control Center			$400,000,000				5.00%	$420,000,000	
Launch costs											
		Launch Vehicles	7		$130,600,000	4	0		15.00%	$522,400,000	
		Launch Vehicles-Falcon Heavy	8		$90,000,000						
		Fairing modifications	7		$40,000,000						
		Propellant			$600,000						
Integration and Test		Integration and Test								$459,617,973	
Program Management											
	Prog Level										
		Mgmt, SE, MA								$714,724,263	
		Reserves (30%)								$2,358,590,069	
Total Non-Recuring Cost										**$10,220,556,965**	

Tab. 3: Recurring Cost Analysis

RECURRING COST										
Operations costs	Earth Seg									
		Ground Comm			$600,000				10.00%	$660,000
		Operations			32080626				30.00%	$41,704,813
	Space	Resupply Launch			$18,000,000				30.00%	$23,400,000
		Resupply Rover			$10,000,000				30.00%	$13,000,000
		Resupply LRS			$14,200,000				30.00%	$18,460,000
Total Recurring Cost										**$78,764,813**
Number of Years										10
										$11,008,205,099

Tab. 4: Cost Analogies and Parameter Value, Units and Sources

Cost Analogies and Paramete	Value	Units	Source
Falcon Heavy Cost	90000000	$/launch	http://spacenews.com/wp-content/uploads/2016/05/spacex-price.gif
Falcon Heavy Mass to GTO	8000	kg	http://spacenews.com/wp-content/uploads/2016/05/spacex-price.gif
New Glenn Mass to GTO	13000	kg	http://spacenews.com/eutelsat-first-customer-for-blue-origins-new-glenn/
New Glenn Cost	146250000	$/launch	Scaled from Falcon Heavy
Atlas V Cost	149000000	$/launch	GTO, Large Payload, https://www.rocketbuilder.com/start/configure
Launch Cost	146250000	$/launch	
Curiosity Rover Annual Cost	$14,000,000.00	2014 Dollars	http://www.space.com/28434-mars-rover-opportunity-nasa-budget.html
# of Curiosities	10	No. Curiosities Equiv.	
Ops Cost	$140,000,000.00		
Falcon 9 propellant	200000		
Mirrors	520000	$	https://www.theguardian.com/world/2013/oct/30/giant-mirrors-first-winter-sun-norway-rjukan
Batteries	45000	$	https://en.wikipedia.org/wiki/Tesla_Model_S#Battery
Mission Control Center	250000000	$	Johnson 1995
Mission Control Center	400000000	$	Johnson w 1.6 inflation
X-band cost	50000	50k per month	http://www.spaceflight.com/wp-content/uploads/2015/10/SF-Networks-Data-Sheet-Aug2015_reduced.pdf
MOCET Operations Model			https://www.nasa.gov/sites/default/files/files/20_MOCET_NASA_Cost_Symposium_Briefing_2015-08-21.pdf
Resources Prospector	$350,000,000	$	http://www.hou.usra.edu/meetings/leag2016/presentations/Wednesday/Colaprete.pdf
	300	kg	
	$1,166,667	$/kg	
MSL	$2,500,000,000.00	$	https://www.jpl.nasa.gov/news/press_kits/MSLLanding.pdf
	3300	kg	https://mars.nasa.gov/msl/mission/spacecraft/
	$757,576	$/kg	

Appendix D – Rover Statistics

Tab. 5: Rover Statistics

Year	Number of rovers	Amount extracted in year (in mT)	Amount extracted till now (in mT)
0	0	0	0
1	6	26.28	26.28
2	12	52.56	78.84
3	12	52.56	131.4
4	12	52.56	183.96
5	12	52.56	236.52
6	12	52.56	289.08
7	12	52.56	341.64
8	12	52.56	394.2
9	12	52.56	446.76
10	12	52.56	499.32
11	18	78.84	578.16
12	18	78.84	657
13	18	78.84	735.84
14	18	78.84	814.68
15	18	78.84	893.52