Team Explorer

CALTECH SPACE CHALLENGE - 2015

Team Explorer
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L-DORADO

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Executive Summary

Project Explorer establishes a concept mission for the exploration and utilization of the asteroid that will be returned by the Asteroid Retrieval Mission (ARM). The mission launches in August 2025 with a science lab flying on a Falcon Heavy. It flies to the asteroid on a ballistic trajectory and takes 8.5 days. In March 2024, a crew of 3 launches in an Orion capsule on a SLS. A service module is contributed by the European Space Agency. The crew returns 39 days later. The mission is divided into three primary phases of crew delivery, asteroid operations, and return. They take 8.5, 22, and 8.5 days respectively. The primary objectives of the operations phase are to characterize the asteroid and its environment, extract and process the asteroid's resources, and to demonstrate the ability of these processed resources to support future space missions. Public outreach is considered a fundamental part of the mission. The mission is designed to operate within a reasonable budget and schedule.

1.0 Introduction

The 2015 CalTech Space Challenge focuses on demonstrating the benefits of bringing an asteroid to a lunar orbit. The asteroid is brought to the moon in 2024 via the Asteroid Retrieval Mission (ARM) at a cost measured in billions of dollars. Many groups, including the government, scientists, and the public, will be keenly interested in feeling that the investment has been wise. These considerations result in the following problem statement:

Problem Statement: In five days, each team is challenged to design a mission to land humans on an asteroid brought back to lunar orbit, extract the asteroid's resources, and demonstrate their use. The launch date of the mission may be no later than January 1st, 2028.

To this end, Team Explorer has developed a concept for maximizing the potential scientific understanding and In Situ Resource Utilization (ISRU).

Beyond the technical and financial considerations lies the passion. Humanity has always had the desire to explore the unknown. Thanks to the technological advancements in the modern era, humans are finally capable of escaping the constraints of Earth's atmosphere and can travel to other celestial bodies. Team Explorer is fueled by the excitement of exploration and has adopted a guiding philosophy in this project of emphasizing creativity and pushing limits over staying strictly within a by-the-numbers approach. The brightest guiding light was the ability to design the mission to enable human missions to Mars. Figure 1 was created specifically to inspire the team in this regard. Even so, the team made a concerted effort to tie the concepts to reasonable budget and schedule considerations. Wherever possible, references and resources have been used as the foundation for making decisions.

Figure 1: The L-Dorado mission is designed to treat the asteroid as a critical stepping stone on the way to Mars.

In broad strokes, the remainder of the report discusses the science objectives, the planned technology demonstrations, the mission schedule and budget, and public outreach activities. Trade studies are discussed in critical topic areas. Equations are supplied primarily in the engineering and trajectory sections. Time constraints prevent extensive justification for every decision and, in these cases, a mix of references, discussions, and rationalization are used to bridge the gaps in knowledge.

2.0 Mission Overview

The mission overview section presents the framework of the mission. Subsequent sections build on the work here.

2.1 Mission Statement

The mission statement defines the overarching goal that the team has worked to. Considering the problem statement and the ground rules provided by the competition, the team formulated the following mission statement:

Mission Statement: Project Explorer brings astronauts to the lunar orbiting ARM asteroid in 20XX to characterize, extract, and utilize its resources in order to demonstrate its potential benefits for humanity and for future exploration missions.

2.2 Objectives

The short term objective of this mission is to demonstrate the technologies necessary to achieve cis-lunar manned missions and extract resources from an asteroid in a lunar DRO orbit. The long term objectives are the transformation of the asteroid or related targets into outposts for future deep space missions and a technology proving ground. During the entire execution of the mission, benefits and involvement of the broader public are of special importance.

The objectives are defined to bind the problem in simple, straight-forward requirements. Table 1 below lists the chosen objectives:

Table 1: Four fundamental objectives define the mission

Mission Objectives	
1	Safely bring and return humans to the ARM asteroid
2	Characterize the asteroid and its radiative environment
3	Demonstrate the feasibility of in situ resource use for long term missions
4	Involve and inspire the public

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2.3 Requirements

The requirements are written to support the objectives. Table 2 lists the requirements underneath the objectives.

Table 2: Mission requirements written to support the functional objectives.

Mission Requirements	
1	**Safely bring and return humans to the ARM asteroid**
1.1	Choose Delta V capacity sufficient to bring the mission elements to their respective destinations
1.1.1	Launch system decisions must be made with regard to mass and volume
1.1.2	Design trajectories to minimize required DeltaV
1.1.3	Account for LEO operations in the calculations
1.2	Communication uplink/downlink capacity must be sufficient for safety and mission objectives
1.2.1	Design to achieve a permanent link to earth
1.2.2	Maximize up- and downlink rates
1.3	The power system has to be sufficient for all mission phases/objectives
1.4	Rendezvous capability is required
1.4.1	GNC needs to provide required accuracy
1.4.2	Structure needs to provide docking structure
1.5	The trajectories have to account for schedule, available deltaV, revisit and human factors
1.5.1	The trajectory should enable a mission abort and crew return in case of an emergency
1.5.2	The trajectory should account for duration of crew stay and radiation environment
1.5.3	The trajectory should abide by the mission timeline
1.6	Safe return to earth for astronauts and samples
1.6.1	The max g-loads during reentry have to be within human tolerable limits
1.6.2	The reentry conditions cannot exceed the manned capsule's heat shield capacity
1.6.3	Design sample retrieval to avoid exposure or contamination
1.6.4	Retrieve at least 100kg of asteroid samples
1.6.5	Design the ECLSS to suffice for the entire mission duration and all operations
1.7	The thermal conditions for all mission phases and objectives have to be within acceptable margins
2	**Characterize the asteroid and its radiative environment**
2.1	Characterize the internal structure of the asteroid
2.2	Quantify the composition of the asteroid
2.3	Characterize the asteroid's radiative environment
3	**Demonstrate the feasibility of in situ resource use for long term missions**
3.1	Have the ability to extract resources
3.2	Have the ability to process extracted resources to useful compounds
3.3	Put short and long term experiments into place that show usefulness of extracted and processed ressources

3.4	Experiments must demonstrate long term foresight within the evolvable solar system exploration strategy
4	**Involve and inspire the public**
4.1	Maximize the outreach capability
4.3	Use international cooperation to leverage mission impact
4.4	Demonstrate that asteroids are viable resources for exploration and prosperity

2.4 Mission Architecture

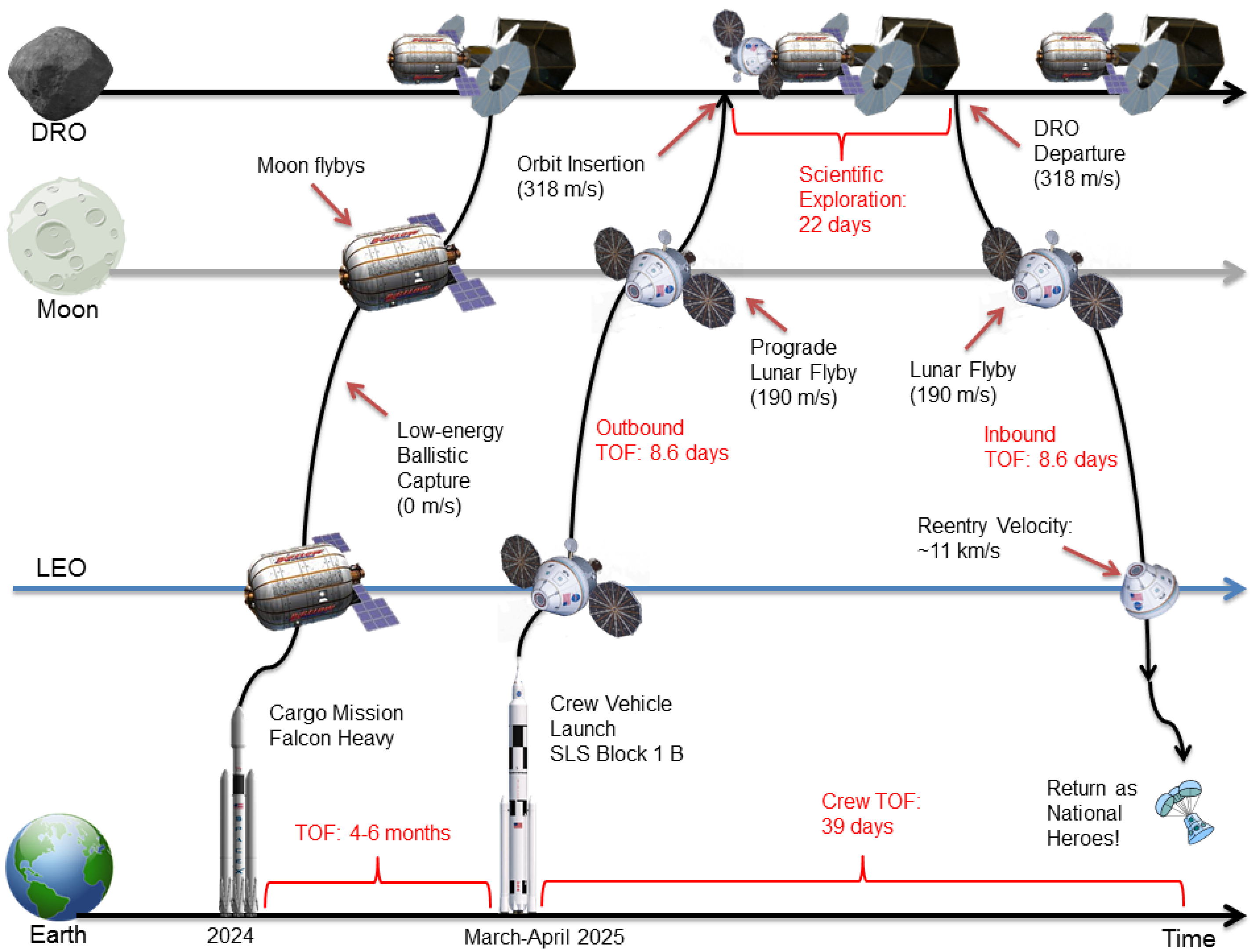


Figure 2: This figure outlines the main phases of the L-Dorado mission

2.5 Mission Design Choices

The mission architecture timeline is shown in Figure 2. Project Explorer plans to launch a cargo mission from Kennedy Space Center using a Falcon Heavy in August 2024. The payload of the cargo launch consists of an inflatable habitat that is used for scientific experimentation at the asteroid, and a service module used for the critical subsystems of the inflatable habitat. Using an inflatable habitat allows for significantly increased work and living space, lower launch mass and superior radiation protection. The technology has already been tested (GENESIS I and II) and an inflatable habitat will be installed in 2015 at the ISS (BEAM). Finally, as this kind of technology will be required for future deep space missions, this mission also serves as a role for technology demonstration.

For the cargo mission, the upper stage of the Falcon Heavy performs the necessary maneuver to inject the spacecraft into a low-energy ballistic capture trajectory. After approximately six months from launch, the cargo mission arrives at the asteroid’s DRO and docks autonomously with the ARM, which is assumed to have brought back an asteroid by 2024. Until the crewed Orion capsule arrives, the docked structure will remain dormant, except for standard operational checks.

The crew mission is launched in March, 2025 so that it arrives at the asteroid six months after the cargo mission launch. An advantageous launch date for the crewed mission has been identified in March/April 2025, to minimize delta V arising from possible inclination changes. A crew of three astronauts departs from Kennedy Space Center on board of the Orion Multi-Purpose Crew Vehicle (MPCV) which is launched into a Low Earth Orbit (LEO) by the NASA SLS Block 1 B. This launch vehicle is assumed flight proven at the proposed launch date and provides the necessary capacity as well as a safety margin with regard to design uncertainties. Additionally, the upper stage of the Block 1 B performs the Trans-lunar Injection (TLI) such that the crew vehicle is placed in a prograde lunar flyby trajectory. Upon arriving at a lunar altitude of approximately 100 km, a propulsive maneuver is performed and the crew vehicle is placed in a trajectory that intersects that of the asteroid’s DRO. Additionally, this maneuver is timed in such a way that the crew vehicle arrives at a location in the DRO when the asteroid-habitat system is approximately at the same location. Upon arrival at the DRO, approximately 8.6 days since launch, an orbit injection maneuver is performed. Successively, the crew vehicle docks to the asteroid-habitat system and inflates the science habitat. The astronauts have a total of 22 days, which corresponds to 2 DRO orbital periods, to perform the necessary scientific experiments and exploration at the asteroid, including EVAs. At the end of the second DRO revolution, the crew vehicle undocks from the asteroid-habitat system and departs the DRO. After executing a propulsive maneuver while doing a flyby of the Moon, the crew vehicle is on an Earth return trajectory. 8.6 days after departing the DRO, Orion arrives at Earth, reenters the atmosphere and splashes down in the Pacific Ocean. The science habitat, experiments and ARM are kept docked with the asteroid and are used for future exploration and technology demonstration missions.

It was attempted to restrict the number of launches as much as possible. However, only one launch using the SLS Block 1B is very close to the required payload. Block 2 would be able to lift more, but the availability is not ensured as no test date has been scheduled so far. Due to uncertainty and to reduce risk it was therefore decided to go with two launches. Nonetheless, as the design matures in the future and uncertainty reduces, the concept might be reduced to one launch.

The Falcon Heavy was chosen because it will be flight proven and, with an estimated cost per launch of $ 255 Million, only about ⅔ of the price of a comparable launcher like the Delta IV Heavy. It is assumed that the payload capability for trans lunar injection (TLI) will be at least 13.2 t, as this is given as the throw mass to Mars.

3.0 Science

The science section discusses the context behind the decisions made, then presents the science objectives and the instruments chosen to achieve them.

3.1 Context

Small asteroids are not hydrostatically adjusted, and are consequently not differentiated and have variable shapes (from ellipsoidal to more complicated morphologies, e.g., contact binaries). A layer of regolith generated by impacting micrometeorites often mantles their surface. The internal structure of asteroids is unknown, but limited surface observations suggest a high variability in porosity, with asteroids ranging from rubble-piles sticking together from low self-gravity to cohesive rock.

C-type asteroids are defined by their spectroscopic properties, such as an extremely low albedo (0.01-0.3) and a strong water absorption at 3.1 microns. Their spectra are most similar to those of carbonaceous chondritic meteorites of CI and CM types, which contain variable but generally high volatile and water abundances, silicates, oxides, and elemental metals (Table 3).

Table 3: A listing of what may be present in the asteroid.

	carbonaceous chondrites							ordinary				enstatite	
	CI	CM	CO	CV	CK	CR	CH	H	L	LL	R	EL	EH
Si	10.5	12.9	15.9	15.6	15.1	15.3	13.3	16.9	18.5	18.9	15.8	18.6	16.7
Ti	0.042	0.058	0.078	0.098	0.13	0.11	0.047	0.060	0.063	0.062	0.05	0.058	0.045
Al	0.86	1.18	1.43	1.75	1.61	1.27	1.06	1.13	1.22	1.19	1.07	1.05	0.81
Cr	0.265	0.305	0.355	0.360	0.366	0.375	0.343	0.366	0.388	0.374	0.362	0.305	0.315
Fe	18.2	21.0	24.8	23.5	23.6	24.0	40.4	27.5	21.5	18.5	24.25	22.0	29.0
Mn	0.190	0.170	0.165	0.145	0.146	0.170	0.106	0.232	0.257	0.262	0.228	0.163	0.220
Mg	9.7	11.7	14.5	14.5	14.8	13.9	12.3	14.0	14.9	15.3	12.9	14.1	10.6
Ca	0.92	1.27	1.58	1.90	1.72	1.38	1.14	1.25	1.31	1.30	1.20	1.01	0.85
Na	0.49	0.41	0.41	0.33	0.319	0.323	0.182	0.64	0.70	0.70	0.659	0.580	0.680
K	0.056	0.040	0.035	0.031	0.029	0.030	0.021	0.078	0.083	0.079	0.068	0.074	0.080
P	0.102	0.090	0.104	0.099	0.043	0.122	~0.1	0.108	0.095	0.085	~0.07	0.117	0.200
Ni	1.07	1.20	1.40	1.34	1.27	1.36	2.45	1.60	1.20	1.02	1.44	1.30	1.75
Co	0.051	0.058	0.069	0.066	0.064	0.067	0.115	0.081	0.059	0.049	0.070	0.067	0.084
S	5.9	3.3	2.0	2.2	1.58	1.31	0.25	2.0	2.2	2.3	4.07	3.3	5.8
H_2O	18.0	12.6	0.6	2.5	0.8	5.7	0.1	–	–	–	–	–	–
C	3.2	2.2	0.45	0.56	~0.1	1.44	~0.8	0.11	0.09	0.12	0.058	0.36	0.40
O	30.0	32.0	36.5	34.8	35.5	31.2	25.6	35.7	37.7	40.0	37.3	31.0	28.0
Fe^{0}/Fe^{tot}	0.0	0.0	0.1	0.1	0.0	0.43	0.85	0.6	0.3	0.15	0.0	0.74	0.65
Ir(ppb)	460	595	735	760	767	642	1030	760	490	360	614	525	565
Au(ppb)	144	165	184	144	136	139	202	215	162	140	183	225	330

CI, CM, CO, CV, H, L, LL, EH and EL groups, data from Wasson and Kallemeyn (1988).
CK group, mainly from Kallemeyn *et al.* (1991). H_2O and O from Mason and Wiik (1992a); their Si/Mg ratio was used with 14.8 wt% Mg to calculate the Si content.
CR group, mainly from Kallemeyn *et al.* (1994). H_2O, C and O from Mason and Wiik (1992b); their Si/Mg ratio was used with 13.9 wt% Mg to calculate the Si content.
CH group: Analysis of Allan Hills 85085 (Wasson and Kallemeyn, 1990). The Si/Mg ratio in bulk silicate (Scott, 1988) was used with 12.3 wt% Mg to calculate the Si content. Carbon content: Grady and Pillinger (1993).
R group: Schultze *et al.* (1994). The Si/Mg ratio from the composition used by Bischoff *et al.* (1994, p. 273) to calculate production rates of cosmogenic nuclides was used with 12.9 wt% Mg to calculate the Si content.

Undifferentiated chondrites contain calcium-aluminium inclusions, which are 4.567 billion years old and are the oldest materials found in the solar system. Consequently, C-type asteroids are believed to be amongst the building blocks of terrestrial planets as well as the rocky cores of gas giant planets. Asteroids surfaces have thus interacted with the space environment for up to several billions of years, and have been subjected to variable degrees of space weathering through the effects of cosmic rays, irradiation and micro-impacts. Space weathering is a poorly understood process, but is thought to decrease the overall albedo of the surface through seeding nanophase iron in the regolith grains. Organic compounds, including amino-acids, were found at the surface of Itokawa by the Hayabusa probe. Amino acids are the building blocks of prebiotic chemistry, and may have formed in-situ from the interaction of UV light with the asteroid material. It has been suggested that life may have originated on an asteroid, which later delivered it to Earth. Consequently, asteroids constitute an outstanding science target for NASA to understand the origin and evolution of the solar system, as well as the origin and evolution of life.

3.2 Science Objectives

The second Planetary Science Decadal Survey (2013-2022) written by the National Research Council for NASA identified three Priority Questions:

1. Building New Worlds - Understanding Solar System Beginnings
2. Planetary Habitats - Searching for the Requirements for Life
3. Workings of Solar Systems - Revealing Planetary Processes through Time.

For primitive bodies in particular, the Decadal Survey further identifies two principal goals:

1. Decipher the record in primitive bodies of epochs and processes not obtainable elsewhere
2. Understand the role of primitive bodies as building blocks for planets and life

The NASA NEO/Phobos/Deimos/Small Bodies Strategic Knowledge Gaps (SKG) report stresses a set of knowledge gaps regarding the radiation environment (gaps I-A,B), and the regolith geotechnical properties (gaps II-C). Several of these gaps coincide with long term Mars objectives identified in the corresponding SKG report.

The science objectives of the mission are designed to build on preliminary science performed on a 500 tons asteroid during the ARM mission, under the assumption that a small asteroid is captured, as opposed to a smaller boulder. The data products required from the ARM mission are:

- Topographic map (5 cm resolution)
- Gravity field (gravity moment to order 6)
- Magnetic field (magnetic moment to order 6)
- Spatially resolved hyperspectral map of the surface (1 m resolution)
- Subsurface resistivity profile(s): During the capture phase, a ground penetrating radar (GPR) is operated at a distance of less than 4 m from the surface to achieve sub-meter vertical resolution in the subsurface. If data is acquired while the asteroid is still spinning, the GPR may be incorporated as a non-mobile part of the capturing system.

These requirements inform the science phase of the mission about potential ice-rich locations, mineralogies of interest, as well as potential terrain and/or operation restrictions.

The science objectives of this mission directly respond to the Decadal Survey three Priority Questions (Table 4), as well as the SKG.

Science objectives of the mission addressing Priority Questions 1 and 3 are:

- Determine the elemental, mineralogical and petrological composition of the asteroid
- Characterize its surface composition and processes

- Probe its internal structure
- Characterize water phases contained by asteroid materials.

A minimum of 100 kg of asteroid material sampled at the surface and at depth is returned to Earth for further analysis responding to Priority Questions 1 and 3, including isotopic composition (e.g., D/H ratio), radiometric dating (e.g., Rb/Sr chronometer), and scanning electron microprobe (SEM) imaging.

Science objectives responding to Priority Question 2 are:

- Identify organic compounds at the surface and within the asteroid.

Opportunistic science is conducted as part of the Technology Demonstration phase (Section 4.0) to characterize the radiative environment at the surface of the asteroid and analyze rock samples acquired at a depth of 4 m.

A detailed timeline for the Science and Technology Demonstration is provided in Section 5.2.

The science payload of the mission is designed to fulfill all measurement requirements dictated by the science objectives (Table 4).

Table 4: The science traceability matrix.

y Question 1. Building New Worlds: Understanding Solar System Beginnings					
y Question 3. Workings of Solar Systems: Revealing Planetary Processes through Time					
ectives	**Measurement Objectives**	**Measurement Requirements**	**Instrument Requirements**	**Data Products**	**Heritage**
he and f the	Measure the elemental chemistry	<1 wt% accuracy	Alpha-particle X-ray spectrometer	Weight abundances of elements	MSL, MER, Philae
	Measure mineralogy	<1 wt% accuracy	X-Ray diffraction	Weight abundances of mineral phases	MSL (CheMin)
e osition s	Acquire HR images of disturbed and undisturbed surface	0.5 cm resolution	High resolution visible camer	Images of the surface	
	Acquire HR microsocopic images of disturbed/undisturbed surface	50 microns resolution	Microscopic Imager	Images of fine structures of regolith	MSL, MER
	Sample surface material for analysis and return	10 kg capacity	Chisel in a cup sampler	Regolith samples	
ternal	Drill a borehole for return	Non-destructive to preserve "stratigraphy", about 2m depth	Percussion-vibration drill	2m log for return, extension rods	ExoMars (Selex Galileo)
	Image the borehole	<1cm spatial resolution	Compact VISIR spectrometer,	Mineral map of the borehole	UCIS (JPL), Ma-Miss (ExoMars)
	Measure the temperature structure	resistance to temperatures <1800 K	Thermocouple	Temperature depth-profile	
	Measure the density structure	60 cm penetration depth	Density logging probe	Density depth-profile	
	Measure the H-content	50 cm penetration depth	Gamma ray/neutron logging probe	H index profiles	
e water ned by rials	Identify and quantify all water phases (ice, adsorbed, hydroxyl)	1 ppm sensitivity	Gas chromatography-Mass spectrometer (GC-MS)	Inventory of hydrated phases	MSL (SAM)
y Question 2. Planetary Habitats: Searching for the Requirements for Life					
anic the ithin	Detect, identify and quantify organic compounds	2 ppb sensitivity for methane	Gas chromatography-Mass spectrometer (GC-MS)	Inventory of organic coumpounds	MSL (SAM)
Demonstration					

rge eroid	Drill and extract bulk material for resource extraction	Provide 1 metric ton of material in less than 4h	Plasma drill with 4 drill bits	1 metric ton of raw material	
ources erial ate its ential	Extract the water	60% extraction efficiency	Pyrolysis chamber	Up to 10 liters of water	
	Perform hydrolysis for potable water, as well as separate O and H for fuel	85% efficiency	Oxygen Generation Assembly (OGA)	Rocket fuel and potable water	ISS
	Separate the oxides		Compact electrostatic separator	Metal oxides separated from regolith	
	Grow lettuce in asteroid regolith with asteroid water	Compare with regular soil and ECLSS water	Experimental Garden	Relative humidity, pH, temperature and dissolved O time series	
	Fly a steam rocket from asteroid water		Steam rocket		
lding ed	Perform an in-situ experiment on radiation shielding	4 experiments, 3 regolith thicknesses, as well as non shielded	10 MeV sensitivity	Calibration for regolith and H radiation shielding capabilities	MSL (RAD)
	Perform an in-situ experiment on heat shielding	11 experiments, 5 sintered, 5 loose regolith, as well as one non shielded		Cooling profiles and shielding power of loose and sintered regolith	

3.3 Science Payload

Non-destructive coring drill: Extracts and stores samples along a 2 m long core with a >2 cm diameter.

Heritage: ExoMars/SELEX Galileo.
Trade space: Mars 2020 drill - cannot drill deep enough.

Gamma-ray, neutron porosity and density logging probes: Measures depth-profiles of hydrogen content and density. Is operated during coring phase. Penetration depth of about 60 cm. Rigid wiring for microgravity operation.

Heritage: Schlumberger probes.
Trade space: remote sensing neutron detector operated from the surface (DAN, MSL) - cannot resolve H vertically.

Compact active-source hyperspectral visible-near infrared (VISIR) camera: Images the borehole at VISIR wavelengths. Is operated during coring phase. Spectral resolution of 0.01 microns, spatial resolution of 2 mm. Camera diameter < 2 cm.

Heritage: UCIS (JPL), Ma-Miss (ExoMars).
Trade Space: inspection of the extracted core - need to unseal the stored sample before Earth return.

Alpha-particle X-Ray Spectrometer: Allows measurement of elemental chemistry. Contact surface area of about 2 cm in diameter. Running time < 30 minutes.

Heritage: MER, MSL, Philae.

X-Ray Diffractometer: Allows measurement of mineral abundances. Analysis of about 50 mm^3 of powdered sample. Running time < 10 hours.

Heritage: CheMIN (MSL).

Gas chromatography-Mass Spectrometer: Allows water and organics detection and identification. Analysis of about 50 mm^3 of powdered sample. Running time < 10 hours.

Heritage: SAM (MSL).

High Resolution Camera: Documents surface samples. Multispectral capabilities in the visible range. Spatial resolution of about 3 mm.

Heritage: Mastcam (MSL).

Microscopic Imager: Documents surface samples. Resolution of about 15 microns.

Heritage: MI (MER), MAHLI (MSL).

Radiation detectors: Identifies radiation levels while in transit and in situ. The detectors are used both in the Science phase and in the Technology Demonstration phase.

The Orion capsule has a built-in detection system comprised of a Radiation Area Monitor (RAM) and a Battery-operated Independent Radiation Detector (BIRD), which furnishes radiation levels for the crew.

Heritage: RAD (MSL).
Trade space: CRaTER (LRO) - RAD measures high-energy protons, ions, neutrons, and gamma rays, whereas CRaTER, despite being more precise, is restricted to galactic cosmic rays and solar energetic protons. RAD is also lighter, less voluminous and more energy efficient.

Thermocouples: Measures temperature in the borehole, on the asteroid, and is used for thermal shielding and sintered regolith experiments during the Technology Demonstration phase.

Scoopers: Allows for the safe capture and handling of surface asteroid materials to avoid contamination with the space suit. Will be used to transfer surface materials to a sealable safe container. This tool is in development.

Chisel-in-a-cup Samplers: Allows crewmember to chip off pieces of rock on the asteroid while containing the debris. A chisel, or sharp surface will be contained within a cup-shaped object which has a handle protruding from the back end of the “cup”. This allows the crewmember the ability to place the cup on the surface of the asteroid, use the protruding handle on the back to operate the chisel in which will chip samples off the surface of asteroid. The cup will be sealable after use to contain the debris. This tool is in development.

Lab Crusher: Crushes small volumes of rock samples in microgravity. Detailed in Section 4.3.4.

Dust Handling Work Station: Provides a clean environment for handling dust in microgravity inside a spacecraft. It consists of a working station with directional airflow that collects loose particles in a vacuum-cleaner-like bag, typically released when handling powders in microgravity. This concept is used in gray rooms on Earth to keep working spaces clean.

Centrifugal Microgravity Sieve: Separates desired grain sizes from crushed samples to be analyzed by the science instruments (APXS, XRD, and GC-MS).

Heritage: US Patent US20130270158.

Sample Packing Boxes: Allow collection of samples in several triple sealed containers, protecting samples from irradiation and contamination. Boxes walls are 1 cm thick (aluminum) with steel meshing inside. Fifteen boxes are on board in two sizes, allowing 500 kg of samples to be stored.

Heritage: Apollo surface missions

4.0 Technology Demonstration

4.1 Context

C-type asteroids constitute a promising resource for human space exploration. Hydrogen extracted from volatiles and hydrated phases, is an efficient insulator against secondary radiation from galactic cosmic radiation (GCR) and solar energetic particles (SEP). Regolith might also be used as a thermal and radiation shield. Oxygen, extracted from volatiles, hydrated phases and oxides, may be used in the atmospheric revitalization system. Water may be used for life support. Both hydrogen and oxygen are efficient rocket fuels. Other phases, including silicates and metals from the oxides may prove to be useful resources too.

Consequently, redirected asteroids may be used as a physical platform to support interplanetary travel, providing fuel, life support fluids, and building material. The mission demonstrates the utilization of new technology that will pave the road to Mars.

4.2 Tech Demonstrations Objectives

The second Planetary Science Decadal Survey (2013-2022) underlines the need for new technology development. For the exploration of primitive bodies, the Decadal Survey recommends the development of “remote sampling and coring devices”.

The NASA NEO/Phobos/Deimos/Small Bodies Strategic Knowledge Gaps (SKG) report stresses a set of knowledge gaps regarding assessment of small bodies regolith geotechnical properties (gaps II-C,D), shielding properties (gaps II-E), as well as excavation, collection and extraction of small bodies resources (gaps IV-A,B). Several of these gaps coincide with long term Mars objectives identified in the corresponding SKG report.

The Technology Demonstrations objectives of the mission are designed to bridge the gap between the need for in-situ asteroid resource extraction emphasized by the Decadal Survey and SKG and the lack of available technology to do so. They are:

- Extract 5 kg of water from asteroid material
- Purify water and demonstrate its utilization potential
- Extract metal oxides from regolith
- Sinter regolith/crushed asteroid material into cohesive bricks
- Measure thermal and radiation shielding properties of asteroid-derived material
- Deploy SPHEREs experiment to assess their surface characterization capabilities

A detailed timeline for the Science and Technology Demonstration is provided in Section 5.2.

The technology payload of the mission is designed to fulfill all measurement requirements dictated by the technology demonstration objectives.

4.3 Technology Demonstration Payload

4.3.1 Ring Plasma Drill

Plasma drilling technology is a recently explored drilling technique that is potentially able to substitute conventional, contact-based rotary drilling systems. It is currently matter of active research and only few companies are investigating IN plasma drilling devices, e.g. Zaptec (ZAPTEC official website) and Geothermal Anywhere (GA) Drilling (GA Drilling official website) (Figure 4).

Figure 3: GA Drilling's PLASMABIT (PLASMABIT, GA Drilling).

Figure 4: GA Drilling's PLASMABIT in action (GA Drilling official website).

Specifically, Zaptec and the US lunar mining company Shackleton Energy (SEC) (Shackleton Energy, Official website) have recently signed a memorandum of understanding (MoU) to explore the potential for developing lunar drilling tools ("Shackelton and Zaptec to develop lunar technology for drilling on the moon", mining-technology.com). In parallel, specific plasma drilling systems are being developed by Zaptec to operate on asteroids and the moons of Mars (Johansen B. W. et al., "A New Plasma Drilling Technology with Applications for Moon, Asteroid, and Mars Exploration and ISRY"). Indeed, conventional drilling technologies used on Earth are difficult to implement in space because of their requirements about mass, volume, and power, and their reliance on gravity (Johansen B. W. et al., "A new plasma drilling technology for the moon, asteroids, and mars"). Conventional deep drilling technologies appear incredibly expensive to be used in space missions.

Therefore, a new plasma drilling technology and approach has been recently developed by the Zaptec Inc. company in order to achieve practical, affordable, and reliable deep-drilling on the Moon, asteroids, Mars, and its moons (Johansen B. W. et al., "A New Plasma Drilling Technology with Applications for Moon, Asteroid, and Mars Exploration and ISRY"). This new technology enables deep subsurface access, exploration and sampling for science and in-situ resource utilization (ISRU). The drilling system consists of a freely advancing drill head (Figure 6) tethered by a power cable to a power source topside and high voltage generator downhole (Johansen B. W. et al., "A new plasma drilling technology for the moon, asteroids, and mars"). The drill advances by generating a high-energy density plasma at the drill head which disintegrates and pulverizes the target rock (Johansen B. W. et al., "A New Plasma Drilling Technology with Applications for Moon, Asteroid, and Mars Exploration and ISRY"). The system is able to deliver high energy plasma discharges with low mass and low volume power transformers located in the drill head section. The fine dust from drilling goes through the unit, is analyzed, and then sprayed into a dust exhaust in contact with the surface vacuum (Johansen B. W. et al., "A new plasma drilling technology for the moon, asteroids, and mars"). The Zaptec system is anticipated to reach 50 to 100 m depths with less than 250 kg of gear topside and 1 kW of peak power on asteroids and on the moons of Mars, (Johansen B. W. et al., "A New Plasma Drilling Technology with Applications for Moon, Asteroid, and Mars Exploration and ISRY"). Zaptec is planning to continue to mature its lightweight, energy-efficient drilling concept with laboratory and field tests over the next years (Johansen B. W. et al., "A new plasma drilling technology for the moon, asteroids, and mars").

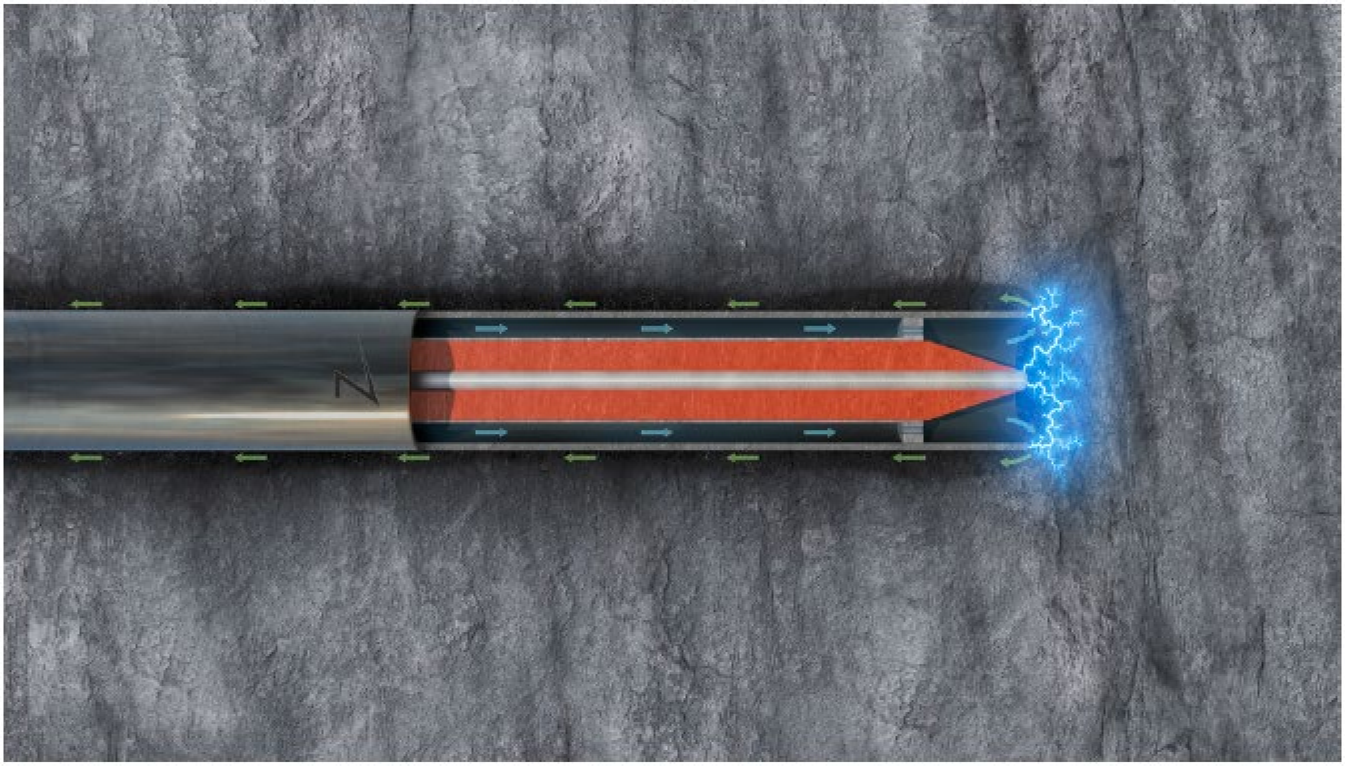


Figure 5: The Zaptec Plasma Drilling System in action: the high voltage, low power sparks delivered through the Zaptec drill head create microscopic plasma channels that explode the rock (Johansen B. W. et al., “A new plasma drilling technology for the moon, asteroids, and mars”).

In order to extract samples from the asteroid sub-surface, six plasma drill heads disposed according to a ring shape having 8 cm radius are used in order to reach a 4 m depth. After the first drilling, the ring is rotated and a new drilling is performed. This procedure is repeated until the asteroid sample is obtained. For the last stages of mining, horizontal drilling is used in order to cut the bottom of the sample.

4.3.2 Pyrolysis Chamber

Assumption: C-type meteorites derive from C-type asteroids (Nelson)

The quantity of ore to be processed depends on what percentage of water is present in the ore. This value varies from 0.5 to 18 % for C-type asteroids, where it is derived from meteorite samples (Hutchinson). Since meteorites are heated during reentry, the asteroids probably contain more water.

Ore quantity = Extraction efficiency*extracted water quantity/water weight percentage

Assuming a conservative scenario of 2 % water content, a extraction efficiency of 0.8 (Zacny) the required ore quantity would be of 320 kg. The pyrolysis chamber described later has a capacity of 500 kg of ore, assuming a density of 3400 kg/m^3.

A pyrolysis system is used to extract the volatiles from the crushed ore, and this system is shown in Figure 7. The use of steam as a working fluid (described later) and a 10 °C condensation chamber has been discussed by (Nichols). Dimensions can be found in Appendix B.

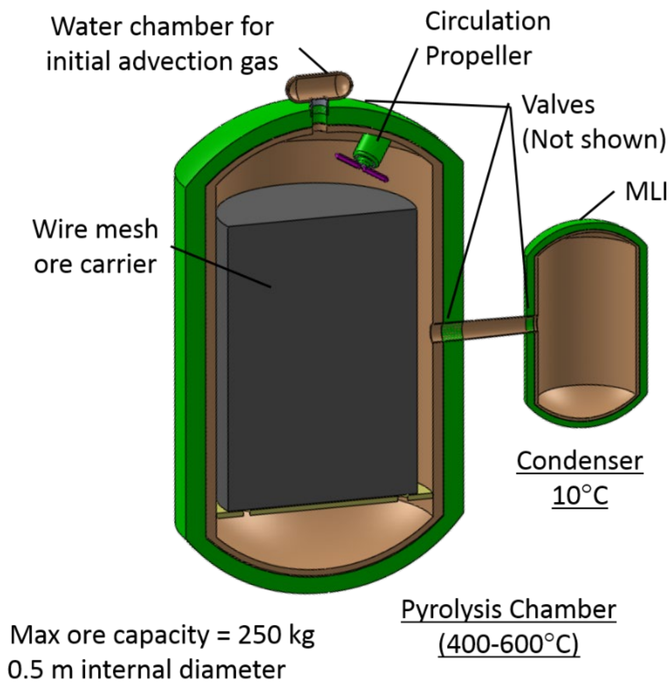


Figure 6: Pyrolysis Water Extraction System

Once the ore is crushed and extracted, it is moved to the ore carrier. The ore carrier is a cylinder 1 m high with a radius of 0.17 m consisting of wire mesh. The top can be removed to allow regolith to be inserted. Once the ore is in the carrier, a cylindrical dust sheath is added to stop regolith escaping through the holes. The astronaut then conveys the carrier to the pyrolysis chamber, and closes the chamber.

The water processing system consists of two chambers - the pyrolysis chamber and a storage container. Both chambers are well insulated with MLI, and a heater keeps the storage container at a constant 10 °C.

Once the ore is in the chamber, it is heated, causing the water to either sublimate or outgas. Once the pressure within the pyrolysis chamber exceeds 1 atm, a pressure valve between the pyrolysis chamber and the storage container opens, allowing vapor to flow. Initially the chamber will contain only water vapor, but as more vapor enters the chamber it will liquefy. This chamber is also insulated and an active heating/cooling system keeps the chamber at a constant 10 °C.

We examined three different methods to heat the regolith - resistive heating, solar thermal parabolic mirrors and a microwave emitter.

Resistive heating: Simple heating resistors are placed around the chamber and powered to heat the chamber. Simplicity and energy efficiency make this solution light, easy to use, hazard free and its development relatively short. An additional water chamber that contains 0.5kg of water is mounted on top of the pyrolysis chamber and connected by a valve. Once the ore is in the chamber and the chamber sealed, the valve between the water chamber and the pyrolysis chamber is opened. The water enters the pyrolysis chamber and becomes vapor upon contact with the vacuum. A propeller mounted in the bottom of the chamber starts circulating the water vapor, ensuring convection occurs between the walls and the ore.

Solar parabolic mirror: This solution consists of directing solar heat into the heating chamber using a large (>20 m^2) parabolic mirror. Its size requires a large, light structure made of deployed hinges to hold the mirror. The astronaut has to bring the oven filled with raw material in place and then the mirrors focus to heat. Once extracted, the mirrors are defocused, the oven removed and emptied. The main drawbacks of the solution are the long operation for the hinge installation, the necessity of a deployment and focusing mechanism. The energy source is abundant however.

Microwaves: With this solution the material is heated by microwaves focused into the chamber. Advantage are that this method does not require either the water chamber or the propeller for circulating the gas, and it will not require as much setup time as the parabolic mirror solution. One drawback is it requires a magnetrode which has to be articulated in order to avoid hot spots. Also, the emitter needs to be cooled and the waste heat radiated away. A commercial 30kW emitter on Earth weighs 7 kg and has an efficiency of power to microwave energy of 78 % (L3). The microwave emitter heats the water and other volatiles, causing them to outgas (Zacny).

Table 5 summarizes this trade effort. The criteria are weighted from 1 to 5 and each solution is given a score based on the performance in regard to the concerned criteria. The three options are a solar parabolic mirror, microwaves and resistive heating.

Table 5: Trade study of different ways of heating regolith in the pyrolysis chamber.

		1	2	3
Heading method **Criteria**	Weight (1 -> 5)	**Solar parabolic mirror**	**Micro-waves**	**Resistive heating**
Weight	5	3	3	5
Crew safety	5	2	3	5
Power	4	5	2	3
Operation simplicity	4	3	4	4
Scalability	4	4	3	5
Processing time	3	3	5	4
Volume	3	3	4	5
Cost	2	3	3	4
Energy efficiency	2	5	2	3
Waste heat	2	5	3	4
Total	34	**97**	**99**	**133**

The most important criteria are the crew's safety (no hazard to crew), its weight and maturity (which drive its cost). Power, Operation simplicity and processing should be optimized to reduce EVA time and energy consumption. They are equally important as scalability which allows the process to be scaled to actual ISRU operations. The first and third solution required an initial presence of a heat conducting gas to allow the heat to flow from the oven walls to the raw material.

Resistance heaters were selected with the trade study shown in Table 5 due to safety concerns and reduced setup time. Parabolic mirrors focusing 30 kW near where astronauts are working is dangerous, and the outside temperature of the pyrolysis chamber would be dangerously hot. The microwave emitter will produce significant waste heat which needs to be radiated away, and the microwave radiation may not stop at the edge of the pyrolysis chamber.

The resistance heater raises the temperature to 100 °C if the regolith contains a high percentage of water ice (>1 %). If the water is contained in hydrated minerals, the temperature will be 400 °C to 800 °C (Sonter). These numbers are because 800 °C is required to extract H_2O from talc and other phyllosilicates, and 500 to 600 °C for gypsum. The type of hydrated minerals present in the regolith has already been determined by the gas chromatographer and mass spectrometer results. The heaters can operate with up to 50 kW of energy from the ARM spacecraft, and can operate for up to 11 days. The total mass of ore that can be processed is 500 kg, assuming a bulk density of 3400 kg/m^3. The energy required to heat 1 kg of ice is shown in Appendix C to provide an order of magnitude calculation on the time required to extract the water. It is expected that 11 days will be plenty of time.

4.3.3 Microgravity Centrifuge

Once a sufficient quantity of volatiles are in the storage chamber, the chamber is removed and inserted into a separate centrifugal system as suggested by (Orenstein). A cyclonic separator is a potential alternative for separation (Sonter). This system is be used to separate the dust, liquid and gas by density. Once the centrifuge is spun up, a valve in the chamber is opened, and the dust, other solids and liquids more dense than water flow out into another chamber due to centrifugal force. Once the dust is removed, the centrifuge is spun down, a fuel bag is placed over the centrifuge opening and the centrifuge spun up. 1 kg of water is allowed to flow into this bag.

4.3.4 Sintering Mirror and Mold

It is known that heat shields fabricated in-situ can provide thermal protection systems for spacecraft that routinely enter a planetary atmosphere. This production from extraterrestrial regolith will greatly diminish costs of launching the heat shield mass from Earth, with orders of magnitude of billions of dollars for future interplanetary missions (Hogue).

This mission is a great opportunity to experiment on in-situ sintering methods through the use of concentrated solar light via a parabolic mirror. It is expected (see Appendix) that, with a 0,3 m^2 mirror, that, with a 0,3 m^2 mirror, one can sinter about 1 kg of regolith every hour with an efficient ray displacement speed.

Sintering process

Regolith will be sintered by placing ARC-crushed regolith samples (in average 2x2x1 cm^3) on a ceramic mold and then redirecting sunlight through the mirror (see Figure 8); it is advantageous to use crushed, baked material as its size will be in general smaller and it will have less water (which could make sintering time longer if in excess). With the above efficiency we assume each sample can be sintered in less than 10 minutes, assuming a 3 g/cm^3 density for the crushed regolith. We have assumed that the production of 10 samples is enough, but depending on real conditions this number might change.

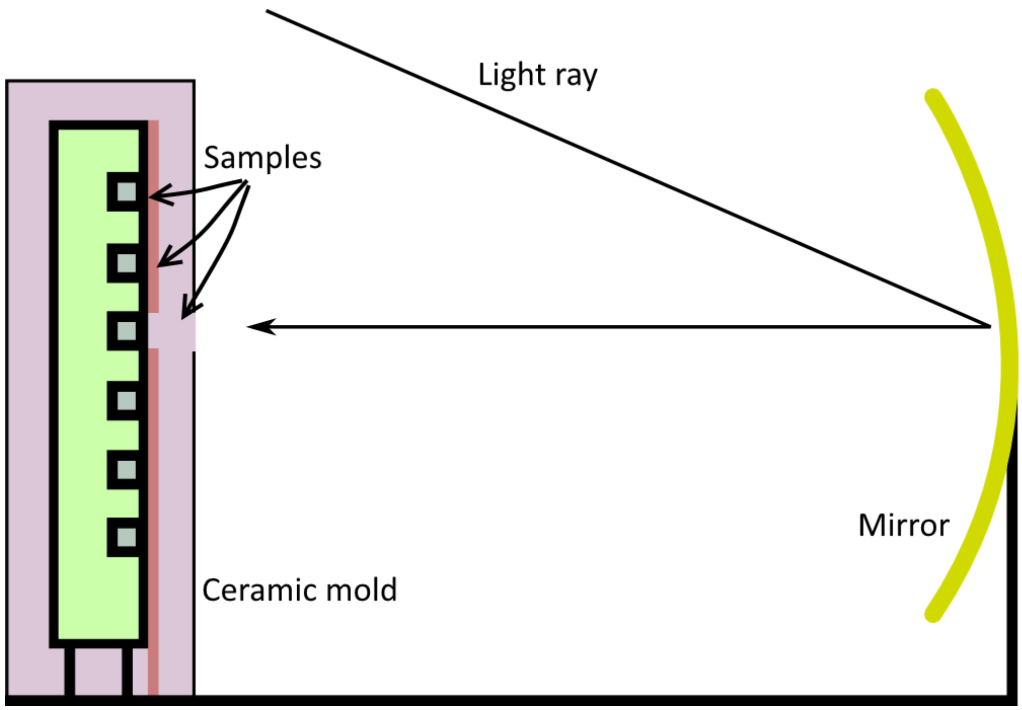


Figure 7: Sketch of the parabolic mirror reflecting sunlight onto the samples, held in a ceramic mold.

Experiments

Sintered material's thermal properties will then be tested using thermocouples. After sintering, the samples are hot, and measuring their temperature while cooling can teach us about their conductivity. They are left on top of the mold, with thermocouples taking measurements on temperature, these data being sent via a telemetry system to the science module and left there after the mission

4.3.5 Radiation Shielding

For testing the properties of the asteroid's regolith in radiation shielding, the astronaut pours various amounts of regolith in bags, each one holding one RAD detector. The bags are then tethered to the asteroid, and information from these detectors are then passed on to the science module via telemetry.

4.3.6 ARC (Asteroid Regolith Crusher)

Since the asteroid material is porous (6-12 % for C-types, (Britt)), the heating process is sufficient to extract its volatiles and the ore does not need to be crushed. Crushing is required for analysis of the rock as well as for the preparation of the sintering samples (see Section 4.3.4). Cone crushers are the most efficient in terms of productivity per mass of crusher. This technology seems adaptable to microgravity using a spring loaded feeder and an inert gas flow to move the grinded powder. The chamber that the powder will be contained in, will have an exit outlet, closed by a bag, much like a vacuum cleaner. The powder is stored in this bag. With the help of electrostatically charged tools, the astronaut moves the acquired powder from the bag to the sintering molds and encloses them. The remaining powder is sealed and bagged to be brought in the science habitat for further examination.

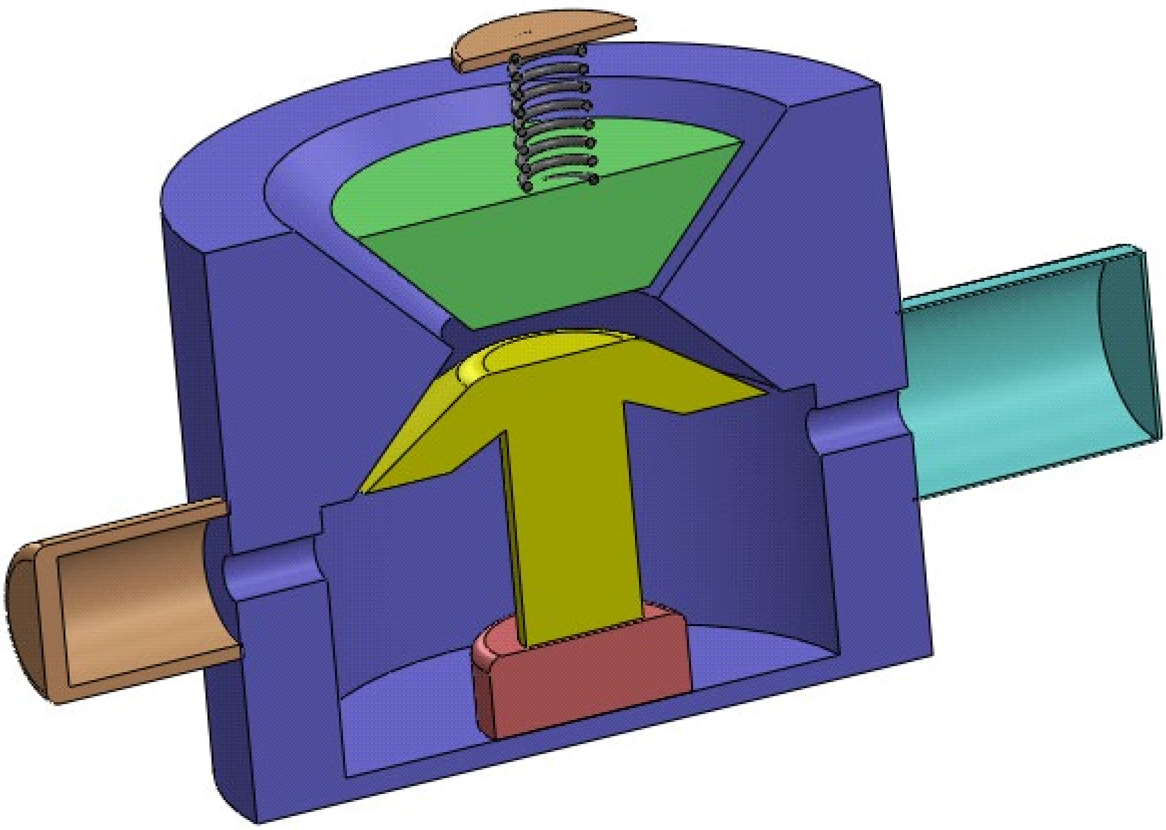


Figure 8: ARC conic crusher concept

4.3.7 EPM2 (Electrostatic Powder Manipulator for Microgravity)

In order to manipulate powder in microgravity, a selectively sticky surface can be used. Electrostatically charged surfaces retain grains of a certain size and particle affinity (Gupta). The influence of the static charge and maybe a high frequency variation of this charge might help to contain the entity of the grain size spectrum. Discharging the tool and charging the receiver tool archives manipulation of the powder. Fully closeable ice-cream-scoops or match boxes that can be opened and closed single handed could be used for bulk handling. This technology is at a very low TRL and needs to be developed. Relative to its complexity this cost is estimated at million USD (comparable to the space drill).

4.3.8 Compact Electrostatic Separator (CES)

To separate oxides from the bulk asteroid material, a process called electrostatic beneficiation (EB) can be employed (Prado). Initially one screens different sized grains by e.g. passing them through a mechanical grinder and sieving them. Given that different materials have different electrostatic affinities, i.e. absorb different amounts of charge depending on their material, their deflection inside an electric field are different as well. EB means to charge the grains via triboelectricity and pass them through an electric field, while having them inside a rotating drum that creates an artificial gravity environment (whose technology is similar to that of the Microgravity Centrifuge) . This process then effectively separates minerals according to their electrostatic affinities and densities. EB has been used efficiently on Earth and is even more efficient in vacuum (Prado).

Earth-based EB usually uses large containers for industrial-scale processing. However, rescaling is certainly possible. Also, and its use for extraterrestrial mining still needs to be demonstrated. One thus requires the development during the next ten years of compact, efficient separators, which we call CES. Power can be made arbitrarily low (low rotation and electric field) and we set a maximum power use of 50 W. Development cost until it reaches TRl 7 is estimated around 40 million dollars.

4.3.9 Steam Rocket

To demonstrate the ability to convert the processed water into rocket fuel, a demonstration steam rocket is included in the mission. 1 kg bags of water are placed in a small 15 kg drone (cubesat derivative). Pressure is applied to the bag, and small quantities are allowed to flow into the rocket chamber.

Parabolic mirrors (~0.5m^2) concentrate sunlight into fiber optics, which feed into the rocket chamber and superheat the water (Nakamura). The design this solution is based on can be found in Appendix D.

While it may appear to be a gimmick, this technology has a great potential for on or near asteroid operations. The fuel requires no complicated equipment to store as ice, requires fewer steps to process than LOX or H2, and is very rugged. Since propulsion only relies on vaporizing the fluid, the water can be heavily contaminated with dissolved material and still perform effectively. Demonstrating this technology with asteroid derived material is a great step forward to future automation of asteroid mining.

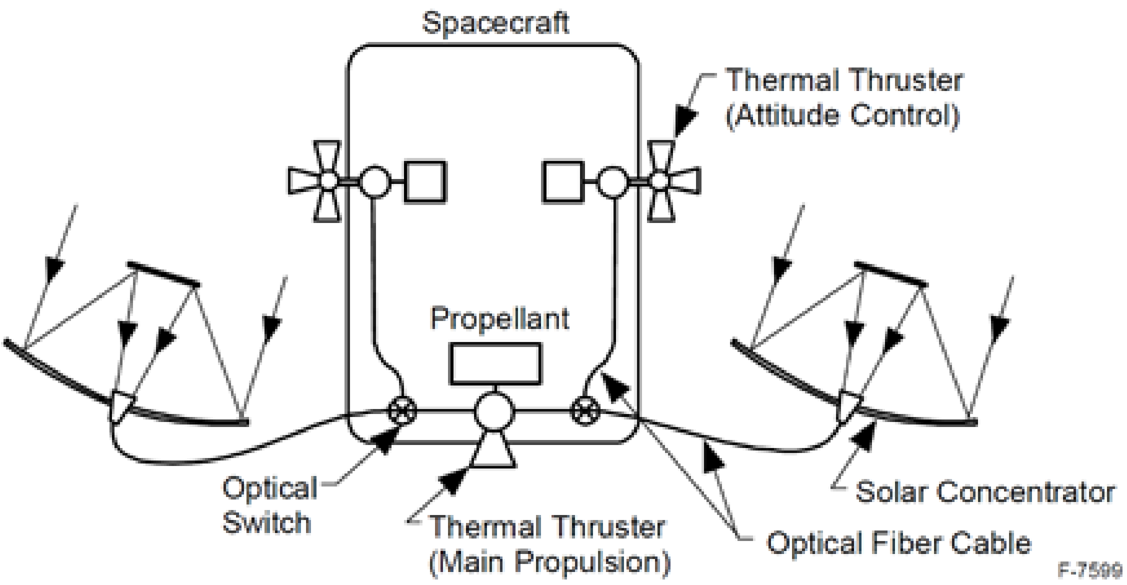


Figure 9: Steam rocket design (Nakamura)

4.3.10 Experimental Garden

The demonstration that a garden can be established on asteroid environment employing regolith as basis would be a great step forward in human advancement into space. The project of a hydroponic garden is proposed as follows. Eight plants are to be grown: four in asteroid regolith and four in Earth soil. In each group, 2 will be watered with asteroid-extracted water and the other two with Earth water. The plants will be illuminated by LED lights with the correct frequency and intensity to optimize their growth.

As for the choice of plant to be grown, lettuce was chosen due to its high bioedible mass fraction as well as a reasonably rapid growth, going from a seed to a baby green in less than a week. In what concerns outreach, this experiment is important to show the viability of growing plants on asteroid soil and/or with water extracted from it, especially if growth is enough for an astronaut to eat it.

The garden structure also includes humidity, pH and temperature detectors, as well as dissolved oxygen probe.

4.3.11 SPHEREs

See Section 6.6.

5.0 Operations

5.1 (Structure as needed for the proposed mission)

The mission starts in 2019, when the ARM spacecraft is launched to capture the NEO that will be the main destination of this mission. Following that, the timely deployment of the herein developed mission depends on the delivery of the asteroid to a lunar distant retrograde orbit in March 2024. A favorable launch window for the crewed Orion vehicle is in March/April 2025. The transfer time to the Asteroid-ARM constellation will be xx days. During a xx day stay the Astronauts will conduct experiments, outreach activities and set up long term science projects. After a xx day return flight to earth the Orion capsule will reenter the earth's atmosphere on xx.xx.xx.

Unmanned habitat?

Return mission?

Additional habitat?

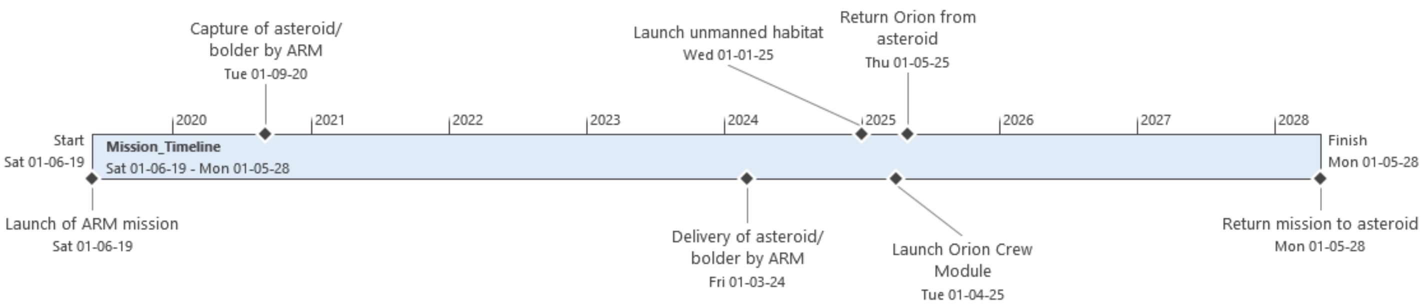


Figure 10: Mission timeline including the ARM

5.2 Science and Technology Demonstration Operation Plan

The Science and Technology Operation plan is detailed in Table 6, and illustrated by Figure 12.

Table 7 details scientific and technology demonstration related EVA activities.

Table 6: Science and Technology Demonstration Plan

Day	Activity onboard	Ongoing activity
1	Adaptation to environment	
2	Begin pre-breathing protocol	
3	EVA 1	science drill campaign (SDC)
4	Sample testing	SDC
5	APXS+XRD+GC-MS of surface samples	SDC
6	APXS+XRD+GC-MS of surface samples	SDC
7	Begin pre-breathing protocol + analysis	SDC
8	EVA 2	pyrolysis
9	Sample analysis	pyrolysis
10	Sample analysis	pyrolysis
11	Cosmic ray experiment prep (3 full hours)	pyrolysis
12	Cosmic ray experiment testing (3 full hours)	pyrolysis
13	Sample analysis	pyrolysis
14	Begin pre-breathing protocol + sample analysis	pyrolysis
15	EVA 3	Thermal/Radiation experiment (TRE)
16	Water purification + garden experiment start	Experimental garden (EG) +TRE
17	Steam rocket flight + H/O generation from hydrolysis	EG+TRE
18	Oxides separation	EG+TRE
19	Sample analysis	EG+TRE
20	Sample analysis	EG+TRE
21	Sample analysis + Astronaut sorbet	EG+TRE
22	Packing and disposing	EG+TRE

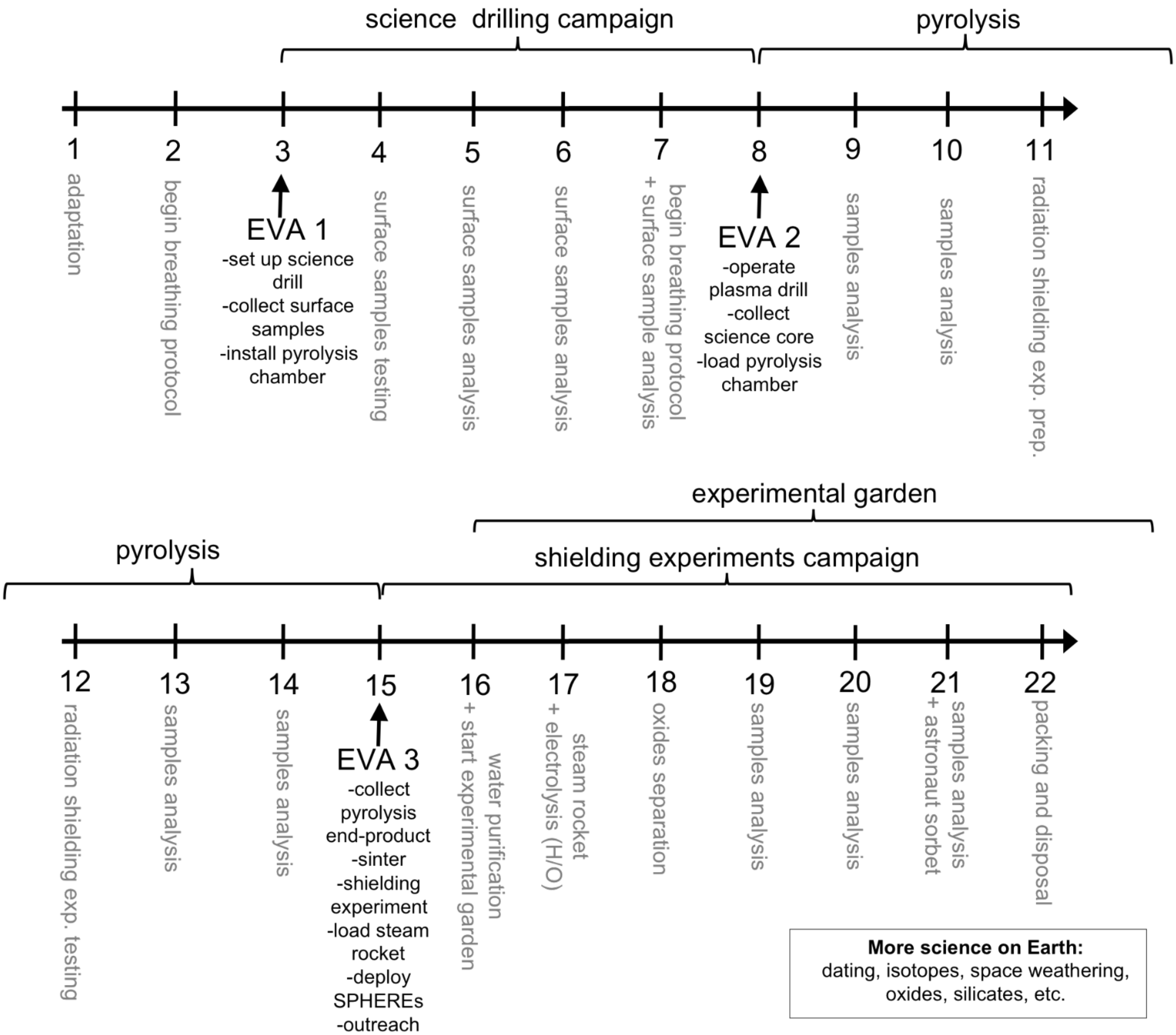


Figure 11: Science and Technology Demonstration Operations Plan

5.3 EVA Detailed Schedule

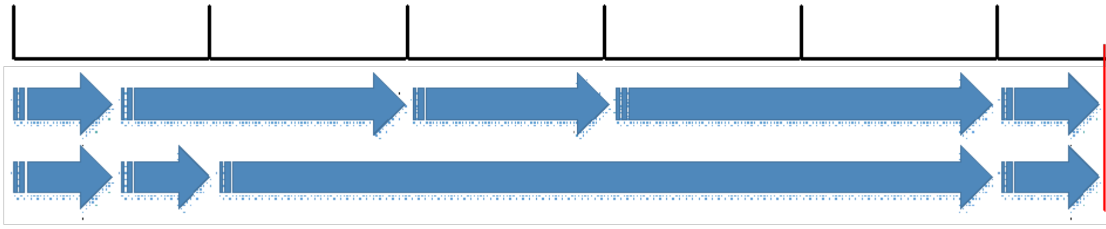
Table 7 details the Science and Technology Demonstration activity schedule for each EVA (EVA 1-3), for each of the two astronauts performing extravehicular activity (EV1 and EV2).

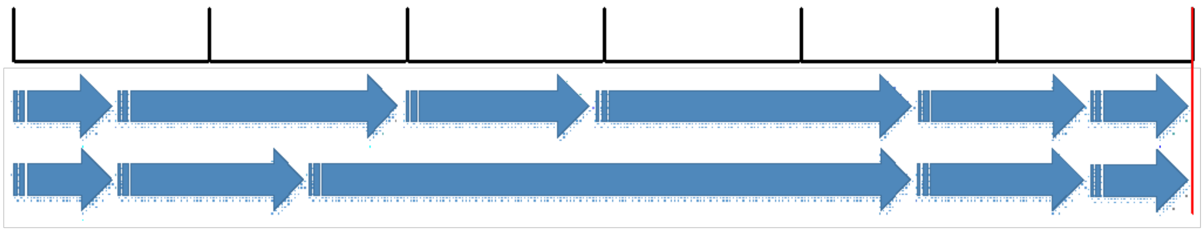
Table 7: Detailed plan for each EVA, by astronaut designation

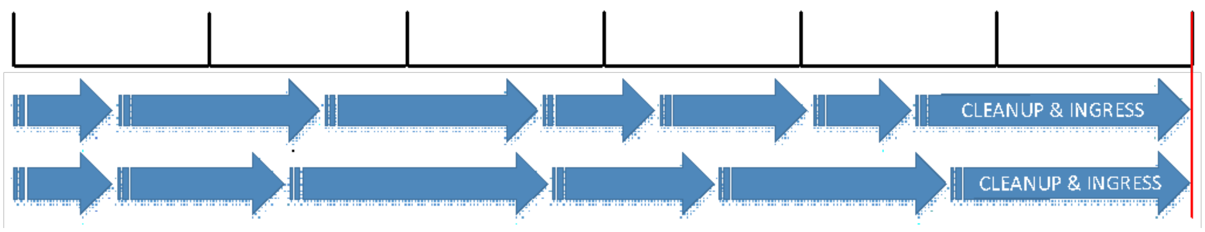
	EV1	Time	EV2	Time
EVA 1	Egress	0.5	Egress	0.5

	Install science drill	1.5	PC Setup	0.5
	Install pyrolysis chamber	1.0	Collect surface samples	4.0
	Collect Surface Samples	2.o	Ingress	0.5
	Ingress	0.5		
	TOTAL	**5.5h**		**5.5h**
EVA 2	Egress	0.5	Egress	0.5
	Collect science drill core	1.5	Set up plasma drill	1.0
	Pack science drill	1.0	Operate plasma drill	3.0
	Collect plasma drill samples	1.75	Pack plasma drill	1.0
	Load pyrolysis chamber	0.75	Ingress	0.5
	Ingress	0.5		
	TOTAL	**6h**		**6h**
EVA 3	Egress	0.5	Egress	0.5
	Collect water from pyrolysis	1.0	collect solids	1.0
	centrifuge it	1.0	set up sintering	1.5
	load steam rocket	0.75	set up cosmic ray experiment	1
	crush solids	0.75	set up thermal experiment	1.5
	deploy SPHERES	0.5	Cleanup & Ingress	0.5
	Cleanup & Ingress	1.5		
	TOTAL	**6h**		**6h**

The following tables give a more detailed look into the EVA timelines, a general EVA summaries, and a detailed procedure list for EVA 1. This shows our team is capable of producing effective and detailed tasks for the crew.







<u>EVA 1 Summary Timeline</u>

PET HR:MIN	EV1	EV2
-- 00:00	<u>EVA 1 ORION EGRESS & BOOM 1 SETUP (00:30)</u>	<u>EVA 1 ORION EGRESS & PC SETUP (00:45)</u> <u>SAFETY TETHER SWAP (00:15)</u>
-- 01:00	 <u>INSTALL SDC (01:30)</u>	<u>COLLECT SURFACE SAMPLES (04:00)</u>
-- 02:00	<u>RETRIEVE PYROLYSIS CHAMBER (00:15)</u> <u>INSTALL PC (00:45)</u>	

-- 03:00	COLLECT SURFACE SAMPLES (02:00)	
-- 04:00		
-- 05:00	EVA 1 CLEANUP AND ORION INGRESS (00:30)	EVA 1 CLEANUP AND ORION INGRESS (00:30)
-- 06:00		

EVA 1 Tools List

EV1	EV2	CREWLOCK
EMU D-ring	EMU D-ring	□ Boom 1
□ 1 - Tether Extender on left	□ 1 - Tether Extender on left	□ SDC
□ 2 – Waist Tethers	□ 2 – Waist Tethers	□ PC
□ 1 – 55-ft Safety Tether	□ 1 – 55-ft Safety Tether	
□ 1 – 85-ft Safety Tether	□ 1 – 85-ft Safety Tether	
□ C/L Bag 1 (3 CiC's)	□ C/L Bag 1 (7 CiC's)	
□ 1 – Surface Sample Bag (SSB)	□ 1 – Surface Sample Bag (SSB)	
MWS	MWS	
□ BRT (left side)	□ BRT (left side)	
□ 8 – Wire Ties, long	□ 16 – Wire Ties, long	

◻ 1 – RET (sm-sm) ◻ 1 – Large trash bag (right side) ◻ 1 – Scooper ◻ 1 – Knife ◻ 1 – SDC BRT Arm ◻ SAFER	◻ 1 – RET (sm-sm) ◻ 1 – Large trash bag (right side) ◻ 1 – Scooper ◻ 1 – Knife ◻ SAFER	

<u>EVA 1 Detailed Timeline</u>

PET HR:MIN	EV1	EV2
-- 00:00	<u>EVA 1 ORION EGRESS & BOOM 1 SETUP (00:30)</u> • Post Depress • Attach 85-ft safety tether to Orion safety HR 1 ◻ Check gate closed ◻ Check hook closed • Egress Orion • Extend translation boom 1 • Install translation boom on Orion and Science Laboratory contact point ◻ Check latch locked	<u>EVA 1 ORION EGRESS & PC SETUP (00:45)</u> • Post Depress • Attach 85-ft safety tether to Orion safety HR 2 ◻ Check gate closed ◻ Check hook closed • Egress Orion • Translate to EVATCB (EVA Toolbox and Capture Bag) • Attach PC to EVATCB HR ◻ Check gate closed ◻ Check hook closed • Unattach PC from MWS (Mini Work Station)
-- 00:30	<u>INSTALL SDC (01:30)</u> • Translate to ARV (Asteroid Retrieval Vehicle) Safety Tether point • Attach 55-ft safety tether to ARV safety tether HR 1 ◻ Check gate closed ◻ Check hook closed • Unattach 85-ft safety tether and attach to ARV secondary safety tether HR 1 ◻ Check gate closed ◻ Check hook closed	
-- 00:45	• Translate to EVATCB (EVA Toolbox and Capture Bag) • Retrieve translation boom 2 from EVATCB • Translate to ARV (Asteroid Retrieval Vehicle) boom contact point 1 • Extend translation boom 2 • Install Boom 2 on ARV contact point 1 ◻ Check latch closed • Translate to end boom 2	<u>SAFETY TETHER SWAP (00:15)</u> • Translate to ARV (Asteroid Retrieval Vehicle) Safety Tether point • Attach 55-ft safety tether to ARV safety tether HR 2 ◻ Check gate closed ◻ Check hook closed • Unattach 85-ft safety tether and attach to ARV secondary safety tether HR 2 ◻ Check gate closed ◻ Check hook closed
-- 01:00		<u>COLLECT SURFACE SAMPLES (04:00)</u>

-- 02:00
 	
-- 02:15
	• BRT to HR 3 at end of boom 2
	
	WARNING
	When using knife, do not touch ends of knife or cut bag towards any crewmember
	
	• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag directly below the end of boom 2
	• Open flap to expose asteroid
	• Ziptie flap to near tether point on bag
	• Use scoop to gather surface rock & materials and place in SSB (Surface Sample Bag)
	• Seal SSB
	• BRT SDC (Science Drilling Campaign) BRT-mount on HR 4 at end of boom 2
	• Install SDC on SDC BRT-mount
	• Position SDC over center of exposed surface
	• Lock SDC position
	  ◻ Check SDC locked
	• Install SDC debris & MMOD cover
	• Check SDC settings
	• Activate SDC
	• Ensure initial operation of SDC
	• Perform glove inspection
	• Release BRT from HR 3
	• Translate to ARV
	
	RETRIEVE PYROLYSIS CHAMBER (00:15)
	• Translate to ARV EVATCB
	• Retrieve PC (Pyrolysis Chamber) from EVATCB and attach to MWS (Mini Work Station)
	  ◻ Check gate closed
	  ◻ Check hook closed
	
	INSTALL PC (00:45)
	• Translate to PC site (TBD)
	• Install PC tether 1 to tether point on asteroid bag (TBD)
	  ◻ Check tether secure
		• Translate to surface site 1 (TBD)
		
		WARNING
		When using knife, do not touch ends of knife or cut bag towards any crewmember
		
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 1
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB (Surface Sample Bag)
		• Seal SSB
		• Use CiC 1 (chisel-in-cup) to gather rock samples
		• Seal CiC 1
		• Place CiC in large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 2 (TBD)
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 2
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 2 (chisel-in-cup) to gather rock samples
		• Seal CiC 2
		• Place CiC 2 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 3 (TBD)
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 3
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag

-- 03:00
 	
-- 04:00
	• Install PC tether 2 to tether point on asteroid bag (TBD)
	  ◻ Check tether secure
	• Install PC tether 3 to tether point on asteroid bag (TBD)
	  ◻ Check tether secure
	• Install PC tether 4 to tether point on asteroid bag (TBD)
	  ◻ Check tether secure
	• Perform glove inspection
	
	<u>COLLECT SURFACE SAMPLES (02:00)</u>
	• Translate to surface site 8 (TBD)
	• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 8
	• Open flap to expose asteroid
	• Ziptie flap to near tether point on bag
	• Use scoop to gather surface rock & materials and place in SSB
	• Seal SSB
	• Use CiC 8 (chisel-in-cup) to gather rock samples
	• Seal CiC 8
	• Place CiC 8 in large trash bag
	• Undo ziptie on flap and place flap in original location
	• Secure flap using ziptie to nearby tether point
	• Translate to surface site 9 (TBD)
	• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 9
	• Open flap to expose asteroid
	• Ziptie flap to near tether point on bag
	• Use scoop to gather surface rock & materials and place in SSB
	• Seal SSB
	• Use CiC 9 (chisel-in-cup) to gather rock samples
	• Seal CiC 9
	• Place CiC 9 large trash bag
	• Undo ziptie on flap and place flap in original location
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 3 (chisel-in-cup) to gather rock samples
		• Seal CiC 3
		• Place CiC 3 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 4 (TBD)
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 4
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 4 (chisel-in-cup) to gather rock samples
		• Seal CiC 4
		• Place CiC 4 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 5 (TBD)
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 5
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 5 (chisel-in-cup) to gather rock samples
		• Seal CiC 5
		• Place CiC 5 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 6 (TBD)

-- 05:00	• Secure flap using ziptie to nearby tether point
	• Translate to surface site 10 (TBD)
	• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 10
	• Open flap to expose asteroid
	• Ziptie flap to near tether point on bag
	• Use scoop to gather surface rock & materials and place in SSB
	• Seal SSB
	• Use CiC 10 (chisel-in-cup) to gather rock samples
	• Seal CiC 10
	• Place CiC 10 large trash bag
	• Undo ziptie on flap and place flap in original location
	• Secure flap using ziptie to nearby tether point
	
	<u>EVA 1 CLEANUP AND ORION INGRESS (00:30)</u>
	• Translate to ARV (Asteroid Retrieval Vehicle) Safety Tether point
	• Attach 85-ft safety tether to waist tether
	  ◻ Check gate closed
	  ◻ Check hook closed
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 6
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 6 (chisel-in-cup) to gather rock samples
		• Seal CiC 6
		• Place CiC 6 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		• Translate to surface site 7 (TBD)
		• Use knife to cut open 3 sides of a 0.5m by 0.5m square on the asteroid retrieval bag at surface site 7
		• Open flap to expose asteroid
		• Ziptie flap to near tether point on bag
		• Use scoop to gather surface rock & materials and place in SSB
		• Seal SSB
		• Use CiC 7 (chisel-in-cup) to gather rock samples
		• Seal CiC 7
		• Place CiC 7 large trash bag
		• Undo ziptie on flap and place flap in original location
		• Secure flap using ziptie to nearby tether point
		
		<u>EVA 1 CLEANUP AND ORION INGRESS (00:30)</u>
		• Translate to Orion crew module
		• Translate to ARV (Asteroid Retrieval Vehicle) Safety Tether point
		• Attach 85-ft safety tether to waist tether
		  ◻ Check gate closed
		  ◻ Check hook closed
		• Unattach 55-ft safety tether from ARV safety tether HR 2 and attach to MWS tether loop
		  ◻ Check gate closed

-- 05:30	• Unattach 55-ft safety tether from ARV safety tether HR 1 and attach to MWS tether loop
	  ◻ Check gate closed
	  ◻ Check hook closed
	• Translate to Orion crew module
	• Ingress Orion crew module
	• Unattach 85-ft safety tether from Orion safety tether HR
	• Pre-Repress
		◻ Check hook closed
		• Translate to Orion crew module
		• Unlock boom 1
		• Collapse boom 1 and attach to MWS
		• Ingress Orion crew module
		• Unattach 85-ft safety tether from Orion safety tether HR
		• Pre-Repress

6.0 Engineering

6.1 Launch

Before considering launch vehicles, the required Δv that will be provided by rocket must be known. Table 8 shows the required Δv for this mission.

Table 8: Required Δv for Launch

	Δv **Required (m/s)**
Launch to LEO	7,800
Losses from Launch	1,600
Trans-Lunar Injection	3,200
Total	12,600

The Δv to LEO corresponds to about a 200 km altitude orbit. The losses from launch due to gravity, drag, and steering are estimated to be between 1,300 m/s to 1,700 m/s and a value of 1,600 m/s was chosen when considering margin and the high thrust provided by next-generation rockets (Wertz 2011). Two trajectories were optimized for TLI. One trajectory with TLI Δv equal to 3.17176 m/s allows for capture by the moon with almost no Δv input by the spacecraft, but takes 4-6 months to arrive at the moon. Another trajectory requires a TLI Δv of 3.1336 m/s and takes 8.6 days to get to the moon, but needs a moderate Δv input to be captured by the moon. A value of 3,200 m/s was taken for this TLI to account for losses such as sloshing, outgassing, center of mass uncertainty, and perturbations.

The masses of the Orion launch system and the supplementary science module, Eureka, are necessary to choose a launch vehicle. The Orion launch system includes a crew module, service module, launch abort system, payload adaptor, jettison fairings, and additional human factors. Table 9 shows a breakdown of the Orion mass and Eureka at launch (NASA/FS-2014-08-004-JSC 2014).

Table 9: Summary of Payloads to be Delivered, Gross Masses

	Payload Breakdown	**Mass (kg)**
Orion	Crew Module	10,387
	Service Module	15,461
	Launch Abort System	7,643
	Payload Adaptor	510

	Jettison Fairings	1,383
	Additional Human Factors	648
	Total	36,000
Eureka	Total	17,200

Current launch vehicles in operation and development are considered for this mission. At this time, the Delta IV Heavy is the most powerful launch vehicle (United Launch Alliance 2015). The Falcon Heavy is set to launch in 2015 and will supersede the Delta IV Heavy for payload capability (SpaceX 2015). The Space Launch System (SLS) Block 1, Block 1B, and Block 2 are in development by NASA for launch in the late 2010s and 2020s and will become the most powerful rockets in operation at their respective launches (NASA 2015). The Long March 9, a Chinese rocket, is planned for its first flight in 2028 and will compete with the SLS Block 2 for launch capability (Lei 2014). A summary of these rockets and their payload capabilities are shown below in Table 10.

Table 10: Summary of Rockets in Operation and Development

Rocket	Phase	First Launch	Mass to LEO (kg)	Mass to GTO (kg)	Mass to TLI (kg)
Delta IV Heavy	Operation	2004	28,370	13,810	<11,500
Falcon Heavy	Development	2015	53,000	21,200	<18,000
SLS Block 1	Development	2018	70,000	~27,500	~21,500
SLS Block 1B	Development	~2021	105,000	~56,000	~47,000
SLS Block 2	Development	2024?	130,000	~63,000	~52,500
Long March 9	Development	2028?	130,000	N/A	N/A

Payload capabilities to LEO for all rockets are provided online (United Launch Alliance 2015, SpaceX 2015, NASA 2015, Lei 2014). The payload capabilities to TLI for the Delta IV Heavy and Falcon Heavy were calculated based on extrapolation of their provided GTO capability using the rocket equation provided in Equation 6.1a. Payload capability for the SLS was based on analysis by stage using the rocket equation and data from Appendix F (Kyle 2015, Gebhardt 2014, Jones). All SLS capabilities assume the Orion is part of the payload, and the launch abort and payload fairings are jettisoned prior to upper stage ignition. The Long March 9 capabilities could not be extrapolated because data for launch mass could not be retrieved.

$$\square_{\square\square\square\square\square\square\square} = \square_{\square\square\square\square\square\square} * \square^{\frac{\Delta \square}{\square_{\square\square} * \square_0}} \qquad (6.1a)$$

Based on Table 10, the Delta IV is not capable of delivering either of the payloads and cannot be used for this mission. Only the SLS Block 1B and SLS Block 2 can deliver Orion. Long March 9 may be able to deliver Orion, but it is a Chinese rocket that will not be available before 2028 and thus will not be considered for this mission. The Orion launch mass is 36,000 kg, which leaves an additional payload mass of about 11,000 kg for SLS Block 1B and about 16,500 kg for SLS Block 2 for a supplementary module. Neither of these masses is sufficient for the 17,200 kg Eureka and thus two launches must be used.

A trade study was conducted based on cost and risk to analyze the best launch option. Risk is the likelihood of completion in time. The non-inflatable habitat/science module (Eureka) would be a custom designed 'can'. The inflatable version would be based off the Bigelow model or be purchased from Bigelow Aerospace. The investigated concepts are listed here and graphed in Figure 12.

- A - Orion & service module in SLS Block 1B and non-inflatable habitat in another SLS Block 1B
- B - Orion & service module in SLS Block 1B and non-inflatable habitat & second upper stage as payload for ballistic TLI (zero Δv to capture) in Falcon Heavy
- C - Orion & service module in SLS Block 1B and inflatable habitat for ballistic TLI in Falcon Heavy
- D - Orion & service module in SLS Block 1B and non-inflatable habitat & SEP (solar electric propulsion) to lunar DRO (moderate Δv capture) in Falcon Heavy
- E - Orion & service module in SLS Block 1B and inflatable habitat & SEP to lunar DRO in Falcon Heavy

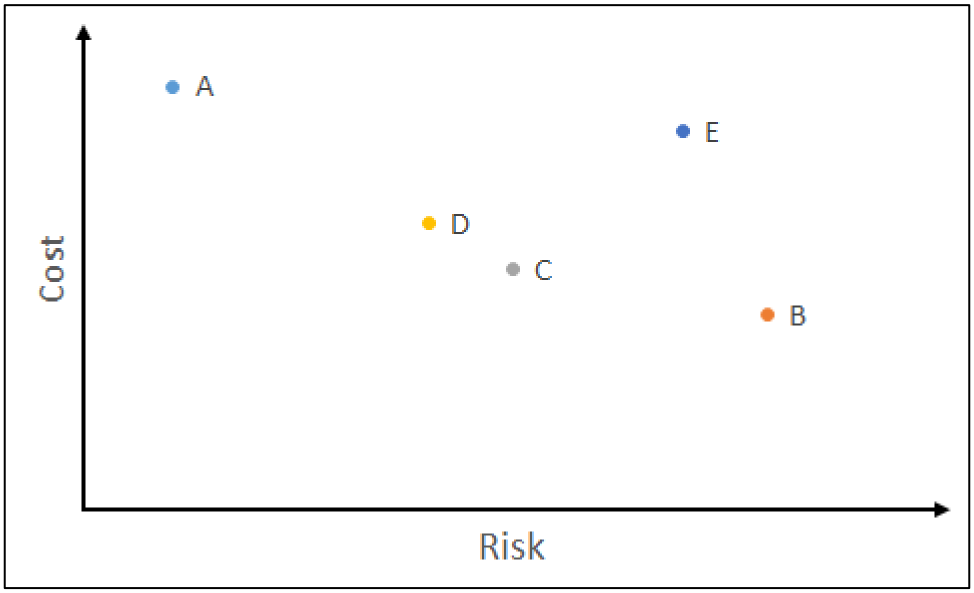


Figure 12: Cost/Risk trade space for the evaluated launch concepts

Option A requires two SLS launches within a year, which may not be possible with NASA's budget. Option B has receives a high risk rating because a large, cryogenic upper stage as payload and a modified payload fairing would be required. Option D is considered more costly than Option C, as a customized non-inflatable habitat as well as another ARM derived SEP module is required. However, can-designs (non-inflatable) are well known and ARM is intentionally developed modularly, making this option less risky. Option E pales in comparison to Option D because of the cost required to develop a custom habitat.

Option C was selected as a good compromise between risk and cost and gives the chance to develop inflatables as a key technology for future exploration missions and large structures in space. Also, this option provides us with enough payload volume and mass, including sufficient margins. If the SLS Block 2 is ready by the launch date and its actual TLI payload capacity is greater then it may be used to launch Orion and inflatable habitat simultaneously, reducing risk.

The volume inside the payload fairings for both the Falcon Heavy and SLS Block 1B was also considered. The payload configurations are shown in Figure 13. Both the Eureka and Orion will fit inside their respective launch systems with additional volume to spare. While there may not be additional mass available for the Falcon Heavy launch, there will be additional mass for the SLS 1B launch. This additional mass and volume can be included to launch cubesats and smallsats at LEO or at the lunar orbit for education or science.

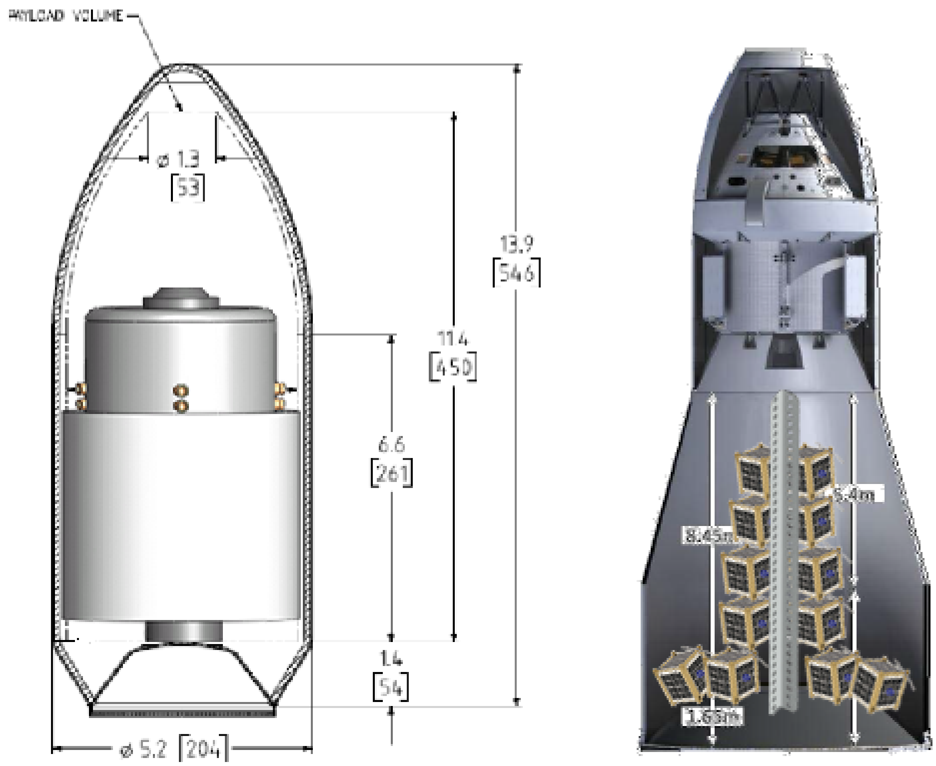


Figure 13: Payload Configurations for Falcon Heavy (left) and SLS Block 1B (right). Cubesats not to scale.

The mission starts August 2024 with a launch of the SpaceX Falcon Heavy from Kennedy Space Center that puts the inflatable Eureka into trans-lunar injection (TLI) for ballistic capture. The Space Launch System (SLS) Block 1B then launches from the Kennedy Space Center on February 20th, 2025 and carries the Orion launch system to TLI on a direct transfer to DRO. BAT charts of the launch sequence are provided below in Figure 13 and Figure 14.

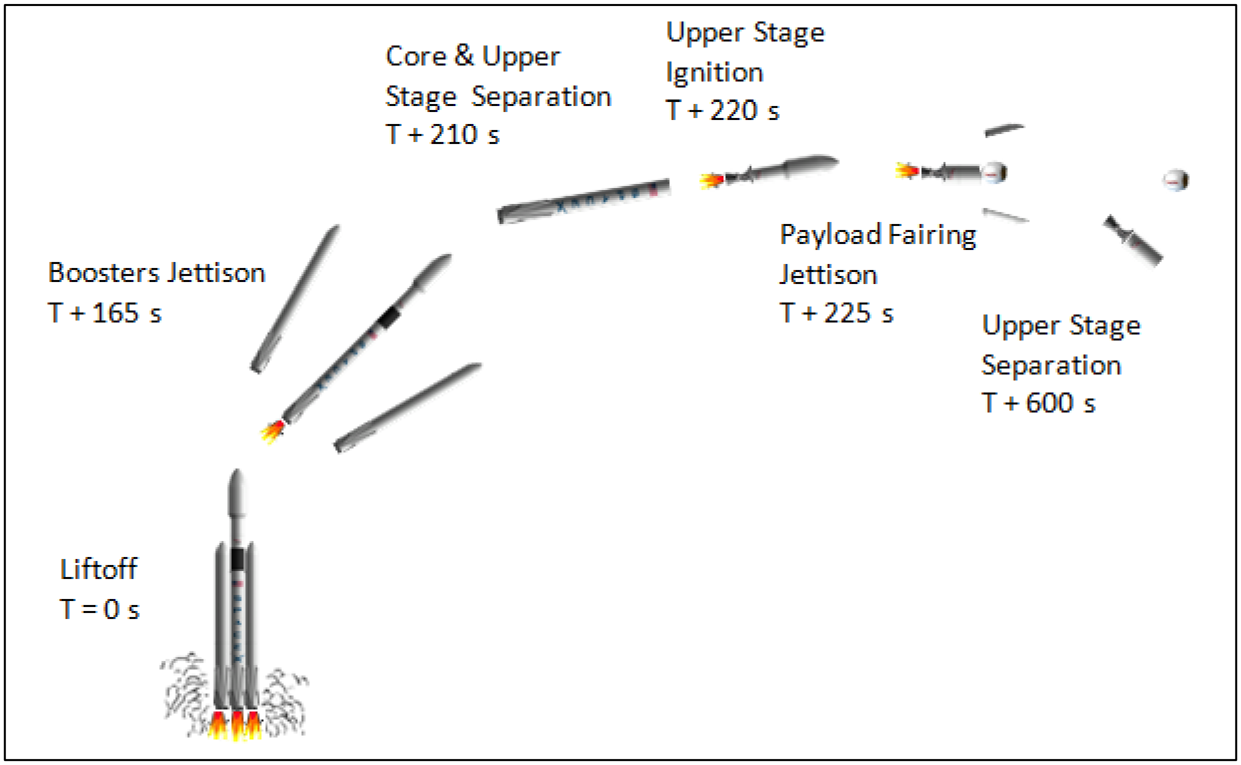


Figure 14: Falcon Heavy Launch Sequence

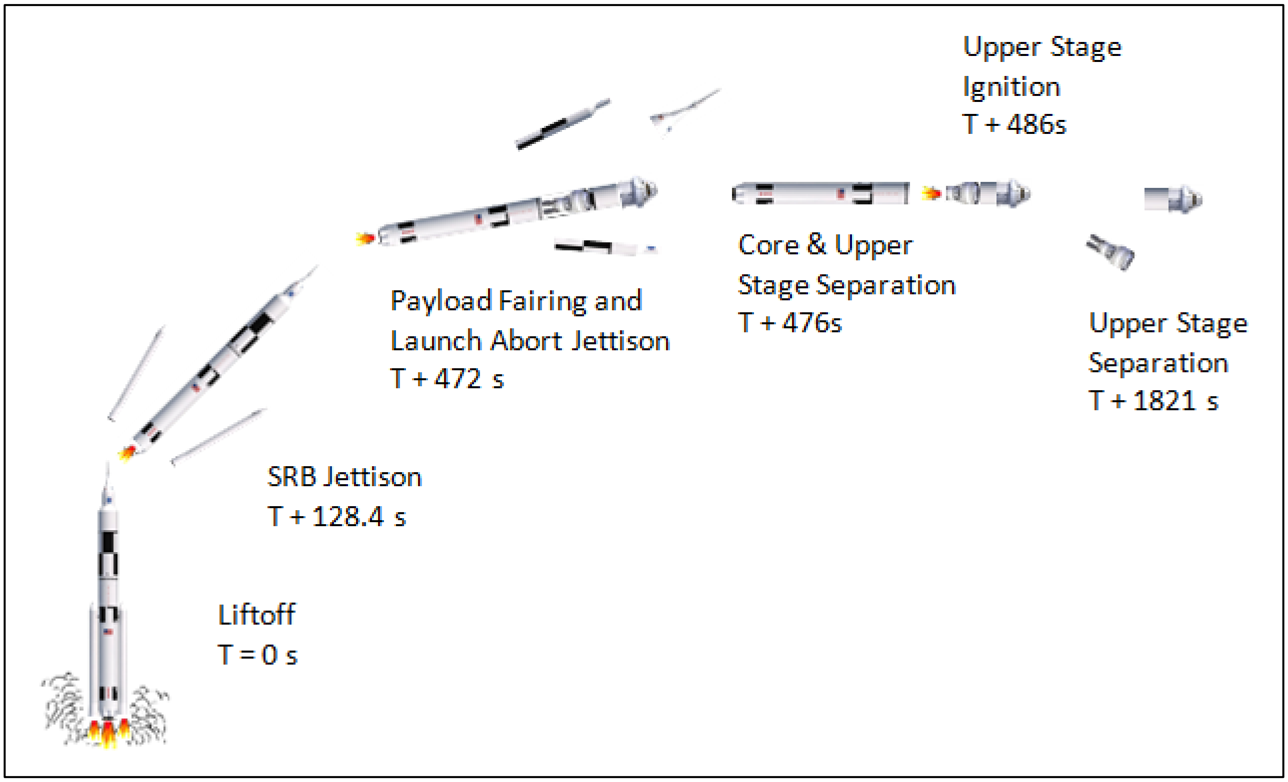


Figure 15: SLS Block 1B Launch Sequence

6.2 Transit

The mission statement defines that the Asteroid Redirect Mission brings an asteroid in a Distant Retrograde Orbit (DRO) around the Moon of mean radius 61,500 km. DROs are defined in the Circular Restricted Three Body Problem (CR3BP).

6.2.1 Theory

The CR3BP arises from considering the gravitational acceleration of two primary masses, m_1 and m_2, on a third smaller body m_3 of negligible mass such that $m_1 + m_2 \gg m_3$. Also $m_1 > m_2$. The primaries are assumed to be in a circular orbit rotating around their barycenter. In the Earth-Moon system, the primaries are the Earth (m_1) and the Moon (m_2) and their orbital period is 27.3217 days. Since the eccentricity of the Moon's orbit is 0.0549, the circular orbit assumption is reasonable Figure 13 shows the geometry of the CR3BP.

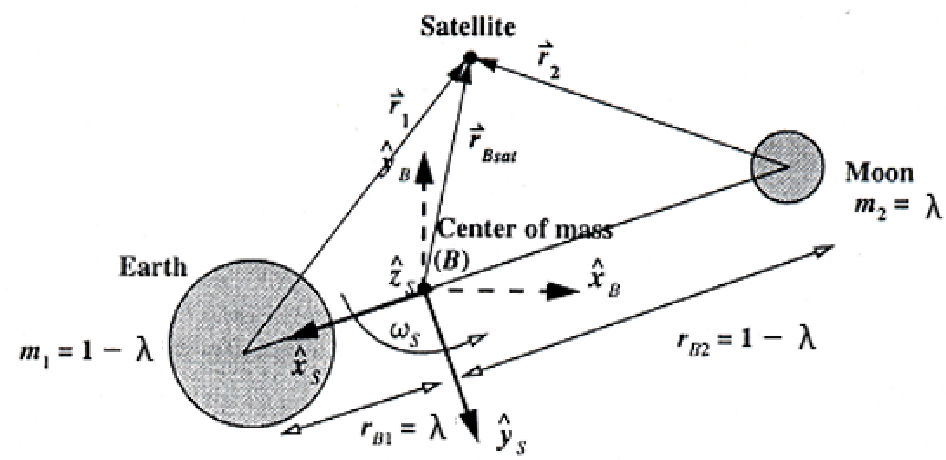


Figure 15: CR3BP general geometry [*CCAR*].

The equations of motion of the CR3BP are:

$$\ddot{x} - 2\dot{y} - x = \frac{-(1-\mu)(x-\mu)}{r_1^3} - \frac{-\mu(x+1-\mu)}{r_2^3} \tag{6.1}$$

$$\ddot{y} + 2\dot{x} - y = \frac{-(1-\mu)y}{r_1^3} - \frac{-\mu y}{r_2^3} \tag{6.2}$$

$$\ddot{z} = \frac{-(1-\mu)z}{r_1^3} - \frac{-\mu z}{r_2^3} \tag{6.3}$$

Where $\vec{r} = [x\ y\ z]^T$ and $\dot{\vec{r}} = [\dot{x}\ \dot{y}\ \dot{z}]^T$ are the position and velocity vectors of the third body respectively, r_1 and r_2 are the distances from m_1 to m_3 and from m_1 to m_2 respectively and are given by the equations:

$$r_1 = \sqrt{(x-\mu)^2 + y^2 + z^2} \tag{6.4}$$

$$r_2 = \sqrt{(x+1-\mu)^2 + y^2 + z^2} \tag{6.5}$$

The equations of motion are written in the coordinate system xyz which generate a rotating reference frame whose origin corresponds to the barycenter of the Earth-Moon system. The x-axis is in the Earth-Moon direction, the z-axis is orthogonal to the Earth-Moon plane, and the y-axis completes the right-handed coordinate frame. Additionally, eq. (6.1), (6.2), (6.3) are written in non-dimensional form with characteristic length L equal to the semimajor axis of the system such that $\mu = \frac{m_2}{m_2+m_1}$. For the Earth-Moon system, $L = 384{,}400\ km$ and $\mu = 0.012151$. Therefore, the non-dimensional locations and masses of the Earth and the Moon are $x_1 = \mu$, $x_2 = \mu - 1$ (pointing in the negative x-direction), $m_1 = 1 - \mu$, $m_2 = \mu$.

The CR3BP has an integral of motion called Jacobi constant which is defined as:

$$C = \frac{1}{2}(\dot{x}^2 + \dot{y}^2 + \dot{z}^2) - \frac{1-\mu}{r_1} - \frac{\mu}{r_2} - \frac{1}{2}(x^2 + y^2) \qquad (6.6)$$

The first term of eq. (6.6), $\frac{1}{2}(\dot{x}^2 + \dot{y}^2 + \dot{z}^2)$, represents the kinetic energy of m_3; the second term, $-\frac{1-\mu}{r_1} - \frac{\mu}{r_2}$, represents the potential energy of m_3; the last term, $-\frac{1}{2}(x^2 + y^2)$, is known as the pseudo potential and is a result of using a rotating reference frame. If the kinetic energy term of eq. (6.6) is set to zero, then letting $C = constant$ gives a contour of zero velocity surfaces. That is, for a given Jacobi constant, only certain areas of the Earth-Moon system are accessible. Figure 14 shows an example of various zero velocity surfaces. Additionally, the location of the Lagrange points are plotted.

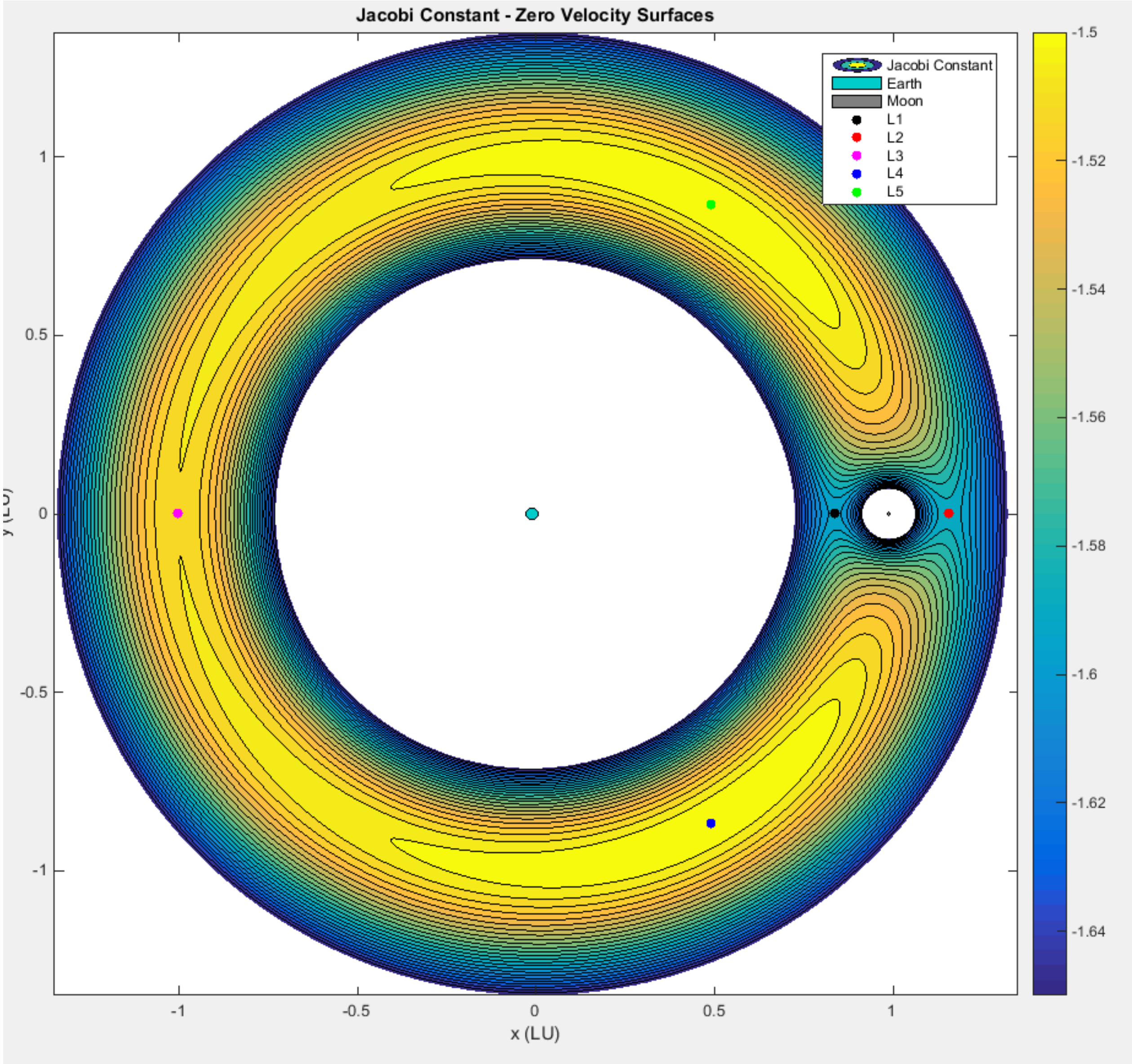


Figure 16: An example of zero velocity surfaces on the Earth-Moon system.

Since an analytic solution of the CR3BP does not exist, various numerical analysis methods were used to study and generate the orbital transfers presented in this report.

The considered DRO is assumed to be a stable periodic orbit in the CR3BP and has a period of approximately 11 days [*Landgraf*]. Figure 15 shows the stability of families of DROs for various perturbation profiles identified by different color lines (Bezrouk). The ranges expressed by the colored lines show the orbits that are stable for 500 years of propagation. It can be seen that the orbit with approximately 11 days cycle is stable, which justifies the assumption for the orbital stability. A DRO of mean radius 61500 km is plotted in Figure 16 in the rotating reference frame of the CR3BP and in Figure 17 in the inertial reference frame.

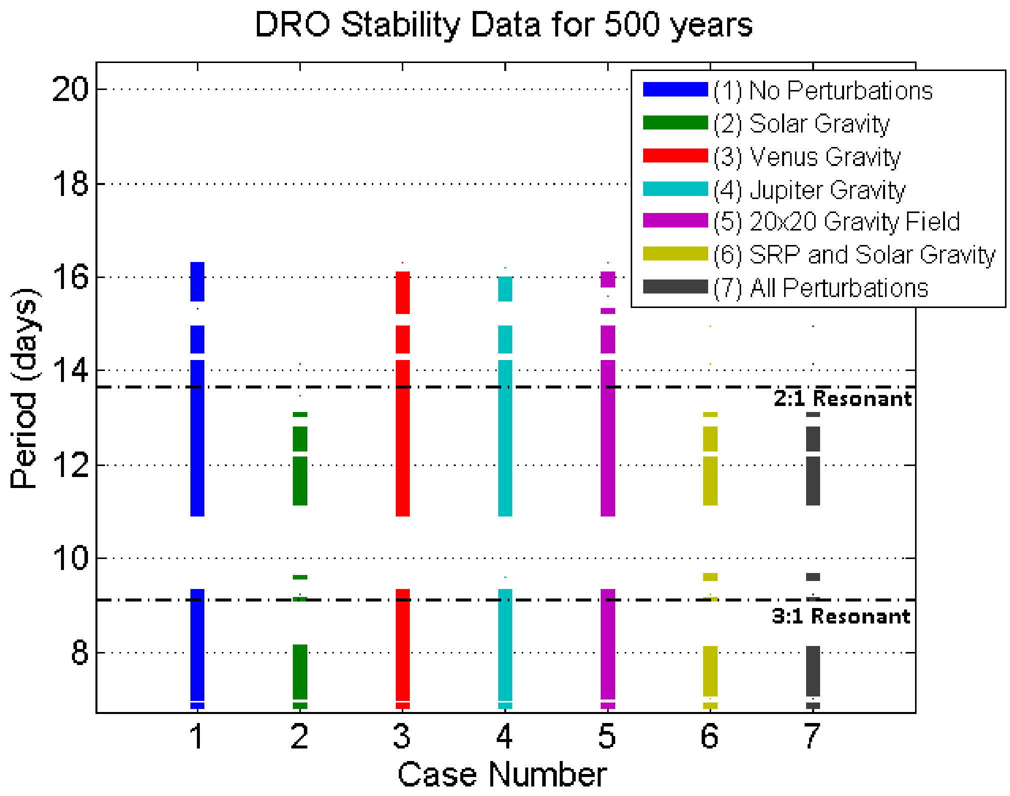


Figure 17: Stability for DROs that have propagated for 500 years [*Bezrouk*].

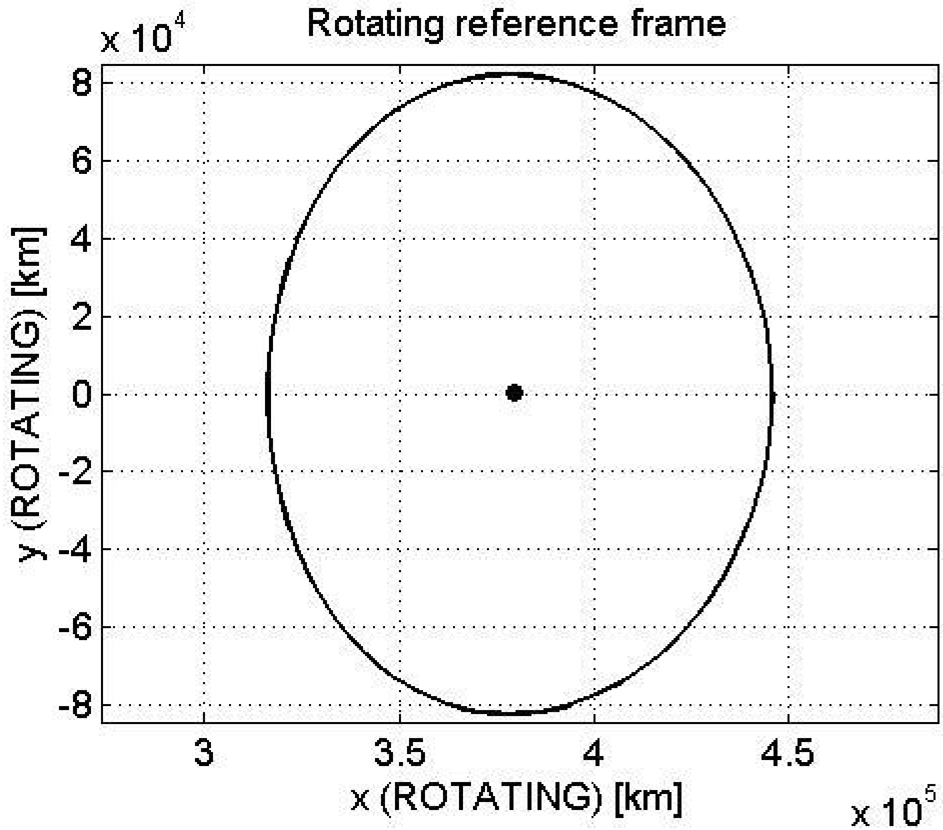


Figure 18: DRO of mean radius 61500 km in the rotating reference frame.

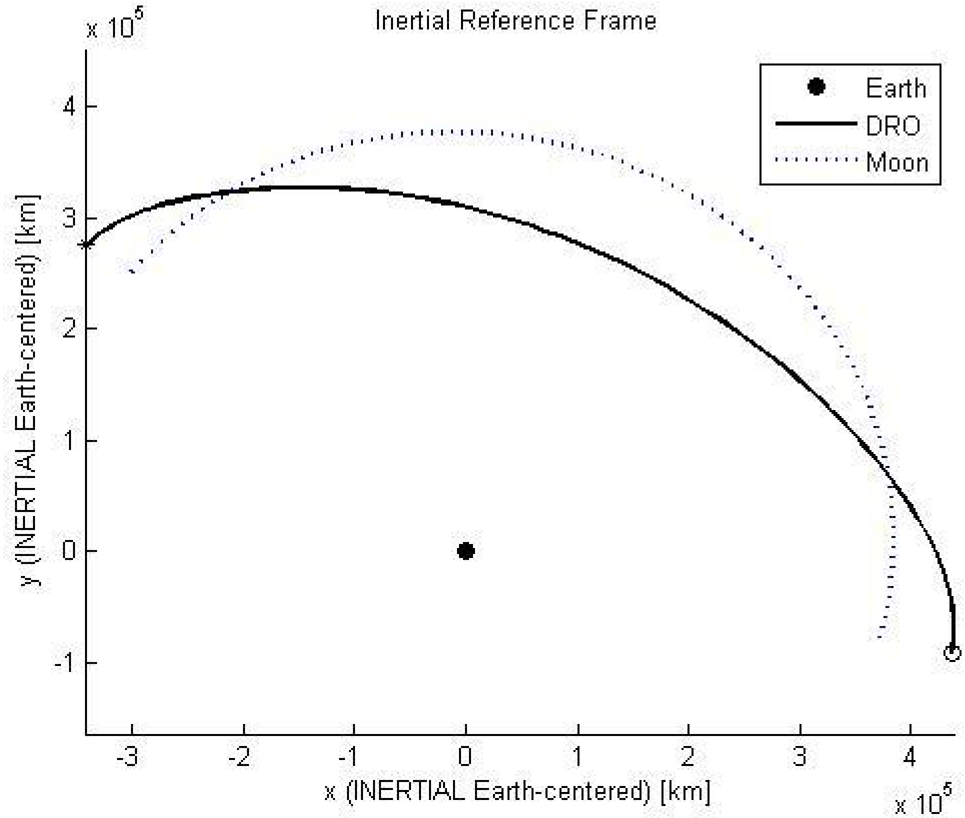


Figure 19: DRO of mean radius 61500 km in the inertial reference frame.

6.2.2 Cargo Mission

As described previously, the baseline requires a robotic cargo pre-deployment mission before the crew one. Unlike a human mission, a robotic mission does not have a motivation for a shorter Time of Flight (TOF); instead, the efficiency (i.e. propellant consumption) would be very important.

Given the above background, the following two options are considered for the cargo mission: the low-energy transfer option and the low-thrust transfer option. The details for each of them are shown below.

Option 1: Low-energy transfer

One attractive option for the cargo pre-deployment is a low-energy ballistic capture transfer. This type of orbit requires almost no ΔV apart from the Earth departure ΔV, resulting in an extremely efficient orbit.

The effectiveness of this approach depends on the stability of the DRO where the redirected asteroid is. The DROs can be categorized into the following four types depending on whether the DROs remain in a bounded distance from the Moon when propagated forward or backward in time: two-way stable, forward stable, backward stable, and two-way unstable.

For the DROs that are forward-stable, the insertion ΔV is essentially zero. This means that the total ΔV is composed of only the Earth departure ΔV, which is supplied by the upper stage of the Falcon Heavy. Even for the stable DROs, which is assumed for this project, the insertion ΔV is much smaller than the direct insertion [*Parker, Roncoli*]. This saving of ΔV results in a large increase of the payload mass, ending up with pre-deployment of a larger mass of science equipment. To perform the required ΔV for orbital adjustment, a bi-propellant propulsion system is on board, which is also used for docking operation as described in Section 6.4.1.

The drawback of this option is its long TOF (e.g. 4-6 months). Thus, the cargo mission is launched at least 4-6 months before the human mission arrives at the asteroid at the DRO. Since this mission is an unmanned one, however, the TOF does not impact the radiation or other health conditions of the crew.

Option 2: Low-thrust transfer

The low-thrust propulsion such as the Solar Electric Propulsion (SEP) is another option for the cargo pre-deployment mission. The concept of SEP makes use of a low thrust engine to follow the trajectory that is not necessarily optimal in terms of Δv or TOF; instead it saves the propellant mass significantly due to its high I_{sp}. The SEP has been used for multiple missions including Hayabusa, which achieved the first sample return mission from the asteroid Itokawa [*JAXA*].

Although the SEP option requires a shorter TOF than the low-energy transfer option, it is still much longer than the direct transfer (~5 months) [*Herman*]. This is not favorable for a crewed mission due to the radiation dose and psychological effects, but it is a valid option for a cargo mission.

One drawback of this option is the complexity of the system. The development of a vehicle with the SEP equipped requires a new vehicle design with the SEP in addition to the crewed vehicle with the chemical propulsion. This requires additional development cost and added risk.

Option selection and Justification

One comparison of these two options is listed in the following table 6.1, and the details of each option is considered in Table 5 and Figure 18 and Figure 19 [*Herman*]. Note that these are only the examples for each option with the same C3 for a fair comparison, but do not show the optimal trajectories for each option. It could be possible to find a trajectory with a shorter TOF or less propellant. The example Herman uses has a payload of 1,800 kg that departs from Earth's orbit. It can be seen that the low-energy transfer does not require any propellant because its insertion ΔV is zero, but requires a long TOF (~6 months). On the other hand, the low-thrust transfer requires a shorter TOF (~5 months), but consumes 54 kg of Xe propellant.

For our purpose, the low-energy option is considered as a preferred option. This is a trade result considering the TOF, propellant consumption, and system complexity. Although the low-energy option has a longer TOF, its effect is not serious for a robotic mission. Having an additional SEP, however, involves an added system complexity and, thus, risk, and therefore is not preferred.

Note that the long TOF of the above options also includes the time spent for rendezvous and docking with the redirected asteroid. Although it is assumed that the spacecraft is inserted into DRO at the same phase as the asteroid, since this docking does not involve human operation, it takes up to two orbital cycle (~22 days) to complete the automatic docking. This ΔV is also performed by the bi-propellant propulsion system on board. The total ΔV for both trajectory adjustment and rendezvous and docking maneuver is assumed to be 15 m/s.

Table 5: Comparison of ballistic (low-energy) and SEP (low-thrust) transfer [*Herman*].

	Ballistic	SEP
Time of flight (days)	187.4	159.3
Final mass (kg)	1800	1746
Launch C_3 (km^2/s^2)	-0.5968666	-0.5968666
Closest lunar approach (km)	34548	7985

Considering the above pros and cons for each mission, the low-thrust transfer is not considered as a baseline for the cargo pre-deployment in this report.

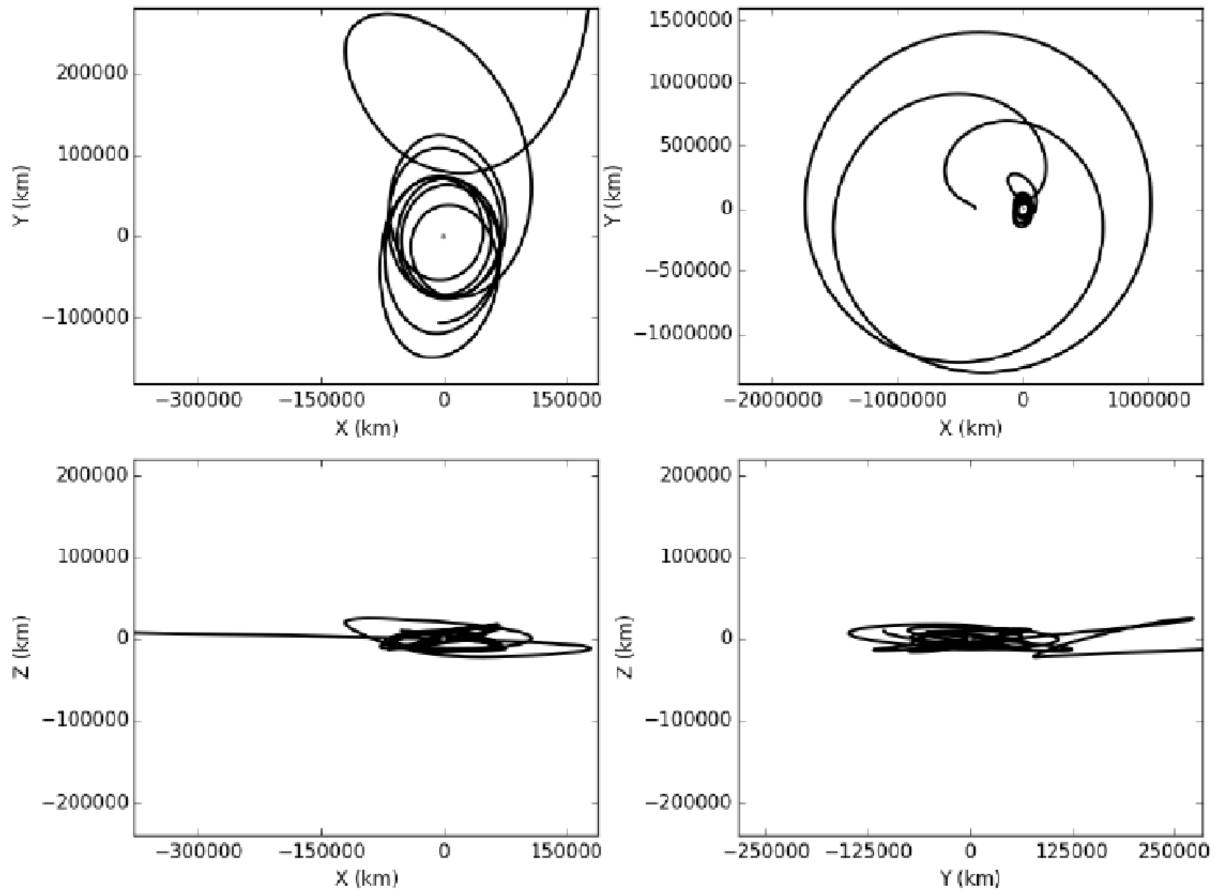


Figure 20: The 6-month low-energy transfer in Earth-Moon rotating reference frame. The XY-plane lies in the Moon's orbital plane, with the X-axis along the Earth-Moon line. The upper left figure shows the same projection as the upper right figure, but zoomed in to illustrate the interactions in the vicinity of the Moon [*Herman*].

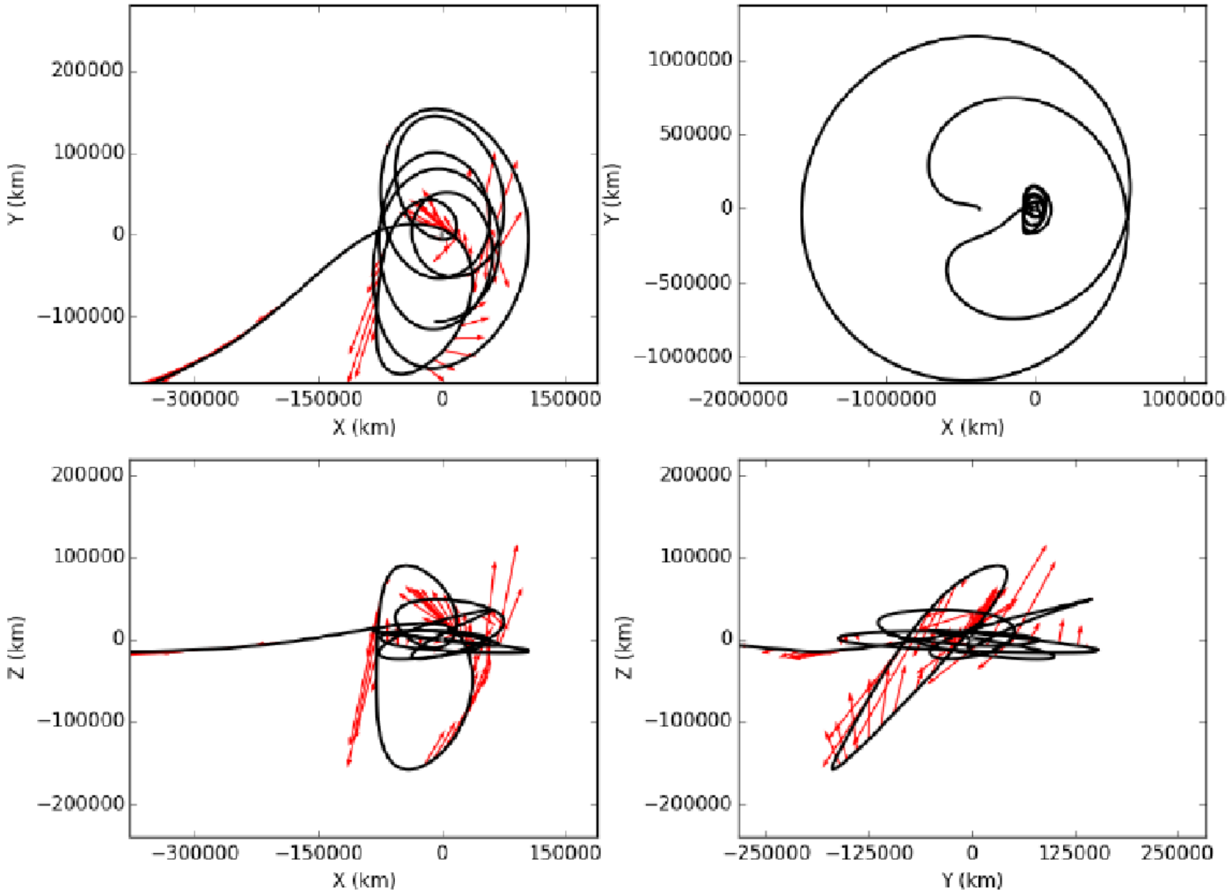


Figure 21: The 5-month low-energy transfer in Earth-Moon rotating reference frame. The XY-plane lies in the Moon's orbital plane, with the X-axis along the Earth-Moon line. The upper left figure shows the same projection as the upper right figure, but zoomed in to illustrate the interactions in the vicinity of the Moon [*Herman*].

6.2.3 Crewed Trajectory

An impulsive transfer trajectory from Earth to the DRO is considered. As in the cargo case, it is assumed that the spacecraft is inserted into DRO at the same phase as the asteroid. Different options have been considered and analyzed for the transfer trajectory to choose the most appropriate transfer.

Option 1: Direct Transfer with Lunar Far-Side Insertion

The simpler way to transfer from LEO to DRO is a direct transfer with injection into DRO in the Lunar Far-Side [*Landgraf*]. This transfer would require two maneuvers, one to depart from LEO and one to insert into DRO [*Capdevila*]. The insertion occurs tangentially along the x axis of the Earth-Moon rotating reference frame.

The Δv and TOF for this kind of transfer have been computed for three conditions:

- Mean Earth-Moon distance (Earth-Moon distance 384,450 km)
- Moon at perigee of its orbit (Earth-Moon distance 357,380 km)
- Moon at apogee of its orbit (Earth-Moon distance 406,020 km)

Table 6: ΔV and time of flight for different Earth-Moon distance.

	ΔV [km/s]	TOF [days]
Moon at mean Earth-Moon distance	0.6317	6.2026
Moon at perigee	0.6014	6.00
Moon at apogee	0.6810	6.3547

Results in Table 6 show that the best timing for the transfer is with the Moon at its perigee. The obtained trajectory is shown in Figure 20 for the Moon at mean distance from the Earth.

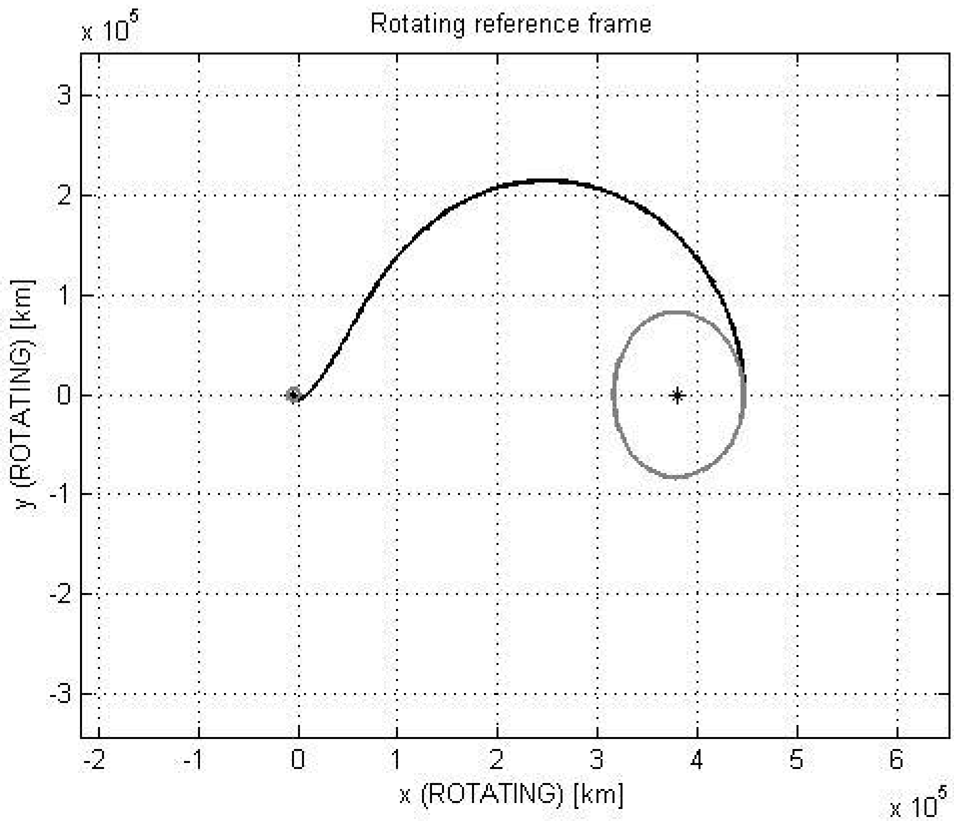


Figure 22: Direct Transfer from Earth to 61,500 km amplitude DRO.

Option 2: Prograde Powered Lunar Gravity Assist

In order to reduce the ΔV for the injection into DRO a powered Moon gravity assist can be considered [*Landgraf*]. Landgraf computed the required ΔV and TOF for this transfer for a 60,000 km amplitude DRO. Two maneuvers are required: a periselenium maneuver of 0.190 km/s and an injection maneuver of 0.318 km/s. The total ΔV would therefore be 0.508 km/s, representing a saving compared with the direct far-side injection. In this case, the TOF would be approximately 8.6 days. The transfer orbit is shown in Figure 21.

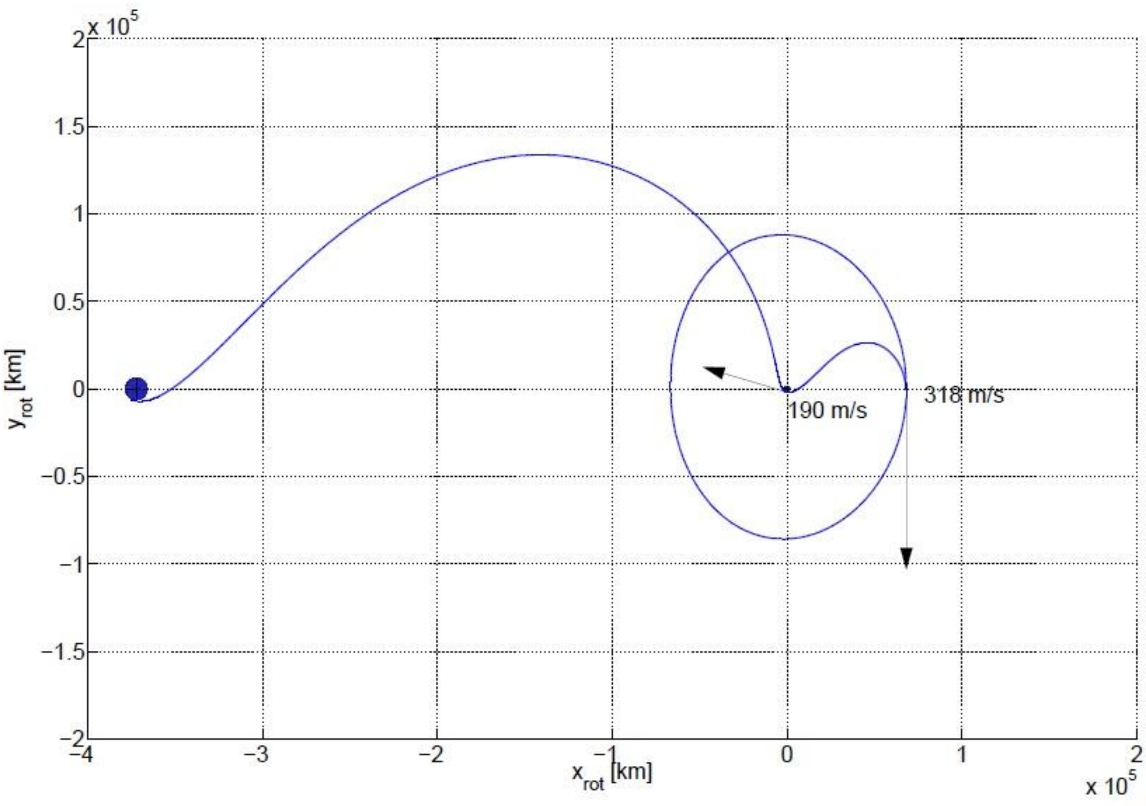


Figure 23: Prograde lunar gravity assist transfer to a 60,000 km DRO amplitude [*Landgraf*].

Option 3: Retrograde Powered Lunar Gravity Assist

The injection maneuvers to the DRO can be realized with a reduced ΔV if a retrograde powered lunar gravity assist is considered. Two ΔV maneuvers are used in this case: 0.201 km/s for the powered gravity assist and 98 km/s for the injection into a DRO of 60,000 km amplitude respectively [*Landgraf*]. The total ΔV is therefore 0.299 km/s. In this case, the TOF is approximately 13.5 days. The transfer orbit is shown in Figure 22.

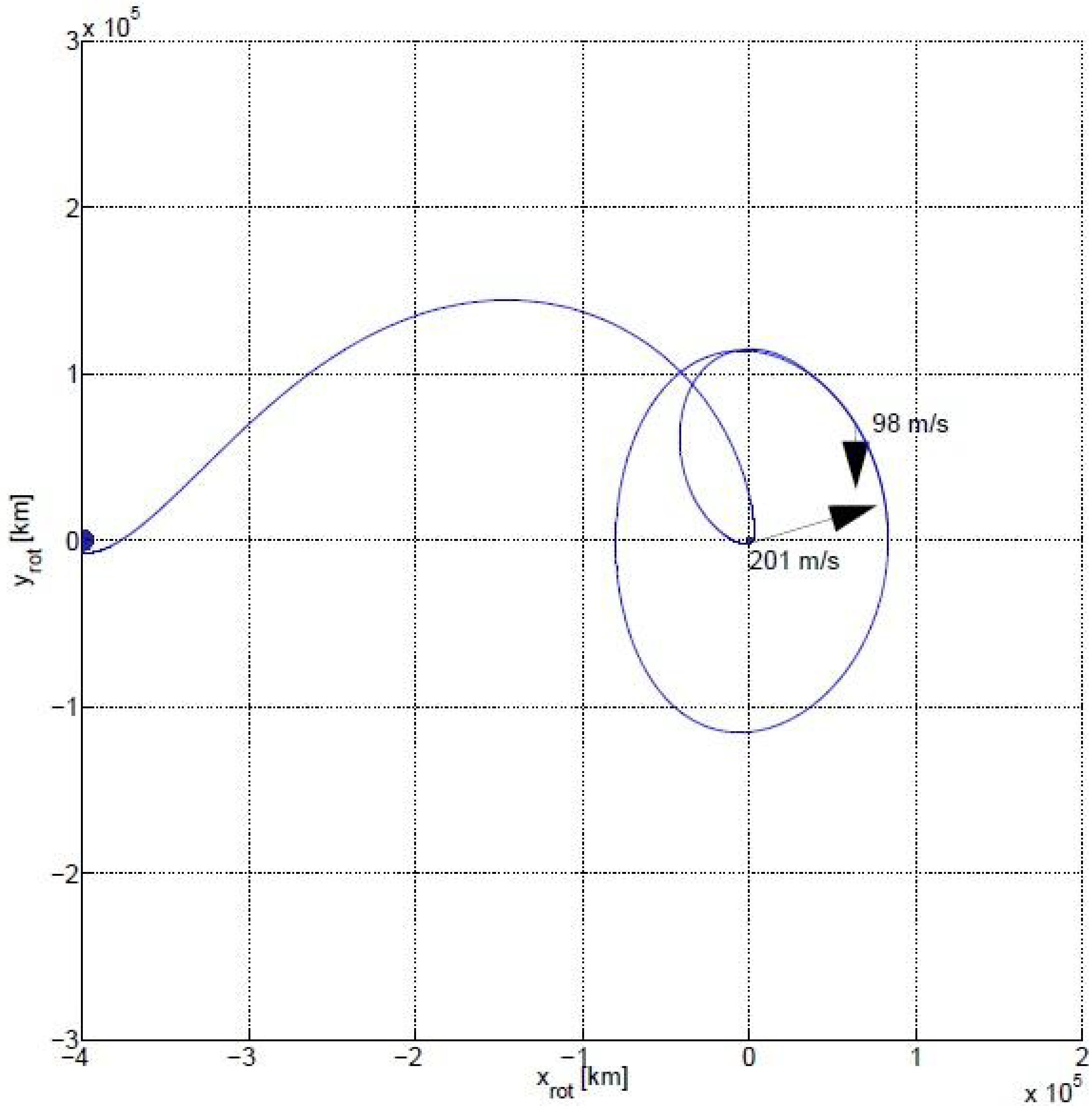


Figure 24: Retrograde lunar gravity assist transfer to a 60,000 km DRO amplitude [*Landgraf*]

Option 4: Direct Transfers to Lunar DRO via L_1 Lyapunov orbit

The last possibility analyzed for the transfer to DRO is an injection from an L_1 Lyapunov orbit. L_1 Lyapunov orbit and DRO overlap in geometry but occupy very different location in the energy space. The size of this energy gap decreases as the orbit extends beyond the vicinity of the Moon [*Capdevila*]. DROs and adjacent L_1 Lyapunov orbit are shown in Figure 23.

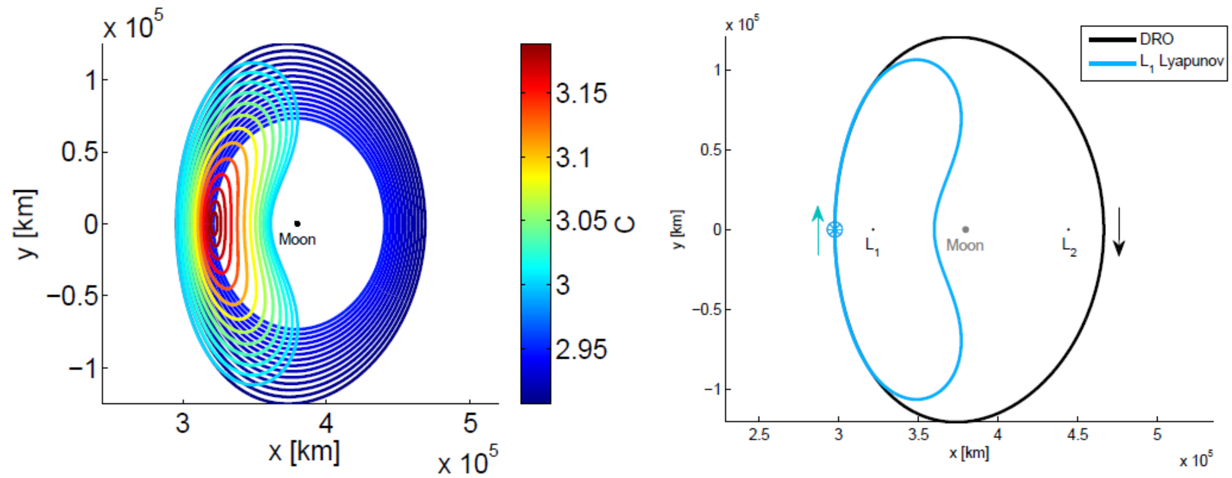


Figure 25: DROs and adjacent L1 Lyapunov orbits [*Capdevila*].

The transfer in this case would be realized with an insertion maneuver into a L_1 Lyapunov orbit where $y = 0$ and $\dot{y} < 0$ in the rotating reference frame. The spacecraft then follows the Lyapunov path for half a period and when the L_1 Lyapunov and the DRO coincide in position, a final maneuver is implemented to insert into the DRO. This kind of transfers are characterized by a total Δv ranging from 1.2023 km/s to 0.4595 km/s, decreasing monotonically as the location of the DRO injection approaches the Moon. The TOF varies from 10 to 15 days, increasing with proximity to the Moon [*Capdevila*]. Due to the considerable transfer time this transfer option have not been considered for this mission.

Chosen Trajectory for the crew mission

With the exclusion of the L1 Lyapunov transfer option, the Δv and TOF for the considered transfers are summarized in Table 7.

Table 7: Summary of Δv and time of flight for the trajectories taken into account

	Δv [km/s]	ToF [days]
Direct Transfer	0.6014	6.0
Prograde Powered Lunar Gravity Assist	0.508	8.6
Retrograde Powered Lunar Gravity Assist	0.299	13.5

The chosen trajectory for both the outbound and the inbound flight is the prograde powered lunar gravity assist, which has been chosen as the best compromise between Δv and transfer time. The direct orbit is considered as a contingency option for a fast return to Earth.

6.2.4 Launch Date Selection

According to the problem statement, the launch dates for the cargo and crewed missions must take place between01/01/2024 and 01/01/2028. In order to determine a preferred launch date, the orbital inclination and RAAN of the Moon's orbit and Sun eclipses were considered. Additionally, the radiation environment was also taken into account as outlined in Section 6.4.4 (ECLSS Subsystem).

Inclination Consideration

The DRO considered for this mission resides in the plane defined by the Earth-Moon. The plane of the orbit of the Moon approximately forms a 5° angle with the plane of the ecliptic, which is tilted by 23.4° with respect to Earth's equatorial plane. The inclination of the plane of the Moon with respect to the Earth equatorial plane varies with a period of approximately 18.6 years, going from 18.28° to 28.58° [*Grayzeck*].

The top plot of Figure 24 shows how the inclination of the Moon's orbit changes during a period of 4 years starting from 01/01/2024. The red line represents the latitude of the launch station, Kennedy Space Center (KSC). In order to avoid an expensive plane change maneuver, the launch date should be selected when the inclination of the orbit of the Moon is close to the latitude of KSC. In the bottom part of Figure 24 the ΔV required to perform the inclination change to match the orbit Moon inclination is shown. It is possible to see that in the first two years of the considered interval of time, the ΔV required to perform the inclination change is considerably low. On the other hand, during the latter two years, the ΔV required to perform a plane change is relatively high. Therefore, the mission's initial date should not be after 01/01/2026 to avoid inclination change Δv.

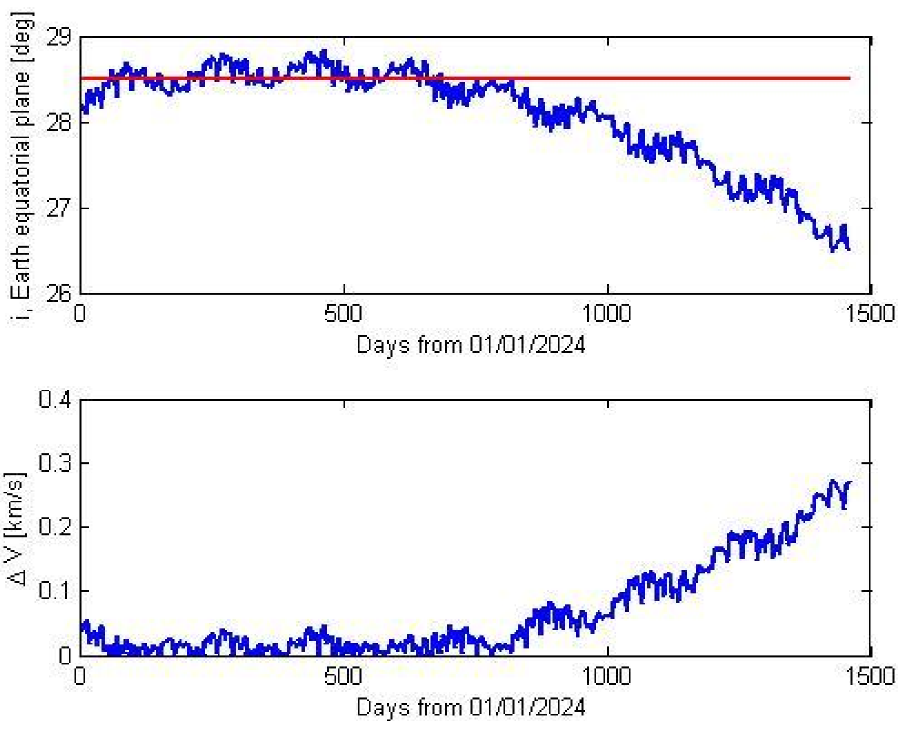


Figure 26: Moon orbit inclination and corresponding Δv for plane inclination change for the considered mission time window.

Right Ascension of the Ascending Node

In Figure 25 the Right Ascension of the Ascending Node (RAAN) of the Moon orbit is shown for the considered mission interval. In order to match the RAAN of the transfer orbit, the launch should be done at an appropriate time of the day to inject the spacecraft into the required orbital plane, i.e. the orbital plane of the Earth and the Moon.

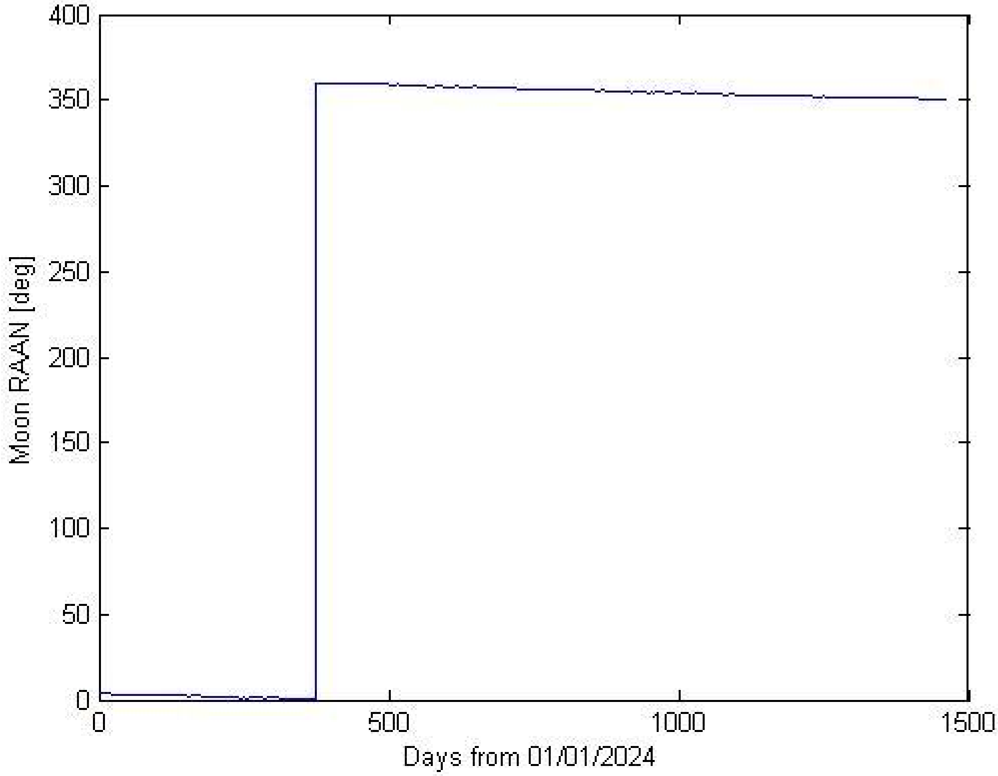


Figure 27: Moon RAAN for the considered mission time window.

Moon position

In order to further reduce the ΔV and the TOF to DRO around the Moon, the DRO insertion should take place when the Moon is at the perigee. Based on Figure 26, perigee occurs on 03/01/2025 and 03/31/2025 during the month of March 2025. Thus, since the selected crew orbital transfer is 8.6 days, the launch opportunities occur on 02/20/2025 and 03/22/2025. The former date is used as the nominal launch date while the latter is kept as a backup date.

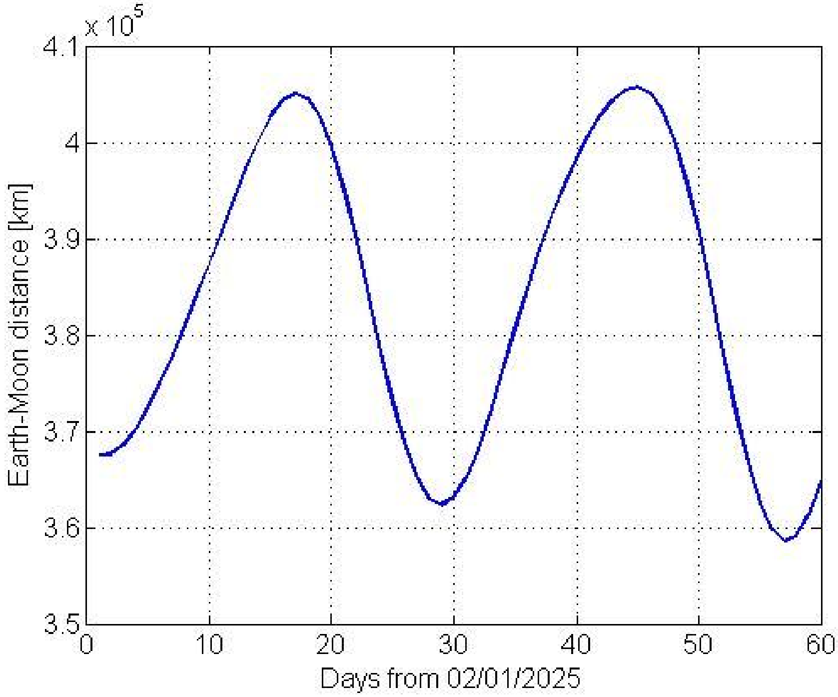


Figure 28: Earth-Moon distance as a function of time.

Eclipse consideration

Time of Sun eclipses must be considered in order to design appropriate power and thermal subsystems, as mentioned in the Section 6.4.5 (Power) and 6.4.8 (Thermal subsystem). As one can see from Figure 27, Sun eclipse durations are of the order of 5 hours each, with longest eclipse times reaching up to 7 hours. Additionally, Earth eclipses were estimated to have an average duration of approximately 30 minutes per orbital period.

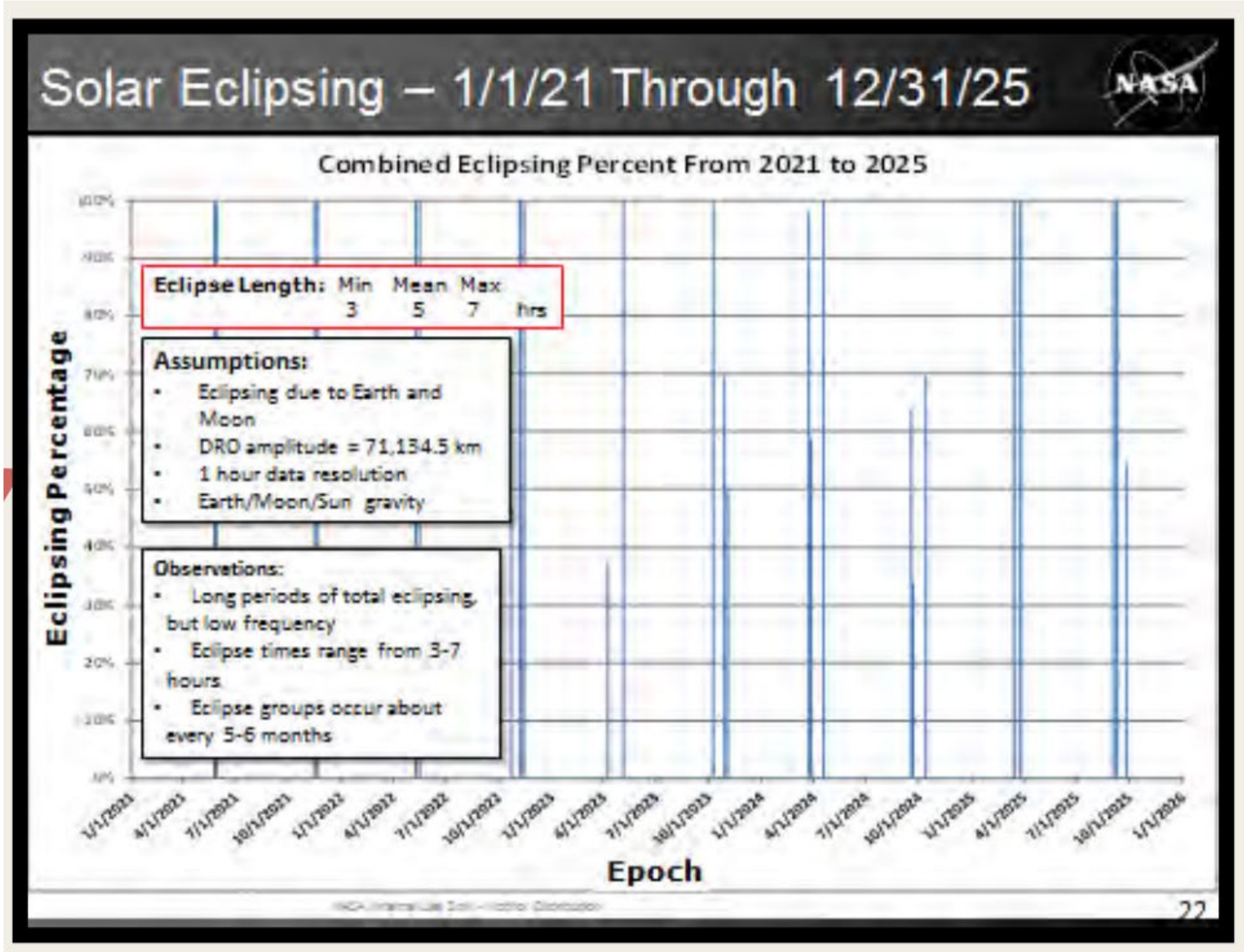


Figure 29: Solar Eclipsing for a DRO amplitude of approximately 71,100 km. Eclipse times for the given 61,500 km altitude DRO are estimated to be of similar lengths.

6.2.5 Reentry

Reentry to Earth has been a challenge for various human space missions. It is very important to ensure that the Orion vehicle can survive the reentry speed from the trajectory analysis as well as navigating itself to the predesigned landing location. Figure 28 shows the plot of the velocity of Orion as a function distance from the surface of the Earth. The result shows that the reentry speed is 11 km/s.

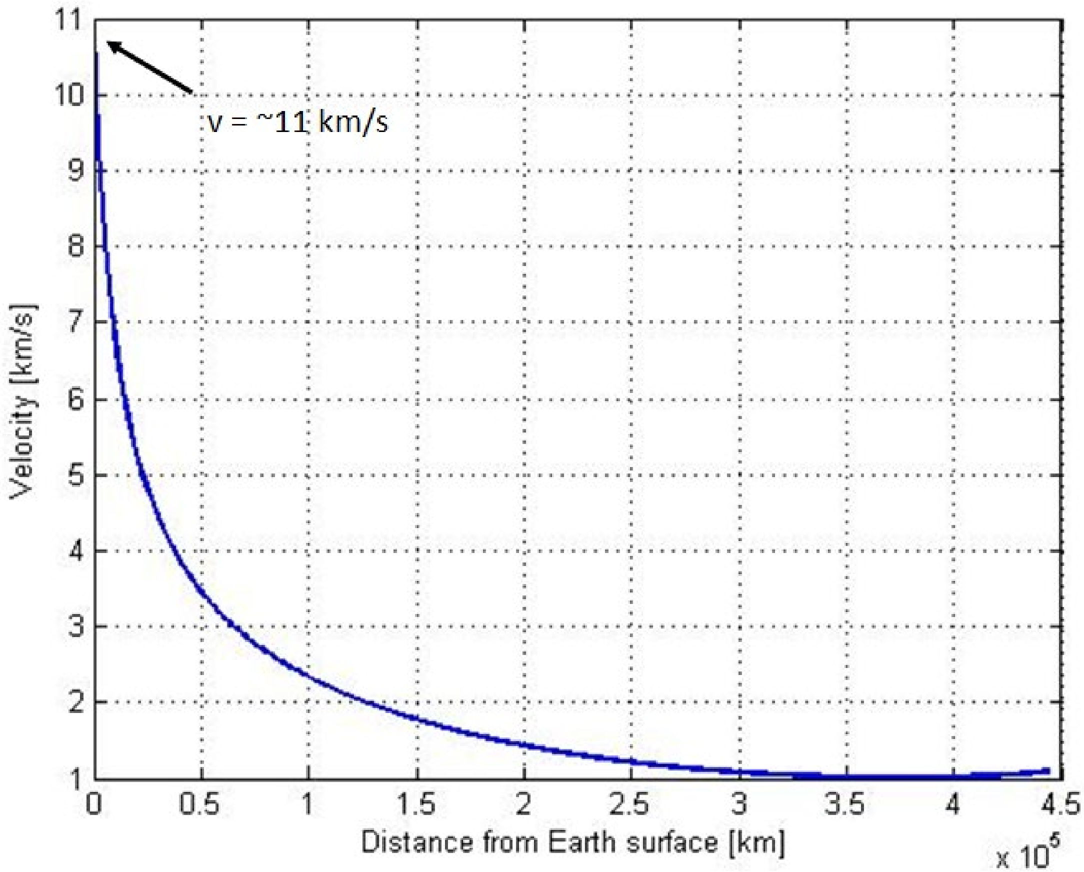


Figure 30: Velocity vs. Distance from Earth Surface for the baseline direct entry option.

The design of the Orion vehicle allows the reentry speed at 11 km/s [*Howell]*. The recent test, Exploration Flight Test 1, has demonstrated its performance of reentry at 8.9 km/s, and the planned Exploration Mission 1 will demonstrate a reentry velocity of 11 km/s in 2018 [*Hill*]. With this specification, the reentry portion of the proposed trajectory can be performed by the Orion vehicle.

The following describes the overview of the reentry system of the Orion vehicle based on the results of the Exploration Flight Test [*Barth*]. The reentry navigation mostly replies on the GPS except the duration of the ionization-induced blackout, during which an IMU system is used. The Orion entry guidance algorithm is based on the Apollo one. During descent, eleven parachutes are deployed and jettisoned. At a predesigned altitude, the Orion attitude controller initiates touchdown heading control. The touchdown is detected using the sensed IMU data. At that point, the RCS commands are all ceased, followed by the deployment of the airbags and the start of a timer to cut the Main parachutes. The Exploration Flight Test has demonstrated the whole sequence of the reentry phase. Figure 30 shows a computer generated image of Orion upon reentry.

Figure 31: Orion with its base heat shield facing Earth as it encounters the first areas of the discernable atmosphere around the planet and the temperature around the spacecraft begins to build up (Siceloff).

6.4 Spacecraft

Both stages of the mission required a different spacecraft, although re-usability of design is encouraged. The first, unmanned, spacecraft requires specific development and fabrication, although the spacecraft heavily uses technology developed for the European Orion transfer vehicle (Multi-Purpose Crew Vehicle European Service Module (MPCV-ESM), a.k.a. Orion Service Module (SM)). Whenever possible, the same subsystems developed for the Orion SM are used in the first spacecraft. This vehicle integrates these capabilities on an inflatable structure, which forms the principal laboratory module.

The second spacecraft combines the manned Orion capsule and the European MPCV-ESM, already under design for support of the Orion and requiring no specific development in the scope of this mission.

The strategy is heavily based on an incremental de-risking approach towards Mars exploration with the Orion capsule. The launch and transfer of the first vehicle adds maturity in terms of power management, guidance, navigation and control, and thermal management. The first vehicle also provides additional maturity for the inflatable structure, valuable for the establishment of Mars colonies. In summary, several advantages to this approach are identified:

· **Reduced cost**: The MPCV-ESM module is under development by the European Space Agency and considered as a European contribution to the international exploration program. Thus the cost of this spacecraft is not considered within this project. Developments specific to the first spacecraft are included.

· **Heritage**: Five Automated Transfer Vehicles (ATV), predecessor to the MPCV-ESM, successfully brought cargo to the International Space Station (ISS), increasingly the TRL of the system

· **International cooperation**: Including the MPCV-ESM and its derived technology in the project strategy will be beneficial to encourage international collaboration in space.

· **De-risking**: Demonstrating the use of the MPCV-ESM and the inflatable module reduces the risk of a later manned mission to Mars.

Table XXX shows an overview of both spacecraft used in the OA-Sys mission. Characteristics of the MPCV-ESM are listed in Table XXX.

Table 14: Characteristics of spacecraft used for the mission

1. Unmanned transfer vehicle (CAMEL)	2. Manned Orion capsule and transfer vehicle (CARAVAN)
Description: "Simplified" service module based on European ATV vehicle heritage, without principal propulsion module Inflatable structure Central avionics unit	Description: Orion capsule MPCV-ESM transfer vehicle

Table 15: Principal characteristics of both spacecraft

Item	EUREKA	Orion + MPCV-ESM
Mass and structure		
Cargo space		0.57 m^3 – 382 kg
Structural mass (kg)	???	4250
Height (m)	4.0 (laboratory) + 2.0 (service module)	4.0 (service module) + 3.3 (Orion)
Diameter (m)	4.5	4.1 (5.2 with external equipment).
Thermal control	Active thermal control using fluid loop and radiators	Active thermal control using fluid loop and radiators
Water storage (kg)	-	280
Oxygen storage (kg)	-	66 (at 275 bar)
Nitrogen storage (kg)	-	33 (at 275 bar)
Extra gas storage (kg)	-	33 (at 275 bar)
Electrical Power System		
Power generated (W)	4 250	11 100
Power consumed (W)		

6.4.1 GNC

In this section the objectives and characteristics of the Guidance, Navigation and Control subsystem are described. Due to the rich legacy that exists in the past spacecraft experience, a standard GNC system serves the objective of this mission. For both cargo and crew missions, the standard Orion system is considered. The following describes the requirement for the GNC subsystem and its design.

The objective of the GNC includes determining and controlling the position and attitude of the spacecraft. In particular there are three large main requirements for the GNC system.

First, the GNC system shall "clean up" the motion after the rocket launch. This maneuver is necessary to inject the spacecraft exactly into the designated orbit and to correct possible error of the launcher. As mentioned in the Transit section, the expected delta-V is about 15 m/s for the Falcon Heavy launch of the cargo, and 20 m/s for the Space Launch System launch of the crew. For both missions, the onboard bi-propellant propulsion system is used to meet that requirement.

During the cruise phase, the GNC shall point the solar panel of the spacecraft toward the Sun when it is not in eclipse. This is important because the spacecraft power system mainly relies on the solar panel. This is a standard operation that the Orion spacecraft is designed for; therefore no large design change is necessary.

Finally, the GNC shall also perform the docking operation when it arrives at the DRO. For the cargo mission, the spacecraft approaches and docks with the asteroid fully autonomously. This process takes two orbital cycle (~22 days). For the crew mission, the docking operation can be performed by the crew. This operation takes about 6 hours, as expected based on the experiences on the International Space Station. Each one of these operations requires about 15 m/s of delta-V, and both of them are performed using the onboard bi-propellant propulsion system.

To meet the above requirements, the following components are considered:

- NASA Docking System
- IMU (Inertial Measurement Unit), which contain both accelerometers and gyroscopes
- Sun sensor
- Star tracker
- GPS receivers
- Vision navigations sensor (LIDAR) for relative navigation
- Docking camera
- HD situational awareness camera
- Vision processing units

For the crew mission, all of these components are necessary. They are incorporated in the Orion spacecraft design, therefore no detailed design is considered here. Additionally, Deep Space Network navigation updates are available when communication with ground is available.

On the other hand, for the cargo mission, only a subset of the above component list needs to be considered. The following Table X shows the necessary component for the cargo mission as well as their mass budget.

Table X. Component list for the cargo mission

Table 16: Component list for the cargo mission

	Amount [-]	Element	Margin	Total	Total + Margin
NASA Docking System	2	320	5%	640	672
IMU (Inertial Measurement Unit)	3	4.44	5%	13.32	13.986
Star Tracker	2	5.83	5%	11.66	12.243
Sun Sensor	1	0.21	5%	0.21	0.2205
Vision Navigation Sensor	2	13	5%	26	27.3
Docking Camera	2	0.77	5%	1.54	1.617
Processing Unit	1	13	5%	13	13.65
Sums				898.73	934.0165

The configuration of the GNC components is shown in Figure XXX

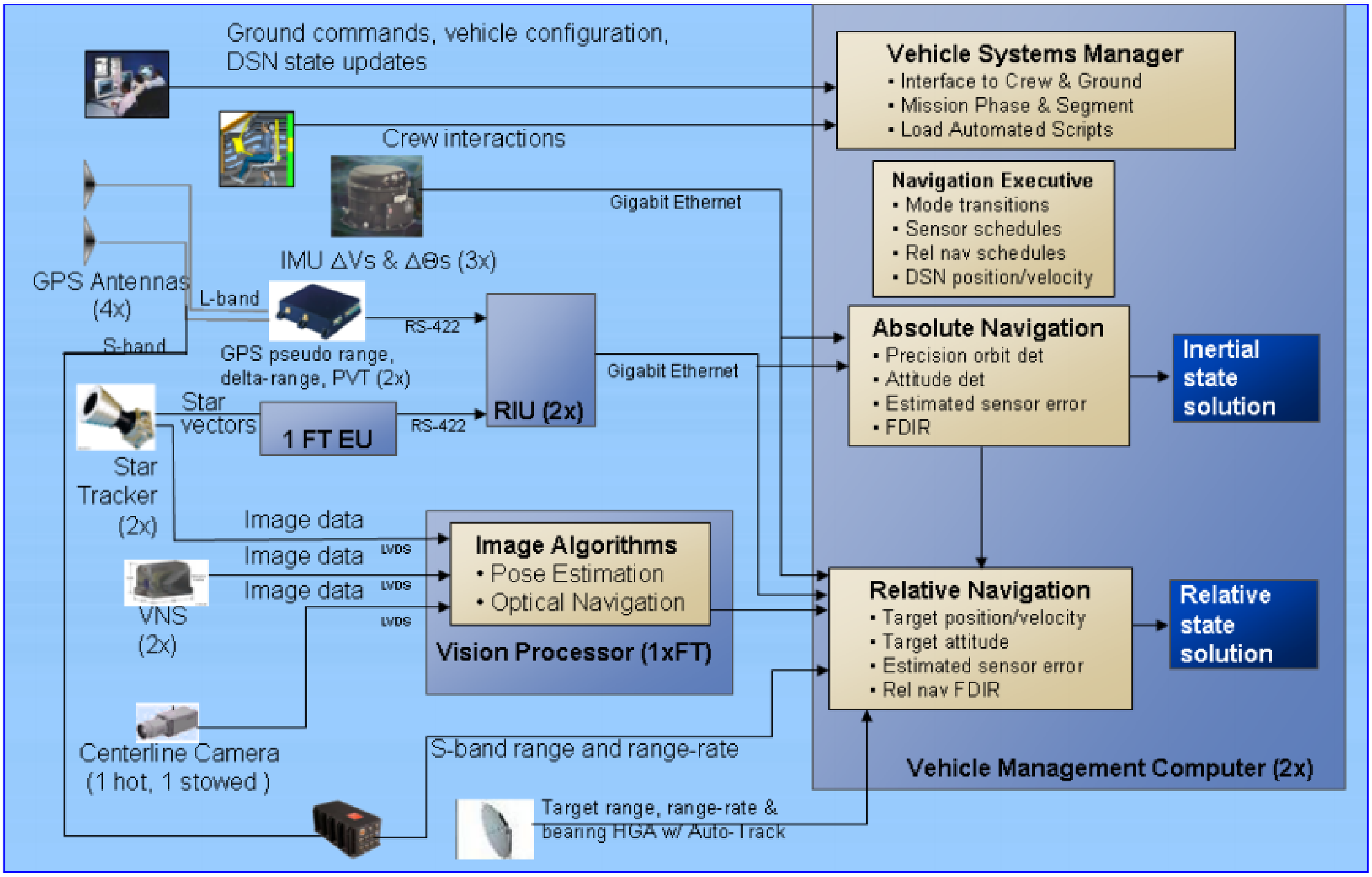


Figure 32: Orion GNC subsystem

6.4.2 Communications, C&DH

For the command and data handling system, rough sizing rules based on system complexity were used. Since the mission contains many different science payloads, HD cameras along with housekeeping data, the data rates cover the whole spectrum. The system complexity is high. The C&DH system is estimated (‘SMAD’, Wertz) to weight 10 kg, measure 15 dm^3 and consume 25 W.

For our mission, the communications system is an important aspect that requires the following links:

1. Telemetry link

   This is the basic and most important link between the exploration vehicles and the ground station. The telemetry data flow involves sensors, instruments, data sampling, and science data amassed from the modules are transmitted via this link, using parsed text messages which are not very data-heavy. However, since we have astronauts on board, it is important to have a link that supports video uplink (for communicating with the ground station, and utilizing the internet) and also downlink (for sending camera image and video data). We use HD quality videos which result in a data rate of 12 Mbps.

2. Crew voice communication link

   This link is used for voice intercommunications between the modules and EVAs. It requires low data-rate, is of close range and with minimal losses.

Architecture

From a preliminary analysis from our link budget (cf. appendix for complete details) it suffices to use direct-to-earth X-band radio link.

Link budget assumptions:

Data rate:	12 Mbps
X-band frequencies:	~8 GHz
Total losses:	12 dB
Thermal noise:	neglected
Ground station characteristics:	15 m dish, 200 W
Various losses outside line losses:	12 dB

Using a S/C dish of 10 cm diameter and a power of 20 W, the link closes with margins of 20 dB on the up- and 11 dB downlink.

Assuming a 30 % efficiency, the Com subsystem requires 20/0.3= 70 W of power. A typical X-Band comm system weights 50 kg (SMAT, Wertz).

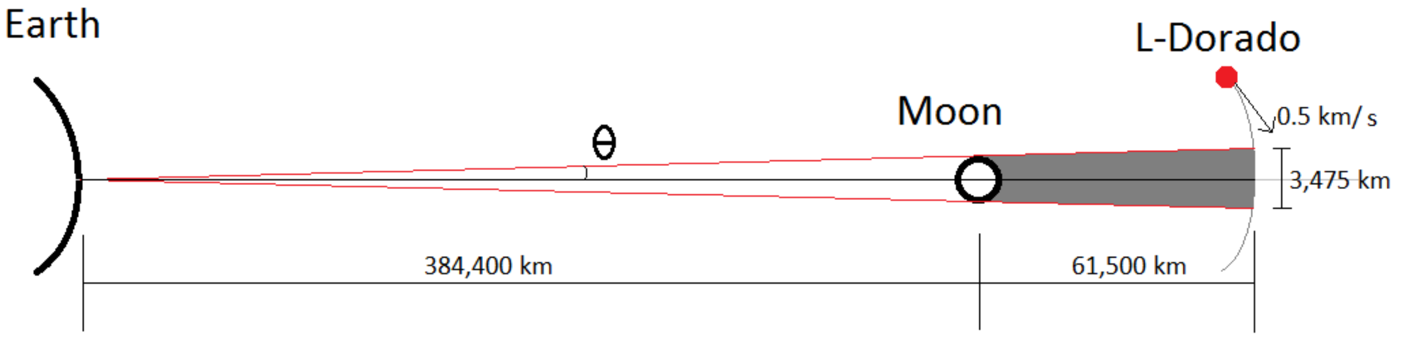


Figure 33: Blackout analysis

Assuming parallel ray propagation from the moon to L-DORADO and using a orbital speed of 0.5 km/s, as well as approximating the orbital path in the black out as straight, the blackout time can be estimated. Also, the rotation of the earth and the orbital movement of the moon are ignored.

D_moon/v_orbital = t_blackout = 1.93 h

6.4.4 Propulsion

For both vehicles, attitude control is ensured by pressure-fed bi-propellant thrusters, as shown in Table 4. Although such a complex system may be overkill for the EUREKA vehicle, using it will increase the technology's maturity, paving the way for the manned mission.

For the EUREKA vehicle, the tanks are designed to provide 193 kg of propellant, enough to provide 50m/s ΔV. Both oxidizers and fuel are separated in two tanks, for mass balancing and redundancy. The oxidizer (MON-3) are 50 liters ($0.05m^3$) each while the fuel (MMH) are 83 liters each ($0.08m^3$), in both cases including 5% margin for volume. The dry mass of the propulsion system, including thrusters, pressurent tank, feeding lines, tank material and valves is estimated at 100kg. For this vehicle, a demand of 5W is estimated for the control of solenoid valves.

The propulsion system for the manned Orion mission vehicle is provided by the MPCV-ESM. With the vehicle mass of 16t, the advertised propellant mass of the service module will provide approximately 1350 m/s.

Table 17: Characteristics of MPCV-ESM, from [Berthe]

Item	EUREKA Vehicle	Orion + MPCV-ESM
Principal propulsion		
Type	None	Shuttle OMS-E dual prop (MON-3/MMH), +/- 6° gimbal
Thrust (N)	None	27 700
Secondary propulsion		
Type	None	8 bi-prop ATV (MON-3/MMH)
Thrust (N)	None	8 x 490
Attitude control thrusters		
Type	24 x bi-prop ATV (MON-3/MMH)	24 x bi-prop ATV (MON-3/MMH)
Configuration	8 modules connected to central reservoirs	8 modules connected to central reservoirs
Thrust (N)	24 x 220	24 x 220
Specific impulse (s)	320	320

Minimum impulse bit (ns)	< 8	< 8
Propellant mass (kg)	193 (MON-3: 53, MMH: 140)	8602 (enough for 1350 m/s ΔV)
Propellant volume (L)	MON-3: 2 x 50, MMH: 2 x 83	N/A
Propulsion system dry mass (kg)	100kg	N/A
Power demand (W)	5W	N/A

6.4.5 Structural Design

The majority of the structural components are provided by sub-contractors. In these cases, the approach is to flow down requirements that they must meet. An example is that they must have positive yield and ultimate margins of safety using factors of safety of 1.25 and 1.40, respectively. Another example would be the need for the vehicles to survive the vibration and acoustic environment. At this time, these levels are undefined for the selected rocket, which makes analysis difficult. In these cases, preliminary loads are defined by NASA's GSFC-STD-7000A "GEVS" document.

Science Habitat

A comparison of traditional can designs and inflatable habitats showed an habitat has a number of additional advantages that make it first choice. TRADE OFF DIAGRAM

The structural mass of the inflatable science habitat is based on available Bigelow data and the NASA TransHab study, see Figure 6.4.7.a. The volume of the habitat developed herein is chosen to accommodate all implemented systems, science equipment and living space for the astronauts, as well as to fit it in the Falcon Heavy payload fairing.

The mass and volume of the required subsystems (GNC, TCS, Communications, EPS) is estimated based on the docking and lunar transfer requirements.

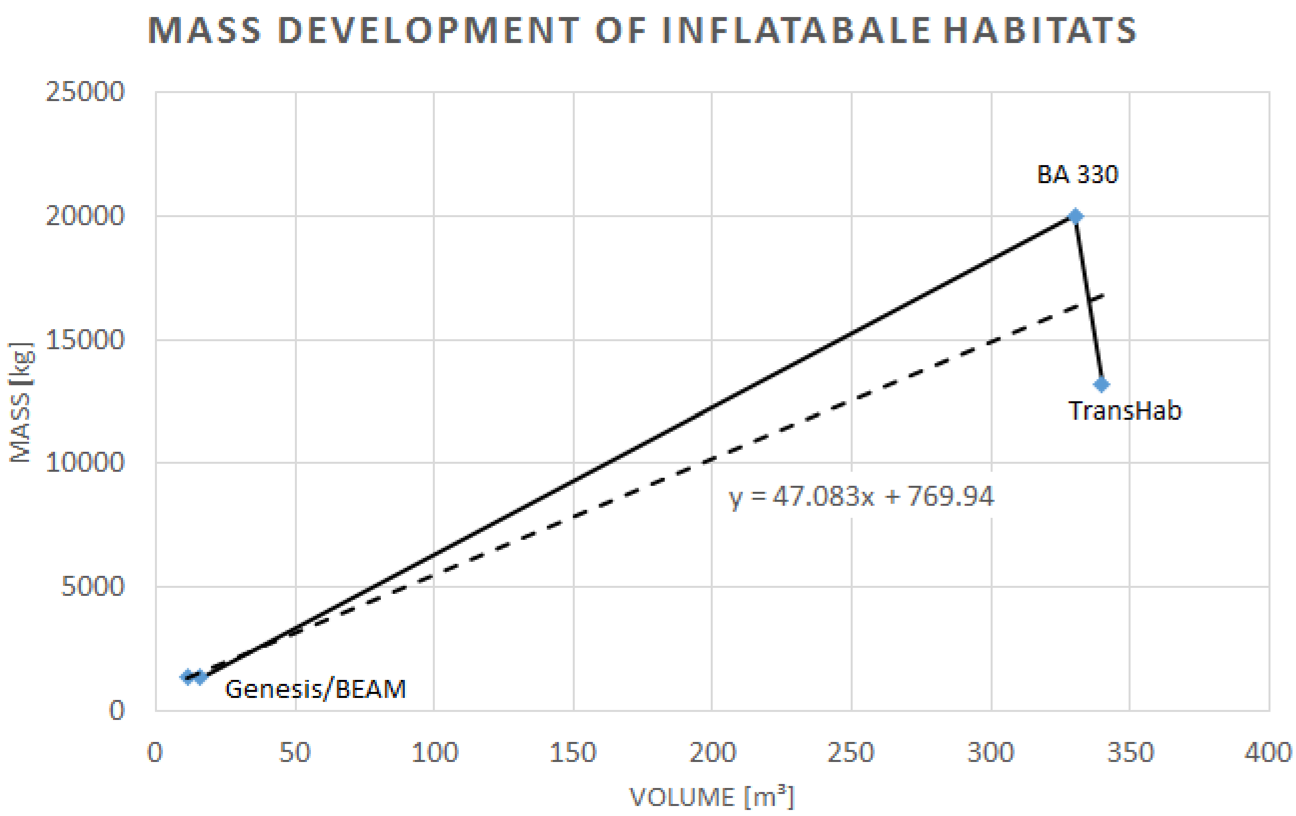


Figure 34: Development of inflatable habitable mass as a function of inflated volume. The difference between the TransHab and the BA 330 stems from the inclusion of an entire service module for the BA 330. Also shown is the linear interpolation curve used in this report.

A traditional can-design is also possible with this mission concept, but a number of science experience what have to be cancelled and the living/working volume would be severely restricted.

6.4.6 Thermal Control

Thermal control of the Orion command module and service module is the responsibility of the vehicle providers. Orion will provide the thermal environment to enable the science, EVA, and human factor goals. It is expected that the thermal requirements for Orion will be achieved through targeted use of thermal paint, radiators, heat pipes, and liquid cooling loops.

For the Eureka science module, the thermal control system must be carefully maintained to preserve the scientific instruments. In addition, Eureka must function in tandem with the Orion command module to provide a comfortable ambient temperature for the astronauts. The thermal environment of Eureka is considered for LEO, transfer to the moon, and the lunar DRO at the asteroid. Eureka is approximated as a 7.5 m spherical spacecraft for first-order calculations.

Eureka will spend most of its time near the asteroid at DRO, and thus controlling the spacecraft at DRO is a priority. Because the DRO is so far away from the Earth and Moon, its view factor to these bodies is approximately zero. Therefore, the albedo and radiative heat transfer with these bodies is negligible compared to the absorbed heat from the sun and the heat radiated to space. In addition, Eureka is not affected by eclipse from the Earth and Moon since this eclipse would have a very short duration. Radiative heat transfer with the ARM and asteroid are also negligible because they are assumed to be at or near zero Kelvin since they are unable to generate heat. Albedo from these objects is also negligible because Eureka will either be perpendicular to the objects or eclipse them. In addition, the reflectance of the ARM and asteroid is small. Therefore, the only heat sources and sinks for the Eureka at DRO are solar irradiance, heat transfer to space, and heat generated by the spacecraft. It is critical that the Orion and Eureka are positioned such that the asteroid and ARM do not eclipse them and such that they do eclipse each other.

At DRO, the Eureka spacecraft must be kept between 20 °C and 25 °C for instrument operation and astronaut comfort. A surface coating of Aeroglaze A276 ($\alpha = 0.90$, $\varepsilon = 0.23$) is used in conjunction with one layer of aluminized Teflon layer MLI ($\alpha = 0.14$, $\varepsilon = 0.74$) to maintain a passive spacecraft temperature of 22.2 °C (Aeroglaze, Wirz 2015). This value was obtained by considering a steady state thermal balance between the solar irradiance, heat radiated to space, and waste heat generated within the spacecraft during normal operation. The layer of insulation gives the spacecraft greater thermal inertia to reduce the effects of transient heat transfer such as eclipses.

For crew and instrument safety, Eureka is also able to regulate its heat generation and radiation transfer through space heaters and deployable radiators. Eureka is equipped with two 1465 W heaters for redundancy that are capable of raising the steady state spacecraft temperature to 35 °C. The heaters compensate for eclipses from the ARM, the asteroid, EVA astronauts, Earth, and Moon. Eureka is also equipped with two deployable 0.715 m^2 radiators for redundancy coated with Z93 ($\alpha = 0.19$, $\varepsilon = 0.92$) that can lower the steady state spacecraft temperature to 10 °C. The radiators allow Eureka to cool off in case of albedo and heat input from the Earth, Moon, asteroid, and ARM. The radiators are carbon-carbon to reduce mass and have a water-glycol fluid loop so that they are non-toxic to the astronauts. Temperature sensors are used inside the Eureka to maintain a feedback control loop. Heat transfer for the solar arrays was also considered and the arrays can easily feather to maintain their operational temperature range of -150 °C to 110 °C while maintaining power requirements (Wirz 2015).

Temperatures for Eureka during LEO and lunar transfer must also be considered. If it is assumed that the Eureka initial temperature at LEO is 20 °C, then the maximum heat gain at sub-solar point (with radiators deployed) and maximum heat loss as eclipse (with heaters on) are ± 10 kW. With a 17,200 kg capsule, a specific heat capacity of 1 kJ/kg/K, and an instrument survivable temperature range of ± 10 °C, it will take over 4 hours for the instruments to become non-functional. Since one 200 km LEO orbit is 1.5 hours, Eureka should be on its way to the moon before its instruments break. The lunar transfer environment is very similar to the DRO because it is very far from the Earth and Moon. Therefore, it can be approximated to be the same as the DRO environment and Eureka will stay between 20 °C to 25 °C. The heaters and radiators can be used to maintain any discrepancies from the desired temperature range.

Table 18: Summary of Eureka Thermal Control System

	Power (W)	**Mass (kg)**	**Volume (m^3)**
*Aeroglaze A276 Surface Coating^*	0	0.5	0.01*
*Aluminized Teflon MLI^*	0	8.6	0.05*
Z93 Coated Deployable Radiators	100	10	1.43*
Space Heaters	2930	29.3	0.10

Temperature Sensors & Control Architecture	1	1	0.1
Sum	3031	49.4	1.71

*: Not included as habitable space, is outside Eureka
^: Included as part of the Eureka structure mass (not TCS mass)

6.5 Science and Technology Demonstration

The following specifications include margins determined as explained in Section 6.7.

	Mass (kg)	Volume (m^3)	Power (kW)
Science Payload	213	8.8	1.8
Tech Demo (of which ISRU)	558 (382)	3.5 (2.7)	43.4 (43.3)

6.6 Additional Robotics

Augmented SPHERES

To provide astronauts a means of prospecting outside the Orion and Eureka spacecraft quickly and safely a robotic, semi-autonomous, swarm system is envisioned. The SPHEREs that are currently being used on the ISS could be augmented with regard to navigation in a vacuum and situational awareness, and used for this purpose. Moreover, the educational aspect of their function that they currently fulfill on the ISS could be maintained as an outreach possibility and educational opportunity where the SPHEREs can be controlled and reprogrammed from the ground. As a technology not core to the functioning of the mission, the successful implementation of SPHERE technology could pave the way for robotic assistance during manned missions in microgravity environments. With onboard equipment such as LIDARs, stereoscopic cameras, attitude control systems (magnetorquers), and propulsion systems, the SPHEREs can make use of CubeSat technologies as a baseline for control and actuation.

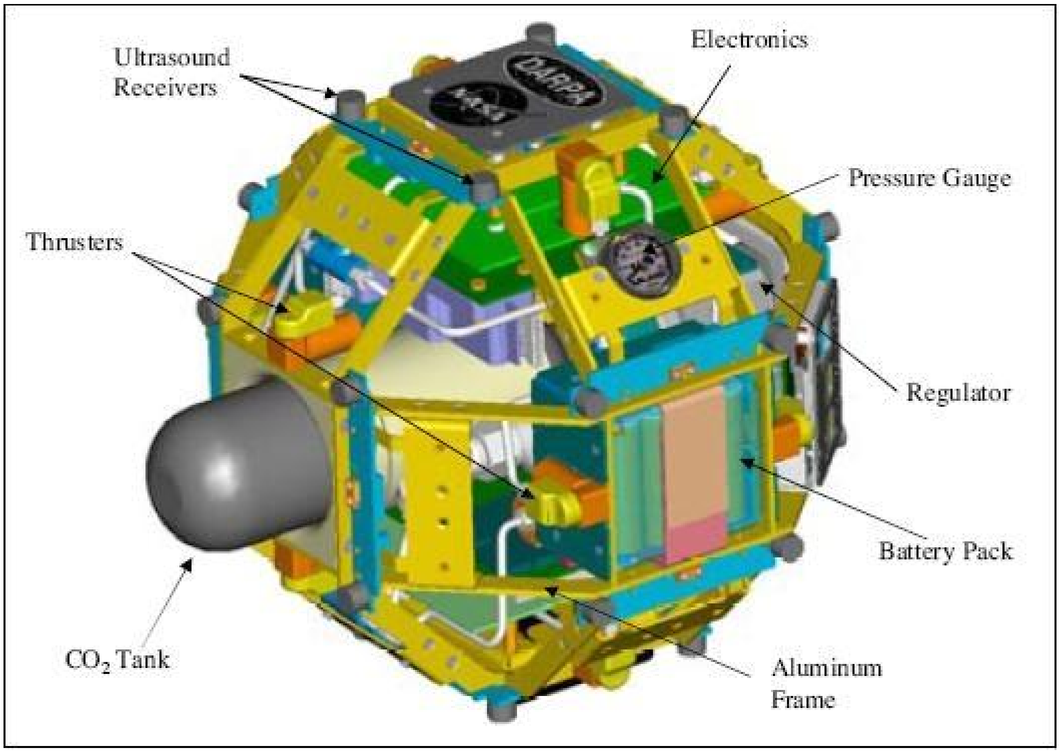


Figure 35: Diagram of SPHERE Onboard the ISS

6.7 Risk and Mitigation

Risk analysis, mitigation and design philosophy follows established standards in space missions. Margins are employed to account for uncertainties in the design process. This can be due to insufficient data, low TRL and to increase safety. Margins are applied according to (Larson) and (NASA/SP-2007-6105) to individual parts/subsystems based on the level of uncertainty estimated by the designer and range from 5% to 25%. If in doubt, a 20% margin is applied. Additionally, a 20% margin is applied to the overall system budgets. Margins apply to mass, power, volume and cost estimates. SCHEDULING? Throughout the design it was attempted to avoid single point of failures (SPOF) through redundancies or other applicable techniques. Loss of Crew (LOC) is deemed unacceptable. Loss of Mission (LOM) can results in failure of achieving the mission objective, but no casualties. In accordance with (Larson) and (NASA/SP-2010-580). Risks are identified by each subsystem and ranked, based on their likelihood and severity. For all identified risks, but especially the ones with high severity and likelihood, mitigation strategies have been developed with respect to (NASA/SP-2011-3422).

7.0 Human Factors, Environmental Control, and Life Support System (ECLSS), EVA

Crew Size, Selection Process & Pre-Flight Training

Crew Size and Makeup

A three-member crew is used for L-Dorado. The three crew members will be composed of two scientists and one engineer. Two scientists are necessary because the mission is designed to experiment-heavy and there is significant amount of potential for scientific discovery. An engineer is necessary because his or her background will complement the understanding and handling of all hardware, software, electrical connections in Orion and Eureka along with the new technologies that are being developed and utilized for scientific demonstration.

Justification

Proper crew size and selection process is key for a successful mission. The objectives of the crew selection process include eliminating potentially unfit applicants and selecting those who will perform and cope in an isolated confined environment optimally. “Select out” and “select in” criteria are used for the process. Select out criteria are focused on identifying candidates with mental problems or who are likely to develop problems. Therefore, specific psychometric tests and personal interviews are used in order to identify and disqualify people with mental issues such as schizophrenia or claustrophobia. On the contrary, select-in criteria focus on those characteristics that are beneficial to the individual and that lead to a successful group functioning. Some of these criteria include:

- communication and interpersonal skills
- interpersonal compatibility
- cross-cultural competence
- ability to work under and cope with the extreme conditions of spaceflight

Therefore, the selection is based on aptitudes, personality, attitudes and experiences. A new selection process based on selecting the crew as a group according to their compatibility and preferences as group instead that on an individual basis is considered. An odd number crew is assumed in order to avoid even numbered decisional splits. People with task-focused styles are preferred since they tend to perform better in short duration missions.

Proper pre-flight training and effective in-flight and post-flight psychological support to the crew and their families, are keys to mission success. Psychological training, survival training, group building activities and training together are the adopted techniques. Then, support to families, food variety, entertainment, stimulation and personal growth are the key for in-flight success.

Physiological Deconditioning in Space

Biomedical records collected from past space flights show significant effects in the cardiovascular, skeletal, muscular and vestibular systems of the human body during and after space travel. The changes that the human body experiences during launch, minutes to hours, hours to days, days to week in microgravity, during landing and post-spaceflight are detailed in the following table:

TABLE: Physiological Effects due to pre-, in-, post- Spaceflight

Human Body Subsystem	Effects during launch	Effects from minutes to hours	Effects from hours to days	Effects from days to weeks	Landing	Postflight
Skeletal		• Biphasic spinal response causes a 1-2” increase in height	• Calcium loss of up to 60%-70%	Bone loss of 1%-2% per month		

Muscle			• Decrease in muscle mass by 20%	Decrease in muscle mass by 30%	Sore and tight muscles	Complete recovery of muscle mass is achievable
Cardiovascular	• Fluid redistribution to upper body	• Loss of hydrostatic pressure, cephalic fluid shift • Increase in heart volume • Decreased heart rate	• Fluid shift continues (body loses up to 15% extracellular fluid by day 2) • Loss of blood plasma volume, loss of total body water • Increase in heart rate over time	Decrease in baroreceptor reflex function Cardiac system gradually stabilizes Heart volume decreases Heart rate continues to increase Possible disturbances in heart rhythm Red blood cell count decreases		• Fluid distribution of body returned to pre-flight conditions
Vestibular			• Space motion sickness • Nasal congestion			

Countermeasure for Physiological Deconditioning from Spaceflight

Countermeasures used by L-Dorado's astronauts for pre-, in-, post- spaceflight and pre-landing have been detailed in table below.

The 39-day mission designed by Team Explorer is categorized as a short-duration mission. Though the mission duration is equivalent to just over a month, the human body experiences cardiovascular, muscular, and skeletal deconditioning in that time period. An astronaut experiences 1-2% bone loss per month with the greatest loss occurring in the hips and legs due to unloading of the skeletal system as is characteristic of a microgravity environment. An astronaut's body is also susceptible to muscle mass loss upwards of 30% in the few short weeks of this mission. In order to decrease the amount of bone and muscle atrophy, a resistive exercise device, the rotary Magneto Rheological (MR) damper has been chose for incorporation into the Orion capsule.

The rotary MR damper will be a part of crew members' seats to allow a means for them to impart rotary motion on their legs and hips. The astronauts' rotary motion in the lower limbs will be resisted by a torque imparted by the MR damper. The rotary MR damper has been chosen for the capsule because it is compact, lightweight of high-variable torque, and will be available for the astronaut's use for the entirety of the mission (Lee 2013).

TABLE: Countermeasures used by L-Dorado for Astronaut Health pre-, in-, post- Spaceflight and pre-landing

Physiological Subsystem	Pre-flight	In-flight	Pre-landing	Post-flight

Skeletal	• Exercise	Rotary Magneto rheological (MR) Damper for Unmanned Vehicle Suspension Systems		• Exercise
Muscle	• Exercise	Rotary MR Damper	MACES suit	• Exercise
Cardiovascular	• Exercise		• Saline fluid loading, MACES suit	
Vestibular				

Mitigating Effects of Radiation during Spaceflight
Van Allen Belts
During travel from Earth to the moon, astronauts experience the most radiation exposure in the Van Allen Radiation Belts (VARB).
The following data table shows the energy fluence measured during outbound and inbound VARB transit durations.
Table. Apollo and Van Allen Belts Energy Fluence Levels (Braeunig, 2014).

Mission Phase	Elapsed Time (min)	Energy Fluence (MeV/cm^2)	
		Electrons	Protons
Outbound VARB Transit	214	2.358e10	7.848e09
Inbound VARB Transit	140	4.913e09	1.472e09

The absorbed dose, D, measured in grays (Gy) can be found using the equation:

D = energy effluence x surface area of body / mass of the crew member

Estimating that the surface area of an astronaut's body is approximately 1.8m^2 and the mass is approximated to be 80kg, the dosage for an outbound VARB mission was calculated to be 0.8948 Gy and for an inbound Varb mission to be 0.1915 Gy. Because the values in the above table were determined for an Apollo 11 mission, the radiation dosage values are different due to changes in orbital inclination in that mission. According to the Office of Health and Services, annual radiation exposure greater than 0.05 Gy is detrimental to the human body. Therefore radiation shielding is necessary for transit (Braeunig 2014).

GCR

Radiation is a large concern for Orion missions. Currently, Orion has a design requirement that any crew member does not exceed 150 mSv during a mission (Plymouth Rock, June 2010). According to Lockheed Martin's Plymouth Rock publication, an astronaut inside Orion will receive a daily effective dose of 1.4 mSv during solar minimums and 0.5 mSv during solar maximums. Looking at the graph below which plots the suggested future solar cycles, as proposed by NASA, we see the year 2025 is suggested to be around the solar maximum of the solar cycle.

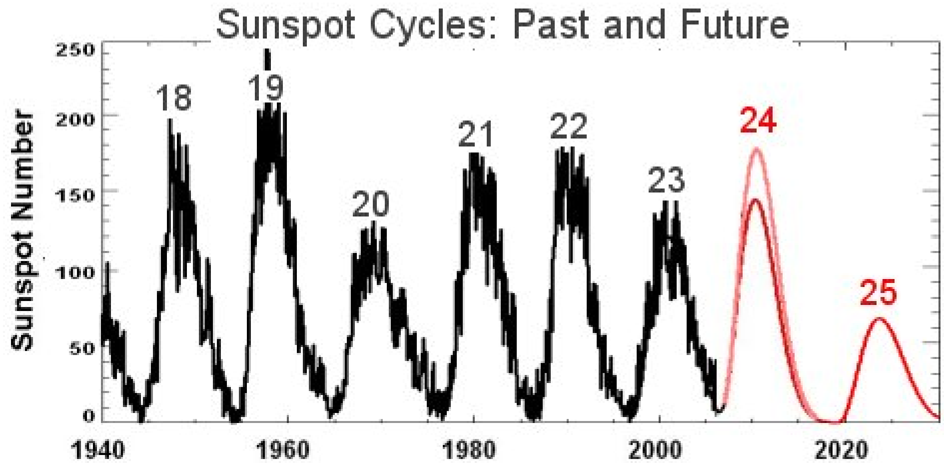


Figure. Sunspot Cycles Graph (Hopkins, 2010).

This would suggest a daily effective radiation dose of 0.5 mSv. We do see however the 25 solar cycle, encompassing the year 2025 is an anomaly and has an extremely small maxima compared to the past several decades. To account for this, we approximated a daily effective radiation dose of 1.0 mSv.

Over the course of a 39 day mission, we would see that the crew would gain a total approximate radiation dose of 39.0 mSv which is well under the Orion design requirement.

SPE

Due to the extended mission duration, the moon and consequently the Asteroid Return Crewed Mission will be out of the earth's magnetosphere at some point for a xx days, as shown in Figure XX. This necessitates additional counter-measures and caution.

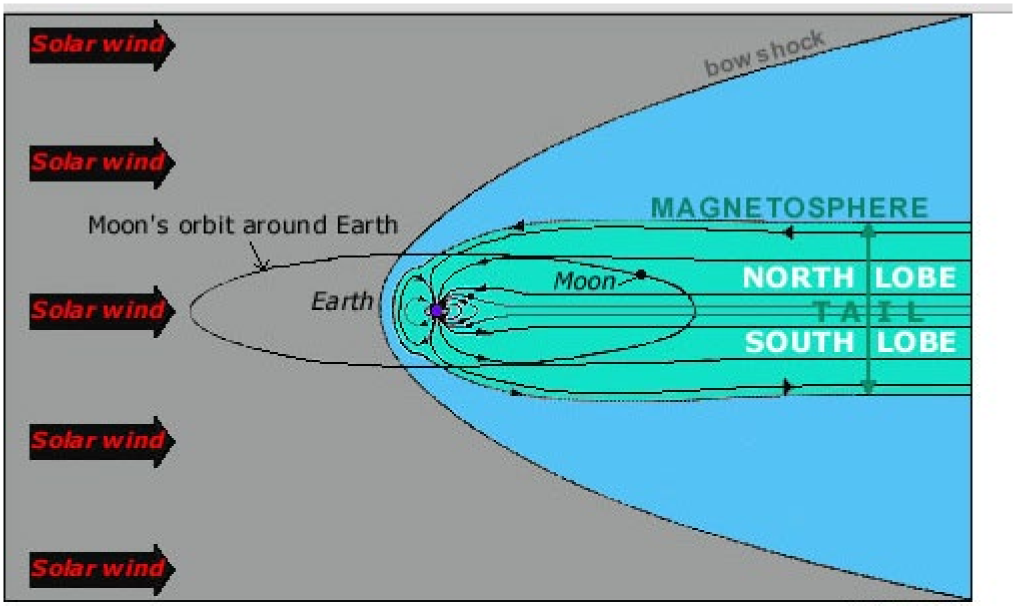


Figure: Illustration of the Moon's orbit with respect to Earth's Magnetosphere

In the case of the solar particle event, astronauts will be required to retreat to one of the inflatable sections of Eureka that will serve as a radiation shelter.

This shelter will be half of the inflatable. The shelter will have walls of 0.05m thickness. These walls are capable of storing approximately 2000kg of water. Extra water reserves, sent in the first launch with Eureka, are pumped into a membrane that is contained in the inflatable walls of the radiation shelter. According to TransHab specifications, after which Eureka has been modeled after, a minimum 5cm packed wall with the inflatable walls is required to protect the astronauts from SPEs.

To minimize radiation exposure during EVA, the communication system will be relaying back X-ray data from the Geostationary Operational Environmental Satellites (GOES) in order to determine the start of solar particle events. In the case that a solar particle is approaching, the astronauts will be warned via communication systems in their suits to retreat to the Orion and then the radiation shelter.

Atmospheric Composition of Orion Capsule and Eureka

The atmosphere of both the Orion Capsule and the Science Lab will be maintained at 14.7psia with an air composition of 78%N2 and 21% Oxygen. 24 hours prior to EVA, two of the three crew members will retreat into Orion that will be depressurized to 10.2 psia as part of the prebreathe protocol that is further discussed in the EVA section. The Science Lab will maintain a 14.7psia environment for the entirety of the period for which the astronauts are utilizing the Science module.

Consumables for Mission

The total consumables needed for the system were determined by a two step process. Water and food amounts were determined by multiplying the rates of max food and water consumption during spaceflight. The hygiene water consumption rate found is significantly large because it is the maximum rate, which considers showers. L-Dorado chose to use maximum rates to accommodate margin. To determine masses of oxygen and nitrogen needed beyond Orion, the ideal gas law was used with the volume, pressure and temperature of Eureka. Approximately 1030kg of N2 and 275kg of O2 are used in Eureka.

Table. Typical consumable consumption rates during Spaceflight. (Klaus 2014)

Consumable	Max Rate (kg/CM-day)
O2	1.85
Potable H2O	7.1
Hygiene H2O	25.58
Food	0.66

Environmental Control and Life Support Systems for Eureka

Factors including habitable volume, layout of science experiments, atmospheric revitalization system for the crew, radiation protection, and capability of storing reservoirs of supplies for future astronauts were considered when designing Eureka.

Habitable Volume

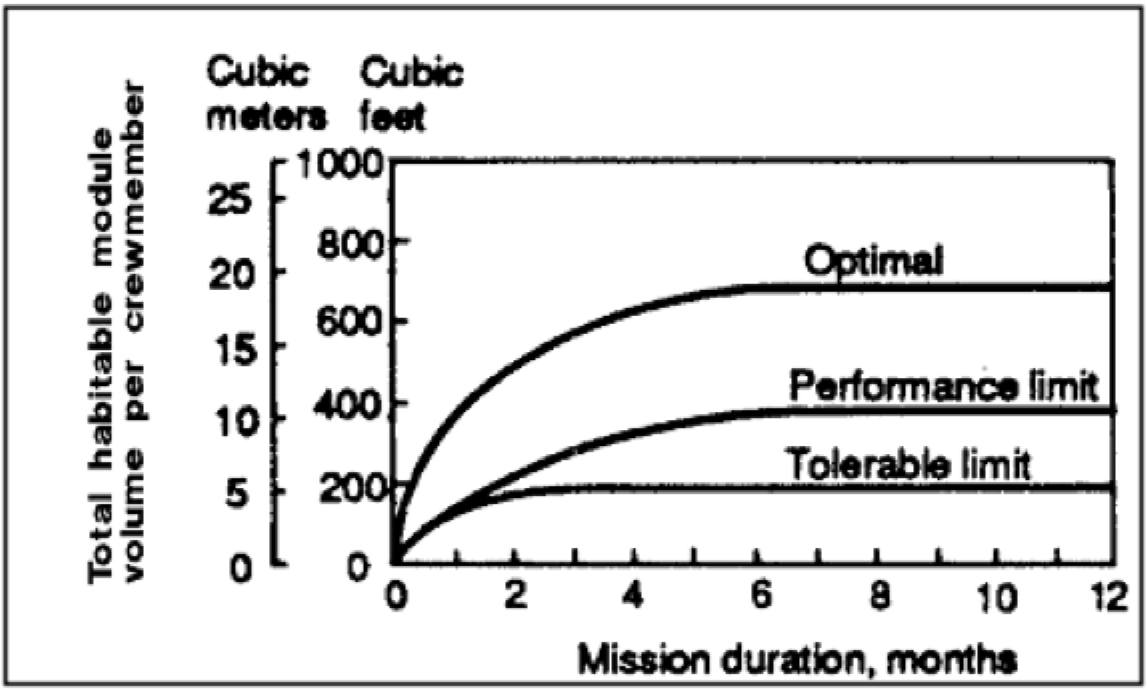


Figure. Celentano curve used to determine habitable volume of spacecraft based on crew size and mission duration (Cohen, 2008).

The Celentano Curve, as shown above, was used to determine a baseline value for habitable volume. Habitable volume is defined as the amount of space accessible and livable by the astronauts. Based off this curve and accounting for a 39-day mission, the crew will require about 24m^3 as an optimal amount of habitable volume. During transit to and from the DRO, the crew will have approximately 9m^3 of habitable volume in the Orion Capsule. The 9m^3 is sufficient for the astronauts during the short transit times and is within performance limits. Upon arrival to the inflatable space habitat, the crew will have access to approximately 35m^3 of habitable volume. This is significantly larger than what the crew requires, however the dimensions of the inflatable were interpolated from dimensions of the Bigelow BA-330 and the TransHab concept developed by NASA. Habitable volume is a necessary consideration for keeping the astronauts healthy, comfortable and productive. Optimal habitable volume will prevent psychological issues from arising as a result of being in an isolated confined environment, improve productivity, reduce noise pollution in the habitat, reduce atmospheric pollution and ensure all functions can be included.

CO2 Removal System in Eureka

After reviewing a variety of CO2 removal and reduction methods and technologies, the Sabatier reactor is utilized in the mission design because of its high efficiency, TRL and water byproduct. Initially beds of 5A zeolite sorbents or LiOH cartridges were considered because a regenerable closed-loop atmospheric maintenance system is not a requirement for the short mission duration. The Sabatier reactor has prior been proposed to be used in a loop with an electrolysis process to produce oxygen from the water byproducts of the reactor.

The Sabatier reactor is used to take the carbon dioxide from the cabin atmosphere in a spacecraft and combine it with hydrogen to produce water and methane. The water produced by this reactor is used in a centrifuge to condense and adsorb such that it can be used in an electrolysis process to form oxygen and hydrogen. The hydrogen that is produced can then be recycled back into the Sabatier reactor. The base method technology used by Sabatier is a chemical reaction. It is an exothermic reaction, limited by thermodynamic equilibrium. The chemical reaction is governed by the equation: $CO_2 + 4H_2 \rightarrow 2H_20 + CH_4$. The Sabatier reactor is composed of a hollow cylinder, hydrogen, and carbon dioxide enter the mixing chamber. The reactants flow over the reuthenium catalyst. Then the heaters around the chamber raise the temperature so the reaction will begin. The integration of parts in the Sabatier reactor is very clean and easy. The CO_2 reduction of the atmosphere is of direct benefit to the crew and the crew does not require any involvement with need to tele-operate anything for this reactors allowing for an easy interface.

Technical papers indicate that the Sabatier would be ideal for long duration missions to Mars for the purposes of oxygen production and methane use for rocket fuel. Sabatier is used by L-Dorado because the water byproduct will be used to demonstrate the ability of a refueling station for future mission. There is no oxygen generation system in Eureka, however there is a electrolysis set up to demonstrate the water collected from the asteroid can be split into oxygen and hydrogen.

The table below shows a comparison between three CO2 reduction systems that were considered for this mission design.

Table. CO2 removal and reduction systems analyzed for Eureka.

	Sabatier	**Bosch**	**LiOH**
Inputs	CO2, H2, [H2/CO2 = 4.5], Heat	CO2, H2, heat	H2O, CO2, N2, O2, LiOH
Outputs	CH4, heat, H2O	C, H2O, heat	H2O, N2, O2, CO2, H2O
Efficiency	96%	NA	NA
TRL	6	4	8
Operability	Autonomous. Only maintenance required involves part replacements after long durations of mechanical wear.	Integration more complex than Sabatier. Catalyst cartridge must be periodically replaced by crew members.	Non-regenerable. The reaction that occurs from the LiOH sorbent is irreversible. The crew will need to replace LiOH cartridges daily making this a poor interface for the crew.

Table. Hardware components of the Atmospheric Revitalization System in Eureka.

Atmospheric Revitalization System	**Hardware**
Air Pressure	Atmospheric Revitalization Pressure Control System (ARPCS) composed of check valves, inlet valves, control switches, sensors, processors.
Oxygen/Nitrogen Storage	Tanks
Humidity Control	Condensing Heat Exchanger
Air Temperature Control	Condensing Heat Exchanger
Carbon Dioxide Reduction	Sabatier Reactor
Trace Contaminant Control	Charcoal Sorbent Bed to Catalytic Oxidizer
Particulate and Microbe Control	Reusable Filters
Air Pressure & Composition Monitoring	Sensor Suite
Fire Detection & Suppression	JPL E-Nose, Alarms, Fire Extinguisher

The following diagram shows the Air Revitalization System in Eureka.

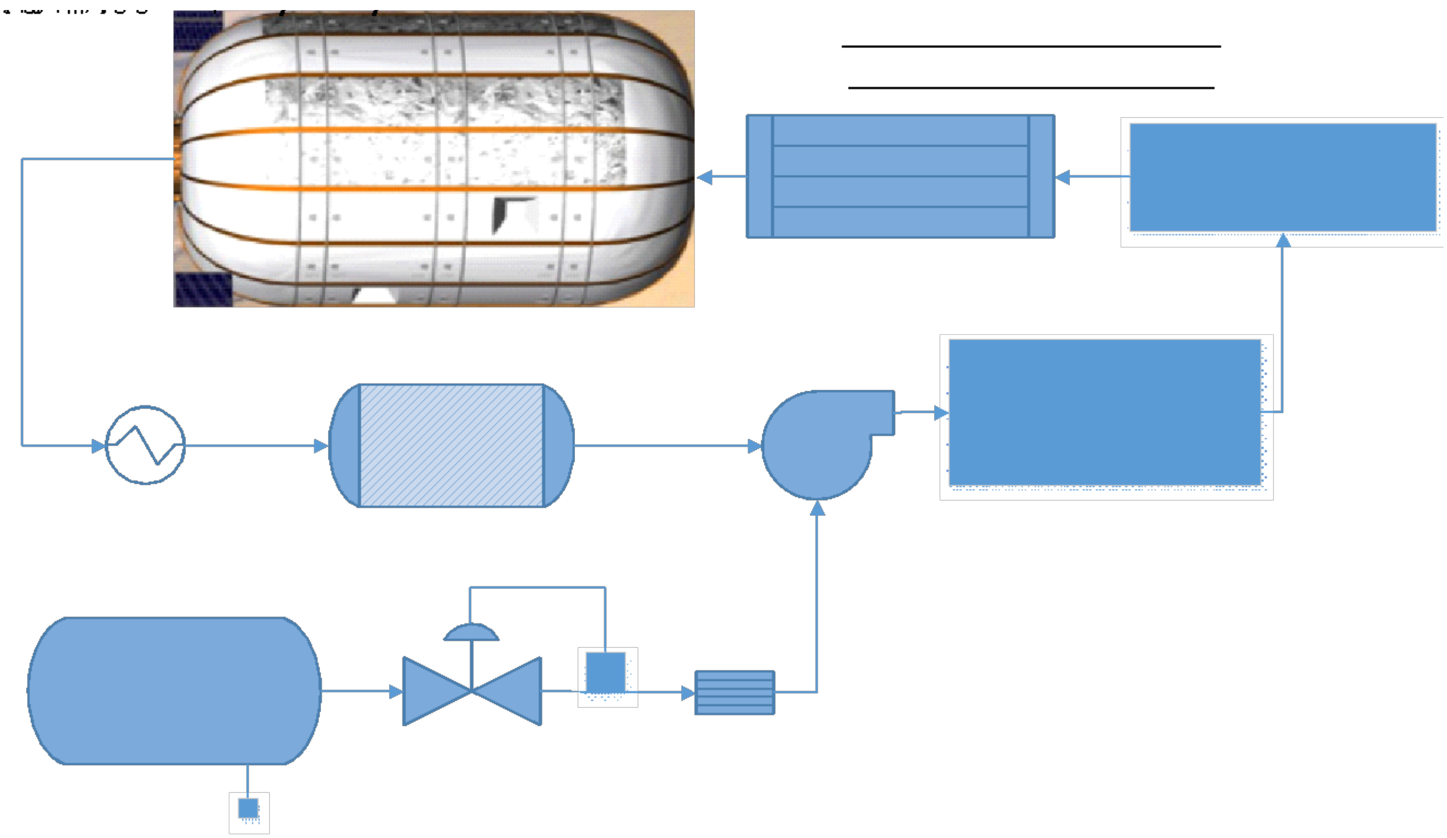


Figure. Air Revitalization in Eureka.

Fire Detection & Suppression

The fire detection and suppression system will consist of JPL Electronic Nose (ENose), smoke alarms to alert the crew, three oxygen masks, and two fine water mist portable fire extinguishers. The ENose is an autonomous sensor suite capable of detecting leaks or spills and was designed to match the actions of the mammalian nose. The ENose works off the method of array-based sensing that is based on detecting patterns and magnitudes of various odors and vapors. Pattern recognition and neural network algorithms makes this technology very robust and reliable for our mission.

ECLSS Demonstrations for Future Space Travel Reservoirs

Water Recovery & Purification

The Vapor Phase Catalytic Ammonia Removal system is being utilized in Eureka for purifying the water recovered from the asteroid pyrolysis of volatiles and contaminants. VPCAR is a phase catalytic process that combines vaporization with high temperature catalytic oxidation of the volatile impurities that vaporize with the water. Recovered water only requires pH adjustments to meet potable standards. The system is broken into two steps:

1) $8NH_3 + 8O_2 \rightarrow 4N_2O + 12H_2O$ [T=250C]
2) $4N_2O \rightarrow 4N_2 + 2O_2$ [450C]

An instrumentation diagram of the VPCAR is shown in the following figure.

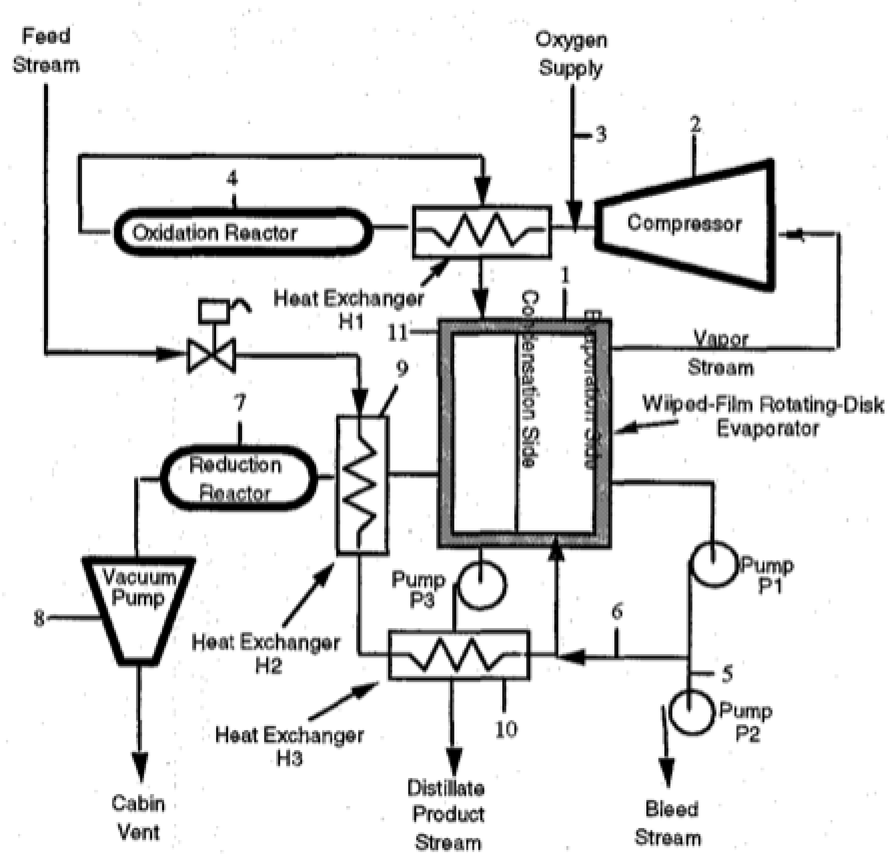


Figure. VPCAR Instrumentation Diagram (Tomes, 2007)

VPCAR was compared with the Water Recovery System on the ISS and a personal Forward Osmosis bag was water purification purposes. The Water Recovery System, part of the Environmental Control and Life Support Systems (ECLSS) on the ISS, is composed of the Urine Processor Assembly (UPA) and the Water Processor Assembly (WPA). This system is capable of processing human wastewater and producing potable water on the ISS. As of 2010, the UPA had processed 2270L of pretreated urine. However, the VPCAR system has the same functionality and is ⅓ of the mass of the WRS. (Yeh)
The Forward Osmosis bag was analyzed for its small-scale water production during L-Dorado's design phase that required one launch. However, to demonstrate that Eureka has the potential to serve as an oasis of resource reservoirs for travelling astronauts, a large scale water purification system with higher TRL was chosen. (Tomes 2010)

Oxygen Generation with H2 byproduct as Rocket Fuel Source

The Oxygen Generation Assembly (OGA) on the ISS is used to split water from the water recovery system to oxygen and hydrogen. L-Dorado's goal is to use the current mission as a step towards building resource reservoirs of oxygen and hydrogen for future astronauts on their way to Mars and the moon. The OGA utilizes Solid Polymer Electrolyte Water Electrolysis (SPEWE) for this function. Eureka has a scaled down version of the ISS SPEWE to demonstrate that an asteroid serves as a source for oxygen for cabin atmospheres and hydrogen for steam rocket fuel. Eureka's system is capable of processing up to 4kg of water in one run.

Experimental Garden in Eureka

The Experimental Garden is a contained group of eight lettuce plants that are exposed to combinations of asteroid regolith as a growth medium, water obtained from the asteroid and then purified in the VPCAR, water from reserves and a earth-based soil as a growth medium. The garden will demonstrate the ECLSS capabilities of providing oxygen to cabin atmospheres and lettuce, a food of relatively high in bio-edible mass. Furthermore, a small-scale greenhouse will have a beneficial effect on astronaut psychology.

ECLSS Environment for Orion

The Environmental Control and Life Support System in Orion will include an amine-based pressure-swing system for CO2 and Humidity Control. Orion's ARS is called the CO2 and Moisture Removal Amine Swing-bed (CAMRAS). In the Orion spacecraft, a pair of CAMRAS systems is used to allow for optimal metabolic load levels. In a CAMRAS unit, air flows from the cabin through a valve , then through the adsorbing bed and then back to the cabin. Two parallel blowers are used to direct the airflow through the sorbent beds. The desorbing bed is isolated and directed to a space vacuum. The adsorption and desorption period is referred to as a half-cycle. The regenerable sorbent, called SA9T, adsorbs the carbon dioxide and water vapor. Each CAMRAS unit contains foam blocks that are filled with porous beads coated with an immobilized liquid amine. These beads are kept in the blocks by aluminum screens. In the Orion system, two CAMRAS assemblies are used for a crew of four and a third is held as a spare (Button 2015). The Active Thermal Control flow diagram of Orion is shown in the following figure:

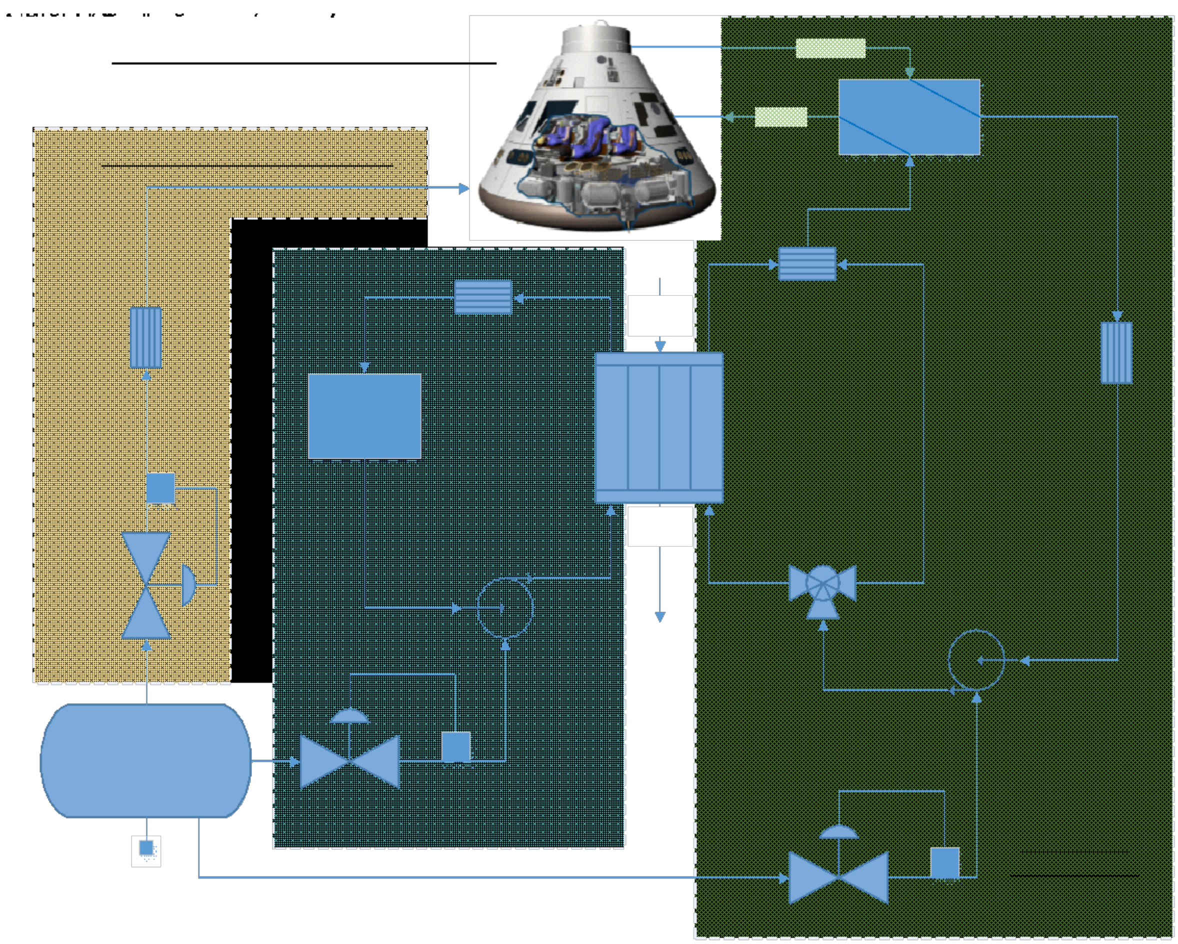


Figure. Orion Active Thermal Control System.

The following diagram shows the Orion Air Revitilization System.

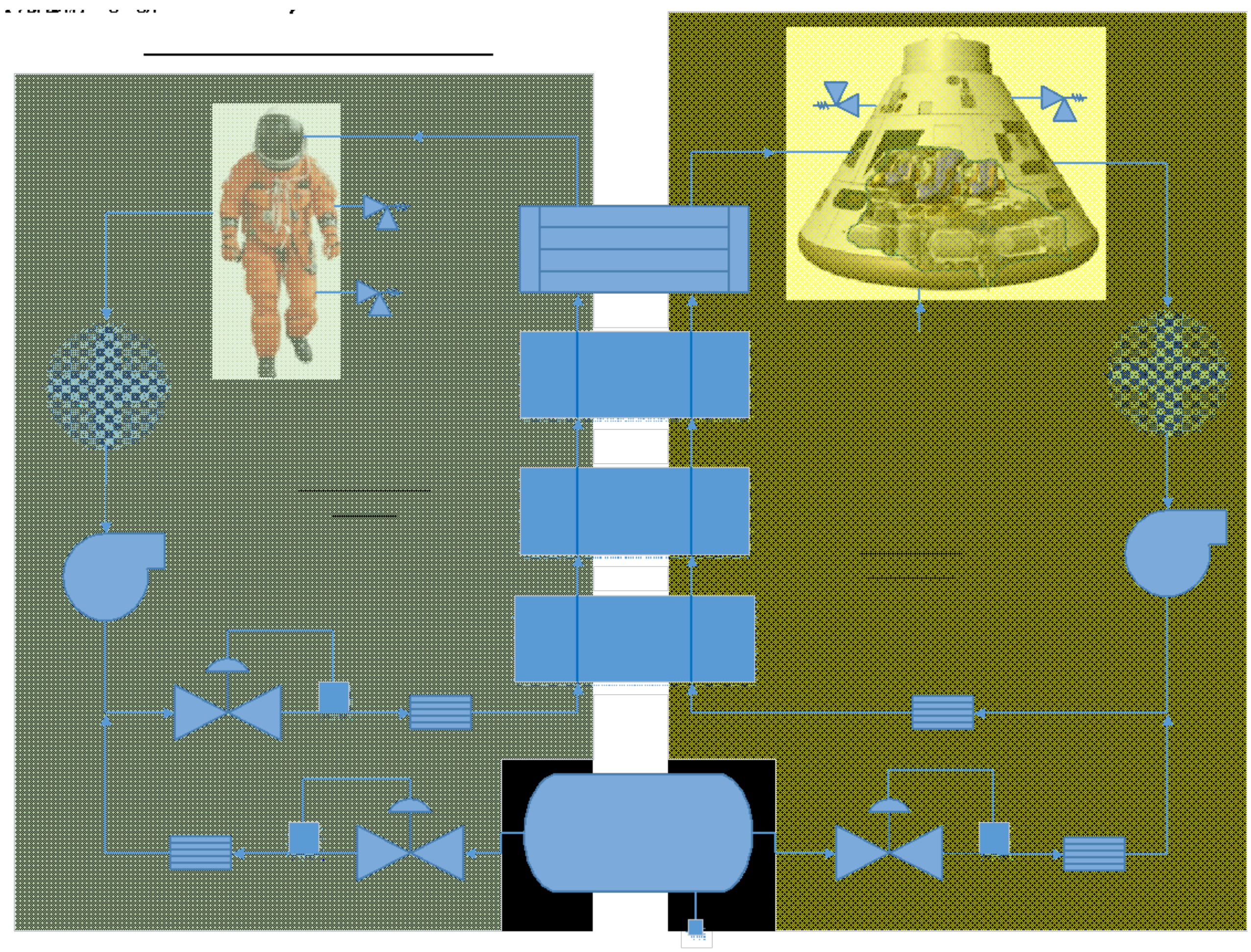


Figure. Orion Air Revitalization System.

Space Suit Selection

The CalTech Space Challenge Mission requires space suits for both IVA (Intravehicular Activity) and EVA (Extravehicular Activity). The following sections will discuss the suit options for both activities as well as rationale as to specific suit selection.

IVA Space Suit

There are several requirements when looking for an IVA suit. The noted requirements for an IVA suit are as follows:

- Provide crew support and protection during launch, reentry, and emergency phases of the mission including:
  - Internal pressurization
  - Supply of breathable oxygen
  - Elimination of carbon dioxide and other dangerous gases
  - Temperature regulation
  - Communications

- Provide emergency life support for up to 10 minutes in the event of LAS (Launch Abort System) emergency
- Provide vehicle life support for up to 168 hours in the event of an emergency cabin depressurization

IVA Suit Options

Currently, there are two suit options that are capable of the IVA requirements listed above. The suits are as follows:

- ACES Suit:

  The ACES (Advanced Crew Escape Suit) Suit is a TRL (Technology Readiness Level) Level 9 piece of equipment. According to NASA, TRL-9 identifies the technology to be “flight proven through successful mission operations” (NASA.gov). The ACES Suit was first introduced in the Space Shuttle STS-68 mission, taking the place of the LES (Launch Entry Suits) Suits. The ACES suit became the baseline IVA suit for the rest of the STS missions and all ISS missions. This suit includes an open-loop demand air system and could be pressurized up to 3.46 psia. It also included an EOS (Emergency Oxygen System).
- MACES Suit:

  The MACES (Modified Advanced Crew Escape Suit) Suit is currently under development at NASA Johnson Space Center. The MACES Suit is an ACES suit with several modifications and upgrades. The changes/upgrades are as follows:
- Feature a closed-loop system instead of an open-loop system using Apollo-inspired equipment
- New suit fit to allow for optimal Orion seat integration
- Relocation and repackaging of EOS for optimal Orion seat integration
- Increase in operating pressure for EVA capability
- Increase in specific suit component customizability
- Various changes in suit fit and components allow for greater crew mobility

The MACES Suit is currently under development and a TRL level 6 as the system has been demonstrated in a relevant ground or space environment through the use of NBL (Neutral Buoyancy Laboratory) tests as well as vacuum chamber testing. The MACES suit is set to achieve TRL level 8 (flight qualification) by 2021.

Suit Comparisons

Topic	ACES Suit	MACES Suit
Current TRL	9	6
2025 Expected TRL	9	8
Total Weight (kg)	41.7	16
Vehicle Design	Shuttle	Orion
Primary Life Support	Vehicle Provided - Open-loop	Vehicle Provided - Closed-loop
Backup Life Support	EOS - 10 minutes	EOS - 10 minutes

Nominal Operating Pressure (psia)	3.46	~4.3*
EVA Capability	No	Yes
Cost	$180,000	~$360,000**
Design		

* Exact operating pressure unknown. Based upon EVA capability and current EVA suit (EMU) nominal operating pressure of 4.3 psia
** Exact value unknown. Development is already at TRL level 6, can assume not much more R&D costs required. Estimated double cost of ACES for margin

<u>IVA Suit Selection</u>

The IVA suit Team Explorer has selected for the Space Challenge is the MACES Suit. This choice was made because of the following facts:

- The MACES suit is scheduled to be flight capable years before our mission date allowing for a safe margin for a time window and time for additional testing
- The MACES suit is specifically designed for the Orion vehicle in mind, providing greater safety, comfort, and mobility when operating the Orion vehicle
- An operating pressure of 4.3 psia is preferable over an operating pressure of 3.46 psia in cases of emergency life support as a pressure of 3.46 is much closer to the unsafe region of pressure for crew
- The MACES suit weighs less than half of the ACES suit
- The closed-loop architecture of the MACES suit allows operation with the Orion closed-loop ECLSS system
- The increase in cost is marginal in the scheme of the other ECLSS equipment and overall mission cost

EVA Suit

There are several requirements when looking for an EVA suit as well. The noted requirements for an EVA suit are as follows:

- Provide crew life support during a maximum of 10-hour periods of EVA activity including:
    - Internal pressurization
    - Supply of breathable oxygen
    - Supply of drinkable water
    - Elimination of carbon dioxide and other dangerous gases
    - Temperature regulation
    - Communications
    - Waste collection device
- Provide crew protection against harsh space conditions including:
    - Radiation
    - MMOD (Micro-meteorite orbital debris)
    - Extreme temperature

- Vacuum conditions
- Provide emergency life support for up to 1 hour in the event of an emergency during EVA activity
- Allow mobility of crew to perform required EVA tasks

EVA Suit Options

Currently, there are three suit options that are capable of the EVA requirements listed above. The suits are as follows:

- EMU (Extravehicular Mobility Unit)

  The EMU is currently a TRL level 9 piece of equipment as it is flight proven through success mission operations. The EMU was first introduced in 1981 and has since then become the baseline EVA Space Suit. The EMU takes advantage of the PLSS (Portable Life Support System).
- MACES Suit

  As stated before, the MACES Suit is currently under development at NASA Johnson Space Center. The MACES Suit is an ACES suit with several modifications and upgrades allowing for EVA capability using an attachable MLI (Multi-layer Insulation) material. The MACES Suit is currently a TRL level 7 as the system has been demonstrated in a operational environment through the use of NBL tests as well as vacuum chamber testing. The MACES suit is set to achieve TRL level 8 (flight qualification) by 2021. The MACES suit in the EVA configuration will also take advantage of the PLSS 3.0 system, the new life support system developed by NASA Johnson Space Center. The new PLSS 3.0 will take the place of the PLSS and is designed for newly developed EVA suits and will allow for greater life support and protection capabilities. This includes longer EVA times, more advanced and reliable life support systems, and improved emergency instruments. A disadvantage of the MACES suit is that it is the least mobility capable EVA suit.
- Z-3 Suit

  The Z-3 suit is the next suit in the Z-series Exploration Suit series. Currently, the Z-3 series suit has not been procured. NASA Johnson Space Center currently has the Z-2 space suit which has a TRL level 7 assuming successful testing in the NBL and human-rated vacuum chamber, but the current TRL level is 6. After Z-2 testing and analysis, the Z-3 series will be procured. The Z-3 series suit will be a TRL level 8 suit (flight qualified) and is expected to be delivered by 2022. The Z-3 series suit will take advantage of the PLSS 3.0 system. The new PLSS 3.0 will take the place of the PLSS and is designed for newly developed EVA suits and will allow for greater life support and protection capabilities. This includes longer EVA times, more advanced and reliable life support systems, and improved emergency instruments. The Z-3 suit is also expected to be the most mobile EVA suit ever developed, allowing easier and faster crew operations during EVAs. It is to be noted that the Z-3 suit and PLSS 3.0 are set to be sent to the ISS (International Space Station) in 2022 for a series of integrated testing EVA's. This will qualify the Z-3/PLSS 3.0 system to be TRL level 9 by the 2025 flight.

Suit Comparisons

Topic	**EMU**	**MACES Suit**	**Z-3 Suit**
Current TRL	9	7	6*
2025 Expected TRL	9	8	9
Total Suit Weight (kg)	55.3	36**	65***

LSS (Life Support System)	PLSS	PLSS 3.0	PLSS 3.0
LSS Weight (kg)	90	Unknown	Unknown
Primary Life Support Time (hours)	8`	10``	10``
Secondary Life Support Time (hours)	0.5`	1``	1``
Nominal IVA Pressure (psia)	0.9	0.9``	0.9``
Nominal EVA Pressure (psia)	4.3	4.3``	4.3``
DCS Pressure (psia)	8.2	8.4``	8.4``
Cost (millions of dollars)	12	100	200
Design		 ^	  ^^

* Current TRL level based upon Z-2 Suit
** Weight estimated using base weight and including the MLI weight (unknown)
*** Weight based upon Z-2 Suit
` Time can vary based upon crewmember metabolic rates
`` Time estimations based upon PLSS 3.0 requirements using an average crew metabolic rate
^ Picture does not include MLI cover
^^ Pictures including Z-1 (left) and Z-2 (right) designs. Z-3 design currently unknown

EVA Suit Selection

The EVA suit Team Explorer has selected for the Space Challenge is the Z-3 Suit with the PLSS 3.0 Life Support System. This choice was made because of the following facts:

- The Z-3 suit and PLSS 3.0 is being designed as the future exploration space suit that astronauts will be using on future missions to Mars, making it a more appropriate choice than the EMU or

MACES suit. Utilizing the Z-3 suit and PLSS 3.0 now will allow for testing of the suit in a non-LEO environment to improve before future missions to Mars.

- The Z-3 suit is expected to be fully flight proven by the 2025 mission
- The PLSS 3.0 will allow for longer and more advanced EVA capabilities than the EMU suit with the original PLSS
- The Z-3 suit is a much more mobile suit than the MACES suit, allowing the crew to be more capable during the missions critical EVAs

As a contingency, if the Z-3 suit fails during development, or is not ready by the flight date, the MACES suit can be used for EVA's. This suit will already be used as the IVA suit and so the only additional material for the suit to become EVA capable is the MLI layer attachment.

8.0 Programmatic Considerations

8.1 Cost

A Rough Order of Magnitude (ROM) cost of the proposed mission is given in this section. The estimate considers the development and procurement of the science laboratory, as well as the operational costs of the launch and operation.

This estimation DOES NOT include:

- Development of the SLS launch vehicle
- Development of the Orion capsule
- Development of Service transfer vehicle
- Development of new space suit
- Annual operating costs of launch facility
- Leasing of NASA facilities as this is a flagship mission

Estimation of the launch cost using the SLS block 1B vehicle is particularly difficult. Web resources hint at values ranging from 2 to 14 billion USD, depending on the launch frequency if the ground station costs are included and the development costs are spread. For this estimate, only the fabrication, integration and test of the actual vehicles used are included, resulting in a cost of 2 billion USD.

Table 19: Rough Order of Magnitude cost estimate for launch vehicles

Item	ROM cost (M$)
2.0 Launch Vehicle	
2.1 First launch (Falcon Heavy)	270

2.2 Second launch (SLS block 1B)	2000

Table 20: Rough Order of Magnitude cost estimate for exploration mission

Item	ROM cost (M$)	**Comment**
1.0 Space Vehicle		
1.1. Service transfer vehicle	0	Assumes European contribution
1.2. Eureka vehicle		
1.2.1. Inflatable structure	150	USCM8: ~(23 k$/kg * 4000kg * 150%)
1.2.2. Laboratory service module	250	USCM8: ~(23 k$/kg * 136kg * 150%)
1.2.3. Science instruments	700	20 instruments @ 20 M$ + 25%
3.0 Ground Command & Control	50	3% of laboratory cost
4.0 Program level		
4.1. System engineering	80	20% of laboratory cost (not instruments)
4.2. Program management	60	15% of laboratory cost
4.3. System integration and test	60	15% of laboratory cost
4.3. Product assurance	20	3% of laboratory cost
4.5. Other	0	
5.0 Flight Support Operations	0	
6.0 Aerospace Ground equipment	0	
7.0. Operations		
7.1. PMSE	60	15% of laboratory cost
7.2. Space segment maintenance	20	
7.3. Ground segment	50	30 engineers + 10 tech, for 2 months
Total cost	**3500.0**	

Table 21: Details of science instruments cost

Item	**Cost**
Chisel in a cup	240

Plasma drill	100
Pyrolysis chamber + crusher	94
Scientific drill + VISIR spectrometer	63
Compact Electrostatic Separator (CES)	48
Lab crusher + dust handling work station	21
Sintering Mirror and Frame	13
APXS	12
XRD	12
GC-MS	12
High resolution camera	12
Gamma-ray log	11
Neutron log	11
Density log	11
Thermocouple	8
Cosmic ray detector	8
Centrifuge	3
Steam rocket	2
Other instruments:	3
Microsopic imager	
Centrifugal sieve	
Sample packaging	
Experimental garden	
Sintering material packaging	
Electrostatic powder handler	
Sintering Mold	
Soil	
Extra margin	16

8.2 Risk

8.3 Political Considerations

International cooperation is a critical aspect of the mission. The expectation is that both the European and Russian space agencies will contribute key instruments, personnel, and supporting resources. Specifically, the expectation is that the Orion Service Module is contributed by ESA. Although cooperation with China is considered desirable, the assumption is that NASA's current congress-mandated ban on working with China will continue to be in effect. It is conceivable that India's growing prowess in space will continue to a point that they can contribute meaningfully. With Congress' approval their participation will be encouraged.

8.4 Planetary Protection

The planetary protection plan consists of two parts: ensuring operations do not allow for ballistic damage to the earth and that operations do not contaminate either the earth or asteroid with bacteria, toxins, or scientifically undesirable pollutants.

A segment of the population will question the wisdom of bringing a relatively large asteroid near earth. It is natural to ask what the risk of impact is and the team expects to address this through a series of activities. Examples include web-based articles and interactive games that help people understand gravity wells and in-person lectures by scientists, engineers, and astronauts.

Contamination of or at the earth is a concern for different groups, albeit for different reasons. Some may be concerned about space-based viruses or bacteria infecting humans or nature. Some may be more concerned about earth-based contaminants polluting returned samples. In all cases, the solution is to seal samples with redundant systems that prevent sample and atmosphere interactions. The samples are contained in these systems until they can be studied in an environmentally sealed laboratory.

8.5 Public Relations and Outreach

Public outreach is considered as important as many of the technical aspects of the mission. It educates and inspires the public, and contributes to let politicians understanding about the importance of space activities. This means that public outreach has a big impact on political decisions and, therefore, on the financial support given to the space sector. A good public outreach plan has the potential to assure future space activities to be conducted and accomplished. Therefore, the value of investing in public support cannot be underestimated.

Specific concepts for public outreach are developed for the XX-name of the mission-XX mission. They include the following main activities:

1. Allowing the public to take space selfies; a green screen is installed during the XX-name of the mission-XX mission with the asteroid behind it. A camera on the vehicle sees both the screen and the asteroid and takes a series of images with the moon both visible and not visible in the background. Then, a on-purpose

developed website allows the public to upload their pictures and a dedicated software stitches the two together

2. Release of CubeSats from Universities after separation of the second stage for educational purposes
3. Astronauts take selfies during the mission and during the EVAs. The pictures are shared through the most popular social-media platforms (e.g. Twitter, Facebook, etc.) since public appearances from astronauts bolster public relations and outreach
4. 3D printable models are created: models of the asteroid, as well as the various modules of the spacecraft, can be printed with a 3D printer by the public
5. Partnership with games companies is created in order to have a dedicated games for the mission so that "arm chair astronauts" can enjoy simulating the mission and the in-orbit activities. Indeed, the Kerbal Space Program demonstrates that there are a large number of "arm chair astronauts" that can become some of the strongest public supporters of the mission
6. Making comic strips about the mission
7. Using returned asteroid material to create art through announced competitions (e.g. statues, paintings, etc.)
8. Creation of souvenirs made with asteroid material and possibility to name them with laser technology
9. Production of "astronaut sorbet" from the processed water retrieved from the asteroid.

In order to raise public awareness about the XX-name of the mission-XX and its goals, a Facebook page has been created and opinions from the public have been asked, e.g. why sending people to an asteroid, what asteroid resources are useful for.

9.0 Conclusions

Team Explorer proposes the L-Dorado mission to characterize and utilize the ARM asteroid. A two-rocket solution allows for substantial science, technology demonstration, and public outreach activities. The mission starts in August 2024 with a launch of the Eureka science module on a Falcon Heavy rocket. The science module takes a 6-month journey on a ballistic trajectory to the asteroid. In March 2025, a 3-person crew launches in an Orion capsule on a SLS rocket. They spend 39 days in space, with 17 days used for traveling to and from the asteroid and 22 days used for science and utilization of the asteroid. Three EVAs and a suite of instruments strike a balance between autonomous and manual operations. Throughout the mission, public outreach and international cooperation are considered critical. A total cost of approximately 4 billion dollars.

Acronym	Definition
ACES	Advanced Crew Escape Suit
APXS	Alpha Particle X-Ray Spectrometer
ARC	Asteroid Regolith Crusher
ARM	Asteroid Redirect Mission
ARPCS	Atmospheric Revitalization Pressure Control System
ATV	Automated Transfer Vehicles
BIRD	Battery-operated Independent Radiation Detector
CAMRAS	CO2 And Moisture Removal Amine Swing-bed
CES	Compact Electrostatic Separator
CES	Compact Electrostatic Separator
CheMIN	Chemistry & Mineralogy
CR3BP	Circular Restricted Three Body Problem
CRaTER	Cosmic Ray Telescope for the Effects of Radiation
DAN	Dynamic Albedo of Neutrons
DCS	DeCompression Sickness
DRO	Distant Retrograde Orbit
EB	Electrostatic Beneficiation
ECLSS	Environmental Control and Life Support System
EG	Experimental Garden
EMU	Extravehicular Mobility Unit
ENose	Electronic Nose
EOS	Emergency Oxygen System
EPM2	Electrostatic Power Manipulator for Microgravity
EPS	Electric Propulsion System
ESM	European Service Module
EVA	Extra Vehicular Activity
GA	Geothermal Anywhere
GC-MS	Gas chromatography–mass spectrometry
GCR	Galactic Cosmic Radiation
GNC	Guidance Navigation and Control
GOES	Geostationary Operational Environmental Satellites
GPR	Ground Penetrating Radar
IMU	Inertial Measurement Unit
ISP	Specific Impulse
ISRU	in-situ resource utilization

ISS	International Space Station
IVA	Intravehicular Activity
JAXA	Japan Aerospace Exploration Agency
JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
LAS	Launch Abort System
L-Dorado	Lunar-reDirected Orbiting Resource Asteroid Demonstration and Operation
LED	Light Emitting Diode
LEO	Low Earth Orbit
LES	Launch Entry Suits
LIDAR	Light Detection and Ranging
LOC	Loss of Crew
LOM	Loss of Mass
LRO	Lunar Reconnaissance Orbiter
LSS	Life Support System
MACES	Modified Adavnced Crew Escape Suit
MAHLI	Mars Hand Lens Imager
MER	Mars Exploration Rover
MI	Microscopic Imager
MLI	Multi layers insulation
MMOD	Micro-meteorite orbital debris
MPCV	Multi-Purpose Crew Vehicle
MR	Magneto Rheological
MSL	Mars Science Laboratory
NASA	National Aeronatics and Space Administration
NBL	Neutral Buoyancy Laboratory
NEO	Neart-Earth Object
OGA	Oxygen Generation Assembly
PLSS	Portable Life Support System
RAAN	Right Ascension of the Ascending Node
RAD	Radiation Assessment Detector
RAM	Radiation Area Monitor
ROM	Rough Order of Magnitude
SAM	Sample Analysis at Mars
SDC	Science Drill Campaign
SEC	Shackleton Energy

SEM	Scanning Electron Microprobe
SEP	Solar Electric Propulsion OR Solar Energetic Proton
SKG	Strategic Knowledge Gaps
SLS	Space Launch System
SM	Service Module
SPEWE	Solid Polymer Electrolyte Water Electrolysis
SPOF	Single Point Of Faillure
TCS	Tele-communication system
TLI	Trans-lunar Injection
TOF	Time of Flight
TRE	Thermal/Radiation Experiment
TRL	Technology Rediness Level
UCIS	Ultra-Compact Imaging Spectrometer
UPA	Urine Processor Assembly
USCM8	Unmanned Space Vehicle Cost Model, Eighth Edition
VARB	Van Allen Radiation Belts
VISIR	Visible Near-Infrared
VPCAR	Vapor Phase Catalytic Ammonia Removal
WPA	Water Processor Assembly
XRD	X-Ray Detector

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Appendices

Appendix A - Selecting H_2O Extraction Method

The three main approaches for extracting water from volatiles are (Mazanek)

1. Pyrolysis: The regolith is heated until the H2O outgasses
2. Ionic liquid acid dissolution
3. H_2SO_4 or HF dissolution

Pyrolysis was chosen to avoid the use of hazardous chemical and because of its relative simplicity.

Appendix B - Regolith Pyrolysis for H2O Extraction

Ore Chamber Size

Required ore mass to undergo pyrolysis is 500 kg. Assume ore density is 3400 kg/m^3.

Mass = Height * pi * (Radius^2) * Density

For ore carrier the dimensions are:

Assuming height = 1 m, radius = 0.22 m

Pyrolysis Chamber Size and Mass

With a 0.02 m clearance around the ore carrier, the pyrolysis chamber dimensions are:

Height = 1 m, radius = 0.24 m

To find the mass of the pyrolysis chamber mass, the pressure vessel stress equation was used:

Wall Thickness = (Internal Pressure * Vessel Radius) / Material Yield Strength

Assuming Titanium, the minimum wall thickness is 250 um. Since this is infeasible we assumed 2 mm, giving this chamber a structural mass of 8 kg. Titanium was selected because of its high strength to weight ratio and high melting point (1668 C).

Condensation Chamber Size

The chamber needs to contain a minimum of 5 kg of water. Here we assume maximum of 25 kg of water per condensation container. Assuming a density of 1000 kg/m^3.

Height = 0.25 m, radius = 0.18 m

Water Chamber Size

At the start of the pyrolysis process, a valve will open between the water chamber and the pyrolysis chamber. Liquid water will undergo a phase change into vapor upon contact with the vacuum in the pyrolysis chamber. This water is required to transfer heat from the resistance heaters on the walls to the ore in the ore carrier. The water is circulated by a propeller in the pyrolysis chamber.

The water chamber is sized so that at 400 °C the pressure in the pyrolysis chamber is 1 atmosphere. Total volume of the inside of the pyrolysis chamber is ~0.24 m^3. At 400 °C and 101.3 kPa (1 atm) the specific volume of water vapor is 3.062 m^3/kg (Tucker). Thus, the mass of water required is 78 grams and volume is 7.8e-5 m^3.

With height of 10 cm, radius is 1.6 cm.

Appendix C - Pyrolysis Energy Calculations

The amount of energy required to extract water from regolith material depends greatly on what the form of water is. Ices may only need to be heated to 100 °C, but if the water is trapped in hydrated minerals, the temperature needs to be raised to 400 °C - 800 °C.

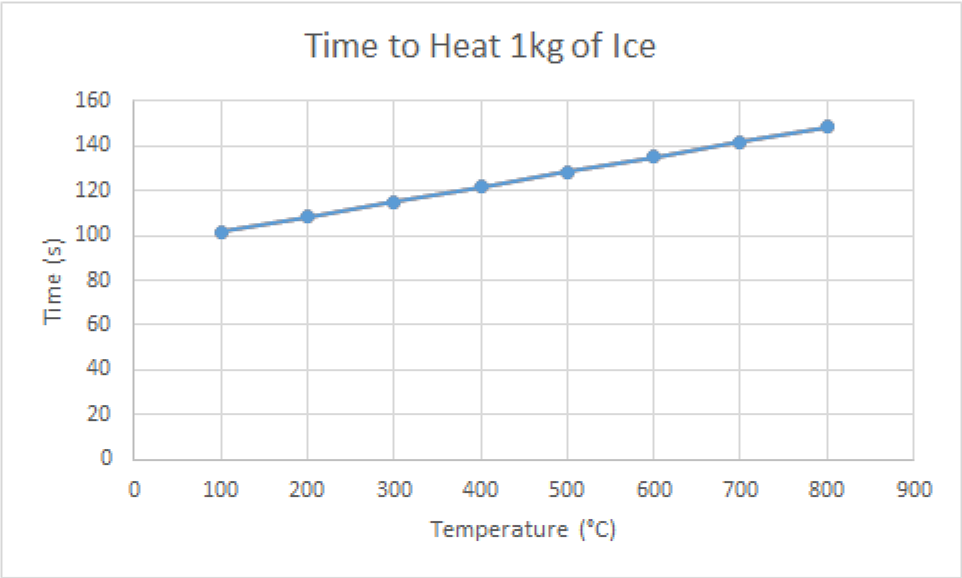

Figure 36: The time to heat 1 kg of ice from 0 Celsius assuming a pressure of 1 atmosphere and a power of 30 kW.

Constant pressure specific heat (liquid) = 4.2 kJ/kg/°C

Constant pressure specific heat (gas) = 2.0 kJ/kg/°C

Latent heat of vaporization = 2.3 MJ/kg/°C

Latent heat of melting = 0.33 MJ/kg/°C

The heating process takes place between EVA 2 and EVA 3, so lower power can be used. Given that the time to heat 1 kg of pure ice to 800 °C with 30 kW is 150 seconds, even if it requires an order of magnitude more energy to release water from hydrated minerals the pyrolysis will be completed before EVA 3.

Appendix D - Steam Rocket

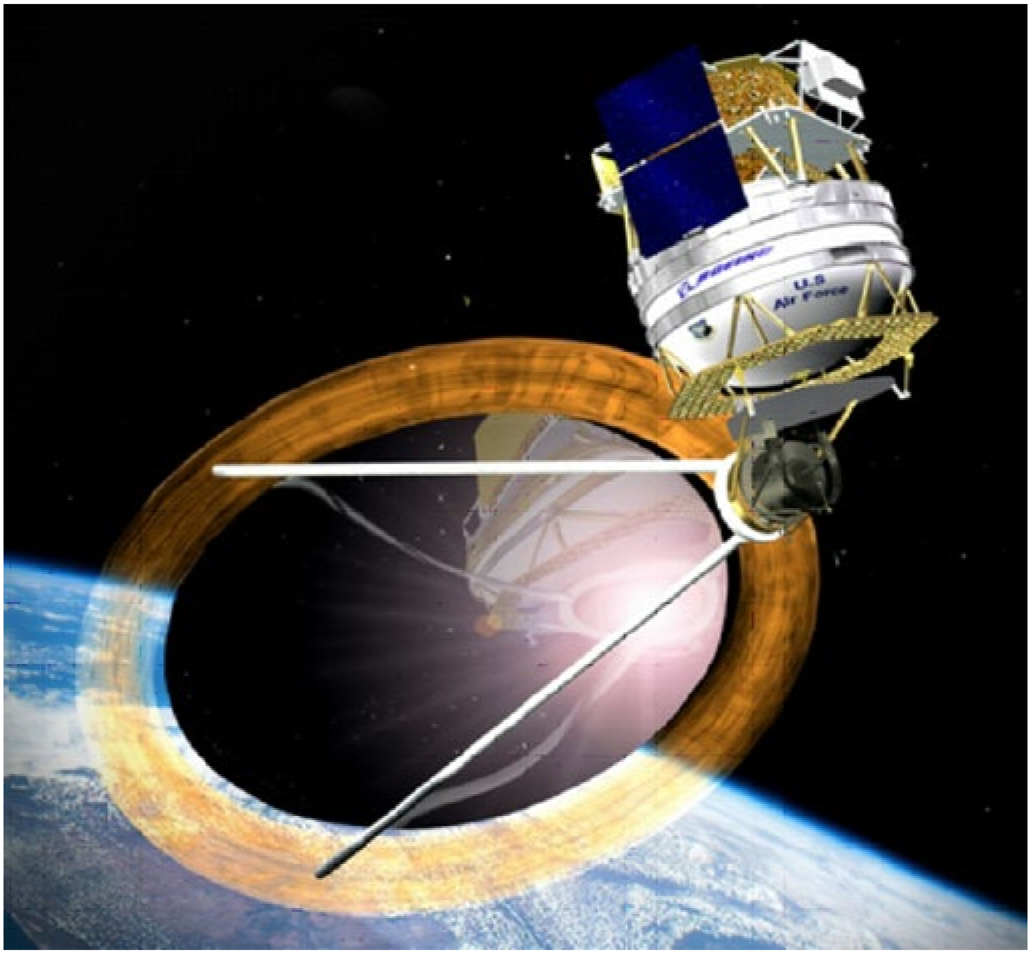

Figure 37: Artist's impression of a steam rocket designed with funding from the US Air Force (Nakamura)

The ISP of a steam rocket depends strongly upon the exit temperature. Estimates range from 200 to 800 seconds (Bolly), with the latter associated with thermal nuclear propulsion. Steam rockets used by hobbyists tend to range from 20 - 40 seconds

Appendix E - Sintering process

We assume that the solar energy flux at the asteroid is of $F = 1360\ W/m^2$, nominally the same as the Earth. Assuming a $\varepsilon = 70\%$ efficient lens but with no loss during heating, power can be obtained by $P = \varepsilon FA$ where A is the mirror's effective cross section.

For calculation's sake, we assume regolith to have specific heat similar to sand, that is $c_v = 900$ J/kg K. We also assume that the post-crushing regolith temperature is around 0 ºC and that sintering temperature is T = 1100 ºC. In this simplified model, we can relate P to the produced mass rate by $P = (\Delta m/\Delta t)\ c_v\ \Delta T$, which gives approximately

$$\Delta m/\Delta t = A*1\ g/s\ = A*60\ g/min$$

for A given in m^2; for a $A = 0.3\ m^2$ mirror, we find $\Delta m/\Delta t = 18$ g/min.

We notice that a higher initial temperature (and thus smaller ΔT) would increase this rate (as the dependence is on 1/ΔT), and that sintered mass goes linearly with area. For a realistic portable 0.3 m^2 mirror, and n samples 2x2x1 cm^3 cubic mass with density 3 g/cm^3 (thus 12n g total mass), one gets about 0.7n minutes for sintering. In a more realistic setting where heat transfer is not ideal, we suppose that this time might exceed 15n minutes.

For testing purposes, we set n = 10 samples with different volumes, but on the average the volume stated above. In that case sintering time is of maximum 2h10min, minimum 1h10min.

This extremely simplified model will greatly profit from future research in sintering processes with sunlight as well as a more realistic description of heat absorption by asteroid regolith-like materials.

Appendix F - Space Launch System by Stage

The SLS is comprised of a core stage, an upper stage, and two boosters. The SLS Block 1B and 2 use an upgraded upper stage compared to Block 1. The SLS Block 2 uses upgraded boosters compared to Blocks 1 and 1B. All three variants use a common core stage.

Stage	**Variant**	**Rocket Engine**	**Thrust (kN)***	**Isp (s)^**	**Burn Time (s)**	**Dry Mass (t)***	**Propellant Mass (t)***
0	1/1B	2x Modified SS SRBs	28024	252.2	128.4	200.8	1263
	2	2x New Composite ATK Boosters	40031	272.5	110	168	1580
1	1/1B/2	4x RS-25D/E	8277	409.1	476	102	979.5
2	1	1x RL-10-B2	110	461.5	1118	3.8	26.9
	1B/2	4x RL-10-C1	425	448.5	1335	15	129

Figure 38: Comparison of the Stages of Space Launch System Blocks

*: Thrust, dry mass, and propellant mass are total values for all engines

^: The Isp values for the 0th and 1st stages are taken as averages between vacuum and sea level values. The Isp value for the 2nd stage is the vacuum value.

Sources: Kyle 2015, Gebhardt 2014, Jones.

Appendix G - Risk matrix label explanation

Below are the risks along with their associated mitigations.

Mining Operations

1. Pyrolysis chamber over pressurizes during heating All chambers contain pressure relief valves, set to 300 kPa (~3 atm). The excess vapor / liquid will vent to vacuum.
2. Dust from mining operations gets into the Orion capsule and causes problems for electronics, breathing etc. A dust shroud will cover all mining operations, and astronaut suits will be electrostatically charged to repel regolith dust.
3. The crusher jams while reducing the mined regolith to dust for the sintering process. The crusher is able to be disassembled for cleaning, some science can still be performed with fewer samples.
4. Large quantities of dust escape through the valve between the pyrolysis chamber and the condensation chamber. The valve is designed to minimize the ability of dust to flow through the valve. The contents of the condensation chamber will be separated out by density using either a centrifuge or a cyclonic separator. Propulsion.
5. Valve failure (on) Redundant valves.
6. Valve failure (off) Redundant feed lines.
7. Reservoir or feed line leakage. Isolation valves.

Astrodynamics

8. Translunar injection maneuver is not fully successful. If the upper stage delivers only a delta-v of 2.92 km/s or lower, the mission cannot be completed (still working on mitigation!)
9. The Lunar flyby maneuver during the outbound trajectory of the crew vehicle is not timed correctly or fails. If Orion's propulsion system is still working, a maneuver can be performed after the failed propulsive lunar flyby to return safely to Earth; TOF is estimated to be 6-10 days; LOM.
10. A subsystem such as ECLSS has a partial failure right after TLI and the crew is required to be back at Earth as soon as possible. if failure occurs within the first 3-4 days of TLI, a delta-v can be performed to change the outbound trajectory to a free-return trajectory. Estimated TOF from TLI to Earth reentry: 10-11 days. If a failure occurs after 3-4 days from TLI, a delta-v can be performed at the lunar flyby to return to Earth safely without exceeding Orion's reentry velocity capability.
11. Docking failure (crew mission). If one or more docking ports are damaged, a direct DRO - LEO trajectory can be used to safely return to Earth. If no critical subsystems are damaged and enough delta-v is available, retry the docking maneuver; this may result in a reduced time for scientific exploration of the asteroid.

Communications

12. Main communications system may fail. Redundancy.

Launch

13. Falcon Heavy launch fails. Ensure the launch vehicle has heritage.
14. SLS launch fails. Test systems extensively.
15. Poor weather. Reschedule launch date.
16. SLS launch capacity reduced. Margin.
17. Falcon Heavy launch capability reduced. Margin.

EPS

18. One solar panel on science habitat fails Margin, redundancy and operation on lower power level.
19. Both solar panels on science habitat fail Attempt to operate on primary and secondary batteries, otherwise LOM.
20. Power supply of ARM fails. Reduce science experiments, omit effected operations.

TCS

21. Eclipse by moon or Earth Include at least one layer of MLI to ensure thermal inertia. Include heating device.
22. Eclipse by asteroid. Dock in such a way that spacecraft is not eclipsed. Include heating device.
23. Coating absorptivity or emissivity will degrade due to solar radiation, galactic cosmic rays. Include heating device and auxiliary radiator.
24. Heater/Radiator fails. Redundancy.

ECLSS

25. EVA Suit failure. Testing and redundancy.
26. IVA Suit failure. Testing and redundancy.
27. Misuse of EVA tools. Thorough training.
28. Loss of cabin pressure. Wear suit and abort mission, gas for suits suffices until return to earth. Structure
29. Inflatable habitat is punctured and cannot hold atmosphere. Perform EVAs to retrieve some of the supplies and science equipment. Shorten mission duration.

Science

30. Not enough water present on the asteroid. Baking power can be increased, which involves risks and consumes more energy, but more material can be processed. A second pyrolysis with increased baking power can be done during EV3 if necessary.
31. Water extraction system non-functional run experiments requiring water (e.g. the experimental garden) with spare ECLSS water.
32. APXS/XRD/GC-MS dysfunctional. Up to 100 kg of samples are returned to Earth for analysis.

Appendix H

```
%-------------- Link Budget -----------------------------------------%
clear all
%-------------- Constants -------------------------------------------%

F_up=7.5e9;      %Frequency up in Hz
F_dw=8e9;        %Frequency down in Hz

P_t_GS=200;        %Ground station power in W
P_t_sat=20;        %Satellite power in W

T=0;             %Noise temperature in K (neglected)
AU=150e9;          %Astronomical unit

Range=384e6+61.5e6;  %Maximal distance between spacecraft & earth
                  %(far side of the moon)

R=12e6;            %Data rate in bps  (HD Video streaminng)
```

```
RdB=10*log10(R);
c=3e8;             %Light speed in m/s


L_a=-3;            %Atmospheric losses in dB
L_r=-3;            %Rain losse in dB

L_m=-6;            %Other losses in dB
D_GS=15;           %Ground station dish diameter in m

D_sat=.1;          %Satellite dish diameter in m
epsilon_sat=.55;   %Antenna efficiency spacecraft
epsilon_GS=.65;    %Antenna efficiency Ground station


lambda_up=c/F_up;
lambda_dw=c/F_dw;
kdB=-228.6;
TdB=10*log10(T);

%Calculate line losses
L_p_up_dB=10*log10((lambda_up/4/pi/Range)^2);
L_p_dw_dB=10*log10((lambda_dw/4/pi/Range)^2);
L_p_dw=10^(L_p_dw_dB/10);

%EIRP_up
G_t1_dB=10*log10(epsilon_GS*(pi*D_GS/lambda_up)^2);
P_t_GS_dB=10*log10(P_t_GS);
EIRP_up=G_t1_dB+P_t_GS_dB

%EIRP_down
P_t_sat_dB=10*log10(P_t_sat);
G_t_sat_dB=10*log10(epsilon_sat*(pi*D_sat/lambda_dw)^2);
EIRP_dw=G_t_sat_dB+P_t_sat_dB

%Reception Satellite
G_R_sat_dB=10*log10(epsilon_sat*pi^2*D_sat^2/lambda_up^2)

%Reception Ground station
G_R_GS_dB=10*log10(epsilon_GS*pi^2*D_GS^2/lambda_dw^2)


%------------- Uplink -----------------------------------%

EbN0_up=EIRP_up+L_p_up_dB+L_a+L_r+L_m+G_R_sat_dB-kdB-RdB

%------------- Downlink -----------------------------------%

EbN0_dw=EIRP_dw+L_p_dw_dB+L_a+L_r+L_m+G_R_GS_dB-kdB-RdB
```