DELPHI: A Lunar Architecture to Enable Exploration, Research, and Commercial Development of Space Beyond LEO

Final Report

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1 Introduction

Humanity has not set foot on a planetary body other than Earth since Apollo 17 in 1972. At the end of the space race, the National Aeronautics and Space Administration (NASA) turned its attention to low earth orbit (LEO) utilization via the Apollo-Soyuz test project, Skylab, the Space Shuttle program, and the International Space Station. With the development of the Space Launch System (SLS) and Orion, NASA is now committed to explore outside of LEO although the immediate destination is still unclear and a matter of controversy [29].

NASA declared the launch of EFT-1 in December 2014 as a first step in the journey to Mars, which is almost inarguably the ultimate goal for human space exploration in the coming decades. NASA's exploration roadmap includes a manned asteroid redirect mission in 2025 as a stepping stone towards a manned Mars mission in the 2030s [20]. The moon is conspicuously absent in this architecture. Although it is not strictly necessary as a stepping stone to reach Mars, the moon is widely considered to provide advantages to a Mars mission architecture in terms of safety and cost [29]. The major safety advantage includes using the moon as a low-risk technology testbed to demonstrate mass-savings for the environmental control and life support system (ECLSS), in-situ resource utilization (ISRU), and propulsion technology required for Mars missions, as well as to build long-term operational experience on extraterrestrial bodies. The moon is considered a low-risk alternative to Mars based on a shorter transit time with reduced radiation exposure, and unrestricted launch windows. The major cost advantage is the potential to utilize ISRU for fuel production and orbital fuel depots to enable significant reductions in Earth launch mass for later Mars exploration missions. Aside from Mars preparation, a moon-first architecture also opens opportunities to the commercial space sector as well as providing planetary and astronomical research benefits.

For these reasons, the Distant Expandable Lunar Permanent Habitation Initiative (DELPHI) has elected to focus on a moon-first architecture. Specifically, **DELPHI seeks to design and validate a mission architecture that enables continuous occupation by 2034 of an eight-person lunar habitat with a rotating crew and minimal resupply to support exploration, research, and commercial development of space beyond LEO.**

2 Functional Objectives and Program Phases

The DELPHI team utilized a functional decomposition method to derive functional objectives from the mission statement and then followed a flowdown process to arrive at specific, quantitative requirements. Ground rules from the RASC-AL committee, as well as assumptions from the DELPHI team, informed the process. To satisfy the mission requirements, DELPHI iterated through multiple architectures and considered multiple design approaches for each component by utilizing formal trade studies. A system of systems approach was stressed throughout the design with greater attention placed on system-level specifications and interactions and less attention placed on component-level design.

Functional Objectives:

1. Safely transport cargo between the Earth's surface and the lunar habitat.
2. Safely transport crew between the Earth's surface and the lunar habitat.
3. Construct the lunar habitat and supporting infrastructure.
4. Maintain and upgrade the lunar habitat to achieve sustainability (resupply mass < 10 t/yr)
5. Conduct projects to support research, exploration, and commercial development.

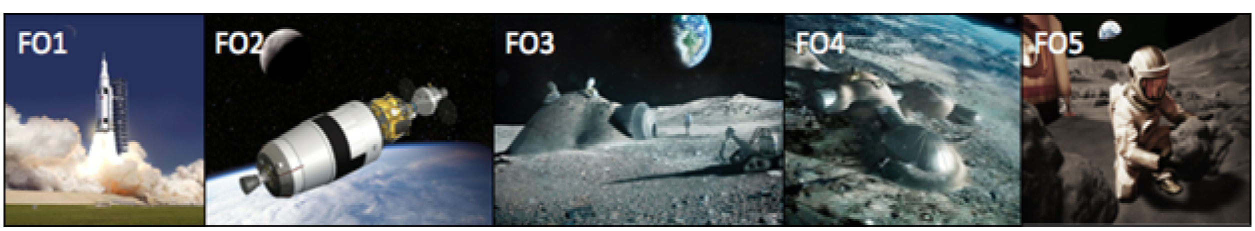

Figure 1: Pictorial representation of DELPHI functional objectives (images courtesy NASA)

To accomplish these objectives, the DELPHI program is divided into phases as follows:

- Development Phase (2015-2022): Technology research, system design, mission planning, and operation training
- Precursor Phase (2022-2024): Uncrewed reconnaissance and preparation missions
- Build Up Phase (2024-2030): Construction and operation of the habitat and sustainable technology testbeds
- Sustainable Test Phase (2030-2034): Implementation and characterization of full-scale sustainable technology
- Sustainable Phase (2034 onwards): Utilization of architecture to support exploration, commercial, and research aims

3 Transportation Architecture

3.1 Surface Habitat Location

Two primary locations were considered to establish the DELPHI lunar surface base: the lunar equator and the lunar south pole. Comparatively, it is easier to land larger payloads at the equator due to the lack of plane changes but it also introduces prolonged night periods. The lunar south pole provides access to Shackleton Crater which contains volatile compounds (including water ice and trace amounts of hydrocarbons distributed throughout the crater) and the potential for continuous solar power. Additionally, multi-base concepts were explored. Ultimately, the south pole was selected as the singular habitat location given that the selected transportation architecture eliminated the need for plane changes. Furthermore, the south pole provides many different geologic features that can be targeted for exploration.

3.2 Concept of Operations

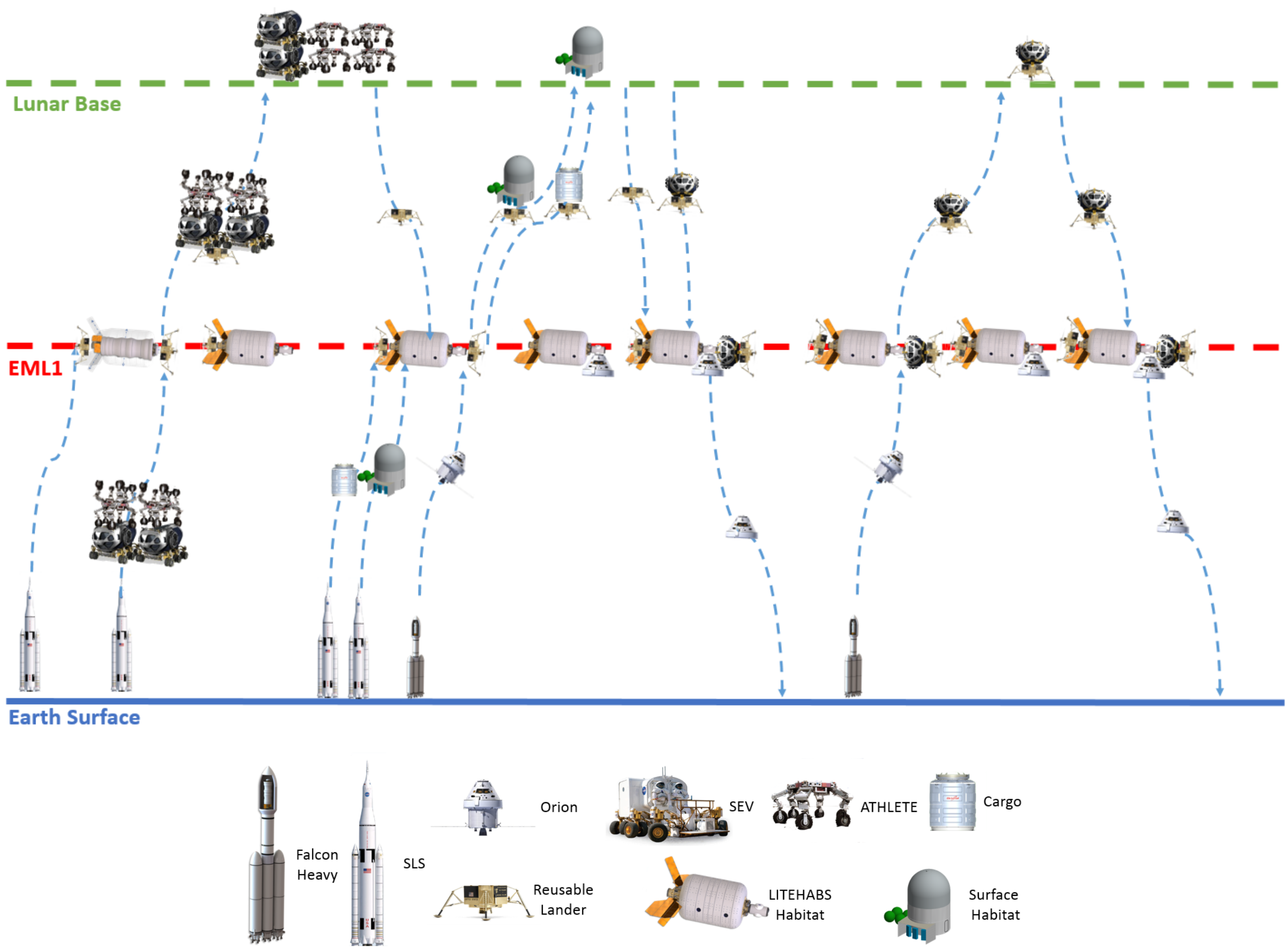

Figure 2: Simplified concept of operations for the DELPHI program

The DELPHI program has iterated through multiple architectures throughout initial design. The overall architectures used in the Apollo program and proposed in the Constellation program were examined in detail. Additionally, less traditional elements such as fuel depots, unpressurized crew landers, LLO and EML1 bases, ISS repurposing, reusable landers, methane propulsion, ion propulsion, and even the controversial EmDrive were considered for integration into the overall architecture. The selected architecture shown in Figure 2 sought to balance several factors:

- Cost (implementation of multi-use and reusable systems to decrease development and manufacturing costs)
- Safety (mission abort and rescue scenarios, system redundancy and simplicity)
- Utilization of existing NASA systems and investments including SLS and Orion in particular
- Overall payload capability to the lunar surface
- High TRLs for components (preferably TRL > 5 but TRLs down to 3 are acceptable with justification)
- Potential to enable future exploration, research, and commercial activity beyond LEO

The selected architecture leverages a cis-lunar waypoint placed into a halo orbit about EML1. This waypoint includes a 4-crew habitat, propellant depot, two reusable Oracle landers, and robotics to assist in docking and spacecraft servicing. Both crew and cargo payloads destined for the lunar surface pass through this waypoint. The SLS Block IB and Falcon Heavy are both utilized depending on program phase and payload type. Once payloads reach the EML1 waypoint, they are transferred to a docked lander and the lander is fueled. For crewed transfers, the Orion remains at the waypoint and the crew transfers into two SEVs mounted to the lander. Until 2030, propellant must be sent from Earth for landing and descent. After 2030, lunar ISRU is utilized to provide complete cryogenic liquid oxygen provision for fueling. Once payloads are attached and the lander is fueled, the lander transfers from the EML1 halo orbit and then lands at the lunar South Pole. One or more ATHLETE rovers are used to unload payload from the lander. Depending on program phase, the lander brings enough fuel for ascent or is refueled on the surface. Once the lander ascends and returns to the waypoint, excess propellant can be transferred to the propellant depot. For crew return, the crew transfers from the two SEVs back into Orion and Orion's propulsion system is used to return to Earth.

3.3 EML1 Habitat and Fuel Depot Specifications

As just described, the EML1 waypoint includes a habitat and propellant depot. The design of the habitat is loosely based on the LITEHABS mission developed by CU Boulder for the 2014 RASC-AL competition [32]. The habitat is located at the EML1 point and consists out of a 14 m long and 4.5 m diameter core cylinder that can be inflated to 17 m length and 7.6 m diameter. The habitat incorporates several mass saving technologies such as lightweight IsoTruss structures coupled with inflatable structures, tethered extra vehicular activity (EVA) capabilities for spacecraft maintenance and a highly regenerable life support system using algal-based water walls. Together with a low energy trajectory it allowed a 30% mass decrease over traditional approaches. The inner core structure provides a safe haven with emergency ECLSS as well as water-based radiation protection. These aspects perfectly suit the habitat's purpose as a lunar waypoint. Upgrades over the previously proposed design include the addition of two docking ports (providing 4 total for landers, crew, and cargo spacecraft) and additional robotics similar to Canadarm2 and Dextre onboard ISS. The LITEHABS habitat is capable of supporting up to four crew members for 30 days. The upgraded habitat is launched in 2022 onboard an SLS along with two reusable Oracle landers.

A propellant depot is established to provide a method of storing propellant between landing/ascent cycles and to transfer propellant to visiting vehicles such as the Oracle lander or other spacecraft such as Mars transit vehicles or fuel tankers. The propellant depot is only intended for cryogenic oxygen storage given the difficulties in collecting large amounts of hydrogen and methane on the lunar surface. Oxygen constitutes 85% of propellant mass for hydrogen-fueled rockets and 66% of propellant mass for methane-fueled rockets. Additionally, oxygen does not experience nearly as much leakage and boil-off as hydrogen. Still, leakage is a problem that should be mitigated to maximize long-term efficiency and support additional propellant types in the future. Several concepts exist for zero boil-off (ZBO) propellant depots utilizing active cooling systems and sun shields [33, 34] and NASA has successfully demonstrated ZBO hydrogen storage systems in ground tests [35]. By 2030, ZBO technology is matured to TRL 7 and a full-scale propellant depot launches to the EML1 habitat. The volume of the propellant depot would ideally be sized to provide a complete oxygen fill to a visiting Mars transit vehicle.

3.4 Oracle Lander Specifications

In early design stages, Oracle resembled a scaled up Altair lander from NASA's defunct Constellation program with a maximum payload of 25 t. The design mirrored the Apollo lander configuration with a two-stage disposable design. Through multiple design iterations targeted at reducing fuel resupply mass by leveraging reusable systems and ISRU fuel production, the current Oracle lander design has evolved to a single stage reusable lander with a maximum payload of 17.1 t. Oracle implements avionics and autonomous landing methods based on the Morpheus lunar lander demonstration project [38] and utilizes the evolved Common Extensible Cryogenic Engine (CECE) RL10 currently in development by Aerojet Rocketdyne with support by NASA [36, 37]. The current version of the CECE includes the ability for deep throttling for landing applications and reusability (50 restarts or 10,000 s of cumulative operation). Given the roadmaps discussed later in this report, it is expected that each lander will be able to perform up to 8 landings before the engine must be serviced or replaced at the EML1 habitat; the engine constitutes just 250 kg of the 6000 kg Oracle lander. The lander is thin enough to fit within the SLS shroud and short enough that its payload deck can be reached with an ATHLETE. The lander supports both cargo and crew payloads. Additionally, the engine is expected to be capable of being evolved to use methane to enable effective utilization of atmospheric ISRU on Mars; this introduces the option of retrofitting the Oracle with a methane-based CECE and using the moon to perform a high-fidelity landing demonstration with the engine before committing it to costly, high-risk Mars missions.

3.5 Communications Architecture

A reliable communications architecture that allows for two-way communication between the South Pole lunar base and the Earth is required. Navigation is also required for autonomous vehicle control on the basis that Earth-based autonomous mining systems use GPS as a standard [12]. Options considered include a single satellite in a halo orbit at either of the Earth-Moon Lagrange point (EML1 or EML2), two satellites in a polar lunar orbit, or a constellation of satellites about the Moon. While there are cost-effective low-energy transfers to place a satellite at EML1 or EML2, this configuration does not allow for any redundancy as it involves only a single satellite. Two satellites in a polar Lunar orbit phased 180° apart with apoapsis located at the South Pole would allow for continuous coverage of the lunar base, with some redundancy in the second satellite. A single failure of one of the paired satellites would result in periodic communication blackouts that could endanger the crew in an emergency. Additionally, this configuration only satisfies the two-way communications requirement and is insufficient for navigation. A constellation of satellites creates the most redundancy in the system, offers constant coverage of the South Pole, and can also provide navigation and science capabilities. A constellation of small satellites with a mothercraft to act as a relay between the small sats and Earth is selected for the communications architecture.

The constellation configuration selected is a series of frozen orbits developed by Folta and Quinn [6]. Frozen orbits eliminate mean drift in the eccentricity and argument of periapsis; therefore, reducing fuel costs for orbit maintenance. The constellation has a total of eight satellites in four inertial orbits, with two satellites per orbit. The constellation satellites will be small satellites that will be deployed from a mothercraft. The mothercraft with the eight small sats and deployment systems are expected to be around 600 kg and will be launched on a Delta II heavy into a geosynchronous transfer orbit (GTO). Once at the Moon, the Mothercraft will make all the maneuvers in the deployment phase, reserving the fuel in the small sats for station keeping maneuvers. Once deployment is complete, the mothercraft will move into a long-term communications orbits to act as a relay between the constellation satellites and Earth. The mothercraft will utilize laser communications. The mothercraft orbit is a small, nearly circular, frozen polar orbit, also developed by Folta and Quinn [6]. The final constellation and mothercraft configuration are shown in Figure 3 with their orbital elements given in Table 8 in Appendix A. The relative orbits created by the constellation are shown in Figure 4 highlighting the total coverage of the lunar surface and the full coverage of the South Pole.

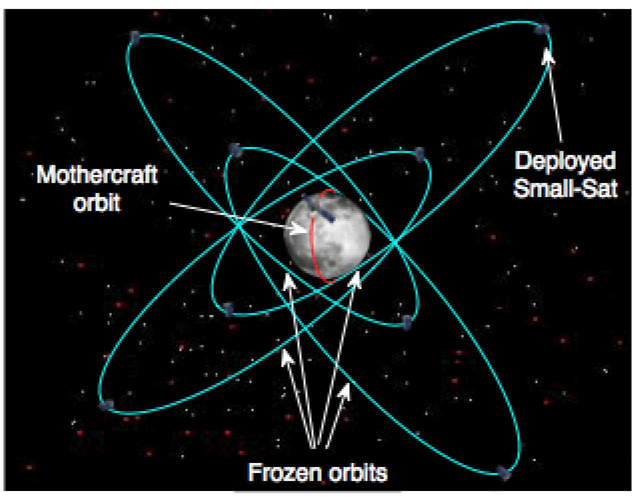

Figure 3: Communication constellation

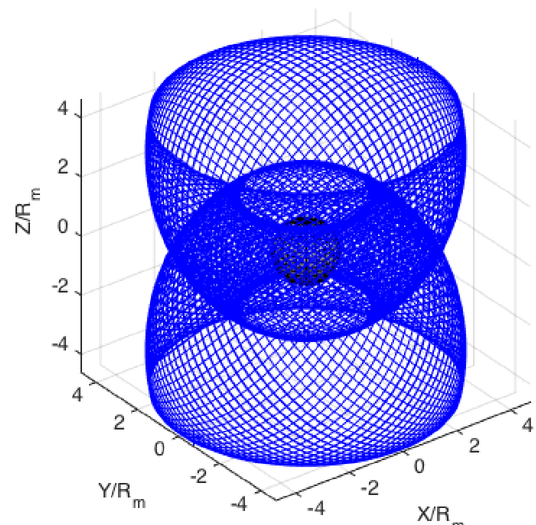

Figure 4: Constellation relative orbits

This configuration allows for global coverage, which not only enables communication between the South Pole lunar base and Earth, but also allows for GPS-like capabilities. Positioning solutions require visibility of at least four satellites: three for location and one for time. On the Moon, astronauts could carry an atomic clock and eliminate the need for the fourth satellites, making positioning solutions feasible even with only eight satellites in the constellation. The global coverage can also be utilized for science purposes. Mapping of the lunar surface or atmosphere with the coverage achievable by an eight satellite constellation could build upon successful science missions like GRAIL and LADEE or study additional features of the lunar environment [7,8].

The small sats will have an initial mass of 10 kg, with a dry mass of 2 kg. The propulsion systems will be small pulsed plasma thrusters built by Busek with a mass of 0.55 kg and a TRL of 5. The thruster specifications are given in Table 9 [9]. A ISIS VHF downlink/UHF uplink full duplex transceiver and deployable antenna from CubesatShop (Table 10 in Appendix A) are used with a combined mass of 0.185 kg [10,11]. This allows for 1.265 kg of any additional required hardware.

Numerical simulations including the point mass effects of the Earth and Sun, as well as the 50x50 gravity field of the Moon were performed to predict the constellation lifetime. With orbit maintenance maneuvers performed every 10 days, the 8 kg of fuel allows for a 5.2 year lifetime of each satellite. Every five years the satellites would need to be replaced.

3.6 Trajectory Analysis

3.6.1 Mission Design Roadmap

DELPHI's mission design makes use of an Earth-Moon Lagrange Point 1 (EML1) Halo orbit propellant depot as a staging ground for both crewed and cargo transfers to the moon. Crewed missions will rendezvous with the Halo orbit using a direct transfer while abiotic cargo will travel along a low energy trajectory, or Ballistic Lunar Transfer (BLT), in order to leverage an increased payload potential to the moon. The crewed transfer architecture was optimized in the framework of the Earth-Moon Circular Restricted Three-Body Problem (EM CRTBP), while the low energy architecture was constructed using patched CRTBP dynamics. Figures of resulting trajectories may be found in Appendix C.

The nominal EML1 Halo orbit was constructed using a Variable Time Single Shooting (VTSS) algorithm, as was the Sun-Earth L2 (SEL2) Halo orbit used in the low energy trajectory design. The total ΔV required to rendezvous with the chosen EML1 Halo orbit from a 400 km altitude Low Earth Orbit (LEO) is found to be 3.587 km/s while that of the BLT is approximately 3.200 km/s. Once a given spacecraft has made rendezvous with the Halo orbit station, a lander will deliver the desired payload to the South Pole Aitken Basin using approximately 2.655 km/s of ΔV. The lander may then ascend alone or with a new payload to EML1. Appendix C outlines the full detailed analysis for the two respective LEO-to-Halo transfer methods along with the landing and ascent scheme. The results of these analyses are summarized below. All analysis was performed in MATLAB with trajectories integrated numerically using ode113 and a user-written derivative function for the equations of motion.

3.6.2 Landing/Ascent Trajectories and Overall Mass Analysis

The low energy trajectory architecture for cargo enables a significant payload savings the EML1 Halo orbit. Using an SLS upper stage with I_{sp} = 448.5 seconds (based on the RL10 engine), and a wet mass of 105 tons with structural mass equivalent to 10% of requisite fuel mass, the following comparison is made.

Table 1: TLI to EML1 transfers

Maneuver	TLI ΔV (km/s)	Halo Rendezvous ΔV (km/s)	Total Fuel Required (t)	Payload to EML1 (t)
Crewed Transfer	3.074	0.513	58.533	40.614
Cargo Transfer	3.200	N/A	54.263	45.311

Once in the Halo orbit, a given lander deploys and delivers payload to the lunar surface. The landing/ascent trajectories are each modelled using three impulse burns, one of which is a significant course correction to target the South Pole Aitken Basin. in the EM CRTBP. The landing itself would in reality be a burn of significant duration. The lander selected will have a structural mass of 6 t tons using the CECE hydrogen engine with I_{sp} = 448.5.

Table 2: Landing and ascent maneuvers

Trip	Halo Depart/Rendezvous ΔV (km/s)	Course Correction ΔV (km/s)	Land/Launch ΔV (km/s)
Landing	0.219	0.752	1.679
Ascent	0.220	0.766	1.670

With a lander in wait for fuel and payload at EML1, the low energy trajectory facilitates significant savings allowing upwards of 18% more surface payload for missions which deploy cargo to the moon where a lander then returns alone to EML1. Identical savings are found for 'land/swap/return' missions in which cargo of a given mass is delivered to the surface via EML1, swapped with an identical mass, and returned with a dry lander to EML1. The table below depicts the maximum permissible masses for the two transfer types using our current architecture.

Table 3: EML1 to surface transfers

Transfer Type	Payload to/from surface (t)	Fuel Required to/from surface (t)	Payload to surface only (t)	Fuel Required to surface only (t)
Crewed	7.959	32.655	14.553	26.061
Cargo	9.365	35.944	17.125	28.186

3.7 Surface Transportation and Construction

When determining the methods for landing and construction architecture, several main factors were taken into account. Incremental buildup of infrastructure is employed to reduce both programmatic risks as well as physical risk to astronauts. For base construction, trade studies were conducted to determine the level of robotic autonomy: fully-automated, fully-teleoperated, teleoperated with on-site astronaut assistance, and fully-manual on-site construction. A trade study of transportation options showed that a pressurized rover that could transport astronauts from the landing craft to the habitat without the need to conduct an EVA was the best option.

The crewed landing architecture utilizes the modular capability of the SEV. Crew will land on the lunar surface using two side-by-side SEV pressurized modules mounted to an Oracle lander via a mechanical interface on the lander's payload deck. The ATHLETE rover can unmount and lower each SEV onto an SEV wheelbase [21]. From the landing site, the SEV can then transport the astronauts to the habitat site without requiring an EVA. Each SEV is capable of supporting up to two astronauts for 2 weeks [26] and will be adequate for the transfer between the lunar surface and orbiting habitat. In the event that one SEV fails during transit, the other SEV can hold all four crew members in an emergency. A hatch on the roof of the SEV enables docking to the EML1 waypoint station for astronaut transfer.

The SEV is intended as the primary surface transportation vehicle for most operations. The SEV also serves to provide additional living area when docked to the habitat and was provides a water-wall based radiation shelter for crew members during solar flares and solar particle events [48]. An alternative mode of transportation for long-distance and long-durations excursions away from the habitat is enabled via the pressurized excursion module (PEM) architecture [46]. In this scenario, astronauts can transfer to the PEM and detach from the main habitat. The PEM can be lifted by an ATHLETE rover and transported in a caravan along with SEVs and other vehicles and supplies. The PEM is capable of providing longer duration physicochemical life support than an SEV provides and can carry more research supplies. When not in transit, the PEM can remain docked to the main habitat and be used for additional living space.

ATHLETE is an integral part of the DELPHI landing and construction architecture, and will be on the very first missions sent to the lunar surface. These vehicles are responsible for unloading all cargo sent from Earth to the lunar surface and can position different mission elements in preparation for human landings such as habitat structures and SEVs. Once astronauts arrive, they can make use of ATHLETE, SEV tool attachments, and space suits on EVA to complete any assembly necessary to make the habitat and supporting infrastructure a working system.

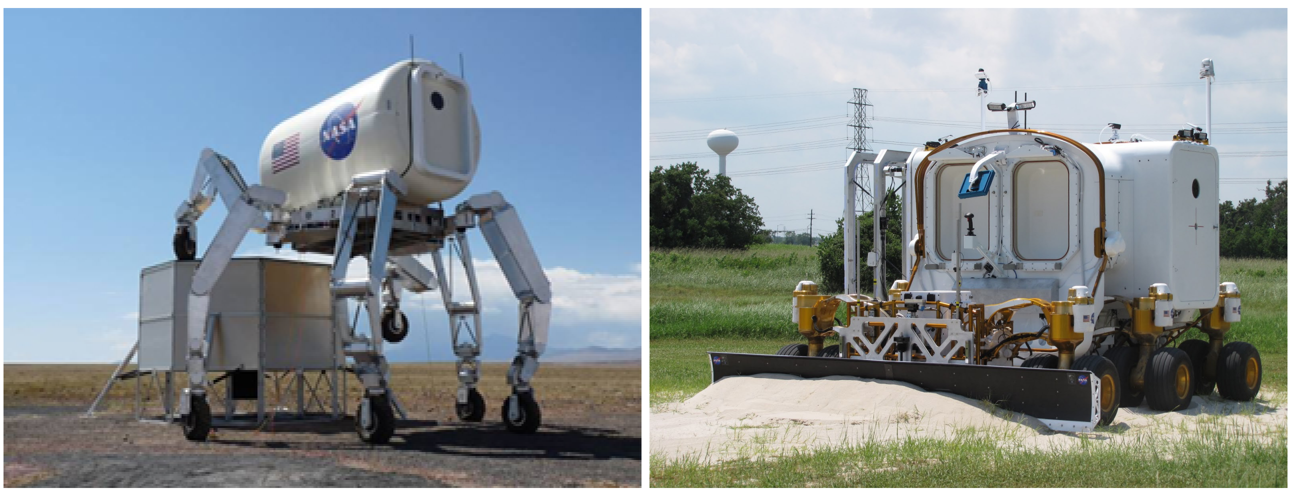

Figure 5: (Left) ATHLETE unloading cargo from simulated lander
(Right) SEV with bulldozer attachment (image credit NASA)

3.8 Abort Options

DELPHI utilizes Orion, SLS, and Falcon Heavy for crew launches. Therefore, the existing launch abort system (LAS) on Orion and in-flight abort scenarios are used for launch aborts. Another abort opportunity occurs in a LEO parking orbit to assess any risks and take immediate action before TLI. Aborts while in TLI exist but will take several days to return astronauts to Earth. The crewed trajectory architecture facilitates a free return trajectory allowing the spacecraft to bypass Halo orbit rendezvous and return to Earth within 5 days should the engines fail to fire at the rendezvous point.

Should the engines fail at the time of course correction onto the landing trajectory (3 days after Halo departure), the spacecraft will not impact the moon but rather enter a lunar escape path which remains in the vicinity between the moon and L1 for two days during which new landing maneuvers or Halo rendezvous may be calculated.

In the event of engine failure at the course correction maneuver following the main ascent burn, the spacecraft would enter a trajectory that impacts within 15 degrees latitude of the north pole of the moon within approximately 30 minutes, though the lunar J2 perturbation may alter this impact time. This trajectory however may be easily adjusted to allow a harmless ballistic insertion into a polar Low Lunar Orbit (LLO); the previously mentioned is merely the worst case scenario. If instead the engine fails upon the Halo orbit rendezvous burn 3 days after ascent, the spacecraft will return to the vicinity over the moon and have 2 days to calculate a safe return to the lunar surface or to a different portion of the Halo orbit before the spacecraft travels unsalvageably far from the moon and earth.

Crewed descent and landing in the near vicinity of the moon are not abortable. With the present design, engine failure during these phases results in loss of crew. There are two mitigation strategies that can be implemented to reduce the probability of engine failure. Firstly, an engine cycling scheme in which crewed landings always use fresh but flight-tested engines. Untested and older engines are relegated to cargo landings. Secondly, the addition of a redundant CECE engine to the lander design would increase lander mass from 6000 kg to at least 6250 kg but enable the lander to cope with a single-engine failure.

The last abort scenario addressed occurs when the crew is in the habitat on the lunar surface. Precautions are taken to ensure that there are enough ascent vehicles to transport every astronaut into lunar orbit. Similarly, the SEV has the capability to act as a lifeboat to transport astronauts from the habitat to ascent vehicles without returning to the base. The SEV would provide up to 2 weeks of life support.

4 Surface Habitat Design

4.1 Module Design and Organization

The standard habitat module is three stories tall, with the first two built within an aluminum cylinder and the third built with an inflatable dome. Figure 6 provides concepts of the habitat and primary features. These features include:

- Each module has a minimum of two doors that allow egress as well as connectivity.
- Up to three suit ports allow for suit operations without bringing dust inside.
- The suits are protected with a Mylar tent to mitigate environmental hazards.
- The various tanks needed for life support are landed separately from the module and connected.
- Eight Standard Payload Racks fit in each floor to support ECLSS, communications, operations, and science.

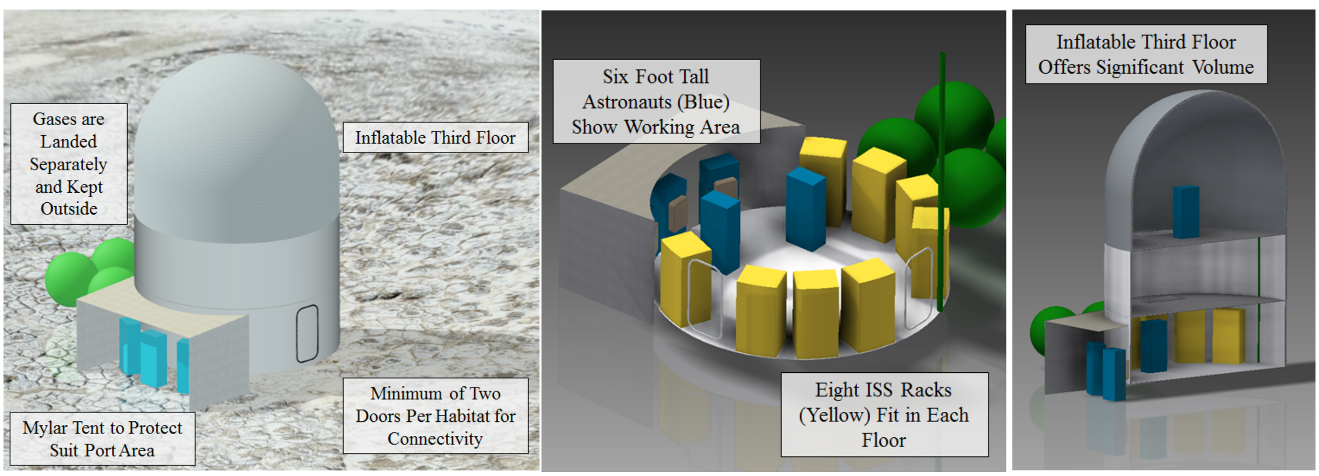

Figure 6: Volumetric Layout of Command Module

The configuration of each module depends on its purpose. For the command module, the first floor hosts the ECLSS hardware. The primary concern is to keep the life support systems close to the spacesuits and the external tanks to minimize connectivity weight and improve efficiency by keeping distances as short as possible. Gases are pumped to the upper floors and fans circulate air. A water source is provided only on the first floor. The kitchen is therefore also designed for the first floor. It is expected that the first floor will be the noisiest and most industrial and therefore will see the least traffic outside of spacesuit operations. The second floor hosts the sleeping quarters and telecommunications equipment. It also has a series of windows for observations built both for operations oversight and astronauts who would like to look out the window during down time. It is expected that most meals will also be enjoyed here with a less noisy environment. The inflatable third floor is a multi-purpose space. Exercise equipment, science payloads, and storage are all options. Different modules have different configurations based on design intent. It is also a useful space for giving the astronauts a feeling of openness. Small windows can be installed to provide additional viewing options. Seven modules types are envisioned for the DELPHI program and are described in Table 11 of Appendix A.

4.2 Structure, Power, and Communications

The structure is primarily aluminum 6061. Titanium 6Al-4V is used where additional strength or thermal isolation is required. The inflatable section is based on technology licensed to Bigelow Aerospace and includes layers of Nomex, woven Kevlar, and inflatable bladders. Regolith supplements radiation mitigation and is an experimental construction process.

The power is provided by triple junction solar cells. An average of 12 kilowatts of power is required for each module, with the greenhouse taking the least and the command module taking the most. The total power required is about 80 kilowatts. The panels are positioned away from the station in an area with optimum sunlight exposure, such as the rim of the Shackleton crater. The panels are designed with drop-in cell technology so that a damaged cell can be replaced easily. Power is stored in lithium-ion batteries for up to 60 hours of nightside operations. Additionally, power can be generated with back-up fuel cell technology.

Communications use a variety of technologies. S-Band and UHF frequencies are the mainstays and serve as the emergency communication channels. For high data rate transmissions, X-Band frequencies are employed. Laser communications directly from the moon to the earth are highly desirable and considered an operational test objective. In all cases, the network of orbiting communication satellites around both the moon and earth assist with transmitting and receiving signals. Although the primary communications path is through the satellite network, the S-Band and UHF channels are capable of back-up two-way communications directly with the earth through boosted power configurations.

4.3 EVA Systems

The DELPHI EVA architecture calls for several main systems. First and foremost when discussing EVA, the suits needed to combine maneuverability and simplicity. Additionally, the use of suit ports were desired to reduce EVA prep time and risk of dust contamination. The most promising option considered was making use of NASA's planned Z-series space suit, which addresses all of the functional requirements set forth by the DELPHI team. One suit for every astronaut will be required, resulting in 8 of these suits.

For transportation across the lunar surface, DELPHI desired a rover that could serve multiple functions. Early in the design process, it was determined that the rover needed to be pressurized, include a hatch for habitat docking, include suit ports, have the ability to be remotely operated, and have different tools to assist in construction or EVA operations. Upon searching for existing concepts, NASA's space exploration vehicle (SEV) perfectly addressed all of these requirements. Additionally, the SEV utilizes a modular design that can detach the pressurized section from the wheel base "Chariot". This presents several advantages including easier maintenance and part replacement as well as the ability to split mass between cargo missions.

In terms of EVA, the mobile command module and ATHLETE rover (both NASA concepts) work together to extend operations of astronauts in an SEV. As discussed earlier, the additional resources provided here allow crew to travel farther from the primary habitat than would otherwise be possible with just SEVs

4.4 ECLSS

The environmental control and life support system (ECLSS) architecture for the DELPHI program implements a hybrid approach of both physicochemical and bioregenerative life support systems. The physicochemical ECLSS serves as the initial life support system with a high level of reliability but also a high mass of consumables. In order to utilize the DELPHI program as a proving ground for Mars missions, the goal is to make the habitat as self-sustaining as possible. Since food cannot be produced with physicochemical systems, DELPHI evolves to use a bioregenerative ECLSS system by the end of the build-up phase. The approach of DELPHI is to iteratively build up the bioregenerative system by continuously decreasing the workload of the physicochemical system. This allow for a smooth transmission between these two systems during the first couple mission. Even though the physicochemical system is no longer needed once the bioregenerative system is fully operational, it will still remain functional for backup purposes. Since this will be the first demonstration of a bioregenerative system in space this approach is crucial for the safety of the crew. Potential risks that are mitigated by this approach include biological cultural collapse of elements of the bioregenerative system due to pathogens as well as long growth periods. The internal atmospheric pressure of the habitat is set at 10.3 psi with 30% oxygen and 70% nitrogen to reduce leakage rate and structural mass requirements for habitat modules. The lower pressure also removes the need for a lengthy prebreathing procedure before EVAs. The systems are described in further detail below.

4.4.1 Physicochemical ECLSS

The main goal of the physicochemical system is reliability. To reduce cost and resources the system uses mostly technology already in use on the International Space Station (ISS). This includes the air revitalization rig including the carbon dioxide removal assembly (CDRA), trace contaminant control system (TCCS), and temperature and humidity control system (THC). The carbon dioxide removed by the CDRA will be used by the oxygen generation assembly (OGA) to reduce it to oxygen. The last system adapted by the DELPHI program is the water recovery system (WRS). As water represents the major consumable mass for the life support system, this system helps to reduce consumable mass even during the buildup phase of the DELPHI habitat. As the initial system represents a physicochemical ECLSS, food is supplied from Earth in a prepackaged form. Whereas the previous mentioned technologies can just be rebuilt from the ISS the thermal control system requires redesign as it is integrated into the structure with no standalone systems. However, the basic principle of heat exchangers that pump heat into an ammonia loop cooled by radiators remains unchanged. Due to the altered location of these systems, different environmental factors have to be taken into consideration. Firstly, the partial gravitational environment is altering fluid dynamics in these systems. All ISS ECLSS systems, however, were demonstrated to be functional in both Earth gravity as well as microgravity, which makes them suitable for operation in partial gravity. The second is performance degradation due to dust coverage on the radiators themselves. In order to prevent this, an electrodynamic dust shield was chosen that is capable of clearing the exposed surfaces from dust. This system is described in greater detail in Section 4.6. The ISS life support system is currently able to provide for a crew size between 3 and 6 [30]. Even though the DELPHI program is planning on ultimately supporting a crew of 8, the ISS ECLSS sizing is still appropriate because the physicochemical system is nominally only in use during initial phases where the habitat only supports 4 crew members [31].

4.4.2 Bioregenerative ECLSS

After the buildup phase, DELPHI will use a bioregenerative ECLSS system in order to reduce the amount of annual resupply required from Earth. The bioregenerative system is designed such that loop closure is approached thus reducing launch cost. A schematic of the bioregenerative ECLSS system is shown in Figure 7. While the technologies used in the bioregenerative system have lower TRLs than the physicochemical technologies, a bioregenerative system ultimately pays for itself in terms of both mass and development cost. Food production, waste water processing and atmospheric management is all provided by an autonomous crop production system. The crop production facility is housed in two separate greenhouses. A variety of crops will be grown with great care to satisfy all nutritional standards. Less than 1% of all water that the crops use will be used to form biomass. The remaining 99% of water taken up by crops is released through transpiration [40]. After initial processing and monitoring to ensure safety to the crops, all waste water is taken up by the crops. The transpired water from the crops is condensed and then used for both potable and hygiene water. Both potable water and hygiene water will be monitored to insure that the water quality is within acceptable standards. The crops remove carbon dioxide from the crewed modules and produce oxygen through photosynthesis [41]. It is important to note that production of carbon dioxide is typically viewed as a disadvantage when conducting a trade study of technologies, by using a crop production facility to provide many of the resources needed to support the crew, carbon dioxide becomes a valuable beneficial resource because of the large amount needed to provide adequate nutrition to the crew. While crops have the ability to remove trace contaminants from the air stream, if air quality is not maintained over extended periods of time, a buildup of toxins can have adverse effects on growth. Therefore, the air stream will flow through a biological air filtration system to remove these toxins before entering the greenhouses [41]. Approximately half of all crop growth is inedible so an aerobic bioreactor is utilized to break down inedible biomatter to carbon dioxide, water, and a sludge. The sludge contains essential nutrients than can fertilize the crops [42]. Fecal matter will be processed using torrefaction, a mild pyrolysis process that uses heat to sterilize and stabilize fecal matter thereby producing carbon dioxide and water as a byproduct [43]. All other waste produced is processed using supercritical water oxidation that functions to produce carbon dioxide, reduce waste volume, and stabilize hazardous materials [44].

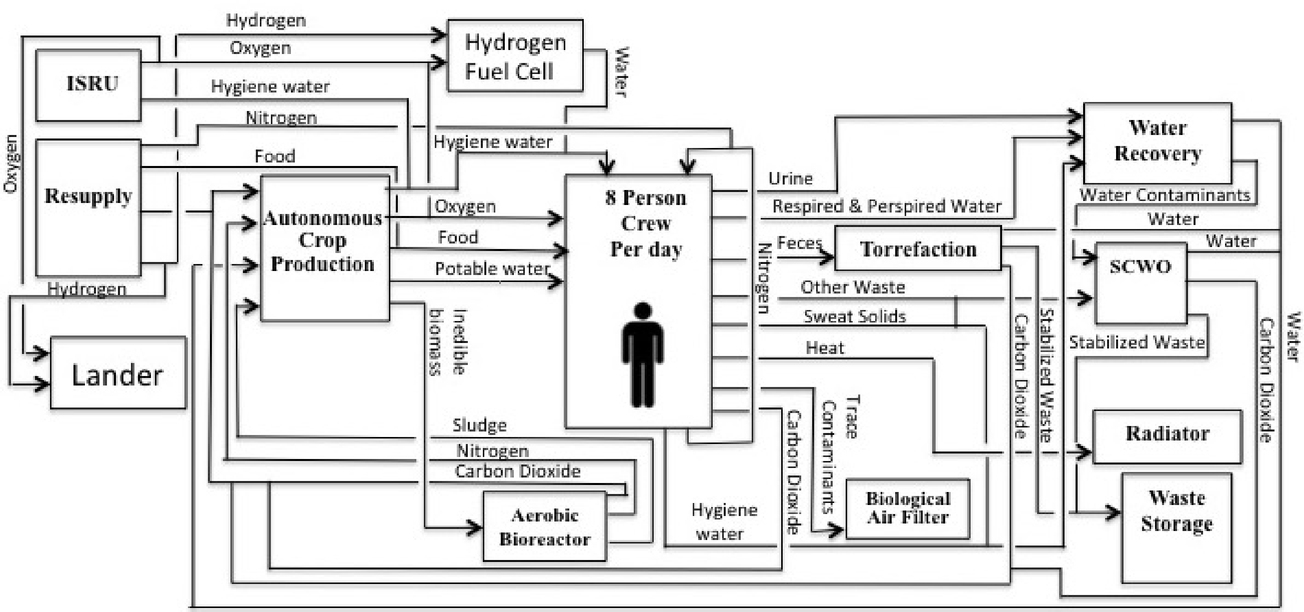

Figure 7: Block Diagram of Bioregenerative ECLSS System.

4.5 ISRU

In-situ resource utilization (ISRU) will be used to take advantage of existing materials on the lunar surface in order to reduce the total mass sent from Earth. The three primary areas that use these resources are ECLSS resupply of consumables, propellant generation for the reusable Oracle lander, and as a construction material. Oxygen is perhaps the most important commodity, as it is important to both of these areas. Hydrogen can also be used as fuel. Carbon exists on the lunar surface in the form of volatiles trapped in permanently shadowed craters. However, the concentration of these hydrocarbons is known to exist in low quantities (<1% of regolith by mass), the exact amount of which is unknown and cannot be relied upon for habitat resupply. Finally, regolith is made up almost entirely of metal oxides [24] which, if refined, can provide a source of material for construction, radiation mitigation, or 3D printing.

DELPHI will employ three major technologies to extract these materials from the lunar surface. First, the carbothermal reduction process is a means of extracting oxygen from the lunar soil. As regolith is made up of metal oxides, oxygen makes up approximately 50% of the lunar soil by weight. The carbothermal reduction process occurs in three steps shown below (with M standing for a generic metal element) and occurs when methane (CH_4) flows over molten regolith consisting of metal oxides, liberating the oxygen atoms and leaving a processed chunk of metal [24].

Step 1: $MO_x + xCH_4 \rightarrow M + xCO + 2xH_2$
Step 2: $xCO + 3xH_2 \rightarrow xCH_4 + xH_2O$
Step 3: $xH_2O \rightarrow xH_2 + 0.5xO_2$
Result: $MO_x \rightarrow M + 0.5O_2$

Carbothermal reduction will be used to resupply the DELPHI habitat with breathable oxygen. Literature cites 1,000 kg/year of oxygen as the design goal for several NASA funded ISRU projects [25]. Orbitec is the recipient of this funding and has demonstrated a prototype oxygen generation unit at Mauna Kea, HI, where the volcanic soil is similar in composition to what would be found on the lunar surface. Working prototypes run off 1kW of energy, provided through a solar energy module [25]. It is assumed that a full scale working unit would be within an order of magnitude of this power specification and have a dedicated solar collector independent of the primary habitat system. The TRL for carbothermal reduction technologies are 6, and could be developed to an operational state for use in the DELPHI habitat from early missions through the lifetime of the lunar outpost.

The second major technology utilized by DELPHI to produce oxygen is molten oxide electrolysis (MOE). In this process, regolith is taken to a completely molten state (as opposed to concentrated melt in carbothermal) and electrolyzed with an anode and cathode material [27]. Oxygen comes directly off the anode to be collected and stored. The remaining liquid metal can be separated and transferred into molds for cooling and later use. Oxygen yields are higher in MOE than carbothermal reduction and is a continuous process instead of batch process. However, TRL for this technology is 3. MOE research is ongoing in laboratory environments (most notably at MIT) and has potential as an industrial process for metal production [29]. To address this gap in technology readiness, DELPHI will employ MOE as a priority technology development and demonstration unit through the early phases of the mission. The final MOE unit is launched in 2029 to aid DELPHI in reaching self-sustainability. The primary function of the final MOE unit will be to produce oxygen for the landing and ascent vehicle, transporting crew and cargo to and from the lunar surface in the form of liquid oxygen. At 25 t of oxygen per refuel and two refuels expected per year, 50 t of oxygen is needed. A 50% yield rate of oxygen from processed lunar regolith means that a total of 100 t of regolith will be processed in one year. In terms of volume, this is approximately equivalent to a 1A series freight container. Liquid hydrogen fuel for the system will be supplied from Earth. Metal byproduct is then used for construction and repair purposes throughout the base.

To move the quantities of regolith needed for carbothermal reduction and MOE, a bulldozer attachment on the SEV will be utilized. Additional attachments are employed as needed to place regolith within hoppers that will feed into the oxygen generation units.

Finally, water will be collected from regolith by a system that utilizes microwaves to heat regolith from the inside to at least a depth of 3 m, thereby sublimating subsurface volatiles (including water ice) that can be collected by a cold plate on the surface [23]. A major benefit of this method of water collection is that there is no need to transport regolith; a rover can collect water without the need for excavation equipment. Shackleton crater is of particular interest for this technology, where the latest information indicates that ice is plentiful but is distributed throughout the lunar soil. Because of these suspected ice deposits, regolith within this crater is thought to approach 10% by weight, as opposed to 1% outside of the permanently shadowed craters [28]. Several NASA funded projects set the goal of water production at 1,000 kg/year [22]. There is still development that needs to go into this technology as its TRL is 4. However, DELPHI does not rely on this technology to keep the base supplied with water during the build-up phase and is not majorly affected by development schedule slip. Water extraction via microwave technology serves to enhance the capabilities of DELPHI and doubles as a research tool to categorize the molecular composition of volatiles in Shackleton crater.

4.6 Dust Mitigation

Lunar regolith dust has been noted by Apollo astronaut John Young as “the number one concern in returning to the moon” due to its detrimental effects to spacecraft systems and crew health [13]. Accumulation can occur through both artificial means (EVA, landings) and natural means (diurnal electrostatic dust levitation phenomena) [14, 15]. DELPHI implements a multipronged approach to prevent dust from accumulating on sensitive habitat, vehicle, and spacecraft surfaces and from coming into direct contact with crew members.

Sensitive hardware surfaces include solar panels, radiators, cameras, viewports, and hatches. The small particle diameter (<50 µm) and radiation induced static charge of lunar dust confounds most known passive and mechanical mitigation approaches such as coatings and brushes as demonstrated through unexpected overheating events during the Apollo missions [15]. For these surfaces, an active approach is selected based on electrodynamic dust shielding technology originally developed at KSC [16, 17, 18]. Electrodynamic dust shields utilize use a thin layer of interdigitated comb electrodes on a surface to electrostatically repel regolith via an alternating electric potential applied to the electrodes. Clearing factors near 100% are achievable with an applied square wave with 10 kV amplitude and 60 Hz frequency. Despite the high voltage, these shields consume minimal power and would only need to be operated for several minutes a day to maintain system performance. The lack of moving parts and ease of automation reduces crew time requirements for system maintenance. This technology has already been successfully integrated with solar panels, radiators, and the habitat demonstration unit in ground-based testing [16, 17, 18].

To demonstrate electrodynamic dust shielding technology, the DELPHI team opted to develop and test a prototype system. A high-voltage square wave function generator with integrated data acquisition was created to power an electrostatic comb shield. JSC-1A lunar simulant was procured for use during testing. Current results using a 5.5 kV 15 Hz square wave are promising but not yet optimal. The prototype system is incredibly effective at clearing small particles but fails to clear larger particles or clumps of particles. The DELPHI team plans to upgrade the system to use a 10 kV square wave before the RASC-AL competition and expects to observe improved performance. The design and evaluation of the current prototype system are further detailed in Appendix B along with videos of the system in action.

To prevent crew member contact with regolith dust, suitports are employed. To prevent spacesuit wear, lotus coatings are also employed to prevent regolith adhesion [47]. Further, a NASA-developed electrostatic dust cleaning wand known as SPARCLE is utilized outside and within dust locks and air locks to provide a method of removing regolith from many material types [19]. Additionally, it is recommended that interior dust locks are utilized along with a directional ventilation system to direct air away from living areas and towards dust locks to prevent any dust accumulation within the habitat. Dust masks and goggles should be provided as PPE in the event of dust contamination within a habitat or vehicle.

4.7 Habitat Mass Analysis

A bottoms-up approach was used to estimate the total amount of mass necessary to be landed on the lunar surface during the build-up phase. The results of the estimation procedure are summarized in Table 4. These estimates includes both permanent infrastructure as well as resupply mass for consumables based on 10 years of operations. The total landing mass requirement is estimated at 231 t. With 25% margin, this is 289 t. A separate heuristic top-down analysis based on habitable volume, crew size, and mission duration provided an estimate of 265 t. Averaging these two values, we expect that a total landing mass of 277 t would be sufficient.

Table 4: Top-down estimate for subsystem mass fractions and total habitat mass

	Structure	Thermal	Power	C3	ECLSS	CA/PA	EVA	ISRU	TOTAL
Mass (t)	93.3	2.12	57.2	0.424	38.2	4.24	14.8	20.9	231
Percent (%)	40.3	0.917	24.8	0.183	16.5	1.83	6.42	9.04	100

4.8 Resupply Mass Analysis

ECLSS consumables typically require the largest fraction of resupply mass. Currently the ISS has to resupply 6.82kg/CM-day [39]. For an eight member crew, that is nearly 20t a year. However, by using bioregenerative ECLSS and recycling resources, as previously discussed, the amount of resupply is greatly decreased, thus leading to a significant cost reduction for the extended life of DELPHI. By optimizing the size of the autonomous crop production facility, the amount of resupply that DELPHI required is significantly reduced. While crop production has the ability to provide nearly all of the crews support, there is a point at which the amount of carbon dioxide, water and nutrients to support crop production outweighs the production of resources from the plants, thus causing an increase in resupply needed to support the entire system. In addition, since DELPHI will be equipped to supply all eight crew members with all needed supplies for 90 days in the event of an emergency, 90 days of pre-packaged food will be available. By using the previous year’s emergency food supply as a supplement to crop production, the total resupply is decreased because the pre-packaged food is not wasted. Through an in-depth analysis, yearly resupply was minimized to 9.02 t of logistics per year, including a 25% margin (see

Figure 8 and Table 5). The crop production facility will be 97 square meters which will provide approximately 61% of the crew's nutritional needs. Therefore, through the use of a bioregenerative ECLSS and ISRU, DELPHI is able to achieve a self-sufficiency with only 9.02 t of logistics resupplied each year. Resupply includes all consumables to support the crew, fuel for descent and ascent from EML1, and materials for research. However, DELPHI is capable of landing 17.1 t to the lunar surface so an additional 8 t per year can be used for extended development.

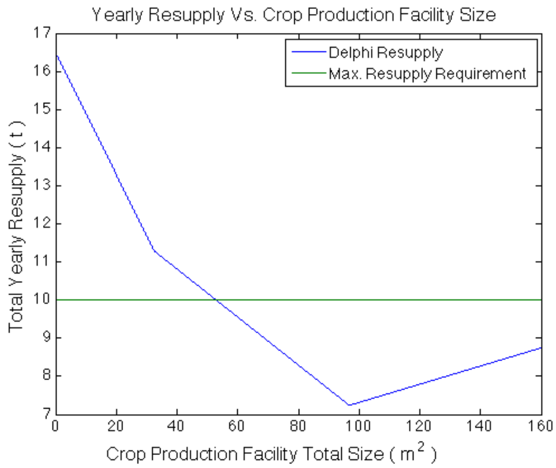

Figure 8: Crop Production Size Optimization

Table 5: Breakdown of Resupply

Logistic	Yearly Resupply Mass (t)
Pre-packaged food (Emergency and Supplementary)	0.95
Nitrogen+tankage	0.14
Carbon Dioxide+tankage	0.05
Hydrogen+tankage	3.77
Research Materials	1.25
Other	1.06
Total	**7.22**
Total +25% Margin	**9.02**

5 Novel Applications to Exploration, Research, and Commerce

There are many applications envisioned for a self-supporting lunar base. DELPHI will be the first permanent human outpost in deep space, and may serve as both a waypoint and base of operations for further solar system exploration, specifically in support of asteroid and Mars missions. Furthermore, the modular design of the habitat means that it can be expanded upon and enhanced as new and innovative technologies emerge. This approach allows the DELPHI system architecture to remain relevant for decades to come, as exploration may reach beyond Mars to destinations such as Europa.

In addition to exploration, research is a primary goal of the DELPHI mission. In particular, testing bioregenerative ECLSS technologies in a space environment is of critical importance for future missions throughout the solar system. Strategically combining these technologies with existing and future spacecraft will reduce resupply needs of permanent Mars settlements and long-term transit vehicles. Another potential use of the DELPHI base is as an astronomy center, supporting new types of deep space telescopes. Finally, educational outreach is an important aspect of the ongoing mission plan and will be conducted in a way similar to the ISS program.

As previously discussed, the DELPHI lunar base can be expanded with new structures and capabilities. While this may be done with NASA funds, it also possible that these expansions may come from the commercial and private industry as well. There are many organizations interested in lunar exploration and the DELPHI mission provides the infrastructure necessary to develop a commercial industry on the Moon. One commonly cited example of a lunar industry is mining for 3He. Allowing commercial companies to synergistically integrate into the existing infrastructure opens up many future possibilities not necessarily accounted for in the DELPHI design process. Commercial companies could also be involved through contracts similar to the Commercial Crew Development (CCDev) Program. Opening up development of architectural elements to contractors allows innovative approaches to be identified and taken. Specifically, a navigation and communications satellite constellation would be very beneficial to have at the Moon, and could be developed under a similar system to CCDev.

Lastly, the fuel production capability and reusable lander concept allow the orbiting fuel depot to be used for both servicing of satellites and as a "gas station" for missions beyond the Earth-Moon system (i.e. near Earth asteroids and Mars). Many geosynchronous satellites reach their end of life only because of fuel depletion. The ability to generate fuel in space and service these satellites provides tremendous benefits, including increased lifespan, reduced orbital debris (fewer satellites), and decreased launch costs. Similarly, any future Mars mission could reduce the launch mass by using the DELPHI fuel depot to gather the necessary fuel for its mission, allowing for more science and cargo payload to reach Mars. In-space production of fuel is a capability of the DELPHI mission that extends far beyond the Moon, and opens up many unique opportunities for the future of space exploration.

6 Schedule and Finance

6.1 Launch Schedule

The launch schedule for DELPHI is shown below in Figure 9. Payloads for each launch vehicle are described in Table 6. The launch schedule was constructed through an ad-hoc iterative process that sought to balance several competing factors such as SLS availability, technology maturation time, first crewed landing, fuel production readiness, and maintaining a steady cadence of launches to prevent budget overruns in any particular year. The schedule enables iterative testing of technology throughout the DELPHI build-up phase. Crew stay durations are gradually increased from one month to two years as medical data is collected to characterize the effect of the lunar environment on human health and operational confidence is established. Altogether, the DELPHI program utilizes 34 launches total throughout the 2-year precursor phase and the 10-year build-up phase placing a total of 91 t of payload at EML1 and landing 370 t of payload onto the lunar surface. This is more than sufficient to cover the 277 t requirement in Section 4.7. If ISRU and fuel depot technologies fail to mature, total payload onto the lunar surface is reduced to 266 t. In that event, a single additional SLS launch would be necessary.

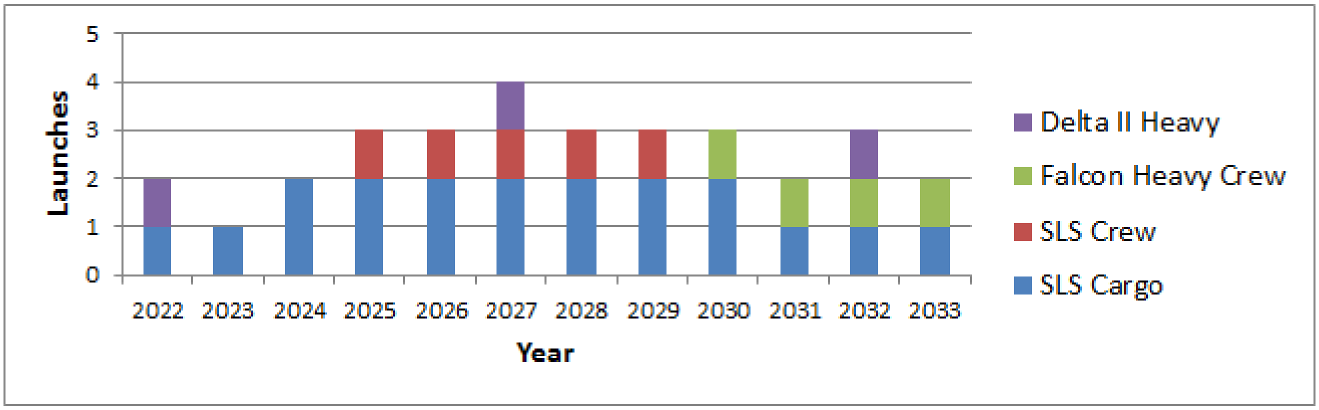

Figure 9: DELPHI Launch Schedule

Table 6: Launch vehicle payloads and destinations

Launch Vehicle	Payload	Destination
SLS Cargo	Variable (see Table 12 in Appendix A)	EML1 Halo Orbit
SLS Crew	Fully-fueled Orion with Oracle landing/ascent fuel	EML1 Halo Orbit
Falcon Heavy Crew	Partially-fueled Orion with no Oracle landing/ascent fuel	EML1 Halo Orbit
Delta II Heavy	Communication satellites	Lunar Orbit

6.2 Technology Development Schedule

Most technologies within the DELPHI architecture are already at TRL 9. Major technological challenges derive from 18 identified technologies with TRLs below 9 as listed in Table 7. The average TRL of these technologies is 5.5 with a minimum TRL of 3. Three technologies are currently below TRL 5 including ZBO fuel depots, molten oxide electrolysis, supercritical water oxidation, and torrefaction fecal processing. Additional time and development resources are allocated to these technologies to help ensure flight readiness before they are needed. None of these three technologies would act as a showstopper if the technology development was delayed or wholly unsuccessful. Four years of margin are provided within the sustainable test phase to allow for testing, troubleshooting, and iteration.

Table 7: Current technology TRLs and projected year to reach TRL 9. TRLs below 5 are marked in red.

Technology	TRL	TRL 9
Suit Port	7	2017
Laser Communication	7	2017
Z-2 Space Suit	7	2018
Pulsed Plasma Propulsion	5	2020
Evolved CECE RL10 Engine	6	2022
Electrodynamic Dust Shielding	6	2022
Mothership Satellite Architecture	6	2022
Inflatable Surface Structures	6	2024
Carbothermic Reduction	6	2024

Technology	TRL	TRL 9
Microwave Water Extraction	6	2024
Astronaut/Rover Interaction	5	2024
Biological Air Filtration	6	2029
Aerobic Bacterial Bioreactors	6	2029
Autonomous Crop Production	5	2029
Supercritical Water Oxidation	4	2029
Torrefaction Fecal Processing	3	2029
Zero Boil-Off Fuel Depots	4	2030
Molten Oxide Electrolysis	3	2030

6.3 Budget and Cost Analysis

In FY 2014, NASA received $17.9 B in federal funding for its activities. Of this $17.9B, we expect that the manned programs of NASA can feasibly draw $8.5B in annual funding. Beyond this primary funding source, RASC-AL has specified that commercial and international participation is required to complete their vision. Using ISS international contributions as a guide (assuming $125B total ISS lifetime cost), we expect NASA to contribute 50%, RKA to contribute 32%, ESA/CSA to contribute 10%, and JAXA to contribute 8% [4,5]. CNSA and ISRO participation is not assumed (but not discouraged). To provide buffer for political disputes, the expected international contribution is halved to 25% leaving NASA to contribute 75% of total costs.

Summing domestic and international contributions, the DELPHI program could feasibly draw $11.3B in annual funding. The annual funding dedicated to the DELPHI program ramps up as the project progresses from concept to operation. As the DELPHI base matures (post-2034), it is envisioned that commercial interests incrementally assume developmental and operational responsibilities allowing for government funding to ramp down. As a baseline estimate, the average annual funding is assumed as 75% of peak funding ($8.5B). For a 20-year development timeline, this provides $170B in total funding. A significant portion of NASA's budget is already encumbered towards the ISS, SLS, and Orion. The ISS will be defunded after 2024 to support the DELPHI program as NASA's flagship human exploration program. The SLS and Orion are funded to full development and are utilized at least throughout DELPHI's development. NASA's budget as well as program costs are expected to keep pace with inflation so all estimations are done in 2015 dollars.

International partnerships introduce additional complexity and risk into the project. Further, the potential financial gains cannot be fully realized due to additional costs related to managing communications between programs and system integration. Partnerships place requirements on crew composition and require delineation of responsibility for different system components. Intellectual property becomes a concern particularly as it relates to ITAR and other nations' policies on space technology. DELPHI recommends a structure in which NASA holds responsibility over core transportation and habitat elements. Other nations would be responsible for interchangeable items on or within these elements. Ideally, system redundancy between space agencies should be minimized to maximize the financial advantage of international partnerships.

For cost estimation, the DELPHI team initially attempted to apply NASA's PCEC costing model to the DELPHI architecture. Ultimately, we decided against this approach due to three primary issues. Firstly, as a system of systems, the design of each system within the DELPHI architecture was not sufficiently specified to provide high-confidence parametric inputs into PCEC. Secondly, PCEC is intended to be used for technologies with a TRL of 6 or higher and many DELPHI technologies fall below that level. Thirdly, PCEC offers a limited range of general systems based on historical spacecraft; it is difficult to accurately represent a lunar surface base within PCEC. A high-fidelity cost estimation tool such as PCEC is recommended to be applied in the future once all system designs have reached sufficient maturity.

To accomplish a low-fidelity cost estimation, a loose and simple heuristic model approach was selected that could be quickly executed and readily understood. Major cost contributors within DELPHI were identified and analogized to similar historical systems. Fixed, variable, or total cost data from these historical systems (adjusted to 2015 dollars) was used to provide cost estimations for DELPHI systems. When a clear system or technology analog could not be identified, educated guesses were made to estimate cost. It is acknowledged that this is a high-uncertainty method of cost estimation so a margin of 25% is provided to cover potential budget overruns.

The major contributors to cost over the lifetime of DELPHI relate to the development and manufacture of the individual orbital and surface systems within the DELPHI architecture as shown in Table 12 in Appendix A. Altogether, system development (both hardware and software), manufacture, and testing costs are estimated at $79.6B. Mission planning and operations through the active 10 year deployment phase between 2024 and 2034 are estimated at $5B per year totaling $50B total. With 25% margin, the total cost to implement the DELPHI architecture is estimated at $162B. With a budget ceiling of $170B, the DELPHI team considers this estimated cost to be acceptable and affordable.

7 Conclusions

As demonstrated in this report, the DELPHI team has provided a feasible and affordable architecture that conforms to the RASC-AL provided requirements and guidelines. Major highlights of the DELPHI architecture are as follows:

- System of systems design approach with emphasis on synergistic interactions between major systems
- Establishment of habitats at EML1 and on the lunar surface to support a variety of potential operations
- Implementation of a highly-reusable lunar landing and ascent architecture coupled with ISRU fuel production and an orbital fuel depot to minimize fuel resupply from Earth and reduce launch mass for Mars mission architectures
- Implementation of a highly-efficient bioregenerative ECLSS system with ISRU resupply of water and oxygen that would constitute a critical capability within a Mars system architecture
- Testbed, proving ground, and research station to support technologies and operations related to asteroidal and Mars exploration, commercial space mining, satellite servicing, planetary science, and astronomy

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9 Appendix A - Supplementary Tables and Figures

Table 8: Communication architecture orbital elements [6]

Communications Constellation						
Spacecraft	***a*** **(km)**	***e***	***i*** **(deg)**	**Ω (deg)**	**ω (deg)**	**ν (deg)**
Mothercraft	1861	0.043	90	90	0	0
Daughtercraft 1	8049	0.4082	45	0	90	0
Daughtercraft 2	8049	0.4082	45	0	90	180
Daughtercraft 3	8049	0.4082	45	0	270	0
Daughtercraft 4	8049	0.4082	45	0	270	180
Daughtercraft 5	8049	0.4082	45	180	90	132
Daughtercraft 6	8049	0.4082	45	180	90	228
Daughtercraft 7	8049	0.4082	45	180	270	132
Daughtercraft 8	8049	0.4082	45	180	270	228

Table 9: Pulsed plasma thruster specifications [9]

Thruster	
Volume	0.5 U
Mass	0.55 kg
Power	2 W
Thrust	0.5 mN
Isp	700 s
TRL	5

Table 10: Antenna and transceiver specifications [10,11]

Antenna		**Transceiver**	
Rf input/output	1-4 SSMCX, female, 50 Ohm	**Transmit power**	22 dBm
Mass	100 g	**Mass**	85 g
Power consumption	20 mW	**Power consumption**	1.7 W (transmitter on) 0.2 W (receiver only)
Frequency range	10 MHz	**Transmitter frequency range**	130-160 MHz
Deployment duration	3 s	**Receiver frequency range**	400-450 MHz
Deployment power consumption	2 W	**Downlink data rate**	1200 bps
Antenna return loss	-10 db	**Uplink data rate**	1200 bps

Table 11: Habitat Module Types

Habitat Module	Quantity	Description
Command Module	1	Self-sufficient module with physicochemical ECLSS system to support initial crews
Pressurized Excursion Module	1	Mobile version of the command module intended for extended surface exploration missions away from the main habitat. The pressurized excursion module can be carried on an ATHLETE rover in a caravan with SEVs or other modules.
Medical/Exercise Module	1	Provides critical medical and exercise module to monitor crew health and keep the crew healthy during prolonged surface stays.
Research Module	1	Enables laboratory research into effects of lunar partial gravity on biological and physical systems. Also enables testbeds for exploration technologies needed in later program phases and beyond.
Living Module	1	Extended habitable volume with additional private quarters and common areas such as kitchens.
Greenhouse Module	2	Provides the technological backbone of DELPHI's final-phase bioregenerative system by housing crops and other bioregenerative technologies
Resupply Module	5	Generic module that can be filled with cargo and later emptied out and repurposed

Table 12: SLS Cargo Launch Payload Manifest

SLS Cargo Launch Number	Launch Year	Payload Description
1	2022	LITEHABS EML1 Habitat, two Oracle landers
2	2023	2 ATHLETEs and 2 SEVs with solar panels
3	2024	2 ATHLETEs and surface infrastructure (solar panels, batteries, radiators, gas tanks, communications equipment, ISRU experiments, etc.)
4	2024	Additional surface infrastructure
5	2025	Command Module
6	2025	Medical/Exercise Module
7	2026	Research Module
8	2026	Resupply Module 1
9	2027	Greenhouse Module 1
10	2027	Resupply Module 2
11	2028	Pressurized Excursion Module
12	2028	Resupply Module 3
13	2029	Living Module
14	2029	Full-Scale ISRU Hardware
15	2030	High capacity ZBO fuel depot
16	2030	Resupply Module 4
17	2031	Greenhouse Module 2
18	2032	Resupply Module 5
19	2033	TBD (margin)

Table 13: Launch Schedule Excel Model

	2022	2023	2024	2025	2026	2027	2028	2029	2030	2031	2032	2033	Totals
SLS Cargo Launches	1	1	2	2	2	2	2	2	2	1	1	1	19
SLS Crew Launches				1	1	1	1	1					5
Falcon Heavy Launches									1	1	1	1	4
Delta II Heavy Launches	1					1					1		3
Cargo Landing/Ascent Cycles		1	2	2	2	2	2	2	1	1	1	1	17
Crew Landing/Ascent Cycles				1	1	1	1	1	1	1	1	1	9
Total Payload to EML1 (t)	45.311	45.311	90.622	109.986	109.986	109.986	109.986	109.986	90.622	45.311	45.311	45.311	957.729
Landing/Ascent Fuel to EML1 (t)	0	28.186	56.372	82.433	82.433	82.433	82.433	82.433	0	0	0	0	496.723
Payload to Lunar Surface (t)	0	17.125	34.25	27.553	27.553	27.553	27.553	27.553	45	45.311	45.311	45.311	370.073
Payload at EML1 (t)	45.311	0	0	0	0	0	0	0	45.622	0	0	0	90.933
Required Fuel Production (t)									54.247	54.247	54.247	54.247	713.711
Fuel Production (t)									100	100	100	100	
Cumulative Payload to EML1	45.311	45.311	45.311	45.311	45.311	45.311	45.311	45.311	90.933	90.933	90.933	90.933	
Cumulative Payload to Surface	0	17.125	51.375	78.928	106.481	134.034	161.587	189.14	234.14	279.451	324.762	370.073	
Cumulative Landing/Ascent Fuel Launched	0	28.186	84.558	166.991	249.424	331.857	414.29	496.723	496.723	496.723	496.723	496.723	
Cumulative Fuel Produced	0	0	0	0	0	0	0	0	100	200	300	400	
Stay Duration (yr)				0.083333	0.5	1	1	1	2	2	2	2	

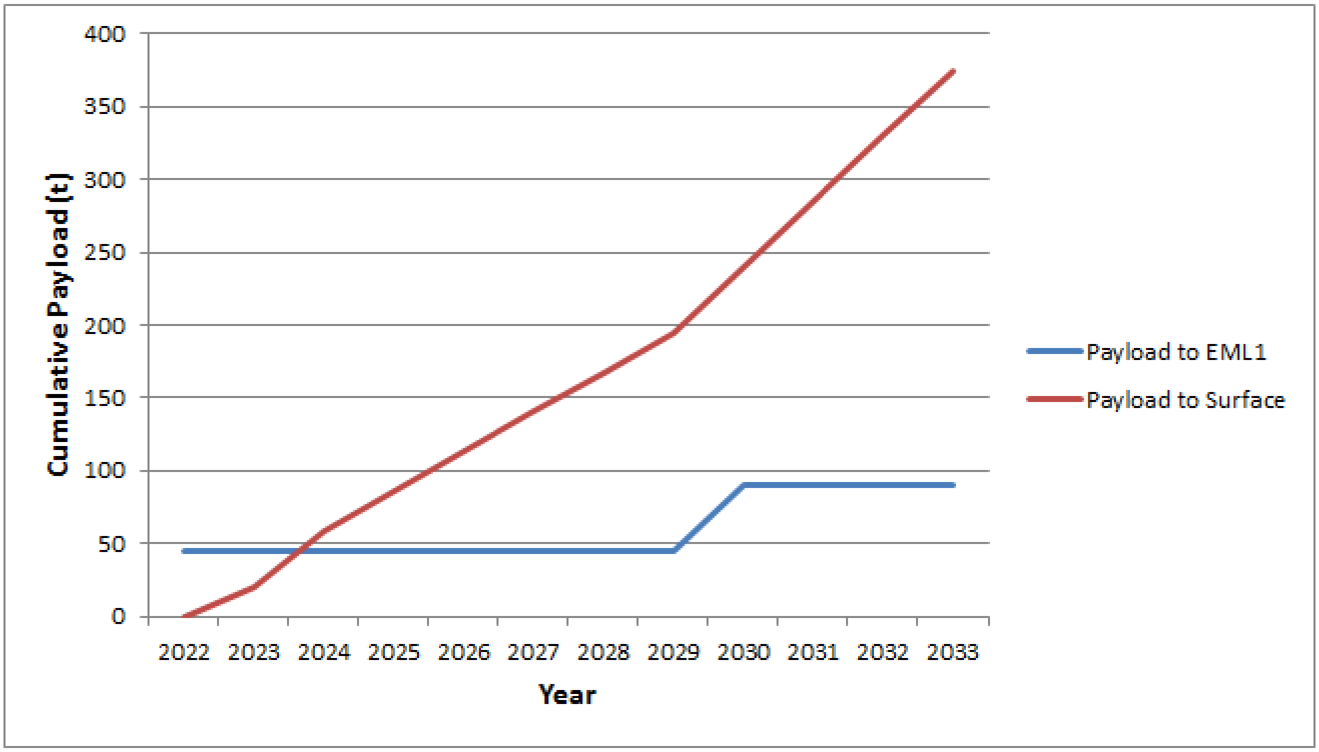


Figure 10: Cumulative payload to EML1 and lunar surface over time

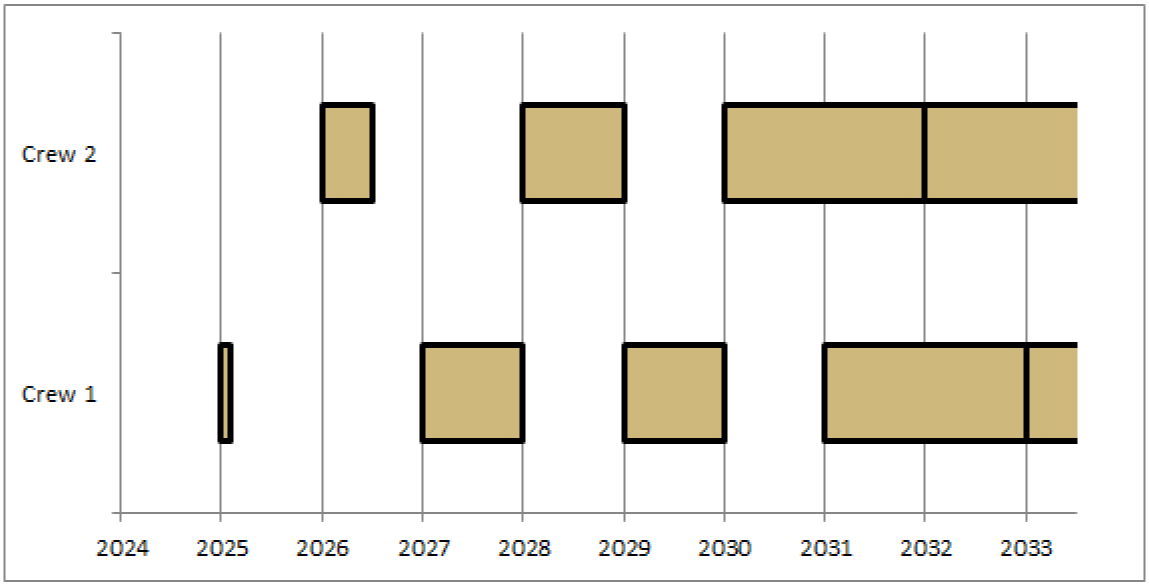


Figure 11: Incremental extension of crew stay durations throughout the DELPHI build-up phase

Table 14: Budget model based on variable and fixed costs

Element/Technology	Quantity	Fixed Cost ($B)	Variable Cost ($B)	Total Cost ($B)	Notes	Expected System Readiness
SLS	24	14.6	0.5	26.6	Fixed cost includes development and launchpad upgrades	2021
Orion	9	10	0.5	14.5		2021
Falcon Heavy	7	0	0.13	0.91	7 crew +, SpaceX funds development privately, $130M per FH launch to EML1 with full payload	2016
Delta II Heavy (7920H-10)	3	0	0.05	0.15	3 comm satellite launches	Immediate
Oracle Lander	3			10	Variable cost includes two replacement CECEs for each lander, cost quoted from Charles Bolden	2022
Habitat/Logistics Modules	12	1	0.1	2.2	For generic empty pressure vessels with short range comm and appropriate I/O connections, based on rough estimate of Cygnus development cost at $600M, $400M in margin added for development, $100M per unit cost assumed	2024
SEV	4	3.058	0.1	3.458	Current budget is $152.9M, doubled and maintained for 10 years, $100M per unit cost assumed	2022
ATHLETE	4	2	0.05	2.2	Development cost based roughly on MSL costs, $50M per unit cost assumed	2022
EML1 Habitat	1			6.1	Cost sourced from 2014 LITEHABS RASC-AL Proposal with 10 year development	2018
EML1 ZBO Fuel Depot	1			2	Total cost assumed as $2B based on ULA study	2022
Comm/Nav Sat Mothership w/ Daughterships	3	0.6	0.3	1.5	Cost based loosely on MRO costs ($720M for single satellite with development)	2022
Bioregenerative ECLSS Technology	1			2.5	Includes development cost plus manufacturing/installation into habitat module	2029
ISRU Tech	1			2.5	Includes development cost plus manufacturing/deployment onto surface	2031
Miscellaneous Surface Infrastructure	1			5	Solar Panels, Radiators, Communications Equipment, etc.	2022
Integration, Mission Planning, and Operations				50	Includes 10 years of operation costs based on ISS historical data at $5B /year over 10 years.	Immediate
			Total Cost	129.618		
			25% Margin	32.4045		
			Total With Margin	162.0225		

11. Appendix B - Electrodynamic Dust Shield Prototype

A two-phase electrodynamic dust shield was designed, built, and tested using JSC-1A lunar regolith simulant to evaluate its suitability for protection of dust sensitive surfaces such as solar panels and radiators on dusty planetary surfaces such as the moon or Mars. The basic principle of operation involves regolith charging via dielectric leakage current and tribocharging. Once charged, the alternating electric field on the shield repels the dust laterally off of the shield. High voltages on the order of kV are required to charge and move the dust. The design specifications for the electrodynamic dust shield prototype were determined through a literature survey of existing electrodynamic dust curtains and dust shields and are described below in Table 15.

Table 15: Electrodynamic Dust Shield Specifications

Property	Specification
Electrode Pattern	Two-Sided Interdigitated Comb
Electrode Spacing	0.7 mm
Electrode Width	0.3 mm
Electrode Pitch	1 mm
Electrode Material	Tin
Voltage Waveform	Anti-Phase Square Wave
Voltage Waveform Amplitude	10 kV
Voltage Waveform Frequency	15 Hz
Dielectric Material	Polyurethane
Dielectric Thickness	< 1 mm
Shield Dimensions	10 cm × 10 cm
Targeted Clearing Factor for JSC-1A at STP	100%

PCB, power circuit, software, and system design schematics were generated to meet these specifications as shown respectively. After an informal design review, components were procured and the system was then built and software was written to control and monitor the system. The completed hardware is shown in Figures 12 and 13.

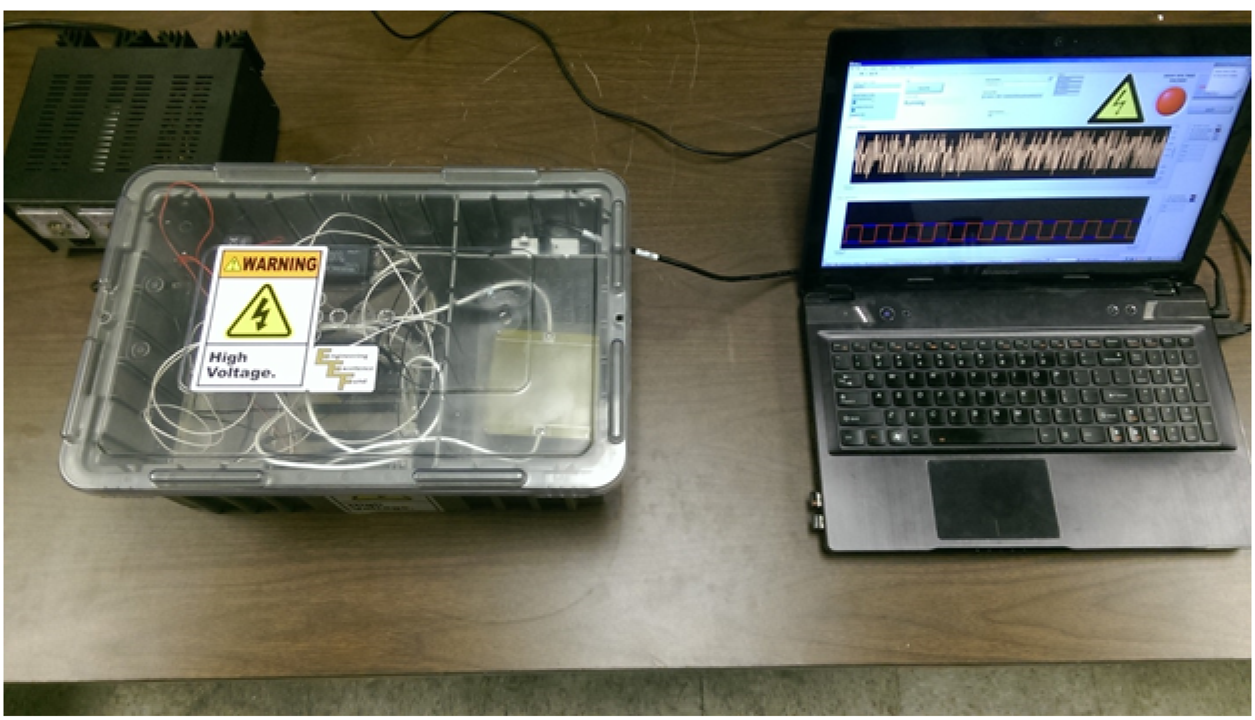


Figure 12: Complete system with external power supply at top-left and external computer at right running control software

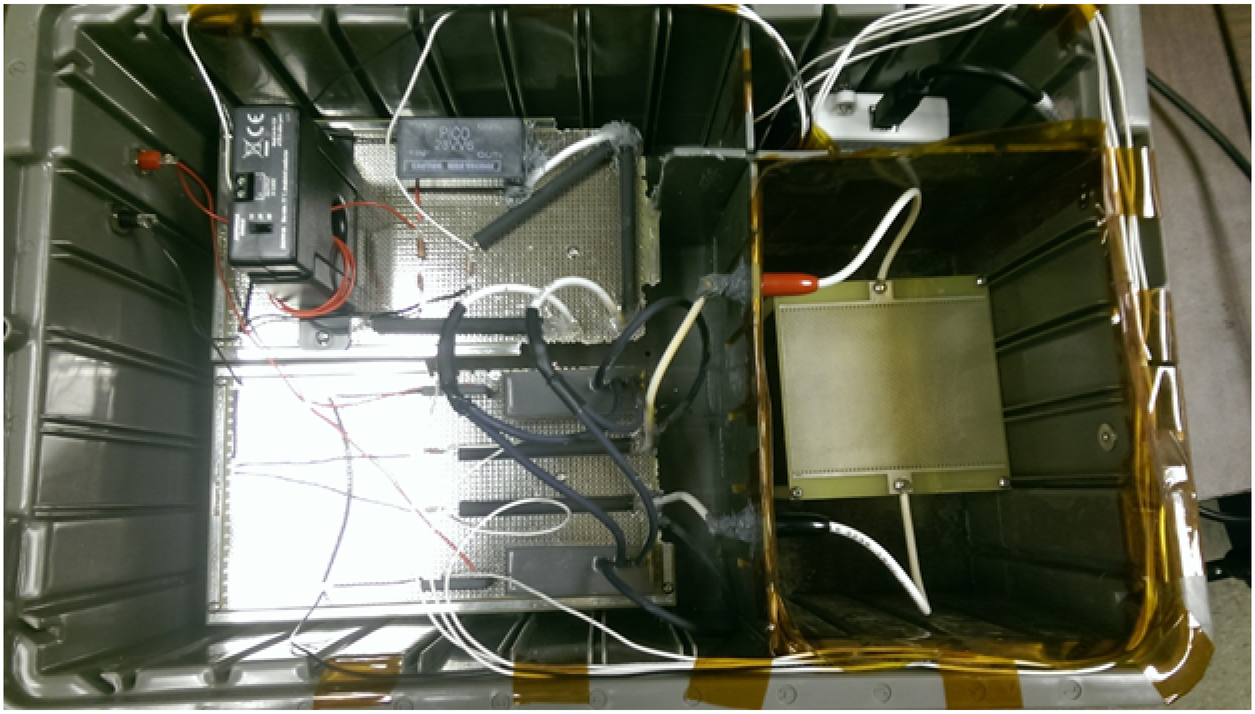


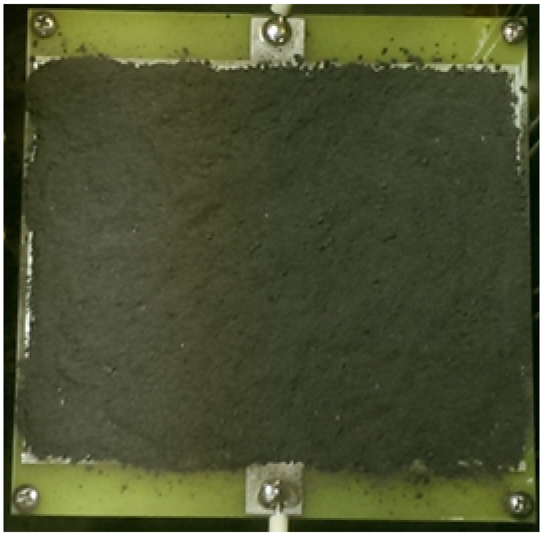
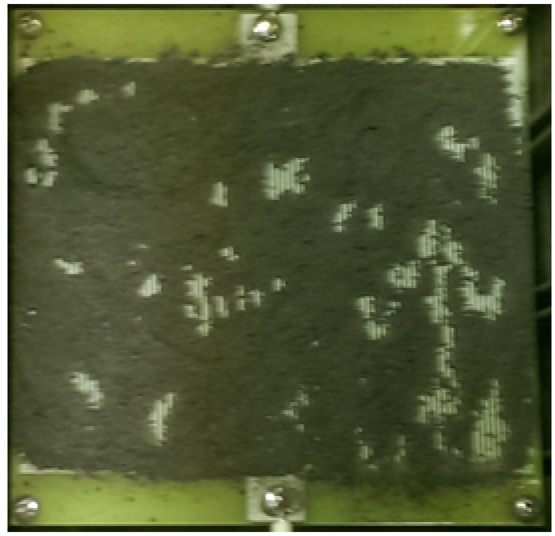
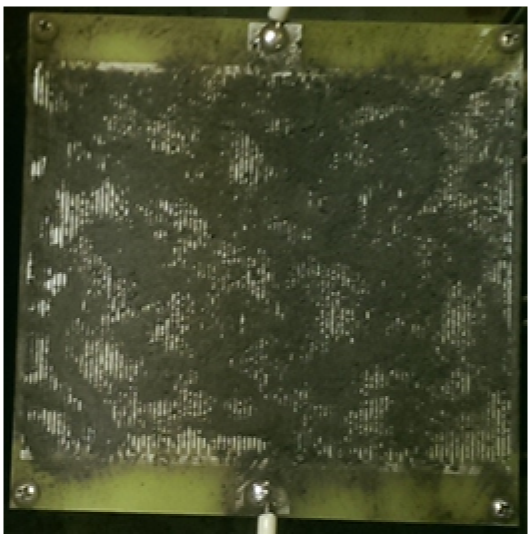
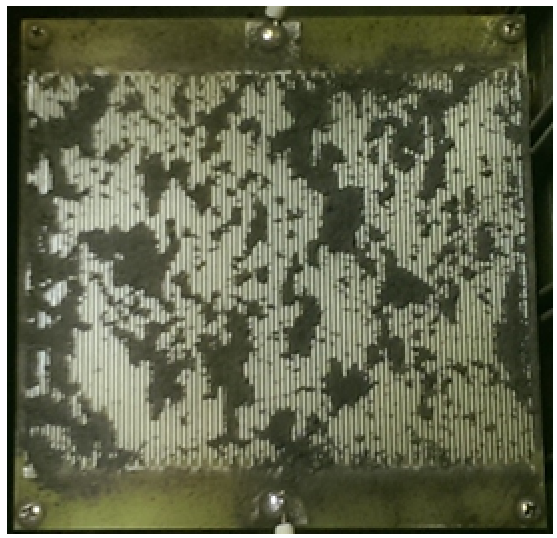
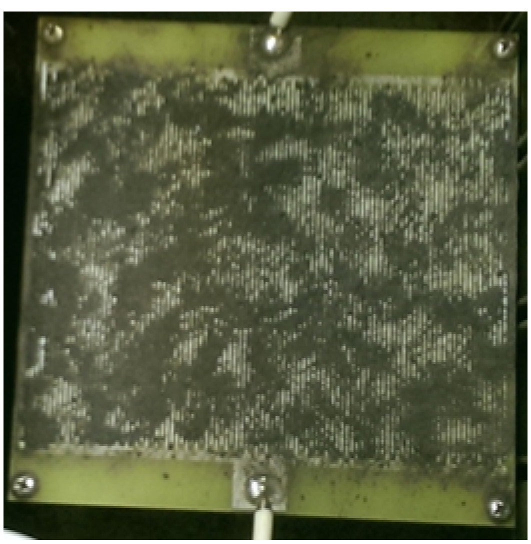
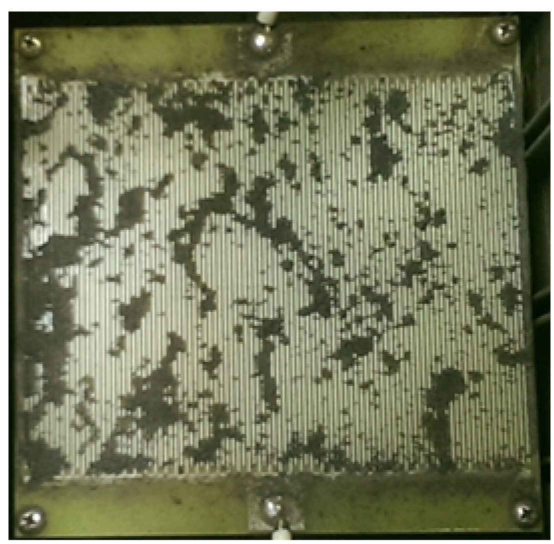
Figure 13: Core dust shield system with high-voltage power circuitry at left, data acquisition module at top-right, and dust shield at bottom left.

As the system was assembled, verification and characterization tests were iteratively performed along the way. Multiple issues were discovered and fixed during this process. Notably, during testing, it was suspected that the 10 kV DC-DC converter had failed. This prompted a replacement using a less expensive 6 kV DC-DC converter due to lack of available funds. Ultimately, the 6 kV DC-DC converter reduced system performance. Still, this was sufficient to demonstrate operation as shown in Table 15 on the next page. Videos of the system in action can be viewed at the following URLs:

- https://youtu.be/cAagbKctZfY
- https://youtu.be/4Fm1yhTfxo8

The system struggles when removing clumps of particles. It performs exceptionally when dust coverage is sparse. The suspected reason for this performance deficit is due to the replacement DC-DC converter that only provides 60% of the originally specified voltage. It is suspected that the apparent failure of the 10 kV DC-DC converter was actually an issue with the data acquisition module in which it was unintentionally exposed to high voltage via a spark that short circuited the analog input channel that typically monitored the output of the DC-DC converter. This gave the false impression that the converter had failed. Before the 2015 RASC-AL forum, we plan to retest the 10 kV DC-DC converter and to reintegrate it into the system. Assuming it works, performance should be improved substantially with up to 100% dust removal. Updated results will be presented at the 2015 RASC-AL forum.

Table 15: Dust coverage on electrodynamic dust shield before and operation using a 6 kV DC-DC converter

Initial	Final
	
	
	

12. Appendix C - Trajectory Analysis

12.1. Crewed Trajectory Analysis

Crewed transfer designs are often concerned with time of flight (TOF) as a primary performance index with ΔV as a secondary. Crewed transfers designed for this mission take five days to/from the EML1 Halo orbit. A Halo orbit is a 3-dimensional quasi-periodic orbit about a collinear Lagrange point in a given CRTBP making two orthogonal X-axis crossings per orbit (in the synodic frame). In the EM CRTBP, Cartesian states along such an orbit in the vicinity of these crossings were found to be the energy-cheapest as rendezvous/departure points to/from the Earth for a direct transfer assuming a constraint of ΔV being employed only in the along-velocity direction. Since the local dynamics change in a roughly linear fashion at this location with respect to a change in velocity with the given constraint, a transfer to or from Earth may be constructed with little intuition for an initial guess trajectory and a desired orbit altitude may be targeted using a form of bisection. The procedure employed to construct crewed transfers was as follows.

1. Select a state along the halo orbit nearby the below-orbit-plane X-axis crossing.
2. Induce an anti-velocity perturbation (to find transfer from LEO to EML1) or along-velocity (to find transfer from EML1 to LEO).
3. Propagate modified state (position unchanged, velocity changed) backwards/forwards in time (LEO to EML1/EML1 to LEO) to perigee.
4.
    a. If perigee > or < desired radius and *did not* previously cross desired orbit altitude, decrement initial ΔV by small multiple and repeat steps 2. - 4. until convergence
    b. Else if perigee > or < desired radius and *did* previously cross desired orbit altitude, change sign of ΔV and cut magnitude in half, then repeat steps 2. - 4. until convergence
5. Repeat the above process until the desired orbit altitude is bisected within some tolerance or the program has run for too many iterations.

The bisection method previously described was used to constructed the large cyan and blue trajectories shown below. The Earth and Moon are plotted to scale in the CRTBP.

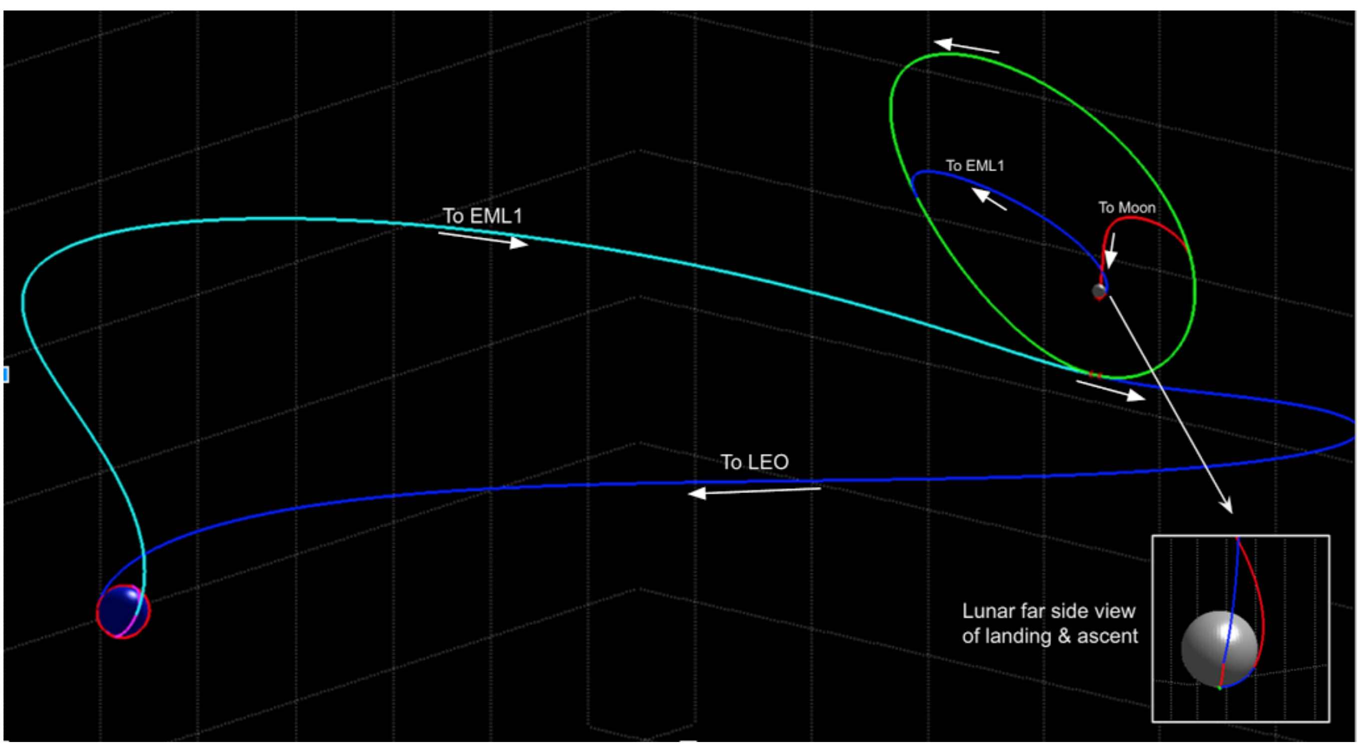


Figure 14

12.2. Cargo Trajectory Analysis

Low energy BLTs leverage a branch of mathematics known as dynamical systems theory to allow a spacecraft to coast to the moon following a single large LEO Translunar Injection (TLI) burn. Periodic and quasi-periodic orbits around equilibrium points in a dynamical system possess a fascinating property of motion known as invariant manifolds. Manifolds in this discussion are surfaces in state-space which are composed of points at which a theoretical particle may be located and exponentially converge onto a periodic orbit as time tends forward to infinity (along the stable manifold) or backward to infinity (unstable manifold). In a CRTBP, the stable or unstable manifold comprises ballistic trajectories to/from a Lagrange point orbit. It may be constructed in 6-element Cartesian position/velocity state space in the following manner.

1) For a given state on a periodic orbit, integrate the State Transition Matrix (STM) $\Phi = dX_i/dX_0$ (where $X_i = [x\ y\ z\ v_x\ v_y\ v_z]^T$) forward in time over a single period of the orbit.
2) Calculate the eigenvectors and eigenvalues of this monodromy matrix. An eigenvalue pair satisfying $\lambda_1\lambda_2 = 1$, $\lambda_1 > 1$, $0 < \lambda_2 < 1$, will be present. The eigenvector corresponding to λ_1 is unstable and that corresponding to λ_2 is stable.
3) Perturb the orbit state by some small fraction of the unit stable (or unstable) eigenvector.
4) Propagate the perturbed state backwards (or forwards for unstable manifold) in time until a desired stopping condition.
5) Repeat for each orbit state; the resultant family of trajectories represents the invariant stable/unstable manifold.

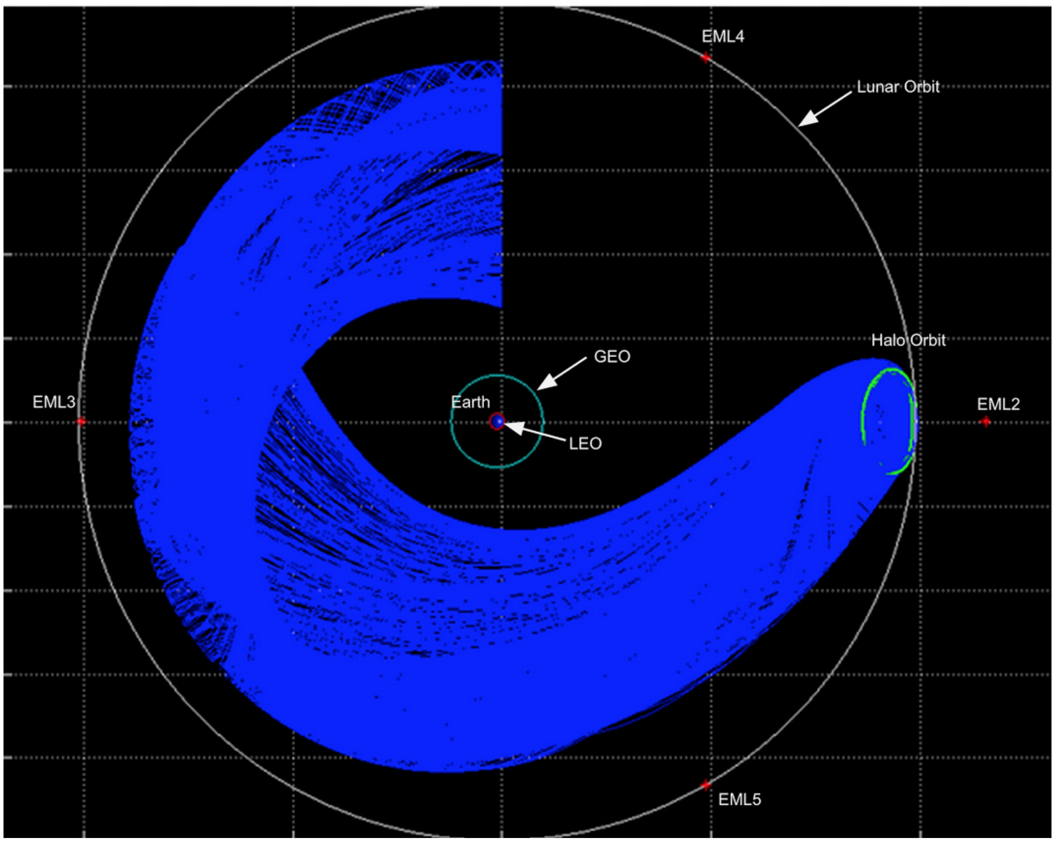


Figure 15

For a Halo orbit in a CRTBP, the resulting manifold of trajectories when plotted in position space takes the appearance of a twisted tube wrapping around the system with one opening at the orbit. The low energy trajectory constructed for this project uses patched CRTBP dynamics to leverage the invariant manifold property of dynamical systems. Analogous to 2-body patched conics, patched 3-body dynamics transitions from one CRTBP to another when crossing the orbit of the secondary body of one of the CRTBP systems. Low energy trajectories take advantage of the sun's influence on raising the spacecraft's semi major axis when in orbit around the Earth. The perturbation is especially significant for spacecraft travelling outside of the moon's orbit radius. Thus this radius shall mark the boundary of system transition for this problem. Using this framework gives us a medium-fidelity solution to model the spacecraft's behavior under the influence of three bodies by restricting it to be perturbed either by the Earth and moon only, or the Earth and Sun only at a given time. A high level outline of the trajectory design process for a BLT to a Halo orbit is as follows.

1) Construct the stable manifold of an EML1 or EML2 Halo orbit (L1 used in this study).
2) Construct a Sun Earth (SE) Halo orbit around SEL1 or SEL2 (L2 used in this study), and its stable and unstable manifolds.

3) Select an initial guess trajectory composed of three segments. Segment 1 is a trajectory on the SEL2 stable manifold which departs the Earth from some LEO radius to the SEL2 Halo. Segment 2 is an SEL2 unstable manifold trajectory departing the SEL2 Halo transferring back to the Earth-Moon system. Both segments so far occur in the Sun-Earth rotating frame dynamics.
4) Rotating the Earth-Moon system as necessary, select an EML1 Halo stable manifold trajectory with one end near the vicinity of the tail end of segment 2. This trajectory transfers to the EML1 Halo in the Earth-Moon rotating frame dynamics. Rotation of the system may be necessary if the EML1 Halo stable manifold does not inherently come close to the SEL2 Halo unstable manifold. The moon orbits the Earth while the Earth orbits the sun, so the EM rotating frame actually rotates in the SE rotating frame. Choice of a rotation angle of the EM X-axis with respect to the SE X-axis can be useful, and any necessary relative orientation repeats monthly. If no "reasonable" initial guess segments can be found, a different combination of Halo orbits can be constructed using an arclength continuation method.
5) With initial guess segments chosen, chain segment 2 to segment 3 using a desired optimization routine (blackbox fmincon, collocation, monte carlo, etc) constraining the position and velocity discontinuity between the end of segment two and start of segment 3 to be zero while varying the velocity vectors of the start point of segment 2 and the endpoint of segment 3. This leads to a quasi-ballistic but fully continuous trajectory from the Earth to SEL2 to EML1 shadowing the manifolds of two Halo orbits. The quasi-ballistic behavior comes from the resulting departure velocity from the SEL2 Halo and the arrival velocity at the EML1 Halo potentially having magnitudes on the order of m/s depending on the initial guess and the optimization scheme.

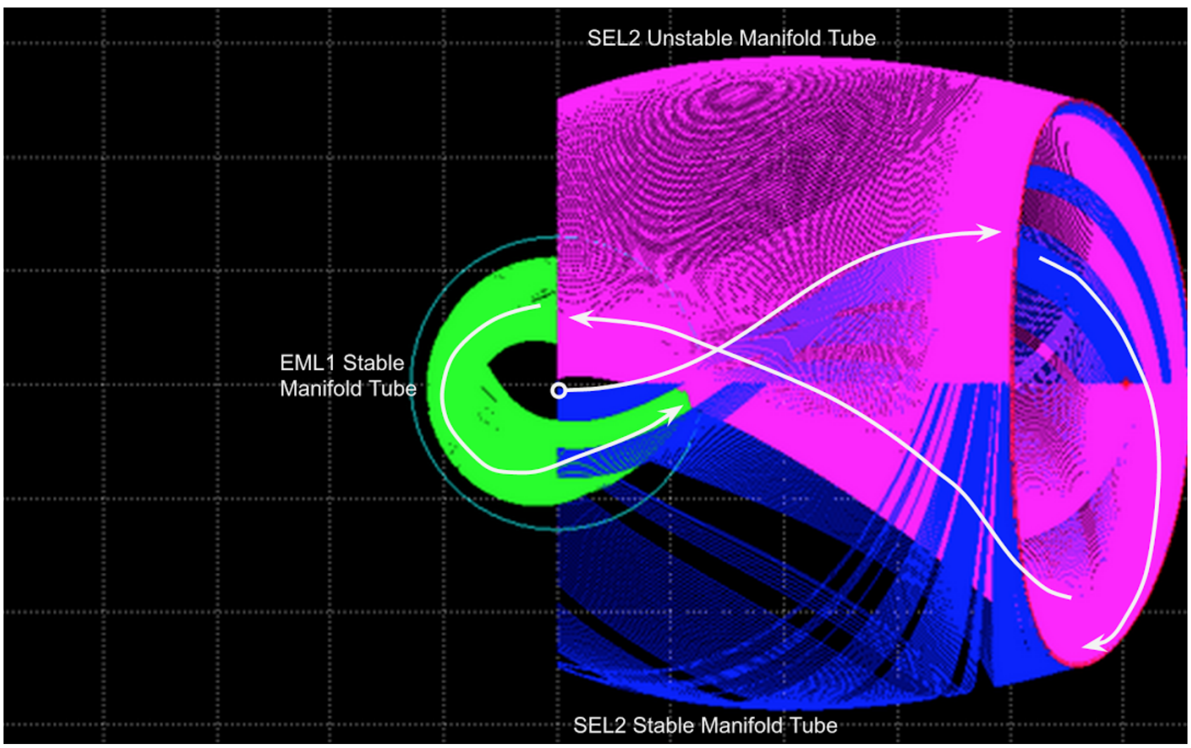


Figure 16:
Search space for potential low energy solutions depicting stable and unstable manifolds of an SEL2 Halo orbit and their interaction with the EML1 Halo orbit stable manifold. Manifolds are truncated at the geocentric y-axis crossing to facilitate the search for patch points.

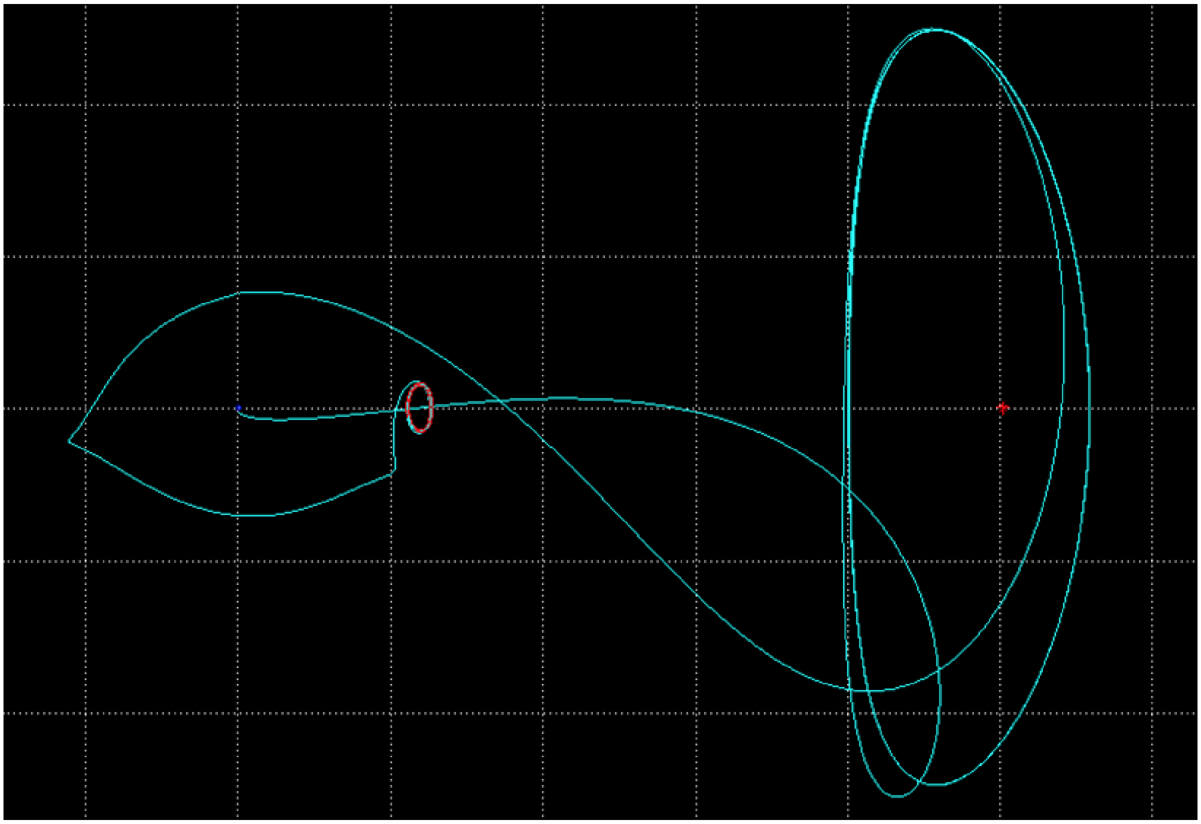


Figure 17

The total ΔV of this trajectory is 3.200 km/s with a TOF of 3.002 years (time of flight was unconstrained in this scenario). Multiple spacecraft may be launched several months apart onto the same trajectory, forming a continuous supply chain to crewed operations at EML1.

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- The CU Boulder Engineering Excellence Fund for financial support of the electrodynamic dust shield prototype
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- RASC-AL and sponsoring organizations

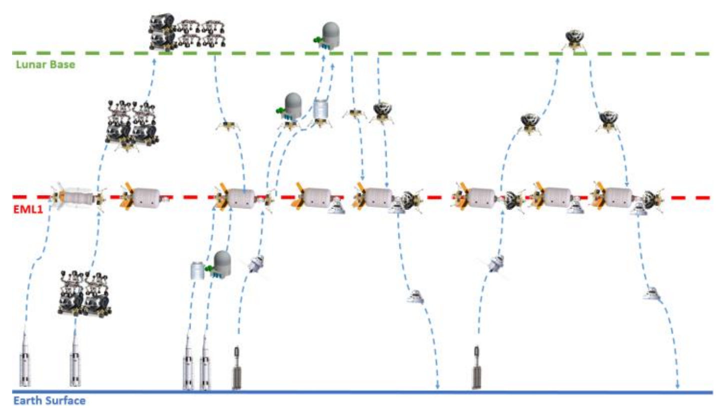
2015 RASC-AL Technical Paper Compliance Matrix

Earth Independent Lunar Pioneering Architecture Theme	Y/N
Is the overall system architecture sufficiently addressed?	Y
Have you proposed synergistic application of **innovative capabilities and/or new technologies** for evolutionary architecture development to enable future missions, reduce cost, or improve safety?	Y
Does your scenario address novel applications (through scientific evaluation and rationale of mission operations) with an objective of NASA sustaining a permanent and exciting space exploration program?	Y
Have you considered unique combinations of the planned elements with innovative capabilities/technologies to support crewed and robotic exploration of the solar system?	Y
Have you addressed reliability and human safety in trading various design options?	Y
Have you identified the appropriate key technologies and TRLs?	Y
Have you identified the systems engineering and architectural trades that guide the recommended approach?	Y
Have you provided a realistic assessment of how the project would be planned and executed (including a project schedule with a test and development plan)?	Y
Have you included information on annual operating costs (i.e., budget)?	Y
Have you given attention to synergistic applications of NASA's planned current investments (within your theme and beyond)? **Kudos given to additional inclusion of synergistic commercial applications**	*Y*
Does your paper meet the 10-15 page limitation?	Y

Team Info	
University	University of Colorado
Abstract Title	DELPHI: A Lunar Architecture to Enable Exploration, Research, and Commercial Development of Space Beyond LEO
Faculty Advisor	Dr. David Klaus
Team Leader	Jonathan Anthony
Competition Category	Graduate

DELPHI

Graphic of Concept/Technology



Summarize Critical Points Addressing Theme Compliance and Innovation

- 8 people continuously living on the surface of the moon, completely self-sufficient beginning in 2034
- Annual crew rotation from Earth after 2034, with 10t of logistics
    - Only 9.02 t required with landing capability for up to 8 t of expansion hardware
    - Fuel-positive operation after 2030 with more fuel sent up than fuel sent down
- Gradual build-up of capabilities, infrastructure and risk reduction
    - Crew durations gradually increase
    - Low-TRL technologies identified and prioritized for development
    - Steady cadence of infrastructure landings
    - Evolution of ECLSS from physicochemical to bioregenerative
- Budget accurately reflects the constraints listed in the themes description
- In-situ Resource Utilization (ISRU) and reusable systems
    - Water and oxygen ISRU production with integration into ECLSS and fuel systems
    - Fully reusable lander, utilization of reusable Falcon Heavy for crewed launches in later program phases, multi-use logistics and habitat modules
- Development of new technologies and infrastructure necessary for ISRU and transportation
    - Development of molten oxide electrolysis, carbothermal reduction, and microwave water extraction
    - Development of zero boil-off fuel depots, reusable landing systems, and autonomous landing capabilities
- Integration of reliability and human safety into design process
    - Multiple abort modes during transit and surface habitation
    - Medical monitoring of crew members to characterize long-term lunar human health effects
    - Multipronged dust mitigation strategy
    - Radiation shielding leveraging regolith ISRU
    - Bioregenerative ECLSS with physicochemical backup
- **Innovation** in crafting a concept that will extend humanity's reach beyond LEO
    - Increase in lunar payload over traditional architectures via low-energy transfers and EML1 waypoint with propellant depot
    - Decrease in resupply requirements versus traditional ECLSS architectures
    - Development of critical technologies for Mars exploration
        - Opportunity to test methane propulsion at minimal cost via the CECE
    - Establishment of a EML1 propellant depot to support Mars transit vehicles and reduce launch mass
    - Opportunities for the commercial space sector to expand within and beyond LEO
    - Research benefits for planetary, astronomical, and cosmological scientists looking outside our own solar system to the universe at large