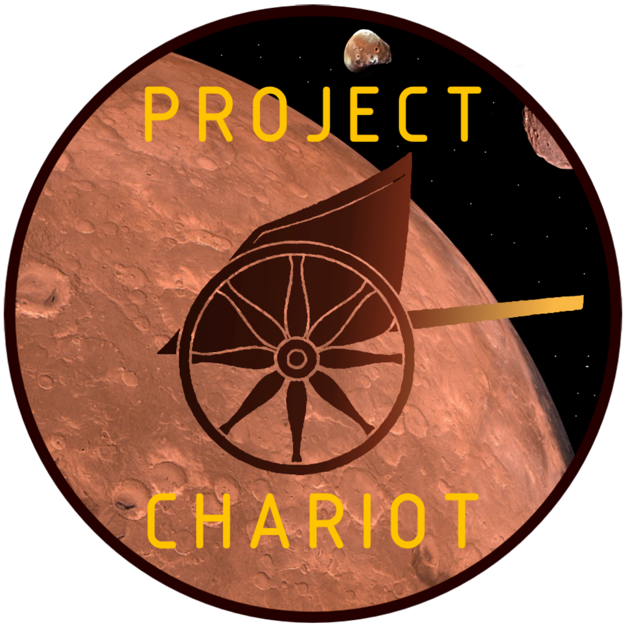
AIAA - Martian Moon Exploration Excursion Vehicle

Project Chariot



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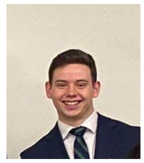

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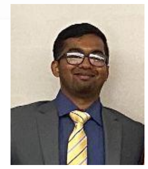

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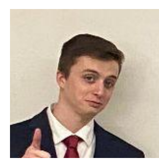

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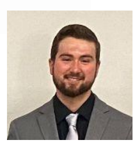

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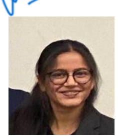

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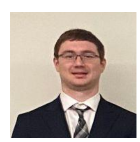

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Executive Summary

Project Chariot is, at its core, both a science mission and a human exploration mission, serving the dual purposes of gaining insight into the origins of Phobos and Deimos and being a stepping stone toward a crewed mission to the Martian surface. To not divert excessive funds away from Mars exploration, Project Chariot will be a short-term, relatively low-cost sample retrieval mission to Phobos and Deimos. The design of Project Chariot involved thorough work from the following seven functional divisions: Science Operations; Human Factors and Life Support; Structures and Launch Vehicle; Attitude, Trajectories, and Orbits; Propulsion; Power, Thermal, and Environment; and Communications, Commands, and Data Handling.

The Exploration Excursion Vehicle (EEV) for this mission must be able to support two crew members, be in a 5-sol parking orbit around Mars before January 1, 2040, autonomously dock with the Deep Space Transport (DST), and perform automated sample retrieval of a minimum of 50 kg from each moon. The mission must stay under a $1 billion budget, quarantine regolith samples from the astronauts before arrival on Earth, last no longer than 30 days from rendezvous with the DST, and not plan for the crew to have any extravehicular activity (EVA). The parameters which can be varied to best suit these requirements are the number of sorties from the DST, mission modes, maneuvers for the round trip, methods chosen for sample retrieval, choice of propulsion system, and launch vehicle.

The science mission of Project Chariot is to collect, analyze, and return at least 50 kg of regolith samples from both Phobos and Deimos. The composition of the regolith will be determined in-situ using an ion trap mass spectrometer (ITMS); the results of this investigation will provide insights into the origins of Phobos and Deimos and the history of Mars. A number of secondary science objectives will also be explored. The atmosphere of Mars and the space around its moons will be investigated for the presence of carbon compounds, volatile compounds, and trace gases using a quadrupole mass filter spectrometer (QMS). The topography of the moons' surfaces will be mapped using a light detection and ranging (LiDAR) system. A dust counter will be used to map the dust fields around each moon. Subsurface regolith will be examined using a ground-penetrating radar (GPR).

The sample acquisition system consists of a combination of a pneumatic system and a coring drill. The pneumatic part of the system is a Honeybee Robotics PlanetVac, which sprays pressurized gas onto the regolith surface and directs regolith particles to a collection chamber. The coring system is Honeybee Robotics' space piezoelectric drill called the Autogropher II, which produces low force drilling suitable for the hyper-low gravity on both Martian moons. The drill provides access to depths of up to 3 m and the PlanetVac allows for the rapid acquisition of regolith samples. Sample collection on each moon can take up to a few hours. As regolith is pulled through the collection system, it is directed into the storage system and

further sorted into Earth storage and analysis storage. Both Earth storage and analysis storage are further divided into Phobos and Deimos samples. Once 50 kg of regolith per moon is acquired, the collection system directs samples into the analysis storage, where the crew is able to access the regolith through a glovebox and analyze it using a mass spectrometer.

The life support system of the EEV includes provisions for the natural and induced environment of the vehicle, crew habitability functions, hardware and equipment, crew interfaces, spacesuits, and clothing. The EEV will be equipped with environmental regulation systems that maintain air supply, temperature, and humidity. The air supply will include one nitrogen tank and two oxygen tanks, along with regulators, air processing units, and pressure gauges. There will be a total of 79.4 kg of oxygen and 40.9 kg of nitrogen brought on this mission, including a 50% safety margin for each.

Other necessary components of the life support system include food, water, personal hygiene, waste management, medical care, sleeping accommodations, and recreation. The crew will have a total water supply of 432 L, including a 50% safety margin. A nutrition plan has been created for this mission, adhering to the minimum required calorie intake for astronauts. A safety margin of 50% is included in the food supply. Taking into account the limited volume of the EEV, the sleeping quarters for the crew include vertical strap-on beds. Several medical packs necessary for a 30-day Martian mission are included in the mission design. The fire-suppression system consists of 15 International Space Station (ISS) smoke detectors and four portable water-mist fire extinguishers. The EEV design also includes a universal waste management system for managing bodily waste, a system to manage trash waste, and a toolkit for any necessary repairs.

The EEV will include interfaces that allow the crew to interact directly with the vehicle, its systems, and the ground station on Earth. Each crew member is responsible for a set of two touchscreen monitors that display mission-critical information at the front of the vehicle. Additionally, the crew will have a physical button control panel and two yokes for manual maneuvering and critical functions.

Each crew member will be supplied with a Boeing Starliner intravehicular activity (IVA) suit and comfortable, moisture-wicking clothing to wear throughout the duration of the mission. One National Aeronautics and Space Administration (NASA) Extravehicular Mobility Unit (EMU) extravehicular activity (EVA) suit will be provided for emergency operations outside the vehicle.

The inside of the EEV is divided into four modules. The first module follows the docking cone on the EEV and serves as the mission control zone; this zone contains computers installed for autonomous operations of the EEV as well as the crew interfaces. The second module is the recreation zone, where collapsible beds are installed on either side of the EEV. The open floor space in this module serves as a location to both administer medical care and perform recreational activities, such as exercising or reading. The third module is the personal hygiene cabin, which houses the universal waste management system. This zone also serves as a changing room, a storage location for clothing, and a place to maintain personal hygiene with features such as water-free showers and edible toothpaste. The fourth and final module is the sample collection cabin in the rear of the vehicle; this zone is unpressurized. While this cabin is typically sealed, it opens to lower the sample collection system down to the surfaces of Phobos and Deimos. The sample analysis module is included in the sample collection cabin with glove ports that allow the crew members to access and analyze the regolith.

The EEV structure consists of two aluminum 6061 shells and eight stringers. The stringers carry the majority of the structural loads. The landing system for the EEV is a custom-designed set of four retractable legs that will dig into the surfaces of Phobos and Deimos. The leg design used for this mission performed the best in an experiment that measured the amount of force required to dislodge a series of eight different leg models from a regolith simulant.

The SpaceX Starship launch vehicle will bring the EEV to the Martian system using a type-II Hohmann trajectory and insert the EEV into the 5-sol parking orbit. Combined, the launch, trajectory, orbit insertion, and any necessary corrections will take 2.17 years and require 14,784 m/s of Δv from the launch vehicle. The 5-sol orbit used in this mission will have an eccentricity of 0.59, a periapsis of 24,000 km, and an apoapsis of 95,630 km.

After being inserted into the 5-sol orbit on January 28, 2039, the EEV will await the rendezvous with the DST and the arrival of the crew on January 1, 2040. The docked EEV and DST will then travel together in the parking orbit until reaching the orbit periapsis. At this point, the EEV will depart from the DST and perform a Hohmann transfer to Phobos that requires 1,121 m/s of Δv and will take 9.11 hours. The landing and takeoff for Phobos will use a total of 48 m/s of Δv. After surface operations on Phobos are complete, the EEV will travel in an 8.88 hour Hohmann transfer to Deimos that uses 756 m/s of Δv. The descent and ascent from Deimos will each take 12 m/s of Δv. After surface operations on Deimos are complete, the EEV will perform a final Hohmann transfer from Deimos to the DST, which uses 373 m/s of Δv and will take 15.45 hours. The maximum wait times for the transfers to the moons are both under a day and the maximum wait time to return to the DST is approximately five days.

Attitude determination for the EEV will be handled by eight RedWire Coarse sun sensors, each with an accuracy of $\pm 1^{\circ}$, three Ball Aerospace CT2020 star sensors, each with an accuracy of 1.5 arcseconds, and two Advanced Navigation Motus Inertial Measurement Units (IMUs).

The propulsion system of the EEV has two primary mission requirements. The first requirement is providing the necessary thrust to travel from the DST to Phobos, Deimos, and back to the DST. This requirement is met using two Transtar III engines, with each engine capable of producing 16.7 kN of thrust. These engines were determined by conducting a trade study of different engine candidates.

The second requirement of the propulsion system is to provide the thrust needed to make attitude adjustments and rendezvous with both moons and the DST. This requirement is accomplished using 16 R-4D-11 thrusters, with each thruster able to produce 445 N of thrust. Similar to the main engine, a trade study was conducted comparing different thruster candidates. The thrusters are organized into four sets of four, located along the midline of the vehicle, with two sets near the nose of the EEV and the other two sets located near the aft.

With a 10% ullage allowance, the total propellant came out to be 8,138 kg. The propellant selected for Project Chariot uses Monomethylhydrazine (MMH) as a fuel and a mixed oxide of nitrogen, MON 25, as the oxidizer. Using a 2.1 oxidizer to fuel ratio, the masses broke down into 2,625 kg of MMH and 5,513 kg of MON 25, which corresponds to 3,000 L of MMH and 3,900 L of MON 25. The oxidizer and fuel will both be stored separately in two tanks. The fuel tanks are each capable of holding 1,500 L and the oxidizer tanks are each capable of holding 1,950 L. There will also be two pressurization tanks using nitrogen gas. The tanks will be split into two groups for added redundancy, with each set containing one fuel tank, one oxidizer tank, and one pressurization tank. To add stability to the vehicle, one set will be located on the top of the EEV and the other set will be on the bottom.

To protect the two astronauts from environmental factors such as radiation and micrometeorite impacts, the EEV will be equipped with protective measures. A lightweight and effective micrometeorite shielding will be constructed out of six layers of Nextel AF62 and six layers of Kevlar. The primary radiation shielding for the EEV will come from a combination of the Kevlar and the 4.8 mm and 2 mm thick aluminum used for the EEV structure. Due to the risk of excess radiation being released during a solar flare event, additional radiation shielding will be in the form of excess water stored in the walls of one of the EEV rooms.

A combination of batteries and solar arrays will be used to power the EEV. The two UltraFlex solar arrays from ABLE Engineering will handle the daily power use and three batteries will be used during peak power usage times, such as during sample collection. The batteries will provide 8.4 kWh and the solar arrays will provide 9.4 kW of power. To regulate the thermal needs of the EEV and the astronauts, the EEV will use multi-layer insulation (MLI), Kapton heaters, and radiators.

The communications architecture of Project Chariot ensures that the EEV can communicate with other assets in Mars orbit and with Earth whenever necessary. The EEV's primary point of contact is the DST, which will relay transmissions to and from the Deep Space Network (DSN) on Earth. When the DST is not in view, the EEV can relay with the Next Mars Orbiter (NeMO) to maintain the link. The bit rates required to transmit the necessary data types for the mission have been determined and met. The EEV is equipped with a suite of antennas designed and optimized for this mission. The ultra-high frequency (UHF) antenna is used to communicate with the DST and NeMO. The X-band high gain antenna (HGA) is used to communicate with the DSN. The Ka-band low gain antennas (LGAs) are used to communicate with the DST and the DSN, backing up the UHF and HGA. Each antenna is tasked with transmitting different combinations of data types at specific times during the mission. The command of the EEV and data storage will be handled by a system of four flight computers.

Because of the Starship's large payload capability, the primary constraint on the mission mass came from budgetary limits. A detailed mass breakdown shows that the dry mission mass of the EEV, with individual growth allowances factored in, totals 5,739 kg. The wet mass, which is shown in section 6.7, comes to a total of 12,889 kg.

The cost breakdown associated with Project Chariot was constrained to \$1 billion. Through the utilization of current equipment and the small size of the EEV, the mission is projected to cost just over \$914 million, including component-specific growth allowances averaging 18.1%.

Due to the inclusion of a crew for this mission, the most important risks revolve around scenarios that result in either injury or loss of life for an astronaut. Two risk analysis summaries are included: one for mission-affecting risks and one for human-affecting risks. Both summaries discuss mitigation strategies incorporated into the mission design to lower either the likelihood of occurrence or the severity of the risk.

Table Of Contents

List of Figures

List of Tables

List of Abbreviations

4BCO2	Four Bed Carbon Dioxide Removal System
8-PSK	8 Phase Shift Keying
ADCS	Attitude Determination and Control System
AIAA	American Intuition of Aeronautics and Astronautics
ATO	Attitude, Trajectories, and Orbits
BER	Bit Error Rate
BPSK	Binary Phase Shift Key
CCAA	Common Cabin Air Assembly
CC&DH	Communication, Command, and Data Handling
CDRA	Carbon Dioxide Removal System
CEA	Chemical Equilibrium Applications
CNR	Carrier to Noise Ratio
CONOPS	Concept of Operations
COPV	Composite Overwrapped Pressure Vessel
DSN	Deep Space Network
DST	Deep Space Transport
Eb/No	Energy Per Bit to Noise Power Spectral Density Ratio
EEV	Exploration Excursion Vehicle
EIRP	Effective Isotropic Radiated Power
EMU	Extravehicular Mobility Unit
ESA	European Space Agency
EVA	Extravehicular Activity
GC	Gas Chromatography
GMAT	General Mission Analysis Tool
GPR	Ground-Penetrating Radar
HD	High Definition
HFLS	Human Factors and Life Support
HGA	High Gain Antenna
IMU	Inertial Measurement Unit
ISRU	In-Situ Resource Utilization
ISS	International Space Station
ITMS	Ion Trap Mass Spectrometer
IVA	Intravehicular Activity
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LiDAR	Light Detection and Ranging
MAVEN	Mars Atmosphere and Volatile Evolution
MLI	Multi Layer Insulation
MMH	Monomethylhydrazine
MOMA	Mars Organic Molecule Analyzer
MON	Mixed Oxides of Nitrogen
MRO	Mars Reconnaisance Orbiter
NAC	Needs, Alterables, and Constraints
NASA	National Aeronautics and Space Administration
NeMO	Next Mars Orbiter
NORS	Nitrogen Oxygen Recharge System
NTO	Nitrogen Tetroxide
O/F	Oxidizer to Fuel Ratio
PLSS	Personal Life Support System
PROP	Propulsion
PSK	Phase Shift Keying
PT&E	Power, Thermal, and Environment
QMS	Quadrupole Mass filter Spectrometer
QPSK	Quadrature Phase Shift Keying
RFP	Request for Proposal
RTD	Resistance Temperature Detector
SDC	Student Dust Counter
SSD	Solid State Drive
S&LV	Structures and Launch Vehicle
TOF	Time of Flight
UDMH	Unsymmetrical Dimethylhydrazine
UHF	Ultra-High Frequency
USD	United States Dollar
UWMS	Universal Waste Management System

List of Symbols

bps	Bits per Second
m^3	Cubic Meters
dB	Decibel
dBm	Decibel Meter
dBW	Decibel Watt
$\circ$	Degree
$^\circ\mathbf{C}$	Degrees Celsius
Δv	Delta V
FPS	Frames per Second
$g's$	G-Forces
GB	Gigabyte
GHz	Gigahertz
GPa	Gigapascal
g	Gram
g/cm^3	Grams per Cubic Centimeter
K	Kelvin
$kbps$	Kilobits per Second
kg	Kilogram
kHz	Kilohertz
km	Kilometer
km/s	Kilometers per Second
km^2/s^2	Kilometers Squared per Seconds Squared
kN	Kilonewton
kPa	Kilopascal
kW	Kilowatt
kWh	Kilowatt Hour
L	Liter
$Mbps$	Megabits per Second
MHz	Megahertz
MPa	Megapascal
m	Meter
m/s	Meters per Second
m^2	Meters Squared
μm	Micron
N	Newton
Pa	Pascal
s	Second
Isp	Specific Impulse
V	Volt
W	Watts
W/kg	Watts per Kilogram
W/m^2	Watts per Meter Squared

1 Introduction

1.1 History and Background

To prepare for the eventual landing of the first humans on Mars, crewed missions must first be flown in deep space to test new technologies and gain insight of what humans are capable of under such conditions. This evaluation is one of the overarching purposes of Project Chariot, right after the science objective of sample return from Phobos and Deimos. The ability of the Deep Space Transport to support a crew in transit to Mars will be tested, and the Exploration Excursion Vehicle will allow this proving ground mission to pursue science objectives and land humans on Phobos an Deimos for the first time. In years past, a mission as involved and groundbreaking as Project Chariot might have been too expensive to serve as a precursor to a larger human mission, but recent advancements in commercial spaceflight have opened the door for producing the systems needed to land on Phobos and Deimos without diverting monetary resources from the larger Mars effort. The goal of Project Chariot is to design and produce a low-cost vehicle capable of landing the first humans on Phobos and Deimos, both to collect and return regolith samples and to establish a proving ground for astronauts and the systems required to support them in deep space [1].

1.2 Problem Statement

As requested by the American Institute of Aeronautics and Astronautics (AIAA) Foundation in the request for proposal (RFP) for the 2022 Team Space Design Competition, Project Chariot has designed an Exploration Excursion Vehicle (EEV) to support two astronauts for a 30-day sample collection mission to Phobos and Deimos. While in a 5-sol orbit around Mars, the EEV must autonomously dock with the Deep Space Transport (DST) vehicle on which the astronauts arrive. After the EEV departs from the DST, it will land on both moons and collect a minimum of 50 kg worth of samples from each to be returned to Earth.

1.2.1 Scope of the Problem

The scope of the problem is defined by the given requirements and constraints from the RFP and the requirements derived by the team to ensure mission success. The RFP defines mission success as landing two astronauts in an EEV onto both Phobos and Deimos and returning with a minimum of 50 kg of samples from each moon [1]. Subsystems on which the RFP requests detail include: science objectives; the mission's trajectories and orbits; launch vehicle selection; EEV structure; power and thermal systems; communications, command, and data handling; life support; and propulsion systems. The scope will include the design of the EEV, interplanetary transfer, and sample collection. Design decisions will be determined using trade studies that evaluate all viable solutions. The mission is constrained by a budget of 1 billion United States dollars (USD) [1].

The mission duration between departure and return to the DST is planned to be no longer than 30 days, and the EEV must be designed to sustain two astronauts on board for at least 30 days to account for potential changes in schedule and emergency situations. The RFP allows for a choice of one or two sorties from the DST to land on the moons [1]. The team has determined the mission will be completed by conducting one sortie to visit both moons, increasing the likelihood of success. This constraint will result in fewer orbit transfers and lower Δv requirements.

The mission shall conduct meaningful science experiments while on the surface of the moons and in orbit around the moons and Mars [1]. As required by the RFP, the EEV must be capable of collecting at least 50 kg of regolith samples from both moons. The EEV will have a propulsion system designed to accommodate maneuvers to and from the moons. The EEV will be designed with no extravehicular activity (EVA) operations in mind. Secondary science objectives can be accomplished using a maximum of 200 kg of scientific equipment that may be transferred from the DST to the EEV along with the crew [1].

The RFP requires that mission operations and a timeline be specified. The EEV shall reach a 5-sol parking orbit around Mars prior to the crew's arrival in the DST, scheduled for January 1, 2040 [1]. The DST will dock with the EEV at the start and end of the 30-day mission to transfer the crew to and from the vehicle. The RFP also states that the transfer orbits to go from Earth to Mars, and the transfers to and from each of the moons, shall be designed [1].

The design of the EEV mission will start with the launch from Earth and continue to the 5-sol orbit insertion and docking with the DST. Travel to Phobos and Deimos, as well as operations on both moons, will take place while the crew is aboard the EEV. The mission ends when the EEV docks with the DST after visiting both moons. The mission architecture will be designed with full consideration of all requirements and constraints of the RFP.

1.2.2 Needs, Alterables, and Constraints

The key parameters that drove the EEV design are listed in Table 1. Needs can be defined as the mission's objectives; failure to meet these needs results in mission failure. Constraints refer to the constant parameters of this mission within which the design must operate to meet the needs of this mission. Alterables can be defined as parameters that can be varied to best suit the EEV to the needs and constraints of this mission.

Table 1: Needs, alterables, and constraints of the mission. Alterables are the parameters that can be changed and constraints are constant parameters.

Needs	***Alterables***	***Constraints***
Supports two crew members	One or two sorties from the DST	30-day mission
Must be able to perform automated sample retrieval	Mission modes	Crew operates from inside the EEV, no planned EVA
Docking with DST	Maneuvers for the round trip	Quarantine samples from crew before Earth arrival
Minimum of 50 kg of samples from each moon	Sample retrieval method	200 kg maximum science equipment delivered to EEV from DST
EEV must be in 5-sol orbit on 01/01/2040	Propulsion system	Cost must stay under 1 billion USD
	Launch vehicle	EEV must dock autonomously

1.3 Mission Requirements

In order to properly determine the scope of the mission, eight requirements were derived from the RFP, shown in Table 2, were determined early in the design process. Several factors were considered when determining the mission requirements, including, but not limited to, the problem statement, the science objectives, and the mission's crew. It is these requirements that set the frame for the subsystem requirements in the following sections.

Table 2: Requirements for the mission design. The index M refers to any overall mission requirements.

Index	***Requirement***
M-1	The mission shall support two crew members.
M-2	The mission shall include up to 200 kg of scientific equipment brought from the DST to the EEV.
M-3	The mission shall collect a minimum of 50 kg worth of samples from each moon.
M-4	The mission shall quarantine each sample from the crew.
M-5	The EEV shall be in a Mars 5-sol orbit by January 1, 2040.
M-6	The EEV shall autonomously dock with the DST.
M-7	The mission shall cost no more than $1 billion USD.

1.4 Overarching Design Decisions

The overall designs of the mission and the EEV were driven by several factors unique to the mission requirements. These include the budget restrictions and resulting mass constraints for the system, the science objectives for the mission, the microgravity of the two moons, and the crew requirements.

One constraint that was a major consideration is the $1 billion cost budget. This budget limited the launch vehicle options drastically, resulting in many mass and cost cuts throughout the design process.

The requirement of collecting at least 50 kg of regolith from both Phobos and Deimos guided the general decisions when it came to the mission design. From this requirement, the EEV is able to collect separate samples from each moon for primary analysis onboard and a more detailed analysis on Earth. The sample collection combines two methods, one for the surface-level regolith, and one for samples collected meters below the surface.

While collecting the regolith samples, the EEV will have to battle the extremely low gravity on Phobos and Deimos. This challenge drove the overall design of the EEV to have a horizontal layout for stability and legs designed specifically for this mission. Tackling this challenge required the testing of several leg designs before determining the ideal leg for the vehicle.

The most prevalent challenge for this mission was the requirement to have two crew members on the 30-day mission. Having a crewed mission increased risks immensely and forced many design decisions such as the habitable volume requirement, oxygen, meals, and other necessities to support the life of the crew members. A visualization of the systems that comprise the EEV is included in Figure 41 in Appendix A.

1.5 Concept of Operations

Project Chariot's concept of operations (CONOPS) is shown in Figure 1. Included on the CONOPS is the projected mission timeline, with the Starship launch vehicle leaving Earth with the EEV on November 19, 2036, inserting the EEV into the 5-sol Mars orbit on January 28, 2039, rendezvousing with the astronauts on the DST on January 1, 2040, and completing the mission by January 27, 2040. Although it is projected to take well under a day to collect samples from each moon, there are several days allotted for staying on each moon as a contingency in case repairs are needed or if sample collection takes longer than expected. In addition to this, the transfers between Phobos, Deimos, and the DST have the maximum possible wait times built into the schedule in case the EEV departs from the DST or one of the moons early or late.

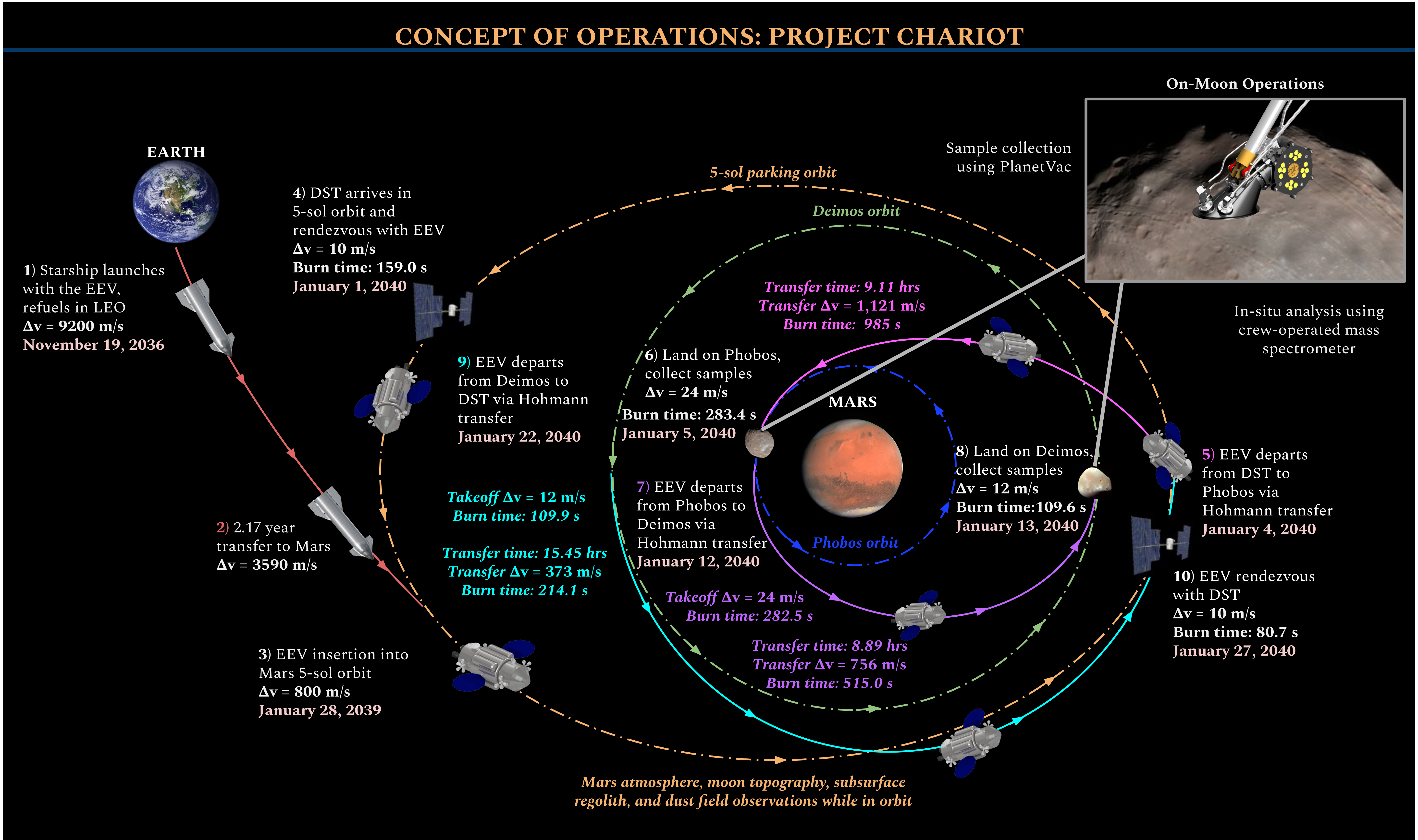


Figure 1: Project Chariot mission concept of operations. This figure details the various mission segments with any relevant dates, transfer times, burn times, and needed Δv.

2 Science Mission

2.1 Introduction

Project Chariot seeks to understand the composition and origins of Phobos and Deimos, guided by the National Academies' Decadal Survey on planetary science for the decade 2013-2022. There are two leading theories regarding the origins of Phobos and Deimos. The giant impact theory speculates that the moons took shape from the accretion of material from a debris disk that formed around Mars as a result of a giant impact. The captured asteroid theory states that Phobos and Deimos are nothing more than asteroids captured by the gravitational pull of Mars [2]. Retrieving samples from both moons presents an opportunity to discover their origins. Project Chariot will additionally pursue other science objectives, including gaining insights on the prospect of in-situ resource utilization (ISRU) with moon material, mapping the topography of the moons, mapping dust fields around the moons, and performing remote sensing of Mars' atmosphere and the space around the moons.

2.2 Science Objectives

Sample collection and return is the primary science objective of the mission. To investigate the origin of Phobos and Deimos, the chemical and mineral composition of the regolith will be analyzed for certain materials. The orbit characteristics of Phobos and Deimos support the giant impact theory. The fragments of Mars kicked up by the giant impact that would have formed the disk were likely heated to approximately 2,000 K. If the moons were formed in this way, the impact and high temperature would result in Phobos and Deimos being composed of glassy or recrystallized igneous materials. The extreme temperature would have caused volatile materials, such as water and organic compounds, to be lost. The captured asteroid theory suggests that Phobos and Deimos could be D- or T-type carbonaceous chondrite asteroids. The presence of chondritic materials such as phyllosilicates, carbonates, oxides, and organic compounds would provide proof of the validity of this theory [2].

A secondary objective related to the primary regolith investigation is to determine the viability of ISRU with Phobos and Deimos material. This objective contributes to the National Aeronautics and Space Administration (NASA) investment in ISRU technologies to support long-duration human spaceflight. The purpose of ISRU is to use planetary, lunar, or asteroid materials to produce spaceflight essentials such as water, propellant, or life support consumables. Project Chariot's ISRU investigation will focus on the possibility of producing propellant from the moons' regolith, as there is unlikely to be a significant amount of water ice on the moons [3].

Project Chariot will pursue a variety of secondary remote sensing objectives. The Decadal Survey advises that trace gases and carbon compounds in the Martian atmosphere and their dynamics should be studied to aid in researching the moons' evolution [4]. The presence of volatile compounds in the Martian atmosphere and around Phobos and Deimos will be investigated to build upon previous missions. The topographies of the moons shall be mapped from orbit in order to gain further insight into the formation of Phobos and Deimos and aid in landing operations. To supplement the sample collection operation, the composition of subsurface regolith that sits deeper than can be accessed by the sample collection system will be studied. Project Chariot will also investigate the presence and density of dust fields around Phobos and Deimos.

2.3 Science Instrumentation

Regolith samples will be analyzed both on the EEV during the mission and on Earth after the mission. Astronauts will use an ion trap mass spectrometer (ITMS) to perform in-situ analysis and obtain preliminary results. To better analyze the complex regolith mixture, gas chromatography (GC) will be used to introduce the sample into the ITMS. ITMS systems are compact and low-mass but have finite ion volume, limited mass resolving capabilities, and produce semi-quantitative measurements of molecular abundances, so the validation of initial results on Earth is necessary [5].

A quadrupole mass filter spectrometer (QMS) will be used to observe the Martian atmosphere for carbon compounds, volatile compounds, and trace gases. The QMS was selected because of its extensive flight heritage investigating planetary atmospheres, including the Neutral Gas and Ion Mass Spectrometer on the Mars Atmosphere and Volatile Evolution (MAVEN) mission [5].

A light detection and ranging (LiDAR) sensor will be used to visualize the topographies of Phobos and Deimos and to aid in landing. A LiDAR system measures the distance between the spacecraft and the target surface using a pulsed laser, resulting in three-dimensional images of the target surface [6]. Because a LiDAR is also useful in the application of rendezvous and docking, the EEV will be equipped with two LiDAR systems: one will be placed on the side with the DST docking port, and the other will be on the side facing the surface of the moons while landing.

The EEV will be equipped with a ground-penetrating radar (GPR) to image below the surface of Phobos and Deimos. The GPR will be operated from orbit and will collect data on the composition of regolith below the depth reachable by the sample collection system [7].

To investigate dust in the vicinity of the moons, the EEV will be equipped with a dust counter. The dust counter system is modeled after the Venetia Burney Student Dust Counter (SDC), built by the University of Colorado Boulder for the New Horizons mission. The SDC uses an array of Polyvinylidene Fluoride detectors that send signals to the attached processor whenever they are struck by a dust particle. This system allows for a map of the dust field around Phobos and Deimos to be constructed [8]. A list of the instruments discussed in this section along with their mass, power, and cost estimates is included below in Table 3. Appendix B includes a view of Project Chariot's overall science mission in Table 43, the science traceability matrix.

Table 3: Project Chariot science instrumentation. This table includes the instruments on the EEV and the objectives they fulfill. Also shown are mass, power, and cost estimations.

Instrument	***Objective***	***Mass** (kg)*	***Power** (W)*	***Cost** (USD)*
PlanetVac	Sample Collection	20	20	2,000,000
ITMS	In-Situ Sample Analysis	12	82	20,000,000
QMS	Atmospheric Remote Sensing	30	125	35,000,000
LiDAR	Topography Mapping	30	35	20,000,000
GPR	Subsurface Regolith Analysis	10	1,000	1,000,000
Dust Counter	Dust Field Mapping	2	10	300,000

2.4 Sample Collection and Handling

Sample collection will be handled by the PlanetVac, a system produced by Honeybee Robotics. The PlanetVac's nozzle touches down on a regolith-covered surface and sprays pressurized gas to redirect regolith into a feed line that leads to storage receptacles. The gas-expended to sample-collected ratio for the PlanetVac is 1:1,000 [9]. To enhance the system, the Autogropher II space-rated piezoelectric drill, produced by the Jet Propulsion Laboratory (JPL) and Honeybee Robotics, will be integrated into the PlanetVac's nozzle. This drill is assisted by a piezoelectric actuator to ensure low force drilling, suitable for the microgravity operations relevant to this mission. The Autogropher II will allow the system to access greater depths and reduce sample collection time by dislodging packed regolith [10]. A diagram of the PlanetVac and drill system is in Figure 2. The path the regolith takes after being collected by the PlanetVac is described in Figure 3.

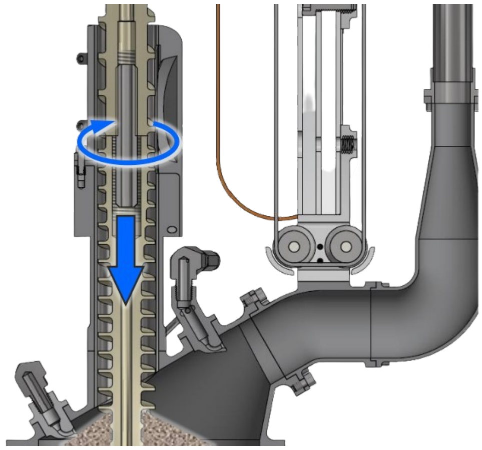


Figure 2: PlanetVac and drill system. This figure shows the schematic of the PlanetVac with attached piezoelectric drill system. The drill dislodges the regolith from the surface and the pressurized gas flow of the PlanetVac directs it up the feed line into a storage unit [11].

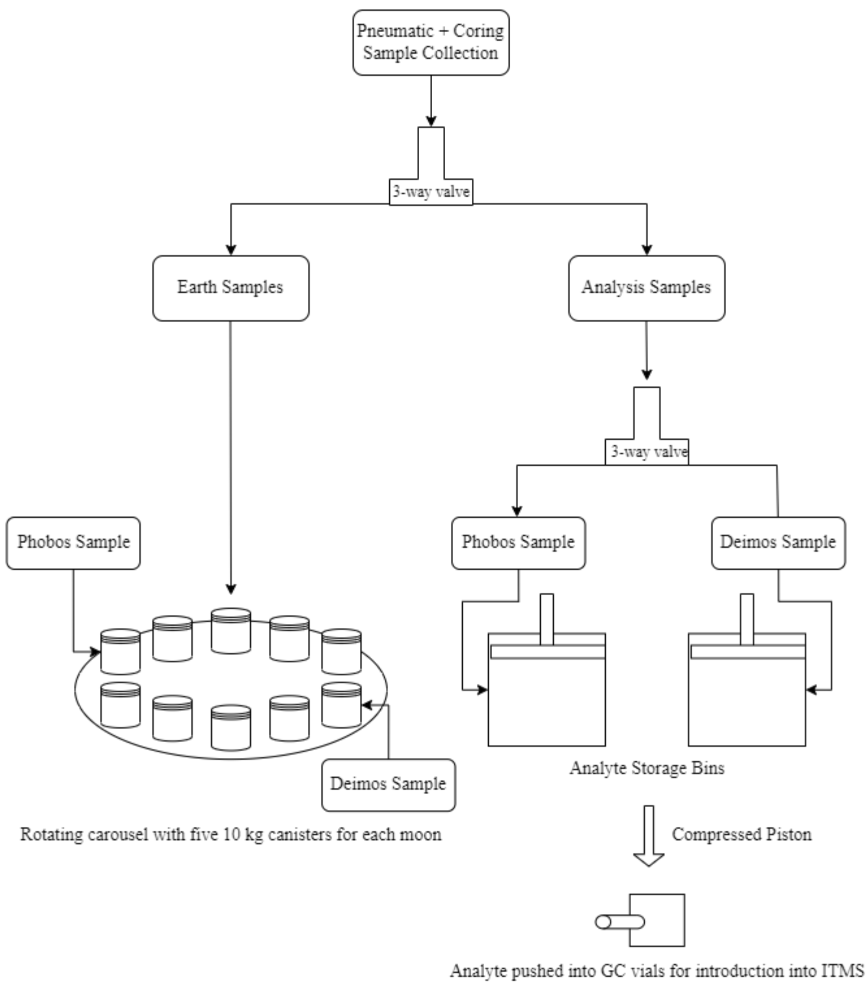


Figure 3: Sample collection and handling system block diagram. This shows the path of regolith sample after it is collected by the PlanetVac.

Regolith samples are directed through a primary three-way valve by the gas flow, first to the area where samples bound for Earth are stored. There are ten 10 kg canisters for sample storage, five for each moon. Splitting the sample storage unit into individual canisters reduces the risk of losing regolith in the case of a broken container. These canisters are fixed to a rotating carousel that turns after the current canister is filled and automatically sealed, positioning the next empty canister underneath the feed line. When the required 50 kg of sample is collected, the flow is diverted to the analysis area to collect about 1 kg of regolith for study by the crew. There is a second three-way valve that divides Phobos and Deimos samples; the extra sample, called analyte, is directed into a storage bin. The storage bins are equipped with pistons that push the analyte into a vial that will be used to introduce the sample into the ITMS by GC. The schematic detailing this process is displayed in Figure 4. The entire system is cleaned by the backflow of gas that originates from the PlanetVac, preventing cross-contamination between the samples of both moons.

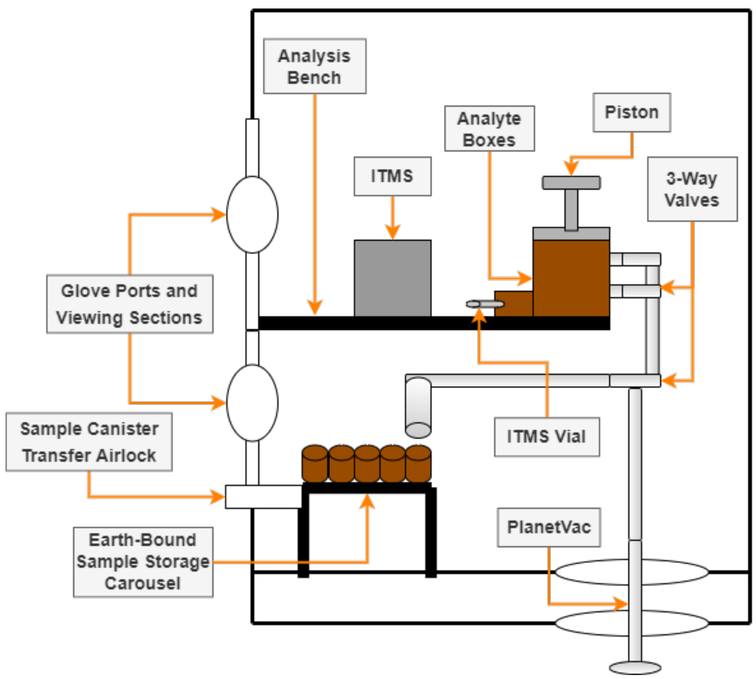


Figure 4: This shows the physical layout of the sample collection and handling system concept. Systems not pictured in Figure 3, such as the glove ports and transfer airlock, are shown here. For a cleaner diagram, five of the ten Earth-bound sample storage canisters are displayed.

The sample collection and handling system is housed in a depressurized module in the rear of the EEV to ensure that accessing the surfaces of the moons can be easily accomplished. The PlanetVac is deployed to the surface by extending through a dual-hatch system that allows the module to be separated from the space environment even if the outside hatch were to become disabled. To assist in the transfer of the sample canisters from the EEV to the DST upon mission completion, a small airlock system will allow canisters to move from the depressurized section of the EEV to the pressurized section. Glove ports and transparent viewing sections are added to the wall of the module to allow the crew to interact with the system without directly handling the regolith. The upper glove ports give the crew access to the analysis bench, which holds the ITMS and analyte storage bins. The lower glove ports allow the crew to move sample canisters from the carousel to the transfer airlock. The design and usage of the glove access system will be held to the same standards of safety and robustness as systems used to handle biohazard materials on Earth.

2.5 Landing Sites

Due to propellant constraints and lack of EVA capability, only one landing site can be accessed on each moon. The possible landing sites must balance scientific interest and mission concerns such as solar power generation and communications access. On both moons, the north pole would receive the most sunlight during the crewed 30-day mission since it will be summer in the northern hemisphere of Mars. A visualization of the sunlight exposure of the north pole on both moons is shown below in Figure 5 [12]. On Phobos, the site of greatest scientific interest is the Stickney Crater, a depression measuring over 9 km in diameter. Due to its diameter relative to the total diameter of Phobos, the impact that created the crater was close to shattering the moon. Despite Phobos' minuscule gravity, streaks observed on the sides of the crater suggest that material has descended the walls over time [13]. Because Project Chariot must characterize the regolith composition of the moons, it will be more useful to land in a more general location to establish a benchmark for regolith composition. Due to its size and difference from other regions of the moon, Stickney Crater deserves its own mission. Deimos is much smaller than Phobos and lacks standout sites such as Stickney Crater, so the decision of where to land is much more straightfoward. Landing in the north polar regions of both moons is preferred to access as much sunlight as possible for power generation.

(a) (b)

Figure 5: Sunlight exposure on the north poles of (a) Phobos [12] and (b) Deimos [12]. These visualizations show where the maximum amount of sunlight shines on Phobos and Deimos during Mars' north hemisphere summer. A landing site in the polar regions would be optimal.

3 Human Factors and Life Support

The Human Factors and Life Support (HFLS) requirements in Table 3 were key driving factors in the design of this EEV. These requirements were taken from NASA Human Spaceflight Standards [14]. A few of the most important requirements mentioned are air supply, water supply, nutrition, habitable climate, medical care, fire suppression, and waste management.

Table 4: The HFLS requirements for making the EEV compatible for two humans for 30 days in space. These are all in accordance with NASA Human Spaceflight Standards [14].

Index	***Requirement***
HFLS-1	The EEV shall be equipped with physical capabilities and characteristics to include provisions such as adaptations for body dimensions, range of motion, and accommodations for external factors such as gravity and pressure.
HFLS-1.1	The EEV shall have a volume habitable for the two crew members along with equipment and other components for the mission.
HFLS-2	The EEV shall be designed with natural and induced environments to support human life.
HFLS-2.1	The EEV shall maintain an internal atmospheric composition that provides safe and breathable air for the crew and the ability to adjust composition given specific needs.
HFLS-2.2	The EEV shall be capable of maintaining a safe atmospheric pressure with the capability to adjust the pressure if needed.
HFLS-2.3	The EEV shall provide the crew with safe and habitable levels of humidity in the cabin.
HFLS-2.4	The EEV shall maintain safe and habitable interior temperatures to support human life.
HFLS-3	The EEV shall be designed with features that support human occupancy.
HFLS-3.1	The EEV shall include provisions for potable water for crew consumption, food rehydration, personal hygiene, and medical needs.
HFLS-3.2	The EEV shall include provisions for storing and maintaining food, nutrition, and its safety throughout the mission in accordance with macronutrient and micronutrient requirements for the crew.
HFLS-3.3	The EEV shall provide the crew with the necessary environments and facilities for oral hygiene, personal grooming, and body cleansing.

Continued on next page

Table 4 – *Continued from previous page*

Index	***Requirement***
HFLS-3.4	The EEV shall be designed to have a human waste management solution isolated from food areas and mission operations for hygienic purposes.
HFLS-3.5	The EEV shall be designed to include an isolated trash management system to stow trash including wet and dry trash, sharp items, and biohazardous and radioactive waste for hygienic purposes.
HFLS-3.6	The EEV shall provide the crew with medical treatments and volume to administer medical care to the crew.
HFLS-3.7	The EEV shall provide surface area and volume for the crew to sleep in expected gravity conditions.
HFLS-3.8	The EEV shall provide the crew with clean, durable, and sufficient clothing to suit crew operations.
HFLS-3.9	The EEV shall provide access to areas for inspection and removal of contaminants as well as sanitization methods and materials.
HFLS-3.10	The EEV shall include recreational capabilities for the maintenance of the crew's behavioral and psychological health.
HFLS-4	The EEV shall be supplied with hardware and equipment for the crew to utilize in various expected scenarios.
HFLS-4.1	The EEV shall be supplied with emergency provisions to support the crew in the event of an emergency situation.
HFLS-5	The EEV shall be equipped with crew interfaces that provide health and system data for the crew to make any necessary adjustments.
HFLS-5.1	The EEV shall contain a display for attitude control information.
HFLS-5.2	The EEV shall be equipped with a display for communication with the ground team on Earth.
HFLS-5.3	The EEV shall employ autonomous mission operations control.
HFLS-6	The crew shall be supplied both IVA spacesuits and an EVA spacesuit to support mission operations.
HFLS-6.1	The crew shall be supplied IVA spacesuits for each astronaut to successfully complete mission operations over the 30-day duration.

Continued on next page

Table 4 – *Continued from previous page*

Index	***Requirement***
HFLS-6.2	The crew shall be supplied with an EVA spacesuit to perform any extra vehicular activity if the situation arises.

3.1 Physical Characteristics and Capabilities

The EEV module must be capable of sustaining two crew members for the entire 30-day mission. Thus, it is necessary to describe life support provisions in the EEV design, including recreation and sleeping, food and nutrition, and personal hygiene.

3.1.1 Task-Based Analysis

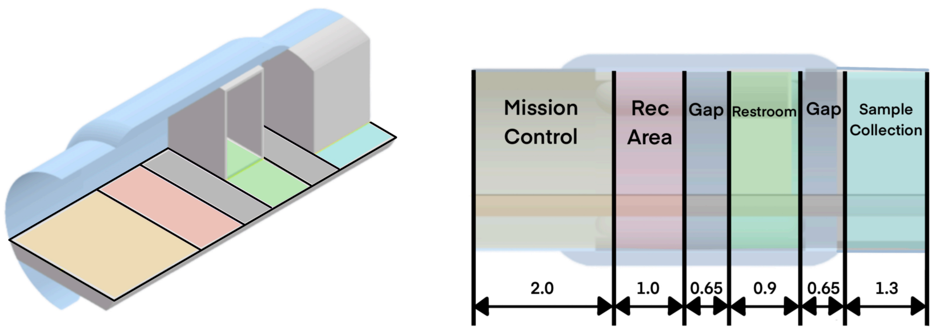
When initially sizing the EEV, the Task Analysis Method was used to first determine the minimum habitable volume needed for the crew. As described by NASA's Human Integration and Design Handbook, the Task Analysis Method involves determining the volumes required for the different tasks the crew will be preforming during the mission, and then combining various volumes to determine the total habitable volume the spacecraft must have [15]. For the purposes of this paper, habitable volume refers to the volume that is accessible by the crew.

Due to the requirement of fitting within the launch vehicle fairing, only tasks essential to the mission and the crew were considered. This includes mission operations, personal hygiene, exercise, nutrition, and sleeping. Furthermore, some tasks can occupy the same volume, such as the crew being able to eat meals in their seats or being able to change clothing in the restroom. The conclusion of the Task Analysis Method for this mission was a minimum habitable volume of 23 m^3. The overall results from this analysis are the primary topic of the next section.

3.1.2 EEV Internal Configuration

The inside of the EEV, as seen in the CAD model in Figure 6a, consists of four modules that are labeled in Figure 6b. The first module follows the conical section of the EEV and serves as the mission control area. While this module is depicted as an open space in the figures, crew interfaces and ergonomic chairs will be installed; the chairs will be able to rotate and serve as a dining area for the crew. The second section is the recreational area, which has two collapsible sleeping quarters with vertical beds installed on the walls of the EEV. This area is shown as an open space in the figures and also doubles as a fitness area.

The third module of the EEV is a personal hygiene cabin which contains waste management tools along with storage for clothing and additional personal hygiene items. The cabin is shown in Figure 6a as an open quarter and also serves as a changing area for the crew. The fourth section of the EEV, shown as a closed area in the back, is the sample collection and analysis module; this section is the only part of the EEV which is unpressurized. The sample collection cabin is accessible to the crew via glove ports which do not interfere with the pressurized sections of the EEV.



(a) (b)

Figure 6: (a) A cross section of the EEV, each section is highlighted in a different color. (b) The side view of the interior cross section, all dimensions are in meters.

3.2 Natural and Induced Environments

3.2.1 Internal Atmosphere

Air supply is a vital component to any crewed space mission. One major concern with air supply is providing adequate oxygen. Humans breathing insufficient oxygen for long periods of time can develop conditions such as pulmonary or cerebral edema. These conditions can affect cognitive function and could result in the astronauts making dire mistakes. In order to mitigate this risk, an adequate supply of oxygen and nitrogen will be provided for the crew. The air supply for Project Chariot is divided as shown in Table 5.

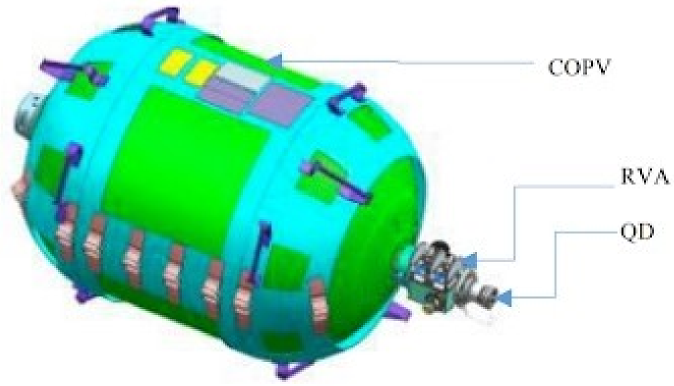
The standard atmospheric composition of air at sea level on Earth consists of 78% nitrogen, 21% oxygen, 0.93% argon, 0.04% carbon dioxide, and 0.03% trace amounts of other gases [16]. The HFLS division plans to provide an environment on the EEV similar to this atmospheric composition, with a mixture of 78% nitrogen and 22% oxygen. This composition was chosen in order to save on cost, mass, and volume by limiting the number of tanks required. The percentages of nitrogen and oxygen were chosen to provide maximum comfort and similarity to atmospheric conditions. These percentages were used along with the volume of the EEV and the fact that each astronaut uses approximately 0.84 kg of oxygen per day to determine the quantity of each gas needed [17]. As a result, the EEV will be pressurized with 22% oxygen and tanks will contain the rest of the supply, including a 50% safety margin. Nitrogen is breathed in and exhaled out by the astronauts, so minimal nitrogen is consumed. The quantities of oxygen and nitrogen needed on the mission are contained in Table 5.

Table 5: Air supply provided for the 30-day duration of the Project Chariot mission broken down by composition. The needed mass of both oxygen and nitrogen are increased by 50% to ensure the crew does not run out of breathable air.

Air Supply	***Makeup (%)***	***Mission Mass (kg)***	***50% Margin (kg)***	***Total Mass (kg)***
Nitrogen	78	27.3	13.6	40.9
Oxygen	22	53.0	26.5	79.4
Total:	**100**	**80.2**	**40.1**	**120.3**

The air supply for this mission will be stored in multiple pressurized tanks, which allows for redundancy in the event of tank failure. Oxygen will be stored in two Northrop Grumman composite overwrapped pressure vessel (COPV) tanks, Figure 7a, that can hold a maximum of 35.4 kg, or 81.4 L, of oxygen at a pressure of 331 bar with a tank mass of 12.7 kg [18]. The two tanks will each be filled with 35.35 kg of oxygen, resulting in a total of 70.7 kg. The remaining 8.8 kg will be stored in the pressurized portion of the EEV in preparation of the crew's arrival. The nitrogen for this mission will be stored in a NASA Nitrogen Oxygen Recharge System (NORS) tank, Figure 7b, which is designed to hold up to 27.2 kg of nitrogen at a pressure of 414 bar for a volume of 68.8 L [19]. Due to the mass of this tank and the decreased importance of nitrogen to human life, only one tank will be used to store the nitrogen. The habitable volume of the EEV shall be pressurized with 27.3 kg of nitrogen and the remaining 13.6 kg will be stored in the NORS tank.

(a)



(b)

Figure 7: Air supply storage tanks for this mission. (a) Northrop COPV oxygen tanks [20]. The composite overwrap can be seen on the pressure vessel along with the valves and gauges to the right. (b) NASA NORS tank for nitrogen storage [18]. The regulators and pressure gauges can be seen to the right of the tank.

3.2.2 Environmental Regulation

The EEV shall be equipped with pressure gauges on the oxygen and nitrogen tanks in order to monitor air supply and pressure. The NORS tank comes pre-equipped with a pressure gauge and the oxygen tanks have gauges on the outflow port [18] [19]. Additionally, the cabin will be equipped with multiple DPG280-100G advanced digital pressure gauges from Omega™ to monitor the cabin conditions [21]. For maximum comfort, the pressure of the habitable zone in the EEV shall be kept at 1.013 bar to simulate the pressure on the surface of Earth. Pressure will be controlled with pressure regulators and valves on the air supply tanks, allowing the proper amount of nitrogen and oxygen into the habitat. Measurements from all pressure gauges will be displayed on the crew interface system which links to the pressure regulators, allowing the crew to control the cabin pressure.

The temperature and humidity of the EEV habitable zone must be maintained within a range that can comfortably support human life. According to NASA human spaceflight standards, the temperature must be between 18 ℃ and 27 ℃ and the humidity must be between 25% and 75% [14]. The temperature will be measured using three ProSense Pt100 resistance temperature detectors (RTD) [22] placed throughout the EEV cabin; the crew interface system will display the temperature. Both the temperature and humidity ranges will be met through the utilization of a Common Cabin Air Assembly (CCAA).

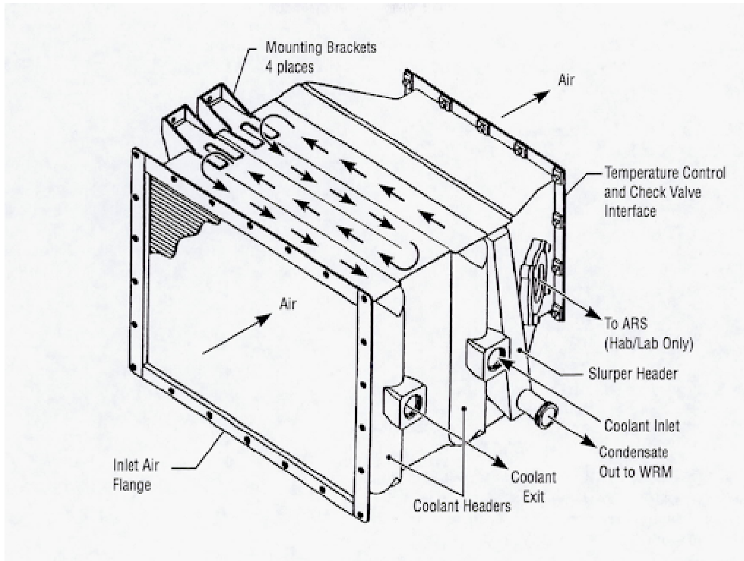


Figure 8: Common cabin air assembly unit outfitted with heat exchanger layers [23]. This will be utilized on the EEV to maintain the temperature and humidity of the cabin.

A CCAA, shown in Figure 8, is an atmospheric heat removal system that removes both heat and humidity from the cabin using a heat exchanger with coolant layers made of stainless steel. The CCAA comes equipped with pressure sensors and fans to cycle air through the system, after which air can be drawn through the coolant layers to condense and extract water vapor from the air while also cooling [23]. Air can also bypass the coolant layers in order to increase cabin humidity or temperature [23]. Along with the CCAA assembly and temperature sensors, the EEV will use the thermal insulation and shielding techniques described in section 6 of this report to maintain the temperature within the specified range.

3.2.3 Carbon Dioxide Removal System (CDRA)

A critical component of supporting human life in space is the regulation of carbon dioxide in the cabin air supply; too much carbon dioxide in the air can have catastrophic consequences for the crew. Due to the importance of carbon dioxide regulation, the EEV will be outfitted with a four bed carbon dioxide removal system (4BCO2). The 4BCO2 is currently being tested on the ISS; assuming the continued success and development of this system, it will be selected for this mission [24]. The 4BCO2 is roughly the size of a small refrigerator, as seen in Figure 9, has a mass of about 200 kg, and uses a four bed system with porous zeolite that sequesters carbon dioxide directly out of the air [17]. On average, a human expels approximately 1 kg of carbon dioxide per day [25]; with a two person crew, this rate becomes 2 kg per day. This system is capable of removing carbon dioxide at a target rate of 3.6 kg per day if operating at 266.645 Pa [17], providing a 80% margin on the needed rate stated above. The 4BCO2 system is designed to be more effective, efficient, and easier to use than previous NASA CDRA systems [24].

Figure 9: Four bed carbon dioxide removal unit [24]. This will be used to ensure the crew is not exposed to dangerous levels of carbon dioxide.

3.2.4 Air Filtration

To keep the cabin air clean and free from dust, bacteria, and pathogens, the EEV will be equipped with an Airocide HD-1500 industrial air sanitizer. The HD-1500 is designed to filter and sanitize the air for spaces of up to 113.3 m^3 by using ultraviolet photo-catalytic oxidation [26]. This air sanitizer was designed with the help of NASA and can remove 100% of mold, airborne dust, pathogens, germs, bacteria, viruses, volatile organic compounds, and other particulate matter [26].

3.3 Habitability Functions

3.3.1 Water

For this mission to be successful, a sufficient supply of water must be provided to the crew for the duration of the mission. To determine the quantity of water needed, water use was broken down into five different categories: hygiene, drinking water, food preparation, dishwashing, and miscellaneous.

Water use for hygiene purposes includes the water the crew must use to ensure health and cleanliness. These include hand, body, hair, and clothes washing as well as teeth brushing, shaving, and toilet use. Methods that NASA uses to decrease the amount of water consumption for hygiene include using a no-rinse cleaning solution and washcloths for hand and body washing, a no-rinse shampoo, and edible toothpaste [27]. To further cut down on water use, the crew will either forgo shaving for the duration of the mission or use electric razors. Clothes washing will also not occur, as the crew will be outfitted with clothing and flight suits that are designed to be repeatedly worn throughout the 30-day mission. Toilet use can be further divided into two categories, solid and liquid waste. Solid waste will be contained in sealed bags and therefore will not require water. Liquid waste will be displaced into waste containers using suction tubes and minimal water. The majority of these hygiene practices will require little to no water and result in a total allotment of 0.5 L of water a day for each crew member's hygiene uses.

The two largest water uses are drinking water and food preparation. According to NASA standards, each astronaut must be provided 2 L of drinking water per day [14]. Much of the food used in this mission will be dehydrated and will need to have water added during preparation [28]. An estimate of 3.8 L total per day will be used for food preparation. The crew will use pre-moistened towelettes to reduce the water used for dishwashing, resulting in a supply of 0.1 L per day for each astronaut. Finally, there may be unforeseen uses for water that arise during mission operations and thus, there is a daily allotment of 0.6 L of water for miscellaneous use. Because an adequate water supply is critical to mission success and the health of the crew, a 50% safety margin is provided. Table 6 details the water supply breakdown and corresponding masses for the 30-day mission.

Table 6: Water use breakdown and volume necessary for the mission duration. Water is broken down for a single crew member in one day and for both astronauts over the course of the mission.

Water Breakdown	***Water per Astronaut per Day*** (L)	***Mission Total*** (L)	***Mass*** (kg)
Drinking	2.0	120.0	120.2
Hygiene	0.5	30	30.1
Food Preparation	1.9	114.0	114.2
Dishwashing	0.1	6.0	6.0
Miscellaneous	0.3	18.0	18.0
Total:	**4.8**	**288.0**	**288.6**
	50% Safety Margin:	**144.0**	**144.3**
	Total with 50% Margin:	**432.0**	**432.9**

3.3.2 Food and Nutrition

Food and nutrition play a vital role in retaining bone density and body mass, especially lean muscle mass, which tend to be affected during space missions [29]. Essential macro-nutrients for maintaining optimal bodily functions include carbohydrates, a readily available energy source; fats, which help cardiovascular health and guard the body against radiation effects; protein, to maintain bone density and replenish muscle mass; and Vitamin B6, for the synthesis of serotonin and catecholamines to alleviate depressive episodes [29].

Project Chariot will include a balanced meal plan, shown in Table 7, that allows each astronaut to consume at least 2,500 calories a day. This menu is a combination of the Gemini Standard Menu and the ISS Menu [30] and includes thermostabilized, re-hydratable, and natural foods; condiments; and beverages. Foods are packaged in 2 layers of plastic packaging, the outer packaging of which can be reused for other purposes such as mixing things [31]. Food packaging and processing techniques will take into account factors such as perishability [32]. Food will be transferred from the DST to the EEV when the crew prepares for their 30 days onboard the EEV. Food will be stored in readily accessible and movable locker trays in order of consumption. The entire food supply will include a 50% margin as a contingency.

Table 7: A five-day nutrition plan which cycles through six times over the 30-day mission for the crew to follow. The meal plan includes three meals per day. The index of days is mentioned in the first row for a representation of cycle repetition [30].

	Day 1, 6, 11, 16, 21, 26	***Day 2, 7, 12, 17, 22, 27***	***Day 3, 8, 13, 18, 23, 28***	***Day 4, 9, 14, 19, 24, 29***	***Day 5, 10, 15, 20, 25, 30***
Meal 1	Scrambled eggs, Bacon, Hash browns, Sausage, Toast, Margarine, Jelly, Apple juice, Coffee, Tea, Cocoa	Cold cereal, Fruit yogurt, Biscuit, Margarine, Jelly, Milk, Cranberry juice, Coffee, Tea, Cocoa	French toast, Canadian bacon, Margarine, Syrup, Orange juice, Coffee, Tea, Cocoa	Hot cereal, Cinnamon roll, Milk, Grape juice, Coffee, Tea, Cocoa	Peaches, Bacon squares, Cinnamon toast bread cubes, Grapefruit drink, Orange drink
Meal 2	Chicken, Oven-fried macaroni and cheese, Whole-kernel corn, Peaches, Almonds, Pineapple juice, Grapefruit juice	Cream of broccoli soup, Beef patty, Cheese slice, Sandwich bun, Pretzels, Fried apples, Vanilla pudding, Chocolate instant breakfast	Cheese manicotti with tomato sauce, Garlic bread, Berry medley, Shortbread cookie, Lemonade	Quiche Lorraine, Seasoned rye krisp, Fresh orange, Butter cookies	Salmon salad, Chicken and rice, Sugar cookie cubes, Cocoa, Grape punch
Meal 3	Beef fajita, Spanish rice, Tortilla chips, Picante sauce, Chili con queso, Tortilla, Lemon bar, Apple cider	Sauteed fish, Tartar sauce, Lemon juice, Pasta salad, Green beans, Bread, Margarine, Angel food cake, Strawberries, Orange drink, Pineapple drink	Sliced turkey breast, Mashed sweet potato, Asparagus tips, Cornbread, Margarine, Pumpkin pie, Cherry drink	Wonton soup, Chicken teriyaki, Chinese stir fry vegetables, Egg rolls, Hot Chinese mustard, Sweet n Sour sauce, Vanilla ice cream, Fortune cookies, Tea	Beef and potatoes, Cheese cracker cubes, Chocolate pudding, Orange drink, Grapefruit drink

3.3.3 Medical Supplies

Project Chariot requires level IV medical care. The five levels of medical care for space missions, as outlined by NASA, are described in Table 8 [14]. Rigorous protocols must be followed to protect the crew from conditions such as bone loss, psychological tolls, and other microgravity effects [33]. Table 9 describes in detail the medical care provisions that are available to the crew.

Table 8: Levels of medical care outlined by NASA according to duration of space missions and space environments [14]. Project Chariot requires level IV medical care.

Level of Care	*Space Environment*	*Mission Duration*	*Capability*
I	Low Earth Orbit (LEO)	<8 days	Space motion sickness, Basic life support, First aid, Private audio, Anaphylaxis response
II	LEO	<30 days	Level I + Clinical diagnostics, Ambulatory care, Private video, Private telemedicine
III	Beyond LEO	<30 days	Level II + Limited advanced life support, Trauma care, Limited dental care
IV	Lunar	>30 days	Level III + Medical imaging, Sustainable advanced life support, Limited surgical, Dental care
V	Mars expedition		Level IV + Autonomous advanced life support and ambulatory care, Basic surgical care

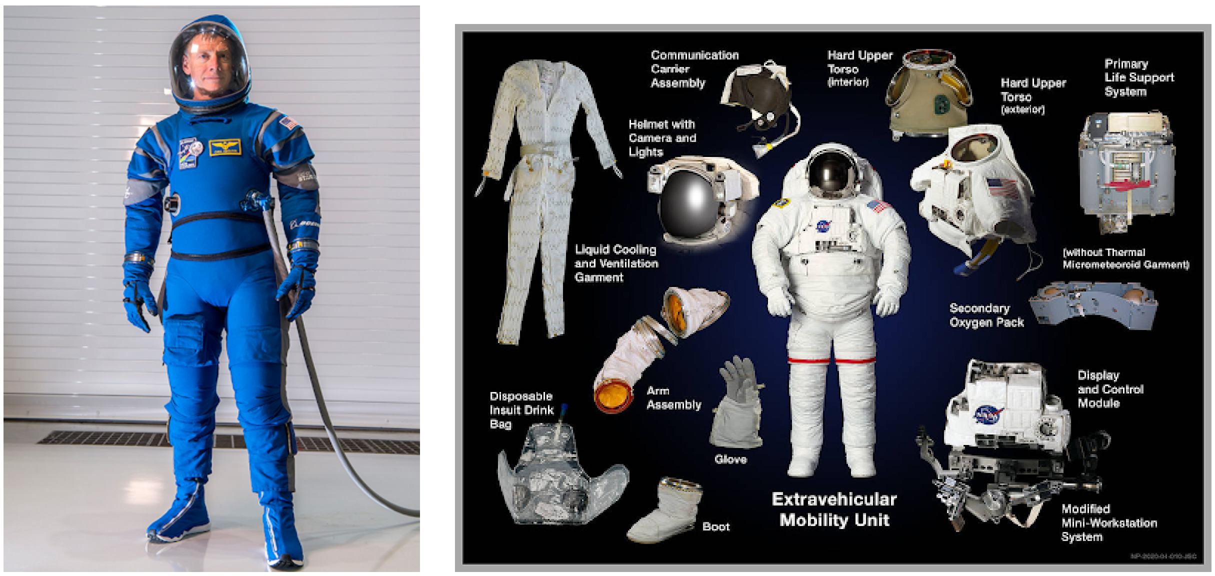
Table 9: Medical packs included in the mission adhering to level IV care of NASA standards. The items in these packs and the purpose of these packs are also outlined [33] [34].

Type of Medical Kit	*Items*	*Purpose*
Advanced Life Support Pack	Airway supply backpack, emergency surgery subpack, IV administration subpack, drug subpack, blood pressure cuffs	To save the life of a crew member, provide basic trauma life support.
Ambulatory Medical Pack	Medication such as antihistamines, tools for wound repair, bandages, bladder catheterization supplies, IV catheterization, physical exam hardware	To provide first aid in non-emergency circumstances, analyze blood samples, perform dental checks.
Crew Contamination Kit	Hand gloves, masks, antiseptic wipes, eyewash	To provide protection from exposure to contaminants.
Crew Medical Restraint System	Restraint straps and base	To stabilize a patient and their spine and to provide electrical insulation for defibrillation.
HMS Ancillary Support Pack	Saline, battery packs, ultrasound gel	To resupply all medical packs.
Respiratory Support Pack	Ambu bag	To provide low flow 100% oxygen to a patient.
Miscellaneous supplies	Biohazard trashbags, multivitamin supplements, gastrointestinal medication, nasal medication, antibiotics	Miscellaneous supplies that are not included in the aforementioned medical packs

3.3.4 Spacesuits and Clothing

The two types of spacesuits used by NASA are the intravehicular activity (IVA) suit and the extravehicular activity (EVA) suit [35]. The IVA suit is worn inside the spacecraft during periods of high risk such as launch or reentry; the EVA suit is worn while outside the spacecraft [35]. Two Boeing Starliner IVA suits, shown in Figure 10a, will be worn during the transfer from the DST to the EEV, landing and departing each moon, return to the DST, and any additional high risk maneuvers conducted during the mission. Despite not having a planned EVA during the mission, one NASA Extravehicular Mobility Unit (EMU) EVA suit, Figure 10b, will be included to allow the crew to perform emergency external repairs to the EEV. The three spacesuits for this mission are assumed to be provided on the DST with the crew and have thus been excluded from the launch mass for this mission.

The Boeing Starliner suit is designed to be comfortable and well-equipped to interface with modern systems. The suit has a smudge-resistant visor, a soft helmet, internal communication capabilities, and flexible, lightweight boots [36]. The suit is equipped with a pressurization valve and an air valve, and is made from Normex fire retardant material and specially designed Gore-Tex™ that keeps air in and allows water vapor to escape [36]. In the event of cabin depressurization, the suit will provide the crew member with fresh air and a pressurized environment. The suit visor can be lifted to stop pressurization, but cool air will continue to circulate [36].



(a) **(b)**

Figure 10: (a) Boeing Starliner suit with two life support valves visible [36]. The crew will be outfitted with this suit for high risk periods of the mission. (b) Boeing Starliner suit with two life support valves visible [36]. The crew will be outfitted with this suit for high risk periods of the mission.

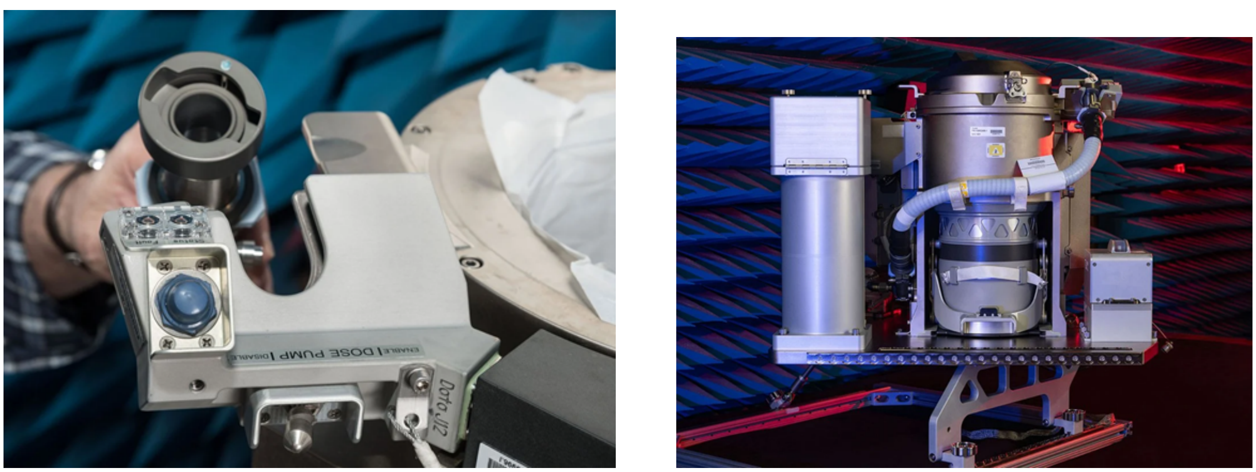
The EMU suit consists of 18 parts and has 14 layers for protection [37]. A personal life support system (PLSS) is included in the suit, and is equipped with oxygen, CO2 removal, a warning system, electrical power, water cooling, and a radio [37]. The suit is pressurized to 29.6 kPa, has a back up supply of oxygen lasting up to 30 minutes, and has a contaminant control cartridge to filter air within the suit [37]. The suit is fully equipped with a communication system to keep in contact with the EEV [37]. Since spacewalks can be very time consuming, the suit also has a small water supply and storage for a small snack bar [37]. Although it is an older suit, the EMU is still extremely capable and offers significant protection for astronauts in the vacuum of space.

A general wardrobe will be provided to the astronauts to wear during low-risk periods of the mission. Many astronauts report increased perspiration during exertion in space [38]. To account for this, the wardrobe for this mission must be comfortable and moisture-wicking. In order to limit the amount of water needed, the clothing must be able to be worn repeatedly throughout the mission without being washed. To meet these qualifications, each astronaut will be provided with a set of athletic clothing that includes four shirts, two pairs of pants, four pairs of shorts, four pairs of socks, and five pairs of underwear.

3.3.5 Human Waste Management

The Universal Waste Management System (UWMS) from NASA will be employed to handle bodily waste for this space mission. The UWMS is constructed with 3D-printed parts using superior corrosion-resistant materials such as titanium and Elgiloy [39]. The newest iteration of the space toilet has recently been adapted for female use. The updated design is 40% lighter and 65% smaller than previous versions. Figure 11a shows the custom hose and funnel included in the UWMS for urine.

The UWMS is equipped with a 0.7 m tall cylinder with a removable waste compactor which can accommodate approximately 30 deposits. Waste containment is ensured by suction that is triggered by the displacement of the lid or the urine hose from its cradle [40]. This suction neutralizes any odor and directs human waste into a receptacle consisting of a disposable bag which can be replaced after each use [41]. The entire UWMS assembly is shown in Figure 11b, with the solid waste cylinder in the center, the urine hose and funnel across the front of the cylinder, and the urine collection tanks on the left. Once the compact cylinder reaches its capacity, waste is commonly emptied into outer space. However, modifications to incorporate further storage can be made as a measure to prevent any possible contamination of space. In the case that the UWMS malfunctions, trash bags will be included for disposal and storage. The EEV will incorporate a personal hygiene stall to ensure privacy while using the UWMS.



(a) (b)

Figure 11: Universal Waste Management System. (a) Urine funnel designed for both men and women that can be used with the UWMS. Also depicted is the cradle in which the hose rests [41]. (b) The UWMS setup. The solid waste cylinder is in the center, with the urine hose across the front. The urine hose cradle is at the top right and the urine collection tank is on the left [41].

3.3.6 Waste Management

Types of waste that may be produced during the mission are: food and food packaging wastes, other consumable wastes, packaging from medical supplies, defective hardware, and payload-generated items [42]. Trash will be sorted into medical and non-medical waste, and then further separated into dry and wet categories. These four divisions of trash will be segregated for effective space management and to properly store any hazardous medical waste.

Commonly, trash is stored in bags inside a container and either returned to Earth or burned up during reentry. Project Chariot will incorporate this strategy, storing trash bags in a designated location and disposing of them once back on Earth. Dry trash can be stowed away in compressible KBO-M bags, which are heavy-duty rubberized cloth bags, each with a metal ring that closes the bag [43]. Food waste can be stowed in OpNOM bags, which are soft rubberized bags that can hold dry and wet waste [43].

3.3.7 Sleeping Accommodations

Due to the microgravity experienced on this mission, beds for the crew can be in any orientation inside the EEV, which saves on space and cost. The sleeping quarters will be part of the recreational area of the EEV module and will consist of sleeping bags with straps to secure the crew to their beds, an example of which is shown in Figure 12. The beds will be enclosed in collapsible sleeping quarters lined with Kevlar for radiation protection [44].

Figure 12: Sleeping bag in the Russian quarters on the ISS with a sleeping bag [44]. Project Chariot is expected to have a similar sleeping bag set-up.

3.3.8 Recreation

To further protect the health of the crew, the astronauts will be provided with various recreational activities, both physical and non-physical. Astronauts are recommended to get 2.5 hours of physical activity per day in space [45]. To limit the extent of damage to the human body due to microgravity, the EEV will include a stationary bike, a set of resistance bands, a full-body resistance trainer, and a workout program designed by professional trainers at NASA. The astronauts will be provided two 512 GB iPad Pros, as well as the associated chargers and headphones [46]. Each iPad will include activities such as games, podcasts, movies, books, and music.

3.4 Hardware and Equipment

3.4.1 Fire Detection and Suppression

The air supply in a spacecraft in microgravity can mimic the air currents that would fuel a flame in Earth's gravity. Therefore, fire detection devices must be placed in the air ventilation system, as the flames will follow the direction of air flow [47]. The smoke detectors chosen for this design are the smoke detectors used on the ISS, manufactured by Allied Signal and Honeywell [48] as shown in Figure 13a [49]. The smoke detector is a 2-pass infrared laser diode forward scattering detector and detects particles as small as 0.3 μm [49].

Fire suppression in space demands a 3-step response from the crew as summarized in Table 10 [47]. Four portable water-mist fire extinguishers, developed by NASA Glenn Research Center, ADA Technologies, and the Colorado School of Mines, will be included onboard as fire suppression equipment. This fire extinguisher, shown in Figure 13b, consists of a metal tank with a 3 L bladder that holds water and a 0.7 L bladder that holds nitrogen gas [50]. It can eject micro-atomized water droplets in any desired orientation. A zero-leak valve designed by the Doering Company LLC seals this fire extinguisher and prevents any pressure leaks that would lead to ineffective fire mitigation [51]. Figure 13c shows the zero-leak valve.

Table 10: NASA's standard crew response for fire suppression in case of an accidental fire involves three steps to prevent the spread of fire. The crew members aboard the EEV will follow these steps if a fire were to occur.

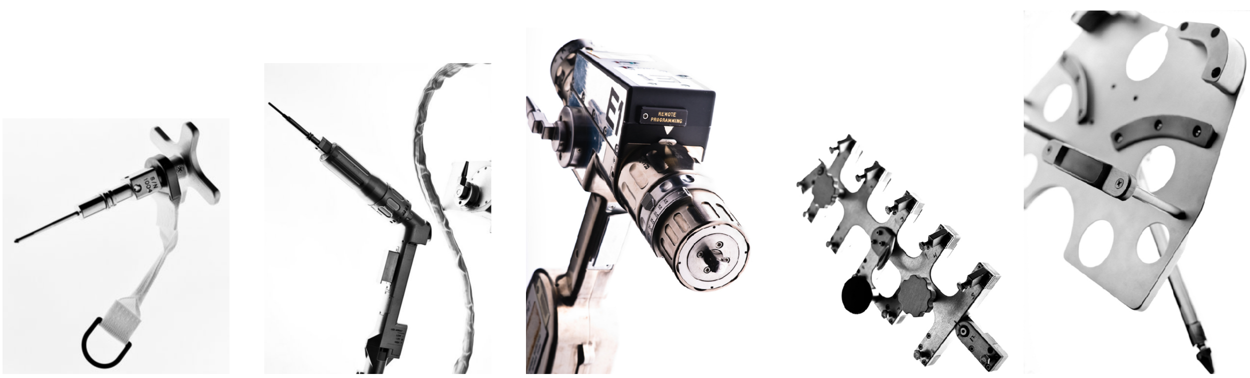
Step	*Course of Action*
1	Ventilation system must be turned off.
2	The effected power unit must be shut down.
3	Fire extinguisher is used on the fire.

(a) (b) (c)

Figure 13: Fire suppression and detection hardware. (a) ISS fire detector with minimum particulate sensitivity of 0.3 μm and optimal sensitivity to particles larger than 1 μm [49]. These will be placed throughout the EEV in the ventilation system. (b) Portable water-mist fire extinguisher designed by NASA Glenn Research Center, ADA Technologies, and the Colorado School of Mines. This extinguisher is sealed by a zero-leak valve designed by the Doering Company LLC [50]. (c) Zero-leak valve designed by the Doering company which prevents pressure loss during ejection of micro-atomized water droplets and extends the life of each fire extinguisher by a maximum of 10 years [51].

3.4.2 Space Crew Toolkit

This mission includes only IVA. However, EVA can be considered a contingency if the external portion of the EEV needs repair. If the science equipment needs repairing, a washer extraction tool, shown in Figure 14a, is included [52]. A mini screwdriver, similar to Figure 14b, will be available [52].



(a) (b) (c) (d) (e)

Figure 14: Examples of tools available to the crew. (a) Washer extraction tool, it has a long aluminum needle to remove washers from instruments [53]. (b) Mini power tool, used to tighten or loosen screws on an instrument [53]. (c) Piston grip tool, used to fasten or unfasten bolts on instruments [54]. (d) Grid cutter tool, used to cut through an instrument's electromagnetic interference grid used to protect said instrument from cosmic rays [54]. (e) Portable foot restraint that secures to the boots of the EVA suit [54].

Other tools include an EVA mini workstation, which can be secured to the EVA suit. The mini workstation can hold manual doorstays to keep bay doors locked during repairs on the external EEV module, a drive ratchet assembly [54], and other miscellaneous supplies such as spare screws, washers, a hammer, and everything in the ISS toolbox, which is shown in Figure 15 [55].

Figure 15: The ISS toolbox is shown here. It contains wrenches, a ratchet, sockets, drive accessories, screwdrivers, specialty sockets, pliers, cutters, tweezers, files, hammers, snips, a saw, and pry bars [55].

3.5 Crew Interfaces

A crew interface will be included in the mission control section of the EEV design to allow the astronauts to control various systems and interact with the EEV. A schematic of the crew interface layout can be seen in Figure 16 and includes four touchscreen monitors, a control panel, and manual maneuvering controls. Monitors 1 and 2 display mission critical information, communication with the Earth ground station, attitude determination and control metrics, and any live video feeds. Monitors 3 and 4 provide more general status updates for the EEV, such as information regarding crew health, the cabin environment, safety, and scientific data. The monitors are high fidelity touchscreen capable and designed to work with the touch screen gloves on the Boeing Starliner IVA suit. For redundancy, below the set of monitors, shown in Figure 16, is a panel equipped with physical buttons for the mission's most critical functions. This panel will be equipped with emergency controls to abort the mission and return to the DST in the event of a catastrophe. Below the physical panel, each astronaut has a manual maneuvering yoke to use if necessary.

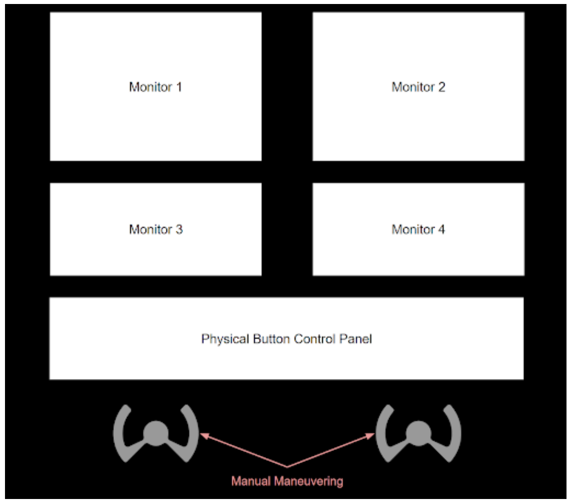


Figure 16: Depiction of the crew interface system for this mission. This will be placed in the mission operations section of the EEV.

4 Attitude, Trajectories, and Orbits

4.1 Requirements

The Attitude, Trajectories, and Orbits (ATO) subdivision was responsible for determining the following: the geometry of the 5-sol orbit, transfer orbits for all parts of the mission, the necessary attitude sensors for the EEV, and how much Δv must be supplied by the EEV. The requirements for ATO can be seen in Table 11. ATO-3.2 is a subteam-imposed requirement intended to ensure that all rendezvous and landings are safe and accurate.

Table 11: ATO requirements. These are the design drivers for the Attitude, Trajectories, and Orbits division.

Index	***Requirement***
ATO-1	The EEV must be in a 5-sol orbit around Mars before January 1, 2040.
ATO-2	The EEV must be able to visit both Martian moons and dock with the DST in 30 days or less.
ATO-3	The EEV must be able to determine its orientation during all orbits, trajectories, and rendezvous in the mission.
ATO-3.1	The EEV must be able to autonomously dock with the DST.
ATO-3.2	The EEV must have a combination of attitude sensors included in the design to determine its orientation within ± 1 degree.

4.2 Earth to Mars 5-Sol Orbit

A variety of transfer orbits were examined to determine which would be the most energy-efficient and still meet the ATO-1 requirement of being in the 5-sol orbit by January 1, 2040. A type II Hohmann trajectory [56] will be used to get to Mars. A simulation of this transfer can be seen in Figure 17. The EEV will launch from Earth on November 19, 2036, after which it will stay in the transfer orbit for 2.17 years and be inserted into the 5-sol orbit on January 28, 2039. In order to perform the launch, transfer, trajectory corrections, and 5-sol orbit insertion, the launch vehicle will use 13,440 m/s of Δv. After adding a 10% growth to this value, the launch vehicle must supply 14,784 m/s of Δv to bring the EEV from Earth to the 5-sol Martian orbit. The options and the accompanying parameters that were considered for the 5-sol orbit are summarized in Table 12. The total transfer times between the DST and the moons varied between each orbit option by less than half a day, meaning that any option would not drastically impact the total mission length; option 3 was chosen to minimize the necessary Δv. The inclinations of Phobos and Deimos, with respect to Mars' equator, are both approximately 1° [57] and as such, the 5-sol orbit will also have an inclination of 1°.

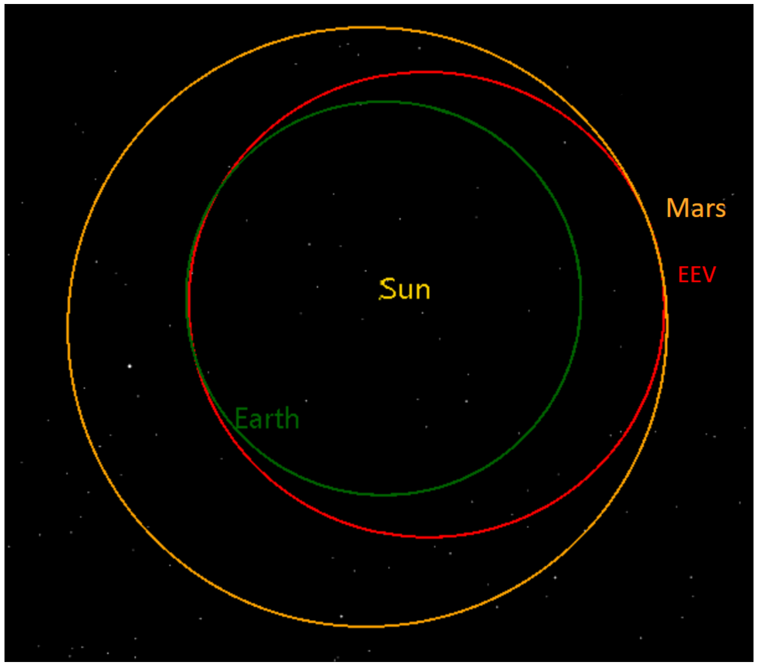


Figure 17: A simulation from GMAT of the type-II transfer from Earth to Mars. The transfer has the EEV leaving Earth in November of 2036 and arriving in January of 2039.

Table 12: 5-sol orbit geometry comparison. Option 3 is bold because it is the chosen geometry.

Option	***Eccentricity***	***Periapsis*** ***(km)***	***Apoapsis*** ***(km)***	***Total*** Δv ***(m/s)***	***Total TOF (days)***
1	0.73	16,000	103,630	2,552	0.9
2	0.83	10,000	109,630	2,961	1.1
3	**0.59**	**24,000**	**95,630**	**2,225**	**1.4**

4.3 Mars Orbit to Moons

A series of calculations were performed using a combination of an original Mathematica script and the use of the General Mission Analysis Tool (GMAT) to determine the trajectories that the EEV will take between the DST, Phobos, and Deimos and their accompanying Δv, as well as the time of flight and maximum wait times for each rendezvous.

Hohmann transfers, shown in Figure 18, will be used by the EEV after the initial rendezvous with the DST in order to minimize the Δv required for the mission. These transfers account for the large majority of the Δv budget. The transfer from the DST to Phobos will require 1,121 m/s of Δv and take 9.11 hours to complete. Traveling from Phobos to Deimos will use 756 m/s of Δv and have a time of flight (TOF) of 8.88 hours. Finally, going from Deimos to the DST will require 373 m/s of Δv and take 15.45 hours.

The Δv necessary for landing and takeoff from Phobos and Deimos was estimated using the escape velocity of each moon because of the lack of atmosphere surrounding them. Due to the microgravity on both the moons, the amount of Δv required for descent and ascent is small compared to the rest of the Δv budget. For this reason, it was inconsequential to reserve double the estimated Δv for these maneuvers as a generous safety margin; the total amount for landing and takeoff accounts for under 3% of the Δv budget, with 48 m/s reserved for Phobos and 24 m/s reserved for Deimos.

The last portion of the Δv budget comes from attitude control and trajectory corrections. Attitude control is important to make sure the EEV is oriented in the correct direction for communications, solar power, and rendezvous with the DST and moons. The additional Δv for corrections allows for small impulsive burns that push the EEV back on the path of its trajectory.

When taking into account all of the different segments of the mission, the total Δv required comes out to 2.34 km/s. With a growth allowance of 10%, this number increases to 2.58 km/s. The individual Δv values for each mission segment can be seen in Table 13.

With the quick orbital periods of both moons, the maximum wait times are, in general, also quite short. The wait time for the rendezvous with Phobos from the DST is only a few hours, the wait for rendezvous with Deimos from Phobos is a maximum of 10.27 hours, and the maximum wait for rendezvous back with the DST is just over five Earth days. With these times in mind, the mission schedule on Project Chariot's CONOPS has been set to allow plenty of time for the orbital transfers and the maximum waits.

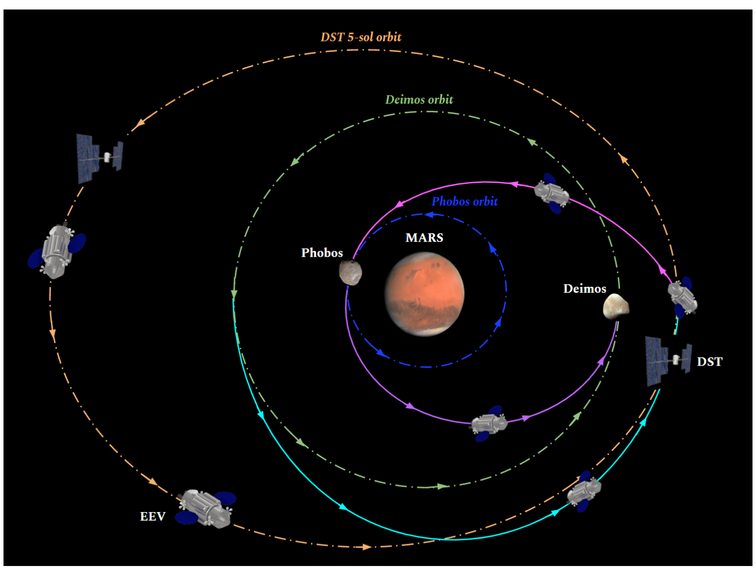


Figure 18: A diagram showing transfers from DST to Phobos (Pink), Phobos to Deimos (Purple), and Deimos back to the DST (Blue). Set orbits are shown in dashed lines.

Table 13: EEV Δv budget breakdown. This shows the required Δv for the transfers for the 30-day mission.

Mission Segment	***Required Δv (m/s)***
Rendezvous with DST	10
Slowed to circular orbit, DST to Phobos	352
Burn to transfer orbit, DST to Phobos	334
Trajectory corrections, DST to Phobos	10
Burn to rendezvous, DST to Phobos	425
Landing on Phobos	24
Takeoff from Phobos	24
Burn to transfer orbit, Phobos to Deimos	417
Trajectory corrections, Phobos to Deimos	10
Burn to rendezvous, Phobos to Deimos	329
Moon landing on Deimos	12
Takeoff from Deimos	12
Burn to transfer orbit, Deimos to DST	8
Trajectory corrections, Deimos to DST	5
Burn to circular orbit, Deimos to DST	8
Speed up to DST orbit, Deimos to DST	352
DST Rendezvous	10
Total:	**2,342**
Total with 10% Growth Allowance:	**2,576**

4.4 Attitude Determination

The Project Chariot EEV will include an attitude determination and control system (ADCS) to meet requirements ATO-3, ATO-3.1, and ATO-3.2. The ADCS is composed of a combination of star trackers, sun sensors, and inertial measurement units (IMUs), as well as attitude thrusters which will be discussed in detail by the propulsion team later on. As stated in ATO-3.2, the sensors on the EEV must have an accuracy of at most $\pm 1°$. This is achieved with high- and low-accuracy sensors. A total of eight RedWire Coarse sun sensors are incorporated in the design, mounted at intervals around the structure. These sensors have an accuracy of $1°$, a field of view of 2π steradians, and project heritage on previous Mars missions such as MAVEN and Mars Odyssey [58]. Three Ball Aerospace CT2020 star trackers will be included, which have an accuracy of $0.00042°$ and a field of view of $33°$ [59]. Finally, two Advanced Navigation Motus IMUs will be onboard. The IMU gyroscope has a maximum drift rate of 0.2 degrees per hour and a maximum range of ± 475 degrees per second; the accelerometer has a maximum range of ± 10 $g's$ [60], well exceeding the acceleration that will be experienced on a crewed mission such as this.

5 Propulsion

Throughout the mission, the EEV is dependent on its propulsion system for travel within the Martian system, including rendezvous with the DST, orbital transfers from the 5-sol parking orbit, landing on both moons, and taking off from both moons. The launch from Earth and the insertion into the 5-sol Martian parking orbit will be handled by the launch vehicle.

5.1 Requirements

Table 14 lists the requirements of the propulsion (PROP) subdivision. PROP-1 ensures that the EEV will be in the necessary 5-sol parking orbit prior to January 1, 2040. PROP-2 ensures that the EEV can travel to each moon and return to the DST. PROP-3 allows the EEV to land on each moon for sample collection. PROP-4 allows the EEV to dock safely and accurately with the DST. Humans can withstand a maximum of 6 $g's$ for 5 seconds [14]; PROP-5 preserves the life of the crew by setting a limit for the acceleration experienced.

Table 14: Five propulsion requirements required for the mission. The requirements primarily focus on the propulsion system being capable of providing necessary Δv for certain actions.

Index	***Requirement***
PROP-1	The EEV shall be inserted into a 5-sol parking orbit around Mars.
PROP-2	The propulsion system shall provide the necessary Δv for the transfers between Phobos, Deimos, and the DST.
PROP-3	The propulsion system shall provide the necessary Δv to allow the EEV to descend and ascend from each moon.
PROP-4	The EEV shall be capable of performing attitude adjustments to ensure safe rendezvous with the moons and the DST.
PROP-5	The maximum acceleration felt by the EEV shall not exceed 6 g's for a period of 5 seconds or longer.

5.2 Method of Propulsion

Chemical and electrical propulsion were considered for use with Project Chariot, as they both have been previously successful in space missions. Through trade studies, the best forms of chemical and electrical propulsion for this mission were determined to be a liquid hypergolic propulsion system and electrostatic thrusters, respectively. These two methods were compared against each other, shown in Table 15. The weights used were calculated using a pairwise comparison matrix with criteria of importance for this specific mission. Comparing both methods of propulsion using average specifications for each, it is clear that liquid hypergolic chemical propulsion is best suited for Project Chariot. As such, the EEV will use a liquid rocket engine with a hypergolic propellant.

Table 15: Table comparing chemical propulsion to electrical propulsion. The propulsion methods are compared using a list of relevant propulsion criteria.

Criteria	*Weight*	*Chemical Propulsion*	*Electrical Propulsion*
Safety	0.39	5	7
Thrust (N)	0.18	614	0.5
Isp (s)	0.10	336	3,600
Reliability	0.06	7	6
Efficiency (%)	0.03	70	80
Transit Times	0.24	9	2
	Score:	**6.5**	**5.0**

5.3 Fuel

Of the three types of spacecraft fuels, hypergolic, petroleum-based, and cryogenic, hypergolic fuel is the optimal choice for this mission. This decision is due to the unrealistic temperature and insulation requirements of cryogenic fuels and the low Isp of petroleum-based fuels. The toxic and carcinogenic nature of hypergolic fuel can be easily dealt with by isolating the fuel from the crew. One benefit of hypergolic fuel is that an ignition source is not needed, which allows for a simplified engine design, and for the engines to restart multiple times with relative ease. Hypergolic fuels are also commonly used in both main engines and attitude thrusters of spacecraft, meaning that, if designed well, the fuel lines that feed the main engines can feed the thrusters as well.

The fuels considered for this mission were hydrazine, monomethylhydrazine (MMH), unsymmetrical dimethylhydrazine (UMDH), and Aerozine 50. Each fuel comes with unique benefits, such as a lower freezing point or a slightly higher Isp. A trade study was performed to find the optimal fuel choice for the mission, shown in Table 16. The Isp and characteristic velocity values in the trade study were calculated using the NASA Chemical Equilibrium Applications (CEA) code [61] and nitrogen tetroxide (NTO) as the theoretical oxidizer. To compare the Isp, a theoretical engine was used with a chamber pressure of 68.94 bars and expansion ratio of 400. Due to its high Isp and low freezing point, MMH is the optimal fuel for the Project Chariot.

Table 16: Fuel trade study conducted between common hypergolic fuels. The criteria used in the comparison are the inherent characteristics of the fuels and engine performance parameters using an arbitrary oxidizer and engine. [61].

Criteria	*Weight*	*Hydrazine*	*MMH*	*UMDH*	*Aerozine 50*
Density (g/cm^3)	0.07	1	0.9	0.8	0.9
Freezing Point ($^\circ C$)	0.30	2	-52	-57	-7
Isp (s)	0.48	306.5	348.0	338.5	344.8
Safety	0.04	3.7	4	2.7	3.2
Characteristic Velocity (m/s)	0.11	1568	1749	1742	1750
	Score:	**2.4**	**8.3**	**6.8**	**6.6**

5.4 Oxidizer

In order to minimize the mass of the propellant, the oxidizer chosen should produce the highest Isp when combined with MMH, and should not require a low storage temperature. The analyzed oxidizers were nitrogen tetroxide (NTO) and three mixed oxides of nitrogen (MON); MON 1.3, MON 3, and MON 25. All of these options are used by NASA or the European Space Agency (ESA). While NTO is commonly used with MMH, its high freezing point [62] and corrosivity are disadvantages that are less prevalent in the MON options. The four oxidizer options were compared in a trade study shown in Table 17. The criteria and conditions in the oxidizer trade study are the same as those of the fuel trade study. The fuel used to analyze the oxidizer options was MMH.

Table 17: Oxidizer trade study conducted between common hypergolic oxidizers. The criteria used in the comparison are the inherent characteristics of the oxidizers and engine performance parameters using an arbitrary hypergolic fuel and engine. [62] [61].

Criteria	*Weight*	*N2O4*	*MON 1.3*	*MON 3*	*MON 25*
Density (g/cm^3)	0.07	1.4	1.4	1.4	1.4
Freezing Point ($^\circ C$)	0.30	-9.0	-11.6	-15	-55
Isp (s)	0.48	348.0	347.7	347.2	341.2
Safety	0.04	3	3	3.1	3.8
Characteristic Velocity (m/s)	0.11	1749	1748	1748	1737
	Score:	**7.1**	**7.3**	**7.3**	**7.5**

5.5 Main Engine

The Aestus, Aestus II, Astris, Transtar III, and XLR 132 engines were considered for the EEV propulsion system. A trade study of these was conducted, shown in Table 18, to find which engine can produce over 10

kN of thrust, have a high Isp, and have a low structural mass .

Table 18: Trade study conducted to compare different main engine candidates. The performance characteristics were found in experimental test runs with the engines. This test runs were performed either by the manufacturer and/or NASA. [63] [64].

Criteria	*Weight*	*Aestus*	*Aestus II*	*Astris*	*Transtar III*	*XLR 132*
Mass (kg)	0.28	111	138	110	47.2	54
Isp (s)	0.52	324	340	320	343	340
Thrust (kN)	0.15	29.6	55.4	27.4	16.7	16.7
Burn Time (s)	0.06	1,100	600	810	4,000	4,000
	Score:	**3.7**	**6.5**	**2.6**	**8.8**	**8.0**

With a high Isp, long burn time, and low mass, the Transtar III, shown in Figure 19, is the preferred engine for Project Chariot. The EEV will have two of these engines incorporated into the center, aft section, one of which is for redundancy. The specifications of this engine are summarized in Table 19.

Table 19: Engine specifications of the Transtar III. This table includes parameters such as the thrust, specific impulse, mass, and burn time of the engine. [64].

Parameter	*Value*
Thrust (kN)	16.7
Isp (s)	343
Chamber Pressure (bar)	98.60
Burn Time (s)	4,000
Oxidizer to Fuel ratio (O/F)	2.10
Expansion Ratio	400
Height (m)	1.2
Exit Diameter (m)	0.6
Mass (kg)	47.17

Figure 19: Colored picture of the XLR 132. The dimensions of the XLR 132 are near identical to that of the Transtar III, thus it is used as a representation of what the engine would likely be like in appearance. [63].

5.6 Attitude Thrusters

The attitude thrusters will be used for docking with the DST, performing attitude corrections, landing on and taking off from Phobos and Deimos. Three Aerojet Rocketdyne thrusters, the R-4D-11, R-4D-15, and the R-42, were compared using a trade study, shown in Table 20.

Table 20: Trade study performed between different Aerojet Rocketdyne thrusters. The criteria used are the same as those of the main engine trade study. [65]

Criteria	*Weight*	*R-4D-11*	*R-4D-15*	*R-42*
Mass (kg)	0.04	4.31	5.44	4.53
Isp (s)	0.16	315	320	305
Nominal Thrust (N)	0.08	490	445	890
Burn Time (s)	0.06	12,000	7,200	27,000
Thrust Range (N)	0.05	378-511	378-511	820-950
Minimum Impulse Bit ($N-s$)	0.35	15.6	35.6	44.48
Number of Restarts	0.20	31950	391	150
	Score:	**5.7**	**4.6**	**5.3**

After considering the options, the R-4D-11 thruster is best suited for Project Chariot due to its reliability, precision in attitude control, and adequate range of thrust outputs. The specifications of this thruster are shown in Table 21 and a diagram is in Figure 20. The EEV will have 16 of these thrusters, divided into four groups of four, as part of the design. This mounting configuration provides thrust in all directions from each corner of the EEV structure.

Table 21: Specifications of the R-4D-11 thruster. This tables shows important parameters such as thrust, specific impulse, mass, and burn time. [65].

Parameter	*Value*
Nominal Thrust (N)	490
Thrust Range (N)	378-511
Minimum Impulse Bit ($N-s$)	15.6
Number of Restarts	31950
Isp (s)	315.50
Chamber Pressure (bar)	7.45
Burn Time (s)	12000
O/F	1.65
Expansion Ratio	300
Height (m)	0.72
Exit Diameter (m)	0.38
Mass (kg)	4.31

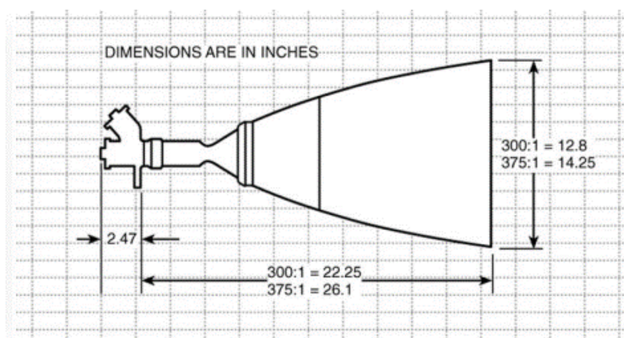


Figure 20: Figure containing a drawing of R-4D-11 thruster. The drawing shows the exit diameter, length of the nozzle, and the length of the combustion chamber. [65].

5.7 Burns

After selecting the fuel, oxidizer, main engine, and the thrusters that will be used on the EEV, the burn times and amount of propellant needed for each section was found. The results of these calculations can be seen in Table 22 below.

Table 22: Burn time and propellant mass required for each individual burn of the mission. Additionally, each burn is segmented into a specific part of the overall mission.

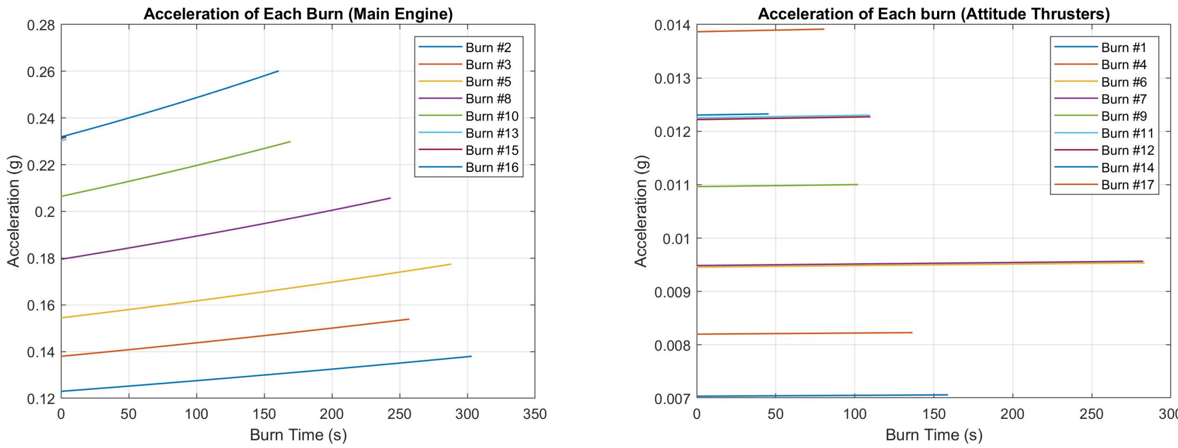
Mission Segment	***Burn***	***Burn Time (s)***	***Propellant Mass (kg)***
DST to Phobos	Rendezvous with DST	159.0	45.1
	Slowed to circular orbit	303.2	1,504.8
	Burn to transfer orbit	257.2	1,276.3
	Trajectory corrections	136.5	38.7
	Burn to rendezvous	288.1	1,430.0
	Moon landing Phobos	283.4	80.1
Phobos to Deimos	Moon take-off Phobos	282.5	80.1
	Burn to transfer orbit	243.5	120.5
	Trajectory corrections	102.1	28.9
	Burn to rendezvous	169.4	840.8
	Moon landing Deimos	109.6	31.1
Deimos to DST	Moon take-off Deimos	109.9	31.2
	Burn to transfer orbit	3.9	19.3
	Trajectory corrections	45.5	12.9
	Burn to circular orbit	3.9	19.2
	Increase velocity to DST orbit	160.8	798.0
	DST rendezvous	80.7	22.9
Entire Mission	10% Ullage	*N/A*	669.41
	Total:	**2,739.1**	**8,137.5**

The total required propellant for the mission, shown in Table 23, was used to determine the EEV wet mass. The dry launch mass of the EEV is 4,751 kg, which requires a base propellant mass of 5,851 kg. A 10% propellant mass margin was added for any additional Δv requirements as a contingency reserve, and a 10% growth in propellant mass was added to account for ullage. With those growth margins, the total wet mass of the EEV is just under 12,900 kg.

Table 23: Total wet mass. The total dry mass of the EEV is tabulated on the left with the propellant mass to the right of it and the wet mass to the far right. Additionally, 10% Δv and 10% is applied.

	Dry Mass (kg)	***Propellant Mass (kg)***	***Wet Mass (kg)***
Base:	4,751	5,851	10,602
10% Δv:	4,751	6,694.1	11,445.1
10% Δv & 10% Ullage:	**4,751**	**8,137.5**	**12,889**

Using the values from Table 22, the acceleration for each maneuver was calculated and plotted against its corresponding burn time in Figure 21a and Figure 21b. The maximum acceleration felt by the crew remains well below both the magnitude and duration limits set in PROP-5.



(a) (b)

Figure 21: Accelerations experienced by the astronauts. (a) Accelerations of the main engine plotted against burn time. The crew will feel these accelerations for the burns using the main engine. (b) Accelerations of the attitude thrusters plotted against burn time. The crew will feel these accelerations for the burns using the attitude thrusters.

5.8 Tank Specifications

The required MMH and MON 25 masses are 2,710 kg and 5,690 kg, respectively; these were derived using the total propellant mass of 8,138 kg and the main engine Oxidizer to Fuel ratio (O/F) of 2.1. MMH requires 3,000 L of volume and MON 25 requires 3,900 L of volume. Table 24 gives the propellant and pressurization tank specifications.

Table 24: Propellant and pressurization tank specifications. Sizing and dimensions of the propellant tanks can be seen.

Parameter	*Fuel Tank*	*Oxidizer Tank*	*Pressurization Tank*
Fluid	MMH	MON 25	Nitrogen
Volume (L)	1,500.00	1,950.00	0.22
Length (m)	3.0	3.0	3.0
Diameter (m)	0.85	0.98	0.30
Mass (kg)	60.0	85.0	145.0
Maximum Expected Operating Pressure (MEOP) (bars)	25.00	25.00	344.73
Number of Tanks	2	2	2

Each tank is cylindrical in shape with hemispheroid end caps. The tanks are 3 m in length, minimizing the diameter and reduces the amount of insulation and shielding required between the tanks and outer space.

The pressurization tank was selected to hold nitrogen gas at a pressure of 206 bars. The EEV will have a total of six tanks; the fuel, oxidizer, and pressurization gas are held in two separate tanks each.

5.9 System Layout

Figure 22 is a schematic of the vehicle's propulsion system, which includes the main engines, attitude thrusters, propellant and pressurization tanks, and a preliminary plumbing layout that connects everything.

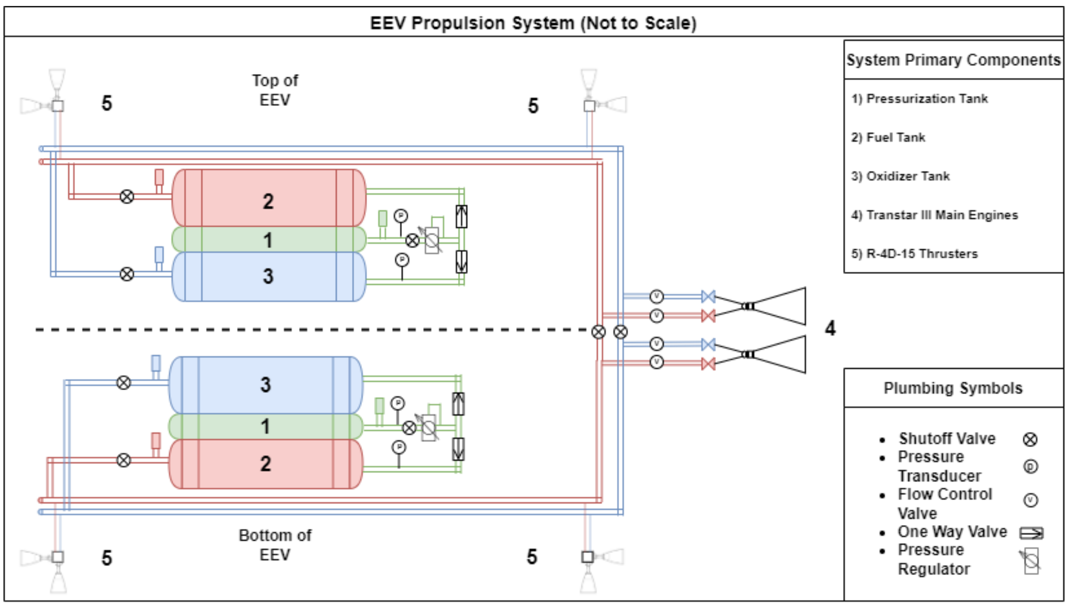


Figure 22: Propulsion system schematic detailing the orientation of the propellant tanks along with their respective pressurization tanks, valves, and piping. Additionally, the main engines and attitude thrusters can be seen. The drawing was created by Propulsion Sub-team in CAD software.

The tanks are divided into two sets, positioned at the top and bottom of the EEV midpoint. This placement ensures that the large mass of propellant does not promote instability in the EEV design. Each tank set consists of one fuel tank, one oxidizer tank, and one pressurization tank.

After the nitrogen gas leaves the pressurization tank, it passes through a shutoff valve and a pressure regulator. Before entering the propellant tanks, the gas passes through a one-way valve, which prevents propellant back-flow into the pressurization lines.

The propellant lines are laid out such that the propellant first reaches the attitude thrusters near the nose of the vehicle and then travels towards the aft of the vehicle where it reaches the aft attitude thrusters. Next, instead of the propellant lines from each set terminating directly into each respective main engine, the primary fuel and oxidizer lines connect while smaller separate lines branch off and feed the engines. This setup results in fewer propellant mass losses and allows one tank set to fuel the main engines if the other tank set fails. If there is ever a need to separate the propellant lines of each set, shutoff valves are placed at the connection point.

5.10 Launch Vehicle Analysis

The launch vehicle options that were considered for this mission were the SpaceX Falcon Heavy by itself, the SpaceX Falcon Heavy with an upper-stage system, and the SpaceX Starship, all of which would insert the EEV into the 5-sol Martian parking orbit. The first Falcon Heavy option required a C3 energy value of 8.1 km^2/s^2 and resulted in a maximum payload mass of 12,825 kg [56], greatly limiting the mission mass capabilities as shown in Figure 23. The second option requires the Falcon Heavy to bring the EEV to low Earth orbit (LEO), at which point an upper-stage propulsion system would take the EEV to the 5-sol orbit. An analysis of this option is shown in Figures 24a and 24b. As long as the wet mass of the EEV stays below the dashed line for the given upper stage Isp, then the EEV can be launched by the Falcon Heavy.

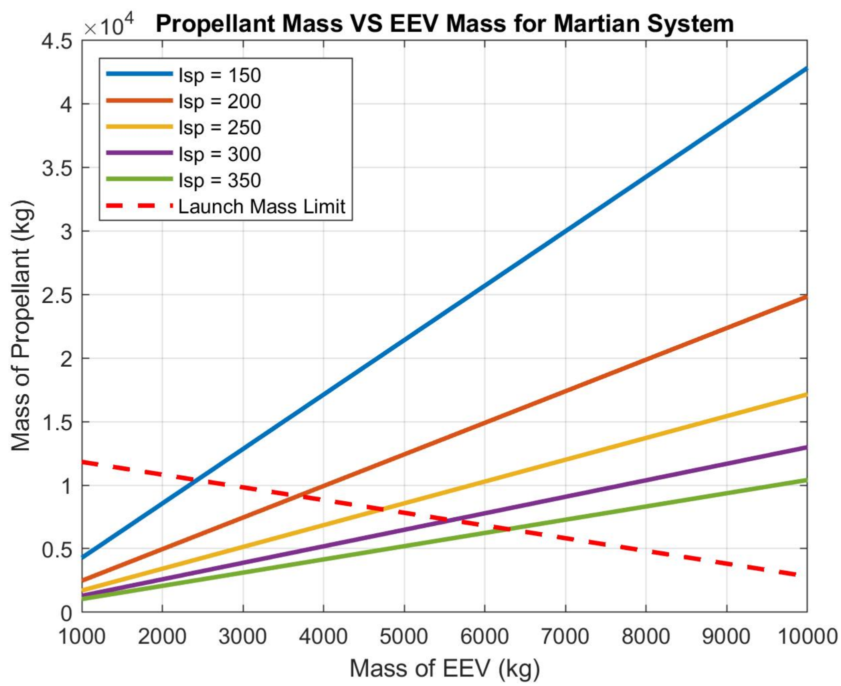


Figure 23: The amount of propellant required by the EEV once in the 5-sol orbit, based on the mass of the EEV. Plotted for multiple average Isp values. The dashed line represents the limits of the Falcon Heavy.

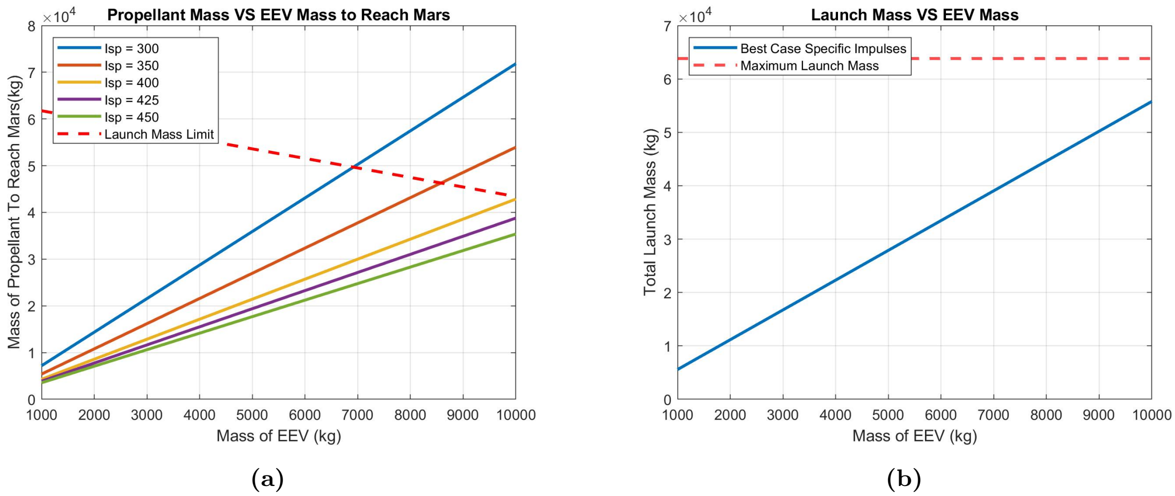


Figure 24: Evaluation of propellant masses with respect to the dry mass of the vehicle. Figure (a) is the propellant required plot, assuming best case Isp, for an upper stage propulsion system to take the EEV to the Martian system with respect to the mass of the vehicle. Maximum propellant mass is shown by the dashed line. Figure (b) is the launch mass required plot for the EEV equipped with an upper stage relative to the mass of the vehicle itself. The maximum possible launch mass of the Falcon Heavy to LEO is shown by the dashed line.

The option to use an upper stage propulsion system to get to Mars greatly increases the mass capabilities for this mission. However, the currently available upper-stage system options would not fit in the Falcon Heavy fairing with the EEV. Therefore, the third option, using the SpaceX Starship launch vehicle, was chosen due to the cost and time constraints preventing the design of a custom upper-stage system.

The Starship will launch into LEO where it is then refueled and sent directly to the Martian system. While this refueling adds risk and complexity, the process is assumed to be perfected by SpaceX prior to the launch of this mission. The main benefit of the Starship is its launch capacity of up to 100,000 kg to Mars [66], which allows Project Chariot to have an 87% safety margin for mass and virtually eliminates any risk of exceeding the mass budget.

6 Power, Thermal, and Environment

6.1 Requirements

The Power, Thermal, and Environment (PT&E) subdivision oversees the power, thermal, and environmental factors acting on the EEV. PT&E-1, the foundation of the PT&E requirements, addresses that Project Chariot must meet the power requirements of the EEV and its equipment. PT&E-2 addresses the thermal requirements for the instrumentation and the two astronauts on board the EEV. Finally, PT&E-3 addresses that Project Chariot must be able to protect the astronauts and the EEV from environmental factors such as radiation and micrometeorite impacts.

Table 25: Power, Thermal, and Environment requirements.

Index	*Requirement*
PT&E-1	PT&E shall meet the power requirements of the EEV.
PT&E-2	PT&E shall meet the thermal requirements of the astronauts and the EEV.
PT&E-3	PT&E shall protect the astronaut and EEV from environmental factors.

6.2 Power

The power requirements of the EEV will be met through the utilization of batteries and solar arrays, fulfilling PT&E-1. Project Chariot will be equipped with two UltraFlex Solar Arrays [67], which can be seen in Figure 25. Each solar array has a diameter of 3.2 m and a surface area of 8 m^2 [67]. Due to the long distance between Mars and the Sun, the solar constant around Mars is approximately 590 W/m^2 [68]. Because the surface area of the solar panel is known, the maximum power that can be generated around Mars is approximately 9.44 kW. Using a mass estimate of 150 W/kg, the solar panels have a combined mass of 63 kg [67]. This type of solar array can retract with some modifications previously used by NASA Johnson Space Center (JSC) on the Wake Shield 04 project [67]. Retracting solar arrays are required during the moon landings since regolith would otherwise be disturbed and cover the panels, reducing their efficiency. The UltraFlex solar panels will also be gimbaled in two axes to get the best sun coverage which can be seen in Figure 26, showing the orientation of the solar arrays during moon operations.

The selected solar arrays are fully capable of managing daily power needs; the power system is designed so that batteries will only be used during peak usage. Project Chariot will use the same Lithium-Ion batteries used by the Orion spacecraft, which are designed to operate in the harsh space environment. The EEV will be equipped with three batteries, each providing 2.8 kWh, for a total power output of 8.4 kWh [69].

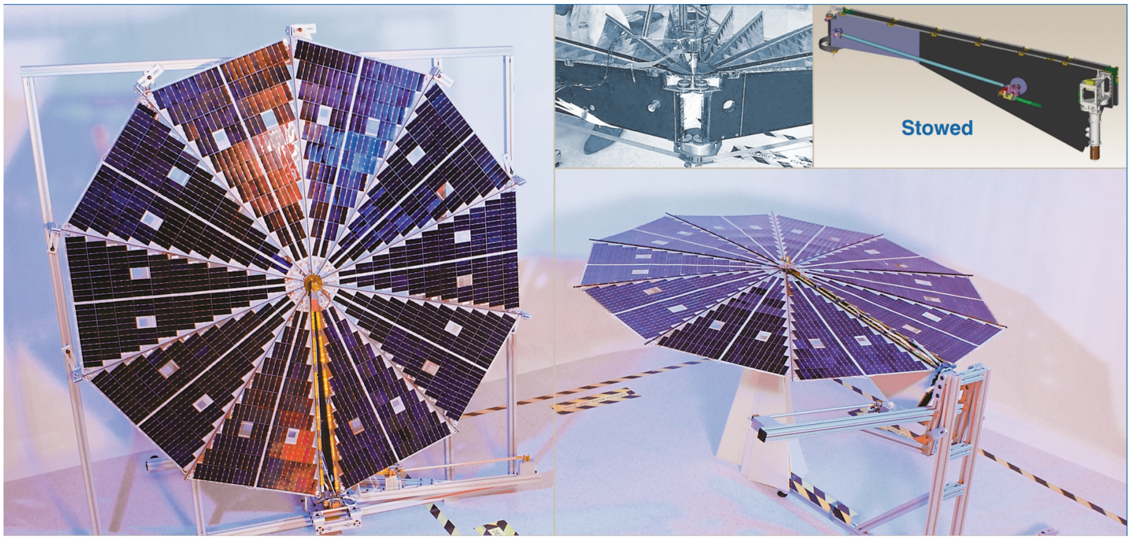


Figure 25: UltraFlex solar arrays [67]. This diagram shows the solar arrays that will be used by Project Chariot with slight modifications to allow the panels to retract.

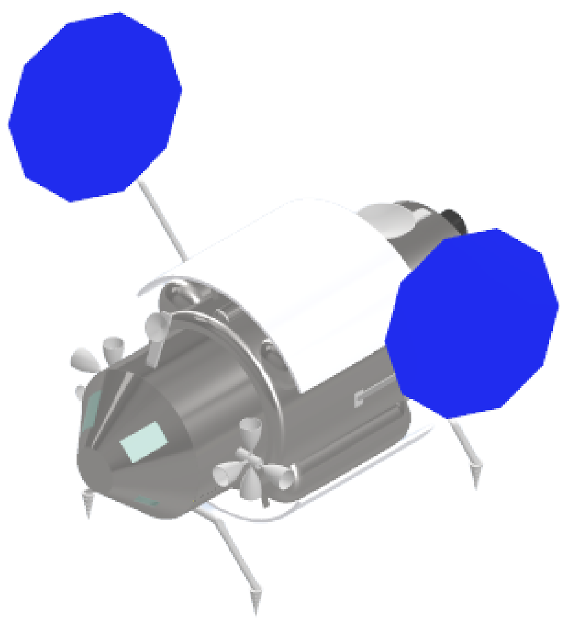


Figure 26: Solar array configuration during Moon Operations Mode. This figure shows the configuration that will be used by EEV while on the moons to maximize solar coverage.

6.2.1 Power Budget

Table 26: Project Chariot power budget. This table breaks down the power requirement by each subdivision of Project Chariot.

Item	***Quantity***	***Power Per Unit (W)***	***Total (W)***
CC&DH			
UHF	1	300	300
X-Band HGA	1	210	210
Ka-Band LGA	8	15	120
Flight Computer	2	250	500
PT&E			
Fluid Pump	2	200	400
Kapton Heater	2	45	90
HFLS			
RTD	3	1	3
CCAA	1	471	471
DPG280-100G	3	4	12
Monitors	4	60	240
Physical Controls	1	10	10
IPad	2	20	40
4BCO2	1	975	975
Airocide HD-1500	1	100	100
Smoke Detector	15	9	135
UWMS	1	270	270
PROP			
PROP Equipment	1	500	500
S&LV			
Mechanical Processes	1	500	500
Science			
GC-ITMS	1	82	82
QMS	1	125	125
LiDAR	2	35	70
GPR	1	1,000	1,000
Dust Counter	1	10	10
PlanetVac	1	20	20
Glovebox	1	1,000	1,000
ATO			
Star Tracker	3	8	24
IMU	2	1.4	2.8

Table 26 shows the power budget based on the power requirements from each subdivision of Project

Chariot. Table 27 shows the four modes the EEV will use: Mars Orbit Mode, Burn Mode, Moon Orbit Mode, and Moon Operations Mode. Mars Orbit Mode is when the EEV is orbiting or transferring to another orbit around Mars. Burn Mode occurs when the main engines are running. When looking at the entire time span of the mission, Burn Mode is under 1% of the total time. Before landing on each moon, the EEV will orbit them to gather scientific data using its LiDAR, GPR, and Dust Counter, which make up the majority of the power consumed in the Moon Orbit Mode. The final mode is the Moon Operations Mode which is when the EEV is on the Moon surface gathering samples and examining the moon's composition.

Table 27: Project Chariot power modes. This table breaks down the power needed depending on which phase of the mission the EEV is in with a 50% growth allowance to factor in unaccounted power usage.

Modes	***Power (kW)***	***Growth (%)***	***Growth (kW)***	***Total (kW)***
Mars Orbit & Transfer Mode	4.20	50	2.10	6.29
Burn Mode	4.70	50	2.35	7.04
Moon Orbit Mode	5.28	50	2.10	7.37
Moon Operations Mode	5.30	50	2.65	7.95

6.3 Thermal

Project Chariot's thermal requirements will be met through a combination of active and passive thermal control. Project Chariot's active thermal controls will consist of Kapton heaters, fluid loops, and radiators. Kapton heaters will primarily be used to heat the instruments requiring thermal regulation. The fluid loops and radiators will work together. The fluid loops carry excess heat generated by the EEV's equipment; some of this excess heat can be used to heat the interior of the EEV and the propellant tanks. The rest of the heat will be radiated away from from the EEV by radiators that will be attached outside the EEV. The radiators must be able to dissipate up to 7.95 W of heat generated by the electronics. Most spacecraft radiators are capable of dissipating between 100 and 350 W of heat per square meter [70]; as a reference, the radiators on the ISS can radiate heat at a rate of 275 W/m^2 [70]. When sizing the radiators, a rate of 325 W/m^2 was used. Since up to 7.95 W of heat needs to be radiated away, the radiators must have a surface area of at least 24.5 m^2. The radiators are designed to fit right above the propellant tanks; they have a radius of 4.2 m, cover 120°, and are 3 m long. Two radiators will be used, covering a total surface area of 26.4 m^2. A diagram of the radiator attachment can be found in Figure 27. Project Chariot's passive thermal control will consist of 25 layers of Multi-Layered Insulation (MLI) and reflective paint.

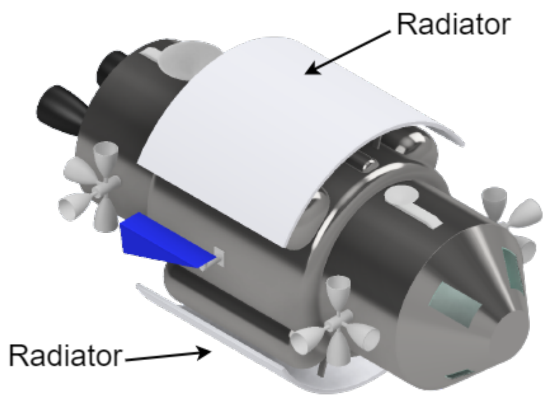


Figure 27: Radiator Configuration. This figure shows the radiator configurations, where the radiators are located about the propellant tanks.

6.4 Environment

As described by PT&E-3, PT&E must be capable of protecting the EEV and the astronauts from environmental factors such as radiation and micrometeorites. The aluminum structure used by the EEV has a total thickness of 6.8 mm which will also serve as radiation shielding. In addition, one room inside the EEV will have walls covered with packages of water for additional radiation shielding in case of a solar flare. To protect the EEV from micrometeoroid impact, Project Chariot will use NASA's wall configuration as described in Figure 28. The micrometeoroid shielding consists of six layers of Nextel AF62 and six layers of Kevlar [71]; the Kevlar layers also act as radiation shielding.

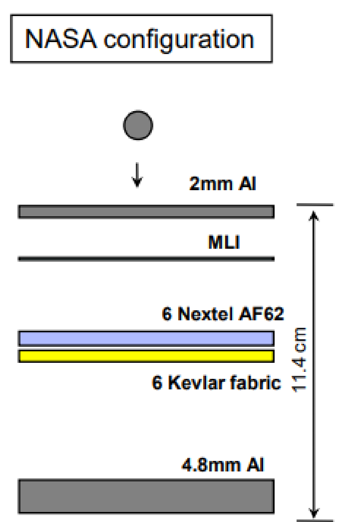


Figure 28: NASA wall configuration [71]. This figure shows the same wall configuration used by Project Chariot.

7 Structures and Launch Vehicle

7.1 Requirements

The Structures and Launch Vehicle (S&LV) requirements were derived primarily from the mission statement and launch vehicle selection. Designing an EEV structure that is compatible with the selected launch vehicle and that can support the science mission was of the utmost importance. The structure requirements for this design are detailed in Table 28. The following discussions in this section will detail how the EEV structure meets these requirements.

Table 28: S&LV requirements. These factored into the internal and external configuration of the EEV.

Index	*Requirement*
S&LV-1	The EEV shall be able to collect at minimum 50 kg of samples from each moon for analysis.
S&LV-2	The launch vehicle shall have the capabilities to lift the payload to orbit.
S&LV-3	The acceleration of the launch shall be small enough to ensure the safety and stability of the payload.
S&LV-4	The EEV shall be able to stay on the surface of both moons under extremely low gravity.

7.2 Launch Vehicle

The two main factors when determining the launch vehicle were payload capabilities and launch cost. After determining the necessary volume and estimated mass of the EEV from its general layout, launch vehicles were compared based on their payload capabilities. After narrowing down the list of vehicles, the only two launch vehicles that remained under the $1 billion budget were the SpaceX Starship and the SpaceX Falcon Heavy.

The launch vehicle that was ultimately chosen for the mission was the SpaceX Starship based on its large projected payload capability of up to 100,000 kg and its low cost relative to the SpaceX Falcon Heavy [66]. The launch cost of the Starship is quoted by Elon Musk to be approximately $10 million per launch. However, because the vehicle is not completed, the cost is listed in this report to be around $100 million with a large growth percentage to account for the uncertainty. Physical characteristics of the launch vehicle include a height of 120 m and a diameter of 9 m [66]. Included in the total height is the payload fairing, which is 17 m tall and 8 m in interior diameter; the top 9 m of the fairing converges conically [72]. These dimensions leave more than enough space to secure the EEV in the cylindrical portion of the fairing. The EEV, utilizing less than half of the available volume, extends vertically with the conical section containing the docking port. Figure 29 shows the payload fairing general dimensions.

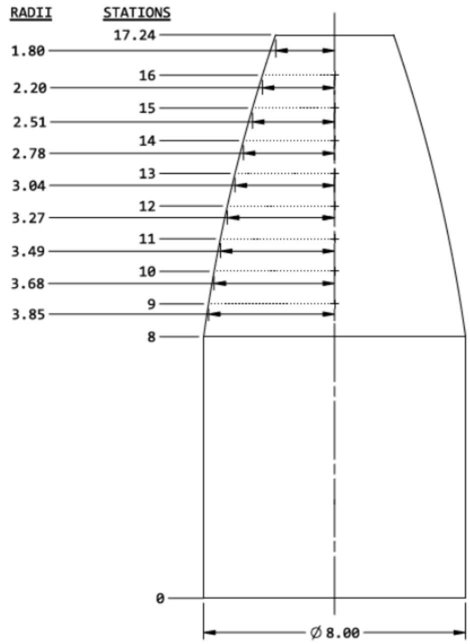


Figure 29: SpaceX Starship payload fairing dimensions [72]. Dimensions are in meters.

7.3 EEV Configuration

After the launch vehicle was selected and the dimensions of the payload fairing were determined, the EEV had restrictions when it came to the complete design. The EEV is composed of aluminum 6061-T6 and contains an inner and outer shell of 2.0 mm and 4.8 mm thickness, respectively, to provide radiation protection as well as structural soundness. The dimensions were restricted by the size and shape of the fairing but not by the mass, given the capabilities of the Starship. The internal configuration was discussed in Section 3.1.2 and the following section will describe the external EEV design.

7.3.1 EEV External Configuration

The external design of the EEV was constrained by the internal dimensions of the payload fairing. Additionally, due to the crewed aspect of the mission, the design was driven by the volume required to support the activities of two astronauts for 30 days. After these factors were considered, the final exterior dimensions of the EEV were chosen to be as shown in Figure 30.

The locations of the propulsion tanks, engines, attitude thrusters, docking hatch, solar arrays, and other vehicle necessities were important to the overall design of the vehicle. The larger center portion of the EEV contains the necessary tanks for propulsion purposes, while the smaller radius of the EEV is the maximum radius in which the crew will be living. Attached to the EEV's front and rear cylinder sections are the thrusters, located at mid-height on the EEV so that they do not become clogged from any regolith displaced when on or near the moons. The solar arrays are attached to the middle of the EEV. These arrays are retracted during the launch but then extend once ejected from the payload fairing. The position and gimbaling of the arrays allow for maximum exposure to the Sun.

The docking hatch is placed at the front of the EEV to allow for the shapes of the EEV and payload fairing to align. Four different locations of the conical section contain windows that allow for manual control of the EEV if necessary. The windows are composed of four panes of fused silica glass, an external debris pane, two internal pressure panes, and an interior scratch-resistant pane. The sample collection hatch is located along the bottom rear section. The EEV will have four lander legs on the bottom; two legs will be placed in the front and two in the back.

The custom legs on the EEV were designed to help combat the microgravity on Phobos and Deimos while collecting regolith samples. The leg design, shown in Figure 31, is meant to dig deep into the moon surfaces. The ridges that wrap around the conic shape allow for the regolith to fill in around these gaps, making it more difficult for the legs to become dislodged. This design was chosen after running tests that determined the force required for different-shaped legs to be removed from a regolith simulant.

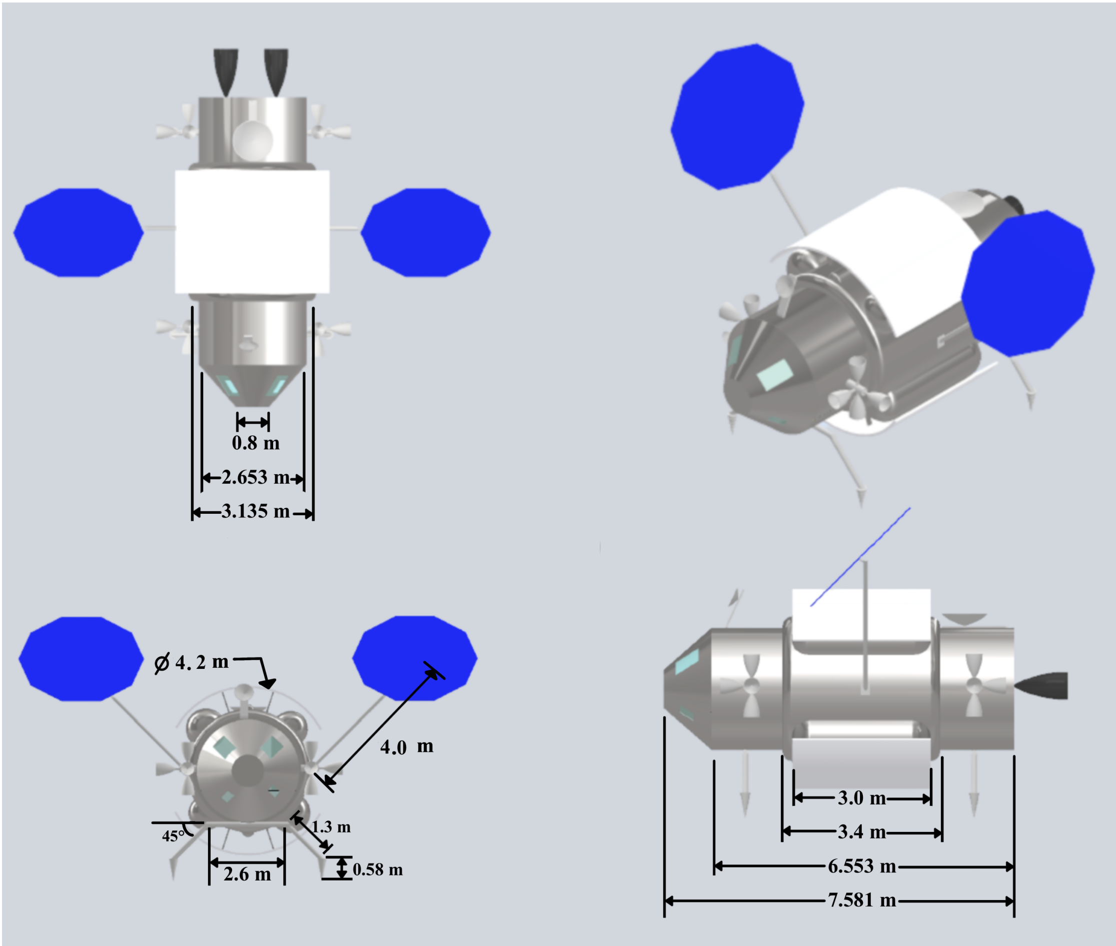


Figure 30: Dimensions of the EEV. Front, side, top, and isometric views of the EEV CAD model are included for completeness.

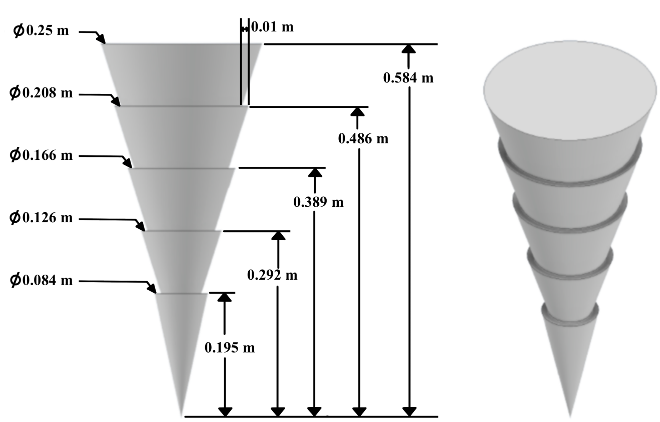


Figure 31: Dimensioned conical leg spike with ridges used for EEV attachment to Phobos and Deimos. This spike was custom-made and tested against other designs.

7.4 Structural Analysis

A stress analysis of the EEV during launch was conducted to determine if any buckling or yielding was expected. To do this analysis, a direct stiffness method Mathematica code was used [73]. The analysis focused on the stringer elements because they will carry the largest stresses. Additionally, the analysis served as an optimization problem to find the number of stringers needed, as well as their radii.

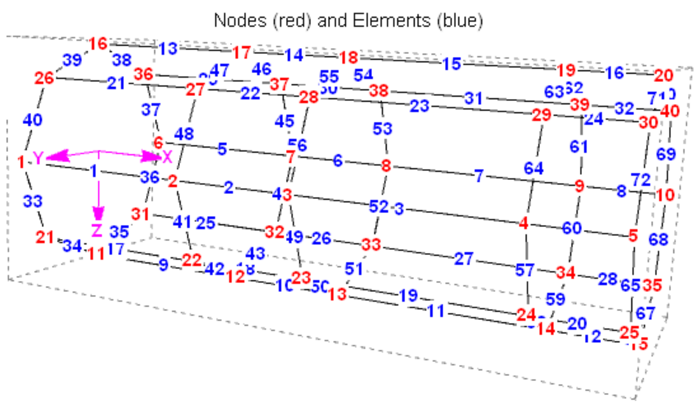


Figure 32: The assembled stringer model [73]. There is a total of 72 elements, indicated in blue, and 40 nodes indicated in red. The x-axis is through the centerline of the EEV, and the axis origin is located at the front of the EEV.

To set up the analysis, a worst-case approach was taken. Using information from Figure 33, the two cases were the maximum axial limit load and the maximum lateral limit load. For the maximum axial load case the values for n_x and n_y were 6.0 $g's$ and 0.5 $g's$, while for the maximum lateral case the values for n_x and n_y were 3.5 $g's$ and 2.0 $g's$ [72]. If the stringers can handle these two worst cases, then they will be able to handle all other loads during the mission. This approach is assuming that the largest loads the EEV will experience are during the launch. Furthermore, the stringers will be made out of aluminum 6061. The values used for the Young's Modulus, Poisson's ratio, the yield strength, and the density of this material can be found in Table 29.

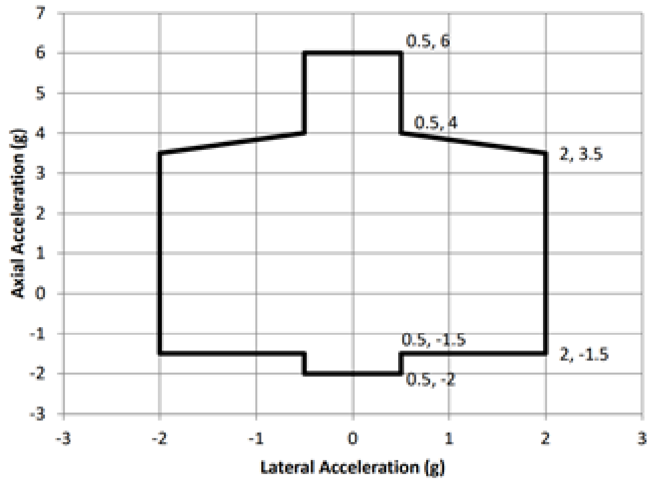
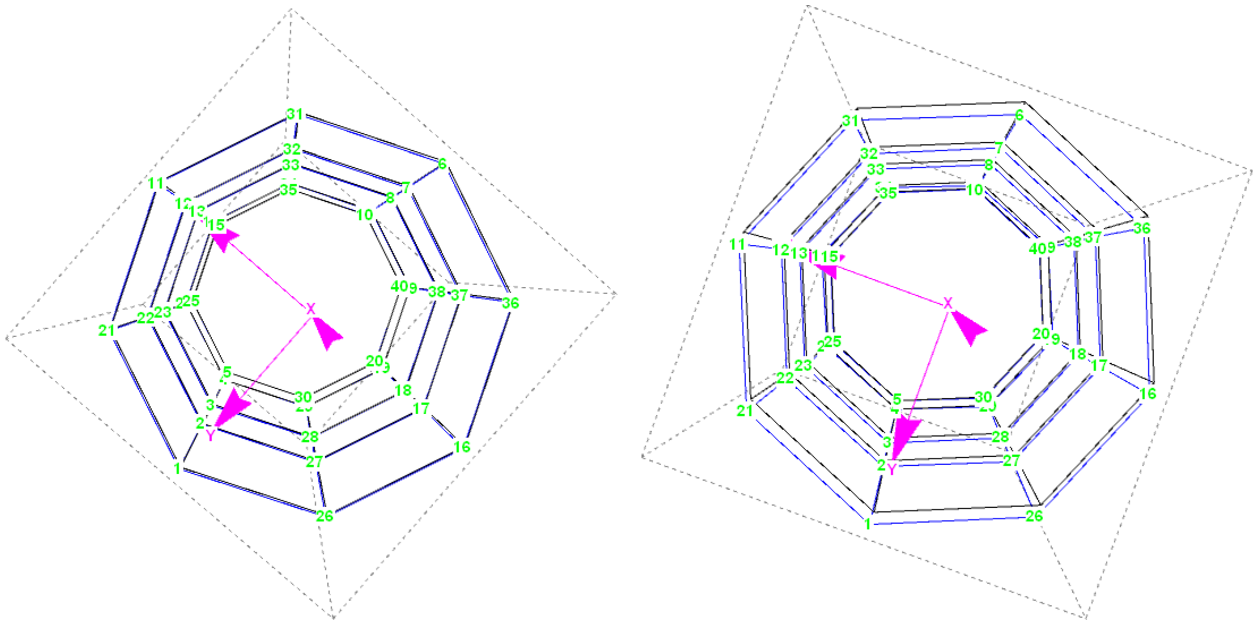


Figure 33: A plot of the load limits for the SpaceX Starship [72]. The maximum axial and lateral loading cases were taken from this plot.

Table 29: Material properties of aluminum 6061. These are used in the direct stiffness method [74].

Property	***Value***
Young's Modulus (GPa)	70
Density (m^3)	2,700
Poisson's Ratio	0.33
Yield Strength (MPa)	240

The resulting deformations of the stringers can be seen in Figure 34. Neither instance shows an unreasonable deformation, and shows that the EEV will not hit the sides of the fairing under these loads. Additionally, there were no elements that buckled or yielded in either case. The closest any stringer element came to failure were with elements 27 and 31, which both had a maximum stress of 86.89 MPa [73]. However, when compared to the yield strength of aluminum, as seen in Table 29, this number would have to be increased by roughly 276% before failure. Lastly, the analysis revealed that using eight stringers with radii of 6.4 mm provided enough strength. When compared to the original design of ten stringers with radii of 9.6 mm, this result saved roughly 26 kg [73].



(a) **(b)**

Figure 34: EEV stringer deformations. (a) The deformation resulting from the maximum axial loading case, indicated in blue [73]. No elements fail and the deformation is reasonable. The point of view is looking from the top of the EEV down the centerline to the bottom where the EEV is attached to the adapter. (b) The deformation resulting from the maximum lateral loading case, indicated in blue [73]. No elements fail and the deformation is reasonable. The point of view is looking from the top of the EEV down the centerline to the bottom where the EEV is attached to the adapter.

8 Communications, Commands, and Data Handling

8.1 CC&DH Requirements

The Communications, Commands, and Data Handling (CC&DH) subdivision is responsible for the communications architecture of Project Chariot, flight computers, and the storage and transmission of data.

Table 30: CC&DH requirements. These requirements apply to Project Chariot's communications architecture and the way data is handled during the mission.

Index	***Requirement***
CC&DH-1	The EEV shall be capable of communicating with the DST and Earth.
CC&DH-2	The EEV shall relay with the Next Mars Orbiter (NEMO) when unable to link to the DST directly.
CC&DH-3	The DST shall act as the primary relay to Mission Control.
CC&DH-4	The EEV shall transmit High Definition (HD) video during important events of the mission.
CC&DH-5	The Deep Space Network (DSN) 34 m antennas shall link with Project Chariot.
CC&DH-6	The DST shall transmit communications and astronaut health data to Earth immediately after reception from the EEV.
CC&DH-7	The DST shall store the data transmitted by the EEV.
CC&DH-8	The link margins of all links shall meet or exceed 10 dB.
CC&DH-9	The bit error rate of the ultra-high frequency (UHF) and high gain antennas shall not exceed 10^{-6}, and the bit error rate of the low gain antenna shall not exceed 10^{-2}.

The communications architecture of Project Chariot is governed by the requirements in Table 30. The EEV must be able to transmit communications and data to the DST and to Earth via the DSN. If the DST and Earth are not in view of the EEV, the NeMO can be used as a relay link. To prevent reliance on a long-range link to Earth, the EEV will primarily link with the DST, which will relay data to Earth. The communications architecture will need to handle video, audio, science data, engineering data, and astronaut health data. A series of operation modes have been devised to control when each type of data is transmitted. To ensure the robustness and quality of Project Chariot's communication links, requirements for link margin and bit error rate (BER) have been imposed. Links with a margin of 10 dB are very robust and should withstand noise perturbations during the mission. The standards for BER determined by the International Telecommunications Union state that 10^{-6} is an acceptable BER for data links and 10^{-3} is acceptable for voice applications. The UHF and X-band high gain antenna (HGA) will abide by the data link standard because of their higher throughput. The requirement is set higher than the standard for the low gain antennas to keep power requirements as low as possible when sending emergency transmissions to Earth while still achieving a robust link margin.

8.2 CC&DH CONOPS

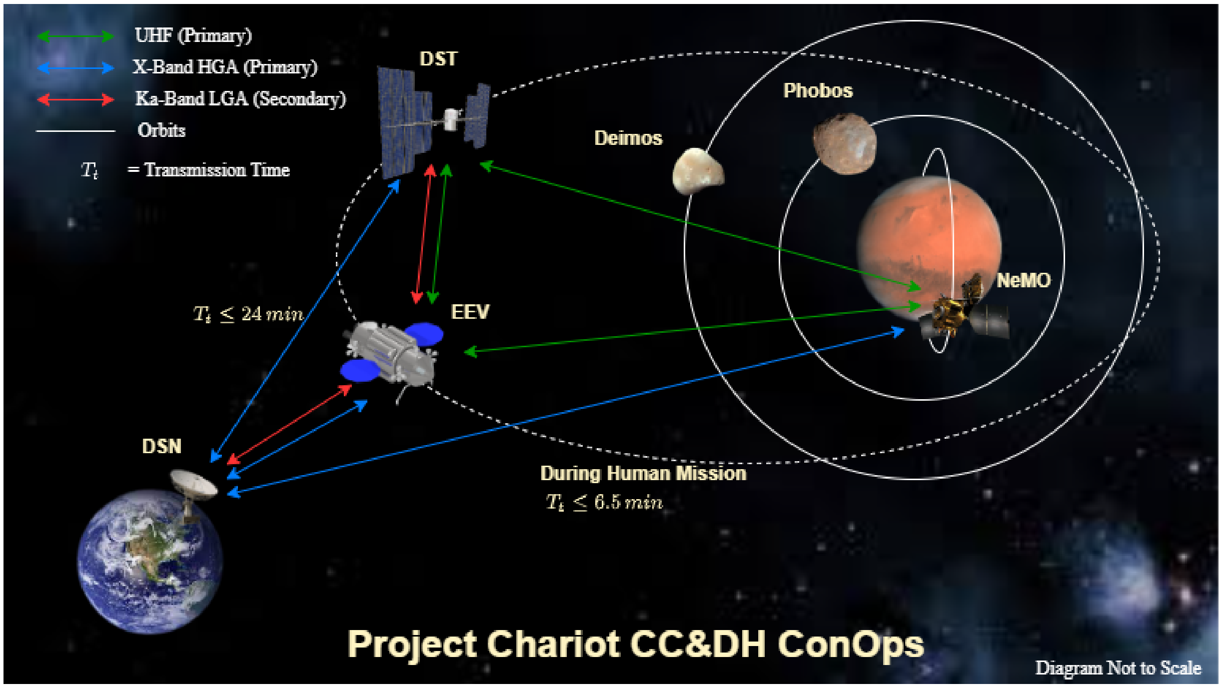


Figure 35: Communications, Commands, and Data Handling CONOPS. This figure shows all the possible links between the EEV, DST, and Earth along with transmission time.

Figure 35 is the Communications, Commands, and Data Handling concept of operations for Project Chariot and shows all the possible links between the EEV, DST, and Earth. After the EEV leaves Earth on November 19, 2036, it will use an X-Band HGA to communicate with Earth until the DST docks with the EEV on January 1, 2040. When the EEV is ready for its 30-day mission, the EEV will primarily use a UHF antenna to transmit to the DST, which then transmits back to Earth. If the DST is not in sight of the EEV, the EEV will use NeMO as a relay to the DST and Earth. The EEV is also equipped with several Ka-Band low-gain antennas (LGAs), which consist of two omnidirectional dipole antennas and six patch antennas, allowing the EEV to have omnidirectional capabilities. The Ka-Band LGAs will be used when the UHF antenna gimbal is not able to reach the DST. The Ka-Band LGAs can also be switched to a low power mode which can be used during emergencies.

8.3 Data Handling

Table 31 shows the data rate layout used by Project Chariot's UHF antenna. The UHF antenna is the primary antenna used during the manned mission. This antenna will transmit HDTV and science data, which has a combined data rate of 13 $Mbps$. HDTV data is compressed using H.264 at 24 FPS with a pixel density of 1280×720 at 8-bit color depth [75]. The EEV will have the capability to store higher quality data locally on solid-state drives (SSDs) which will be retrieved at the end of the mission. Project Chariot will also use its UHF antenna to transmit science data during "off-times" when HDTV is not being used. Because science data can require up to 10 $Mbps$, transmitting during these "off-times" will conserve power. The UHF antenna also has two HD audio channels for each astronaut that can be used throughout the mission. In addition, the UHF antenna contains two channels for engineering data and health data which will be transmitted every 30 minutes. Two channels are also dedicated to emergency use if other antennas fail.

Table 31: UHF antenna data rate layout [70] [75]. These are the data types that are transmitted by the UHF Antenna and their respective data rates.

Description	***Data Rate*** *($Mbps$)*	***Channels***	***Total Data Rate*** *($Mbps$)*
HDTV	3	1	3
Science Data	10	1	10
Description	***Data Rate*** *(bps)*	***Channels***	***Total Data Rate*** *(bps)*
HD Audio	192,000	2	384,000
Emergency	10	2	20
Engineering	10,000	2	20,000
Health Data	10	2	20

The X-Band HGA is the primary antenna for Project Chariot during the uncrewed mission segments and in the scenario that the Earth is in direct sight and the UHF antenna is unable to relay to the DST or the NeMO. Table 32 shows the X-Band HGA data rate layout, which is very similar to that of the UHF antenna but without HDTV or science data. The X-Band HGA is not designed to transmit high data rates because the UHF antenna is the primary antenna during the crewed portions of the mission.

Table 32: X-Band HGA data rate layout [70]. These are the data types that are transmitted by the HGA and their respective data rates.

Description	***Data Rate*** *(bps)*	***Channels***	***Total Data Rate*** *(bps)*
HD Audio	192,000	2	384,000
Emergency	10	2	20
Engineering	10,000	2	20,000
Health Data	10	2	20

Table 33 depicts the data rate layout for Project Chariot's Ka-Band LGA. The Ka-Band LGA consists of two omnidirectional dipole antennas and six patch antennas. Ka-Band LGA are secondary antennas that are used when the UHF antenna gimbal is not able to reach a target. The Ka-Band LGA also serve as emergency antennas when other antennas fail.

Table 33: Ka-Band LGA data rate layout [70]. These are the data types that are transmitted by the LGA and their respective data rates.

Description	***Data Rate*** *(bps)*	***Channels***	***Total Data Rate*** *(bps)*
Audio	22,000	1	22,000
Emergency	10	1	10
Engineering	10,000	1	10,000
Health Data	10	1	10

The EEV will be equipped with four flight computers, where two of the computers are primary and two are backup. Each flight computer is radiation-hardened to survive the harsh environment of space. The two primary flight computers will work independently from each another to validate and verify information received by sensors. The flight computer will also process all data, which will be stored physically on an SSD. The important information will be displayed on the four monitors.

8.4 Links and Antennas

The EEV is equipped with eight antennas: a UHF antenna, an X-band HGA, and six omnidirectional Ka-band LGAs. The UHF antenna is used to downlink to the DST and NeMO at a frequency of 2 GHz and is the main antenna that the EEV will use in Martian space during the crewed phase of the mission. The UHF will be gimbaled to aid in pointing. The X-band HGA is used to downlink to the DSN at a frequency of 8.45 GHz and is also gimbaled to aid in pointing. The HGA is the primary antenna that communicates with Earth directly. During the crewed phase of the mission, the HGA will be used if the DSN is in view while the DST and NeMO are not.

The Ka-band LGAs are used to downlink to the DSN at a frequency of 32.30 GHz. The LGAs are the secondary antennas that communicate with Earth directly. These antennas will be used in emergency situations and when pointing is difficult. Two of the LGAs are dipole antennas mounted on the HGA dish to allow them to gimbal and the other six LGAs are patch array antennas placed around the surface of the EEV to ensure complete coverage. The specifications of these antennas are listed in Table 34.

Table 34: Downlink antenna specifications. These are the properties of the antennas installed on the EEV.

Antenna	*Frequency Band (GHz)*	*Diameter (m)*	*Gain (dB)*	*Modulation*
UHF	2	0.7	25.30	BPSK
HGA	8.45	1	37.39	BPSK
Dipole LGA	32.30	0.05	16.99	BPSK/QPSK
Patch Array LGA	32.30	N/A	20.00	BPSK/8-PSK

The EEV's UHF antenna will uplink to similar antennas on the DST and NeMO. It is assumed that the DST will be capable of including a UHF antenna with the specifications shown in Table 35. The NeMO's communication system cannot be assumed to be built to these specifications, but it can be assumed that it will have a similar system to the Mars Reconnaissance Orbiter (MRO). At the time of the crewed phase of Project Chariot, the NeMO will be maintaining the satellite infrastructure in Mars orbit in lieu of the MRO. Because of this, it is assumed that NeMO will be equipped with a UHF antenna, similar to the one used by the MRO to link with rovers on the surface of Mars, and any antenna that meets or exceeds the specifications in Table 35 will close the link.

The DSN is an array of large dish antennas operated by NASA to communicate with spacecraft beyond cislunar space, such as the MRO and Voyager. The DSN is composed of sites in Goldstone, CA; Madrid, Spain; and Canberra, Australia, each housing multiple antennas and spaced equidistantly around Earth to ensure that a DSN antenna is available to point toward a spacecraft at any time [76].

Table 35: Uplink antenna specifications. These are the assumed properties of the antennas on the DST, NeMO, and DSN that will link with the EEV.

Antenna	*Frequency Band (GHz)*	*Diameter (m)*	*Gain (dB)*	*Modulation*
DST/NeMO UHF	2	1	24.87	BPSK
DST Ka-Band	32.30	0.5	43.01	BPSK
DSN	7.19	34	66.62	BPSK
DSN	34.70	34	80.29	QPSK/8-PSK

Project Chariot's communications architecture uses the modulation technique of phase shift keying (PSK) to regulate each link for link margin and BER requirements. Two varieties of PSK are used: binary phase shift keying (BPSK), which carries 1 bit per symbol, and quadrature phase shift keying (QPSK), which carries 2 bits per signal. BPSK is useful for decreasing BER because it increases as the bits per symbol of the link increases. QPSK increases the throughput of a link and is useful when it is most important to fulfill the link margin requirements. 8 phase shift keying (8-PSK) increases the throughput even further but also further increases the BER of the link. BPSK is used for the UHF antenna because, as the EEV's primary link while in the crewed mission phase, it must be robust and resistant to errors. Since this antenna is in close proximity to its receivers relative to the other antennas, it can still fulfill the link margin requirements while sacrificing throughput for BER. BPSK is used for the HGA because its data rate and diameter allow it to fulfill the link margin requirements while prioritizing low BER. Furthermore, the proximity of the EEV to Earth during the crewed phase of the mission allows for significant power savings when using BPSK over QPSK. Since fulfilling the link margin requirements is most important when operating with low gain at long distances, the dipole LGA will use QPSK and the patch array LGA will use 8-PSK when communicating with the DSN. When communicating with the DST, both LGAs will use BPSK because the path is much shorter.

8.5 Transmission Time and Operation Modes

8.5.1 Transmission Time

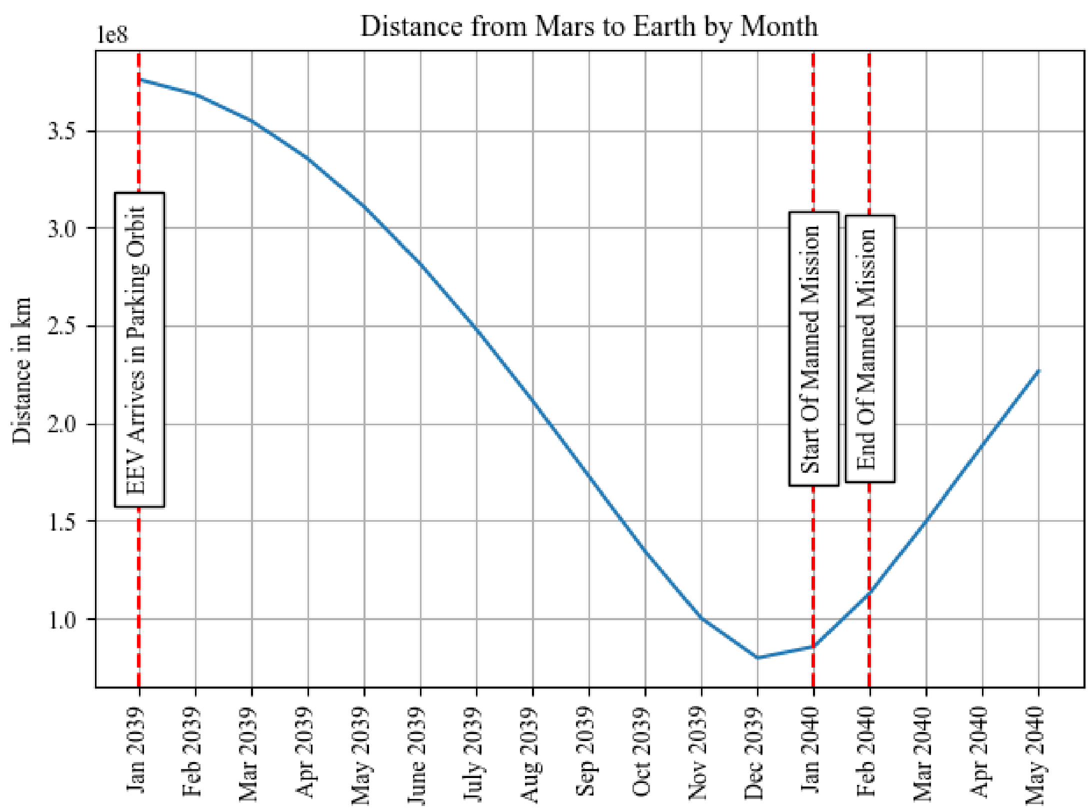


Figure 36: Relative distance from Mars to Earth. This figure shows the relative distance from Mars to Earth starting from the month the EEV arrives in its parking orbit until a few months after the mission ends.

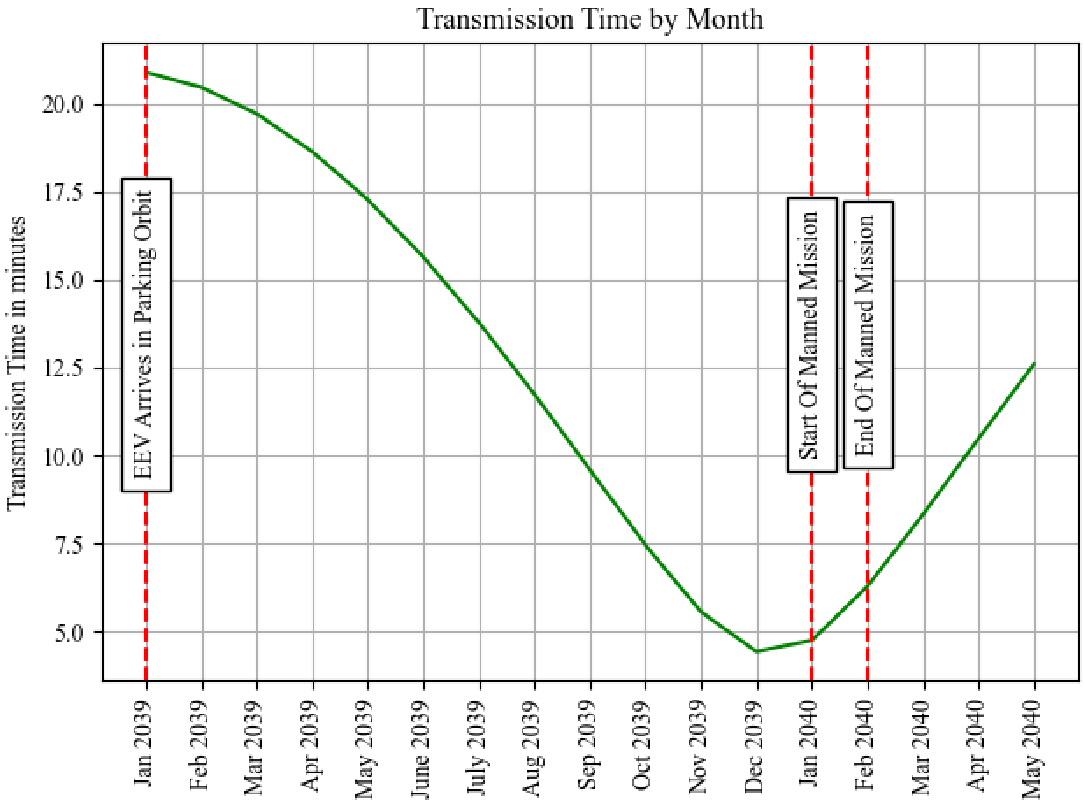


Figure 37: Transmission time from Mars to Earth. This figure shows the relative transmission time from Mars to Earth, starting from the month the EEV arrives in its parking orbit until a few months after the mission ends.

Figure 36 and 37 shows the relative distance and transmission time from Mars to Earth from the month the EEV enters a Martian orbit to the end of the human mission. Project Chariot is scheduled to be in a Martian orbit on January 28, 2039, which is also the time during this mission when Mars and Earth are the furthest apart, with a transmission time of approximately 22 minutes. Fortunately, when the astronauts arrive on January 1, 2040, Mars and Earth are relatively close with a transmission time of under 5 minutes. By the end of the crewed mission, the transmission time is approximately 6.5 minutes.

8.5.2 Operation Modes

A series of operation modes have been established to allow the EEV's antennas to transmit the data required without requiring oversized, all-purpose antennas with unreasonable power requirements. These operational modes are determined by the data types required, the distance of the link path, and the presence or absence of the crew. Power requirements, bit error rate, and link margin were calculated using the link budget analysis as shown in Table 44 in Appendix C. The UHF antenna has three operation modes, as displayed in Table 36. These modes are dependent on the types of data that must be transmitted at a given time. The UHF antenna will always transmit essential data such as HD audio, emergency data, engineering data, and health data, regardless of mode. In Science Mode, the UHF antenna transmits up to 10 $Mbps$ of science data. In Video Mode, the UHF antenna transmits HDTV video feed. In Passive Mode, the UHF antenna only transmits essential data. The UHF antenna will never transmit science data and video simultaneously.

Table 36: UHF antenna operation modes. These modes control when the UHF antenna transmits certain data types to save power.

Parameter	*Science Mode*	*Video Mode*	*Passive Mode*
Date Rate ($Mbps$)	10.40	3.40	0.40
10% Growth	1.04	0.34	0.04
Total Data Rate ($Mbps$)	11.44	4	0.44
Total Bandwidth (MHz)	11.44	4	0.44
Power Required (W)	300	100	15
Bit Error Rate	2×10^{-8}	2×10^{-8}	7×10^{-9}
Link Margin (dB)	10.57	10.65	11.70

The HGA has two operation modes, as displayed in Table 37. These modes are dependent on the presence or absence of the crew. During the uncrewed phase of Project Chariot, there is no need to transmit audio communications. This provision relieves the HGA from transmitting data at the furthest point from Earth, which would require a much larger antenna and much more power. During the crewed phase of the mission, Earth and Mars will be much closer together, allowing the transmission of HD audio to use much less power.

Table 37: HGA operation modes. These modes control when the HGA transmits certain data types to save power.

Parameter	*Crewed Mode*	*Uncrewed Mode*
Date Rate ($kbps$)	404	20
10% Growth	121	6
Total Data Rate ($kbps$)	525	26
Total Bandwidth (kHz)	525	26
Power Required (W)	450	210
Bit Error Rate	3×10^{-8}	4×10^{-8}
Link Margin (dB)	10.29	10.14

The LGAs have two operation modes, as displayed in Table 38 and Table 39, that are dependent on whether or not audio will be transmitted. Transmitting audio with the weaker LGAs will require more power than the HGA, but the ability to transmit without pointing is essential. Because the LGAs are the appointed antennas in times of emergency, they must be able to transmit with low power requirements.

Table 38: Patch array LGA operation modes. These modes control when the patch array LGA transmits certain data types to save power.

Parameter	*Audio Mode*	*Silent Mode*	*DST Mode*
Date Rate ($kbps$)	32	10	32
10% Growth	3	1	3
Total Data Rate ($kbps$)	35	11	35
Total Bandwidth (kHz)	18	6	35
Power Required (W)	250	80	15
Bit Error Rate	1×10^{-2}	1×10^{-2}	1×10^{-8}
Link Margin (dB)	10.02	10.13	11.36

Table 39: Dipole LGA operation modes. These modes control when the dipole LGA transmits certain data types to save power.

Parameter	*Audio Mode*	*Silent Mode*	*DST Mode*
Date Rate ($kbps$)	32	10	32
10% Growth	3	1	3
Total Data Rate ($kbps$)	35	11	35
Total Bandwidth (kHz)	18	6	35
Power Required (W)	190	60	5
Bit Error Rate	5×10^{-5}	5×10^{-5}	1×10^{-8}
Link Margin (dB)	10.08	10.13	11.64

8.6 Access Time Analysis

To test the robustness of the EEV-DST-NeMO relay in Martian space, an access analysis was done in STK using the north polar landing sites on each moon. The time periods in which the EEV will be on each moon are displayed below in Figure 38.

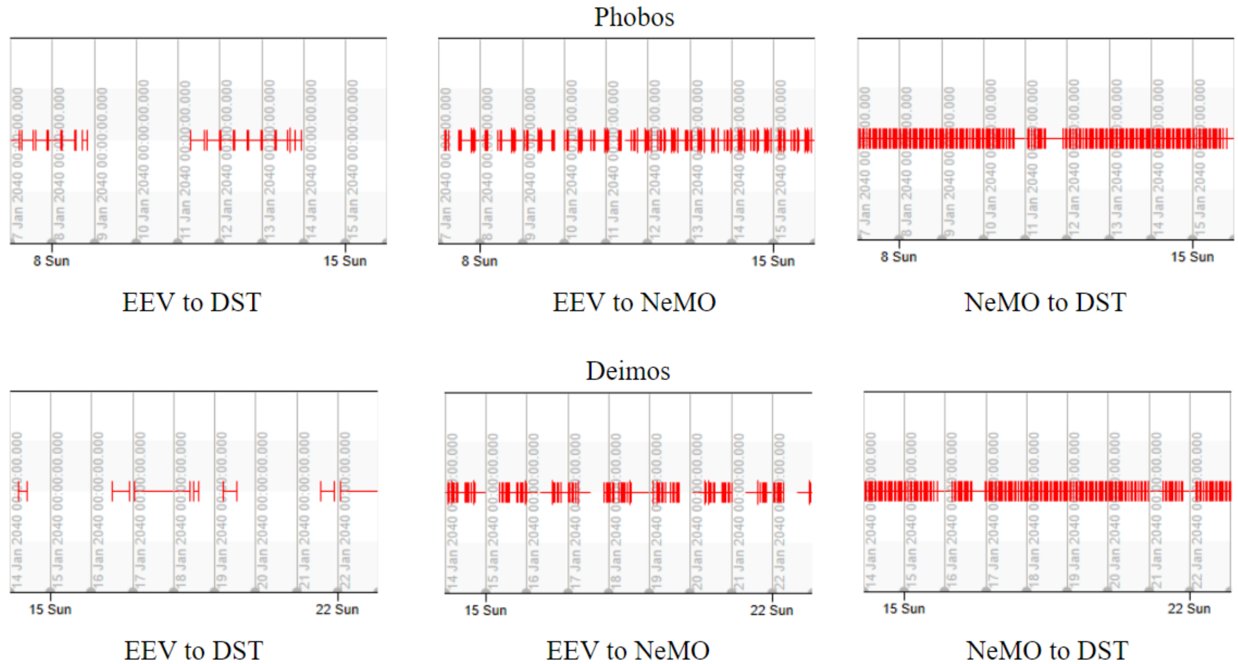


Figure 38: Communications architecture access analysis. This figure shows the time intervals in red where each asset in the communications architecture can link with each other. The time periods shown are chosen because that is when the EEV will be performing surface operations on the moons.

Analysis for the DST to DSN is not shown because the model demonstrated that the DST is in view of at least one DSN site at all times besides when the DST is behind Mars. The EEV to DST charts for each moon show why the inclusion of NeMO is necessary for this architecture, as there are long periods where the EEV is unable to see the DST from the landing site. NeMO supports much more frequent communication between the EEV and the DST.

9 Mission Summary

9.1 Budgets

9.1.1 Mass Budget

Table 40 is the mass budget table containing all the divisions of Project Chariot and their components. The table shows that the majority of the Mass is taken up by the HFLS, S&LV, and PT&E divisions. The cells in blue represent items that will be transferred over from the DST to the EEV. The final mission mass is approximately 5,739 kg with growth allowances. Subtracting the masses of the items in the blue cells will give the launch mass which is approximately 4,751 kg with growth allowance. Including the mass of the propellant, the total wet mass at launch is 12,899 kg.

Table 40: Project Chariot total mission mass budget. This table breaks down the individual mission masses from each subdivision of Project Chariot. The cells in blue represent items that will be transferred over from the DST to the EEV.

Item	***Quantity***	***Unit Mass (kg)***	***Total (kg)***	***Growth (%)***	***Growth (kg)***	***Total Mass (kg)***
ATO						
Sun Sensor	8	0.1	1.0	5	0.1	1.1
Sun Tracker	3	3.0	9.0	5	0.5	9.5
IMU	2	0.1	0.2	5	0.0	0.2
		ATO Total:	**10**	*N/A*	**1**	**11**
CC&DH						
X-Band	1	6.0	6.0	45	2.7	8.7
Ka-Band Dipole	2	1.0	2.0	45	0.9	2.9
Ka-Band Patch	6	0.1	0.6	10	0.1	0.7
UHF	1	1.5	1.5	25	0.4	1.9
Transceiver	4	2.8	11.2	25	2.8	14.0
HPA	4	3.1	12.4	25	3.1	15.5
Diplexer	4	0.6	2.4	25	0.6	3.0
Switching Network	4	1.0	4.0	25	1.0	5.0
Antenna Gimbal	2	6.0	12.0	25	3.0	15.0
Coax Cable	4	3.0	12.0	25	3.0	15.0
Flight Computer	4	20.0	80.0	25	20.0	100.0
Wire Harness	*N/A*	*N/A*	55.0	75	41.3	96.3
		CC&DH Total:	**148**	*N/A*	**39**	**283**
HFLS						
Water	*N/A*	*N/A*	214.7	50	120.9	362.6
Food	*N/A*	*N/A*	149.4	50	74.7	224.1

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Table 40 – *Continued from previous page*

Item	***Quantity***	***Unit Mass (kg)***	***Total (kg)***	***Growth (%)***	***Growth (kg)***	***Total Mass (kg)***
Air Supply	*N/A*	*N/A*	80.2	50	40.1	120.3
Air Tank	3	*N/A*	72.6	5	3.6	76.2
Air Processor	1	8.2	8.2	5	0.4	8.6
Medical Supply	*N/A*	*N/A*	14.1	1	0.1	14.2
Toolkit	1	50.0	50.0	2	1.5	51.5
UWMS	1	70.0	70.0	20	1.4	71.4
Fire Management	1	15.0	15.0	2	0.3	15.3
CO_2 Removal	1	199.6	199.6	5	10.0	209.6
RTD	3	0.5	1.5	3	0.0	1.5
CCAA	1	35.0	35.0	5	1.8	36.8
EVA Space Suit	1	118.0	118.0	10	11.8	129.8
IVA Space Suit	2	9.0	18.0	5	0.9	18.9
Monitor	4	4.5	18.0	5	0.9	18.9
Physical Control	1	20.0	20.0	15	3.0	23.2
Water Storage	33	1.2	33.0	5	1.7	34.7
Clothing	*N/A*	*N/A*	9.0	2	0.2	9.2
Astronaut	2	62.0	124.0	3	3.7	127.7
		HFLS Total:	**1322**	*N/A*	**279**	**1,601**
			PROP			
Thruster	16	5.4	87.0	5	4.4	87.0
Main Engine	2	50.0	100.0	5	5.0	105.0
Propellant Tank	4	*N/A*	273.7	10	27.4	301.1
Pressurization Tank	2	150.0	300.0	15	45.0	345.0
Fuel Lines and Valves	*N/A*	*N/A*	75.0	20	15.0	90.0
		PROP Total:	**836**	*N/A*	**97**	**928**
			PT&E			
MLI	*N/A*	*N/A*	6.0	10	0.6	6.6
Nextel AF62	*N/A*	*N/A*	474.0	10	47.4	521.4
Kevlar	*N/A*	*N/A*	165.0	10	16.5	181.5
Radiator	2	35.0	70.0	30	21.0	91.0
Kepton Heater	*N/A*	5.0	5.0	25	1.3	6.3
Solar Array	2	31.5	630	15	9.5	72.5
Battery	3	44.8	134.4	15	20.2	154.6
		PT&E Total:	**913**	*N/A*	**116**	**1,029**
			S&LV			
Inner Al Shell	*N/A*	*N/A*	385.0	5	19.3	404.3
Outer Al Shell	*N/A*	*N/A*	925.0	5	46.3	971.3
Water Walls	*N/A*	*N/A*	10.0	3	0.3	10.3
Window	4	13.8	55.0	3	1.7	56.7

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Table 40 – *Continued from previous page*

Item	***Quantity***	***Unit Mass (kg)***	***Total (kg)***	***Growth (%)***	***Growth (kg)***	***Total Mass (kg)***
Bulkhead	*N/A*	*N/A*	125.0	20	25.0	150.0
Stringers	*N/A*	*N/A*	14.7	5	0.7	15.5
Lander Leg	4	15.0	60.0	10	6.0	66.0
		S&LV Total:	**1567**	*N/A*	**99**	**1,666**
Science						
ITMS	1	12.0	12.0	10	1.2	13.2
Glovebox	1	10.0	10.0	20	2.0	12.0
PlanetVac	1	20.0	0.0	5	1.0	21.0
Sample Storage	10	5.0	50.0	5	2.5	52.5
LiDAR	2	30.0	60.0	20	12.0	72.0
QMS	1	30.0	30.0	20	6.0	36.0
GPR	1	10.0	10.0	20	2.0	12.0
Dust Counter	1	2.0	2.0	10	0.2	2.2
		Science Total:	**194**	*N/A*	**27**	**221**
		Total Mission Mass:	**4,990**		***Final Mission Mass:***	**5,739**

9.1.2 Cost Budget

Table 41 is the cost budget table for Project Chariot, which includes all the subdivisions and their component costs. Most of the budget is dedicated to the Science and S&LV divisions where research, development, and labor require the majority of the cost. Combining all the divisions for Project Chariot, the final cost is approximately $914 million. which is $76 million less than the $1 billion budget and allows for 7% growth on top of the component-specific growth allowances.

Table 41: Project Chariot total mission cost budget. This table breaks down the individual costs from each subdivision of Project Chariot.

Item	***Quantity***	***Unit Cost (USD)***	***Total (USD)***	***Growth (%)***	***Growth (USD)***	***Total Cost (USD)***
ATO						
Sun Sensor	8	$50,000	$400,000	30	$120,000	$520,000
Sun Tracker	3	$200,000	$600,000	30	$180,000	$780,000
IMU	2	$8,200	$16,400	10	$1,640	$18,040
Labor	*N/A*	*N/A*	$5,000,000	30	$1,500,000	$6,500,000
		ATO Total:	**$6,020,000**	*N/A*	**$1,800,000**	**$7,820,000**
CC&DH						
X-Band	1	$250,000	$250,000	30	$75,000	$325,000
Ka-Band Dipole	2	$150,000	$300,000	30	$90,000	$390,000

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Table 41 – *Continued from previous page*

Item	*Quantity*	*Unit Cost (USD)*	*Total (USD)*	*Growth (%)*	*Growth (USD)*	*Total Cost (USD)*
Ka-Band Patch	6	$45,000	$270,000	30	$81,000	$351,000
UHF	1	$300,000	$300,000	30	$90,000	$390,000
Transceiver	4	$1,500,000	$6,000,000	30	$1,800,000	$7,800,000
Flight Computer	4	$290,000	$1,160,000	20	$232,000	$1,392,000
R&D	*N/A*	*N/A*	$25,000,000	30	$7,500,000	$32,500,000
Labor	*N/A*	*N/A*	$1,000,000	30	$300,000	$1,300,000
		CC&DH Total:	**$34,280,000**	*N/A*	**$10,170,000**	**$44,450,000**
HFLS						
Water	*N/A*	*N/A*	$432	5	$22	$454
Food	*N/A*	*N/A*	$180,000	10	$18,000	$198,000
Nitrogen	*N/A*	*N/A*	$6,135	5	$307	$6,442
Oxygen	*N/A*	*N/A*	$1,588	5	$79	$1,667
Air Tank	3	$3,500	$10,500	5	$525	$11,025
Air Processor	1	$3,000	$3,000	3	$90	$3,090
Medical Supply	*N/A*	*N/A*	$100,000	5	$5,000	$105,000
iPad	2	$1,100	$2,200	3	$66	$2,266
Fitness Equipment	3	*N/A*	$405	3	$12	$417
Toolkit	1	$100,000	$100,000	5	$5,000	$105,000
UWMS	1	$23,000,000	$23,000,000	5	$1,150,000	$24,150,000
Fire Management	1	$5,000	$5,000	5	$250	$5,250
CO_2 Removal	1	$1,000,000	$1,000,000	5	$50,000	$1,050,000
RTD	3	$30	$90	3	$3	$93
CCAA	1	$100,000	$100,000	5	$5,000	$105,000
Pressure Gauge	3	$650	$1,950	3	$59	$2,009
EVA Space Suit	1	$15,000,000	$15,000,000	15	$2,250,000	$17,250,000
IVA Space Suit	2	$5,000,000	$10,000,000	15	$1,500,000	$11,500,000
Monitor	4	$500	$2,000	3	$60	$2,060
Physical Control	1	$10,000,000	$10,000,000	5	$500,000	$10,500,000
Water Storage	*N/A*	*N/A*	$660	3	$20	$680
Clothing	*N/A*	*N/A*	$2,000	5	$100	$2,100
Labor	*N/A*	*N/A*	$50,000,000	30	$15,000,000	$65,000,000
		HFLS Total:	**$109,120,000**	*N/A*	**$20,480,000**	**$130,000,000**
PROP						
Thruster	16	$100,000	$1,600,000	3	$48,000	$1,648,000
Main Engine	2	$10,000,000	$20,000,000	5	$1,000,000	$21,000,000
Propellant Tank	4	*N/A*	$500,000	15	$75,000	$575,000
Oxidizer	2	$4,000,000	$8,000,000	3	$240,000	$8,240,000
Fuel	2	$4,000,000	$8,000,000	3	$240,000	$8,240,000

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Table 41 – *Continued from previous page*

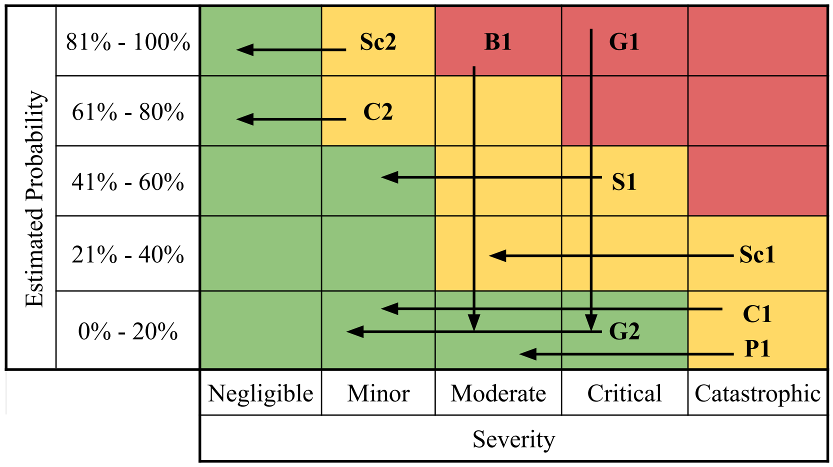
Item	***Quantity***	***Unit Cost (USD)***	***Total (USD)***	***Growth (%)***	***Growth (USD)***	***Total Cost (USD)***
Pressurization Tank	2	$500,000	$1,000,000	10	$100,000	$1,100,000
R&D & Labor	*N/A*	*N/A*	$50,000,000	30	$15,000,000	$65,000,000
		PROP Total:	**$89,100,000**	*N/A*	**$16,700,000**	**$105,800,000**
PT&E						
MLI	*N/A*	*N/A*	$500,000	35	$175,000	$675,000
Nextel AF62	*N/A*	*N/A*	$750,000	30	$225,000	$975,000
Kevlar	*N/A*	*N/A*	$750,000	30	$225,000	$975,000
Radiator	2	$3,000,000	$6,000,000	30	$1,800,000	$7,800,000
Kepton Heater	*N/A*	*N/A*	$50,000	50	$25,000	$75,000
Solar Array	2	$2,000,000	$8,000,000	35	$2,800,000	$10,800,000
Battery	3	$1,000,000	$3,000,000	15	$450,000	$3,450,000
R&D & Labor	*N/A*	*N/A*	$35,000,000	30	$10,500,000	$45,500,000
		PT&E Total:	**$54,050,000**	*N/A*	**$5,700,000**	**$70,250,000**
S&LV						
Al Shell	*N/A*	*N/A*	$2,378,880	10	$237,888	$2,616,768
Window	4	$500,000	$2,000,000	5	$100,000	$2,100,000
Structural Elements	*N/A*	*N/A*	$500	10	$50	$550
Starship	1	$100,000,000	$100,000,000	50	$0	$150,000,000
Lander Leg	4	$50,000	$200,000	10	$20,000	$220,000
R&D & Labor	*N/A*	*N/A*	$100,000,000	30	$30,000,000	$130,000,000
		S&LV Total:	**$254,580,000**	*N/A*	**$30,360,000**	**$284,940,000**
Science						
ITMS	1	$20,000,000	$20,000,000	30	$6,000,000	$26,000,000
Glovebox	1	$10,000,000	$10,000,000	30	$3,000,000	$13,000,000
PlanetVac	1	$2,000,000	$2,000,000	30	$600,000	$2,600,000
LiDAR	2	$20,000,000	$40,000,000	30	$12,000,000	$52,000,000
QMS	1	$35,000,000	$35,000,000	30	$10,500,000	$45,500,000
GPR	1	$1,000,000	$1,000,000	30	$300,000	$1,300,000
Dust Counter	1	$300,000	$300,000	30	$90,000	$390,000
R&D & Labor	*N/A*	*N/A*	$100,000,000	30	$30,000,000	$130,000,000
		Science Total:	**$208,300,000**	*N/A*	**$32,490,000**	**$270,790,000**
		Total Mission Cost:	**$755,850,000**	***Final Mission Cost:***		**$914,050,000**

9.2 Risks

Figure 39 below shows all risks related to the mission with their mitigation strategy and resulting likelihood and severity. The biggest risks the mission faces are going over budget and not being able to stay on the moon surfaces. The current mitigation strategies for these risks are removing non-critical systems and using a combination of special spikes on the landing legs and attitude thrusters to resist any upward force on the EEV, respectively. The probability of these risks occurring after the mitigation strategy drops drastically, while the severity remains the same. Additionally, there are three mission risks that would have catastrophic consequences but are relatively unlikely to happen. One of these risks is a loss of power: if this were to occur without the mitigation strategy, all electronics would shut down and the crew would essentially be stranded. To help minimize this risk, the EEV will be equipped with a backup generator, moving the severity from catastrophic to moderate.

Similar to the mission risk diagram, Figure 40 shows the risks that can affect the crew, along with mitigation strategies, likelihoods, and severities. There are three human risks that have a catastrophic effect and a significant likelihood. The risk with the highest likelihood is that a habitable climate is not maintained on the EEV, which would put the crew's life in danger. To decrease the likelihood of this risk to $<0.01\%$, backup thermal controls will be available. There is also a chance of a medical emergency, which again could result in loss of life. To help prevent this outcome, a supply of medical supplies in accordance with NASA human spaceflight guidelines will be included.

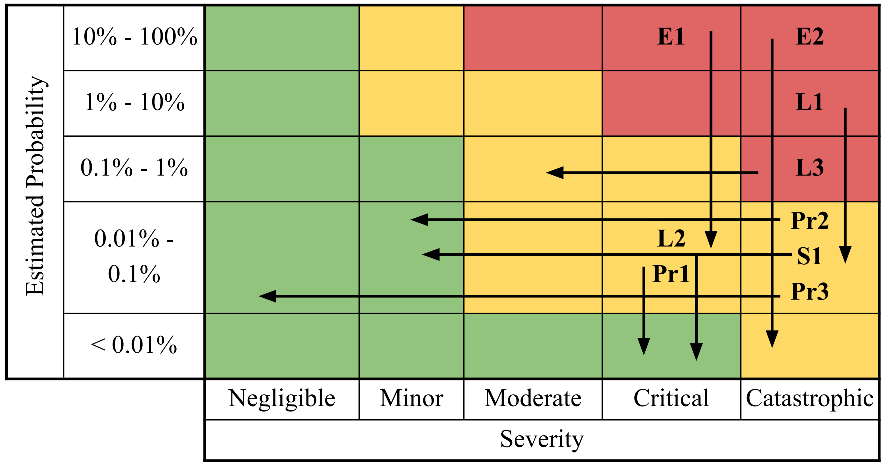
Risk Classification	Category
B	Budget
C	Communications
G	General Mission
P	Power
S	Structural
Sc	Scheduling



Index	Risk	Consequences	Mitigation Strategy
B1	If the mission goes over cost budget	Then mission may be delayed. **(Moderate)**	Remove non-critical systems to lower costs.
C1	If there is a communications failure,	Then crew cannot communicate and exchange data with Mission Control. **(Catastrophic)**	The EEV will be equipped with redundant communications systems.
C2	If there is a communications blackout caused by Mars, Phobos, or Deimos blocking the signal,	The crew cannot communicate and exchange data with Mission Control for a certain amount of time. **(Minor)**	The EEV will relay with the DST and NEMO to communicate with Mission Control.
G1	If the EEV cannot stay on moon surfaces,	Then samples cannot be collected and the EEV must return to the DST. **(Critical)**	EEV legs will be fitted with spikes to dig into the regolith. Thrusters will push the EEV down on the surface.
G2	If the launch vehicle is not fully developed by 2036,	Then the mission may be delayed or not launched. **(Critical)**	The EEV will be launched using an upper stage on a completed launch vehicle.
P1	If there is loss of power,	Then there will be electronics failure. **(Catastrophic)**	Backup generator or solar panels will be included.
S1	If there is damage to the EEV during landing or sample extraction,	Then extra-vehicular operations are demanded for repair. **(Critical)**	EVA Spacesuits will be stowed on-board, or escape module on the EEV can be used.
Sc1	If there is a manufacturing delay resulting in missing launch window,	Then mission will be canceled. **(Catastrophic)**	Launch date will be chosen with backup launch dates.
Sc2	If there is a weather delay,	Then there will be a delay in launch. **(Minor)**	Launch when the weather clears.

Figure 39: These charts show each mission-affecting risk with its corresponding mitigation strategy. Each mitigation strategy decreases decreases either a risk's likelihood or a risk's severity.

Risk Classification	Category
E	Environment
L	Life Support
Pr	Propulsion
S	Structural



Index	Risk	Consequences	Mitigation Strategy
E1	If there is excessive radiation exposure,	Then crew health and/or life endangered. **(Critical)**	The EEV is equipped with Multi Layer Radiation Insulation.
E2	If habitable climate is not maintained in the EEV due to equipment failure,	Then crew health and/or life endangered. **(Catastrophic)**	Backup thermal control will be available on-board.
L1	If there is insufficient oxygen supply for the crew,	Then crew health and/or life endangered. **(Catastrophic)**	Provide a margin of safety for the oxygen supply.
L2	If there is insufficient food or water supply for the crew,	Then crew health and/or life endangered. **(Critical)**	Add a 50% safety margin to food and water supply.
L3	If a crew member has a medical emergency,	There is a chance of loss of crew. **(Catastrophic)**	Medical supplies are included.
Pr1	If there is an inadequate amount of propellant available for the EEV,	Then there are chances of loss of crew and loss of mission. **(Critical)**	Provide a safety margin to the amount of propellant supplied for the mission.
Pr2	If there is main engine failure on the EEV,	Then the crew may be unable to reach the intended destinations. **(Catastrophic)**	The EEV will be equipped with more than one main engine.
Pr3	If there is a failure in supplying propellant to engine(s),	Then there are chances of loss of crew and loss of mission. **(Catastrophic)**	Backup pressurized gas systems or backup turbopumps can be used.
S1	If the EEV legs get stuck in the moon surface,	Then the crew will have limited supplies, endangering their health and/or life. **(Catastrophic)**	The EEV legs will be able to detach on command.

Figure 40: These charts show each human-affecting risk with its corresponding mitigation strategy. Each mitigation strategy decreases decreases either a risk's likelihood or a risk's severity.

9.3 Conclusion

As evident throughout this report, Project Chariot has met the RFP requirements, summarized in Table 39, and given thorough details explaining each subsystem. The mission stays under the $1 billion USD budget and mitigation strategies for all levels of risk have been developed. In all ways, Project Chariot is a thoroughly optimized response to the challenge set in the RFP.

Table 42: Project Chariot compliance table, containing the design project requirements and constraints, ways they have been met, and where each compliance strategy is discussed in the report.

Index	*Requirement*	*Explanation*	*Compliance*	*Section #*
RFP-1	The EEV has the ability to support two astronauts for the mission.	The EEV contains all the necessary life support provisions and is large enough for a crewed, 30-day mission.	Yes	3
RFP-2	The mission duration is 30 days or less, including travel to and from the DST.	Including extra time reserved for moon operations and extra time to account for any possible transfer wait times, the mission will last less than 30 days from leaving the DST to the final DST rendezvous.	Yes	1.5
RFP-3	The EEV can collect and store at least 50 kg of samples each from both moons.	The EEV is designed to include multiple sample collection tools and a designated storage area, with space for over 50 kg from each moon.	Yes	2.4
RFP-4	The mission will not include any planned EVA.	The EEV is capable of performing all mission operations without requiring the crew to leave the vehicle or DST.	Yes	N/A
RFP-5	The EEV will allow the astronauts to conduct scientific exploration of both moons.	The EEV contains an interface that allows the crew to perform scientific experiments on the collected samples.	Yes	2
RFP-6	The science objectives performed on the mission will advance both deep space travel capabilities and understanding of the moons.	The samples collected by the EEV will give insight into the origins of the moons.	Yes	2
RFP-7	The EEV will include the required science equipment to meet the science objectives.	A series of scientific instruments are included in the EEV design that are fully capable of meeting all of the planned science objectives.	Yes	2
RFP-8	The astronauts may bring up to 200 kg of science equipment from the DST to the EEV.	For this design, all of the science equipment utilized on the EEV will be built in prior to launch.	N/A	N/A

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Table 42 – *Continued from previous page*

Index	***Requirement***	***Explanation***	***Compliance***	***Section #***
RFP-9	The sample retrieval and storage methods must be described; the samples must be quarantined and brought to the DST to be returned to Earth.	The EEV is equipped with sample collection and storage tools. A method to transfer the samples to the DST is included.	Yes	2
RFP-10	The EEV autonomously docks with the DST.	The EEV has a docking hatch compatible with the DST.	Yes	7.3
RFP-11	The maneuvers to reach the moons must be discussed.	The orbital maneuvers between the DST and the moons are shown and described.	Yes	4.3
RFP-12	The surface operations and time required to complete them must be described.	The sample collection system is described in detail.	Yes	2.4
RFP-13	Determine how the EEV will travel to Mars from Earth, including the launch, propulsion system, and interplanetary trajectory used.	All maneuvers done by the EEV and or the launch vehicle are described in detail.	Yes	4.2
RFP-14	The EEV must be in the 5-sol orbit around Mars by January 1, 2040 when the DST arrives with the crew.	The EEV will arrive in the 5-sol orbit nearly a year before the DST is scheduled to arrive.	Yes	1.5
RFP-15	Discuss the launch vehicle selection process.	The launch vehicle options considered and a description of the deciding factors are included.	Yes	5.10
RFP-16	Include trade studies for systems and subsystems included in the mission design, ideally using current technologies with mission heritage.	Trade studies are included when necessary, and most components on the EEV have been used in previous missions.	Yes	N/A
RFP-17	The total vehicle and launch cost must be under $1 billion USD.	The current cost for the EEV and launch vehicle is just under $915,000,000 USD with growth allowances.	Yes	9.1.2

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Appendix A: System Block Diagram

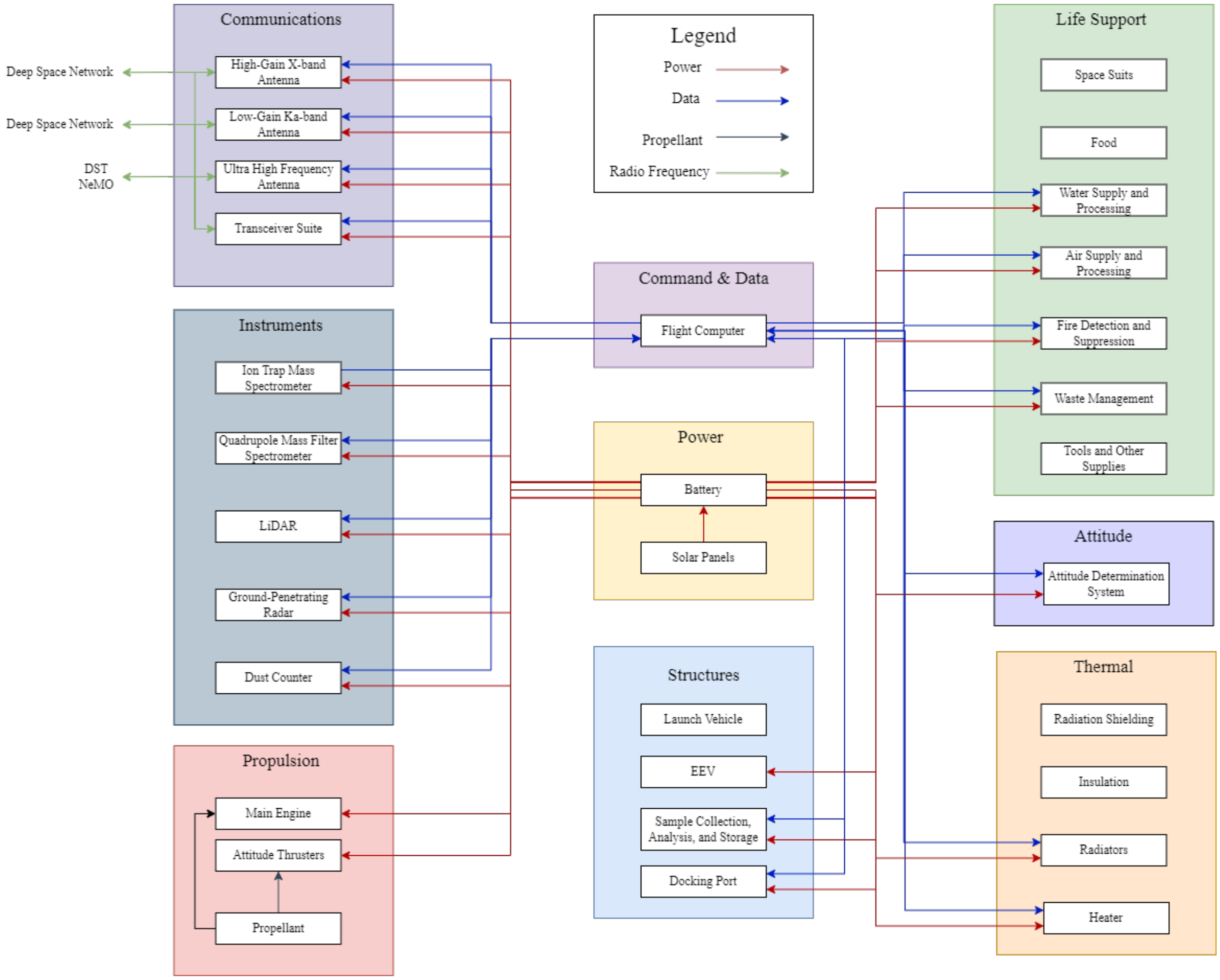

Figure 41: System block diagram. This figure lays out the systems the EEV currently has in place and how each system interacts with others via the transfer of power, data, propellant, and radio frequencies.

Appendix B: Science Traceability Matrix

Table 43: Science traceability matrix. This table traces each overarching science goal through the objectives for measurement down to the instruments selected to accomplish the objectives.

Science Goals	***Measurement Objectives***	***Instrumentation***
Determine regolith composition	Measure chemical and mineral composition by mass	Ion trap mass spectrometer
Return regolith samples	Return samples to Earth to be distributed to researchers for more intensive study	PlanetVac and Autogropher II
Test hypotheses of the origins of Phobos and Deimos	Measure chemical and mineral composition by mass; look for glassy, recrystallized igneous materials for giant impact theory; look for chondritic materials for captured asteroid theory	Ion trap mass spectrometer
Study Mars atmosphere and space around moons to build off of MAVEN	Measure abundance of carbon compounds, volatile compounds, and trace gases in Mars atmosphere and space around moons while in orbit and in transit	Quadrupole mass filter spectrometer
Study moon topography	Map the topography of Phobos and Deimos from orbit	LiDAR
Determine viability of ISRU with Phobos and Deimos material	Measure chemical and mineral composition of samples, look for and chemicals that could produce rocket fuel and determine if they can be extracted	Ion trap mass spectrometer
Determine composition of subsurface regolith	Measure chemical and mineral composition deeper below where the sample collection system can reach	Ground-penetrating radar
Determine the presence and abundance of dust around Phobos and Deimos	Map dust distribution and density around Phobos and Deimos	Dust counter

Appendix C: UHF Downlink Link Budget

Table 44: UHF Downlink Link Budget. This table includes the values and calculations used to design the UHF antenna and its corresponding link to the DST. All link budgets used in Project Chariot follow this format.

Parameter	***Value***	***Unit***	***Notes***
Downlink Frequency	3	GHz	Input
Downlink Wavelength	0.0999	m	Calculation
Modulation	BPSK	N/A	Input
Target Bit Rate	11.44	Mbps	Input
Bandwidth	11.44	MHz	Calculation
Transmit Power	300	W	Input
Transmit Power	54.77	dBm	Calculation
Transmit Losses	1	dB	Assume
Transmit Antenna Diameter	0.7	m	Input
Transmit Antenna Efficiency	70	%	Assume
Transmit Antenna Gain	25.30	dB	Calculation
Transmit Pointing Losses	0.1	dB	Assume
Transmit EIRP	78.97	dBm	Calculation
Link Distance	95630	km	Input
Downlink Path Loss	201.60	dB	Calculation
Polarization Loss	0.1	dB	Assume
Atmospheric Losses	0.2	dB	Assume
Receive Antenna Diameter	1	m	Input
Receive Antenna Efficiency	70	%	Assume
Receive Antenna Gain	28.39	dB	Calculation
Receive Pointing Losses	0.1	dB	Assume
Receive Waveguide Loss	0.1	dB	Assume
Received Power	-124.74	dBW	Calculation
Mars Noise Temp	29	K	Assume
Antenna Noise Temp	27	K	Assume
Background Noise Temp	3	K	Constant
System Noise Temp	59	K	Calculation
Noise Bandwidth	11.44	MHz	Calculation
Noise Power	-140.31	dBW	Calculation
CNR	15.57	dB	Calculation
Eb/No	15.57	dB	Calculation
BER	2×10^{-8}	N/A	Approximation
Minimum Receiver CNR	5	dB	Assume
Link Margin	10.57	dB	Calculation