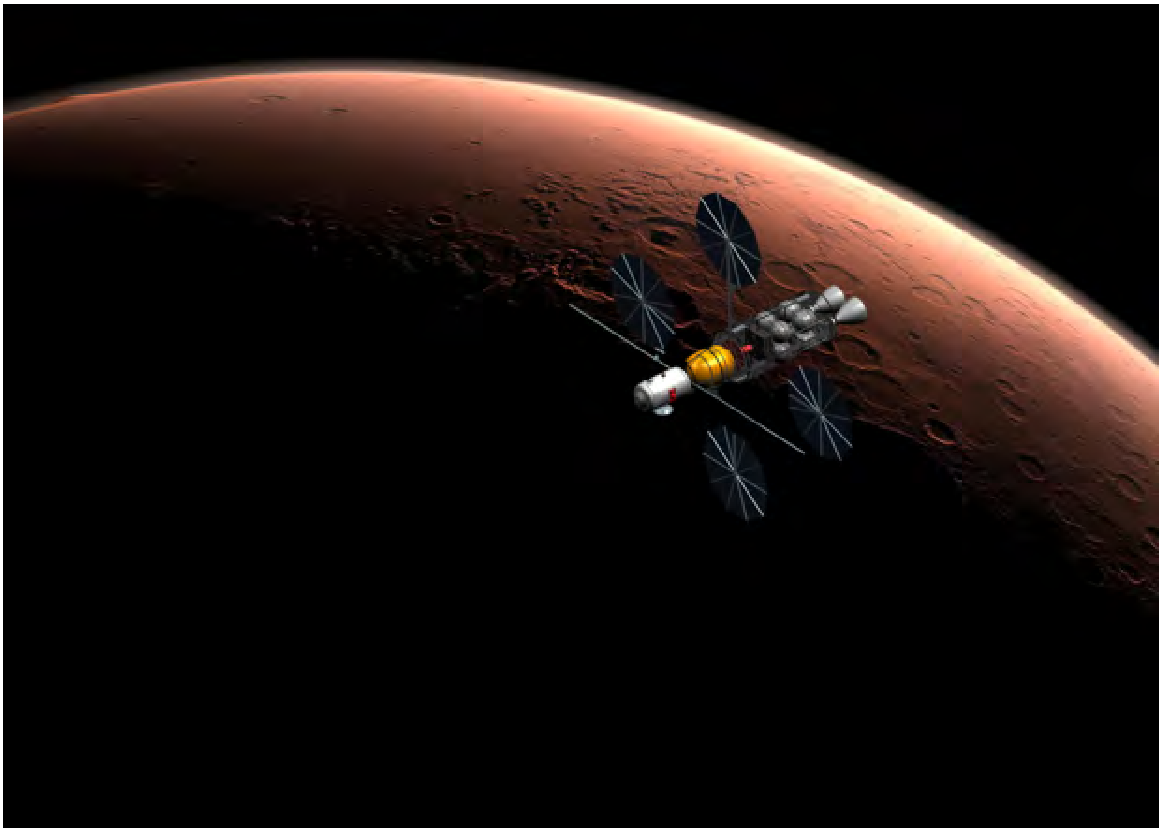
University of Kansas



AIAA 2017 Undergraduate Spacecraft Design Competition Submission

Mars Orbiter Operating Near Satellites

Team Lead: Bailey Miller

Supervised by: Dr. Mark Ewing

University of Kansas Team:

Arno Prinsloo Brian Frew Brooke Reid

Colin Murphy Conner Murphy Philip Guzman

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Acknowledgments

A dreamer sits, her heels dig into soil.
The Kansas wind knocks wheat that ebbs and flows.
The sky this night is infinite and royal -
She wonders where the spiral arms might go.

A grit of teeth, a flutter in her chest;
The pressure in the lock is counting down.
Her feet again move dirt from where it rests -
The ground this time a rusted red, not brown.

From years of time we spent in lecture halls
Our hearts are heavy with the thoughts of planes.
The atmosphere is laden with the call
To let our love for stars and planets wane.

To dreamers now who still chase after stars,
We give this plan to make that future ours.

Signatures

Bailey Miller

Member Number: 688234

Signature

Arno Prinsloo

Member Number: 820199

Signature

Brian Frew

Member Number: 820209

Signature

Brooke Reid

Member Number: 510348

Signature

Colin Murphy

Member Number: 544961

Signature

Conner Murphy

Member Number: 675877

Signature

Philip Guzman

Member Number: 808450

Signature

Dr. Mark Ewing

Member Number: 100975

Signature

Contents

List of Figures

List of Tables

List of Symbols

AA Actuator Assembly

ADCS Attitude Determination and Control System

AIAA American Institute of Aeronautics and Astronautics

BEAM Bigelow Expanding Activity Module

BEES Buzzword Extraterrestrial Exploratory Satellites

BPF Band Pass Filter

C&DH Command and Data Handling

CEVIS Cycle Ergometer with Vibration Isolation and Stabilization

CMG Control Moment Gyros

COLBERT Combined Operational Load-Bearing External Resistance Treadmill

DDMO Deimos Distance Mars Orbit

DSN Deep Space Network

ΔV Change in Velocity

ECLSS Environmental Control and Life Support System

EEG Electroencephalograph

EGS Exploration Ground Systems

EPR Earth Penetrating Radar

ESA European Space Agency

EUT Electra UHF Transceiver

EVA Extra Vehicular Activity

GPR Ground Penetrating Radar

HD High Definition

HGA High Gain Antenna

HiRise High Resolution Imaging Science Experiment

HIVE High Intensity Vehicle Expulsor

ICES International Conference on Environmental Systems

ISRU In Situ Resource Utilization

ISS International Space Station

LEO Low Earth Orbit

LGA Low Gain Antenna

LMO Low Mars Orbit

LOX Liquid Oxygen

MAVEN Mars Atmospheric and Volatile EvolutioN

MCC Mission Control Center

MIT Massachusetts Institute of Technology

MOONS Mars Orbiter Operating Near Satellites

NASA National Aeronautics and Space Administration

QPSK Quadrature Phase Shift Keying

RAM Random Access Memory

RFAE Radio Frequency Amplifier Enclosure

RFP Request for Proposal

ROCKY Resistive Overload Combined with Kinetic Yo-Yo

RTG Radioisotope Thermoelectric Generators

SDST Small Deep-Space Transponder

SLS Space Launch System

SSR Solid State Recorder

TOF Time-of-Flight

TRL Technology Readiness Level

TWTA Traveling Wave Tube Amplifiers

UHF Ultra-High Frequency

USO Ultra-Stable Oscillator

VMC Vehicle Management Computer

VR Virtual Reality

VSHM Vector/Scalar Helium Magnetometer

Chapter 1: Mission Description

The purpose of this section is to discuss the mission described by the American Institute of Aeronautics and Astronautics (AIAA) Request for Proposal (RFP) for the manned Mars orbital mission design.[17] The RFP for this competition states that the purpose of this competition is to develop a precursor orbital mission to Low Mars Orbit (LMO). This mission is said to be analogous to the Apollo 8 mission that went before the historic Moon landing that took place in 1969. Like the Apollo 8 mission, this mission to Mars will hopefully lead to follow-on surface missions, but new technologies need to be tested before that can happen. This team has determined that such technologies include extended periods in stasis, methane-based propulsion, nuclear power generation, and inflatable habitats. If these technologies can properly mature, it is believed that not only will humans be ready for a visit to Mars, but possibly the farthest reaches of our solar system. The purpose of maturing these technologies is to ensure safe, reliable, and scalable missions for future astronauts.

Apart from the overarching goal of putting humans safely in LMO, the RFP specifies a couple of other requirements that a successful design must address. A proposed design should address solutions to make the mission and its architecture safe, reliable, affordable, and operable. Trade studies should also be performed to determine the best compromises for the mission. From these a vehicle architecture can be specified and basic science missions can be set forth. For the vehicle, all of the systems and instruments should be described in detail with an emphasis on the ECLSS. The trajectory of the vehicle should also be laid out from launch to return and every maneuver in between. The ground operations over the course of the mission should also be described. The last requirement is that the described mission should not exceed a total cost of five billion US dollars. The above requirements are described as the firm requirements. Throughout the report, some derived requirements will be presented and discussed on a per system basis.

Chapter 2: Executive Summary

The expansion of mankind from Earth to Mars will be a journey of many steps, the first of which is demonstrating the capability to safely get humans there and back. An orbital mission which follows in the footsteps of Apollo 8 will set the precedent for follow-on missions that aim to land humans on the face of Mars. The first mission to take humans to Mars orbit should demonstrate the same technology as later missions to prove the methodology and pave a path to human habitation of Mars.

This was the driving ideology behind the MOONS mission concept. Every system was selected to be safe for the crew and scalable to the larger missions that will eventually land humans on Mars. The proposed science objectives all directly enable future, longer manned missions to the Mars system, and eventually habitats on Mars and its moons. The combination of emerging technologies that will be necessary for regular Mars missions, such as inflatable habitats and crew stasis chambers, with proven systems seen on the International Space Station (ISS) provide safety to the crew while still taking advantage of the latest developments in manned spaceflight. A full four-view of our configuration can be seen in Appendix D, with our mass and power distributions shown in Table 2.3.

The MOONS mission will use a trajectory optimized for low TOF to carry a crew of four to the Mars, Phobos, and Deimos system. Of our four crew, two will be male and two will be female to better characterize how gender impacts the effects of deep space on the human body. The vehicle will be assembled in Earth orbit via three launches from Earth. The propulsion system uses methane, both for its high density compared to traditional rocket fuels, which is important as the MOONS vehicle is volume limited, and for its potential to be refueled using resources harvested on Mars in future missions. The mission will have a total ΔV of 11.7 km/s and a TOF of 777 days, departing Earth on or near the 8th of November, 2026. ΔVs for the mission are broken down in Table 2.1.

Table 2.1: Mission ΔV and TOF Breakdown

Mission Leg	**ΔV (km/s)**	**TOF (days)**
Earth to Mars	5.56	304
Mars Orbit Transfer	0.595	149
Mars to Earth	5.55	324
Total	11.7	777

Table 2.2: System Mass and Volume Breakdown

	Volume of Propellant (m^3)	Volume of Tanks (m^3)	Mass of Fuel (kg)	Mass of Tanks (kg)	Total Mass (kg)
CH4	74	100	28846	488	29334
LOX	87	126	92308	420	92728

The majority of our habitable space is provided by an inflatable habitat deemed the B125, similar to the Bigelow Expanding Activity Module (BEAM) currently on the ISS. Inflatable space habitats allow comfortable volumes of habitable space while reducing the mass and required volume in launch vehicles. With their plans to make inflatable habitats for deep space missions already, working with Bigelow Aerospace to develop the B125 is a natural fit for our mission architecture. The B125, along with the Kilopower reactor, CMG assembly, and propulsion system, is located inside the external truss structure. The truss is composed of two out-of-phase helical structures with four longerons along its length and ring frames at endpoints and key mounting points. Truss members are composite tubes to increase torsional strength and resistance to out-of-plane loads while reducing the total weight of the structure. The portion of the truss that houses the B125 contains telescoping members attached to hinges that allow for expansion and contraction while the B125 habitat is inflated or deflated.

The MOONS spacecraft will also incorporate a hardshell module, similar in construction to ones used on the ISS, in the form of a torpor and Extra Vehicular Activity (EVA) module. This module will serve as the primary module for containing mission critical systems and computers, as well as torpor chambers and externally mounted suits for EVA. In transit, the crew will demonstrate the capability of torpor chambers currently under development by SpaceWorks. These chambers induce a stasis that reduces the volume of consumables required for human travel. While the consumables savings for only four crew are small, a system of this type will be essential to bringing enough people to Mars for a sustainable colony. Building in these systems allow later missions to use MOONS as a precedent for large scale torpor use. This torpor chamber will also utilized externally mounted Z-2 suits for EVA, debuting the externally mounted space suits commonly seen in Mars surface mission architectures. EVA was deemed a necessary system for the MOONS mission. Without EVA, crew would have no way of reliably performing repairs to the spacecraft exterior in the case of a system failure or micrometeorite impact. EVA capability mitigates risk across all systems. A Dragon II capsule will be used to ferry the crew between the spacecraft and the ground, and left in Earth orbit when the spacecraft is en route to Mars. The ECLSS is powered by a series-Bosch reactor, recovering 90% of used water and oxygen throughout the duration of the mission.

Subsystems on the MOONS vehicle were designed to meet the requirements of safe, long-duration manned space flight. The command and data handling system uses Ka-band and X-band antennae to communicate science and system data, as well as personal communication to and from Earth. An Ultra-High Frequency (UHF) antenna will be used for communication during EVA and with probes. It will be capable of an average data rate of 12.6 Mbps in Mars orbit, allowing for streaming High Definition (HD) videos and voice calls with Earth. At its furthest from Earth, the communication system will have a significant time delay, so communication will be largely in the format of voice and video messages as opposed to live streaming.

Attitude control is provided by a five L3 Double Gimbal CMGs to deliver 4760 N-m-s of angular momentum. The power system uses lightweight, radially packed Orbital ATK Megaflex solar panels in conjunction with a Kilopower nuclear reactor and lithium-ion batteries. This combination provides redundancy for failure and scalability with the use of nuclear systems, which can be scaled to much larger missions once the technology is proven. This power system will be capable of delivering an approximate total of 35 kW.

While near Mars, the crew will observe the surface of the red planet by using instruments on the spacecraft as well as deployable CubeSats. The Mars observation objectives tackle questions concerning the geology and climate of Mars using synthetic aperture, which will allow better planning for Mars bases. These tests are specifically to find areas rich in resources such as methane or liquid water which would allow for more bases sustainable by In Situ Resource Utilization (ISRU) on the surface of Mars. The MOONS mission will continue by entering an orbit that can perform similar observations on Phobos and Deimos. These instruments include Earth Penetrating Radar (EPR), magnetometry readings, and camera imaging. Completing these science objectives will better characterize these moons and allow intelligent planning for future Phobos and Deimos bases.

Upon return, the MOONS vehicle will dock with the Dragon II capsule left in orbit to allow the crew to return to Earth. At this point, the ground crews will work to quantify the wear on the the systems and plan servicing missions for the spacecraft while it is in LEO. After it is cleared for reuse, the spacecraft can be refueled and future missions using the same vehicle can be performed. Variants of this mission using the same propulsion system with longer times of flight would allow for larger usable dry masses. A variant bringing a Mars lander using the MOONS vehicle is proposed.

The MOONS mission has a total estimated cost of $4.16 billion, below the $5 billion allotted by the RFP. Considerations are made to the business aspects of the mission and how the MOONS mission fits into the current state of space exploration from an economic and socio-cultural standpoint, showing a positive forecast for our mission feasibility due to optimistic economic outlooks and rising public interest and approval of space exploration.

These core elements define what separate the MOONS mission from others. The vehicle is designed to be reusable and scaled to larger crew sizes, and develops the technology to do so. While in the Mars system, it leverages its scientific capabilities to seek out the potential locations for future landing sites, clearing the way for the next steps towards colonization. Upon return to Earth, it is then readily modified to fly those next missions, and eventually see mankind to its first steps on the red planet. It is for these reasons that we believe the MOONS mission is representative of how any first manned Mars orbital mission should be conducted. Not only for its dedication to crew safety, but for its choice to actively prepare mankind for an interplanetary existence.

Table 2.3: Mission Mass and Power Breakdown

Subsystem	**Mass (kg)**	**Power (kW)**
Dragon II	12,000	-
Torpor	10,100	9
ECLSS	7,670	3.3
Propulsion	128,000	0.13
C&DH	18.8	0.07
Communication	228	2.41
ADCS	1,430	2.48
Thermal	1,190	1.5
Power	2,440	0.13
Science	176	20
Launch	163,000	-
Spacecraft	151,000	35

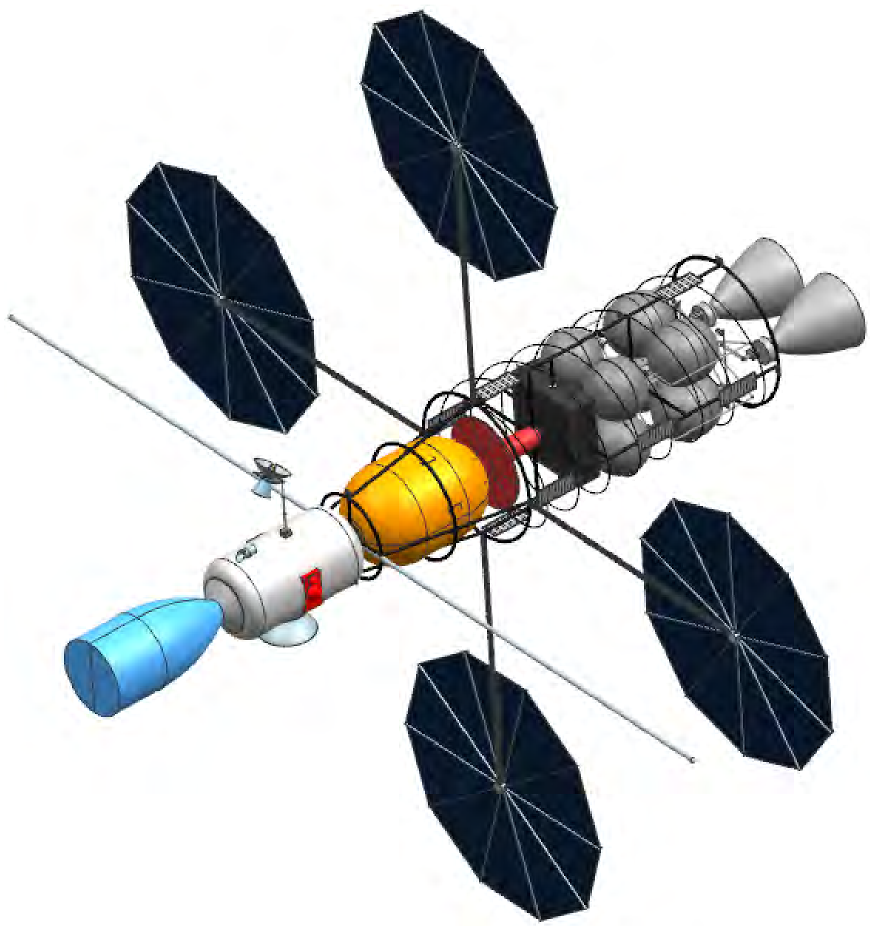


Figure 2.1: MOONS Vehicle Configuration

Chapter 3: History

While there is a long history of exploration and discovery in regards to Mars, this mission will satisfy several firsts, namely the first manned mission to Martian orbit, as well as the first mission to ever focus primarily on the Martian moons of Phobos and Deimos. In this section, a history of Martian exploration will be examined. By knowing where humanity has been, the next steps can be more effectively chosen to avoid replicating completed experiments or repeating past mistakes.

The first meaningful forays into the Martian environment outside of science fiction arose during the golden age of space exploration in the 1960s. The US and USSR both attempted several unmanned probes, many of which were unsuccessful in even reaching Earth orbit. The first probes to actually arrive in Mars orbit successfully were the Mariner missions. The various Mariner probes provided the first-ever close-up photos of the planet, as well as testing the atmospheric conditions and composition. In the midst of the Mariner missions, the USSR launched their Mars program. After some failures, Mars 3 became the first lander to reach the planet successfully. However, it broke down almost immediately after landing. The corresponding orbiter continued to collect temperature and atmospheric composition data, however. Subsequent missions in the Mars series were attempted, but almost all of them failed in some way.

By the mid-1970s, the USA's Mariner program had ended, and the National Aeronautics and Space Administration (NASA) began a new program, called Viking. The Viking program included landers which searched for microorganisms on Mars. In addition to this, color photos were taken from the surface, and the weather patterns were monitored over time. Meanwhile, the orbiters mapped the planet's surface. The Viking probes recorded data for several years before failing. In the late 80's, the USSR attempted one of the few missions dedicated to the Mars moon of Phobos in the creatively-named Phobos missions. Two missions were launched, but neither was successful.

In 1996, the United States launched Pathfinder, which delivered a lander to the surface of Mars. This lander deployed a rover named Sojourner, who explored the surface and gathered various types of data. The real focus of this mission, however, was to exemplify that low-cost landings could be done on Mars. Several years later, in 2001, the 2001 Mars Odyssey reached Mars orbit. This important orbiter acted as a communications relay for future Mars missions and has been in operation ever since. This probe was used in most subsequent Mars missions to help communicate back to Earth.

In 2003, the Spirit and Opportunity rovers launched for Mars. These rovers traversed the Martian surface, taking pictures and gathering data. Communications were lost with Spirit in 2011, but Opportunity is still active. The Mars Reconnaissance Orbiter was launched in 2005, and is currently orbiting the planet. This orbiter has served as an updated relay satellite for later missions. In 2007, the Phoenix Mars Lander was launched. This lander focused on studying the history of water on Mars, as well as the habitability potential on or in the surface. This mission ended in 2010. In 2011, the Russian Phobos-Grunt mission launched, one of the few Phobos-focused missions to date. This probe was to land on Phobos and return samples to Earth. Unfortunately, the probe failed in Earth orbit and was left on the wrong trajectory, never completing its mission. Also in 2011, the Mars Science Laboratory was launched, including the rover Curiosity. This mission focused on testing the environment to see if microbes were ever present on Mars. This mission is still active to date. In 2013, the Mars Atmospheric and Volatile EvolutioN (MAVEN) mission was launched. MAVEN was designed to test the Martian atmosphere to understand the climate change over Mars' history. MAVEN was the last NASA mission to travel to Mars.

Many of our science objectives were based off of these missions, in particular missions that failed or could be updated with current technology, such as Phobos-Grunt. The science objectives section of this report will go into detail describing this.[18]

Chapter 4: Trajectory

The purpose of this section is to discuss the launch, on-orbit rendezvous, and trajectory of the MOONS spacecraft.

4.1: Launch

The MOONS spacecraft will launch in three stages using the Space Launch System (SLS) Block 1B, the Falcon 9, and the Falcon 9 Heavy launch vehicles. The first launch will be of the SLS and will contain all of the propulsion system tankage with most of the propellant as well as the two vaccuum rated Raptor engines, the nuclear reactor, the solar panels and booms, the B125 module, and the truss structure. The B125 will be deflated for launch and the truss structure surrounding the B125 will be collapsed via telescoping beams and a hinge mechanism to reduce the overall length. The tanks will be filled so that the SLS is filled to capacity by mass. The launch capacity is assumed to be 130,000 kg to LEO. The Falcon 9 will launch second and will carry up the torpor module to LEO. The launch capacity to LEO for a Falcon 9 is 22,800 kg. The remaining capacity after torpor is loaded will be 12,700 kg. This Falcon 9 is being used because the fairing on the Falcon 9 Heavy is just too short to fit the torpor and the Dragon II modules. Future follow-on missions will only require a Falcon 9 to ferry the crew to the spacecraft that has been left in LEO. The Falcon 9 launch described would not include the required propellant to refuel the spacecraft. The remaining propellant mass for this mission will be launched using an auxiliary tank on the Falcon 9 Heavy. The Falcon 9 Heavy will have the remaining propellant, any consumables, and the Dragon II capsule with the

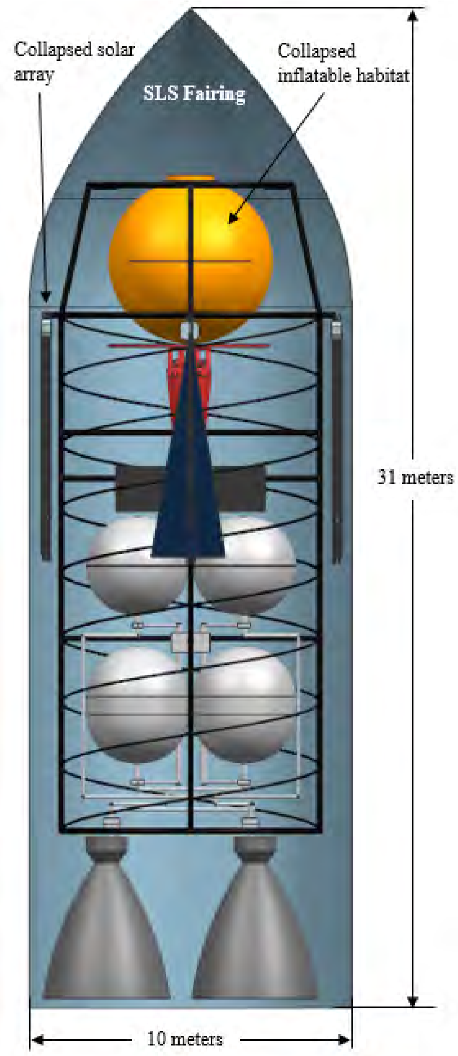


Figure 4.1: SLS Launch Configuration

crew. The Falcon 9 Heavy launch has a capacity of 63,800 kg. This launch will not be filled to capacity and should have a margin of 11,700 kg. The three launch loads can be visualized in Figures 4.1 and 4.2.

4.2: Rendezvous

The full MOONS vehicle will need to be assembled on-orbit because of the split in launches. The SLS and Falcon 9 launches will leave well before the specified launch opportunity. While it is in orbit, the B125 module will be inflated and all of the power generation systems will be checked. Once every thing is determined to be in working order from the ground, the Falcon 9 will be launched to connect the torpor module. After torpor is connected and all systems are determined to be in working order, the Falcon 9 Heavy will be launched. Once in orbit, the two crafts will rendezvous with the Dragon II capsule connecting to the torpor module. At this time the propellant tanks will be filled to capacity using the auxiliary tank attached to the Dragon II capsule. Once the spacecraft has been checked, filled, and checked again, the Dragon II capsule and auxiliary tank will be disconnected to be left in LEO for the remainder of the mission. The crew will then wait any remaining time for the transfer opportunity.

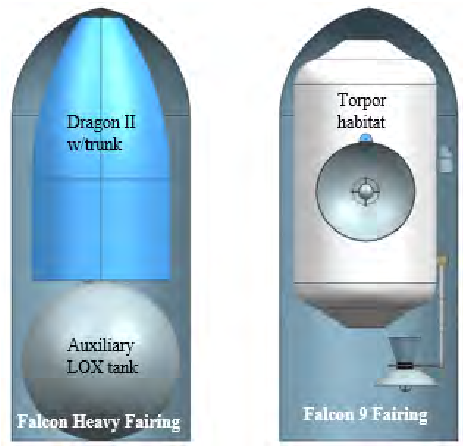


Figure 4.2: Falcon 9 and Falcon 9 Heavy Launch Configuration

4.3: Mars Transfer

Through this section multiple solutions will be discussed as well as the simulation that was constructed to find the optimal trajectory. In order to optimize the trajectory a couple of requirements need to be defined. In an effort to minimize any physical degradation of the crew, the TOF should be as low as possible. With a low TOF, the trajectory chosen should still have as low of a ΔV as possible. Lastly, while at Mars there will need to be a Hohmann transfer from a highly inclined LMO parking orbit to a Deimos Distance Mars Orbit (DDMO) to pursue science objectives around Mars' moons.

When discussing orbital mechanics, the most efficient transfer maneuvers are Hohmann transfers. These maneuvers require propulsive burns at the perigee of the initial orbit and secondary burns at the transfer orbit's apogee. When considering rendezvous it is important to find the synodic period. For a Hohmann transfer to or from Mars, the synodic period is 780 days. This means that if a launch date is scrubbed then another launch could be done 780 days later. The main reason a Hohmann transfer was not selected is because the TOF would be very large. The ΔVs for this launch as well as TOF are shown in Table 4.1.

The first trajectory solution that should be discussed is one presented by NASA. NASA is looking to put people on Mars near the year of 2033 using one of two trajectory options. These options are named the Conjunction and Opposition trajectories. The conjunction trajectory is characterized by an extremely low ΔV but high TOF. The opposition trajectory is similar but has a much lower TOF and a higher ΔV. This is achieved by departing Mars and then using a gravity assist around Venus to reach Earth. All of the characteristics of these trajectories are laid out in Table 4.1. The reason neither of these trajectories was selected is because the ΔVs from the report are lower than the Hohmann trajectory. It's believed that there is missing information in the report[19] so without concrete numbers these cannot be used.

With the introduction of the opposition trajectory, there was a mention of Venus gravity assist. This gravity assist can be used on a return trajectory from Mars to reduce the TOF further. But this flyby opportunity isn't available for every Mars return. There are also a couple other issues with the gravity assist. The structure required for the assist would increase the mass of the spacecraft by no small margin. The closer distance to the Sun would greatly increase the radiation exposure to the astronauts.

Table 4.1: Candidate Trajectory Overview

Trajectory	**Earth-Mars ΔV (km/s)**	**Mars-Earth ΔV (km/s)**	**Total ΔV (km/s)**	**Total TOF (days)**
Hohmann	4.90	4.26	10.3	1037
Conjunction[19]	1.75	1.06	3.41	1005
Opposition[19]	2.36	3.33	6.29	560
Optimized	5.56	5.55	11.71	777

To find the optimal trajectory, a script was developed using Matlab to solve Lambert's problem. Lambert's problem was to find the ΔV required at departure and arrival of two different points in a gravitational field traveling for a set amount of time. This problem is developed and described in a paper by Massachusetts Institute of Technology (MIT) graduate students.[20] By brute force iteration, they found the optimal launch times to Mars and the ΔVs associated with each launch date. This data was used as a baseline because a similar method was used for finding the optimal trajectory to Mars for this mission. As an added complexity, the script developed for this design iterated through the wait times at Mars to find the optimal arrival and departure dates. The script output the tell-tale porkchop plot, which can be seen in Figure 4.3. Figure 4.3 shows the optimal Earth to Mars ΔV map with the X-axis showing departure dates, the Y-axis showing the TOF between Earth and Mars, and the color representing the total ΔV required for that leg. Similar plots were output for each wait time increment for the Mars to Earth return. Using these outputs the optimal trajectory was found.

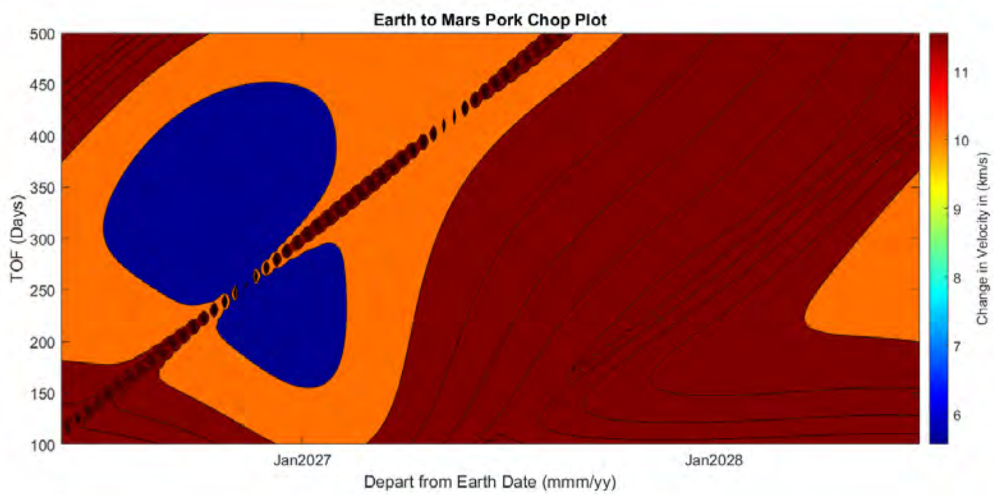


Figure 4.3: ΔV Map for Earth to Mars Leg

The trajectory that the MOONS spacecraft will take is shown as the light blue curves in Figure 4.4. The arrows point to each burn and the TOF is labeled for each leg of the mission. All of the information is summarized in Table 4.1. During the stay at Mars, there will be a transfer from the highly inclined LMO arrival orbit to the DDMO departure orbit. This transfer will require a departure burn of 318 m/s at LMO and a circularization burn of 276 m/s at DDMO arrival. The total ΔV for this transfer, 595 m/s, has been tacked onto each trajectory total seen in Table 4.1 because no matter how the spacecraft gets to Mars it will have to perform this maneuver to achieve the science objectives. The timing of this burn is dependent on the trajectory chosen but will be specified so that the satellites of Mars can be observed and the optimal return trajectory can be achieved.

The optimal trajectory described was found using a brute force Lambert solver that iterated over launch date, TOF, and wait time at Mars. The optimal trajectory was found to have a launch window on November 8th of 2026. The total TOF of the mission is 777 days including a 149 day

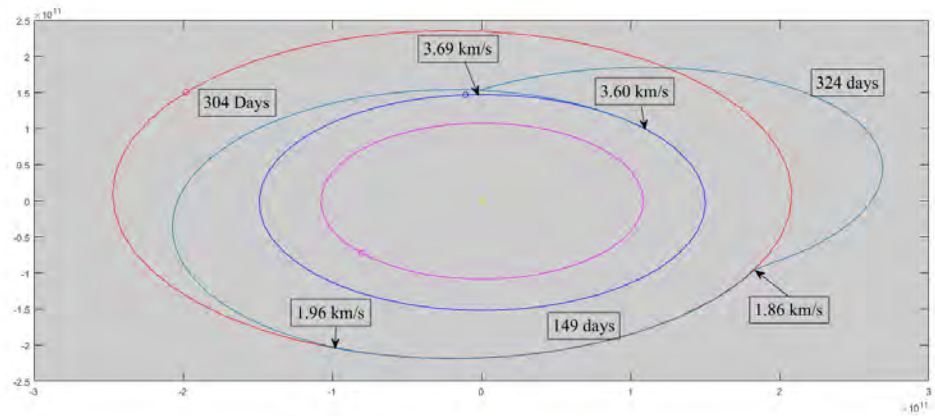


Figure 4.4: Spacecraft Trajectory Overlaid with the Planets

stay in Mars orbit. The total ΔV required is 11.7 km/s including a Hohmann transfer at Mars from LMO to DDMO requiring 595 m/s of ΔV. The total value slightly exceeds the Hohmann transfer mission ΔV of 10.3 km/s lending validity to the results as being optimized. Both of these results greatly exceed the total ΔV laid out in the NASA trajectory report.[19] It is believed this report did not give all of the information required to define a full mission trajectory. Furthermore, a Venus gravity assist was not used because of the complexity in trajectory as well as concerns for the radiation exposure of the crew and the mass gained from additional structure required.

4.4: Earth Return

Once the spacecraft has returned from Mars, it will rendezvous with the Dragon II capsule that has been parked in LEO. Once their, the crew will board the Dragon with any souvenirs and physical science, such as the Solid State Recorder (SSR). After the crew and science has been transferred, the Dragon II will deorbit for re-entry. This leaves the MOONS spacecraft in orbit for future missions to refuel and reuse.

Chapter 5: Human Habitation and Safety

This chapter will discuss all mission aspects regarding keeping the crew alive and comfortable throughout the mission. The first section will discuss the habitat selection process to decide which modules will be used. The second section will cover the fitness and health accommodations throughout the mission. Finally, the radiation measures taken to protect the crew will be described.

5.1: Habitat Selection

For this mission, several habitats were considered. A variety of habitats were initially examined and qualitatively ranked comparatively to each other based on power, mass, volume, and cost. This information is shown in Table 5.1. In this table, the lower numbers represent more desirable characteristics. These modules were ranked solely considering habitability potential in terms of power usage, mass, volume, and cost.

Table 5.1: Qualitative Analysis of Various Habitats

Vehicle	Power Usage	Mass	Volume	Cost
Orion	8	8	6	8
Dragon II	1	2	7	2
CST-100	2	4	8	6
ATV	7	7	3	7
Tranquility	4	6	2	5
Destiny	5	5	1	4
Columbus	6	3	5	3
Cygnus	3	1	4	1

While these numbers were obtained after relatively rough estimations, the general trends stand. Clearly, Orion is incredibly costly in terms of power, mass, and cost. However, there are only a few actual crew modules considered here (Orion, Dragon II, and CST-100). By comparing only these three options, it is clear that Dragon II is the best candidate from a power, mass, and cost standpoint. While the volume is lower than may be ideal, it will be supplemented by other habitats, so this is not a significant factor. For these reasons, Dragon II was chosen as the primary crew module. Next supplementary modules were chosen. The other habitats considered above are mainly ISS modules. While useful, ultimately the decision was made to pursue torpor technology, so Spaceworks' torpor module was chosen. This will be discussed further later in this section. The decision was also made to abandon the hard module technology observed in the above table to instead examine using inflatable modules by Bigelow Aerospace. The selection process for the inflatable module will also be discussed later in this section. Once torpor and inflatable modules were considered, a new trade study was performed examining the various configurations of modules. This information is shown in Table 5.2. Ultimately, Space X's Dragon II will serve as the

Table 5.2: Second Iteration of Habitat Sizing

Configuration	Max Diameter, m	Length	Total Volume, m3	Habitable Volume, m3	Power used, W	Liftoff Mass, kg
Orion + Torpor + BEAM	5.02	17.53	169.9	54.95	7380	42359
Orion + Torpor + B330	6.7	27.22	736.9	368.95	27005	60999
Orion + BEAM	5.02	10.03	69.9	24.95	-9620	24760
Orion + B330	6.7	19.72	636.9	338.95	10005	43400
Dragon 1 + Torpor + BEAM	4.3	18.71	158	57	14000	30449
Dragon 1 + Torpor + B330	6.7	28.4	725	371	33625	49089
Dragon 1 + BEAM	3.7	11.21	58	27	-3000	12850
Dragon 1 + B330	6.7	20.9	625	341	16625	31490
Dragon 2 + Torpor + BEAM	4.3	19.61	157	56	14000	30959
Dragon 2 + Torpor + B330	6.7	29.3	724	370	33625	49599
Dragon 2 + BEAM	3.7	12.11	57	26	-3000	13360
Dragon 2 + B330	6.7	21.8	624	340	16625	32000

primary crew module, while Spaceworks' proposed Torpor stasis chamber is intended to house both a medical bay, science objectives, and EVA suit ports. In addition to these, a custom inflatable module based on the B330 module by Bigelow Aerospace will be used. Finally, an updated version of NASA's Unity Module can hypothetically be added to the spacecraft for scalability reasons. As such, it will also be discussed. The following paragraphs will go into more detail about the habitats chosen, including discussing the tradeoffs between other comparable modules.

The capsule used to ferry the crew to orbit will be Space X's Dragon II spacecraft. The Dragon II is capable of carrying up to seven crew members, but since only four will be used on this mission, the remaining space will be used for additional systems and instrumentation. The Dragon II contains both a large crew chamber but an even larger unpressurized storage area, which will be used to stow extra fuel to keep the capsule in orbit. The Dragon II is also designed to be reusable, a key tenet of our design philosophy. Dragon also boasts several other systems that lend themselves to Mars travel, given that Space X designed it specifically for interplanetary voyages. These features include a state of the art ECLSS for climate control, a built-in launch abort system (capable of both safely saving the crew from danger anywhere from launch to orbit as well as being used for propulsive landings), and an advanced carbon ablative heat shield, among other features. The Dragon II has not yet flown in orbit, but is scheduled to do so in late 2017 or early 2018. In addition to this, Dragon II is very similar to its predecessor, Dragon I, which is an established and proven vessel for unmanned space travel. For this mission, Dragon II will be used to get the crew to the primary vehicle, and then left in orbit until the crew returns. Table 5.3 shows some properties of the Dragon II spacecraft, as well as comparing the values to those for the Orion capsule.

Table 5.3: Comparison between Dragon II and Orion[6]

Parameter	**Dragon II**	**Orion**
Diameter, m	3.7	5.02
Length, m	8.1	6.02
Total Volume, m^3	24	36.9
Habitable Volume, m^3	10	8.95
Power Required, kW	4	11
Liftoff Mass, kg	12000	23400
Crew Capacity	7	6
Cost, millions of dollars	280	360

As this table shows, Orion boasts comparable crew capacity, but has a lower habitable volume and essentially double the liftoff mass. In addition, Orion costs an extra 120 million dollars. The investment for Orion would ultimately need to be even higher than this however, as Orion was designed to fly aboard NASA's SLS, while Dragon II can launch from the much smaller (and cheaper) Falcon 9 and Falcon 9 Heavy launch vehicles. Beyond this, Dragon also has a technical edge over Orion, being newer and therefore having more advanced technologies. Even though Dragon technically isn't available in today's market, the technology should be more than ready by the launch date.

The next habitat to be used in the mission is Spaceworks' torpor-inducing transfer habitat. Their proposed module is expressly designed to house human stasis pods which can keep crew members in a state of hibernation for rotating 2-week-long shifts. This is done to cut down on the amount of consumables needed for the trip, as well as to help crew members psychologically endure long periods of travel. The stasis process has been tested and proven on Earth, where it is used to sustain critically injured people. The torpor module also serves as a radiation safety zone due to being constructed from radiation-resistant materials. Torpor technology has never flown in space before, however. To help ensure that the system is ready in time, a portion of the budget for this mission will be going towards advancing Spaceworks' development of the torpor module so it will be ready in time for the mission. This is discussed further in the cost section of this report. By working with Spaceworks, the specific module needed for this mission can be developed, as the stock module that Spaceworks is proposing contains chambers for six crewmembers, while this Mars mission will only use two pods, leaving two crewmembers active at all times for safety reasons. By removing four of the stasis pods of Spaceworks' original design, more volume is freed up for exercise and medical equipment. Figure 5.1 shows a proposed schematic for the torpor module.

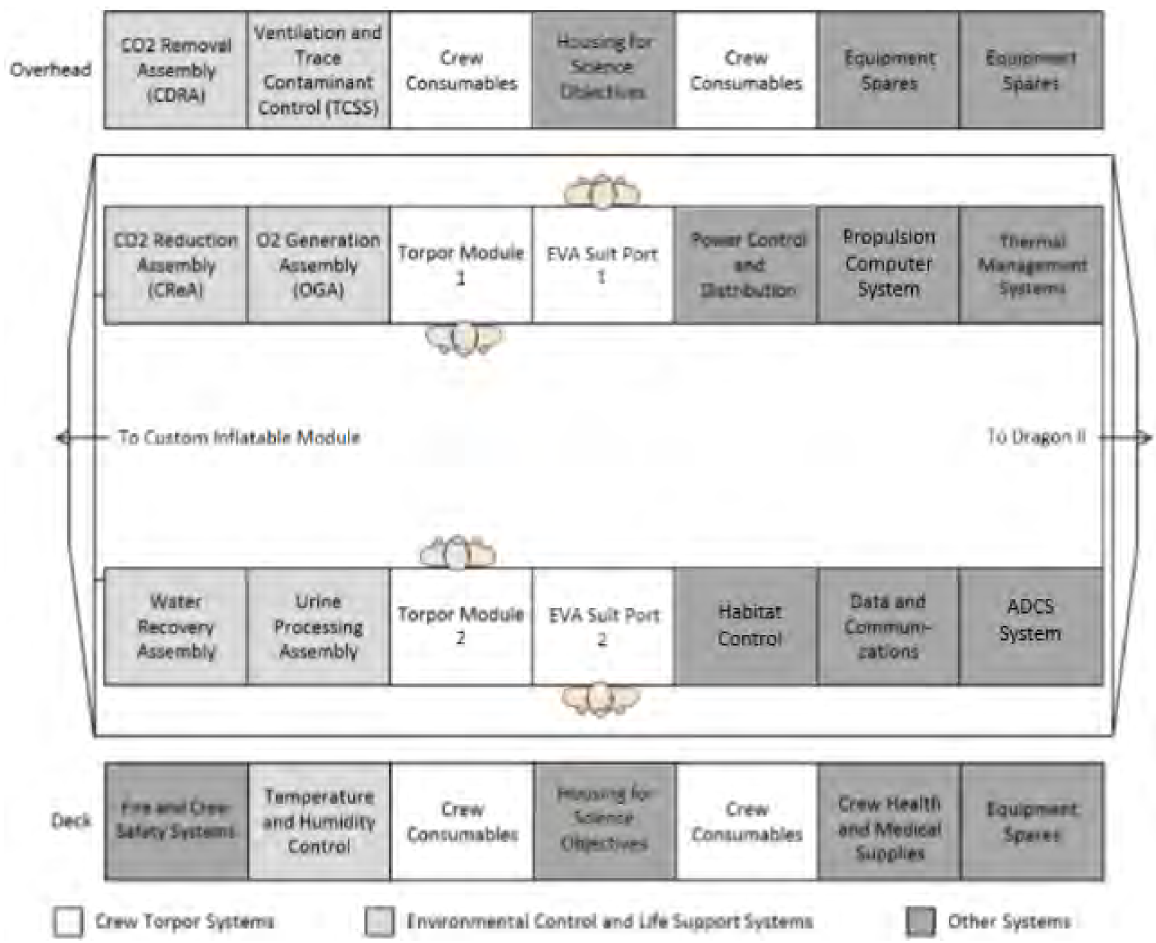


Figure 5.1: Schematic for the torpor module[1]

As this figure shows, there are two torpor pods, one on each side of the module. The torpor module will attach to the inflatable module on one end, and the Dragon II module on the other end when the crew is moving into the main mission vehicle. Adjacent to the stasis pods are EVA suit ports. These ports use a self-contained airlock built into the back of the space suit to allow for EVAs without using up precious volume on a large and intensive air lock. These suit ports haven't been tested in space yet, but are expected to be used on the ISS at some point. If so, they will be more than ready by the time the mission is occurring. Table 5.4 displays dimensional data for the torpor module.

Table 5.4: Data for the Spaceworks torpor module[1]

Diameter, m	4.3
Length, m	7.5
Total / Habitable Volume, m^3	100 / 30
Power required, kW	17
Liftoff Mass, kg	17599

Overall, the torpor module is one of the most advanced and high-risk technologies present on the mission. Despite the concerns of what a failure would mean when it is so integral to the entire mission architecture, it is believed that by the time the mission launches, the technology will be up to an acceptable level, especially if money from the overall budget is used to bolster the torpor development. These risks are worth taking as inducing torpor is the next step towards interplanetary and ultimately interstellar space travel. By proving the process on such an important mission, the road will be paved for further exploration of the concept, and eventually torpor technology may be as ubiquitous as computers onboard any long-duration spacecraft.

Attached to one end of the torpor module is a custom inflatable module, created by Bigelow Aerospace. This module is based off two other inflatable modules by Bigelow. The first of these is the BEAM. BEAM was created to test the expandable module concept, and was used on the ISS in 2016. During the test, the module was slowly inflated and left so for several months. This mission was considered a success, and Bigelow has plans for other, more ambitious modules in the future. The most developed of these is the second module used as a basis for the B125. This module, called the B330, is significantly larger than BEAM, and much more intensive in terms of equipment and storage located within. While still in the preliminary phases, the B330 is scheduled to be launched in 2020. Based on the information for these two modules, interpolation was used to calculate the proportional dimensions for a custom module with a desired habitable volume of 125 m^3. Table 5.5 presents the dimensions for these two modules. Based on the data in these tables, a new inflatable

Table 5.5: Data for the Bigelow Modules[7,8]

Parameter	BEAM module	B330 module
Diameter, m	2.36 (packed), 3.23 (expanded)	6.7 (expanded)
Length, m	2.16 (packed), 4.01 (expanded)	13.7 (expanded)
Total / Habitable Volume, m^3	33 / 16	600 / 330
Power required, kW	1	20
Liftoff Mass, kg	1360	20000

module was sized, dubbed the B125 due to it's 125 m^3 habitable volume. The values found are rough estimates, as they are based upon only two data points (and even some of those are suspect to an extent, as there isn't exact data for several aspects of the B330 module). Table 5.6 presents the dimensions for the B125.

Table 5.6: Derived Data for the B125 Module

Diameter, m	6.3 (expanded)
Length, m	7.37 (expanded)
Total / Habitable Volume, m^3	250 / 125
Power required, kW	8
Liftoff Mass, kg	7830

This module will ensure that there is enough habitable volume for each crew member, even in the event of the torpor module being rendered completely unusable. According to NASA guidelines, a crew of four requires at least 25 m^3 per person for a mission of this duration.[2] In total, these habitats provide about 150 m^3 of habitable volume.

While it will not actually be used on the first MOONS mission, there is one other proposed component to the habitat system in the form of the Unity module. The current Unity module is part of the ISS, and was the first US module sent up. This was because it was designed solely to be modular, so all the other ISS modules could connect to it. To that end, Unity has six berthing ports to connect to other structures. Adapters can be fitted for virtually any spacecraft desirable. If a new Unity module were constructed and included on the spacecraft for future missions (and attached to the torpor module), the scalability of the mission would vastly improve. Habitable volume could easily increase if more inflatable modules branched off from Unity, meaning more room for science or consumables, leading to longer mission durations. While not in use on the first MOONS mission, the potential applications are very attractive from a scalability standpoint.

5.2: Fitness and Health

Keeping fit through exercise is vital on any space mission, as the lack of gravity causes bone degradation and atrophy in the human body. This is especially important on such a long mission. In lieu of artificial gravity being applied to the MOONS spacecraft, the decision was made to use exercise equipment to keep the astronauts fit. The first piece of equipment to be used is the treadmill known as Combined Operational Load-Bearing External Resistance Treadmill (COLBERT). This is the primary means of exercise on the ISS, and is therefore at an established TRL of 9. COLBERT works well for astronauts, as the crew is physically strapped in using bungee cords. This requires the astronauts to apply force with their legs to work out.

The next piece of equipment to be used is the Resistive Overload Combined with Kinetic Yo-Yo (ROCKY) exercise system. This device serves as a sort of rowing machine that is well-designed to exercise the upper body. While lower TRL than COLBERT, ROCKY is designed to be the primary exercise device on Orion, so it should be at a TRL of 9 at the time of launch.

In addition to these exercise devices, healthy meals will be prepared for the entire mission, and vitamin supplements will be given to the crew daily. The science objectives portion of this report discusses more about the vitamins.

5.3: Radiation Protection

Table 5.7: LEO Career Dose Limits in Sv[2]

Age	25	35	45	55
Male	0.7	1.0	1.5	2.9
Female	0.4	0.6	0.9	1.6

Radiation protection is critical for manned missions in deep space to keep the crew alive and healthy for the mission duration, as radiation dosages are much higher in space than within Earth's atmosphere. Recent advances in shielding technology have shown that polyethylene-based plastics provide better protection from solar radiation and cosmic rays compared to traditional metal shielding. Initial tests have shown that RXF1,[21] when compared with aluminum, provides 15% better protection from cosmic rays, 50% better protection from solar flares, exhibits three times the tensile strength, and is more than twice as light.[22] Tables 5.7 and 5.8 show the various radiation limits for astronauts as defined by NCRP-132 ("BFO" = blood forming organs). Figure 5.2 shows a comparison of various materials and their shielding properties for variable areal densities.

Table 5.8: Recommended Organ Dose Limits[2]

Exposure Interval	BFO Dose Equivalent (cSv)	Ocular Lens Dose Equivalent (cSv)	Skin Dose Equivalent (cSv)
30-day	25	100	150
Annual	50	200	300
Career	See Table 2	400	600

Assuming that LEO dose limits are enforced for deep space travel, the shielding will vary from 2-5 cm thick with the areal density range shown below for the hard shell torpor module. Variable radiation shielding is chosen so that the weight of the armor is optimized. The 5 cm thick and 10 g/cm^2 dense armor will surround the two torpor chambers to provide effective shielding for crew members during torpor shifts. Crew members will be required to wear woven polyethylene radiation jumpsuits during waking shifts to supplement the 2 cm thick and 7 g/cm^2 dense armor. The radiation shielding mass for the torpor habitat is included in the habitat structure mass. The inflatable Bigelow habitat has woven polyethylene radiation shielding integrated into its shell, creating a fully protected habitable enclosure. The inflatable habitat will be the safety point for any high radiation events that may occur during the mission.

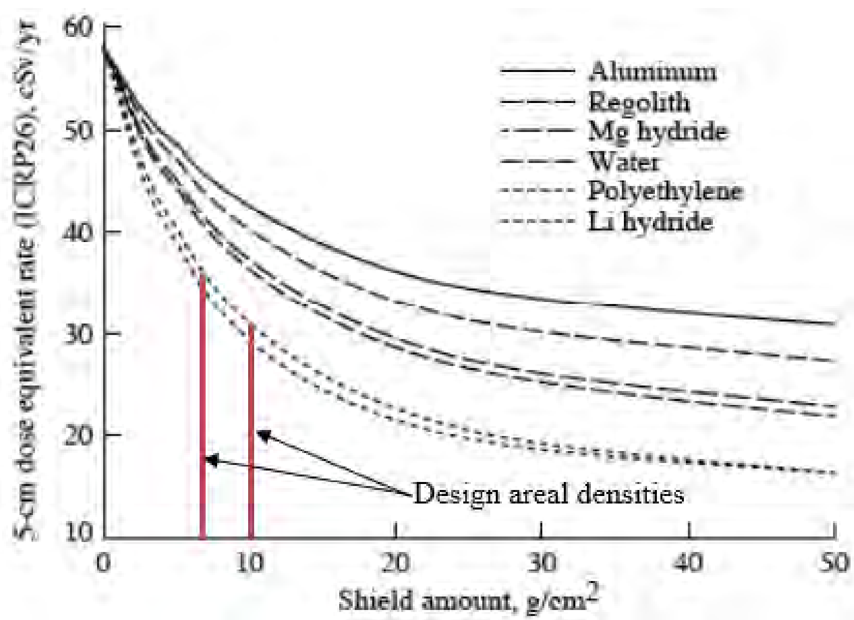
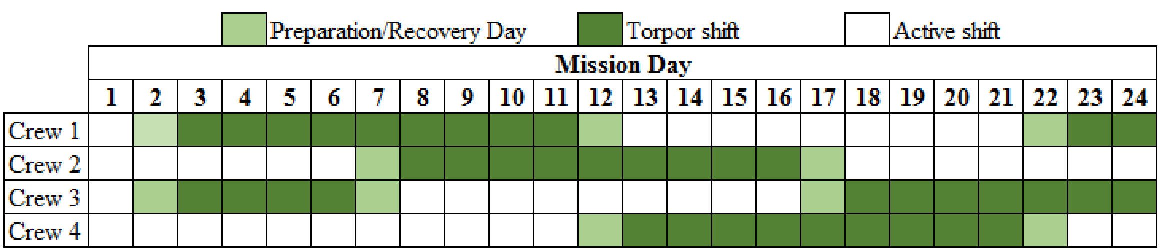


Figure 5.2: Radiation Shielding Material Comparison[2]

5.4: Crew Psychology

Deep space travel also has many psychological effects to be considered for manned flight. Crew composition must be carefully considered to determine if crew members have compatible personalities and temperaments appropriate for deep space travel. A crew of four was chosen for the MOONS mission because an even number reduces the chance of the 'odd man out' scenario where one crew member is alienated by the rest of the crew. A torpor schedule for a crew of four allows for two crew awake and two crew in torpor at all times. The active crew members will be tasked with monitoring the torpor system and performing science objectives and other necessary tasks. The torpor schedule is staggered to allow crew members to rotate their active partner during waking shifts. This mitigates conflict between crew members and increases social contact during the long travel times to and from Mars orbit. Table 5.9 shows a preview of the torpor schedule for the first 24 mission days.

Table 5.9: Torpor Schedule for First 24 Mission Days



This mission was designed for each crew member to have a dedicated role in the mission, based on a military-style hierarchy structure. The mission commander will have military leadership experience and pilot experience. This person will be tasked with mission supervision, safety, and maneuvering the spacecraft as needed. The pilot/flight engineer will have experience in both of those fields, and will be responsible for monitoring and troubleshooting systems and assisting with maneuvering and payload operation. The third crew member will serve as payload commander. This person will have a geology and software design background, and is charged with preparing and analyzing the payload systems and data. Finally, the mission specialist will be a medical expert in charge of monitoring the torpor systems and crew health, as well as processing medical research data. Cross-training will be necessary to ensure that all able crew members are capable of completing the mission if a crew member is incapacitated. Due to the long term isolation nature of the mission, thorough psychological studies of potential crew members is recommended to ensure crew compatibility and performance in high stress scenarios.

A report from NASA specified that at least one window must be included on the spacecraft for crew comfort.[21] The MOONS spacecraft features two EVA suit ports with visors that will act as windows. The crew will also have Virtual Reality (VR) goggles available that will allow them to 'see through' the spacecraft walls by using external cameras mounted around the vehicle. The VR system will also act as an entertainment system to increase crew comfort and also allow the crew to virtually access data and science applications during the mission. The spacecraft will also have voice and video streaming capabilities to keep crew members in contact with friends and relatives on Earth to ease feelings of isolation and disconnect.

Chapter 6: ECLSS

For the MOONS mission, the ECLSS must be responsible for maintaining atmospheric pressure, fire detection, oxygen levels, waste management, and the water supply. ECLSS reactors function based on chemical processes that recycle human breath, sweat, and waste into usable water and oxygen. The current chemical process used on the ISS reactor is the Sabatier reaction. However, many experts believe the Bosch process is more efficient and would make better reactors in the future. Figure 6.1 shows each step of the Bosch process.

RWGS	$CO_2 + H_2 \leftrightarrow H_2O + CO$	$\Delta H°_{rxn} = 41$ kJ/mol (1)
CO Hydrogenation	$CO + H_2 \leftrightarrow H_2O + C(s)$	$\Delta H°_{rxn} = -131$ kJ/mol (2)
Boudouard	$2CO \leftrightarrow CO_2 + C(s)$	$\Delta H°_{rxn} = -172$ kJ/mol (3)
Bosch Process	$CO_2 + 2H_2 \leftrightarrow 2H_2O + C(s)$	$\Delta H°_{rxn} = -90$ kJ/mol (4)
Water electrolysis	$2H_2O \rightarrow 2H_2 + O_2$	$\Delta H°_{rxn} = 242$ kJ/mol (5)

Figure 6.1: Each Step of the Bosch Process[3]

To start the process, CO_2 and H_2 are needed. The CO_2 is generated from the crew's breathing, and while the H_2 must initially be provided, ultimately the process generates more H_2, closing the loop. To most efficiently provide life support, a series-Bosch reactor will be used. In this configuration there are actually two reactors operating different steps in the process simultaneously as opposed to one reactor performing each step sequentially. This speeds up the process, increasing efficiency. Figure 6.2 shows a diagram laying out where each step of the process above factors into the reactor.

As this figure shows, the process starts with the reverse water gas shift reactor in accordance with Figure 6.1 before proceeding through two filters, Polaris and Proteus, which extract CO_2 and H_2, respectively. Finally, the products proceed through a carbon formation reactor to complete the process. Excess water is sent to an electrolysis chamber to be split into oxygen and hydrogen for the reactor. The dimensions of the reactor were assumed from a NASA report describing Bosch technology. Table 6.1 presents this information.

This information is based on a two-person crew; however, it is not expected that two more crew members will change the sizing of the reactor dimensions, just the amount of consumables

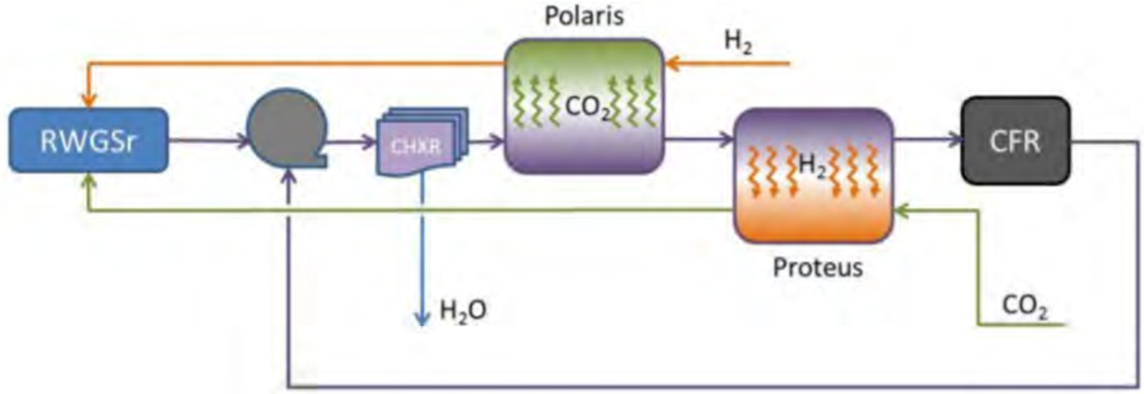


Figure 6.2: Diagram Showing the Flow of the Bosch Reactor[4]

Table 6.1: Series-Bosch Reactor Subsystem Breakdown[9]

Subsystem	Mass, kg	Volume, m^3	Power, kW
Air Subsystem	1,127	2.56	1.60
Food Subsystem	0	0.00	0.00
Thermal Subsystem	247	0.74	0.34
Waste Subsystem	187	4.54	0.01
Water Subsystem	1,110	3.20	1.12
Human Accommodations	316	0.50	0.23
Totals	**2,986**	**11.54**	**3.29**

needed. These consumables were sized using unit masses and volumes found from the same NASA report, multiplied by the number of crew members and total mission duration. This number was then multiplied by the projected efficiencies of the reactor to arrive at a final mass and volume estimate. Table 6.2 shows this information.

Combining these numbers results in a total ECLSS mass of about 8000 kg, and a volume of about 42 m^3. This volume will fit inside the torpor module as designed.

Table 6.2: Series-Bosch Reactor Subsystem Breakdown[9]

# CM	4					
TOF	777					
% O2 recovery	0.9					
% H2O recovery	0.9				After Recycling	
Items	Mass, kg/CM-day	Volume, m3/CM-day	Mass, kg	Volume, m3	Mass, kg	Volume, m3
Food	0.617	7.57E-03	1918	23.53	1918	23.53
Food Packaging (Food x 15%)	0.093	1.14E-03	289	3.53	289	3.53
Oxygen	0.84	2.77E-03	2611	8.62	261	0.86
Oxygen tankage	0.31	1.01E-03	963	3.14	96	0.31
Water	4.17	4.17E-03	12960	12.96	1296	1.30
Water tankage	0.834	8.34E-04	2592	2.59	259	0.26
LiOH and packaging	1.75	5.00E-03	5439	15.54	544	1.55
Oxygen leak	0.0088	2.90E-05	27.4	0.09	2.74	0.01
Oxygen leak tankage	0.0032	1.10E-05	9.95	0.03	0.99	0.00
Nitrogen leak	0.0353	1.16E-04	110	0.36	10.97	0.04
Nitrogen leak tankage	0.0196	6.50E-05	60.9	0.20	6.09	0.02
		Totals:	2.70E+04	7.06E+01	4.68E+03	3.14E+01

Chapter 7: Propulsion System

Table 7.1: Fuel Storing Temperatures[10]

Fuel Type	Storing Temperature (C)
H2	-253
CH4	-161
LOX	-183

The main requirements that went into the design of the propulsion system were the abilities to send a crew to Mars and back safely and to restart the engines when need be. The major design features of the propulsion system are liquid methane fuel, a heat exchanger to store methane and oxygen in liquid form, a pressurant tank to keep the fuel flowing steadily from the tanks, and four tanks each for methane and oxygen for redundancy and safety purposes. The propulsion system (along with the CMG assembly, Kilopower reactor, and B125) is housed in and mounted to the external truss structure. The truss is composed of two out-of-phase helical structures with four longerons along its length. Ring frames are located at key mounting points. Members in the truss are composite tubes to increase torsional strength while minimizing weight.

7.1: Reasons for Selecting Liquid Methane

Table 7.2: Pressure Needed to Keep Fuel a Liquid

Fuel Type	PSI Needed to Keep Propellants Liquid
CH4	2429
LOX	2485

The main reason for selecting liquid methane and liquid oxygen as the fuel and oxidizer is that this will prove ISRU technology, paving the way for the future of rocket propulsion. Some other benefits of methane are that a methane and Liquid Oxygen (LOX) mixture is a good choice due to its non-toxicity and similar storage temperature for the fuel and oxidizer, which reduces complexity. Storing temperatures are shown in Table 7.1. To keep the methane fuel in liquid form, a pressurant tank and a heat exchanger are needed.[23] A heat exchanger will be attached to a radiator on the spacecraft to get the energy needed to cool the fuel. However, since the fuel needs to be at such a low temperature there is a cooling process involved. First, cooled water will be condensed, cooling the chemical propylene. The propylene will then be cooled and condensed to cool ethylene. This is important because ethylene has the ability to get cold enough to keep the methane a liquid in the heat exchanger. Since the storing temperatures of liquid methane and oxygen are similar, the pressures needed to keep them liquids are also similar, as shown in Table 7.2. Since the design is volume limited, the large density impulse is significant because it means smaller storage tanks will be needed. Table 7.3 shows the difference in specific impulse and density impulse between liquid hydrogen and liquid methane. Even though the specific impulse for methane is lower than that of hydrogen, the density impulse is significantly higher which is beneficial for this mission. This contributed to liquid methane being selected as a fuel source along with the fact that the power and pressure needed to cool and keep it a liquid is lower.

7.2: Reasons Against Alternate Propulsion Systems

Table 7.3: Fuel Impulse Comparison

Fuel Type	Specific Impulse (s)	Density Impulse (kg*s/m^3)
H2O2	450	31860
CH4O2	350	147350

Nuclear propulsion was also looked at as an option for the propulsion system along with liquid hydrogen and liquid methane. After doing research and risk analysis it was determined that liquid methane was the most logical choice for fuel, while nuclear propulsion and liquid hydrogen had too many flaws to overlook. One of the benefits of having methane instead of hydrogen is that it is much easier to store than hydrogen. Passive cooling can keep methane cryogenic while hydrogen needs to be constantly cooled and will still vent over time.[24] This is a big factor in deep space travel or space missions with long duration periods. Methane is also denser than hydrogen, which means it can be stored in a smaller tank for the same mission. It is also easier to use in an engine because of its higher density than hydrogen, which means less pressure is needed to pump the fuel to the engines. A significant factor for choosing methane as a main source of propulsion is that it could potentially be produced on Mars by having imported hydrogen mixed with carbon dioxide to create methane. This re-emphasizes the point on why methane propulsion was chosen to prove ISRU science and technology. Also, opposed to kerosene, when methane is burned it leaves little to no residue on the engine, which increases the reusability compared to kerosene.

Using nuclear power as the main source of propulsion was ruled out at the beginning of the

Table 7.4: Volume and Mass of Selected Fuel

	Volume of Propellant (m^3)	Volume of Tanks (m^3)	Mass of Fuel (kg)	Mass of Tanks (kg)	Total Mass (kg)
CH4	74	100	28846	488	29334
LOX	87	126	92308	420	92728

Table 7.5: Mass Difference Depending on Change in Velocity

Fuel Type	Delta V = 10 km/s	Delta V = 11 km/s	Delta V = 12 km/s
LOX	76424 (kg)	84287 (kg)	93168 (kg)
CH4	24092 (kg)	26346 (kg)	29076 (kg)

selection process. While nuclear has a lot of potential upsides and the ability to launch much more weight into space while also taking up significantly less volume in the spacecraft, the technology is just not far enough along. The TRL for most nuclear propulsion systems is only three, with no real life testing really taking place until 2018 at the earliest. Even with initial testing beginning, that leaves a long time before it can be tested on actual spacecraft, especially with human lives on board. There are too many risks to take with a nuclear propulsion system, such as a failure on the launch pad that could be catastrophic to allow this system to be used. With our 2026 launch window, the decision was made that the nuclear technology would not be ready. The team acknowledges that farther in the future, with successful testing, the use of nuclear propulsion is most likely the best option for deep space travel due to the much faster attainable speeds and smaller volume needed.

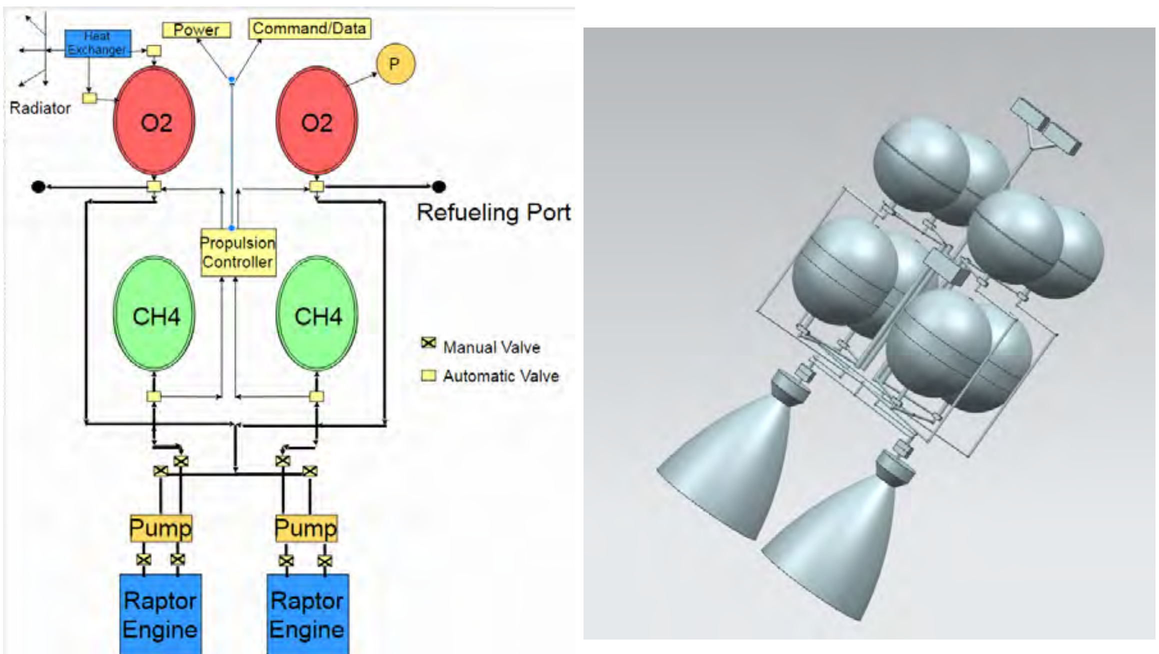
7.3: Fuel Mass and Volume

Due to the route that was chosen to get the spacecraft and crew to Mars it was determined that the total change in velocity needed for this mission is 11.7 km/s. To make this mission possible the propulsion system will require a total mass of roughly 121,000 kg. A breakdown of the volume and mass of the liquid oxygen and methane tanks are shown in Table 7.4. The higher the change in velocity the mission has, the more fuel needed, adding weight to the spacecraft. Since the launch vehicles have a limited payload space, it is crucial to keep the weight as low as possible. Table 7.5 shows the weight changes between 10 and 12 km/s for total change in velocity.

The engines used in this mission are two SpaceX Raptors. They are being tested and will be ready for the projected launch date. This engine was chosen because it was designed to use methane as fuel. There will be four liquid oxygen and four liquid methane tanks providing fuel for the Raptors. There are multiple reasons for this tank configuration. The size of these tanks was limited by the SLS fairing that is taking up the main propulsion system in the first of two launches. With a diameter of roughly 10 meters and only a limited amount of space, the mission is volume limited as opposed to mass limited in trying to fit systems into the SLS. This led to the idea of having four LOX tanks and four methane tanks. The reasoning for this is so that there can be redundancy in the propulsion system. By having four of each tank, the mission can still proceed even if one tank is critically damaged. If one tank of either LOX or methane is rendered unusable, the other three fuel tanks of the appropriate type have enough propellant to return the crew home safely. These tanks will be placed between the nuclear power generator and the living habitats so that if the nuclear generator malfunctions it will not be devastating to the crew on board.

7.4: Fuel Valves and Power, Command, and Data Lines

In the propulsion system there are many valves which are both manual and automatically controlled that provide different levels of redundancy and add safety to the system. Multiple valves in the propulsion system regulate the flow between the liquid oxygen and methane tanks to the Raptor engines. A radiator will be used to keep the methane at a storable temperature. This is important because methane and liquid oxygen need to be kept at specific temperatures so that they can be a usable fuel source. A pressure source is also attached to the tanks to compress the fluids inside the tanks to make it easier for liquids to flow into the system. Having a multitude of valves throughout the propulsion system allows for redundancy if a valve does not work somewhere and to make sure the correct fuel mixture ratio is being routed to the Raptor engines. The valves will need power command and data connectivity. The power line will be so that the valves can be turned on and off automatically and the data line will be so that the computer knows when enough liquid oxygen, methane, and heat from the radiator have gone through the system. There will also be two refueling ports so that the liquid oxygen and methane tanks can be easily refueled without having to move parts around or taking them out. The block diagram of the propulsion system can be seen in Figure 7.1a. The valves are also made so that they can be removed easily and fit somewhere else in the system if needed. For example, if one tank is compromised and no longer usable and a different tank's valve does not work, it is possible to take the valve from the compromised tank and replace the broken valve.



(a) Block Diagram of Propulsion System

(b) CAD of Propulsion System

Figure 7.1: Overview of Propulsion System

Chapter 8: Command and Data Handling

The C&DH subsystem includes the Vehicle Management Computer (VMC) and three SSRs. A block diagram of the system is shown in Figure 8.1. Required Random Access Memory (RAM) and throughput of the system is shown in Table 8.1 with a breakdown of individual system memory and throughput requirements. The system is sized to store 32.5 terabytes of data in the event that data cannot be transmitted to Earth and thus must be saved in the onboard SSRs until Earth return.

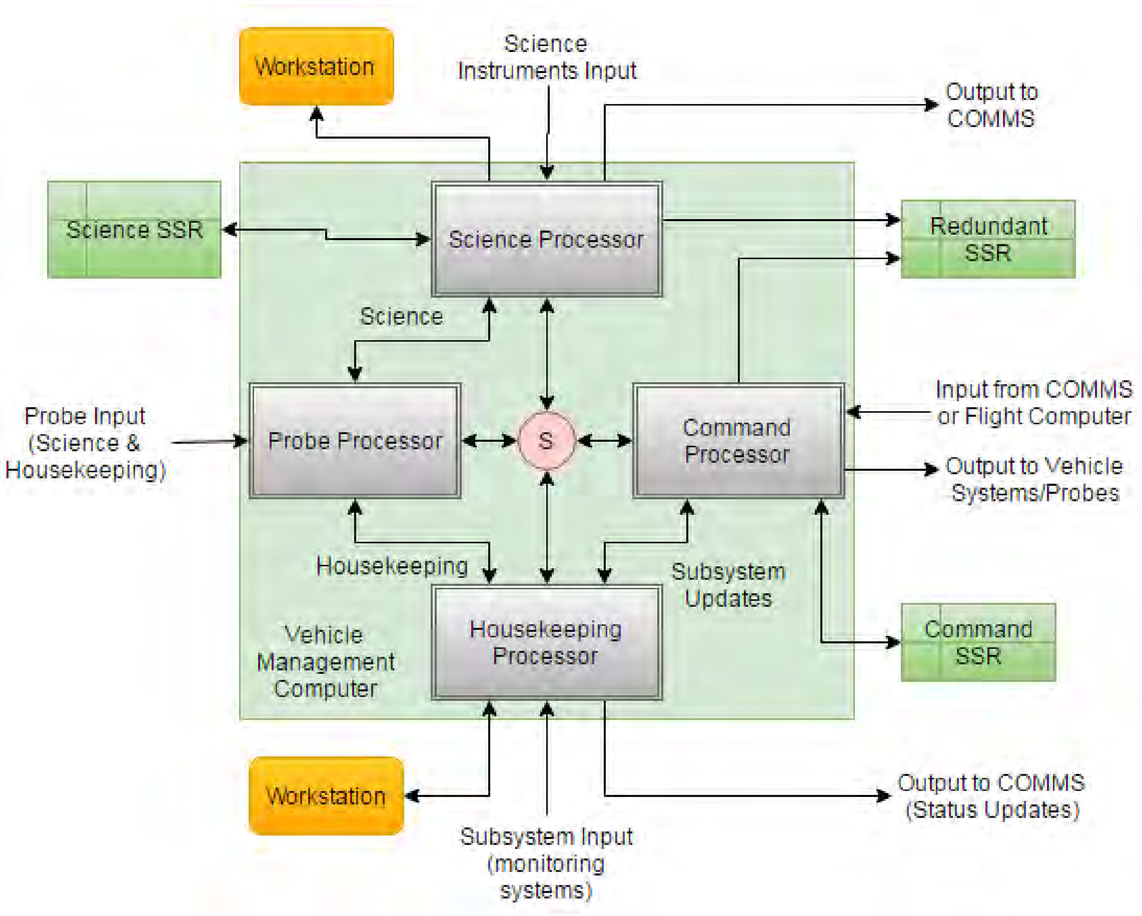


Figure 8.1: C&DH Subsystem Block Diagram

8.1: Vehicle Management Computer

The VMC contains many internal redundancies due to the harshness of the radiation environment and the importance of this subsystem. The VMC contains:

- Two redundant computing modules tasked with executing flight control and other software
- An ethernet network router module with hardline connections to all vehicle controllers operating onboard systems
- Processor unit with four independent modules (shown in the block diagram) which delivers the processing capability for the spacecraft to receive and execute commands, monitor various subsystems, and communicate with science probes and sensors

The modules of the processor unit operate independently but can distribute and assume the tasks of another processor in the event of an individual module failure. This connection is shown by the switch block in the block diagram shown in Figure 8.1. Mass and power requirements of the VMC are shown in Table 8.1. RAM and throughput requirements are shown in Appendix A.

Table 8.1: CDS and Telecommunications Mass and Power Breakdown

System	Mass (kg)	Power (W)
COMMAND AND DATA HANDLING		
VMC	8	35
SSR (3)	6.5	26
Data Links, Misc.	4.3	6
Totals	**18.8**	**67**
TELECOMMUNICATIONS		
SDSTs (2)	6	16
USOs (2)	1.7	5
RFAE1	116	1214
HGA1	14.7	0
Emergency Array	5.5	0
X-band TWTAs (2)	2	200
Ka-band TWTA	0.8	1000
Power Converters	4.5	0
Diplexers (2)	1.8	0
Misc. Components	3.1	0
Gimbals and Motors	45	14
Structure and Boom	38.7	0
RFAE2	85	1012
HGA2	33	0
X-band TWTA	1	0
Ka-band TWTA	0.8	1000
Power Converter	2.3	0
Diplexer	0.9	0
Misc. Components	2.2	0
Gimbals and Motors	37	12
Structure	8	0
UHF Subsystem	19.4	152
EUTs (2)	10.2	72
UHF Antenna	7	80
Misc. Components	2.2	0
Totals	**228.1**	**2399**

8.2: Solid State Recorders

There are two main SSRs used on the spacecraft; one for science data storage and one for command and housekeeping storage. A third SSR is included for redundancy and will have the capability to store both science and command data. Science data will be transmitted to Earth continuously throughout the mission and data that has been transmitted will no longer be stored if it is not critical to the mission. Command history is stored as well as future commands, recurring commands, and housekeeping data that has not yet been transmitted to Earth. Mass and power requirements of the SSRs are shown in Table 8.1.

Chapter 9: Communications

The telecommunications subsystem employs a primary antenna (HGA1) operating on Ka-band, a secondary antenna (HGA2) with the capability to operate on X- or Ka-bands, an emergency antenna (LGA) operating on X-band, and a proximity signal transmitter (UHF Antenna) for communication with probes and during EVAs. The primary and secondary antennas are parabolic dishes while the emergency antenna is a horn design. The system uses Quadrature Phase Shift Keying (QPSK) modulation. This configuration is chosen because it allows for communications over X- and Ka-band frequencies with either the 34-m or 70-m Deep Space Network (DSN) antennas. This system provides full redundancy and allows the spacecraft to utilize one or two high gain antennas at a time for data transmission, increasing its maximum transmitting data rate and also providing the option of arraying the antennas to receive high data rate signals from Earth. Each high gain antenna is sized to act as the primary antenna if the other antenna fails during the mission.

The spacecraft will be behind Mars for about one-third of every orbit and will lose line-of-sight with the DSN on Earth, causing a break in communications. This can be combatted by using science and communication satellites already orbiting Mars as a node to transmit critical housekeeping data to Earth during this blackout period. The remaining orbit time will allow a direct connection between the spacecraft and Earth. A constant link with a DSN antenna is necessary for the mission because it will be necessary for the crew to have the ability to communicate with Earth at any given time. The minimum amount of data will be transmitted during transit to and from Mars. The maximum amount of data will be transmitted once a day for 2-6 hours (depending on file sizes of important science data) during the stay at Mars. Table 9.1 shows a breakdown of data rates to show the range of transmitting rates expected for the mission. Figure 9.1 shows transmitting data rates for various cases as shown. The data shows that the spacecraft's minimum and maximum transmitting rates exceed the required and estimated mission rates. The spacecraft is designed to be scalable for deep space flight beyond Mars, thus data rates at Europa for the given telecommunications system are

Table 9.1: Data Rate Breakdown

Transmitted Data	kbps
Monitoring	1
TTC	2
Voice	4.8
Engineering	5
Minimum	12.8
General	200
Images	1000
Science	3000
Total	4212.8
HD Video	12000

shown in Figure 9.1 to advertise the capability of the subsystem. Video streaming data rates can be accommodated with the minimum required data rate.

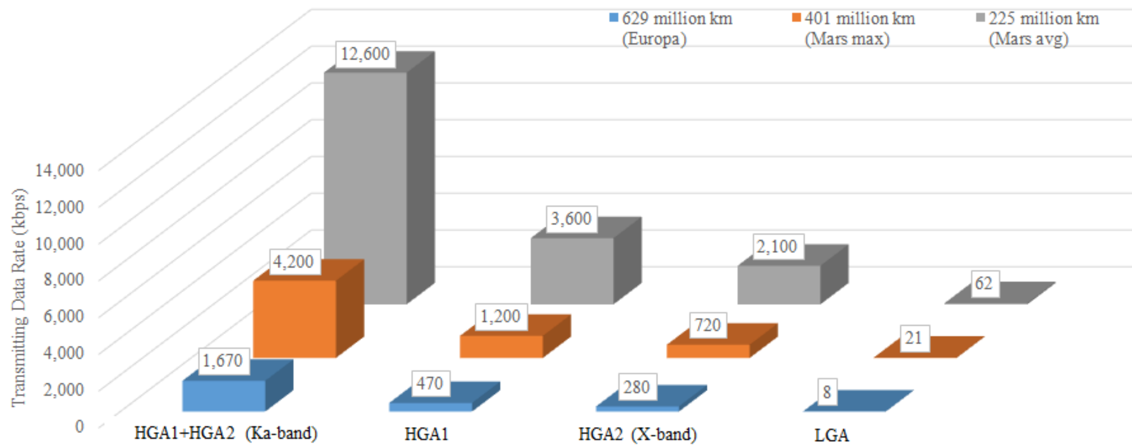


Figure 9.1: Transmitting Data Rates

Figure 9.2 shows a block diagram of the telecommunications subsystem. Individual units in the system are explained and described in the following subsections. Table 9.2 shows a link budget for each antenna at the specified frequencies. Table 9.3 shows the mass and power requirements for the subsystem.

Table 9.2: Link Budget

Item	Units	HGA1	HGA2 (X)	HGA2 (Ka)	LGA
Diameter	m	2.0	3.0	3.0	3.0
Frequency	Hz	32.0	8.42	32.0	8.42
Transmitter Power	dBW	30.0	30.0	30.0	23.0
Transmitting Antenna Gain (net)	dBi	53.4	45.6	47.8	57.2
Equiv. Isotropic Radiated Power	dBW	82.9	75.1	78.9	58.1
Space Loss	dB	-294.6	-283	-294.6	-283
Receiving Antenna Gain (net)	dBi	77.9	71.2	77.9	71.2
Eb/N0	dB	12.4	12.3	12.4	12.3
Required Eb/N0	dB	9.6			
Misc. Losses	dB	-2.5			
Margin	dB	0.3	0.2	0.3	0.2

9.1: High Gain Antenna

Two High Gain Antenna (HGA) are used to provide a fully redundant primary communications system. The primary antenna is HGA1 and is only Ka-band compatible to reduce the size and weight of the antenna while still transmitting the maximum data rate for the same power input. Ka-band frequencies provide higher data transmission rates than X- or S-band frequencies

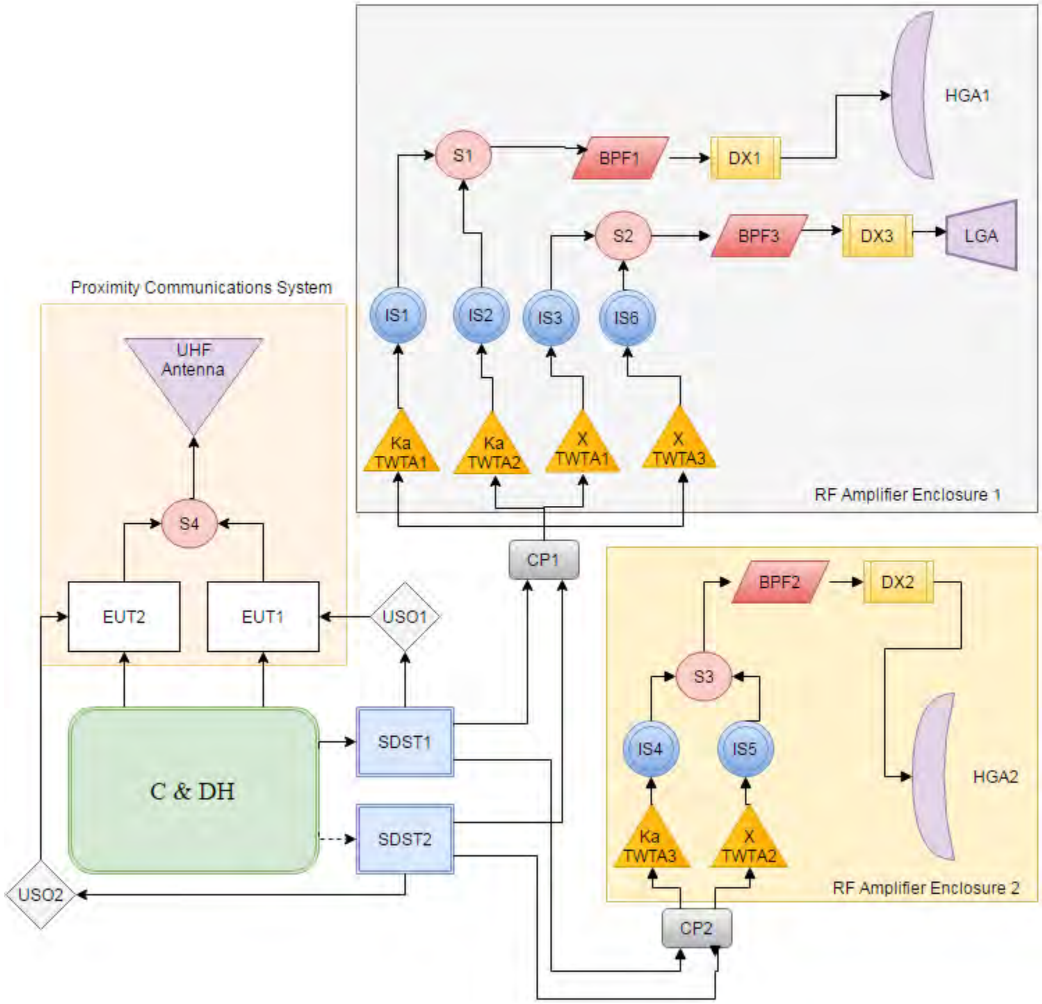


Figure 9.2: Telecommunications Subsystem Block Diagram

and have less traffic. Currently, only the 34 meter antennas of the DSN are capable of Ka-band frequency reception; however, the DSN and other communication networks are being developed to accommodate Ka-band communications for future missions.[25] The HGA1 system consists of a disc-on-rod feed (Ka-band compatible), an ellipsoidal subreflector, and a two-meter offset parabolic main reflector. HGA1 is mounted on a one-axis gimbal mechanism on the end of a telescoping boom (shown in Figure 9.3) to allow freedom of movement for the antenna to point at Earth during any spacecraft position or maneuver. The boom is attached to a two-axis gimbal mechanism.

The secondary antenna (HGA2, shown in Figure 9.4) is X- and Ka-band compatible for redundancy and to ensure compatibility with signal receivers on Earth. It is designed to act as a full replacement for HGA1 for critical data transmission if needed. HGA2 can also supplement HGA1

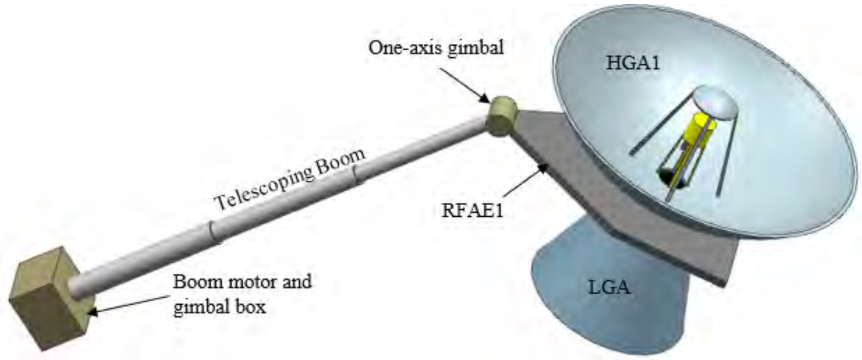


Figure 9.3: HGA1 Assembly

and increase the overall maximum transmitting data rate (critical and science data). The HGA2 system consists of a corrugated horn feed (X-band compatible), a disc-on-rod feed, an ellipsoidal subreflector, and a three-meter offset parabolic main reflector. The system is mounted on a two-axis gimbal mechanism attached to the torpor habitat. This location allows for full communication capabilities in the event that the main antenna boom experiences a critical failure and renders HGA1 and Low Gain Antenna (LGA) systems unusable.

9.2: Low Gain Antenna

A LGA is mounted on the primary HGA boom system and moves with the HGA system. The LGA is a one-meter diameter horn design and is attached behind the primary HGA dish. The purpose of the LGA is to provide low rate communication during emergencies and special events (launch, MOI, EOI). The LGA has a low data rate capability because it is small in diameter and focuses the beam more broadly than the HGAs. It transmits at X-band frequencies to ensure compatibility with any DSN antenna.

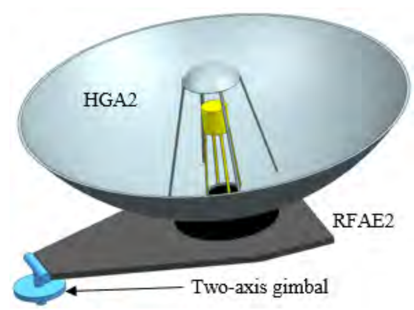


Figure 9.4: HGA2 Assembly

9.3: Transponders

The Small Deep-Space Transponder (SDST) performs functions such as tracking the uplink carrier, command demodulation, convolutional coding, producing subcarrier frequencies, modulating telemetry, X-band and Ka-band excitation, and generating differential one-way ranging tones. Two identical SDSTs are used on the spacecraft for redundancy, with only one transponder powered on at a time. The transponder is composed of a digital processing module, a downconverter module, a power module, and an exciter module. It also contains an internal low-pass filter to reduce the amplitude of high-frequency components to avoid interference with other functions. Both transponders are each connected to an Ultra-Stable Oscillator (USO) to use as a clock and frequency reference.

9.4: Radio Frequency Amplifier Enclosures

There are two Radio Frequency Amplifier Enclosure (RFAE)s, one attached to each HGA system. These enclosures connect the gimbal mechanism to the parabolic antenna. RFAE1 is attached to HGA1 and contains:

- Four Traveling Wave Tube Amplifiers (TWTA)s, two for Ka-band and two for X-band frequencies to amplify the transmitted RF signals and provide adequate redundancy
- Four isolators to protect the TWTAs in the event of a temporary short in the transmit path to the antenna
- Switches S1 and S2 to toggle between redundant Ka-band TWTAs for the HGA1 and X-band TWTAs for the LGA
- Two Band Pass Filter (BPF)s to filter amplifier interference before the signal is transmitted
- Two diplexers (DX) to implement frequency-domain multiplexing for the transmitted signal

RFAE2 contains the same components as RFAE1 excluding all the components for the and excluding many redundancies since the entire system encapsulated in RFAE2 is a redundant system. Note that the couplers and diplexers do not require redundancy due to them being passive devices with no moving parts nor electronics and thus have a very low rate of failure.

9.5: UHF Communications Assembly

A UHF antenna assembly is used for proximity communications during EVAs and to communicate with probes and data acquisition satellites when in Mars orbit. A UHF system is used to reduce the required antenna size for proximity communications. The system consists of redundant Electra UHF transceivers, a switch to toggle between the transceivers, and the UHF antenna. The whole UHF assembly is fixed on the exterior of the spacecraft for signal reception and transmission.

The Electra UHF Transceiver (EUT) assembly consists of five modules: a half-duplex overlay receiver filter and UHF diplexer, a filtering and switch unit, a UHF radio frequency module, a baseband processor module, and a power supply module. The tasks of the assembly consist of receiving and transmitting signals, filtering signals, and processing the signals.

Chapter 10: Attitude Determination and Control System

The ADCS of the spacecraft will provide all control and attitude determination needed for the duration of the mission.

10.1: System-Level Requirements

The system level requirements of the ADCS system include handling all control requirements from external disturbance torques and having the ability to point the spacecraft with enough accuracy for the telecommunications system in case of emergency (0.16^o). It should be noted that the antennas of the telecommunication system are independently gimbaled, so in Normal Mode, the ADCS system is not required to orient the spacecraft for these antennas.

The spacecraft will also carry multiple payloads that will observe Mars, Phobos, and Deimos such as the HiRISE camera and JunoCam. These payloads will have their pointing dependent instruments independently gimbaled to decrease complexity of the ADCS system and overall spacecraft pointing accuracy.

The power system is mission critical, but given the amount of redundancy in the power subsystem, the solar panel pointing accuracy is not considered when looking at the system-level requirements.

10.2: Spacecraft Control Selection

According to the system-level requirements, only the antenna pointing accuracy of 0.16^o on all three axes during the Emergency Mode needs to be considered when selecting a spacecraft attitude control method. Therefore, the spacecraft will use a three-axis control technique. Passive control techniques do not provide high enough pointing accuracies and although spin control techniques provide a high enough pointing accuracy, they cannot do it for all three axes, only two.[11]

The three-axis control techniques considered are bias momentum, thrusters only zero momentum, reaction wheel zero momentum, and CMG zero momentum. Since the telecommunications antenna pointing accuracy requires movement about all three axes, and bias momentum constrains the yaw maneuver, it was eliminated as a potential control technique.[11]

The ADCS system is a mission critical system, so proper redundancies are needed. It was determined that the system will either use at least two zero momentum techniques (either reaction wheels and thrusters, or control moment gyros and thrusters) or an overly redundant thruster or actuator system. This brings the overall pointing capability of the spacecraft to 0.001^o to 0.1^o and creates no constraints on for attitude maneuverability. The lifetime limit of the ADCS system will be determined by the thruster propellant (if used), sensor life, or wheel bearing life (if used).[11]

10.3: External Disturbance Environment

External disturbance torques were assumed to be the ruling torques with internal disturbance torques being negligible because they're easier to control. Internal disturbance torques the spacecraft will experience include: uncertainty in center of gravity, thruster misalignment, thruster output mismatch, rotary machinery, liquid sloshing, and thermal shocks with flexible appendages.[11]

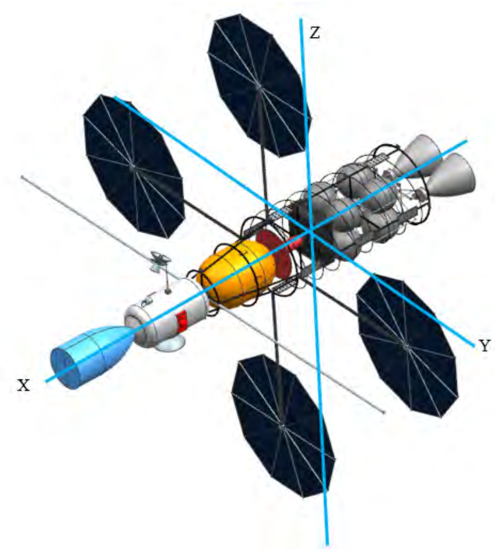


Figure 10.1: Spacecraft Configuration and Axes Orientation

The total torque was calculated using the worst-case external disturbance scenario for the spacecraft. To determine this external disturbance, solar, gravity gradient, aerodynamic drag, and magnetic torque were calculated at Earth, Mars, and Venus (due to the potential fly-by for future flights for scalability purposes). To simplify calculations, the spacecraft was assumed to be in a circular orbit around Earth, Mars, and Venus at 300 km, 7,660 km, and 135 km (Venus Express) respectively. At these altitudes and orbits, the spacecraft velocity at Earth, Mars, and Venus is 7.73 km/s, 1.97 km/s, 7.25 km/s respectively. Since there is a potential Venus fly by, the spacecraft will experience at most, one half orbit around the planet.

The assembly of reaction wheels or CMGs was assumed to be placed external of habitable modules so the assembly would contribute to the overall length and mass. For the initial assumption, L3 Double-Gimbal CMGs in a pyramidal orientation were chosen as the AA because they are currently used on the ISS[26] along with its thruster based Reaction Control System. The CMGs have a total length of 1.37 meters and mass of 272 kg each.[27] Since the pyramidal orientation is being used, four CMGs will be used in the initial AA assumption.

The total length of the AA was assumed to be 2.74 meters and have a mass of 1,088 kg. Therefore, the total mass of the spacecraft is 158,000 kg, the total length is 40 m, and the location of the solar panels from the end raptor engines nozzles is 24.7 m. Figure 10.1 shows the configuration of the spacecraft and how its axes were named.

Table 10.1: Planetary Worst Case External Disturbance Torques

External Disturbance Torque on Spacecraft	Planet	Resulting Torque (N-m)
Solar	Earth	0.095
	Mars	0.041
	Venus	0.182
Gravity Gradient	Earth	0.643
	Mars	0.015
	Venus	0.659
Aerodynamic Drag	Earth	6.33×10^{-6}
	Mars	0
	Venus	1.45×10^{-2}
Magnetic	Earth	5.40×10^{-3}
	Mars	2.90×10^{-7}
	Venus	5.62×10^{-6}
Total	Earth	0.740
	Mars	0.057
	Venus	0.855

10.3.1: Worst Case External Disturbance Torques

The worst case situation for solar torque will be when the spacecraft is oriented in its xz-plane towards the Sun with its solar panels facing the sun. The worst case situation for magnetic torque will be if the spacecraft has the maximum possible residual dipole, 20 A-m^2,[28] plus an order of magnitude. Table 10.1 shows the four different external disturbance torques and total torques at Earth, Mars, and Venus.

10.3.2: Momentum Sizing

The orbit period at Earth, Mars, and Venus was calculated using the orbit assumptions seen in Section 10.3 of this report. The orbital period of the spacecraft at Earth, Mars, and Venus (half orbit) are 5.43 E+03 seconds, 3.53 E+03 seconds, and 2.68 E+03 seconds respectively. From this information and the total torque at Earth and Mars, the total momentum buildup during one orbit due to external disturbance torques is reflected in Table 10.1.

Table 10.2: Spacecraft Total Momentum Buildup During One Orbit

Planet	Total Momentum Buildup During One Orbit (N-m-s)
Earth	4038
Mars	2000
Venus	2294

Since the Earth total momentum buildup during one orbit is larger than the Mars momentum, it was determined to be the ruling situation. From this point on, all sizing will be based off of the 4038 N-m-s per axis seen in Table 10.2.

10.3.3: Thruster Sizing

Thrusters will potentially be used as a redundancy to the AA system to increase safety, redundancy, and to de-saturate the AA if reaction wheels are used. There would be four thrusters located on each side of the spacecraft, giving 24 thrusters in total. Using the total torque at Earth (ruling situation) seen in Table 10.3, the force required per thruster for a fully redundant system is 0.0052 N. A trade study was performed on a propellant combination of liquid oxygen/methane (propulsion system and methane made on the spacecraft), nitrogen tetroxide/UDMH (ISS), and Dinitrogen Tetroxide/MMH (Space Shuttle). The results can be seen in Table 10.4.

Table 10.3: Thruster Propellant Trade Study

Propellant Combo	Total Mass Needed for Mission Duration (kg)	Total Volume Needed for Mission Duration (m^3)
Oxygen/Methane	7,854	10.05
Nitrogen Tetroxide/UDMH	8,362	7.49
Dinitrogen Tetroxide/MMH	8,362	7.47

Based on the trade study, an oxygen and methane propellant combo would be used as the combination of propellant for the thruster system because the spacecraft is mass limited, not volume limited. However, possible thruster implementation will be determined when choosing the actuator for the AA.

Table 10.4: Preliminary Reaction Wheel vs. Control Moment Gyroscope Trade Study[11]

Actuator	Accuracy (°)	Typical Performance Ranges (N-m-s)	Attitude Maneuverability	Weight Consideration	Complexity Consideration
Reaction Wheel	±0.001° to ±1°	0.4 to 400	No Constraints	Normal Weight	Normal Complexity
Control Moment Gyroscope	±0.001° to ±1°	0.5 to 5000	No Constraints High Rates Possible	Increased Weight	Increased Complexity

10.4: Hardware Selection

Hardware selection for the spacecraft will be based off of the Emergency Mode telecommunications pointing accuracy (0.16^o), ability for redundancy, safety, complexity, mass, and power.

10.4.1: Actuator Assembly Trade Study

Preliminary considerations of either a thruster/reaction wheel or a thruster/CMG combination yielded that the thruster/CMG combo should be used because of the momentum buildup.[11] Reaction wheels are usually only capable of producing 0.4 to 400 N-m-s of momentum buildup per wheel.[11] Assuming a reaction wheel was considered that was capable of producing 400 N-m-s, 11 reactions wheels would be needed per axis (and 33 wheels per assembly without any axis redundancy) to be able to handle the external disturbance torques properly. CMGs in a pyramidal configuration have built in redundancy, but CMGs add weight and complexity to an ADCS system. However, a wheel assembly with 33 wheels and no axis redundancy drastically increases weight, complexity, and lowers redundancy and safety. Regardless, a trade study was performed to compare different reaction wheels and CMGs. A preliminary trade study of reaction wheels vs. CMGs can be seen in Table 10.4 before any hard calculations were performed.

Table 10.4 shows that reaction wheels are the preferred wheels when performing preliminary data analysis. However, a more in-depth trade study was performed that took into consideration the total number of components needed and therefore total weight, power, and complexity.

The initial assumption that the spacecraft will use the same AA as the International Space Station came fairly close to the required momentum buildup. The spacecraft needs to be able to handle 4,038 N-m-s of momentum buildup during one orbit, and the L3 CMGs can handle 4,760 N-m-s which also leaves about a 10% margin. The amount of components was based off of a three-axis configuration sized to the total momentum buildup for reaction wheels and a pyramidal configuration (four sides) sized to the total momentum buildup. The trade study can be seen in Table 10.5.

Based on the trade study, the L3 Reaction Wheel Assembly is the best option for reaction wheels and the L3 Double Gimbal CMG is the best option for CMGs. It should be noted that the author was limited to published reaction wheel and CMG datasheets and there may be better options available for use, but the data is not available to the public.

Table 10.5: Reaction Wheel vs. Control Moment Gyroscope Trade Study[12–16]

Type of Wheel	Actuator	Angular Momentum (N-m-s)	Number of Components	Mass (kg)	Power (W)	Volume (m^3)
Reaction Wheel	Honeywell HR 0610	4038	1010	5000	81,000	-
	L3 Reactional Wheel Assembly	4038	33	460	5,000	0.19
	Blue Canyon RW8	4038	1514	5500	121,00	3.83
Control Moment Gyro	L3 Double Gimbal CMG	4760	4	1088	2400	2.23
	Honeywell M50	4038	215	6000	24,000	2.46

The L3 CMGs have the lower number of components needed to satisfy the total momentum buildup and therefore has lower complexity compared to the L3 reaction wheels. The L3 reaction wheels have the lowest mass of the two, but the extra complexity is not worth the risk. Pyramidal CMGs also have an extra redundancy compared to three-axis reactions wheels, so the L3 Double Gimbal CMG was the chosen actuator for the AA.

Figure 10.2: AA Configuration with Five CMGs

Based on the total mass of the individual CMGs (272 kg), an extra CMG was decided to be added to the AA instead of using a fully redundant thruster system because of the mass the thruster system adds (7,854 kg). An example of the AA configuration can be seen in Figure 10.2.

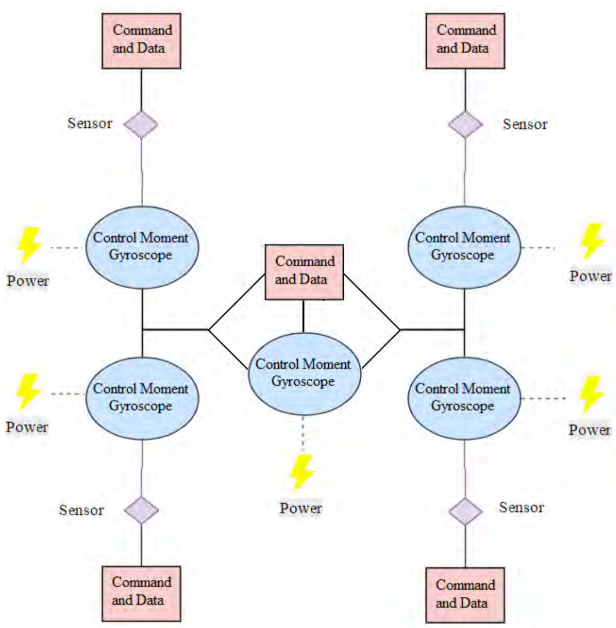
10.4.2: Sensors

ADCS will utilize three sun sensors, two inertial measurement units, and two star trackers to help determine the spacecraft attitude and control data. Multiples of each type of sensor are for system redundancy.

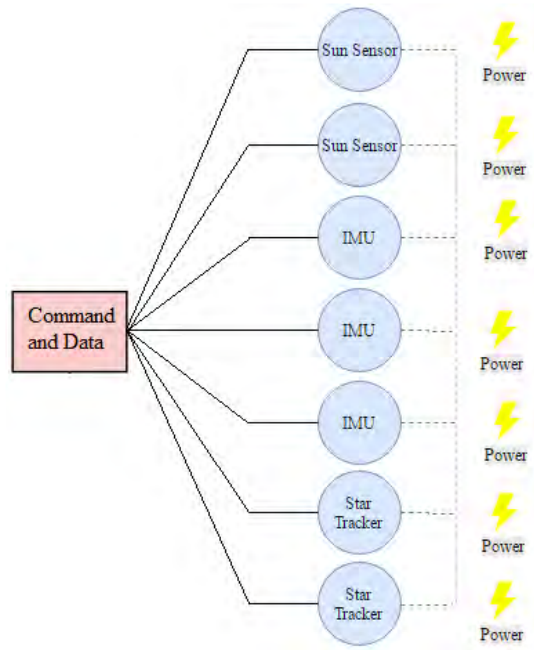
Due to the pointing accuracy requirement (0.16^o) of the telecommunications system in Emergency Mode, the Honeywell HG9900 was chosen for the spacecraft's inertial measurement unit because the IMU is accurate to 0.003^o/hour.[29]

10.4.3: ADCS Hardware Block Diagrams

The diagrams seen in Figure 10.3 demonstrate a high level overview of how ADCS interacts with itself, the power subsystem, and the command and data handling subsystem.



(a) CMG Assembly



(b) Sensors

Figure 10.3: ADCS Block Diagrams

10.5: Actuator Assembly Placement

The AA placement was determined using information about the external spacecraft components and the calculated information from the AA in Section 10.4.1.

There are two extreme mass stages that the spacecraft can be in: full tanks and empty tanks. Table 10.6 shows the mass and length of each external spacecraft component considered when calculating the spacecraft center of gravity. The order component appear in this table is the order the spacecraft is configured from Raptor engine end to Torpor Module front.

Using the information above, the spacecraft center of gravity was calculated at the two mass extremes and averaged to make an educated assumption about where the AA should be placed relative to the end of the Raptor engine nozzles. Due to the length of external spacecraft components, the AA centroid was placed as close as reasonably possible to the average spacecraft center of gravity. The information is reflected in Table 10.7.

Despite placing the AA at a reasonable location, the spacecraft will still experience unbalanced torques when firing coupled thrusters, and unwanted torques when performing translational movements. This is assumed to be negligible.

Table 10.6: External Spacecraft Components

Configuration Component	Total Length (m)	Total Mass (kg)
Raptor	4.97	1.00E+04
8 Propellant Tanks (Full/Empty)	6.66	1.28E+05 (Full)/ 3,740 (Empty)
Nuclear Reactor	1.4	1.36E+03
Actuator Assembly	4.62	1.54E+03
Inflatable	7.37	5.77E+03
Structure	20.05	3.74E+03
Topor	7.5	9.44E+03
Solar Panels	7.00	3.00E+02
Solar Panel Structure Connection (Mass Included in Structure)	7.00	-
Total	32.5	1.53E+05

Table 10.7: AA Placement and Spacecraft Center of Gravity

	Full Tanks	Empty Tanks
Spacecraft Center of Gravity (m)	10.3	13.8
Average Center of Gravity (m)	12.0	
Actuator Assembly Center of Gravity (m)	12.3	

Chapter 11: Thermal

Solar radiation, infrared radiation from planets, internal power systems, and the crew themselves are all generating heat that needs to be removed from the spacecraft. Additionally, propellants like methane and liquid oxygen require low temperatures to maintain their liquid state, so cooling systems are needed. This constitutes the need for a thermal system. This is split into passive and active systems.

11.1: Passive Systems

The passive systems transfer heat as a result of their physical properties without input from electrically powered systems. This includes radiative surfaces to expel heat from the spacecraft in form of external panels and dedicated radiators, insulation, and heat pipes. Heat pipes are designed to transfer heat to cooler areas through an internal evaporation-condensation cycle, allowing heat to be moved without pumps.

Heat will be radiated away in dedicated radiators, as well as panels placed on the truss structure. The radiators will be placed radially around the truss, and fold flat against the surface rather than holding a scissor configuration like seen on the ISS. This creates a louvering effect, where closing and deploying the radiators can be used to regulate the flux from the panels beneath them. This allows the total radiative area of the spacecraft to be adjusted by deploying and retracting the radiators without the added weight of traditional louvers. Aluminized FEP Teflon was selected as our outermost radiative surface for our radiators and as surface for our truss panels, due to its low absorptivity of 0.163 and its emissivity of 0.8.[28] The panels also include Kapton-Mylar insulation. This was selected for its widespread use on previous spacecraft, low weight, and thermal properties.[28] Figure 11.1 shows the layers that make up these truss panels.

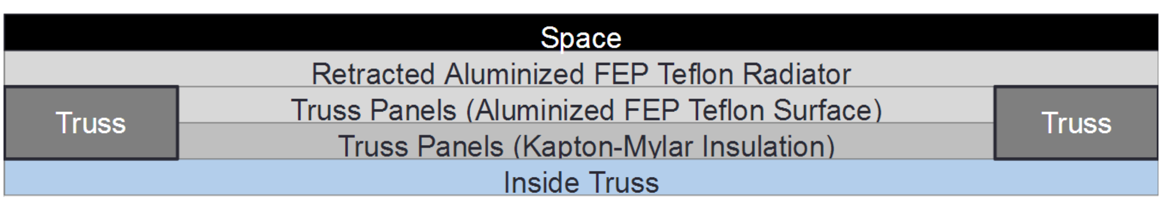


Figure 11.1: Thermal Truss Panel Laying

Heat pipes will be used to move heat from the side of the spacecraft exposed to the sun to the colder, shadowed side of the spacecraft, providing an even temperature across the surface.

Heat pipes will also be used to passively cool the nuclear power system, thermally connecting the Kilopower reactor to its own set of radiators.

11.2: Active Systems

The active system uses two pumped ammonia loops. Pumped loops were selected for their ability to precisely regulate the heat of each component of the thermal loop separately. Ammonia was selected for its thermal properties as well as its proven use in manned stations like the ISS. The system on the MOONS vehicle uses two separate evaporation-condensation loops, one regulating the cooling of the propellant tanks and the other carrying heat from the habitable environment and power systems to the spacecraft exterior and radiators. A heat exchanger in the cooling loop carries heat away from the propellant tanks and into the habitable environment. A schematic of this system can be seen in Figure 11.2.

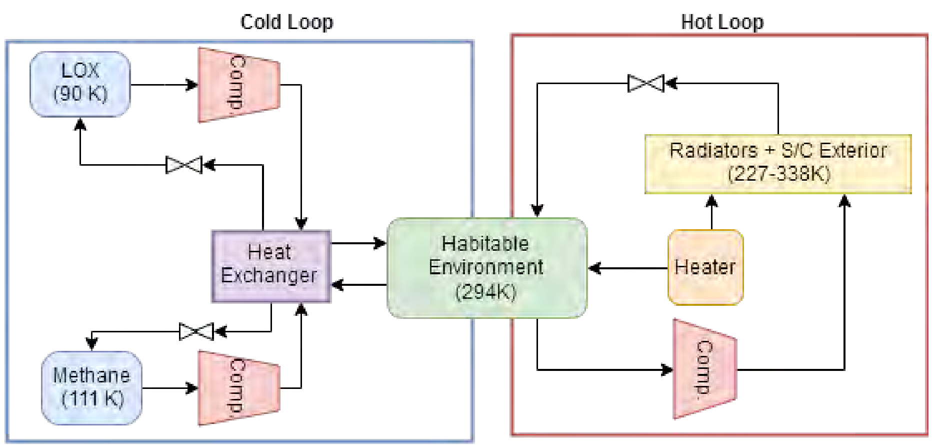


Figure 11.2: Active Thermal Loops

Both the active and passive systems are visualized by the heat flow diagram shown in Figure 11.3.

11.3: Sizing

The thermal system was sized to the following requirements.

- External temperature range of structure within $\pm 40^o$C[28]
- Radiation capability must be greater than the power system output, power required for propellant cooling, absorbed solar and planetary radiation, and crew energy output (864 kW)
- Worst Case Hot: Earth day-pass, Direct Solar = 1370 W/m^2, Planetary IR[30] = 298 W/m^2

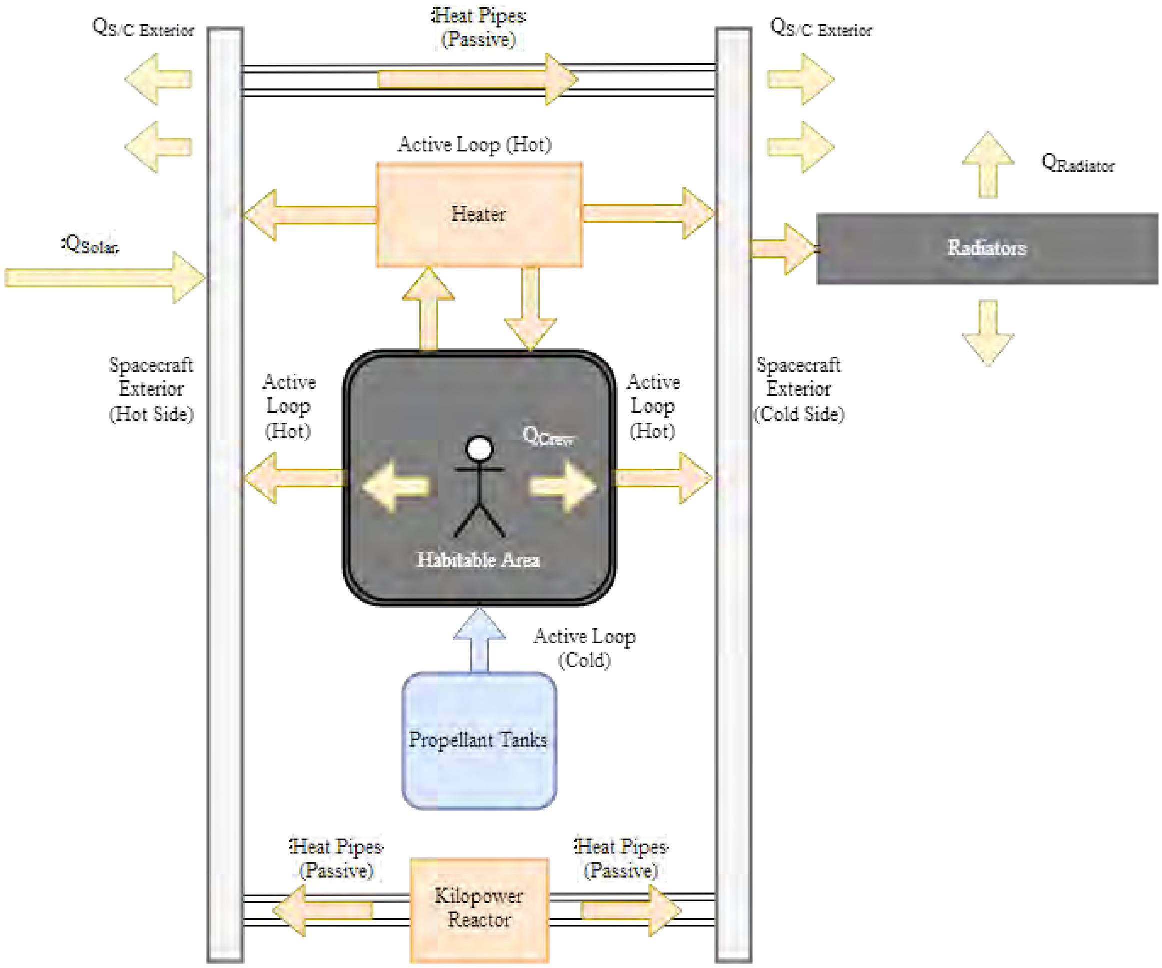


Figure 11.3: Heat Flow Diagram

- Worst Case Cold: Mars night-pass, Planetary IR[30] = 30 W/m^2

The thermal sizing was completed using a simulation that performed Stefan-Boltzmann analysis described in Section 11.5 of Space Mission Analysis and Design.[11] This generates the heat rejection curves of the spacecraft, and sizes the heater and radiators to generate and reject enough energy to stay within the temperature bounds. Mass is estimated in accordance with the Spacecraft Design textbook.[28] The results are shown in Figure 11.4 and Table 11.1.

Table 11.1: Thermal System Characteristics

Radiator Area	42 m^2 (89% of ISS Radiator Size)
Heater	115 kW
Est. Mass	1500 kg

11.4: Design, Risk Assessment, and Risk Mitigation

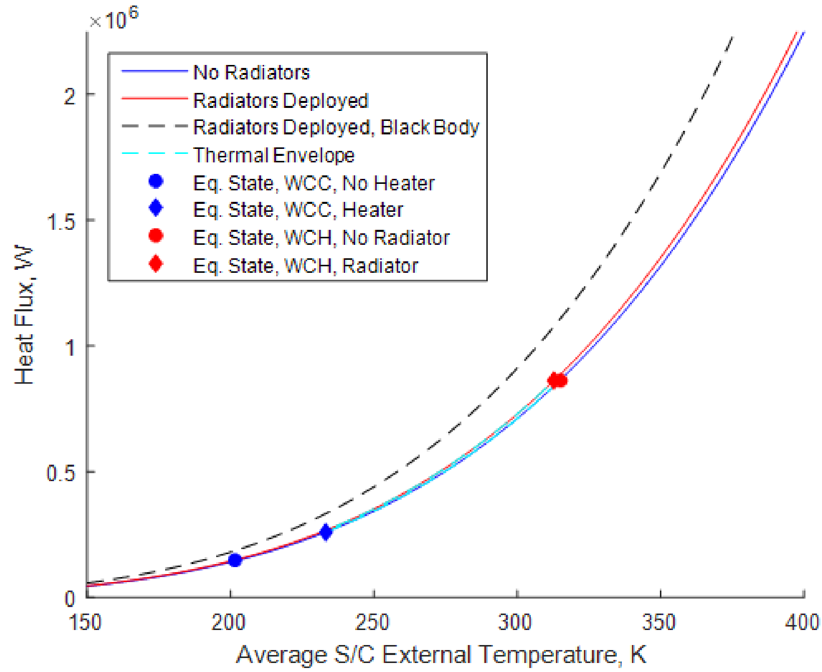


Figure 11.4: MOONS Vehicle Thermal Radiation Curves

This thermal system is inherently low risk by only utilizing materials and heat transfer techniques that have been used in manned space applications in the past, giving our thermal system a TRL of 9. The heat will be expelled through ten two-sided radiators, each 1 m by 2.3 m in size. This provides an area of 46 m^2, allowing a redundancy in case a radiator fails to deploy during a worst case hot scenario. Using multiple smaller radiators allows us to scale our system to meet higher thermal requirements as it grows in size in future missions, so that the thermal capacity of the spacecraft can be increased by simply adding more radiators.

Additionally, the active system will be doubly redundant, so that if there are any pump or valve failures within the system, all loops will still be operational. Finally, the worst case hot scenario is currently sized for the broad side of the spacecraft to be facing the sun while in Earth orbit. The MOONS spacecraft will keep its engine facing towards the Sun at the most thermally intensive parts of the mission, allowing the thermal conditions of the spacecraft to be regulated by the attitude of the spacecraft as well. The attitude control system is sized large enough to consistently provide this level of pointing accuracy throughout the mission, meaning the spacecraft would only be in the worst-case hot scenario when orienting itself for a burn. This provides another level of redundancy for the thermal systems.

Chapter 12: Power

With multiple modules on the spacecraft, a substantial amount of power is needed to fully accommodate all systems. The power is routed from two sources to seven subsystems: Thermal, ADCS, Power, C&DH, Communications, Propulsion, and Mechanisms. The total power requirements of these subsystems are derived from the procedures laid out on Pg. 33 of Brown's Spacecraft Design book.[28] The values are primarily a function of payload power requirements. Early in the design process, the payload power was found to be 23 kW. This value comes from an interface document populated by subsystem designers. Using the estimated power consumption equation of the Other category in Table 9 of Brown's Spacecraft Design book,[28] the full operation power consumption was found. The subsystems power was then found by taking the difference of power consumption and payload power. Lastly, a contingency of 10% was added to find the total power required. These values are summed in the Table 12.1.

Table 12.1: Power Budget

Estimated Power Consumption (W)	27,203
Subsystem Power Allocation (W)	6,439
Contingency (W)	1,360
Total(W)	35,002

The minimum power is what the MOONS vehicle can function from in case of emergency. It is what the crew would need to return to Earth at any point in the mission. This is also assuming the batteries are not functional. The chosen essential components were: Thermal Control, Basic Habitat Requirements/ECLSS, Secondary Comm Array, Propulsion, Attitude Determination, and Active CMGs. The combined power draw of these systems resulted in 23 kW. This is roughly half of the total power requirements.

The subsystems' power requirements are all found as a percentage of the subsystem power allocation amount. These percentage values are taken from Table 2.10 in Brown's text.[28] It is important to note that the subsystem fractions prescribed[28] are different than the chosen values. This is due to the MOONS vehicle having a different mission than those prescribed. The percentage values and power values are shown in the Table 12.2.

Table 12.2: Subsystem Power Allocation

Subsystems Power Requirements (W)		Fraction of Subsystem	Brown Subsystem Fractions
Thermal	1,046	.16	.33
Attitude	2,485	.39	.11
Power	128	.02	.02
CDS	74	.01	.15
Communications	2,414	.38	.30
Propulsion	128	.02	.04
Mechanisms	161	.03	.05

12.1: Solar Panels

Solar panels are often seen as the hallmark of a spacecraft and usually act as the primary source of power. For this spacecraft, the Orbital ATK Megaflex Solar Panels will be used. The Megaflex arrays leverage the legacy of the successful Ultraflex arrays, which have flown on the Cygnus service module. The Megaflex arrays are ultralight in their mass and have a high specific power. The array is stowed along the truss structure of the craft, and deploys circumferentially about a center point. This minimization of moving parts helps to ensure a successful deployment. The Megaflex arrays currently sit at a TRL of 5, but with some investment from the allotted five billion dollar budget, it is certain that the TRL can be raised quickly since the groundwork has already been taken care of.

For the MOONS mission, four Megaflex Panels were chosen. Each panel has a diameter of 7 m and outputs 10 kW. Combined, the panels all produce 40 kW and have a nonstructural weight of 300 kg. It is important to note the primary reason for choosing four panels is safety. The chances of more than one panel breaking away from the craft is minimal.

Table 12.3: Four Wing Solar Panel Characteristics

Solar Panel Characteristics (4 Wings)	
Array Power (kW)	40
Array Mass (kg, Non Structural)	300
Wing Diameter (m)	7

12.2: Nuclear Reactor

While many applications have flown in space that have used Radioisotope Thermoelectric Generators (RTG), there are few that have flown with nuclear reactors. The Soviet Union/Russian

Federation have flown nearly 40 nuclear reactors into space, while the United States has only sent one.[31] The Los Alamos National Laboratory is developing a new system to provide power in deep space travel. The Kilopower project,which can be seen in Figure 12.1, is aimed at developing a kilowatt class nuclear reactor for space travel.[32] The system uses thermal energy generated by the nuclear reactor and the cold of space to drive sterling engines. These engines provide power to the spacecraft. A nuclear reactor was chosen for the simple reason of distance from the sun. As the craft moves further from the sun, the solar panels will provide less energy. Figure 12.2 shows where the MOONS power and duration lie in a popular power selection graph.

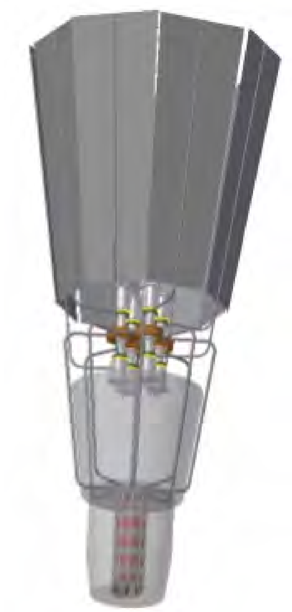


Figure 12.1: Kilopower Concept

The red identifier shows a rough estimation of where the mission will operate. It becomes clear that a combination of nuclear and solar power will be needed to accomplish the mission. This is one of two driving factors in the selection of the Kilopower system. The other driving factor is the inherently designed safety of the system. Since Kilopower runs on simple parts, only one control rod made of boron-carbon is needed. This rod allows for the system to be started whenever the mission requires it. The rod is also automated to allow for an added degree of safety. If the system starts to reach critical levels the rod is automatically inserted into the reactor by a triply redundant computer system.

Table 12.4: Kilopower Characteristics

Nuclear Reactor Characteristics (1 Reactor)	
Power (kWe)	10
Mass (kg)	1544
Diameter (m)	1.5
Length (m)	3.3
Volume (m^3)	2.0

For this mission a single reactor sized to one solar panel outputting 10 kW has been selected. That means one 10 kW reactor will make the journey to Mars. In case of emergencies, such as destruction of the solar panels, the nuclear reactor will supply the needed power of 10 kW. The sizing of the nuclear reactor was based on the idea of offering redundancy or providing purely minimal power. If there was one reactor sized to output what one solar panel outputs, it is sized for redundancy. If the reactor were sized to minimal power it would be much heavier, but would be used in cases of emergency. The choice was made to size the reactor for one solar panel due to the unlikelihood that more than one panel will become nonfunctional. The characteristics of the nuclear reactor are defined in Table 12.4.

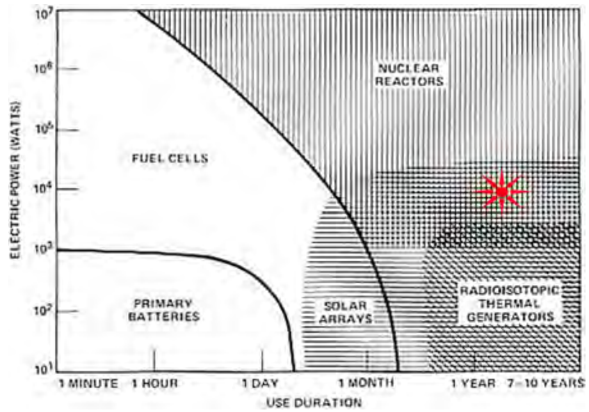


Figure 12.2: Power Selection Graph

12.3: Batteries

Since the spacecraft will be orbiting Mars and Earth, it will experience eclipse periods. In these periods the main power draw will come from the installed batteries. A trade study was done to determine the optimal battery to use on the MOONS craft. The breakdown can be seen in the Table 12.5.

Table 12.5: Battery Characteristics

Battery Type	Mass / 140 kW (kg)	Safety Index	TRL	Percent Total Mass of Craft (%)
Ni-Cad	4,202	High	9	2.7
Li-Ion	611	Low	9	.39
Li-Po	380	Med	8	.24
Solid State Li-Ion	205	High*	3*	.1

In this mission a combination of lithium ion and solid state lithium ion batteries will be installed. Saft[33] Li-Ion batteries will provide power during eclipse periods. The full eclipse load sits at 27 kW. The Li-Ion batteries are sized for the full eclipse load plus the thermal heater load. Lithium-ion batteries have significant weight savings, but do also house a flammable electrolyte solution. To mitigate this flammability problem, the batteries are stored in a reinforced box that has been put under vacuum. This helps to mitigate the probability of puncturing the batteries and causing catastrophic failure. The characteristics of all four battery types are summed in Appendix B.

Solid state lithium ion batteries are a new technology that mitigates the risks of lithium batteries. These batteries prevent dendrites from growing in the electrolyte solution by getting rid of the solution entirely. The specific energy is stated to be triple that of lithium ion at around 735 W-h/kg.[34] They also are one-fifth the cost of typical Li-Ion batteries, along with a highly extended cycle life of nearly 1,200 cycles. An investment will be made to bring the current TRL of 3 to a much more acceptable level. The MOONS mission will act as a heritage flight for a set of 25 kW solid state lithium ion batteries. These batteries have been sized to meet minimum power requirement.

12.4: Power Block Diagram

A block diagram of the power system is detailed below. The diagram is split up into four sections. The first is the power sources, including the solar panels, nuclear reactor, and batteries. The solar panels take commands from the power management system to fold and unfold as needed. From here, the diagram runs into the power management system. This system collects all the power and routes it towards the power bus. The power management system is a duplex system. This stacked system is a backup management system that is idle until the main system cannot complete its task. If there is excess power it is routed to the dump which then routes it to the radiators. The radiators dump the excess power as heat. In the next section, the bus distributes the allotted power to each subsystem. The subsystems then run to the computer system. This health processor accurately measures the power draw of each system and reports back to the power management system. The power management system then adjusts the power it sends to the bus as needed. The computer system also constantly monitors the status of the power sources, primarily the nuclear reactor. If there is ever an emergency, the computer can shut down the reactor and relay the information to the power management system. Similar to the power management, the computer has a stacked duplex backup that sits idle until the main system cannot completes its task.

12.5: Design Down Selection

Since the power system is highly scalable, a trade study was done on the mass and volume requirements of the system with a variety of configurations. Those configurations are shown in Figure 12.4. It is important to note the energy produced across the top is for total solar panel

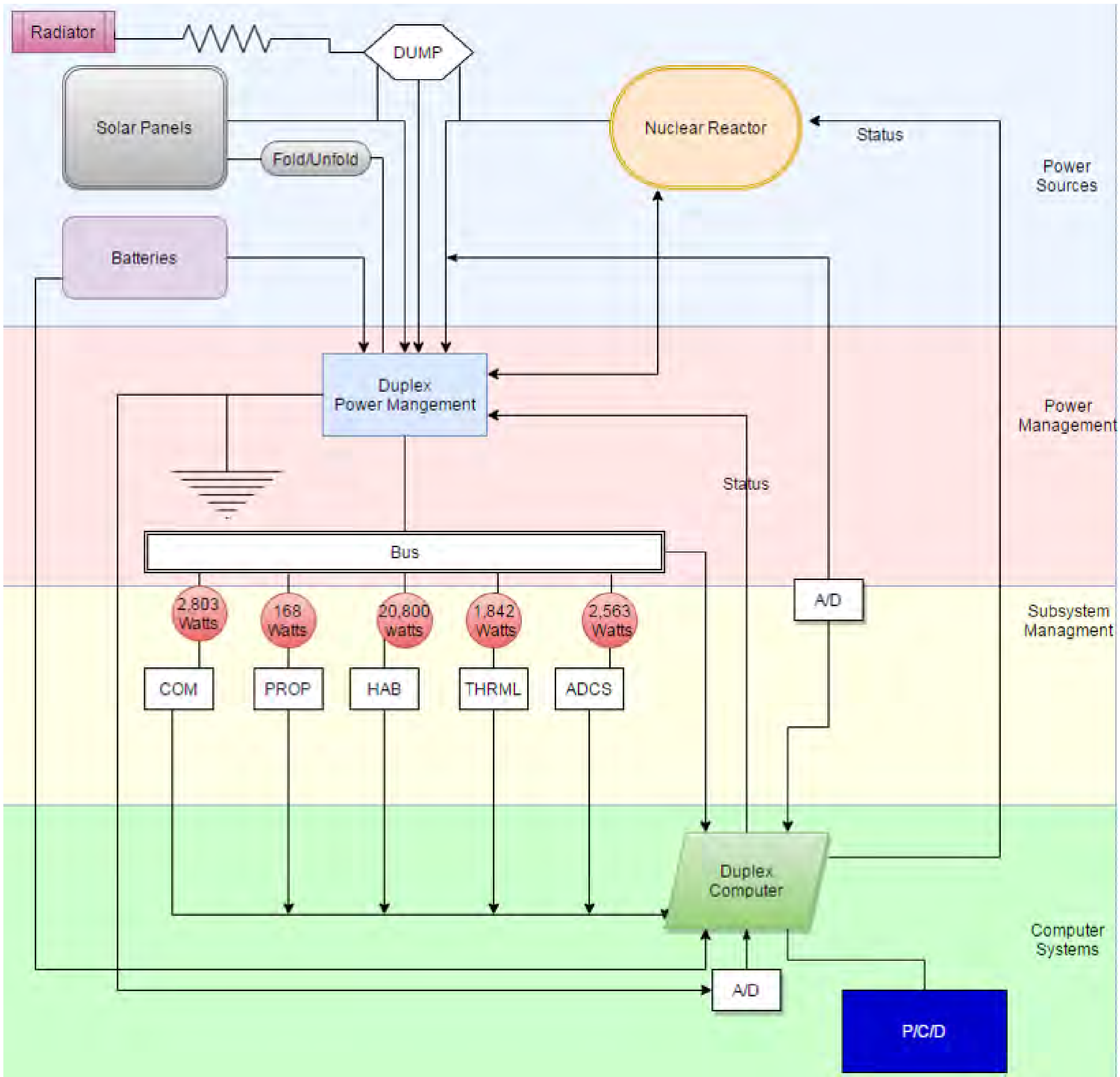


Figure 12.3: Power Block Diagram

output. The solar panel configuration is across the far left side. That is then followed by the amount of nuclear reactors and what the reactors are sized to. Each power column breaks down the mass of a configuration and the stowed volume. The power provided by the nuclear reactor is also noted.

The highlighted strip is the configuration that will be implanted on this first mission. Other configurations can be used in later missions, or as the design evolves. Four solar panels provide power with a sense of redundancy. Together the panels produce 40 kW of power. There is one reactor that is outputting 10 kW of electrical energy. This combination allows for the best growth and shrinkage of the combined power system in regards to the mass and volume.

Configuration			Solar Panel Power Size								
			20 kW			40 kW			60 kW		
Solar Panels	Nuclear Reactor	NR Sized	NR Power (kW)	Mass(Kg)	Volume (m^3)	NR Power (kW)	Mass(Kg)	Volume (m^3)	NR Power (kW)	Mass(Kg)	Volume (m^3)
2	0	NA	NA	133	0.5		267	1		400	1.5
4	0	NA	NA	200	0.75		300	1.25		500	1.75
2	1	1 Solar Panel	10	1677	2.4	20	2549	5.1	30	3286	8.3
2	2	1 Solar Panel	5	2155	1.8	10	2280	2.4	15	2986	7.5
4	1	1 Solar Panel	5	1211	1.9	10	1844	3.2	15	2432	4.7
4	2	1 Solar Panel	2.5	1702	2.4	5	1802	2.9	7.5	3086	4.9
4	4	1 Solar Panel	1.25	1926	2.1	2.5	1668	2.7	3.75	3961	5.5
2	1	Min Power	22	2415	4.6	22	2549	5.1	22	2682	5.6
2	2	Min Power	11	3221	4.4	11	3355	4.9	11	3488	5.4
4	1	Min Power	22	2482	4.9	22	2582	5.4	22	2782	5.9
4	2	Min Power	11	3288	4.6	11	3388	5.1	11	3588	5.6
4	4	Min Power	5.4	4244	5.5	5.4	4344	6.0	5.4	4544	6.5

Figure 12.4: Configuration Table

Chapter 13: Science Objectives

Traveling to Mars will provide ample opportunities to pave ways in deep space science. The science objectives can be split up into three categories: non-biological craft-based, non-biological autonomous, and biological. These objectives are all detailed in the following sections.

13.1: Non-Biological Craft Based

Several of the non-biological science objectives are based on the MOONS craft. These objectives are built into the structure of the MOONS vehicle. Their primary focus will be on the planetary bodies the MOONS craft will be orbiting. The first science objective is a set of two Ground Penetrating Radar (GPR) arrays. These radar will peer into the surface of Mars, Phobos, and Deimos, as there have been only a few studies looking beneath the surface of Mars. The primary mission for the GPR is to identify if Mars has an extensive dried lava tunnel network. Subsurface lava tunnels could reveal clues about Martian history, as they could have been used by past life on Mars. They could also provide clues as to how Mars' internal dynamo shut down, causing mass extinctions across the planet. Furthermore, intact lava tunnels could be explored further by future manned missions to the Martian surface.

The first GPR will use a 3 MHz pulse lasting 55 nanoseconds. This frequency will penetrate 100 m into the Martian soil. Since GPR1 uses a lower frequency and penetrates deeper, it will have a lower resolution of 25 m. The characteristics of GPR1 are summed in the Table 13.1.

Table 13.1: Ground Penetrating Radar 1 Data

Penetration Depth (m)	100
Resolution (m)	25
Frequency (MHz)	3
Sampling Interval (ns)	55
Dipole Length (m)	50
Power Draw (W)	65

The second GPR will use a 12.5 MHz pulse lasting 70 nanoseconds. The GPR2 are a series of bowtie antennas. They form a synthetic aperture. This frequency will penetrate 20 m into the Martian soil. GPR2 uses a higher frequency to achieve a higher resolution of 6 meters. The tradeoff is a much lower penetration depth. Since the frequency is higher, multiple booms can be placed along the longitudinal axis of MOONS craft. This will create a synthetic aperture, which will in turn allow GPR2 to create 3D images of the area it scans. Combining the scans from the radars, composite images from below Mars' surface can be constructed. Besides looking at the Martian subsurface, GPR1 and 2 will also focus their attention to the satellites, Phobos and Deimos. Neither of the two moons have ever been a focus of study on a successful mission. Due to this relatively little is known about the two moons. Deimos is untouched that only two physical features have been named. Scanning the subsurface of these two will provide never before seen insight and information.The characteristics of GPR2 are summed in the Table 13.2.

Table 13.2: Ground Penetrating Radar 2 Data

Penetration Depth (m)	20
Resolution (m)	6
Frequency (MHz)	12.5
Sampling Interval (ns)	70
Dipole Length (m)	12
Antenna Separation (m)	36
Power Draw (W)	10

The next instrument taken on the MOONS craft will be a two part magnetometer. The magnetometer will be used to study the Martian magnetosphere and ionosphere. Since Mars no longer has an inner dynamo, it produces no magnetic field. The Mars Global Surveyor used a small magnetometer to observe large crustal fields on Mars. These crustal fields are thought to give rise to localized mini- magnetic fields. The magnetometer of MOONS is significantly more powerful than that of Mars Global Surveyor. The first part of the Mars Orbiter Operating Near Satellites hardware is a fluxgate magnetometer. This is a similar system as to what is used on Cassini. A fluxgate magnetometer uses an alternating current to determine the background magnetic field. The dynamic range of the fluxgate magnetometer is from ± 44,000 nT to ± 40 nT, with a resolution of 5.4 nT and 4.9 pT respectively. It will be supported on a boom 7 meters away from the craft. Its characteristics are summed in Table 13.3.

Table 13.3: Fluxgate Magnetometer Data

Mass (kg)	.44
Power (w)	5
Frequency (Hz)	30
Boom Length (m)	7
Resolution/ Dynamic Range	4.9 pT, +- 40 nT
	48.8 pT, +- 400 nT
	1.2 nT, +- 10,000 nT
	5.4 nT, +- 44,000 nT

The second onboard magnetometer is a Vector/Scalar Helium Magnetometer (VSHM), a dual-mode magnetometer that can function in a vector mode where three orthogonal magnetic fields are generated and changes are calculated using Pythagoras' equation. In the scalar mode, the magnetometer measures the Larmor frequency proportional to the magnitude of the magnetic field. The resolution of the VSHM, ranges from ± 32 nT to ± 384 nT with resolutions of 3.9 and 36 pT respectively. Its characteristics are summed in Table 13.4.

Table 13.4: VSH Magnetometer Data

Mass (kg)	.71
Power (w)	6
Frequency (Hz)	10
Boom Length (m)	12
Resolution/ Dynamic Range	36 pT, 256, 384 nT
	3.9 pT, +- 32 nT
	31.2 pT, +- 256 nT

The final aspect of the main craft based science objectives is the camera system. The MOONS mission will be traveling with four cameras designed to capture new images of Mars and its moons. The first camera is similar to the camera on the Mars Reconnaissance Orbiter. The HiRise is the largest camera ever flown outside of Earth's sphere of influence. The MOONS craft will carry a secondary HiRise camera to Mars and its moons. Its main mission will be to scout out future landing sites for manned surface missions. The HiRise will have the new capability to work with the information garnered from the GPR's to scout out surface analogies. The HiRise camera is unparalleled in deep space based satellite resolution. At an altitude of 200 km, HiRise has a resolution of only 90cm/pixel. This gives MOONS the ability to survey Phobos and Deimos as never before. HiRise's characteristics are in Table 13.5.

Table 13.5: HiRise Camera Characteristics

Quantity	1
Power (W)	60
Mass (kg)	65
Volume (M^3)	0.54
Cost($)	$ 40,000,000.00
Aperture (cm)	50
Resolution cm/pixel	90
Swath Width (km)	3
Color Band	570-830 nm RED
	<580 nm BG
	> 790 nm NIR

The second camera system flying on MOONS will be a set of three Red Cameras. These cameras are significantly smaller than HiRise and serve slightly different function. They are designed to provide sweeping details of Mars and the moons. These details can be studied inside the MOONS craft and provide location information for the HiRise camera to focus on. The Red Cams will also play an important role in outreach. These cameras will constantly upload any new pictures to social media outlets in an effort to increase the awareness and excitement of space travel. As MOONS is the first manned mission to orbit Mars, constant media will bolster enthusiasm both on the ship and on Earth. The full swath of the Red Cam characteristics are summarized in Table 13.6.

Table 13.6: Red Cam Characteristics

Quantity	3
Power (W)	5.9
Mass (kg)	3.7
Volume (M^3)	0.001
Resolution km/pixel	3
Color Band	480 nm BLUE
	533 nm GREEN
	689 nm RED
	893.3 nm NIR

13.2: Non-Biological Autonomous

Since one of the primary objects of the MOONS craft is to study the Martian moons, a good portion of the science objectives will focus on Phobos and Deimos. The autonomous non-biological studies involve leaving cube satellites in orbit around the two moons. The Buzzword Extraterrestrial Exploratory Satellites (BEES) will continue to gather scientific data as MOONS heads home. The BEES will be based on designs from Aalto University based in Finland. These are 3U satellites carrying sophisticated data collection equipment. Onboard will be a spectral imager capable of taking images from 500 to 900 nm. With a resolution of 240 m/pixel, the satellites can accurately monitor the moons for any changes caused by asteroid impacts. The second piece of equipment on the BEES is a miniaturized particle telescope. This telescope can monitor the proton and electron radiation environment of the moons. Continuous data from the radiation environments could not only impact future missions, but also provide new information about deep space solar radiation. Placing these satellites in orbit around the moons will help gather new images that can be used for studying. These images can also be used to decide if a manned Martian moon base would be viable or beneficial. The satellites and their payload are summed in Table 13.7.

To place the satellites into orbit, they will need to be launched from the MOONS craft in some fashion. This will be done with the High Intensity Vehicle Expulsor (HIVE). HIVE is based on the NanoRaks Cube Sate Deployer currently used on the ISS. It is capable of launching satellites

Table 13.7: BEES Characteristics

Count	8	
Mass (kg)	40	
Volume (m^3)	0.03	
Cost ($)	$ 800,000	
Spectral Imaging		
	Resolution m/pixel	240
	Swath Width (km)	120
	Color Band	500-900 nm
Miniaturized Particle Telescope		
	Radiation Environment	
	Proton Energy Range	1-100 MeV
	Electron Energy Range	1-15 MeV
	Resolution	<5 MeV

from 1U size up to 6U. Since the BEES are 3U satellites, each HIVE is capable of launching 2 BEES. To have provide complete coverage of the Martian moons, a total of 8 BEES will be taken. This equates to 4 HIVEs. HIVE's salient characteristics are in Table 13.8.

Table 13.8: HIVE Characteristics

Nano Sat Launcher (HIVE)	Count	4
	Power (w)	120
	Mass (kg)	68
	Volume (m^3)	0.070

13.3: Biological Studies

Due to the pioneering nature of the MOONS mission, there will be many unknown factors of deep space travel such as cosmic radiation, long-term bone degradation, and brain damage. MOONS will provide key data points for future missions as a heritage flight in many of these areas. The first study will be done on the vitamin supplements to the crew. In several Earth-based studies, it has been theorized that the harmful effects of deep space travel, particularly deep space radiation, can be mitigated through the use of vitamin and mineral supplements. MOONS will provide the perfect base for such studies to be done. A proposed list of supplements and their benefits can be found in Appendix C. It is important to note that there are multiple supplements that provide similar benefits. For this reason the crew will be placed on a controlled regiment to monitor the effects of the supplements. Doing this will provide data on which supplements provide the most comprehensive protection. Having this information will be important to future deep space missions. The vitamin regiment is provided in Table 13.9. There are three sections. Crew members

A and B belong to the Red group, while members C and D belong to the Blue Group. All crew members will be taking supplements from the green group.

Table 13.9: Crew Supplement Regiment

Supplement Regiment		
Crew Members	**Taken Supplements**	**Protection**
A & B	Vitamin A	Radiation Damage, Vision Loss
	Arginine	Radiation Damage, Cardio-Vascular
	Lysine	Radiation Damage
	Beta Carotene	Radiation Damage, Vision Loss
	Adenine	Rapid Tissue Repair, Immune Response
	Guanine	Rapid Tissue Repair, Immune Response
	Lipoic Acid	Memory Loss, Aerobic Metabolism
C & D	Vitamin B	Vision Loss
	Proline	Radiation Damage
	Citruline	Radiation Damage
	Lutein	Radiation Damage, Vision Loss
	Resveratrol	Bone Loss, Vision Loss
	Querctin	Radiation Damage
	Cytosine	Rapid Tissue Repair, Immune Response
	Thymine	Rapid Tissue Repair, Immune Response
All	Vitamin D	Diseases, Memory Loss, Depression
	Vitamin K	Bone Loss
	Vitamin C	Cardio Vascular, Scurvy
	Magnesium	DNA Stabilization, Fcn of Mitochondria
	Zinc	Radiation Damage, Memory Loss
	Proline	Radiation Damage
	Selenium	Radiation Damage
	Iodine	Thyroid Health
	Astaxanthin	Radiation Damage, Anti-Inflammatory
	Carnitine	Steroid

The next set of biological studies comes in the form of exercise. Currently the ISS uses the COLBERT, the Cycle Ergometer with Vibration Isolation and Stabilization (CEVIS) cycle, and a resistive exercise device for fitness regimens. Together, these three systems weigh more than 1,300 kilograms. To maximize weight savings, MOONS will be dropping CEVIS and the resistive exercise device. Instead it will take a similar system to the COLBERT treadmill, and a newly developed resistive device called ROCKY. ROCKY can be used for both aerobic and anaerobic exercises. ROCKY is able to simulate upwards of 180 kilograms of resistance while only weighing 9 kilograms. Its small size makes it the ideal system to replace current systems. Crew members can perform a range of exercises, including squats, rows, deadlifts, heel raises, and curls. Examining the benefits of ROCKY would be a new study which could greatly help NASA in the future. MOONS provides the platform to test the device and potentially use improved versions for future flights.

The final set of biological experiments revolves around the brain. Due to the high levels of cosmic radiation, astronauts on their way to Mars could suffer from a condition dubbed space brain. The highly charged particles can place crew members at risk of early onset dementia and mental impairment. To monitor such things, MOONS will be taking four EEG machines. These EEG's are designed by Emotiv and are lightweight alternatives to their industrial counterparts. The Emotiv Epoch+ uses a 14 channel headband with a 43 Hz bandwidth. This allows the EEGs to gather data with a resolution of 14 bits. To filter out background noise, the Epoch+ includes a 5th order Sinc filter and notch filters at 50 and 60 Hz. Crew members in torpor cycles will be wearing the EEGs to monitor brain function. This will gather data for the body's response under torpor and monitor the effects of deep space radiation on the brain.

Chapter 14: Cost Analysis

The cost analysis for this mission was based upon the cost breakdown for spacecraft in the Space Mission Analysis and Design textbook.[11] The cost breakdown is split into six major sections. These sections are space system characteristics; advanced technologies cost, space segment cost, launch segment cost, ground segment cost, operations and maintenance cost, and life cycle cost. The given budget was $5 billion and after all costs were taken into account the mission's total cost is $4.22 billion, resulting in an overhead of 15.6%. This can be used to invest in technologies to improve TRLs before launch date and to cover any unforeseen costs. The following tables will detail what went into each section and give a total cost for each section.

Table 14.1: System Segment Costs

System Segment	Subsystems	Cost (Millions)
Advanced Technologies		**676**
	Torpor Module	234
	Dragon 2 Capsule	123
	Nuclear Reactor	169
	Bigelow Inflatable Module	150
Space Systems		**156.7**
	Payload	20
	Spacecraft	42.92
	Structure	5.28
	Thermal	1.15
	Electrical Power System	31.25
	Telemetry	6.48
	Command and Data Handling	7.94
	Attitude Determinations	9.64
	Propulsion	7.29
	Integration/Assembly Test	6.95
	Program Level	11.45
	Ground Support	3.3
	Launch/Operations Support	3.05
Launch Vehicle		**650**
	Falcon 9	60
	Falcon 9 Heavy	90
	SLS	500

Ground Segment		**2645.77**
	Flight and Ground Software	655
	Facilities	117.9
	Equipment	530.55
	Software	655
	Logistics	98.25
	Management	117.9
	Systems Engineering	196.5
	Product Assurance	98.25
	Integration and Test	157.2
	Earth Terminals, Antennas, and communication	19.22
Operations and Maintenance		**262.04**
	Maintenance	260.69
	Contract Labor	0.8
	Government Labor	0.55

Table 14.2: Life Cycle Costs

Life Cycle Segment		**Cost**
	Advanced Technologies	507,000,000
	Space Segment	156,700,000
	Launch Segment	650,000,000
	Ground Segment	2,645,770,000
	Operations and Maintenance	262,040,000
Total Cost		**4,221,510,000**

Chapter 15: Marketing

The five major factors that influence the behavior of any market are technology, competition, regulatory, economic, and socio-cultural. This chapter will discuss the marketability of the MOONS mission in terms of each of these factors.

15.1: Technology

Space technology has advanced extensively in the past decade in areas such as propulsion, power, habitation, and launch vehicles. The MOONS spacecraft is a combination of flight proven technologies and newer technologies with a reasonably high TRL. Examples of flight proven technologies on this spacecraft include the entirety of the ADCS, telecommunications, CDH, and thermal systems. The proven technologies mostly comprise mission critical subsystems, allowing for newer technologies to be used in other areas of the spacecraft or in the initial launch/mission end such as launch vehicles and crew modules. A summary of new technologies and their TRL's can be seen in Table 15.1.

This mission will also serve as a heritage flight for the nuclear reactor and solid state lithium batteries. Since there is a reasonable amount of redundancy in the solar panel and and battery system, these components are not considered high risk to mission success and their lower TRL is considered acceptable.

Table 15.1: New Technology TRL Summary

Technology	Company	Product	TRL
Engines	SpaceX	Raptor Engines	6
Nuclear Reactor	Los Alamos Laboratory	Kilopower	4
Inflatable Modules	Bigelow Aerospace	BEAM	9
Habitation	Spaceworks	Torpor Modules	2
Solar Arrays	Orbital ATK	MegaFlex	5
Launch Vehicle	NASA	SLS	6
	SpaceX	Falcon 9 Heavy	6
Crew Modules	SpaceX	Dragon 2	6
Average TRL			5.5

15.2: Competition

The space industry is a highly competitive industry that has experienced numerous technological advancements and announcements in recent years that have spurred other companies to compete. Competitive areas of the space industry include launch vehicle and crewed capsules. Companies involved in the launch vehicle market include NASA, SpaceX, United Launch Alliance, and Blue Origin. Other companies are involved in the crewed capsule market, such as Lockheed Martin, Blue Origin, SpaceX, and Boeing. Usually competition among products and companies produce better products, so it is assumed that this is the case for this mission.

15.3: Regulatory

There are two major regulatory units that directly affect this spacecraft. NASA standards (such as NASA STD 3001 Volume 2) directly influenced how different subsystems were sized, their capabilities, and also their limitations. However, this body does not change regulations, nor does it hold as much influence as the other body: government.

One of the few defined requirements for this spacecraft is a five billion dollar budget and a focus on proven technologies (and therefore technology development deadlines). This spacecraft relies on the use of SLS for one of the launch vehicles and is therefore directly influenced by the development of the Orion capsule, Exploration Ground Systems (EGS), and different congressional bills.

On March 21 of this year, President Trump signed the NASA Transition Authorization Act of 2017. This act calls for EM-1 and EM-2 to be accomplished by 2018 and 2021, respectively. NASA committed to these dates. It also requires a feasibility study into a 2033 Mars mission using SLS and Orion, as well as a budget and engineering effort report on how to develop and use the 130-ton cargo variant of SLS.[35]

However, congressional acts do not guarantee that subjects called out will be completed on time. In the NASA Authorization Act of 2010, Orion was directed to be fully operational and flight tested by December of 2010.[36] As of the submission of this report, this operational goal was still not met. New delays and recommendations also arose while this spacecraft was being designed. On April 27, 2017, the United States released their findings on a like delay for EM-1. Due to low schedules and cost reserves for Orion, SLS, and EGS, the office recommended that NASA confirm or deny the feasibility of a November 2018 launch for EM-1 and propose a new launch date, potentially in the next fiscal year (2019).[37] NASA agreed with and confirmed that EM-1 will be delayed.

The spacecraft does rely on one SLS launch. However, in response to this news, if the SLS is not available by the time of launch, then the SLS can be replaced with two Falcon heavies instead of the proposed configuration in this report.

15.4: Economic

As of this writing, the United States is in a bull market, where prices rise due to optimistic economic outlook. It will only benefit the success of the mission for the economy to continue to be in a bull market until the end of mission. However, since there is no tool (yet) to predict when a bear market (or crash) will occur, it is assumed that the bull market trend will continue throughout the life of the mission.

15.5: Socio-Cultural

A socio-cultural analysis involves identifying the largest and/or most influential stakeholder in the system and focusing on the beliefs and behaviors of this stakeholder. In this case, the most influential stakeholder is taxpayers.

Cornell University's Roper Center[5] is one of the largest archives of public opinions. Since 1990, they've polled United States Citizens about their NASA approval. The latest data published includes the year 2014. The results can be seen in Figure 15.1.

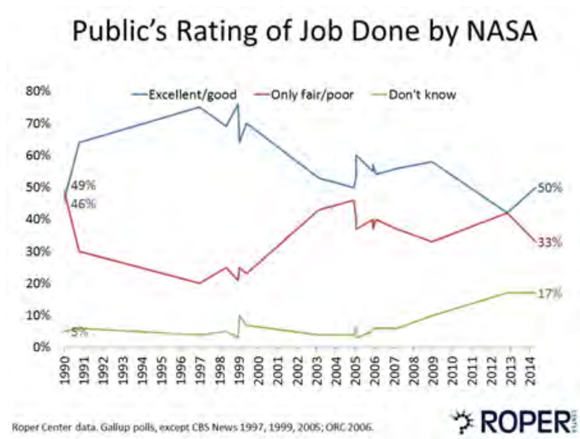


Figure 15.1: Public's Rating of Job Done by NASA Conducted by the Roper Center[5]

Although, the public's most recently published approval rating was 50%, their opinion most likely has increased in the past three years due to developments in the space industry through SpaceX and even internationally like the European Space Agency (ESA). Regardless, a mostly positive opinion from the major stakeholder will only help the mission.

Chapter 16: Mission Operations Plan

The mission operations plan details the step-by-step process of how the mission goals will be accomplished. The plan is broken down into a hierarchy of phases, mission segments, and mission items. Mission phases follow the mission chronologically, changing with the completion of a major milestone, for instance, the ground development or assembly of the space vehicle. These phases are broken down into mission segments, which represent the broad goals that must be accomplished in each phase. The mission items represent the smallest level of the mission architecture, and are the individual, itemized objectives that are completed by the ground and crew throughout the mission. The mission operations plan is as follows.

Phase 1: Research, Development, Testing, and Evaluation

Phase 1 includes the preparation of all necessary systems, crew selection, and crew training, which all occur prior to beginning the launches. By the end of this phase, all spacecraft systems will be developed, ground systems outfitted, crew fully trained, and everything will be prepared for launch and assembly in orbit. All segments of this phase happen concurrently.

Table 16.1: Mission Segment 1.1: Torpor System Co-Development

1.1a: Torpor Habitat Ground Trials	Extended ground trials, mimicking the two-week cycles planned for the MOONS mission.
1.1b: Torpor Habitat Space PoC Development	Earth orbit proof of concept mission for the torpor system is developed and prepared for launch.
1.1c: Torpor Habitat PoC Launch	Launch of the Earth orbit proof of concept mission for the torpor system.
1.1d: Torpor Habitat PoC Orbital Mission	Torpor system tested in Earth orbit, following the same two-week cycles as planned for the MOONS mission.
1.1e: Torpor Habitat PoC Deorbit	Reentry of the Earth orbit proof of concept for the torpor system.
1.1f: Torpor Habitat PoC Analysis	Analysis of the effects of Torpor cycles on human physiology and an assessment of the risk that the system poses to our mission architecture is conducted.

Mission Segment 1.1: Torpor System Development

The torpor system being developed by SpaceWorks is integral to future spaceflight to Mars. Torpor systems can greatly reduce the weight of consumables required per person, and can be seen as necessary for missions carrying a large crew, as for colonization of the red planet. As a mission that is meant to lay the foundation for large scale habitation of Mars, demonstrating the capability of these systems in long-term, deep space missions is key. Increasing these systems to a TRL of 9 will allow for the use of torpor chambers with confidence on the MOONS mission. To ensure this, the mission architecture includes the time and budget to partner with SpaceWorks to adapt their development schedule to this mission's needs. Mission segment 1.1 follows this partnership.

Mission Segment 1.2: Primary System Development

This segment separates each mission critical system and slots time and a portion of the budget for each system. As MOONS is using an inflatable habitat, Bigelow Aerospace will become a partner to develop an inflatable habitat for this mission. Bigelow already has the manufacturing facilities and proven inflatable space habitats, making the decision to contract them preferable to developing an independent inflatable habitat facility. Mission segment 1.2 follows the development of each mission critical system.

Table 16.2: Mission Segment 1.2: Primary System Development

1.2a: Bigelow Contracting Period	Interfacing with Bigelow to outline our needs and follow/monitor the development of the inflatable habitat
1.2b: Propulsion System Dev. + Manufacturing	Development and manufacturing of system
1.2c: Power System Dev. + Manufacturing	Development and manufacturing of system
1.2d: ECLSS System Dev. + Manufacturing	Development and manufacturing of system
1.2e: Structural System Dev. + Manufacturing	Development and manufacturing of system
1.2f: Telecom System Dev. + Manufacturing	Development and manufacturing of system
1.2g: GNC System Dev. + Manufacturing	Development and manufacturing of system
1.2h: Thermal System Dev. + Manufacturing	Development and manufacturing of system
1.2i: Command + Data System Dev. + Manufacturing	Development and manufacturing of system
1.2j: EVA System Dev. + Manufacturing	Development and manufacturing of system
1.2k: Science Instrument Dev. + Manufacturing	Development and manufacturing of system

Mission Segment 1.3: Ground System Development / Crew Training

This segment follows the processes of preparing the ground systems to support the MOONS spacecraft and crew for the duration of the mission. The first launches for the MOONS mission will not occur until after the ISS is decommissioned in 2024. These facilities are assumed to be available for use with the MOONS mission, as these facilities are already equipped with the systems necessary for training crew and providing operations support to manned space missions. This mission is an obvious progression towards NASA's pathway to Mars, making a partnership and the use of their facilities a natural choice over creating new ground facilities. Just as is done with the ISS, NASA's Johnson Space Center will be used as the primary training facilities for the MOONS astronauts and host ground control facilities in Mission Control Center (MCC) Houston.

Table 16.3: Mission Segment 1.3: Ground System Development / Crew Training

1.3a: Mock-Up Development	Development of ground mock-ups of the MOONS vehicle for crew training.
1.3b: Crew Selection	Application, interview, and selection process for astronaut candidates. A total of 12 will be selected and trained, with 4 being selected for the final mission. This allows 8 trained astronauts to be prepared for any follow up missions or last minutes emergencies with selected crew.
1.3c: Crew Training	Crew trained for key procedures and contingency procedures for the MOONS mission and instructed on operation of all critical systems.
1.3d: Johnson Space Center Crew Contracting Period	This represents the time and resources required to use Johnson Space Center facilities such as the Neutral Buoyancy Laboratory and Space Vehicle Mockup Facility.
1.3e: Ground System Outfitting	Transition of current Johnson Space Center Facilities from ISS operations to the systems and software required to operate the MOONS mission

Phase 2: Orbital Assembly

In this phase, the space vehicle will be launched and assembled in orbit and the crew will inhabit the vehicle. By the end of this phase, all launches will be completed and systems verified as ready for the Mars mission. The spacecraft and crew will be ready for Mars injection at the end of this phase.

Mission Segment 2.1: Launch Progression

This phase follows the launch and in-orbit assembly of the primary systems.

Table 16.4: Mission Segment 2.1: Orbital Assembly

2.1a: SLS Launch	The first launch is taken in an SLS, including the engine, propellant, radiator system, truss, solar panels, nuclear reactor, and the inflatable habitat.
2.1b: Falcon 9 Launch	The second launch in a Falcon 9 Heavy carries the torpor/EVA Airlock with all of the major systems for the spacecraft.
2.1c: Falcon 9 Heavy Launch	The third launch carries the Dragon capsule, the remaining LOX tank, and crew. This occurs after all critical systems in the first two launches are verified as operational.
2.1d: Docking	The second launch will dock directly with the truss and inflatable habitat. The LOX tank will then fill the remaining room in the tanks of the spacecraft and deorbit to burn in the atmosphere.

Mission Segment 2.2: Outfitting and System Verification

This segment follows the steps that prepare all vital systems for use and ensure that all systems are go for injection to a Mars transfer orbit. At the end of this stage, the Dragon capsule will be undocked and allowed to orbit Earth for the duration of the mission, reducing the thrust requirements for transit.

Table 16.5: Mission Segment 2.2: Outfitting and System Verification

2.2a: Solar Array Deployment	After the first launch is in its desired orbit, solar arrays are deployed and the power system is initialized.
2.2b: Radiator Deployment	After the power system is operational, the radiator systems will also be deployed.
2.2c: ECLSS Initialization and Verification	With power and thermal systems operational, the ground will initialize the Environmental Control and Life Support System to verify that it is prepared to sustain crew.
2.2d: Habitat Inflation	Once the ECLSS system is operating, the ground will begin the inflation of the habitable space using oxygen and nitrogen tanks brought up with the first launch.
2.2e: Crew Verification of Habitat Inflation and Solar Array Deployment	After the second launch, crew will visually verify that all power, thermal, and habitable systems are functioning nominally.
2.2f: Habitation Equipment Unloading and Installation	The crew will take the equipment brought up in the Torpor/Airlock and Unity modules and outfit the inflatable habitable space with all equipment used throughout the duration of the mission.
2.2g: Crew Verification of Systems with Spacecraft and Ground	Crew will walk through all critical systems with the ground to verify that all mission critical systems are operating nominally and ready for Mars transfer orbit injection.
2.2h: Crew Verification of Science Instruments	Crew will walk through all instruments required for science objectives to ensure systems are functioning and to refresh on the science objective procedures.
2.2i: Pre-Transfer Habitation Period	Before Mars injection, the MOONS vehicle is inhabited for a week long period to allow crew to acclimated to the space environment and provide a confidence in all mission critical systems.
2.2j: Dragon Undock	Dragon is undocked from the spacecraft to orbit Earth until it returns

Phases 3-5: Mars Mission

This phase follows the mission from Mars transfer injection to the recovery of the crew. By the end of this phase, all science objectives will be complete and the crew will be recovered on Earth. Segments in this phase occur in chronological order.

Phase 3: Earth-to-Mars Transit

This phase and mission segment follows the crew from the Earth-escape burn into the Mars injection orbit to just before entering the first Mars moon observation orbit.

Table 16.6: Mission Segment 3.1: Earth-to-Mars Transit

3.1a: Mars Injection Burn	The first burn to put the MOONS vehicle into a Mars transfer orbit.
3.1b: Initialization of Torpor Cycles	The first crew member is placed in stasis. The crew will take staggered two-week shifts in stasis to preserve consumables and demonstrate the technology in an interplanetary context.
3.1c: Deployment of Solar Panels Required for Mars Orbit	As the mission goes through its Mars transfer and grows more distant from the Sun, the solar arrays are continuously deployed to provide a constant amount of power.
3.1d: Mars Approach	The MOONS spacecraft begins to enter the Mars system and prepares for a burn to enter Mars orbit with the same semimajor axis as Deimos.
3.1e: Attitude Adjustment for Mars Observation Orbit Burn	The spacecraft orients itself for the Mars orbit burn.

Phase 4: Mars Orbit

Mission Segment 4.1: Mars Observation

This segment outlines the process of completing the observational science objectives at Mars.

Table 16.7: Mission Segment 4.1: Mars Observation

4.1a: Mars Observation Orbit Retrograde Burn	Burn to stabilize the Mars Observation Orbit.
4.1b: Full Crew Awoken from Stasis	Crew fully woken so that all four are able to assist in Mars observation tasks.
4.1c: In-Spacecraft Observation of Mars for Landing Sites	Observations of Mars are made using instruments on the station to characterize its surface and identify suitable destinations for future landing missions.
4.1d: In-Spacecraft Observation of Mars Climate and Geology	Observations pertaining to the geology and climate of Mars are made using instruments on the MOONS vehicle.
4.1e: Attitude Adjustment for Earth Transfer Orbit Burn	Spacecraft is oriented for the burn to inject to Earth transfer orbit.

Mission Segment4.2: Mars Moon Observation

This mission segment follows all mission operations pertaining to the observation of Phobos and Deimos.

Table 16.8: Mission Segment 4.2: Mars Moon Observation

4.2a: Moon Observation Orbit Burn	A burn stabilizes the MOONS spacecraft in an orbit suitable for observation of Phobos.
4.2b: In-Spacecraft Observation of Phobos	Observations of Phobos are made using instruments on the station to characterize its surface and identify suitable destinations for future landing missions.
4.2c: Deployment of Phobos Probes	Ground probes for observation of the surface of Phobos are deployed by launches on the outside of the MOONS vehicle.
4.2d: Attitude Adjustment for Deimos Observation Orbit Burn 1	The spacecraft is oriented for the first burn to transfer from Phobos orbit to Deimos orbit.
4.2e: Deimos Observation Orbit Prograde Burn 1	The first of two burns to heighten orbit from Phobos orbit to Deimos orbit.
4.2f: Attitude Adjustment for Phobos Observation Orbit Burn	The spacecraft is oriented for the second burn to transfer from Phobos orbit to Deimos orbit.
4.2g: Phobos Observation Orbit Prograde Burn 2	Second burn to stabilize an orbit at Deimos distance.
4.2h: In-Spacecraft Observation of Deimos	Observations of Deimos are made using instruments on the station to characterize its surface and identify suitable destinations for future landing missions.
4.2i: Deployment of Deimos Probes	Ground probes for observation of the surface of Phobos are deployed by launches on the outside of the MOONS vehicle.
4.2j: Attitude Adjustment for Earth Injection Burn	The spacecraft is oriented for the transfer to the Mars observation orbit.

Phase 5: Crew Return

This segment follows the crew from the burn into Earth transfer orbit to just before entering Earth orbit.

Table 16.9: Mission Segment 5.1: Mars-to-Earth Transfer

5.1a: Earth Injection Burn	Burn to enter Earth transfer orbit.
5.1b: Reinitialization of Torpor Cycles	Crew reenters same torpor cycles as their Mars-to-Earth transfer phase.
5.1c: Earth Approach	The MOONS spacecraft begins to enter Earth orbit.
5.1d: Attitude Adjustment for Earth Orbit Burn	Spacecraft is oriented for the burn to stabilize in Earth orbit.

Mission Segment 5.2: Crew Recovery

This segment follows the Earth orbit stabilization and recovery of the crew. This is the final segment of the primary mission.

Table 16.10: Mission Segment 5.2: Crew Recovery

5.2a: Earth Orbit Burn	Burn to enter Earth orbit.
5.2b: Dragon Rendezvous and Docking	Dragon rendezvous with the MOONS spacecraft
5.2c: All Personal Items Stowed in Dragon	All items that are to be brought back to the ground are loaded into Dragon.
5.2d: Crewed Dragon Capsule Sealed and Readied for Reentry	Crew enters Dragon and readies for reentry. Ground teams verify that all reentry systems are safe and ready for reentry.
5.2e: Dragon Deorbit	Dragon is deorbited with crew.
5.2f: Crew Recovered from Dragon	Crew is recovered from the Dragon at sea.

Chapter 17: Future of Spacecraft

After the MOONS spacecraft reenters Earth orbit and the crew are returned to Earth in the Dragon II capsule, it will remain in Earth orbit while ground crew performs an assessment of the craft's reusability. This assessment will begin by monitoring the efficiency of the ECLSS and habitable modules for their maintained habitability. This series of analyses will be completed with the intention of ensuring it is safe enough to send future crews up to verify and repair the systems on the MOONS vehicle. If the spacecraft is cleared for human habitation, then future crews can be sent up to study the efficacy of other systems and make repairs through EVA as needed. Freighter missions to bring up additional fuel for the vehicle allow for future missions to use the same spacecraft used in the original mission to continue trips to orbit Mars.

This design emphasis on scalability combined with the mission plan optimized for time of flight allow for larger, longer missions to take place using the same base vehicle in the future. Configurations with more mass can make trips to Mars using the same propulsion system if they take less energy intensive trajectories, such as a Hohmann Transfer, which would offer the maximum amount of dry mass for a given propulsion system. If later missions using the MOONS spacecraft were to use a Hohmann Transfer rather than a trajectory optimized for TOF, an additional dry mass of 8,000 kg could be used. The amount of available dry mass available will also increase as torpor systems mature, allowing for a further reduction in the mass of consumables. This additional mass can be used to increase the number of science payloads or increase the crew size of future missions.

17.1: Manned-Lander Configuration

In missions focusing on bringing large crews, an unmodified SpaceWorks Torpor module can be used to induce the whole crew into stasis, for crews of up to 6. This will eliminate the need for most of the equipment in the habitable environment as well as the sizing requirements for habitable volume, allowing all of the volume and weight of the B125 module to be removed. These savings total to an additional 7,300kg that can be used in addition to the bonuses from the trajectory, for an added total of 15,300kg. In this variant, the mass and volume formerly used for the B125 module can be purposed towards a manned lander. The Apollo 11 Lunar Module had a mass of 15,102kg,[38] so a similarly sized Mars lander could be delivered to Mars using the MOONS vehicle.

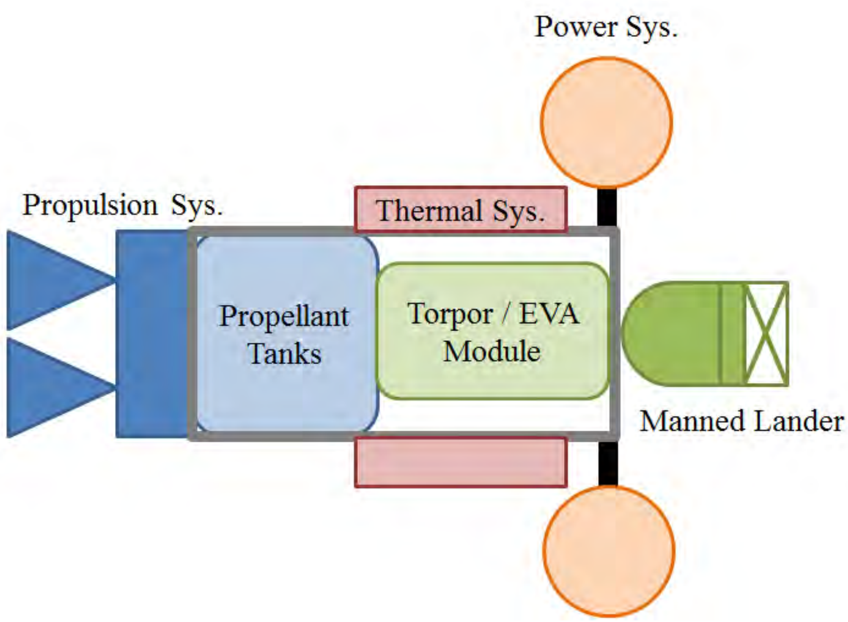


Figure 17.1: Lander Configuration Concept

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Appendices

Appendix A: C&DH

Subsystem	Task	LOC (ADA)	Code (kbits)	Data (kbits)	Throughput (KIPS)
CDS	Telemetry Processing	1500	960	240	3
	Command Processing	3000	1920	480	7
	Polling/multiplexing	1200	768	192	0.5
	Formatting	600	384	96	0.5
	Configuration table	1100	704	176	1
Telecommunication	Uplink processing	900	576	144	8
	Downlink processing	800	512	128	10
Attitude Control	Attitude determination	1500	960	240	14.8
	Attitude control	2400	1536	384	50
	Ephemeris processing	975	624	156	4
Articulation	Solar array control	900	576	144	9
	Secondary HGA	500	320	80	0.4
	UHF antenna	1200	768	192	1.2
	HGA	1200	768	192	2
Fault Protection	Safing	1500	960	240	0.8
	CDS fault protection	1800	1152	288	5
	ACS fault protection	11500	7360	1840	5
Operating System	OS	1000	640	160	64.5
Utilities	Utilities	4000	2560	640	1.1
Environment Control and Life Support	Air Content Regulation	1100	704	176	2
	Interior Thermal Control	400	256	64	0.8
	Humidity regulation	400	256	64	0.2
	Water filtration	800	512	128	1
	Waste control	600	384	96	0.5
	Sabatier reaction control	1000	640	160	5
	Stasis monitoring systems	1400	896	224	7
	Stasis system control	4000	2560	640	12
Science Objectives	On-board sensor control	3000	1920	480	14
	Data processing	2500	1600	400	10
	Satellite network control	2700	1728	432	20
	Totals	55475	35504	8876	260.3
	Total RAM Required	88.76	Mbits		

Appendix B: Power

Nickel Cadmium Batteries	
Power (kW)	140
Mass (kg)	4202
Cell Number	814
Discharge Voltage (V)	1.2
Volts per Cell (V)	4
Cycle Life (Cycles)	10000
Transmission Efficiency (%)	97
Specific Energy (W-h/kg)	24
Lithium Polymer Batteries	
Power (kW)	140
Mass (kg)	380
Cell Number	10
Discharge Voltage (V)	33.3
Volts per Cell (V)	3.7
Cycle Life (Cycles)	800
Transmission Efficiency (%)	97
Specific Energy (W-h/kg)	265
Lithium Ion Batteries	
Power (kW)	140
Mass (kg)	611
Cell Number	482
Discharge Voltage (V)	4.1
Volts per Cell (V)	3.7
Cycle Life (Cycles)	800
Transmission Efficiency (%)	97
Specific Energy (W-h/kg)	165
Solid State Lithium Ion Batteries	
Power (kW)	25
Mass (kg)	25
Cell Number	10
Discharge Voltage (V)	33.3
Volts per Cell (V)	3.7
Cycle Life (Cycles)	1200
Transmission Efficiency (%)	97
Specific Energy (W-h/kg)	735

Appendix C: Science

Supplements		Sources	Role	Theoretical	NASA Recommended Amount (mg)
Fat-Soluble Vitamins	A	Carrots, Sweet Potatoes, Kale	Radiation Damage, Vision Loss	X	0.9
	D	THE SUN	Diseases, Memory Loss, Depression		0.025
	E	Nuts, Avocados, Sweet Pot	Radiation Damage, Cardio-Vascular	X	15
	K	Kale, Spinach, Leafy Greens	Bone Loss		0.12
Water Soluble Vitamins	B	Legumes	Vision Loss	X	1.7
	C	Kale, Banana, Avocado	Cardio Vascular, Scurvy		90
Minerals	Magnesium	Nuts, Cocoa, Spinach, Kale	DNA Stabilization, Fcn of Mitochondria		420
	Zinc	Wheat, Nuts,	Radiation Damage, Memory Loss		11
	Selenium	Nuts, Fish	Radiation Damage		0.4
	Iodine	Potatoes	Thyroid Health		0.15
Amino Acids	Arginine	Granola, Nuts	Radiation Damage	X	-
	Lysine	Legumes, Dairy	Radiation Damage	X	-
	Proline	Synthetic	Radiation Damage	X	-
	Citruline	Synthetic	Radiation Damage	X	-
Carotenoids	Beta Carotene	Carrots, Sweet Potatoes, Kale	Radiation Damage, Vision Loss	X	-
	Lutein	Kale, Spinach, Leafy Greens	Radiation Damage, Vision Loss	X	-
	Astaxanthin	Shrimp, Salmon	Radiation Damage, Anti-Inflammatory		-
Non Vitamin Nutrients	Lipoic Acid	Spinach Broccoli	Memory Loss, Aerobic Metabolism	X	-
	Carnitine	Carrots, Sweet Potatoes, Kale	Steroid		-
Polyphenols	Resveratrol	Red Wine, Grapes	Bone Loss, Vision Loss	X	-
	Quercetin	Spinach Broccoli	Radiation Damage	X	-
Supplemental Nucleotides	Adenine	Legumes, Seafood	Rapid Tissue Repair, Immune Response	X	-
	Guanine	Legumes, Seafood	Rapid Tissue Repair, Immune Response	X	-
	Cytosine	Legumes, Seafood	Rapid Tissue Repair, Immune Response	X	-
	Thymine	Legumes, Seafood	Rapid Tissue Repair, Immune Response	X	-

Appendix D: Spacecraft Views