Crewed Orbit and Ascent Surface Transportation (COAST)

AIAA Design Competition Mars Dual Lander Conceptual Design

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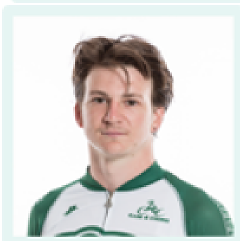

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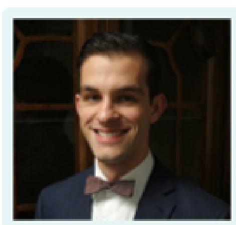

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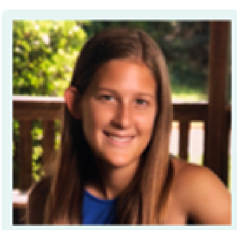

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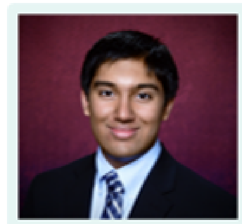

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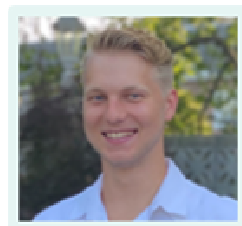

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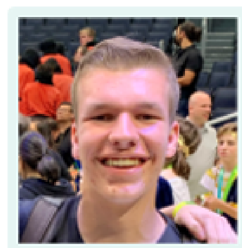

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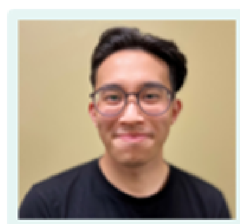

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I. Executive Summary

The objective of Program COAST is to create a Martian dual-lander ascent system. Two humans will ascend from the Martian surface to the Deep Space Transit (DST) in orbit. This objective will be achieved by designing two landers for the mission, one housing the Mars Ascent Vehicle with a life support system for human occupation, and a support lander carrying the ascent propellant. The mission success criteria include providing a launch method to transport these humans and 50 kg of scientific samples from the surface of Mars to the Deep Space Transit in Mars orbit and providing autonomous refueling and launch support operations for the MAV.

Stakeholders for this mission at this stage include NASA, Congress, Mission Partners (European Space Agency), Georgia Tech, and the American Institute of Aeronautics and Astronautics (AIAA).

The first essential mission element is the Space Launch System (SLS), required in the AIAA Announcement of Opportunity for launch from Earth. The MAV is used to transport the crew from the surface of Mars to orbit, where it can rendezvous with the DST to bring the humans back to Earth. The decided upon method for the propellant transfer process was three rovers, called Propellant Hauling Integrated System Hardware (PHISH). Additionally, a scaled-up version of MOXIE (SMOXIE) will be used to produce Oxygen on the surface of Mars. The final elements are the two landers, between which the propellant transfer will occur. The two landers designed for this mission were nicknamed "Land-On-Mars" and "ISLE". Land-On-Mars was designed to contain the propellant for the transfer process. The ISLE lander contains the MAV, PHISH, and SMOXIE.

The first phase is mission launch, which is accomplished using the SLS. The travel to Mars will take around 1 year, and upon arrival at Mars the landers will begin aerobraking for a duration of around 10 months. The next phase is entry, descent, and landing which will takes a duration of approximately 7 minutes. The landers will deploy upon landing at a distance apart of around 1 kilometer. Upon deploying, the PHISH rovers will begin the propellant transfer process. This entire process will take around 2 years, during which the two SMOXIE units will be producing LOX for the propellant oxidizer. After a period of two years, the crew of two humans will arrive in mid-2040 ready for launch. The MAV will ascend from Mars, carrying the crew and 50 kg of samples, and ultimately rendezvous in orbit with the Deep Space Transit after a period of 3 days.

The Attitude Determination and Control System (ADCS) was split into modes for each phase of the mission. Some of these modes require no rotational maneuvers to exist correctly; for example, the antenna on ISLE is omnidirectional, and thus needs little help from the ADCS system to fulfill its pointing requirement.

However, required rotational maneuvers for the remaining modes include reorientation burns for the landers during entry, landing, and descent, TCMs to adjust the launch trajectory of the MAV towards the DST, and reorientation burns for the MAV during rendezvous with the DST.

The ADCS system in the landers is primarily used for entry, landing, and descent. As such, it needs sensors capable of gathering position data relative to the surface of Mars, rather than relative to objects in space like most sensors. As such, the landers contain an IMU and 3 optical navigation cameras for position determination; the IMU gathers position data using the accelerometer, and the optical navigation cameras gather initial position data to supply to the IMU and supplemental data as redundancy for the IMU accelerometer measurements. The landers also have such a large mass that it is unlikely that a control moment gyro will be capable of making the trajectory adjustments required during entry, landing, and descent. As such, the landers also contain 6 110N hydrazine monopropellant thrusters, with 2 thrusters aligned along each axis for redundancy. The ADCS system in the MAV is primarily used for launch and for rendezvous with the DST in the 5-sol orbit. In this case, sensors fixed on other celestial bodies can be used, and in most cases are more efficient. As such, the primary sensor used on the MAV is a star tracker to gather relative position to nearby bodies. An IMU is additionally used to supplement the position data gathered by the star tracker and to gather acceleration data from launch.

The Environmental Control and Life Support System (ECLSS) is applicable to the MAV, as that is the only vehicle which will need to support human life. The main human needs which were accounted for in the ECLSS system are water supply, oxygen, pressurization, waste elimination, radiation shielding, CO2 removal, dust/contaminant mitigation, and food. Following a series of trade studies, the selected options for ECLSS were stored water in a tank, stored oxygen at high pressure, stored nitrogen at high pressure, lithium hydroxide CO2 canisters, Meals Ready to Eat (MREs), waste bags, a metallic radiation field, and fans and dust suits for ventilation.

The Landers have a dry mass of 10,422.680 kg (Land-On-Mars) and 10,440.550 kg (ISLE) as well as a landed mass of 19041.109 kg (Land-On-Mars) and 24321.4844 kg (ISLE) with a mass contingency of 30%. The thin wall for the landers was calculated by assuming an internal pressure of 1 ATM (101325 Pa) and Mars's surface pressure (external pressure) of 600 Pa. To allow for easy refueling and launching strategy the ISLE incorporates a petal landed configuration which allows the MAV a stable platform to launch.

To accommodate every subsystem, the MAV is 12.6 meters tall with an overall volume of 152.6 m3. The MAV's main structure is made of Aluminum 2024-T3 due to its high yield strength.

Thermal control is a challenge both in space and on the Martian surface due to large temperature fluctuations. With this environment in mind, notably the 400K difference in internal and external temperature, a physical implementation scheme may be developed. In particular, the cryogenic storage tanks would require special attention due to their required 5 W/cm emitted heat flux in order to maintain favorable operation temperatures. By employing Multilayer Insulation (MLI), this important part of the MAV may be adequately specified as using 10 layers of insulation to provide the required heat flux. The supplementary methods to be used in the Thermal Protection System will be louvers, heat pipes, and radiators where best specified at the sub component and part level.

Entry, descent, and landing is another challenge with the thin atmosphere of Mars. The overarching principle is to use a HIAD with 15m diameter along with a heat shield of nose radius 10m with overall diameter of 8.4m as the primary components. RCS thrusters will provide necessary corrective thrust during the entire phase, with distinct emphasis at 1km and below.

The robustness of our power systems for a mission of this scale are absolutely critical to smooth and successful operation. By using the previously shown subsystem power budget as well as knowledge of the system architecture and operation, a table of power requests for each subsystem over the course of the mission was created. Using the required power as a guide, the power systems for our mission were chosen. A 10 kilowatt fission power unit is required to be the main power source of the mission. Different battery systems were chosen for each vehicle in order to store power needed for operation. For the main MAV power system, primary LiCl batteries were chosen because of their high specific energy and low degradation rate. The main power source for Lander Two is the 10 kilowatt fission reactor, but this is supplemented by secondary backup batteries. These backup batteries have a 10 kilowatt capacity in case of any small emergencies or contingencies. Lander One uses advanced Mars solar panels for its main power source, although it requires a fairly small amount of power. The PHISH Rovers also use solar panels to supplement their power, however most of their power comes from the fission power unit on Lander Two. During propellant transfer, the PHISH on-board batteries are charged enough to complete their next trip.

Propulsion systems need to be considered for the MAV as well as the landers. The MAV needs a main propulsion system to launch into orbit and it also needs a propulsion for attitude determination and control. The landers need a retro-propulsion system for the Entry Descent and Landing (EDL) process. The landers will have six 250 kN bipropellant engines that are similar to the RS-72 engine. The engines will use a combination of MMH and MON-25 propellant. There are multiple check valves as well as relief valves across the schematic for safety and redundancy. There is also a quick disconnect for safety while loading the tanks. For ADCS, 6 Aerojet MR-107T thrusters are utilized. This is a 110N monopropellant hydrazine thruster. A monopropellant thruster is chosen for ADCS for its simplicity and ability to fulfill the purpose. Much like the main MAV propulsion system, redundancies and safety checks are built in with the various valves and other mechanisms.

Communication systems and budgets must be allocated for each of the vehicles and for each mode of communication the vehicles undergo during the mission duration. The communication budgets were broken into four main types of transmission: data, telecommands, health status of vehicles, and emergency signals. The main communications strategy for the mission is similar to past missions to the surface of Mars. Data that is not time sensitive will be relayed through the DST before heading back to Earth, rather than being sent to Earth directly. Telecommands transmit actions and autonomous operations to the vehicles for operation during communications blackouts and delays. These allow the vehicle to operate separate from human communication links, especially during the refueling operations where no humans are involved yet. Such autonomous events include the refueling operations on the surface of Mars, MAV docking with the DST, SMOXIE oxygen production, anomaly detection and handling, and general operations during communication blackouts during orbit and during EDL.

Some challenges are presented in the original problem statement, such as accounting for communication blackouts during entry, descent, and landing. With such concerns comes a need for redundant communication systems to ensure successful operation of the robotic refueling mission and the human launch mission. Such a strategy requires each vehicle to have two methods of communicating with Earth. Within the mission architecture, the DST is used as a relay throughout the duration of the autonomous and human mission on Mars.

The mission cost estimate encompasses the total cost for the design, development, and operation of all vehicle components as well as the cost for the launches using the SLS. This resulted in a total cost of $9.13 billion for the entire mission. The AIAA prompt specifies that the cost of the landers and launch vehicles do not need to be factored in for the total cost. Therefore, this brings the mission budget to a total of $3.73 billion.

The most critical risk to the mission was determined to be abnormalities associated with performing a remote launch of the MAV. Since the ultimate goal of the mission is to return the crew and samples from the surface of Mars, any problems associated with that launch would prove detrimental to the mission as a whole. Although the likelihood that this would happen to an extent that the mission would be derailed is low, the severity of this risk would be high. The second most important risk was associated with the entry, descent, and landing stage of the mission. Since this mission requires landing the largest payload ever onto the surface of Mars, it is expected that there will be risks associated with this maneuver. Both of these aforementioned risks must be accepted, as they are essential to the successful completion of the mission. The third risk addresses the possibility of a rover failure during propellant transfer. The strategy to address this risk is mitigation, which was done to a great extend by introducing the concept of redundancy with the three rover strategy. While it is unlikely that there will be enough rover failures to greatly impact the propellant transfer process, the result of that occurrence would greatly impact the next steps to be taken for the duration of the mission.

The COAST team has prioritized risk mitigation and reliability throughout the mission architecture development. The system has been strategically designed to address risks such as dust and temperatures from the Martian environment, reduced LOX production rates, and rover abnormalities. The use of three propellant transfer rovers utilizes the concept of redundancy to ensure a successful operation on the Martian surface. Future work on this project includes further analysis of the system architecture, and final component and configuration selection.

II. Introduction

A. Mission Objective

The objective of Program COAST is to successfully land the Mars Ascent Vehicle (MAV) and fuel the vehicle while on the surface of Mars. At the end of a crewed mission to Mars, these elements will be used by two humans to ascend from the Martian surface to the Deep Space Transit (DST) in orbit. This objective will be achieved by designing two landers for the mission, one housing the Mars Ascent Vehicle with a life support system for human occupation, and a support lander carrying the ascent propellant. The mission success criteria include providing a launch method to transport these humans and 50 kg of scientific samples from the surface of Mars to the Deep Space Transit in Mars orbit and providing autonomous refueling and launch support operations for the MAV.

B. Stakeholders

The mission is currently at a Pre-Phase A life-cycle stage. Stakeholders for this mission at this stage include NASA, Congress, Mission Partners (European Space Agency), Georgia Tech, and the American Institute of Aeronautics and Astronautics (AIAA). Stakeholders during other phases of the mission are shown in Table 1.

Table 1 Stakeholders during different phases of the mission.

Life-Cycle Stage	Stakeholders
Pre-Phase A	NASA, Congress, Mission Partners (European Space Agency), Georgia Tech, AIAA
Phase E	NASA Scientists, College Research Labs, Scientific community interested in Mars data
Phase F	Science Community, Planetary Protection

C. Science Traceability Matrix

The Science Traceability Matrix developed for this mission is based on science goals outlined in the announcement of opportunity. As this mission was further developed, the specifications were updated to match the parameters necessary for its successful completion.

Science Goals	Science Objectives	Scientific Requirements		Instrument Requirements		Projected Performance	Mission Requirements (Top Level)
		Physical Parameters	Observables				
Maintain adequate conditions for safe crew transport	Provide breathable air for 2 crew members	Monitor oxygen and carbon dioxide levels	Relative concentration of O_2 and CO_2	Sensor resolution	+/- 30 ppm	+/- 30 ppm	Must provide 2 kg of oxygen / day
	Characterize risks o f ionizing radiation on surface	Ionizing radiation exposure to crew	Dosimetry levels and type of radiation	Dosimeters to limit exposure; atmospheric sample analysis	<0.1 mSv	Safe levels across mission entirety	Astronauts' exposure must be below limits
	Maintain an appropriate cabin temperature	Determine temperature of the cabin	Air temperature	Thermometer resolution	0.01 deg C	0.01 deg C	Cabin temperature must remain between 5 and 40 deg C
	Provide water supply	Determine water needs of crew and any applicable systems	Consumption rate vs supply	Tank sensor	0.001 %	0.001 %	Must provide 30L of potable water
Reach the desired orbit	Reach a Mars 5-sol parking orbit	Closed orbit	C3	Accelerometer, Star Tracker	Less than Mars escape velocity (4.25 km/s) to achieve a negative C3 value	Negative C3 value for entirety of orbit	Must enter a closed (circular or elliptic) orbit
		Period	Radius / Velocity				Must have an orbital period of 5 Martian Days
Use efficient power systems to travel	Use required fission surface power unit	Specific impulse and thrust requirements	Fuel and flow rates required	Enough propellant and coolant to operate fission power unit within safe limits		Surface portion of mission (generation and transfer), ascent	Must have fission surface propulsion unit to power necessary systems in fulfillment of mission; ensure safe transfer of fuel
			Thrust and ISP	10kW subdivided across spacecraft subsystems and electrical subcomponents		Adequate power provided as defined by subsystem requirements	
Transport Martian samples	Capacity to return 50 kg of Mars samples	Dimensions and weight	MAV dimensions, maximum weight of samples	Weight capacity	50 kg Mars samples	50 kg Mars samples	Must have capacity to contain samples and adequate thrust to carry samples
Transfer propellant from lander to the MAV	Transfer propellant from Land-On Mars to ISLE	Distance from target area of the crewed vehicle	Infrared proximity sensor measurement s in X, Y, Z directions	Relative distance	< 50 mm	50 mm	Must recognize the target location for propellant on crewed vehicle and the relative distance from that location
	Define position and power required for propellant transfer	Location on the Martian surface	Point-to-point distance measurement	Relative distance	0.001 km < Z < 1 km	0.001 km < Z < 1 km	Must be able to navigate to the same location on the surface of Mars for transfer of propellant
		Power required to reach the crewed lander	Power required to travel 1 km on the surface	Power	3 kW	3 kW	
	Automate lander and payload transfer	Ability to fill	Fill rate	Flow rate	1 L/m < Z < 100 L/m	1 L/m < Z < 100 L/m	Must be able to autonomously refill MAV without manual intervention

Fig. 1 Science traceability matrix for COAST mission.

D. Baseline and Threshold Mission

The baseline science mission is the mission that fulfills the full science objectives of the mission, while the threshold science mission describes the minimum achievements for the mission to be a worth investing in. The descope describes the difference between the baseline science mission and the threshold science mission. Each one of these science missions can be found in Table 2.

Table 2 Baseline versus threshold mission.

Baseline Science Mission	The baseline science mission is to land a refueling mission for transportation of humans from the surface of Mars by mid-2040 and return the humans and collected samples back to Deep Space Transit (DST) safely, leaving the support landers behind to continue collecting data about the Martian environment.
Threshold Science Mission	The threshold science mission is to return humans and Martian samples to DST safely from Mars.
Descope	Alterations that limit the ability to continue science observations past the point where humans depart the surface of Mars.

III. Mission Architecture

A. Literature Review

A literature review was initially conducted to provide background information based on similarly performed missions and relevant mission proposals. The initial sources used for this review were shared in the AIAA Announcement of Opportunity [1, 2] and ultimately expanded as additional sources were found regarding the development of a MAV system designed for transporting humans from the surface of Mars. These sources provided the baseline for the conceptual design of the mission, and more specific sources relevant to each subsystem were used as the concept design matured.

B. Mission Elements

The first essential mission element is the Space Launch System (SLS), required in the AIAA Announcement of Opportunity for launch from Earth. The MAV is used to transport the crew from the surface of Mars to orbit, where it rendezvous with the DST to bring the humans back to Earth. The decided upon method for the propellant transfer process was three rovers, called Propellant Hauling Integrated System Hardware (PHISH). Additionally, a scaled-up version of MOXIE (SMOXIE) will be used to produce Oxygen on the surface of Mars. The final elements are the two landers, between which the propellant transfer will occur. The two landers designed for this mission were nicknamed "Land-On-Mars" and "ISLE". Land-On-Mars was designed to contain the propellant for the transfer process. The ISLE lander contains the MAV, PHISH, and SMOXIE.

C. Concept of Operations

The concept of operations for the mission is shown below. The first phase is mission launch, which is accomplished using the SLS. The travel to Mars will take around 1 year, and upon arrival at Mars the landers will begin aerobraking for a duration of around 10 months. The next phase is entry, descent, and landing which will takes a duration of approximately 7 minutes. The landers will deploy upon landing at a distance apart of around 1 kilometer. Upon deploying, the PHISH rovers will begin the propellant transfer process. This entire process will take around 2 years, during which the two SMOXIE units will be producing LOX for the propellant oxidizer. After a period of two years, the crew of two humans will arrive in mid-2040 ready for launch. The MAV will ascend from Mars, carrying the crew and 50 kg of samples, and ultimately rendezvous in orbit with the Deep Space Transit after a period of 3 days.

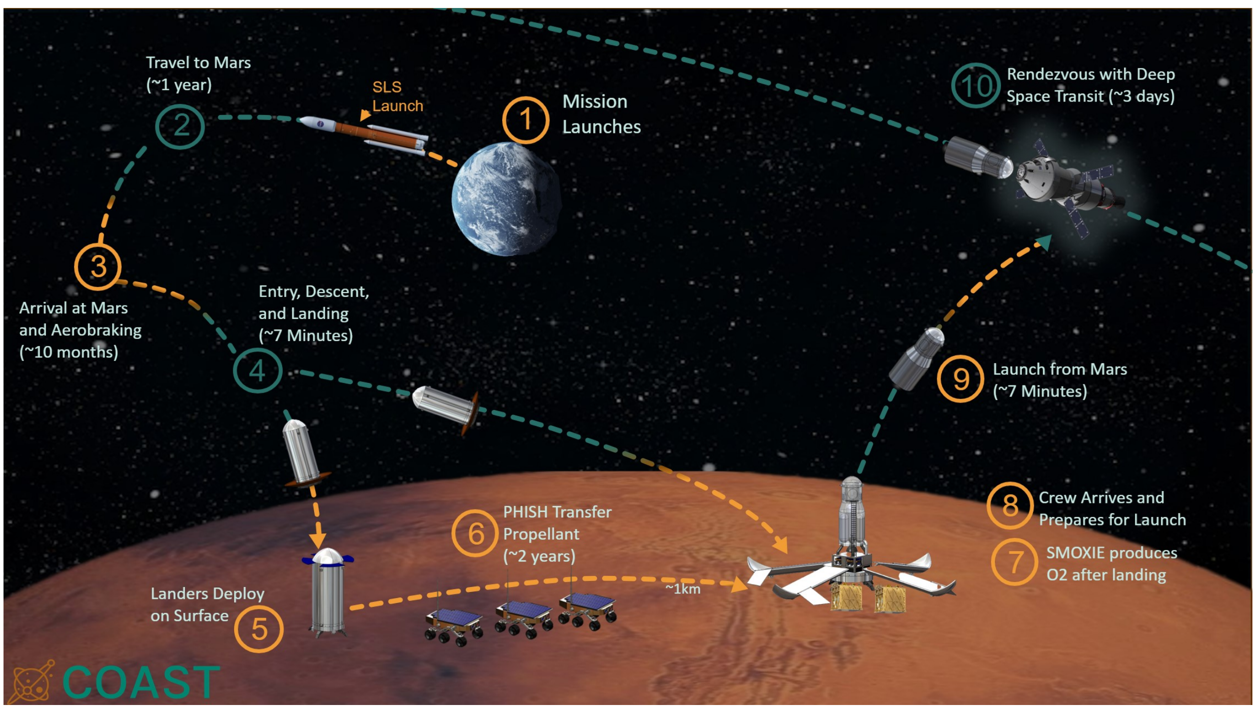


Fig. 2 Concept of operations.

D. Landing Site

The landing site selected for this mission is the Elysium Planitia. The main considerations for landing site were its elevation, proximity to the equator, and surface conditions. With a latitude of 4.5 degrees North [3], the landing site will have a beneficial impact on delta V provided during launch. Additionally, the elevation of around 1.65 km below the reference zero elevation on Mars will result in additional time for atmospheric drag to take effect during entry, descent, and landing. Finally, this landing site is a proven concept: the Insight lander has already touched down in this location, choosing it for its safety due to its flat surface. A site lacking in hills and large rocks will result in a smoother propellant transfer process for the PHISH.

E. Power Budget

The power budgets were established based on each vehicle based on the amount of power available during each phase of the mission. In addition to the 10 kW fission power unit provided, it was decided to use additional solar panels on the Land-On-Mars lander to provide power for the mission. The following power budgets detail the allocated power for each vehicle including a contingency margin of 30%. More detailed power budgets separated by mission phase will be provided in more detail in the power subsection below.

A. MAV	Power Percentage	CBE (W)	Contingency	Allocated Power (W)
A1.0 Payload (Total)	-	**10**	-	**13**
A1.1 People (x2)	0.0%	0	30%	0
A1.2 Mars Samples (Preservation)	0.7%	10	30%	13
A2.0 MAV Power (Total)	-	**1340.77**	-	**1743**
A2.1 Structure and Mechanisms	0.7%	9.5	30%	12.35
A2.2 Thermal Control	29.6%	400	30%	520
A2.3 Power	24.5%	331	30%	430
A2.4 Comms	12.7%	171	30%	222.3
A2.5 ODH	7.7%	104.5	30%	135.85
A2.6 ADCS	10.5%	142.5	30%	185.25
A2.7 Propulsion	3.5%	47.5	30%	61.75
A2.8 ECLSS	10.0%	135	30%	175.5
MAV Total	-	**1350.77**	-	**1756**

Fig. 3 MAV total power budget.

B. Lander One (Land-On Mars)	Power Percentage	CBE (W)	Contingency	Allocated Power (W)
B1.0 Payload (Total)	-	**60**	-	**78**
B1.1 Transfer Mechanisms (4)	12.9%	60	30%	78
B2.0 Lander Power (Total)	-	**403.5**	-	**524.5**
B2.1 Structure and Mechanisms	1.7%	7.7	30%	10
B2.2 Thermal Control	16.6%	77	30%	100
B2.3 Power	29.9%	138	30%	180
B2.4 Comms	10.8%	50	30%	65
B2.5 ODH	6.5%	30	30%	39
B2.6 ADCS	10.8%	50	30%	65
B2.7 Propulsion	7.6%	35	30%	45.5
B2.8 EDL	3.3%	15	30%	20
Lander One Total	-	**463.5**	-	**602.5**

Fig. 4 Land-On-Mars total power budget.

C. Lander Two (ISLE)	Power Percentage	CBE (W)	Contingency	Allocated Power (W)
C1.0 Payload (Total)	-	**9483**	-	**10828**
C1.1 Transfer Mechanisms (4)	0.5%	60	30%	78
C1.2 MAV	Table A	Table A	Table A	Table A
C1.3 Fission Power Unit	0.0%	0	30%	0
C1.4 PHISH Rovers (Charging)	16.3%	1923	30%	2500
C1.5 SMOXIE	63.4%	7500	10%	8250
C2.0 Lander Power (Total)	-	**2337**	-	**3038.5**
C2.1 Structure and Mechanisms	6.3%	750	30%	975
C2.2 Thermal Control	2.1%	250	30%	325
C2.3 Power	6.7%	792	30%	1030
C2.4 Comms	0.7%	80	30%	104
C2.5 ODH	0.7%	80	30%	104
C2.6 ADCS	0.4%	50	30%	65
C2.7 Propulsion	0.3%	35	30%	45.5
C2.8 EDL	2.5%	300	30%	390
Lander Two Total	-	**11820**	-	**13866.5**

Fig. 5 ISLE total power budget.

F. Mass Budget

The mass budgets for this mission were initially developed using a top down approach from similar missions and literature focused on similar proposed missions. As the mission design progressed, adjustments were made where necessary to accommodate the needs of each individual subsystem. The finalized versions of these budgets for each designed vehicle are shown below.

G. PHISH and SMOXIE

The mission architecture includes three rovers, known as PHISH, for the propellant transfer mechanism. Since NASA has sent several rovers to traverse Mars, the mechanics supporting a rover concept are well

D. PHISH Rover (Single Rover)	Power Percentage	CBE (W)	Contingency	Allocated Power (W)
D1.0 Payload (Total)	-	**17**	-	**21.5**
D1.1 Science Instruments	5.2%	5	30%	6.5
D1.2 Propellant Storage	12.0%	12	30%	15
D2.0 MAV Power (Total)	-	**79**	-	**103**
D2.1 Structure and Mechanisms	19.3%	18	30%	24
D2.2 Thermal Control	16.1%	15	30%	20
D2.3 Power	13.7%	13	30%	17
D2.4 Comms	11.2%	11	30%	14
D2.5 ODH	10.4%	10	30%	13
D2.6 Navigation Systems	12.0%	12	30%	15
PHISH Total	-	**96**	-	**124.5**

Fig. 6 PHISH total power budget.

A. MAV	Mass %	CBE (kg)	Contingency (%)	Allocated (kg)
A1.0 Payload (Total)	-	**455**	-	**611.5**
A1.1 People (x2)	7%	400	35%	540
A1.2 Mars Samples	1%	55	30%	71.5
A2.0 MAV Mass (Total)	-	**5255.7**	-	**6743.8**
A2.1 Structure and Mechanisms	32.9%	1990.0	35%	2686.5
A2.2 Thermal Control	21.0%	1266.0	30%	1645.8
A2.3 Power	11.6%	700.0	30%	910.0
A2.4 Comms and ODH	0.3%	18.6	30%	24.1
A2.6 ADCS	2.8%	171.8	30%	223.3
A2.7 Propulsion	15.8%	955.0	30%	1053.4
A2.8 ECLSS	2.6%	154.4	30%	200.7
A3.0 Total Dry Mass	-	**5710.7**	-	**7355.3**
A5.0 Other (wires, etc.)	5%	332.0	30%	431.6
MAV Total	**100%**	**6042.7**		**7786.9**

Fig. 7 MAV total mass budget.

established. The use of one rover results in a major point of failure, however, as any system failure of the rover could result in an incomplete transfer process, leading to a scrubbed MAV launch. The mitigation of this risk was accomplished by proposing the use of multiple rovers for the propellant transfer. When performing calculations for two rovers, however, there were still major concerns that arose. The first included the added mass of the propellant to the structure of the rover. These combined weights were far heavier than any previous NASA rover, which resulted in concerns about the wear and tear on the treads and other rover mechanisms while carrying such a heavy load. Additionally, the same issue remained concerning the single point of failure: if either of the rovers malfunctioned, the entire timeline for the mission would be pushed back. Ultimately these issues were resolved by adding an additional rover for a total of three PHISH. This mitigated the risk of moving a large mass across the surface of Mars by reducing the overall weight of each

B. Land-On-Mars	Mass %	CBE (kg)	Contingency (%)	Allocated (kg)
B1.0 Payload (Total)	**-**	**7833**	**-**	**10574.6**
B1.1 Propellant for Transfer	42.7%	7803	35%	10534.1
B1.2 Transfer Mechanisms	0.2%	30	35%	40.5
B2.0 Lander Dry Mass (Total)	**-**	**10422.7**	**-**	**13090.5**
B2.1 Structure and Mechanisms	31.2%	5700.0	35%	7695.0
B2.2 Power	3.7%	674.0	30%	876.2
B2.3 Comms and ODH	0.2%	28.7	30%	37.3
B2.4 ADCS	0.5%	100.0	30%	130.0
B2.5 Thermal Control System	3.3%	600.0	30%	660.0
B2.6 Propulsion Subsystem	1.1%	200.0	30%	260.0
B2.7 EDL	17.1%	3120.0	30%	3432.0
B3.0 Total Dry Mass	**100%**	**18255.7**	**-**	**23665.0**
B4.0 Propellant	-	13702.1	30%	17812.7
B5.0 EDL (ejected before landing)	-	11000.0	30%	14300.0
B6.0 Other (wires, etc.)	-	785.4	30%	1021.1
Total Landed Mass	**-**	**19041.1**	**-**	**24686.1**
Lander Total	-	43743.2	-	56798.8

Fig. 8 Land-On-Mars total mass budget.

C. ISLE	Mass %	CBE (kg)	Contingency (%)	Allocated (kg)
C1.0 Payload	**-**	**13486.9**	**-**	**13086.9**
C1.1 Rovers	3.4%	731.6	0%	731.6
C1.2 MAV (Dry Mass)	33.8%	7355.3	0%	7355.3
C1.3 Fission Power Unit	23.0%	5000.0	0%	5000.0
C1.5 MOXIE	1.8%	400.0	30%	520.0
C.2 Lander Dry Mass	**-**	**8280.6**	-	**11049.7**
C2.1 Structure and Mechanisms	26.2%	5700.0	35%	7695.0
C2.2 Thermal Control	2.8%	600.0	30%	780.0
C2.3 Power	3.2%	700.0	30%	910.0
C2.4 Comms and ODH	0.1%	20.6	30%	26.7
C2.5 ADCS	0.5%	100.0	30%	130.0
C2.6 Propulsion Subsystem	0.9%	200.0	30%	260.0
C2.7 EDL	4.4%	960.0	30%	1248.0
C3.0 Total Dry Mass	**100%**	**21767.5**	**-**	**24136.6**
C4.0 Propellant	-	13975.1	30%	18167.6
C5.0 EDL (ejected before landing)	-	13160.0	30%	17108.0
C6.0 Other (wires, etc.)	-	663.0	30%	861.9
Total Landed Mass	**-**	**22030.4**	**-**	**24998.5**
Lander Total	-	49165.6	-	60274.1

Fig. 9 ISLE total mass budget.

component, and it addressed the single point of failure by establishing a timeline where one PHISH could fail upon deployment and the other two would still have enough capacity to complete the transfer process within the timeline of the mission. The calculations performed for this final architecture are shown in Figure 11.

D.Rover (PHISH)	Mass %	CBE (kg)	Contingency (%)	Allocated (kg)
D1.0 Payload	-	**45.0**	-	**58.5**
D1.1 Science Instruments	5.7%	10.0	30%	13.0
D1.2 Propellant Storage	20.1%	35.0	30%	45.5
D2.0 Rover Dry Mass		**129.6**		**174.9**
D2.1 Structure and Mechanisms	20.1%	35.0	35%	47.3
D2.2 Thermal Control	16.0%	28.0	35%	37.8
D2.3 Power	25.2%	44.0	35%	59.4
D2.4 Comms and ODH	7.2%	12.6	35%	16.9
D2.5 ADCS (Navigation Systems)	5.5%	10.0	35%	13.5
D3.0 Total Dry Mass	**100%**	**174.6**	-	**233.4**
D4.0 Other	-	7.3	30%	10.4
Total	-	**181.8**	-	**243.8**
Total for 3 Rovers	-	**545.5**	-	**731.4**

Fig. 10 PHISH total mass budget.

The other main architecture element is the SMOXIE, which will be used to produce liquid oxygen on the surface of Mars. Calculations were initially performed to address bringing the oxidizer to Mars, however the added mass to the system was so large that this quickly proved to be infeasible with the lander mass constraints. This led to the idea of using a scaled version of MOXIE, a device that has proven to successfully produce oxygen on the surface of Mars. The specifications for this device were driven by the required amount of LOX for the MAV launch as well as NASA predictions for a future, larger MOXIE based off the success of the currently operational device. Concerns surrounding depreciation of the device and need for inoperative times led to the architecture requiring two devices to cycle on and off for the duration of the mission. The calculations for the SMOXIE devices are shown in Figure 11.

PHISH and SMOXIE Calculations	
Total Travel Time for 2 km	13 Hours
Total Time for Propellant Transfer	2 Hours
Total PHISH Propellant Capacity	50 Liters
Total Number of Trips per Rover	70 Trips
Total Amount of LOX Required	22000 kg
Time Required to Produce LOX	18 Months
SMOXIE Total Mass	1040 kg
SMOXIE Total Power Consumption	8 kW

Fig. 11 PHISH and SMOXIE utilization calculations.

IV. Attitude Control

A. Control Modes and Pointing Requirements

Each spacecraft in the system have control modes. These modes are described below.

- Descent and Landing Mode (ISLE and Land-On-Mars): Actuators on the landers are operating to keep the landers oriented normal to the surface of Mars.
- On Surface Normal/Nominal Mode (ISLE): ISLE is sending data to the DST for transmission back to Earth.
- Launch Mode (MAV): Launch and orbit insertion are occurring. Actuators are completing TCMs as necessary to launch towards the desired 5-sol orbit for rendezvous with the DST, and status updates are being transmitted to the DST.
- Rendezvous Mode (MAV): Actuators are orienting the MAV for rendezvous with the DST, and telemetry data is being sent to the DST.
- Safe Mode (MAV): Telemetry data and status updates are being sent to the DST, and life support systems are on.

Within these control modes, the related systems have pointing requirements for accuracy. The values of these pointing requirements were determined by the related system architecture and situational requirements. These pointing requirements are demonstrated in the table below [4].

Table 3 Control modes and pointing requirements for all vehicles.

Control Mode	**X-Axis Pointing Requirement**	**Y-Axis Pointing Requirement**	**Z-Axis Pointing Requirement**
Descent and Landing Mode (ISLE and Land-On-Mars)	Actuators: 0.75°	Actuators: 0.75°	Actuators: 0.1°
On Surface Normal/Nominal Mode (ISLE)	Comms: 3°	Comms: 3°	Comms: 3°
Launch Mode (MAV)	Comms: 0.3° Actuators: 0.75°	Comms: 0.3° Actuators: 0.75°	Comms: 0.3° Actuators: 0.75°
Rendezvous Mode (MAV)	Comms: Actuators: 0.01°	Comms: Actuators: 0.01°	Comms: Actuators: 0.01°
Safe Mode (MAV)	Comms: 0.1°	Comms: 0.1°	Comms: 0.1°

Some of these modes require no rotational maneuvers to exist correctly; for example, the antenna on ISLE is omnidirectional, and thus needs little help from the ADCS system to fulfill its pointing requirement.

However, required rotational maneuvers for the remaining modes include reorientation burns for the landers during entry, landing, and descent, TCMs to adjust the launch trajectory of the MAV towards the DST, and reorientation burns for the MAV during rendezvous with the DST.

B. Torques

There are a large number of torques that affect the MAV while it is in motion. Externally, these include the Martian gravity gradient, atmospheric drag, and solar radiation pressure, which can be calculated using equations 1, 2, and 3, respectively; the values required to calculate these torques come from physical constants of Mars and constants related to the worst possible cases the MAV may be subject to in its orbit.

$$T_g = \frac{3\mu}{2R^3}|I_z - I_x|sin(2\theta) \tag{1}$$

$$T_a = (\frac{1}{2}\rho V^2)C_d A(c_{pa} - cg) \tag{2}$$

$$T_s = \frac{SAcos(i)}{c}(1+q)(c_{ps} - cg) \tag{3}$$

Internally, these include sloshing, uncertainty with the center of mass, and thruster misalignment. The magnitude of these torques is stated below [4].

Table 4 Magnitudes of external and internal torques.

Torque	**Value**
Gravity Gradient	$3.11 * 10^{-4}$ Nm
Atmospheric Drag	$9.694 * 10^{-4}$ Nm
Solar Radiation Pressure	$2.23 * 10^{-4}$ Nm
Sloshing	on the scale of 10^{-4} Nm
Uncertain Center of Mass	on the scale of X cm
Thruster Misalignment	0.3°-0.5°

C. MAV Mass Characteristics

The MAV has a particular set of mass characteristics. The center of mass of the vehicle, from the bottom of the vehicle, is [3.78, -0.85, 6.15] m. Additionally, the moment of inertia matrix is $\begin{bmatrix} 2.07 & -0.16 & 0.995 \\ -0.16 & 2.32 & -0.23 \\ 0.995 & -0.23 & 1.04 \end{bmatrix}$

* 10^6 kg*m^2. Through calculation, it was determined that the required momentum storage for the MAV is approximately 7.22 Nms, using equation 4.

$$h = FNP(T_a + T_s + \frac{\sqrt{2}}{4}T_g) \tag{4}$$

Knowing this, the MAV requires about 0.017 Nm torque for the 7 minute launch and 3.26*10^{-5} Nm torque for the entirety of the rendezvous with the DST to desaturate the acrewed momentum; this was calculated using equation 5.

$$F = frachLt \tag{5}$$

Additionally, to desaturate the MAV from these internal torques, desaturation maneuvers are required. It was determined that these maneuvers would have a $Delta$V of approximately 2-5 mm/s. To complete this, the required fuel amounts to 154.63 kg, calculated using equation 6

$$m_{prop} = \frac{2Ft}{g_0 I_{sp}} \tag{6}$$

D. ADCS System Architecture

1. Landers

The ADCS system in the landers is primarily used for entry, landing, and descent. As such, it needs sensors capable of gathering position data relative to the surface of Mars, rather than relative to objects in space like most sensors. As such, the landers contain an IMU and 3 optical navigation cameras for position determination; the IMU gathers position data using the accelerometer, and the optical navigation cameras gather initial position data to supply to the IMU and supplemental data as redundancy for the IMU accelerometer measurements. The landers also have such a large mass that it is unlikely that a control moment gyro will be capable of making the trajectory adjustments required during entry, landing, and descent. As such, the landers also contain 6 110N hydrazine monopropellant thrusters, with 2 thrusters aligned along each axis for redundancy.

2. *MAV*

The ADCS system in the MAV is primarily used for launch and for rendezvous with the DST in the 5-sol orbit. In this case, sensors fixed on other celestial bodies can be used, and in most cases are more efficient. As such, the primary sensor used on the MAV is a star tracker to gather relative position to nearby bodies. An IMU is additionally used to supplement the position data gathered by the star tracker and to gather acceleration data from launch. Again, the MAV has such high mass that it is highly unlikely that the gyroscopes in the IMU will be capable of making the magnitude of trajectory adjustments required for launch, and as such, thrusters are required. Again, 6 110N hydrazine monopropellant thrusters were used with 2 thrusters aligned along each axis for redundancy. During rendezvous, however, when the maneuvers need to be smaller and more precise, the gyroscopes in the IMU will be the primary actuator. An image of the ADCS system in the MAV is shown below.

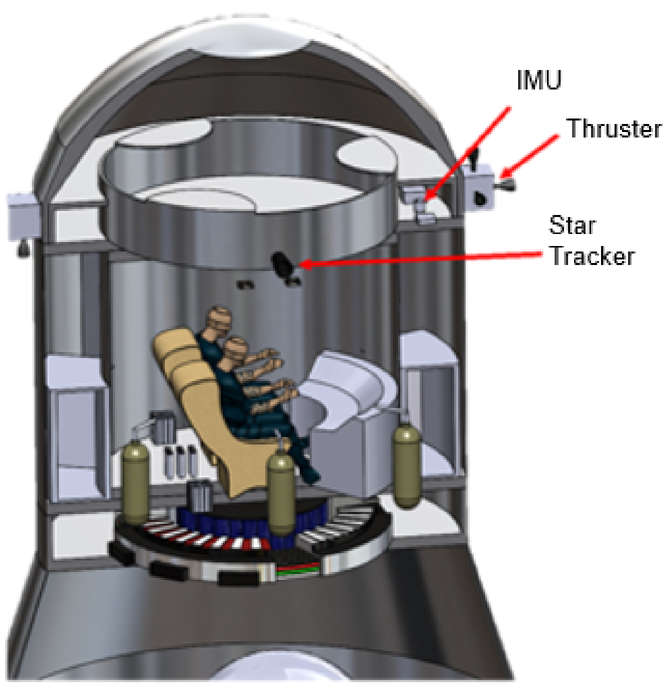


Fig. 12 ADCS system within the MAV.

3. *Trades*

Many different sensors and actuators were considered for these vehicles before reaching the final architecture. In terms of actuators, control moment gyros, momentum wheels, and thrusters were all considered. Both the gyros and the momentum wheels have a higher degree of accuracy than the thrusters, but require greater power and are greatly affected by the mass of the system. Thrusters, however, do not require power and supply greater force than the other two systems and as such, are less affected by mass. In terms of sensors, primarily IMUs and star trackers were considered. An IMU was not determined as the primary sensor for the MAV because it is less accurate in position determination long term, as it only uses an accelerometer for position determination and requires frequent supplementary position data to ensure accuracy. However, it was most efficient for the landers because they only need to land oriented correctly.

V. Environmental Control and Life Support Systems

The Environmental Control and Life Support System (ECLSS) is applicable to the MAV, as that is the only vehicle which will need to support human life. The main human needs which were accounted for in the ECLSS system are water supply, oxygen, pressurization, waste elimination, radiation shielding, CO2 removal, dust/contaminant mitigation, and food. In this section, the choices for each of these categories and the methodology used to make those choices will be summarized.

ECLSS Design Alternatives			
Water Supply	Stored Water	Stored Water + Recycling	Fuel Cells
Oxygen Supply	Stored Oxygen (cryogenic)	Stored Oxygen (high pressure)	Collect from SMOXIE
Cabin Pressure	High Pressure N2	Cryo-stored N2	Pure Oxygen
CO2 Removal	Regenerable	Non-regenerable	
Food	Meals Ready to Eat (MREs)	Dehydrated food	Hydroponics
Waste	Ejection from MAV	Store in waste bags	
Radiation Shielding	Metallic shield	Water layer	
Dust/Contaminants	Ventilation with filter	Dust suit (removed before entry)	

Fig. 13 ECLSS morphological matrix.

A. CO2 Removal

There are two main categories of CO2 removal systems: regenerable and non-regenerable. Regenerable methods include Molecular Sieves, Solid Amine Water Desorption, and Electrochemical Depolarization

Concentrators [5]. Non-regenerable methods include Lithium Hydroxide, Sodasorb, and Superoxides [5]. For a short term mission, a non-regenerable option is preferred since the mass of the system needed exceeds the mass saved from using recyclable material [6]. Within the non-regenerable category, Lithium Hydroxide canisters were chosen as they have flight heritage in the Apollo program. 2.28 kg/CM-day of LiOH are required [6], which leads to a total of 17.8113 kg with 30% margin. This amount will be split into 8 canisters, each containing 2.227 kg of LiOH. There will be 2 units capable of holding the cartridges placed at opposite ends of the cabin for redundancy. This allows the canisters to be exchanged at an offset to prevent large fluxes in CO2 concentrations.

B. Water

Following the same logic as outlined above, a simple water tank is preferable over a recycling system on a short duration mission. Additionally, the oxygen, hydrogen, and general architecture needed for a fuel cell would not be worth the water produced when the water needed is only 2.7kg/CM-day [6]. This water amount assumes minimal hygiene use, which is applicable for a short mission.

C. Oxygen

The astronauts require 1kg/CM-day of oxygen [6]. Three options for providing this oxygen include storing it in a cryogenic tank, storing it in a high pressure tank, and using some oxygen from the SMOXIE unit which is producing liquid oxygen for the propulsion system. Due to the thermal requirements and potential leakage of a cryogenic tank as well as the relatively small amount of oxygen needed, a high pressure tank was selected. This is also preferable to the SMOXIE option because it prevents the need for a complicated transfer mechanism and keeps the propulsion system isolated from ECLSS. Given a 50% margin, 9 kg O2 are required. Following an iterative process to allocate mass and volume around the MAV for balancing, the total oxygen will be split into 3 tanks, kept at 34.474 MPa (5000 psi) each [7]. At this pressure, a carbon fiber composites tank is the optimal material selection [7]. In addition to the oxygen tanks, one oxygen candle will be stored on board for emergencies.

D. Cabin Pressure

For a cabin pressurized by nitrogen, the same two tank options apply as for oxygen: high pressure and cryogenic. Alternatively, the cabin pressurization could be supplied entirely by a pure oxygen environment.

However, as demonstrated in the Apollo 1 fire, this produces an immense flammability risk. Another consideration is the astronaut transfer between the DST and the MAV. If the DST is to be kept at similar conditions to the ISS (21% oxygen) and the spacesuits are kept at 100% oxygen, the MAV can serve as a transition period for the 72 hour trip [6]. Taking the average of the two yields an oxygen-nitrogen mix of 60/40. The cabin pressure will be kept nominally at 56.5 kPa, which follows the guidelines for a Mars lander outlined by NASA's Exploration Atmospheres Working Group in 2005 [8]. Finally, the nitrogen will be stored in a composite high pressure tank following the same reasoning as oxygen for avoiding a cryogenic tank when possible.

E. Food

Three options for feeding the astronauts during their trip to the 5-sol orbit are MREs, dehydrated food, and a hydroponic/plant system. While a hydroponic system would be useful for providing fresh food and providing other resources such as oxygen, it would require substantially more mass and volume than the other alternatives. Therefore, it is not feasible for a 3 day mission. Between MREs and dehydrated food, Fig. 14 compares the mass of each, given that dehydrated meals are lighter but also require a rehydration system [9]. Since 20 meals are needed, the MREs were chosen.

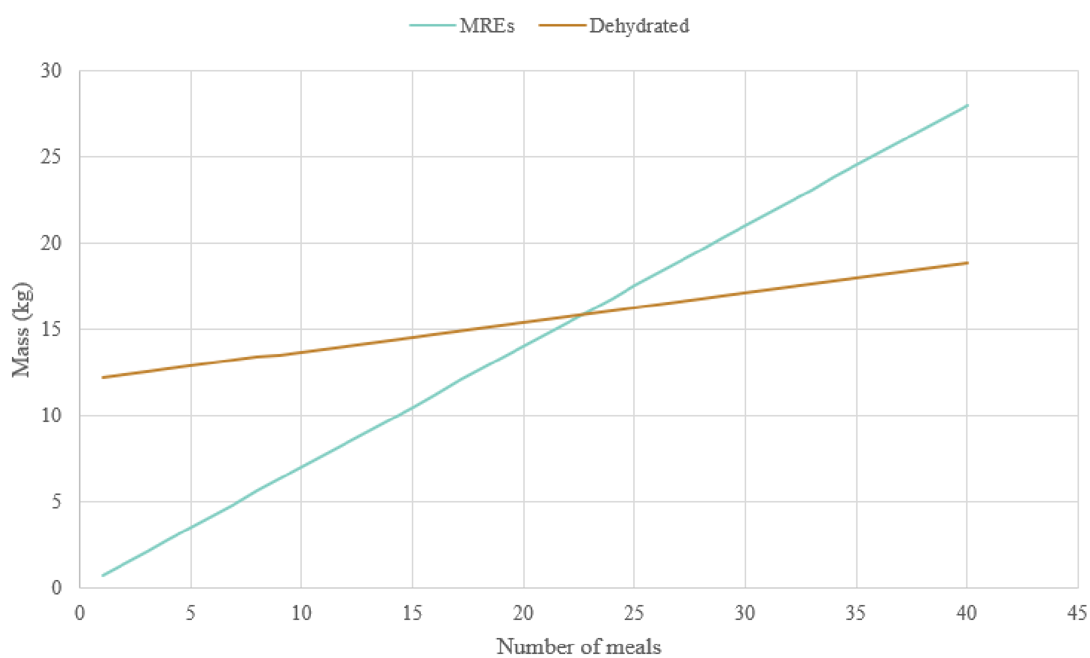


Fig. 14 Masses of MREs vs dehydrated food.

F. Waste

The two options for waste removal are ejection from the MAV and storage inside the MAV. For a 72 hour mission, storage is preferable to reduce the complexity of the system. The waste bags will include a liquid germicide to reduce the chance of contamination.

G. Radiation Shielding

Two options for mitigating the astronauts' radiation exposure in the 72 hours they are travelling in the MAV are a metallic shield and a water shield. Water is a relatively strong radiation blocker, and is convenient due to its alternative use for crew consumption [10]. However, the water required to be carried in this mission is small (21.06 kg), which when spread over an area of 25.5 m^2 (the surface area of the cabin) would only be 0.0823 cm thick. Given that a thickness of at least 7 cm is required to be an effective shield, 84.7 times the amount of water carried would be needed [10]. When compared with the mass of a 6mm aluminum layer, the metallic shield makes more sense [11]. Additionally, the metallic shield is much easier to implement and has less failure modes than an encapsulating water tank.

H. Dust/Contaminants

Two options for mitigating the dust circulating in the MAV cabin are having fans with ventilation filters attached and providing the astronauts with a protective dust garment that they will wear while exploring the surface and then remove prior to entry. As both these options are feasible and beneficial, both will be implemented in the design.

I. ECLSS Summary

Fig. 15 shows the summary of the choices made for the ECLSS subsystem. Given the quantities described in this section, the total mass budget for ECLSS is 200.72 kg with a 30% contingency, and the power budget is 175.5 W with a 30% contingency.

ECLSS Design Alternatives			
Water Supply	Stored Water	Stored Water + Recycling	Fuel Cells
Oxygen Supply	Stored Oxygen (cryogenic)	Stored Oxygen (high pressure)	Collect from SMOXIE
Cabin Pressure	High Pressure N2	Cryo-stored N2	Pure Oxygen
CO2 Removal	Regenerable	Non-regenerable	
Food	Meals Ready to Eat (MREs)	Dehydrated food	Hydroponics
Waste	Ejection from MAV	Store in waste bags	
Radiation Shielding	Metallic shield	Water layer	
Dust/Contaminants	Ventilation with filter	Dust suit (removed before entry)	

Fig. 15 ECLSS completed in morphological matrix.

VI. Structures and Configurations

A. Land-On-Mars, Isle

Each lander will be launched via a separate SLS launch. Each lander is designed to fill the space within SLS block 2. The Landers have a dry mass of 10,422.680 kg (Land-On-Mars) and 10,440.550 kg (ISLE) as well as a landed mass of 19041.109 kg (Land-On-Mars) and 24321.4844 kg (ISLE) with a mass contingency of 30%. Due to the similar landed mass of these vehicles, the same overarching structural design will be used. For the landers, it is a valid assumption that the radius is much larger than the wall thickness. Therefore, the computations and engineering assumptions for a thin walled pressure are valid, so the basic equations for a thin walled pressure vessel will be used. Constraining the stress below 65 MPa was implemented as aluminum has poor fatigue properties and an effort to keep the max stress below a known fatigue limit of 10 million cycles [**?**]. The Landers center of Gravity (CG) was designed to be in front of the Center Of Pressure (COP) to maintain a stable system, it was also designed to be co-linear so there are no off axis moments produced during launch, to simplify the wiring hardness installation systems that rely on each other are placed near each other.

The thin wall was calculated using the following method: assuming an internal pressure of 1 ATM (101325 Pa) and Mars's surface pressure (external pressure) of [12] 600 Pa the radial stress is determined by

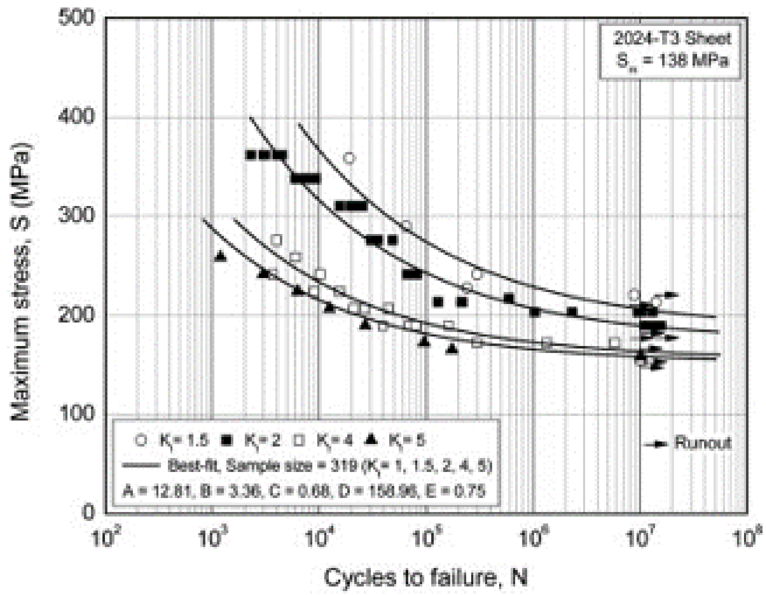


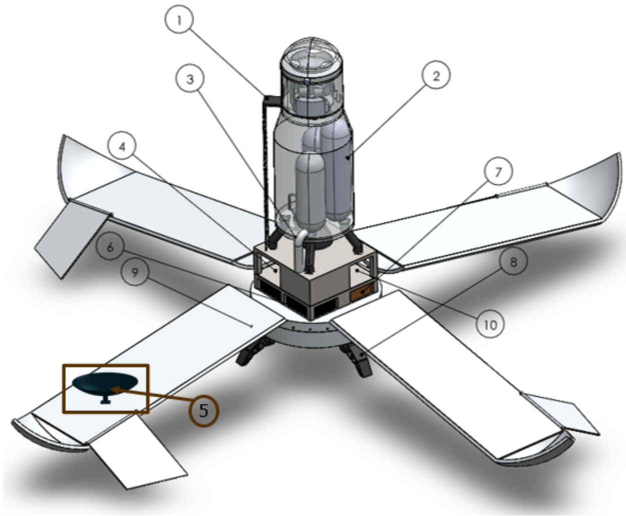
Fig. 16 2024-T3 fatigue analysis.

$$radial\ stress = \frac{R_i^2 p_i - R_e^2 p_e}{R_e^2 - R_i^2} - \frac{1}{r^2}\frac{(p_i - p_e)R_i^2 R_e^2}{R_e^2 - R_i^2} \tag{7}$$

As radial stresses are the landers' major mode of failure, this stress component will be the designing constraint. Ensuring the radial stress is below the set target of 65 MPa will guarantee structural integrity of the landers. The wall thickness needed was computed to be 0.067 m with an additional 0.03 m to withstand the abrasive sand storms on mars as shown in Table 5. Because of the additional abrasive layer, the stiffness of the landers will increase. By increasing the stiffness of the lander structure this also increases their natural frequency, so the frequency produced by the rockets are unable to excite the modal failure modes of the structure.

Table 5 Lander design parameters and specifications.

Height	27 m
Wall Thickness	0.067m
Abrasive wall thickness	0.03m
Structural Material	Aluminum 2024-T3
Structural Mass	10,422.680 / 10,440.550 kg
Landed Mass	19041.109 / 24321.4844 kg
Volume	1187 m^3
Peak Load	61 MPa
Peak deflection	0.0197 m



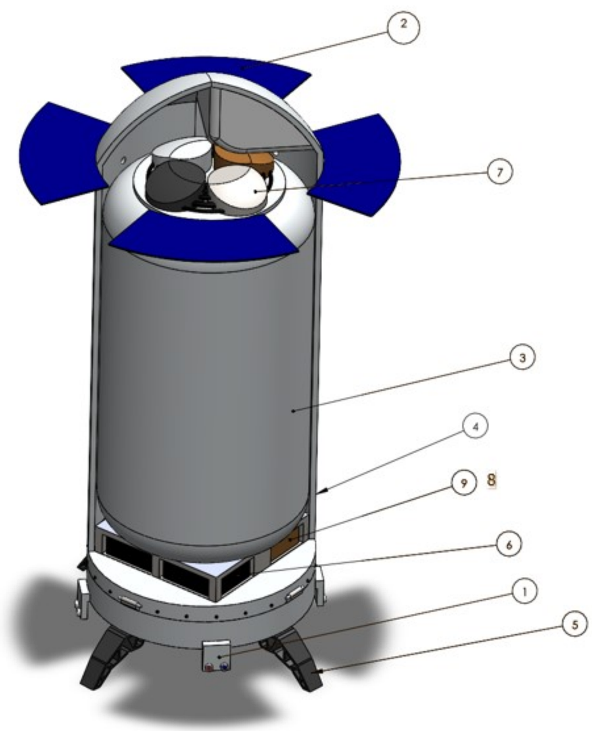
(a) Petal configuration

1	SCAFFOLDING/ LADDER	Welded	Supporting Structure for MAV
2	MAV	Exploding bolts and quick disconnects	MAV feet / lander surface
3	SMOXIE/FUEL TANK PIPE	Valve	MAV Tanks/SMOXIE Valve
4	PHISH	Exploding bolts	Lander Shell wall / PHISH Wheels
5	X-BAND ANTENNA	Bolts	Lander Walls
6	FISSION REACTOR	Fixed to floor in Secure compartment	Secure Floor
7	SMOXIE	Fixed to floor	Secure Floor
8	LANDER FEET	Hinge	Lander Outside wall
9	PETALS	Hinged to Lander base	Lander base / edge
10	BATTERY CELLS	Bolted	Lander Walls / Fuel cell Walls

(b) ISLE interfaces

Fig. 17 ISLE deployed, on-surface configuration.

To allow for easy refueling and launching strategy the ISLE incorporates a petal landed configuration which allows the MAV a stable platform to launch.



(a) LAND-ON-MARS configuration

Reference	Attachment	Method	Interfacing Surfaces
1	PHISH FUEL CONNECTION	High pressure hose deliver	Lander Shell wall / PHISH Wheels
2	SOLAR PANELS	Hinge	Lander shell solar panel edge
3	FUEL CELLS	Bolted	Lander Walls / Fuel cell Walls
4	LANDER SHELL	Welded	Lander base
5	LANDER FEET	Hinge	Lander Outside wall
6	FISSION REACTOR	Fixed to floor in Secure compartment	Secure Floor
7	COMMS	Threaded to fuel cell	Secure Floor
8	BATTERY PACK	Bolted	

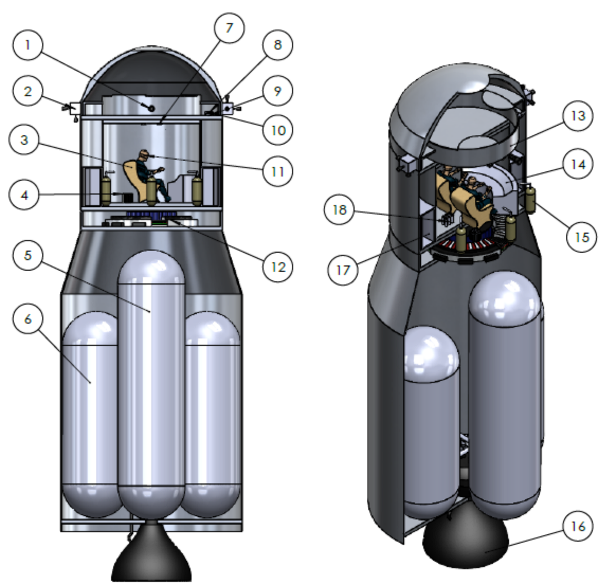
(b) LAND-ON-MARS connections

Fig. 18 LAND-ON-MARS landed configuration.

The proposed landed configuration of the propellant lander is pictured in Figure 18, a low CG and a wide base were designed so that, in the event of high winds, the propellant lander would not fall over potentially damaging fuel cells. The abrasive wall thickness is also implemented on the propellant lander so the fuel can survive on Mars for the duration of the mission.

B. Mars Ascent Vehicle (MAV)

The Mars Ascent Vehicle (MAV) is a rocket that will be used to transport samples and a crew of two from the surface of Mars to a 5-sol parking orbit, where it will rendezvous with the Deep Space Transit (DST). A diagram of the MAV with locations of different subsystems are shown in Figure 19. It has a structural mass of 2686.5 kg and a total mass of 7602.525 kg, as shown in Table 6. The peak power consumption of the MAV's structural components is 12.35 W. The mass budget of structures and mechanism includes a 35% contingency. Power consumption for the MAV includes three phases - launch, docking, and travel. All three phases require 12.35 W of power.



(a) Balloon diagram of the MAV

1	Star Tracker
2	ADCS Thruster
3	Seat
4	CO2 Remover
5	Fuel Tank
6	LOX Tank
7	Fan
8	Onboard Computer
9	Antenna
10	IMU
11	Crew
12	Power
13	Docking Mechanism
14	Instrument Panel
15	ECLSS Tank
16	Engine
17	Payload Compartment
18	CO2 Canister

(b) Subsystems within the MAV

Fig. 19 Configuration of the MAV.

The first consideration when configuring the MAV is the symmetry of where each individual component is placed. Symmetry minimizes disturbance torques and simplifies ADCS design. Another consideration is placing large and heavy components near aft. For example, the fuel and LOX tanks are the largest and heaviest

Table 6 MAV specifications.

Height	12.6 m
Structural Material	Aluminum 2024-T3
Yield Strength	344.738 MPa
Structural Mass	2686.5 kg
Total Mass	7602.525 kg
Total Volume	637.743 m^3
Peak Power (Structures)	12.35 W

parts of the MAV. Placing them towards the bottom of the spacecraft minimizes bending moments at launch, allowing higher natural frequencies of vibration and reducing structural mass. Furthermore, to reduce the mass of wires, subsystems that require wiring are placed near each other. For example, ADCS sensors, ODH, and communication systems are all located close together. Since control sensors need a stiff and thermally stable surface, they are mounted in the nose cone of the MAV. Additionally, the star tracker is placed in a location where its view will not be blocked by other components. Doing so reduces pointing errors.

The MAV will be transported by the ISLE lander. Hold-down arms and explosive bolts will be used to secure the MAV on a platform in ISLE [13]. The hold-down arms and explosive bolts detach from the MAV during the launch sequence, which are shown in Figure 20. Mass distribution throughout the MAV is also a key factor in keeping the MAV balanced while stored in ISLE.

Fig. 20 MAV with hold-down arms and explosive bolts.

To accommodate every subsystem, the MAV is 12.6 meters tall with an overall volume of 152.6 m^3. It has three different sections of varying diameters, shown in Table 7. The section of the MAV holding the cabin is 3.6 meters in diameter. Having a smaller diameter reduces overall structural mass of the MAV and empty space inside the MAV. The load applied to this section of the MAV was derived from the decibels of a rocket launch, which is 204 dB according to NASA's measurements of the Saturn V launch [14]. The number of decibels was converted into sound pressure in Pascals using the following equation

$$SPL = 20 \times log(\frac{P}{P_{ref}}) \tag{8}$$

where SPL is the sound pressure level in dB, P is the sound pressure in Pascals, and P_{ref} is the reference sound pressure of 0.00002 Pa. The resulting sound pressures, calculated using Equation 8, are shown in Table 7. However, the sound pressure experienced by the MAV will be less than Saturn V due to the engine of choice. The wall thickness was determined by performing a pressure vessel test. The resulting wall thickness and factor of safety are shown in Table 7.

The same method was used to test the 3.6 to 4.6 m diameter section of the MAV, as well as the 4.6 m diameter section. The wall thickness column of Table 7 shows that as the diameter of the MAV increases, the wall also needs to be thicker. As for the cabin, 1 ATM (101325 Pa) was applied when conducting the pressure vessel test because the Earth's surface atmospheric pressure is that value [15]. This is to ensure the crew's comfort during their trip. The wall thickness of the cabin is shown in Table 7.

Table 7 Table of loads applied to MAV walls and the resulting thicknesses.

Section	Applied Load	Wall Thickness (m)	Von Mises (MPa)	Displacement (mm)	FOS
3.6 m	316979 Pa	0.017	56.56 MPa	1.118	1.3
3.6 to 4.6 m	316979 Pa	0.01	60.01 MPa	0.3	1.3
4.6 m	316979 Pa	0.03	54.96 MPa	1.561	1.4
Cabin	1 ATM (101325 Pa)	0.005	20.55 MPa	0.259	3.7

The MAV's main structure is made of Aluminum 2024-T3 due to its high yield strength of 344.738 MPa. The primary alloying element in Aluminum is copper [16]. The copper content in the alloy contributes to its high yield strength, meaning that the material will not permanently deform under high stress. This is important because the MAV will be under high stress conditions during launch. The good machinability and ease of surface finishing of Aluminum 2024-T3 also makes it the ideal material for the outer structure of the MAV [17]. Furthermore, Aluminum 2024-T3 has a high strength to weight ratio, which aids in weight reduction without sacrificing structural integrity of the spacecraft [16]. Additionally, Aluminum 2024-T3 has high fatigue resistance, meaning that the MAV can withstand the many deflection cycles experienced at launch [18].

VII. Thermal Control

To provide adequate thermal protection to all components of the mission, adequate environmental characterization must be in order. In line with prior Martian missions throughout the past four decades as noted in [19] and [20], the martian environment has been thoroughly characterized from many aspects. Of particular importance are the temperature ranges on the surface, averaging from -100K to 200K with a pressure of roughly 80 Pa at the surface [21] with periodic dust storms to boot. Notwithstanding, the surface characteristics are not as extreme as the re-entry conditions encountered in the Entry, Descent, and Landing phase.

The thermal protection system is first developed for the Mars Ascent Vehicle (MAV), as its transit from the surface to the aforementioned parking orbit signifies passage into space for which the vehicle must be thermally prepared for. To estimate the internal and external temperatures of the MAV for a hot and cold case, the two-node evaluation presented in [22] provides a useful framework supplanted by the respective code provided in the AE4342 Canvas Page by Prof. Romero-Calvo. The properties of interest are listed in the below table and figure. Note that a key assumption is that the MAV is spherical; the internal and external geometries of the MAV were evaluated and extrapolated to spherical equivalency.

Table 8 Unique parameters used in calculation of nodal analysis.

Variable	Value
S	50 m^2
F_{sp}	0.27
J_s	590 W/m^2
a	0.25
T_p	210 K
R_{int}	1.9947 m
R_{ext}	1.75 m

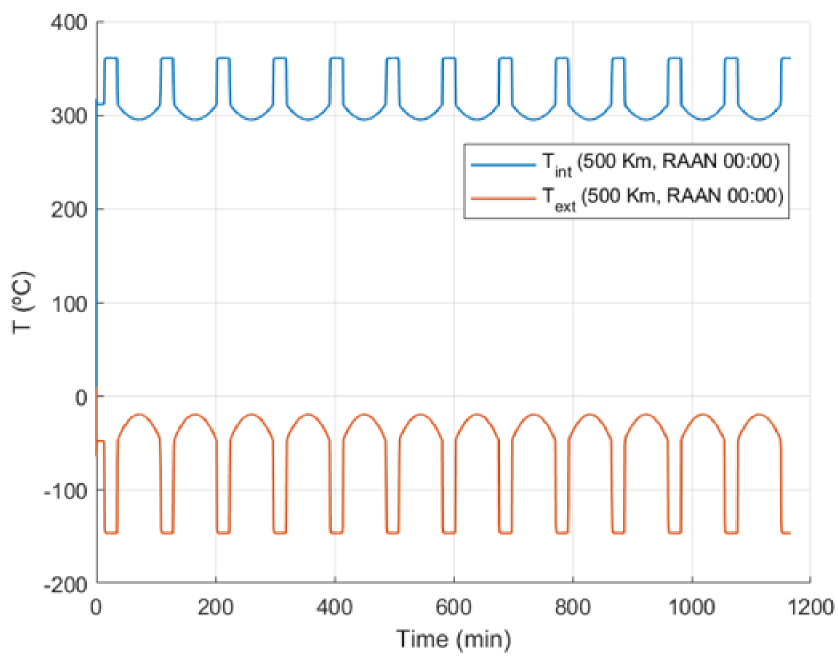


Fig. 21 Nodal analysis for the MAV.

With these results in mind, notably the 400K difference in internal and external temperature, a physical implementation scheme may be developed. In particular, the cryogenic storage tanks would require special attention due to their required 5 W/cm emitted heat flux in order to maintain favorable operation temperatures [22]. By employing Multilayer Insulation (MLI) as described in the below table and through the implementation and paremeterization in [23], this important part of the MAV may be adequately specified as using 10 layers of insulation to provide the required heat flux.

Table 9 MLI properties.

MLI Layer Density	20 layers/cm
MLI Mass Density	.023 kg/m^2
VCS Mass Density	2.0 kg/m^2
Absorbtivity/emissivity ratio	.4/.09

The supplementary methods to be used in the Thermal Protection System will be louvers, heat pipes, and radiators where best specified at the sub component and part level. The following figure offers a graphic for a lover and its placement radially along the MAV, with a projected heat flux of 150 W/m^2. A passive method used throughout both the MAV and landers is the Chemglaze A276, which has a ratio of emissivity to absorbtivity of .25/.88 [24], allowing for conjugate operation with the aforementioned active methods.

The thermal control for the fission power unit will employ louvers and heat pipes to provide adequate cooling to its components when operating on the surface and during flight. The PHISH modules benefit from the same technology, yet in a downscaled implementation with proliferation of passive methods such as conductive strips and reliance on adequate dust-resistant multilayer insulation for surface operation.

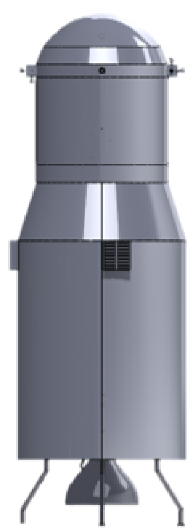


Fig. 22 MAV rendering showing lover.

A. Entry, Descent, and Landing

As a corrolary to the Thermal Control Section, the Entry, Descent, and Landing (EDL) phase is described with emphasis to phases and components.

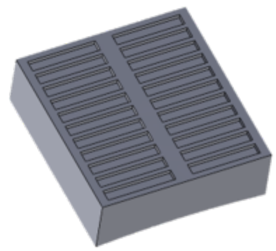
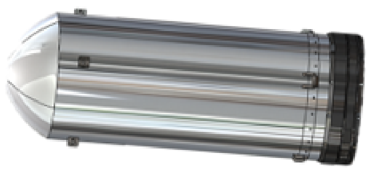


Fig. 23 Louver detailed model.

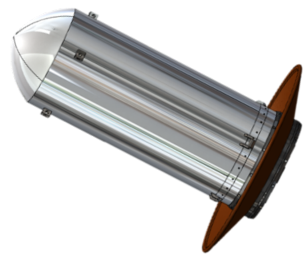
Both landers, weighing at 55,000 and 60,000 kg (approx), currently exceed the record for heaviest landed object on Mars. To best allot for chances of success in EDL whilst significantly reducing the landing weight, a shallow, long-duration, lower-velocity entry is employed which uses hypersonic parachute technology described in [25] to effectuate a landing void of heavy Retropropulsion in the final stages. The procedure is described as follows, which is roughly identical for both landers with modifications to compensate for mass effects. The overarching principle is to use a HIAD with 15m diameter along with a heat shield of nose radius 10m with overall diameter of 8.4m as the primary components. The ballistic coefficient is approximately 261 kg/m^2 in accordance with [26] which yields a drag coefficient of 1.3 with a 300kg/m^3 packing density. The weight of the Aeroshell is estimated to be approximately 11,000 kg and the HIAD another 1000kg based on SIRCA ablation material specifications and geometric optimization. Note altitudes are listed above ground level. Also, RCS thrusters will provide necessary corrective thrust during the entire phase, with distinct emphasis during phases 5 and 6.

- Phase 1: Arrival and Entry at 133km Entry and 5km/s velocity, to meet the Atmosphere at 120km Altitude at a 2-degree angle of attack. Propulsive force to achieve this will be applied prior, with the respective rockets and fuel tanks jettisoned.
- Phase 2: Cruise and Peak Heating for some 10 minutes with a 3-minute Peak Heating Duration with a heat flux of 50 W/cm^2 at the base of the heat shield, with allocation for 75 W/cm^2 under extreme conditions .
- Phase 3: Deploy HIAD at L/D=.25 at 100km altitude at M 4.5
- Phase 4: Jettison Heatshield and HIAD at 25km altitude, deploy chutes at 10km at M 3 and at a dynamic pressure of 1.2 kPa.
- Phase 5: Begin Terrain Relative Navigation at 1km altitude and M=0.5
- Phase 6: Land On Mars after first deploying airbags to absorb initial impact.

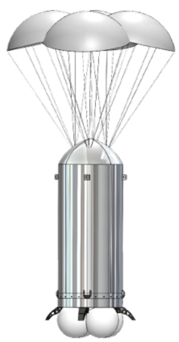
The following figures depict the configuration of both landers during distinct EDL phases.



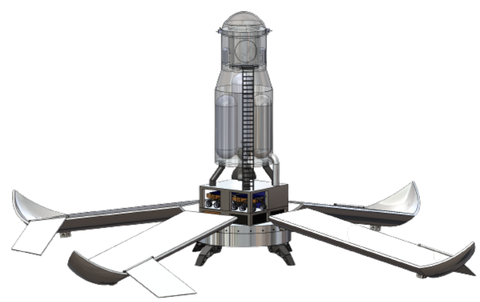
(a) Entry



(b) HIAD deployment during Phase 3



(c) Parachute Deployment



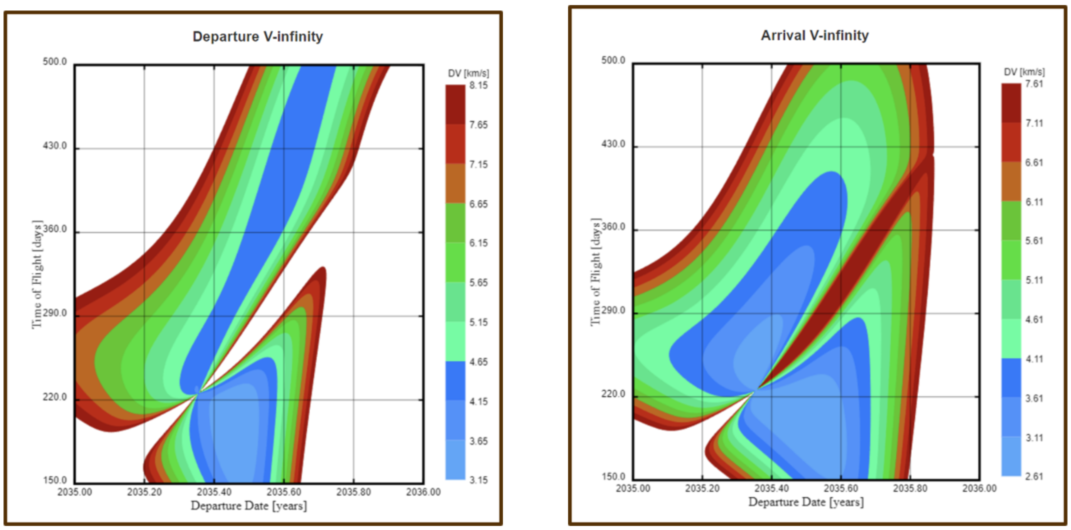
(d) Landed

Fig. 24 Entry, descent, and landing strategy for COAST mission

VIII. Trajectory Design

A. Interplanetary Trajectory

All systems must depart from Earth on the SLS with a 8.4m diameter payload fairing. To determine the departure and arrival parameters for the interplanetary transfer, the following pork-chop plots generated from [27] will define the transfer orbit properties.



(a) Departure pork-chop plot **(b) Arrival pork-chop plot**

Fig. 25 Pork-chop plots for Earth-Mars transfer

Looking at figure 25, to optimize the efficiency of the transfer, the expected launch window will be May 1^{st}-7^{th}, 2035 with an arrival window of Nov. 9^{th}-15^{th}, 2035 (time of flight of 192 days). The backup launch window is a month later on June 1^{st}-7^{th}, 2035 with a correlating arrival window of December 10^{th}-16^{th}, 2035. Note that this backup launch/arrival window will provide a very identical transfer orbit (same time of flight, hyperbolic excess velocity for departure/arrival, etc.). Both landers will depart from Earth with a V_∞ of 3.15 km/s, C_3 of 9.92 km^2/s^2, and arrive at Mars with a V_∞ of 2.65 km/s.

Using the provided C_3 value, various potential launchers launching from Cape Canaveral, Florida ([28], [29], [30], [31]) and their payload capability by mass/volume could be evaluated (see figure 26 and table 10) and selected based on the landers' mass and size.

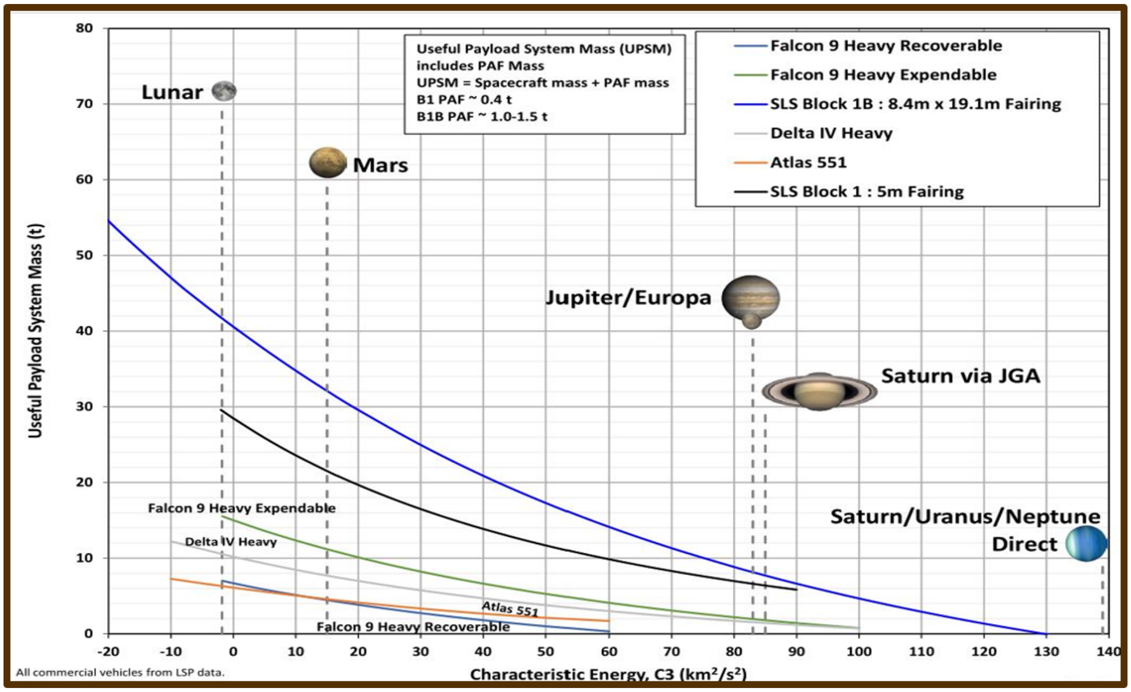


Fig. 26 Launcher C_3 capability with payload capacity

Table 10 Potential launchers for Earth departure showing payload fairing sizing.

Launcher	Max. Payload Capacity (MT)	Diameter (m)	Length
SLS Block 1	21.8	5.0	19.1
SLS Block 1B	34.3	8.4	19.1
SLS Block 2	37.6	8.4	27.4
Falcon 9 Heavy Exp.	10.89	5.2	18.7 (Extended)
Falcon 9 Heavy Rec.	4.54	5.2	18.7 (Extended)
Delta IV Heavy	7.71	5.0	19.1
Atlas 551	4.54	5.0	16.5 (Long)

According to table 10 and the landers' mass in figures 8 and 9, the SLS block 2 provides the optimal payload fairing by physical size and mass. It should be noted that four SLS Block 2 launchers will be required for this mission being that both landers mass with propellant exceeds 37.6 metric tons.

B. Mars Arrival/Aerobraking

Upon Mars arrival, the landers will be on a hyperbolic trajectory (V_∞ of 2.65 km/s) and will require thrusters to place them into a closed orbit. To calculate the ΔV required to place the landers directly into a

200km altitude circular orbit, the following set of orbital mechanics equations must be used:

$$\epsilon = \frac{V_{\infty}^2}{2}, \tag{9}$$

$$\epsilon = \frac{V^2}{2} - \frac{\mu}{r}, \tag{10}$$

and

$$\epsilon = -\frac{\mu}{2a} \tag{11}$$

where ϵ is the specific energy of an orbit, a is the semi-major axis of the orbit, μ is the gravitation parameter for Mars (42,828 km^3/s^2 [20]), r is the position of the spacecraft on the orbit (note: Mars radius is 3,396km [20]), and V is the velocity of the spacecraft at position r. Using equations 9-11, the ΔV required for a direct insertion would be $\Delta V_{NoAerobraking} = 2,103 m/s$. To compare this value with the ΔV required with aerobraking, it is essential to appropriately size the first orbit after Mars insertion. This initial orbit will determine the $\Delta V_{Aerobraking}$ value and the amount of time it takes for the landers to aerobrake to a 100km altitude circular orbit. Previous Mars Spacecraft ([32],[33],[34]) that utilized aerobraking had a similar periapsis altitude of approximately 100km. By iterating through various size initial orbits (changing apoapsis altitude while keeping periapsis altitude 100km), figure 27 can be generated noting that the period of the initial orbit can be calculated using the following equation:

$$\tau = 2\pi\sqrt{\frac{a^3}{\mu}}. \tag{12}$$

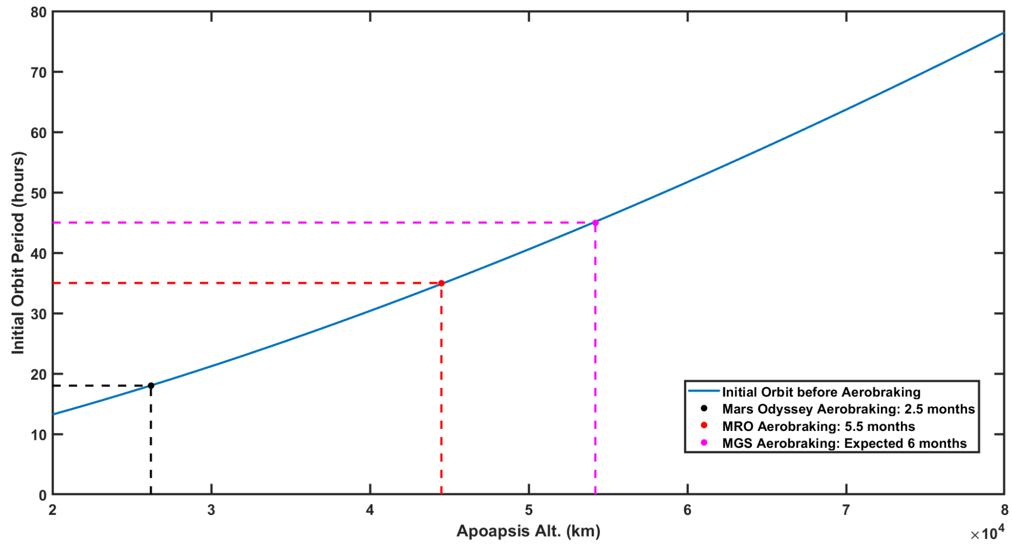


Fig. 27 Initial orbit size/period and aerobraking duration for various Mars aerobraking spacecraft.

Calculating the required ΔV to place the landers into an initial orbit with an apopasis altitude from 27 will result in figure 28.

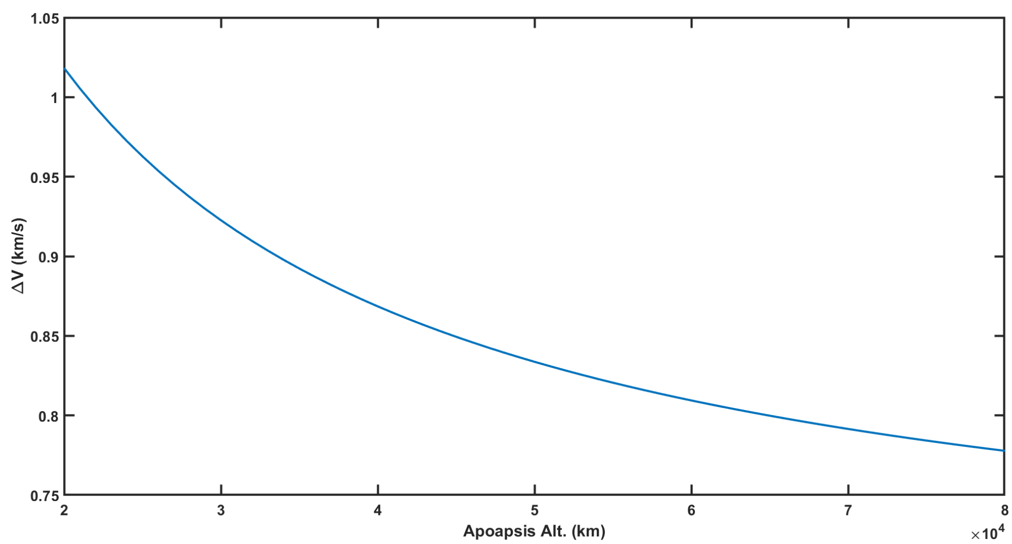


Fig. 28 Required ΔV to place landers into an initial orbit for a particular apoapsis altitude (periapsis altitude is 100km).

Considering that the landers have much larger inertia than previous Mars aerobraking spacecraft, an initial orbit with an apoapsis altitude of 35,000km (initial orbit period of 25.66 hours) will result in an aerobraking duration of approximately 10 months when comparing with aerobraking durations of other spacecraft (figure 27). This apoapsis altitude will require a Mars orbit insertion ΔV of 892 m/s as seen in figure 28. The perturbations that the landers will undergo throughout the aerobraking phase includes drag (100N maximum value [35]) which will dissipate the energy of the orbit, solar radiation which affects the translational/rotational motion of the spacecraft ($1.96\mu N$ worst case scenario for both landers [36]), gravity gradient which affects the attitude of the spacecraft, and J_2 perturbations which will rotate the orbit's line of apsides due to the oblateness of Mars (-0.264 deg/day worst case scenario [37]). Taking these perturbations into account and using the source [38], the ΔV budget for the landers can be tabulated as seen below:

Table 11 Landers' ΔV budget.

Maneuver	ΔV (m/s)
Mars Orbit Insertion	892
Aerobrake Walk-in Insertion	15
Correction Maneuvers/Velocity Adjustments	65
Deorbit	255
Attitude Control	10
Total	1,237

Using the ΔV needed for aerobraking from table 11, the landers will be saving $\Delta V_{Saved} = \Delta V_{NoAerobraking} - \Delta V_{Aerobraking} = 2,103m/s - 1,237m/s = 866m/s$! With this in mind, the landers will begin aerobraking around November 15^{th}, 2035 and will finish aerobraking (begin EDL phase) on September 15^{th}, 2036.

C. MAV Ascent Trajectory

The primary goals of designing an ascent trajectory is to achieve the lowest possible ΔV and time of flight for the transfer from the Martian surface to 5-sol orbit. To design the transfer orbit, a simple Hohman transfer directly from the Martian surface to the 5-sol orbit will be used as a zeroth-order approximation, thus, this transfer will require two burns: one at the Martian surface and the second at the apoapsis of the 5-sol orbit. Because the desired orbit is 5-sol (meaning the orbit has a period of 5 Martian days), the semi-major axis of the target orbit must be 59,790km as calculated from equation 12. There are a variety of 5-sol periapsis/apoapsis combinations to use to achieve this semi-major axis value, so a graphical analysis will be required to choose the optimal orbit configuration. To start, the initial orbit velocity induced on the MAV by the planet's rotation could be determine using the equation: $V_i = \omega R_{Mars} \cos i$ where ω is the angular velocity of Mars ($7.088*10^{-5}$ rad/s from [20]), R_{Mars} is the mean radius of Mars, and i is the latitude of the launch location $i = 4.5$ deg (which will also be the inclination of the transfer and 5-sol orbit) which results in an initial velocity of 0.24 km/s. The velocity at the periapsis of the transfer orbit and apoapsis of the 5-sol orbit can be calculated using equations 10 and 11, noting that the semi-major axis equation is $a = \frac{r_a + r_p}{2}$.

Using these values for various 5-sol periapsis/apoapsis combinations results in the following ΔV and time of flight profile:

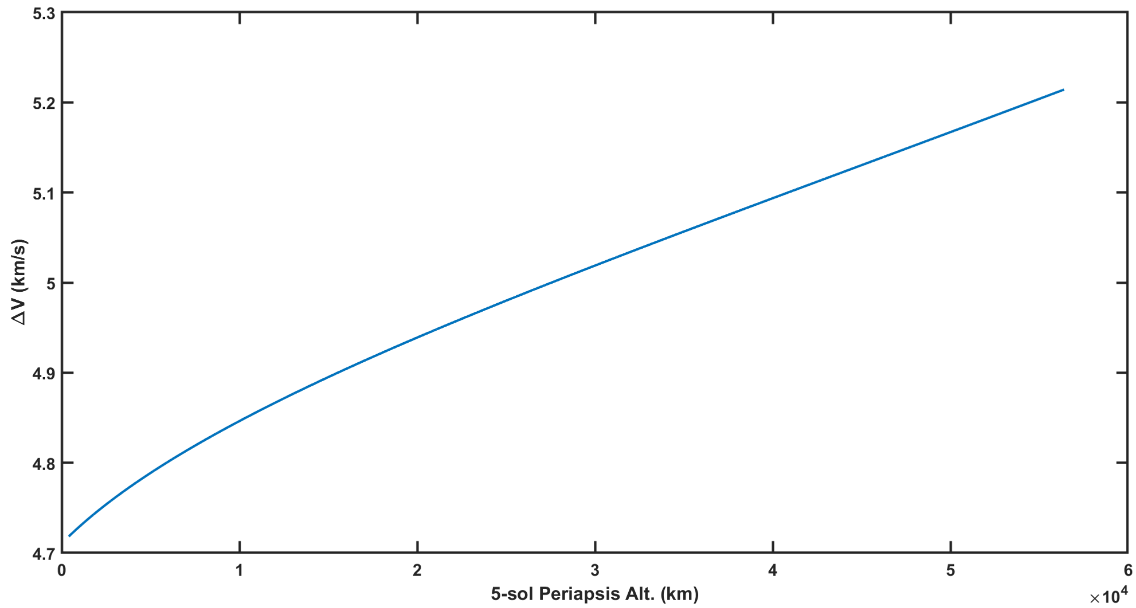


Fig. 29 Hohman transfer ΔV for a given 5-sol periapsis altitude.

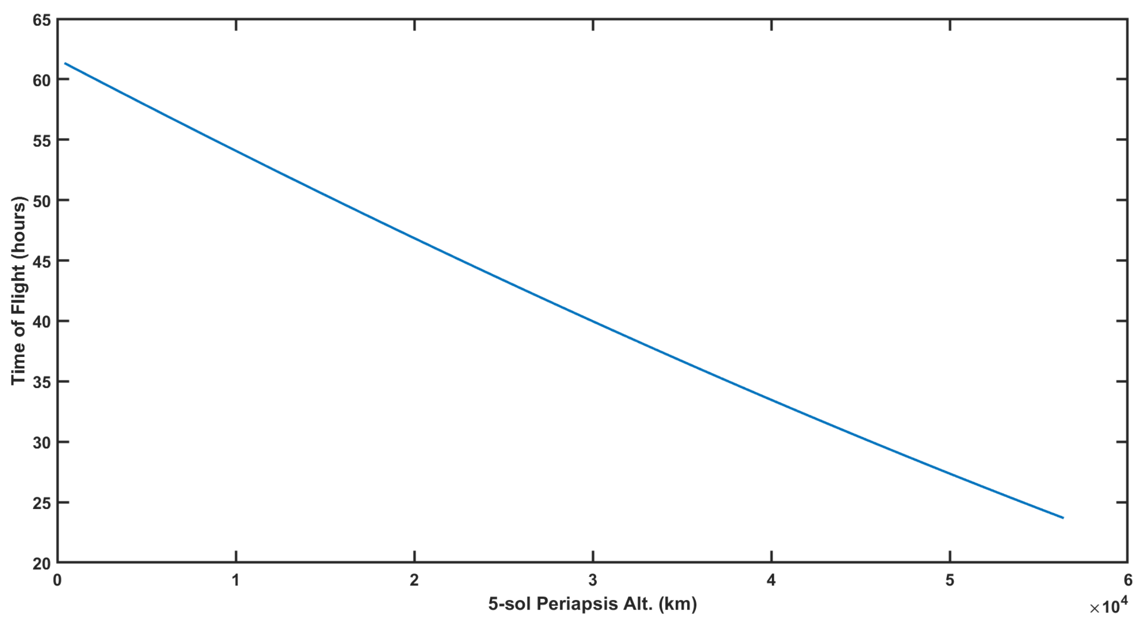


Fig. 30 Hohman transfer time of flight for a given 5-sol periapsis altitude.

Figures 29 and 30 show that the optimal periapsis altitude of the 5-sol orbit should be 400km (note that Mars atmosphere is negligible above 200km [35]), resulting in a ΔV of 4,718 m/s and time of flight of 61.34 hours. A bi-elliptic transfer from the Martian surface to the 5-sol orbit could also be calculated and compared to the Hohman transfer for various combinations of 5-sol periapsis/apoapsis values. This calculation however would introduce another independent variable to iterate through which would be the semi-major axis of the transfer/intermediate ellipse. Following a similar procedure for calculating ΔV and time of flight for the bi-elliptic transfer, and noting that this type of transfer would require three burns: one at the Martian surface, the second at the apoapsis of the transfer ellipse, and the third at the periapais of the 5-sol orbit; the following plots are generated:

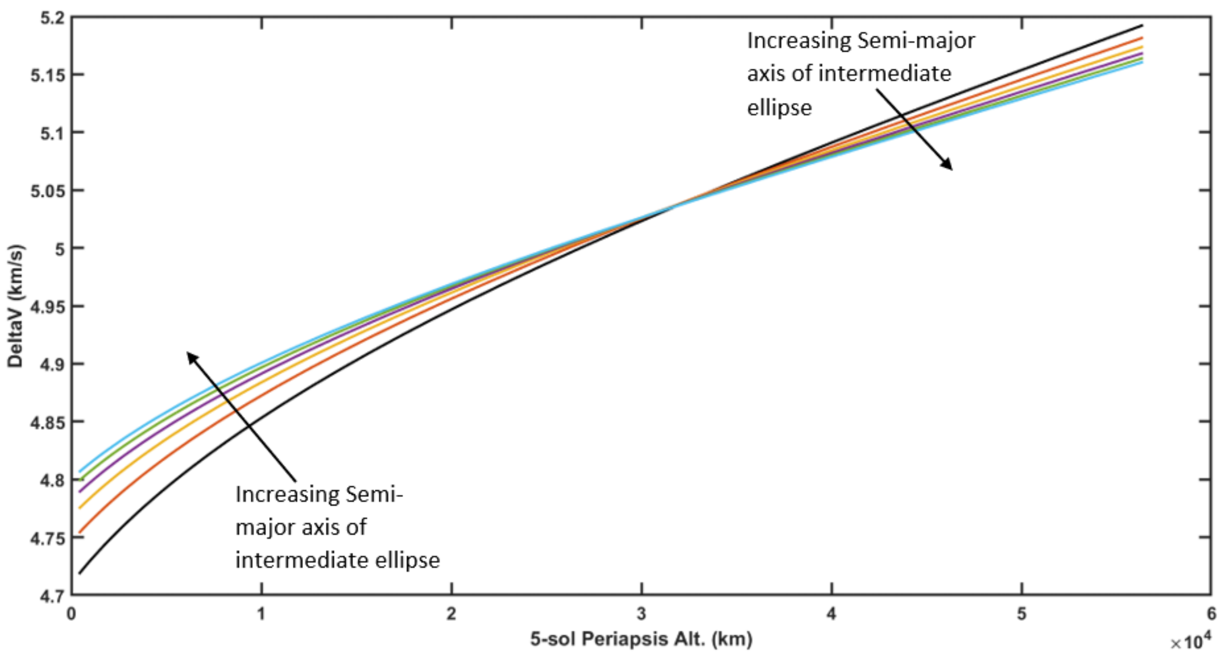


Fig. 31 Bi-elliptic transfer ΔV for a given 5-sol periapsis altitude and various semi-major axis values of transfer ellipse.

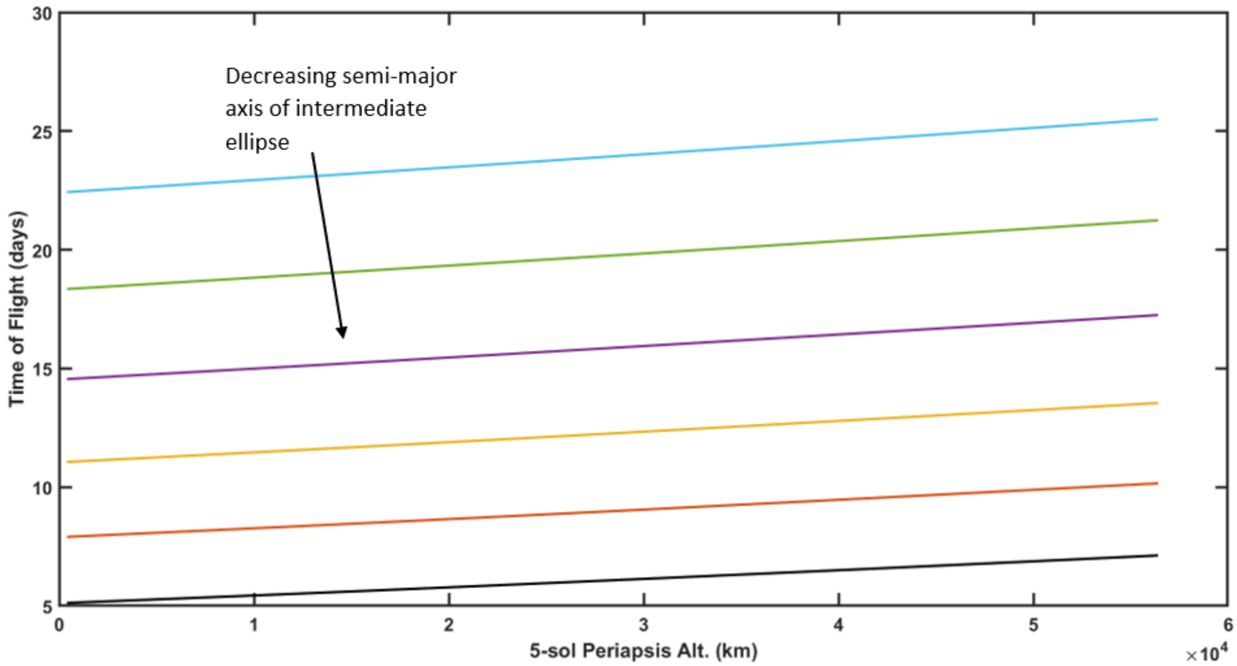


Fig. 32 Bi-elliptic transfer time of flight for a given 5-sol periapsis altitude and various semi-major axis values of transfer ellipse.

As seen in figures 31 and 32, the optimal bi-elliptic transfer would need a ΔV of 4,730 m/s similar to the optimal Hohman transfer ΔV, but the time of flight would be much longer (5 days), thus the MAV will be using a Hohman transfer trajectory for this mission. Taking into consideration the same perturbations that were analyzed for the landers: max. aerodynamic drag for MAV at launch 20kN [35], solar radiation pressure of 1.96μN worst case scenario [36], and J_2 perturbation of 5-sol orbit -0.045 deg/day [37]; the ΔV budget for the MAV can be tabulated as shown below [36]:

Table 12 MAV ΔV budget.

Maneuver	**ΔV (m/s)**
Hohman Transfer	4,718
Station Keeping	10
Counteract Aerodynamic Drag	50
Turn Flight Path from the Vertical	150
Correction Maneuvers/Velocity Adjustments	50
Attitude Control	1
Total	4,979

Finally, the following tables display the desired transfer orbit and 5-sol orbit characteristics:

Table 13 MAV transfer orbit characteristics.

Transfer Orbit Characterstic	Value
Semi-Major Axis	59,590 km
Time of Flight (Half Orbit Period)	61.34 hours
Inclination	4.5°
Eccentricity	0.936
Apoapsis Altitude	112,388
Periapsis Altitude (Mars Surface)	0
C_3	-0.7187 km^2/s^2

Table 14 5-sol orbit characteristics.

5-Sol Orbit Characterstic	Value
Semi-Major Axis	59,790 km
Period	5-Sol (5 Martian Days)
Inclination	4.5°
Eccentricity	0.937
Apoapsis Altitude	112,388 km
Periapsis Altitude (Mars Surface)	400 km
C_3	-0.7163 km^2/s^2

Because the humans are required to depart from Mars by July 1^{st}, 2040, the MAV will depart from Mars with an expected launch date on June 10^{th}, 2040. Backup launch dates will be every 5 Martian days (June 15^{th}, 20^{th}, etc.) preceding this date up to the required departure date.

IX. Mission Analysis

Sun aspect angles (SAAs) are used to determine the efficiency of a solar panel and is also used for thermal analysis/control. In orbit, the incident angle of the Sun's rays with respect to the normal vector of a system's panels is heavily dependent on the attitude control of the system. To supply max power to a system, the SAA should be ideally zero degrees. When the MAV, landers, and PHISH are on the surface of Mars, the SAA for each system is dependent on the time of year (Martian season) and the geometric latitude of the systems. Because the latitude of the landing/launch site is 4.5 degrees, the minimum/maximum SAA for all grounded systems could be geometrically determined for each Martian season:

Table 15 Sun aspect angle throughout Mars' seasons.

Mars Season	SAA during Equinox/Solstice	SAA Throughout Season	
		Minimum	Maximum
Spring (7 Months)	4.5°	0°	20.5°
Summer (6 Months)	20.5°	0°	20.5°
Fall (6 Months)	4.5°	4.5°	29.5°
Winter (4 Months)	29.5°	4.5°	29.5°

Throughout the entirety of the mission, there will be communication losses due to various sources of interference. For example, interference with the landers, MAV, and PHISH that occurs approximately every 0.75 Martian days is expected due to the rotation of Mars and the DST traversing around the 5-sol orbit. This interference should last no longer than 0.5 Martian days. If needed, during this period of interference with the DST, all ground systems could communicate with other Mars orbiting objects such as the Mars Relay Network or the Mars Reconnaissance Orbiter, which has an inclination of 93°and a period (much faster than the 5-sol period of the DST) of 112 minutes [19]. Interference with Mars and Earth is expected to occur twice throughout the entire mission due to solar conjunction as seen in NASA's orbit viewer tool [39]: once during the aerobraking phase on September 27^{th}, 2036 and the other occurs during the propellant transfer phase on November 12^{th}, 2038. Communications between the Earth and Mars systems (including those in orbit) will not be possible during these occurrences, which last approximately 9-12 consecutive days.

X. Power

A. Power Requirements

The robustness of our power systems for a mission of this scale are absolutely critical to smooth and successful operation. By using the previously shown subsystem power budget as well as knowledge of the system architecture and operation, a table of power requests for each subsystem over the course of the mission was created. This is shown in the figures below.

		Phases		
A. MAV	Peak Power (W)	Launch Power (W)	DST Docking Power (W)	Travel Power (W)
A1.0 Payload (Total)	**13**	**13**	**13**	**13**
A1.1 People (x2)	0	0	0	0
A1.2 Mars Samples (Preservation)	13	13	13	13
A2.0 MAV Power (Total)	**1743**	**1557.75**	**1681.25**	**1681.25**
A2.1 Structure and Mechanisms	12.35	12.35	12.35	12.35
A2.2 Thermal Control	520	520	520	520
A2.3 Power	430	430	430	430
A2.4 Comms	222.3	222.3	222.3	222.3
A2.5 ODH	135.85	135.85	135.85	135.85
A2.6 ADCS	185.25	0	185.25	185.25
A2.7 Propulsion	61.75	61.75	0	0
A2.8 ECLSS	175.5	175.5	175.5	175.5
MAV Total	**1756**	**1570.75**	**1694.25**	**1694.25**

Fig. 33 MAV phase power requirements.

		Phases				
B. Lander One (Land-On Mars)	Peak Power (W)	Orbit Power (W)	EDL Power (W)	Deployment Power (W)	Idle Power (W)	PHISH Transfer Power (W)
B1.0 Payload (Total)	**78**	**0**	**0**	**0**	**0**	**78**
B1.1 Transfer Mechanisms (4)	78	0	0	0	0	78
B2.0 Lander Power (Total)	**524.5**	**494.5**	**514.5**	**394**	**384**	**384**
B2.1 Structure and Mechanisms	10	0	0	10	0	0
B2.2 Thermal Control	100	100	100	100	100	100
B2.3 Power	180	180	180	180	180	180
B2.4 Comms	65	65	65	65	65	65
B2.5 ODH	39	39	39	39	39	39
B2.6 ADCS	65	65	65	0	0	0
B2.7 Propulsion	45.5	45.5	45.5	0	0	0
B2.8 EDL	20	0	20	0	0	0
Lander One Total	**602.5**	**494.5**	**514.5**	**394**	**384**	**462**

Fig. 34 Land-On-Mars phase power requirements.

C. Lander Two (ISLE)	Peak Power (W)	Phases Orbit Power (W)	EDL Power (W)	Deployment Power (W)	Idle Power (W)	PHISH Transfer Power (W)	Post Mission Power (W)
C1.0 Payload (Total)	**10828**	**0**	**0**	**0**	**8250**	**8378**	**0**
C1.1 Transfer Mechanisms (4)	78	0	0	0	0	78	0
C1.2 MAV	Table A	-	-	-	-	-	-
C1.3 Fission Power Unit	0	0	0	0	0	0	0
C1.4 PHISH Rovers	2500	0	0	0	0	2500	0
C1.5 SMOXIE	8250	0	0	0	8250	5800	0
C2.0 Lander Power (Total)	**3038.5**	**1683.5**	**2063.5**	**2538**	**1563**	**1573**	**1238**
C2.1 Structure and Mechanisms	975	10	0	975	0	10	0
C2.2 Thermal Control	325	325	325	325	325	325	0
C2.3 Power	1030	1030	1030	1030	1030	1030	1030
C2.4 Comms	104	104	104	104	104	104	104
C2.5 ODH	104	104	104	104	104	104	104
C2.6 ADCS	65	65	65	0	0	0	0
C2.7 Propulsion	45.5	45.5	45.5	0	0	0	0
C2.8 EDL	390	0	390	0	0	0	0
Lander Two Total	**13866.5**	**1683.5**	**2063.5**	**2538**	**9813**	**9951**	**1238**

Fig. 35 ISLE phase power requirements.

D. PHISH Rover (Single Rover)	Peak Power (W)	Phases Operation Power (W)	PHISH Transfer Power (W)	Post Mission Power (W)
D1.0 Payload (Total)	**21.5**	**21.5**	**0**	**6.5**
D1.1 Science Instruments	6.5	6.5	0	6.5
D1.2 Propellant Storage	15	15	0	0
D2.0 PHISH Power (Total)	**103**	**103**	**113.1**	**49**
D2.1 Structure and Mechanisms	24	24	0	24
D2.2 Thermal Control	20	20	20	2
D2.3 Power	17	17	17	8
D2.4 Comms	14	14	14	5
D2.5 ODH	13	13	13	5
D2.6 Navigation Systems	15	15	0	5
PHISH Total	**124.5**	**124.5**	**113.1**	**55.5**

Fig. 36 PHISH phase power requirements.

These power request tables were used to make decisions on required power system technologies and sizing.

B. Technology Selection

Using the required power as a guide, the power systems for our mission were chosen. A 10 kilowatt fission power unit is required to be the main power source of the mission. Different battery systems were chosen for each vehicle in order to store power needed for operation. For the main MAV power system, primary LiCl batteries were chosen because of their high specific energy and low degradation rate. 563 kilograms of primary batteries are needed in order to sustain the maximum power consumption of the MAV for 4 days. While on the ground, the MAV draws any small amount of needed power from Lander Two. The main power source for Lander Two is the 10 kilowatt fission reactor, but this is supplemented by secondary backup batteries. These backup batteries have a 10 kilowatt capacity in case of any small emergencies or contingencies. Lander One uses advanced Mars solar panels for its main power source, although it requires a fairly small amount of power. These solar panels have a coating that can be electrically charged to repel Martian dust that will accumulate on their surfaces. This power is also stored by secondary batteries. The PHISH Rovers are able to transfer small amounts of power to and from the lander in order to supply any unforeseen power needs. The PHISH Rovers also use solar panels to supplement their power, however most of their power comes from the fission power unit on Lander Two. During propellant transfer, the PHISH on-board batteries are charged enough to complete their next trip. Solar panels are used as a backup power source. All of our vehicles use fully regulated bus PPT power management systems. Shown in the figures below is the sizing for each power subsystem.

A. MAV	Area/Size	Mass	Power/Energy Before Contigencies	Efficiency	Degradation	Margin	Power/Energy After Contigencies
A1.0 Power Supply Tech	-	-	-	-	-	-	-
A2.0 Power Storage Tech	-	-	-	-	-	-	-
A2.1 Primary Batteries (LiCl)	470 dm^3	563 kg	110 kWh Stored	60%	5%	20%	225 kWh Stored
A3.0 Power Control	-	-	-	-	-	-	-
A3.1 PPT PMAD System	-	143 kg	330 W Used	83%	5%	30%	430 W Used

(a) MAV technology selection.

B. Lander One (Land-On Mars)	Area/Size	Mass	Power/Energy Before Contigencies	Efficiency	Degradation	Margin	Power/Energy After Contigencies
B1.0 Power Supply Tech	-	-	-	-	-	-	-
B1.1 Solar Panels	60 m^2	125 kg	1320 W Provided	20%	6%	20%	1100 W Provided
B2.0 Power Storage Tech	-	-	-	-	-	-	-
B2.1 Secondary Batteries (NiH)	135 dm^3	110 kg	Must Hold 5.5 kWh	60%	5%	20%	Must Hold 10 kWh
B3.0 Power Control	-	-	-	-	-	-	-
B3.1 PPT PMAD System	-	140 kg	145 W Used	83%	5%	30%	180 W Used

(b) Land-On-Mars technology selection.

C. Lander Two (ISLE)	Area/Size	Mass	Power/Energy Before Contigencies	Efficiency	Degradation	Margin	Power/Energy After Contigencies
C1.0 Power Supply Tech	-	-	-	-	-	-	-
C1.1 Fission Power Unit	5 m^3	5000 kg	14 kW Provided	90%	2%	20%	10 kW Provided
C2.0 Power Storage Tech	-	-	-	-	-	-	-
C2.1 Backup Batteries (NiH)	135 dm^3	110 kg	Most Hold 5.5 kWh	60%	5%	20%	Must Hold 10 kWh
C3.0 Power Control	-	-	-	-	-	-	-
C3.1 PPT PMAD System	-	900 kg	880 W Used	83%	5%	30%	1030 W Used

(c) ISLE technology selection.

D. PHISH Rover (Single Rover)	Area/Size	Mass	Power/Energy Before Contigencies	Efficiency	Degradation	Margin	Power/Energy After Contigencies
D1.0 Power Supply Tech	-	-	-	-	-	-	-
D1.1 Solar Panels	1.3 m^2	4.5 kg	240 W Provided	20%	6%	20%	213 W Provided
D2.0 Power Storage Tech	-	-	-	-	-	-	-
D2.1 On Board Batteries (NiH)	45 dm^3	35 kg	Most Hold 1.8 kWh	60%	5%	20%	Must Hold 2.4 kWh
D3.0 Power Control	-	-	-	-	-	-	-
D3.1 PPT PMAD System	-	8 kg	32 W Used	83%	5%	30%	41 W Used

(d) PHISH technology selection.

Fig. 37 Mission technology selection.

		Phases		
A. MAV	Peak Power (W)	Launch Power (W)	DST Docking Power (W)	Travel Power (W)
A1.0 MAV Power (Total)	**1800**	**1800**	**1800**	**1800**
A1.1 Primary Batteries	1800	1800	1800	1800

(a) MAV power allocation.

		Phases			
B. Lander One (Land-On Mars)	Peak Power (W)	Orbit Power (W)	EDL Power (W)	Daytime Idle Power (W)	Nighttime Idle Power (W)
B1.0 Lander Power (Total)	**1900**	**0**	**600**	**0**	**550**
B1.1 Lander One Solar Panels	1300	1300	0	1100	0
B1.2 Secondary Batteries	600	0	600	0	550

(b) Land-On-Mars power allocation.

		Phases			
C. Lander Two (ISLE)	Peak Power (W)	Orbit Power (W)	EDL Power (W)	Daytime Idle Power (W)	Nighttime Idle Power (W)
C1.0 Lander Power (Total)	**10600**	**10000**	**10000**	**10000**	**10000**
C1.1 Fission Power Unit	10000	10000	10000	10000	10000
C1.2 Backup Batteries	600	0	0	0	0

(c) ISLE power allocation.

		Phases		
D. PHISH Rover (Single Rover)	Peak Power (W)	Daytime Operation Power (W)	Nighttime Operation Power (W)	PHISH Transfer Power (W)
D1.0 PHISH Power (Total)	**2878**	**213**	**187**	**2500**
D1.1 PHISH Solar Panels	213	213	0	0
D1.2 Lander Two (ISLE)	2500	0	0	2500
D1.3 PHISH On Board Batteries	165	0	187	0

(d) PHISH power allocation.

Fig. 38 Mission power allocation.

XI. Propulsion

Propulsion systems need to be considered for the MAV as well as the landers. The MAV needs a main propulsion system to launch into orbit and it also needs a propulsion for attitude determination and control. The landers need a retro-propulsion system for the Entry Descent and Landing (EDL) process.

Space propulsion systems generally fall into three major categories: cold gas, chemical, or electric [24]. There are several parameters that are utilized when comparing different space propulsion systems and propellants. Thrust is the force applied to the rocket due to the propulsion system [24]. Specific impulse (I_{sp}) is a propellant dependent property and is a measure of how efficient the propellant is to create thrust. Specific impulse is defined using Equation 13 where F is Force, $\dot{m}$ is the mass flow rate, and g_0 is the gravitational acceleration constant of Earth.

$$I_{sp} = \frac{F}{\dot{m} g_0} \tag{13}$$

A. Propulsion Type Selection

Cold gas propulsion is the simplest propulsion mechanism and has low I_{sp} values and low thrust values. Electrical propulsion utilizes electrical power to accelerate something to produce thrust [24]. Electric propulsion systems have high I_{sp} values. However, electric propulsion systems require a lot of power and produce low thrust making it infeasible for applications that require a lot of thrust.

Chemical propulsion systems utilize chemical propellants and are more complex than cold gas propulsion. Chemical propulsion systems can have higher I_{sp} values and can produce high thrusts. For this reason, chemical propulsion systems are considered for the needs of this mission. Chemical propulsion systems can be broken down into solid motor, liquid monopropellant, liquid bipropellant, and hybrid systems. One of the disadvantages of solid rocket motors is that they cannot be stopped once started so they are not very useful for things such as ADCS. Additionally, due to the unique nature of the propellant transfer problem, liquid chemical propulsion systems are chosen. It is hard to transport a solid rocket motor from the landers to the MAV.

Table 16 Comparison of different liquid bipropellant systems. Numbers taken from [24]

Propellant	Isp	Mixture Ratio (By mass)	Mixture Ratio (By Volume)	Propellant Mass Ratio	Fuel Mass Ratio	Oxidizer Mass Ratio
LCH4/LOX	360	3.2	1.19	3.12	0.74	2.38
LH2/LOX	390	3.4	0.21	2.69	0.61	2.08
LH2/LF2	410	4.54	0.21	2.47	0.63	2.02
RP-1/LOX	301	2.24	1.59	4.44	1.37	3.07
Hydrazine/LOX	313	0.74	0.66	4.10	2.93	2.17
RP-1/H2O2	276	3.4	0.75	5.34	1.21	4.13

B. Main MAV Propulsion System

1. Propellant Type Selection

Table 16 shows a comparsion of different bipropellant systems. The column labeled *Propellant Mass Ratio* shows the amount of propellant that is necessary with respect to the weight of the MAV. If m_f is the final dry mass of the MAV and m_p is the propellant mass, the ratios given in the table are $\frac{m_p}{m_f}$. Tsiolkovsky's Equation given by Equation 14 is utilized to calculate the $\frac{m_p}{m_f}$ ratio.

$$\Delta V = I_{sp} g_0 \ln \frac{m_f + m_p}{m_f} \tag{14}$$

The calculations in Table 16 are performed for a ΔV of 4,979 m/s for launch. This ΔV is a fixed value.

The higher the I_{sp} of a propellant, the less propellant is necessary. Since all of the propellant has to be carried to Mars, landed, and transferred to the MAV, it is best to select a solution which requires the least amount of propellant.

Cryogenic propellants are propellants that need to be stored and carried at extremely low temperatures such as 120K. Recently, there have been many advances made to store and carry cryogenic propellants [40, 41]. However, for the guidelines of this mission, it was deemed too complicated and low TRL levels to both transport cryogenic propellant to Mars and for the propellant transfer mechanism on Mars while keeping propellant cool the entire time. Therefore, no cryogenic propellants are transported to Mars from Earth. Examining Table 16 shows that cryogenic propellants have significantly higher I_{sp} and significantly lower propellant mass ratio than propellants that are purely not cryogenic. Additionally, it was determined that with the weight of the MAV, landing and transferring both the fuel and oxidizer would be a lot of weight. For many of these mixtures, significantly more oxidizer is necessary than fuel. On the latest NASA Mars Mission with the Perseverance rover, NASA demonstrated the technology to produce Liquid Oxygen on Mars through MOXIE [42]. A scaled up MOXIE is a way to produce LOX on the surface of Mars as the oxidizer so only the fuel has to be transported to Mars from Earth and transferred to the MAV. This is the approach that is taken. RP-1 is chosen for the fuel as RP-1 is not cryogenic and has a high I_{sp}. The RP-1/LOX combination is chosen for fuel.

2. Propulsion System Specifics

In order to have a high TRL and to simplify the design, existing engines were examined for the propulsion system. The SpaceX Merlin 1D Engine is a modern RP-1/LOX pump fed gas-generator engine [43]. The Merlin 1D Engine produces 854 kN of thrust [43]. The weight of the Merlin 1D Engine is given in Table 17. The total weight of the MAV on Mars is around 110 kN so 854 kN of thrust will be more than enough for the MAV with one Merlin 1D type engine. A gas generator cycle is good as it can be utilized to power the pumps for the pump fed system.

Table 17 Propulsion system mass budget.

Component	Mass (kg)
LOX	21,786
RP-1	11,254
Pressurant (He)	24
Engine	470
Tanks	299

Table 17 shows the mass budget of the various components of the propulsion system. The mass of RP-1 and LOX is calculated utilizing the dry mass of the MAV and the ratios given in Table 16. Margins are added to the value derived from the Tsiollkovsky equation to account for Ullage, PMDs, Residuals, and Reserves. The calculated propellant already accounted for a contingency of 30% from the mass budget so there is already 30% reserves accounted for in the design. In addition to this, an extra 5% of propellant was added for further margins such as the propellant that cannot be emptied from the tank or other unforeseen circumstances.

The tank volume was initially sized utilizing the density of the propellants. The minimum volume of LOX is calculated to be 19.1 m^3 and the minimum volume of Fuel is calculated to be 14.1 m^3. Spherical tanks are utilized in space applications to distribute the pressure on the tank evenly. However, with space constraints, a combination of a cylindrical and spherical tank is utilized. To meet space constraints, two LOX tanks and one fuel tank is included in the system. The radius of each tank is selected to be 0.85 meters based on the diameter of the MAV and the overall engine. The height of the fuel tank is selected to be 5.1 m. and the height of the two LOX tanks are selected to be 3.5 m. to meet the minimum volume requirement with some contingency.

The minimum propellant tank pressures are calculated using Equation 15 p_t is the tank pressure in MPa and V_t is the volume of the tank in m^3.

$$\log_{10} p_t = -0.1068[\log_{10} V_t] - 0.2588 \tag{15}$$

The minimum pressure necessary for the LOX tank is calculated to be 0.40 MPa and the minimum pressure necessary for the fuel tank is calculated to be 0.413 MPa. Utilizing the minimum pressures, and the chosen radius of 0.85 meters, the minimum thickness for a think wall spherical pressure vessel is calculated using Equation 16. In Equation 16, r is the radius of the tank, p_b is the burst pressure, and σ_a is the allowable tensile ultimate stress. The burst pressure includes a factor of safety of 2 over the expected pressure in the tank.

$$t = \frac{p}{2\sigma_a} \tag{16}$$

The tank is made from an Aluminum Alloy 2195 due to its high allowable tensile stress and its use in many existing spacecraft propellant tanks. The ultimate tensile stress of Aluminum 2195 is 590 MPa [44] and its density is 2710 kg/m^3 [44]. The minimum required thickness if the system was a sphere is calculated to be 0.000577 meters. To account for a factor of safety, extra tolerance, and the fact that the tanks are really a cylindrical sphere, a thickness of 0.001 meters is utilized.

The mass of an empty tank, m_t is given by Equation 17 where SA is the surface area of the tank, ρ is the density of the material, and η is 0.2.

$$m_t = (1 + \eta) \cdot SA \cdot t \cdot \rho \tag{17}$$

The calculated mass of the tank is listed in Table 17.

The mass of pressurant is calculated using the perfect gas law given in Equation 18 where P is the pressure of the tank, V is the volume of the tank, R is the specific gas constant, and T is the temperature. Helium is utilized as the pressurant gas as it is inate and leads to the lowest mass. The amount of pressurant necessary is

Table 18 Propulsion system power budget.

Component	Power (kW)
Pumps	7,500
Valves, etc.	0.2

listed in Table 17.

$$m_{gas} = \frac{PV}{RT} \tag{18}$$

Table 18 shows the Power budget for the propulsion system. The power values were derived from information about the Merlin 1D Engine [43] as well as the typical power consumption of things such as valves.

3. Propulsion System Schematic and Configuration

Figure 39 shows the propulsion schematic for the main MAV propulsion system. The gas generator cycle is represented by the pre-burner and turbine. There are multiple check valves as well as relief valves across the schematic for safety and redundancy. There is also a quick disconnect for safety while loading the tanks. Multiple sensors are also shown in the schematic to collect necessary data.

C. MAV ADCS Propulsion

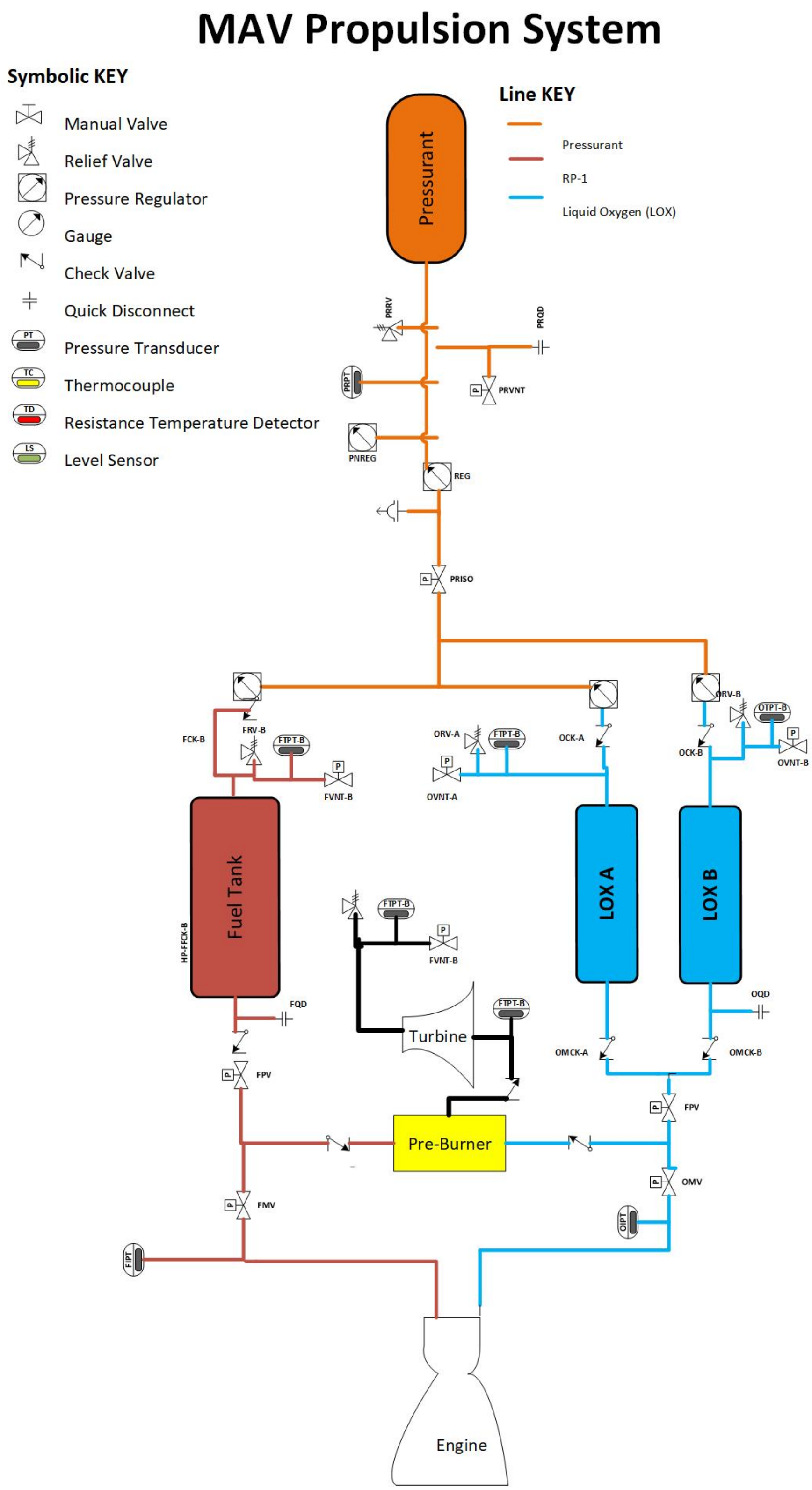


Fig. 39 MAV main propulsion system schematic.

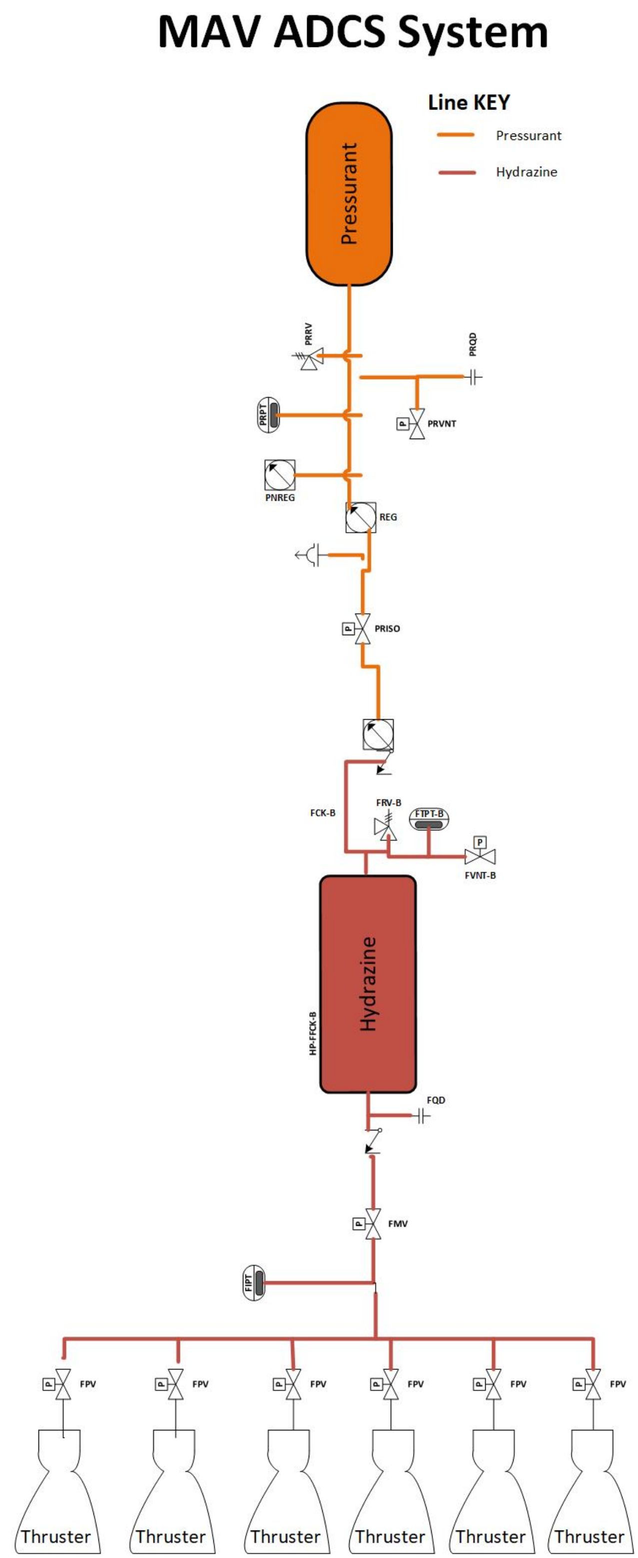


Fig. 40 MAV ADCS propulsion system schematic.

For ADCS, 6 Aerojet MR-107T thrusters are utilized [45]. This is a 110N monopropellant hydrazine thruster. A monopropellant thruster is chosen for ADCS for its simplicity and ability to fulfill the purpose. Figure 40 shows a schematic for the ADCS propulsion system. Much like the main MAV propulsion system , redundancies and safety checks are built in with the various valves and other mechanisms.

D. Lander Propulsion

Table 19 Propulsion system mass budget.

Component	Mass (kg)
Propellant	21,300
Pressurant (He)	55
Engine	2,628
Tanks	277

The propulsion system for the lander is based on the propulsion system outlined in reference [26]. The reader is asked to read reference [26] for more details regarding the propulsion system. The landers will have six 250 kN bipropellant engines that are similar to the RS-72 engine. The engines will use a combination of MMH and MON-25 propellant. Table 19 shows the mass budget of the lander propulsion system. Figure 41 shows a schematic of the lander propulsion system. Similar features are incorporated in this schematic as the MAV main propulsion system for both safety and redudancy. Additionally, this propulsion system is also a gas generator cycle to power the pumps and there are 2 MMH tanks and 4 MON-25 tanks.

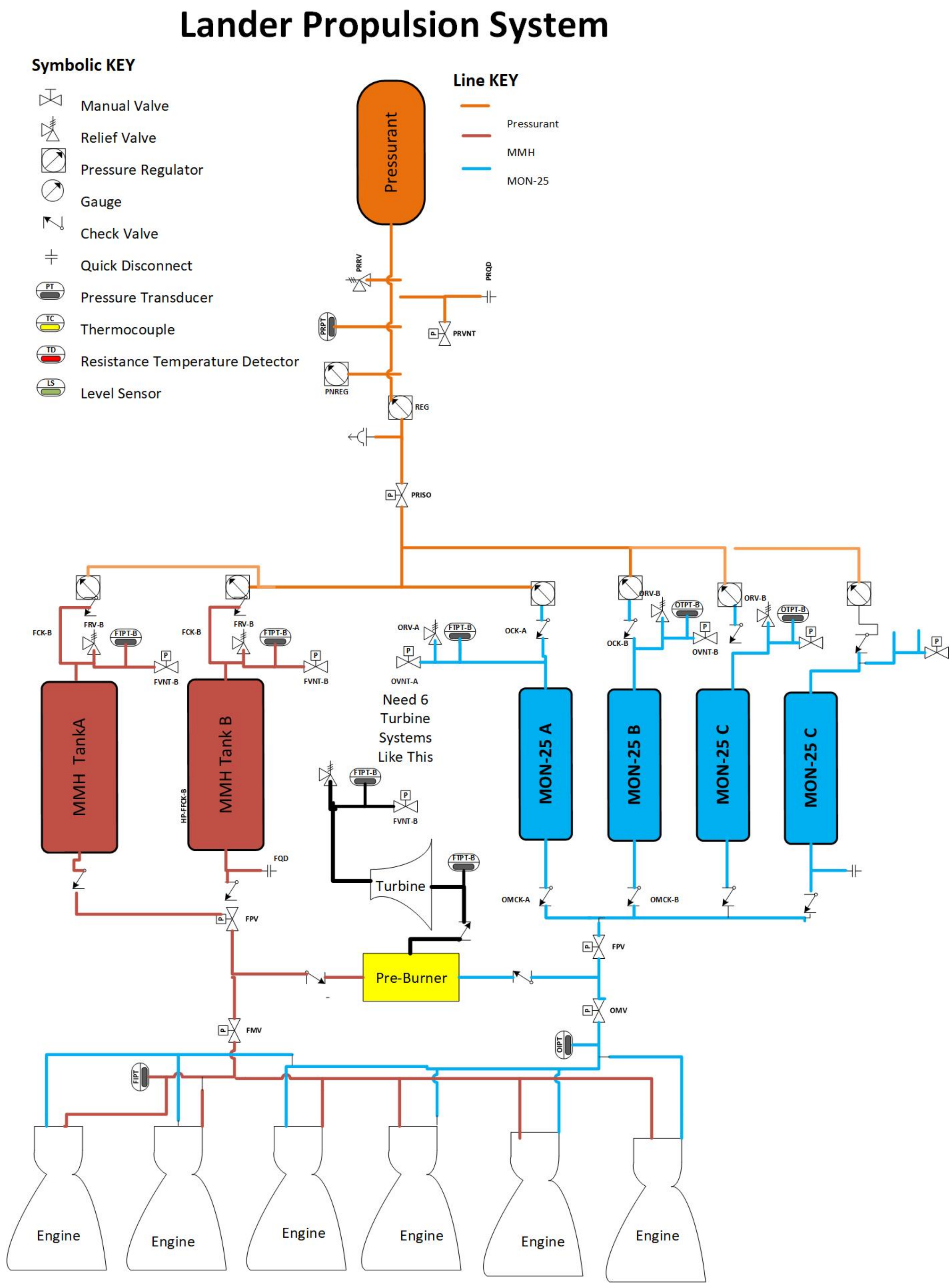


Fig. 41 Lander propulsion system schematic.

XII. Communications and On-Board Data Handling

A. Communication Budgets and Requirements

Communication systems and budgets must be allocated for each of the vehicles and for each mode of communication the vehicles undergo during the mission duration. The communication budgets were broken into four main types of transmission: data, telecommands, health status of vehicles, and emergency signals. The approximate data transmission requirements for each datatype are found in Table 20 [46].

Table 20 Data transmission requirements.

Data Type	Transmission Requirement
Data	1.5 Mbps
Health/status	10 kbps
Emergency	300 bps
Telecommand	40 kbps

Of the three transmission types, data transmission will take the most bits per second and the most power to transmit. Because most data is not time sensitive, these packages will be relayed through the DST before heading back to Earth, rather than being sent to Earth directly. Telecommands transmit actions and autonomous operations to the vehicles for operation during communications blackouts and delays. These allow the vehicle to operate separate from human communication links, especially during the refueling operations where no humans are involved yet. Such autonomous events include the refueling operations on the surface of Mars, MAV docking with the DST, SMOXIE oxygen production, anomaly detection and handling, and general operations during communication blackouts during orbit and during EDL.

B. On-Board Data Handling Budgets and Hardware

The on-board data handling requirements for the mission vary depending on the phase and the vehicle involved. The PHISH and MAV themselves handle very little data acquisition and instead house a significant amount of autonomous capabilities and codes. Figures 42-44 show the data budgets for each vehicle and their operations during the mission [**?**] [47].

These estimations are based on the selected technologies onboard each vehicle as well as the worst case scenario for holding on to data before transmission. During the mission, two conjunctions with the Earth and the Sun will significantly impact communication abilities on the surface of Mars, as notes in the Mission Analysis section. The longest time period during which the vehicles would have to hold onto data without

Function	Size		Throughput (KIPS)	Frequency (Hz)	Saved Data
	Code	Data			
Communications					
Command Processing	7	10	7	10	360
Data encoding	50	150	903	1	1.2
Vehcile Communication Collection	10	30	100	0.05	200
EDL Functions					
Thruster Control	1	1	1.5	1	3
IMU	1	1	18	1	3
LiDAR Sensor Control	1	1	10	1	3
Error determination	1	0.1	12	10	3
Autonomy					
Fault Detection	9	10	0.008	0.05	1.2
Fault Correction	9	10	0.001	0.05	1.2
Complex Autonomy	15	10	20	10	60
Operative System					
Subsystem data management	4	10	1	0.07	2
Executive	2	3.5	7	-	2
Run-time kernel	8	4	-	-	-
I/O Device Handlers	4	10	11000	-	-
Built-In test diagnostic	1	1	0.5	-	3
Math utilities	1	1	-	-	-
Other					
Kalman filter	40	48	8	0.1	0.6
EPS Management	3	3	5	0.001	0.03
Power Management	1.2	0.5	5	1	2
Thermal Control	0.8	1.5	3	0.1	1.2
Total bits/sec					642.6
Total bytes/sec					80.325
Total bytes					83280960
Total RAM	**83.28096**	**Mbyte**			
Total ROM	**36.64362**	**Mbyte**			
Total saved data	**6.94008**	**Mbyte/day**			
Throughput	**12.10101**	**MIPS**			

Fig. 42 Data handling requirements for both landers.

downlinking to Earth would be 12 days. The longest time period the on-surface assets will go without connection to the DST in orbit is approximately 10 hours. These values are was were used to size the amount of required RAM for the processors in the on-board data handling subsystem. Based on these calculated, the MIL-STD-1553B bus was selected to house the on-board data handling system. This is a proven and flight tested system that has high reliability, which is crucial for such a complex mission. A LM RAD750 processor was chosen for the landers and the MAV because of the processing speed and the storage capabilities. This is a slightly newer version of the LM RAD6000 processor, which will be onboard the PHSIH rovers. These are slightly slower, but the PHISH don't require as much processing power.

Function	Size (Kwords) Code	Data	Throughput (KIPS)	Frequency (Hz)	Saved Data (bit/sec)
Communications					
Data Compression	20	50	50	1	1.5
Data Encoding	50	150	903	1	1.5
Vehcile Communication Collection	10	30	100	0.05	300
Command Processing	1	4	7	10	400
Telemetry Processing	1	2.5	3	10	40
ADCS Functions					
Error Determination	4	6	120	10	3
Orbit propogation	7	10	200	10	7.2
IMU	1	1	18	1	5
Thruster Control	1	2	1.5	1	5
Star Tracker	2	15	2	0.01	5
Autonomy					
Simple Autonomy	2	1	1	1	20
Fault Detection	4	1	15	5	1.5
Fault Correction	2	10	5	5	1.5
Power and Thermal					
Kalman filter	40	48	8	0.1	0.6
EPS Management	3	3	5	0.001	0.03
Power Management	1.2	0.5	5	1	2
Thermal Control	0.8	1.5	3	0.1	1.5
Operative System					
Subsystem data management	4	10	1	0.07	2
Executive	2	3.5	7	-	2
Run-time kernel	8	4	-	-	-
I/O Device Handlers	4	10	11000	-	-
Built-In test diagnostic	1	1	0.5	-	3
Math utilities	1	1	-	-	-
Total bits/sec					802.33
Total bytes/sec					100.2913
Total bytes					6661345
Total RAM	**6.661345**	**Mbyte**			
Total ROM	**2.930992**	**Mbyte**			
Total saved data	**8.665164**	**Mbyte/day**			
Throughput	**12.455**	**MIPS**			

Fig. 43 Data handling requirements for MAV launch vehicle.

Function	Size (Kword)		Throughput	Frequency	Saved
	Code	Data	(KIPS)	(Hz)	Data (bits/sec)
Communications					
Command Processing	1	4	7	10	400
Positioning	1	2.5	3	10	40
Navigation Functions					
Navigation system	5	5	7	1	5
Error determination	1	0.1	120	10	3
Autonomy					
Simple Autonomy	2	1	1	1	20
Complex Autonomy	15	10	20	10	60
Fault Detection	4	1	0.01	5	1.5
Fault Correction	9	10	0.001	5	1.5
Operative System					
Subsystem data management	4	10	1	0.07	2
Executive	2	3.5	7	-	-
Run-time kernel	8	4	-	-	-
I/O Device Handlers	4	10	11000	-	-
Built-In test diagnostic	1	1	0.5	-	3
Math utilities	1	1	-	-	-
Other					
Kalman filter	40	48	8	0.1	0.6
Power Management	1.2	0.5	5	1	2
EPS Management	3	3	5	0.001	0.03
Thermal Management	0.8	1.5	3	0.1	1.5
Total bits/sec					540.13
Total bytes/sec					67.51625
Total bytes					121529.3
Total RAM	**0.121529**	**Mbyte**			
Total ROM	**0.053473**	**Mbyte**			
Total saved data	**5.833404 Mbyte/day**				
Throughput	**11.18751**	**MIPS**			

Fig. 44 Data handling requirements for PHISH rovers.

C. Communication Architecture

The main communications strategy for the mission is similar to past missions to the surface of Mars. Some challenges are presented in the original problem statement, such as accounting for communication blackouts during entry, descent, and landing. With such concerns comes a need for redundant communication systems to ensure successful operation of the robotic refueling mission and the human launch mission. Such a strategy requires each vehicle to have two methods of communicating with Earth. Within the mission architecture, the DST is used as a relay throughout the duration of the autonomous and human mission on Mars. Figure 45 shows the overall communications architecture for the mission.

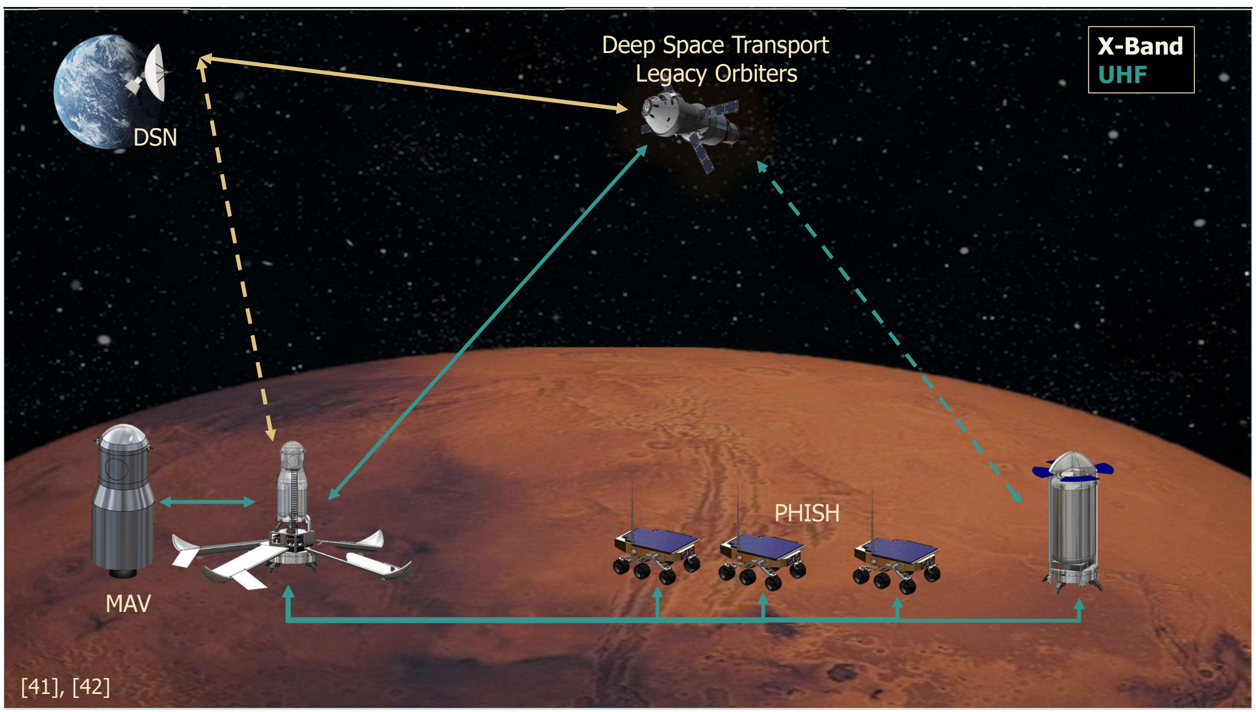


Fig. 45 Communication links and architecture.

D. Communication Link Budgets

The link budgets for each communication link were evaluated for the mission. For each budget, the worst-case scenario was used as the base case to ensure that the link would close between different communication nodes. For example, the maximum slant range was used to calculate the link budget for communications from the surface of Mars up to Earth. Similarly, the largest distance from Mars to Earth was used to calculate the space losses for the Earth-Mars surface link.

	Telecommand	Health/Status	Emergency	Units
BER	0.0000001	0.00001	0.00001	
Modulation Scheme	BPSK Viterbi	BPSK Viterbi	BPSK Viterbi	
Eb/No(required)	5.5	4	4	dB
Required Datarate	40000	10000	300	bps
Compressed Datarate	25000	6250	187.5	bps
Uplink Frequency	7190000000	8425000000	8425000000	Hz
Downlink Frequency	7167500000	8425000000	8425000000	Hz
Pin	130	17	0.5	W
D_tr	0.75	0.75	0.75	m
A_eff_tr	0.243	0.243	0.243	m^2
Transmitter Gain	1756.311	2411.478	2411.478	
Transmitter Gain (dB)	32.446	33.823	33.823	dB
Communication Efficiency	0.9	0.9	0.9	
Reciever Gain (dB)	74.4	74.4	74.4	dB
light speed	299792459.0	299792460.0	299792460.0	m/sec
Signal Wavelength	0.0	0.0	0.0	m
System Noise Temperature	460.6	460.6	460.6	K
K	1.38065E-23	1.38065E-23	1.38065E-23	
Atmospheric Losses	-0.5	-0.5	-0.5	dB
Attenuation Losses	-273	-273	-273	dB
Implementation Loss	-3	-3	-3	dB
Eb/No(design)	8.642	7.205	7.119	dB
Margin	3.142	3.205	3.119	dB

Fig. 46 Surface to Earth communication budgets.

	Data Transmission	Telecommand	Health/Status	Emergency	Units
BER	0.00001	0.0000001	0.00001	0.00001	
Modulation	BPSK Viterbi	BPSK Viterbi	BPSK Viterbi	BPSK Viterbi	
Eb/No(required)	4	5.5	4	4	dB
Required Datarate	1500000	40000	10000	300	bps
Compressed Datarate	937500	25000	6250	187.5	bps
Uplink Frequency	2500000000	2115000000	2500000000	2500000000	Hz
Downlink Frequency	2500000000	2115000000	2500000000	2500000000	Hz
Pin	330	23	2.2	0.07	W
D_tr	0.4	0.4	0.4	0.4	m
C_tr	1.256637061	1.256637061	1.256637061	1.256637061	m
h_tr	0.444752127	0.376260299	0.444752127	0.444752127	dB
Transmitter Gain (dB)	17.6065834	16.15399066	17.6065834	17.6065834	dB
Pointing Error	2.5	2.5	2.5	2.5	dB
Half-Power Beamwidth	20.98547206	24.80552253	20.98547206	20.98547206	
Pointing Loss	-1.429560408	-1.209408105	-1.429560408	-1.429560408	dB
Communication Efficiency	0.9	0.9	0.9	0.9	
D_r	1.5	1.5	1.5	1.5	m
A_eff_r	0.971930227	0.971930227	0.971930227	0.971930227	m^2
Reciever Gain	849.3438852	607.8890082	849.3438852	849.3438852	
Reciever Gain (dB)	29.29083564	27.83824291	29.29083564	29.29083564	dB
Light Speed	299792458	299792458	299792458	299792458	m/sec
Signal Wavelength	0.119916983	0.141745843	0.119916983	0.119916983	m
System Noise Temperature	385	385	385	385	k
K	1.38065E-23	1.38065E-23	1.38065E-23	1.38065E-23	
Atmospheric Losses	-0.5	-0.5	-0.5	-0.5	dB
Implementation Loss	-3	-3	-3	-3	dB
Space Losses	-203.0517772	-203.0517772	-203.0517772	-203.0517772	dB
Eb/No(design)	7.026	8.513	7.026	7.282	dB
Margin	3.026	3.013	3.026	3.282	dB

Fig. 47 Surface to orbit communication budgets.

	Data Transmission	Telecommand	Health/Status	Emergency	Units
BER	0.00001	0.0000001	0.00001	0.00001	
Modulation	BPSK Viterbi	BPSK Viterbi	BPSK Viterbi	BPSK Viterbi	
Eb/No(required)	4	5.5	4	4	dB
Required Datarate	1500000	40000	10000	300	bps
Compressed Datarate	937500	25000	6250	187.5	bps
P_in	0.0003	0.000006	0.000001	0.0000001	W
Frequency	2300000000	2115000000	2300000000	2300000000	Hz
light speed	299792458	299792458	299792458	299792458	m/sec
Signal Wavelength	0.13034	0.141745843	0.130344547	0.130344547	m
Atmospheric Losses	-0.5	-0.5	-0.5	-0.5	dB
Space Losses	-99.68234	-99.68234	-99.68234	-99.68234	dB
Implementation Loss	-3	-3	-3	-3	dB
Eb/No(design)	7.41533	9.16595	7.40503	9.63382	dB
Margin	3.41533	3.66595	3.40503	5.63382	dB

Fig. 48 Surface to surface communications budgets.

XIII. Project Management

A. Project Schedule

The project schedule for this mission was developed from NASA's Project Life Cycle schedule for a Human Space Flight mission [48]. The schedule details the phases of the program, beginning in Pre-Phase A, which is currently taking place in 2023. This phase includes identifying stakeholders, developing the baseline Concept of Operations, identifying risks, and preparing program proposals. Phase A, beginning in 2025, centers around developing proposed mission and system architectures that are responsive to program constraints and requirements. Phases B and C focus on the design of the system before system assembly taking place in Phase D. The phases are spaced out according to the launch date required in 2035. Phase E, Operations and Sustainment, will take place between 2035 and 2040 as specified in the AIAA Announcement of Opportunity and detailed in the Concept of Operations.

Project Life-Cycle Phases	Pre-Phase A: Concept Studies (2023)	Phase A: Concept and Tech Development (2025)	Phase B: Preliminary Design and Technology Completion (2030)	Phase C: Final Design and Fabrication (2033)	Phase D: System Assembly, Integration and Test, Launch (2035)	Phase E: Operations and Sustainment (2035-2040)	Phase F: Closeout (2040)
Project Lifecycle Major Events	Key Decision Point A (KDP A) Project Requirements	KDP B	KDP C	KDP D	KDP E Launch	KDP F End of Mission	Final Archival of Data
Human Space Flight Project Reviews	Mission Concept Review (MCR)	System Requirements Review (SRR) System Definition Review (SDR)	Preliminary Design Review (PDR)	Critical Design Review (CDR) System Integration Review (SIR)	Operational Readiness Review (ORR) Safety Mission and Success Review (SMSR)	Critical Events Readiness Review (CERR)	Disposal Readiness Review (DRR)

Fig. 49 Project schedule by phases.

B. Mission Cost Estimate

The mission cost estimate encompasses the total cost for the design, development, and operation of all vehicle components as well as the cost for the launches using the SLS. This resulted in a total cost of $9.13 billion for the entire mission. The AIAA prompt specifies that the cost of the landers and launch vehicles do not need to be factored in for the total cost. Therefore, this brings the mission budget to a total of $3.73 billion. The methodology resulting in these figures is detailed below in sections pertaining to each individual vehicle.

1. MAV

The NASA costing software was used to develop the cost of the instrumentation necessary for the design and development of the MAV. This total was found to be around \$1.22 billion, and a 25% margin was applied to this figure to account for additional interfaces and components necessary for the vehicle, bringing the total to \$1.52 billion. The costs not directly associated with the vehicle, such as the operations, systems engineering, and program management costs were then calculated. This was done using the budget breakdown of the Apollo program [49]. Each indirect cost was calculated as a percentage of the total spacecraft cost for the Apollo mission, and that percentage was then multiplied by the MAV spacecraft cost to determine a figure for the indirect costs associated with developing the MAV. This number was found to be around \$1.50 billion. The total cost for the design, development, and operation of the MAV was therefore found to be \$3.02 billion.

2. PHISH

The cost of the design and development for the PHISH rovers was determined by a historical mission analysis of the previous NASA rovers landed on Mars. The two metrics used for this analysis were the mass and complexity of the rovers. The mass was determined in kilograms from released mission information [50]. The complexity weighting was determined from the number of scientific instruments developed for the rover. These metrics were then plotted against the cost of the rover development, adjusted for inflation to Financial Year 2023.

The PHISH were determined to have an individual mass of 243 kg and a complexity of 1, since they are primarily used for propellant transfer and do not require any additional scientific instruments. By taking the average of the cost predicted from each linear fit, the cost of an individual rover was found to be \$0.18 billion for a total of \$0.54 billion for the design and development of the three PHISH rovers.

3. Landers

The cost of the landers was determined by a historical mission analysis of various landers. Initially, these missions only included landers sent to Mars, as shown in the first subplot below [50]. Unfortunately, due to the magnitude of the mass of the landers used for this project, the costs extrapolated from these missions were unreasonably high. Therefore, the Human Landing System was added as a data point to this analysis using its current projected budget as the total cost [51]. This extrapolation led to a more reasonable figure of \$2 billion for the development of the landers, however this figure comes with a large factor of uncertainty due to the

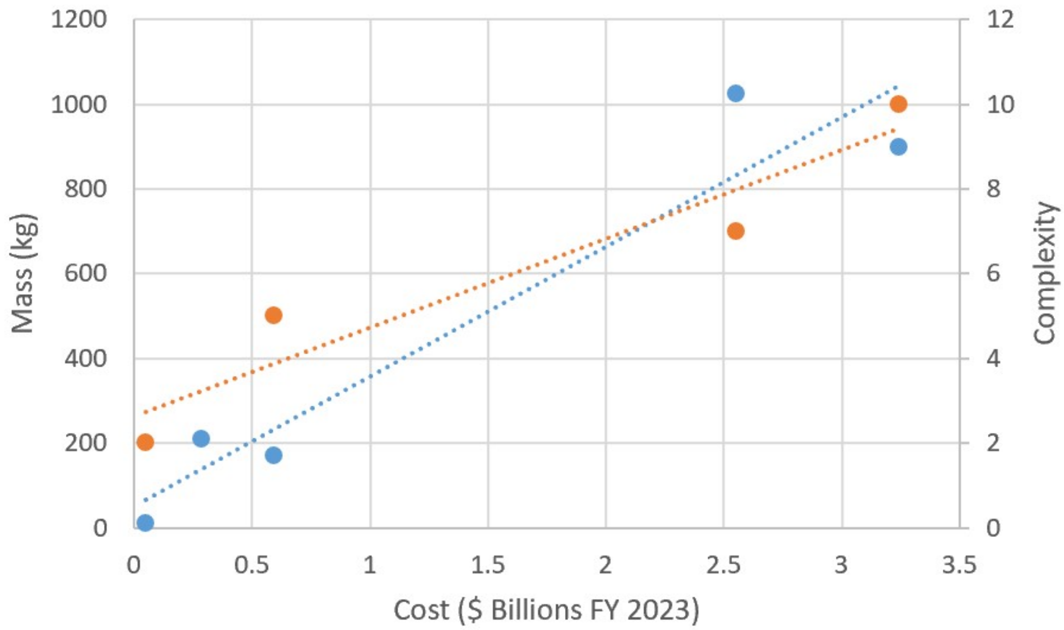
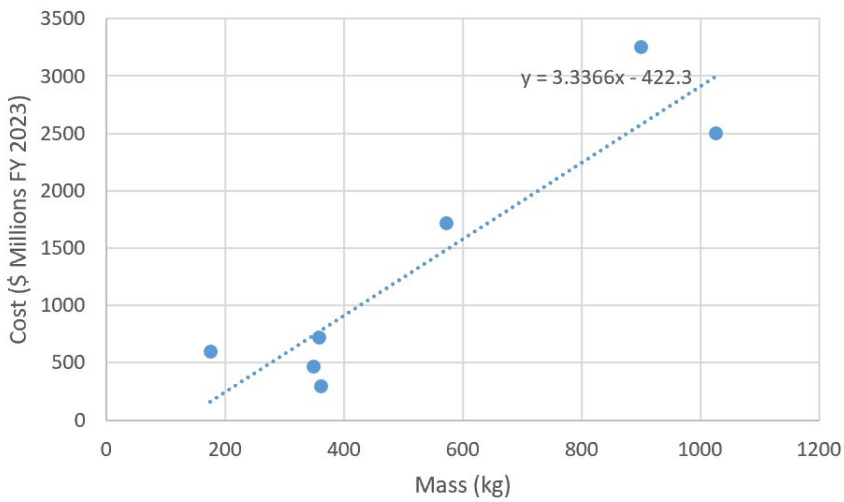
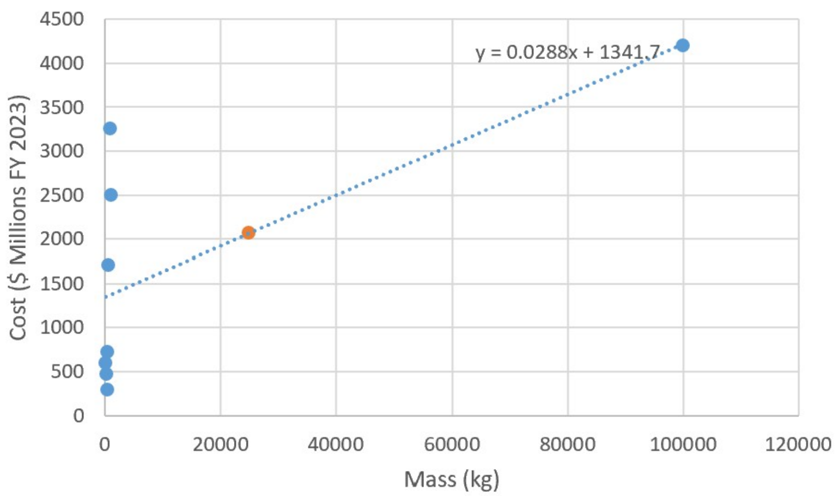


Fig. 50 Historic rover analysis.

extreme mass of this mission compared to previous Mars missions. The mass and budget for the landers is plotted with the orange dot in subplot b shown below.



(a) Analysis of Mars missions.



(b) Analysis including HLS conceptual design.

Fig. 51 Historic Lander Analysis

4. SLS Launch Vehicle

As noted earlier, it was determined that four SLS launches were needed for an overall mass on this scale. Although there are no official NASA estimates for cost per SLS launch, there are several papers citing the predicted future cost for SLS launches [52, 53], which result in an estimated figure of $0.85 billion per launch. This results in a total launch vehicle cost of $3.4 billion.

5. SMOXIE Development

The cost for development of the currently operational MOXIE was $50 million [54]. It was estimated that a number similar to this figure would be required to develop a larger version of MOXIE, which will be the main cost for the two SMOXIE units. Therefore, a total figure of $0.05 billion was estimated for the SMOXIE units.

6. Surface Operations

The surface operations cost was determined from historic mission analysis of previous rover missions. It was found that surface operations typically encompassed around 20% of the cost of the rover design [50]. Therefore, the cost of the units being used during surface operations (PHISH and SMOXIE) was multiplied by a factor of 20% to estimate the cost of their operations over the course of the propellant transfer process. This figure did not include operations of the MAV since that operational cost was previously determined as outlined in the MAV section above. The results of this analysis led to a surface operations cost of $0.12 billion.

C. Risk Management

The most impactful risks for the mission were assessed and displayed in a Risk Matrix. This matrix evaluates the probability and severity of consequence for each risk on a scale from 1 to 5. In completing this assessment, the mission assurance guidelines developed by NASA for Class A missions were used [55]. The risks developed and analyzed for the system along with their mitigation strategies are shown below.

Table 21 Risk titles and management approaches.

Rank & Trend	Approach	Risk Title
1, D	A	Remote MAV launch abnormal
2, D	A	Large mass impacts on EDL
3, U	M	Rover failure during propellant transfer
4, D	W	LOX production
5, U	R	Environmental impacts on equipment and operations
6, U	M	Schedule integration problems between subsystems

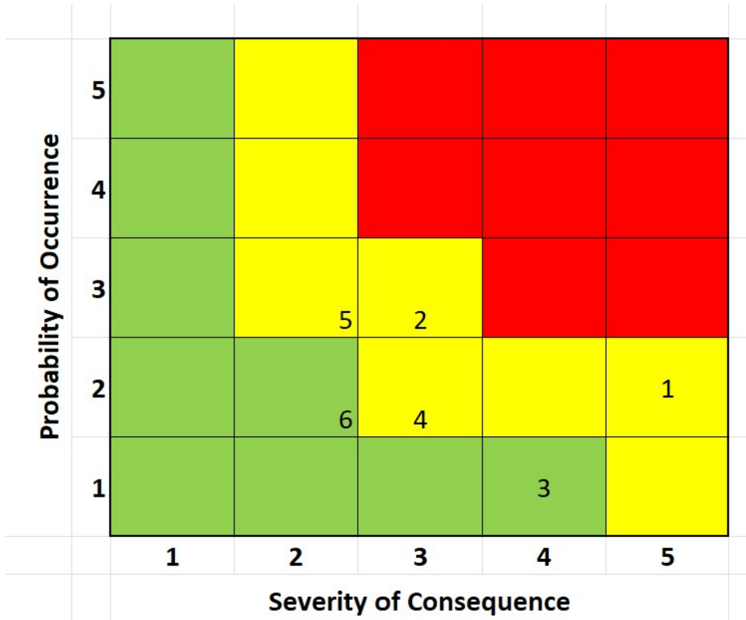


Fig. 52 Risk assessment matrix.

The most critical risk to the mission was determined to be abnormalities associated with performing a remote launch of the MAV. Since the ultimate goal of the mission is to return the crew and samples from the surface of Mars, any problems associated with that launch would prove detrimental to the mission as a whole. Although the likelihood that this would happen to an extent that the mission would be derailed is low, the severity of this risk would be high. This is depicted graphically in the Risk Matrix diagram. The second most important risk was associated with the entry, descent, and landing stage of the mission. Since this mission requires landing the largest payload ever onto the surface of Mars, it is expected that there will be risks associated with this maneuver. Both of these aforementioned risks must be accepted, as they are essential to the successful completion of the mission. The third risk addresses the possibility of a rover failure during propellant transfer. The strategy to address this risk is mitigation, which was done to a great extend by introducing the concept of redundancy with the three rover strategy. While it is unlikely that there will be enough rover failures to greatly impact the propellant transfer process, the result of that occurrence would greatly impact the next steps to be taken for the duration of the mission. Another risk associated with this mission results from the use of a scaled up version of MOXIE to produce LOX in-situ. This risk will be watched as the technology continues to develop, however the specifications we have outlined previously seem reasonable based on the current technology. Additional risks are associated with environmental impacts on equipment and operations due to the harsh temperature and dust conditions on the Martian surface. While it is likely the environment will have an impact on equipment, it is somewhat unlikely that these impacts will be outside the tolerances the equipment is prepared to encounter. Additional research can be done on long-term impacts of Martian conditions by examining the wear and tear on the current rovers deployed on Mars. The final risk concerns the logistics around the integration of the many components needed for this mission. Since there are many factors that must come together to make this mission a success, it is likely that certain components will delay others. This risk has been mitigated by allotting extra time to the planning missions in the project schedule, however it is unreasonable to expect that this will eliminate all schedule delays.

D. Business Case

One of NASA's many mottos is "exploring the secrets of the universe for the benefit of all". We are lucky enough to live in a time when human exploration of Mars is not just theoretical- with current technology, this centuries long dream can finally be achieved. Not only will this mission increase our knowledge of both Mars and the universe by supporting the first ever human exploration of the planet, but it will benefit everyone with its concurrent economical impact. NASA's Moon to Mars Campaign has an estimated economic output of $20.1 billion annually. Additionally, for every civil service job located at NASA centers related to Moon to Mars, nearly 37 additional jobs are supported throughout the US economy [56]. Not only will this mission further the goals of science, but it will simultaneously increase economic output. The COAST team has prioritized risk mitigation and reliability throughout the mission architecture development. The system has been strategically designed to address risks such as dust and temperatures from the Martian environment, reduced LOX production rates, and rover abnormalities. The use of three propellant transfer rovers utilizes the concept of redundancy to ensure a successful operation on the Martian surface. The total programmatic cost of $9.13 billion will encompass the design, development, and operation phases of this mission. Although this sounds like a large figure, this total cost is only 3.5% of the Apollo Missions [49].

XIV. Conclusions

Balancing all of the elements of a manned mission to Mars is a challenging task that NASA hopes to undertake in the coming decades. In order for this mission to be successful, a valid refueling method must be determined for refueling the human ascent vehicle. The solution proposed in this report offers a redundant system for the entire two-year mission in order to successfully complete this task. Many challenges came about in the design of this system, including balancing the requirements from all the subsystems, identifying areas of criticality, and integrating the subsystems into one cohesive system. Future work on this project includes further analysis of the system architecture, and final component and configuration selection.

Appendix A: Requirements

System Level Requirements

COAST-F-1.0 Minimum Lifetime
COAST shall be designed to have a minimum lifetime of 3.5 years from launch from Earth to rendezvous with the DST.
Justification: Requirement aligns with the timeline dictated by the AO.

COAST-F-2.0 Crew Capacity
COAST shall support two crew members during the ascent from Mars surface to the DST.
Justification: Requirement aligns with the crew capacity required by the AO.

COAST-F-3.0 Autonomous Refueling
COAST shall autonomously refuel the MAV with propellant required for ascent.
Justification: Requirement aligns with autonomy requirements dictated by the AO.

COAST-F-4.0 Communications
COAST shall provide communications with Mission Control on Earth.
Justification: Requirement aligns with communication discipline required by the AO.

COAST-F-5.0 DST Docking
COAST shall reach a 5-sol parking orbit around Mars to dock with the DST.
Justification: Requirement aligns with orbit requirements described by the AO.

COAST-F-6.0 Return Sample Payload
COAST shall have a minimum scientific return payload capacity of 50 kg.
Justification: Requirement aligns with sample capacity requirements in the AO.

Subsystem Level Requirements

COAST-MO-1.0 Lander Payload Capacity
Each of the two COAST landers shall have a payload capacity of 25 metric tons.
Justification: Aligns with capacity requirements in the AO.

COAST-MO-2.0 End-of-Life Operations
COAST shall implement end-of-life operations after crew delivery at the DST.
Justification: Aligns with Planetary Protection Protocols for operation on Mars.

COAST-MO-3.0 System Deployment
COAST shall deploy from the descent/stowed position once on the surface of Mars.
Justification: Follows logical operation when landing on Mars.

COAST-MO-4.0 Launch Vehicle
COAST shall launch from Earth onboard the Space Launch System.
Justification: Aligns with launch requirements given in the AO.

COAST-MO-5.0 Surface Condition Survival
COAST shall survive Mars weather events during the duration of the mission.
Justification: Necessary for the success of the mission.

COAST-MO-6.0 Launch Survival
COAST shall survive launch from Mars surface into orbit.
Justification: Surviving launch from Mars surface into orbit is crucial to the crew's safety.

COAST-MO-7.0 Launch Configuration
COAST shall launch in a configuration such that Martian orbit can be reached.
Justification: The launch vehicle requirement places constraints on the configuration in order to reach the required destination.

COAST-MO-8.0 Deployed Configuration
COAST shall deploy in a configuration such that the propellant transfer rovers are able to autonomously exit the lander.
Justification: The rovers must autonomously exit the lander to complete the propellant transfer process.

COAST-MO-9.0 Risk Management
COAST shall perform immediate action to mitigate any risks rated "high" or "critical".
Justification: The system operations must be in compliance with NASA Mission Assurance Guidelines.

COAST-EPS-1.0 Fission Power Unit Transportation
COAST shall transport a minimum of 10 kW Fission Surface Power unit for the duration of surface operations.
Justification: The 10 kW Fission Surface Power unit is used to provide necessary power to execute the mission.

COAST-EPS-2.0 Communication Power
COAST shall allocate enough power for mission communications to Mars orbit and Earth.
Justification: Necessary for successful communications for the success of the mission.

COAST-EPS-3.0 Power Regulation
COAST shall regulate power application to all internal devices.
Justification: The power consumption must be regulated in order to not exceed the power requirements.

COAST-EPS-4.0 Mode Support
COAST shall meet or exceed the power needed for each system component to operate in the required modes.
Justification: The power provided must meet or exceed the required power for each system component to operate successfully.

COAST-EPS-5.0 Power Storage
COAST shall provide a means to store the required power to operate the MAV for 72 hours.
Justification: Since the fission power unit will be left on the surface of Mars, a power storage device must be used to supply the MAV with power for the duration of the ascent through rendezvous period.

COAST-ECLS-1.0 Life Support
COAST shall contain an Environmental Control and Life Support System for two crew members for 72 hours.
Justification: Requirement given in the AO.

COAST-ECLS-2.0 Dust Mitigation
COAST shall mitigate the Martian dust within the cabin designed for crew members,
Justification: The presence of dust within the cabin is hazardous to the health of the crew members during ascent from Mars.

COAST-ODCS-1.0 Heat Flux
COAST trajectories shall reduce exposure to heat flux.
Justification: Necessary for the thermal safety of the system.

COAST-ODCS-2.0 DST Rendezvous
COAST shall rendezvous with the DST in a 5-sol parking orbit around Mars.
Justification: Requirement given in the AO.

COAST-ODCS-3.0 Launch Window
COAST shall provide a primary and backup launch window for ascent from both Earth and Mars.
Justification: The system must plan for adverse weather conditions and potential scrubbed launches.

COAST-ADCS-1.0 Spacecraft Positioning
COAST shall provide accurate determination of the spacecraft's position.
Justification: Necessary for the successful completion of the mission.

COAST-ADCS-2.0 Attitude Adjustment
COAST shall be capable of adjusting attitude to rendezvous with the DST.
Justification: Rendezvous requirement given in the AO.

COAST-ADCS-3.0 Position Relative to DST
COAST shall determine position relative to the DST.
Justification: Necessary for the rendezvous requirement given in the AO.

COAST-ADCS-4.0 Autonomy During Blackouts
The COAST navigation system shall function autonomously during periods of communication outages up to two hours.
Justification: Aligns with the communication blackout mitigation requirement provided in the AO.

COAST-TMC-1.0 Communication Delays
COAST shall account for communication delays between the Mars sphere of influence and Earth Mission Control.
Justification: Aligns with the communication delay requirement provided in the AO.

COAST-TMC-2.0 DST Data Transmission
COAST shall transmit and receive data from the DST in orbit around Mars.
Justification: Necessary for successful rendezvous.

COAST-TMC-3.0 DSN Communications
COAST shall transmit and receive data from the DSN.
Justification: Necessary for successful communication.

COAST-TMC-4.0 Ascent Communication Blackouts
COAST shall mitigate the likelihood of communication blackouts during Mars ascent.
Justification: Aligns with the communication blackout mitigation requirement provided in the AO.

COAST-TMC-5.0 Positioning Communication Blackouts
COAST shall mitigate the likelihood of communication blackouts due to Earth-Mars positioning.
Justification: Aligns with the communication blackout mitigation requirement provided in the AO.

COAST-TMC-6.0 MCN Communication
COAST should communicate with the Mars Communication Network.
Justification: Utilizing the MCN would be helpful for achieving a successful communication system.

COAST-TMC-7.0 Emergency Communication
COAST shall provide emergency communication to Earth in the event of a system failure that jeopardizes the health of the crew.
Justification: Necessary for the human safety requirement in the AO.

COAST-TMC-8.0 Earth Antenna Bandwidth
COAST shall have an antenna bandwidth capable of communication with Earth.
Justification: Necessary for successful communications.

COAST-OBDH-1.0 On-Board Data Storage
COAST shall store scientific measurement data on onboard computers for at least one day (TBR).
Justification: Storage of data is necessary between periods of data uplink.

COAST-OBDH-2.0 Human Health Data
COAST shall store human health measurements on onboard computers for at least one day (TBR).
Justification: Human health measurements are critical to determining the crew's wellbeing during the mission.

COAST-PS-1.0 Propulsion System Storage
The COAST propulsion system shall be stored for a minimum of two years on the surface of Mars.
Justification: Aligns with the mission duration requirements outlined in the AO.

COAST-PS-2.0 Delta Velocity
The COAST propulsion system shall generate the ΔV required to reach a 5-sol orbit.
Justification: Aligns with the required orbit outlined in the AO.

COAST-PS-3.0 Propellant Safety Margin
COAST shall carry enough fuel for MAV mission duration, including a safety margin of 15% (TBR).
Justification: Necessary for the successful launch of the MAV.

COAST-PS-4.0 Propellant for ADCS
COAST shall provide the necessary propellant for all required ADCS maneuvers.
Justification: Propellant is needed for ADCS in addition to the transferred propellant.

COAST-TCS-1.0 Thermal Control
The COAST Thermal Control System shall regulate the temperatures onboard both landers and the MAV.
Justification: Necessary for the safety of mission systems and the crew.

COAST-TCS-2.0 Power Unit Heat Dissipation
COAST shall dissipate heat generated by the 10kW Fission Surface Power unit.
Justification: Necessary for the safety of mission systems.

COAST-TCS-3.0 Electronics Temperature Regulation
COAST shall regulate the temperatures of critical electronic and mechanical components.
Justification: Necessary for the safety of mission systems.

COAST-TCS-4.0 Entry, Descent, and Landing
COAST shall not exceed required internal temperatures during the entry, descent, and landing process.
Justification: The EDL process will expose the system to extreme heats which must be accounted for in the thermal system.

COAST-RB-1.0 Lander Relative Position
COAST shall provide accurate position determination of the landers relative to each other.
Justification: Necessary for successful propellant transfer.

COAST-RB-2.0 Hazard Navigation
COAST shall navigate to avoid hazards on the surface of Mars.
Justification: Avoiding hazards is critical to the success of propellant transfer.

COAST-STR-1.0 Payload Compartment
COAST shall have a payload compartment volume of at least 20,000 cm^3.
Justification: This accounts for the density of 50 kg of Martian rock samples, the most dense possible samples to be collected. 50 kg of return samples is dictated in the AO.

COAST-STR-2.0 Touchdown
COAST shall resist fracture from touchdown.
Justification: Fracture from touchdown can cause damage to instrument and landers, which are used in the propellant transfer process.

COAST-STR-3.0 Human Accessibility
COAST shall be accessible for astronauts to enter the Mars Ascent Vehicle before ascent from Mars,
Justification: There must be provisions for the crew to enter the cabin prior to launch in order to be returned safely.

Appendix B: Quad Chart

Georgia Institute of Technology

Crewed Orbit and Ascent Surface Transportation

Objectives & Technical Approach:

- The objective of Program COAST is to successfully land and fuel the Mars Ascent Vehicle (MAV) while on the surface of Mars, which will be used by two humans to ascend from the Martian surface to the Deep Space Transport in orbit.
- Performance goals include autonomous refueling and launch of MAV, reliable communication systems throughout nominal operations and blackout conditions, and safe transportation of humans and samples to DST.
- Technologies include scaled up MOXIE, ECLSS, 10kW Fission Surface Power unit

Image:

Key Design Details & Innovations:

	MAV	Land-On-Mars	ISLE	PHISH
Mass (kg)	7786.915	56798.763	60274.143	243.87
Power (W)	1756	602.5	13866.5	124.5
Volume (m^3)	637.743	1187	1187	50
Data Rates (kbps)	500	1000	1500	0.3

- Technologies, such as MOXIE, reduces landed mass by producing LOX on the surface of Mars
- The MAV showcases the capability of returning humans from Mars, enabling future crewed missions to Mars

Schedule:

- Baseline ConOps, stakeholders 2023
- Requirements, architecture, design 2025
- Landers depart Earth 2035
- Landers arrive on Mars July 2038
- Crew arrives on Mars Mid-2040
- MAV ready to support crew July 1, 2040

Cost:

- Total budget for the proposed concept: $9.13 Billion

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