Mars Sample Retrieval

Jaden Hernandez, Alayna Miller, Shreesh Nalatwad, Nischal Suhas

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***Abstract*— As the Perseverance rover on Mars continues to collect regolith samples, it needs a method to return those samples back to Earth for scientific evaluation. Subsequently, a system must be designed to use an existing launch vehicle to reach low Earth orbit (LEO), then propel itself from Earth's orbit towards Mars through a transfer orbit, finally entering into a capture orbit where it should deploy a landing vehicle to gather the samples from Perseverance. From here, the samples should be launched in a sample recovery capsule back into Low Mars Orbit, where they will ultimately return to Earth via direct reentry. Through extensive research and calculations, the optimal solution was determined to be a three-stage system composed of a kick stage, an orbiter, and a lander. More specifically, this system was developed based on required ΔV estimations at different mission phases, stage mass estimations, and mission timeline calculations, all of which were confirmed based on historical values from previous spacecraft such as the Atlas V, Centaur upper stage, Hayabusa, and OSIRIS-REx. With the mission officially starting on January 16, 2025, the system would attempt to leave LEO using the kick stage on November 29, 2026 based on calculated synodic periods of Mars and Earth. Shedding the kick stage after entering transfer orbit, the orbiter arrives at the Mars capture orbit on August 15, 2027, while simultaneously detaching the lander to retrieve the samples. By November 12, 2028, the lander will have obtained all samples from Perseverance, transferring those to the orbiter via a launch to low Mars orbit (LMO) which allows the orbiter to finally depart back towards Earth. By April 14, 2030, the mission will have been successfully completed, with the samples arriving back to Earth from the orbiter. Although the estimations and assumptions are not perfect, the design report attempts to justify each calculation made to explain the process behind the development of the system in order to ensure the success of the mission.**

Table of Contents

Figures

Tables

1 Introduction

The Mars Sample Return project was a semester-long design project that utilized all the concepts learned from throughout the AAE251 course to develop a complete Martian sample return. The project combined numerous aerospace engineering principles, starting from problem-definition and risk analysis, to intermediary calculations based on basic orbital dynamics and rocket sizing, and finally to 3-D modeling and preliminary design work. While completed in the context of an academic assignment, this project has numerous real-world applications as current Mars sample return missions are currently in various stages of development across the globe.

Mars has been a planet of immense interest ever since it was discovered. Its red hue captivated public interest and the belief of life on Mars inspired countless stories of Martian inhabitants. But it wasn't just the public that had a particular interest in Mars, scientists throughout the years have attempted to understand the Martian environment and its unique history. Rovers and probes such as Perseverance on the surface have provided incredible insight into Mars' chemical composition and have even proven that water existed at some point in the distant past.

However, there are limitations to the level of testing a rover or probe can do as opposed to a lab here on Earth. Scientifically significant samples from Mars, such as rock, soil, and atmosphere, need to be collected and returned from Mars so that scientists may complete more thorough and detailed analyses to better understand the mysteries of Mars. A sample return mission would help answer the questions of how much water existed on Mars, where it went and why, and also did life ever exist in the form of microbial organisms or potentially something bigger? Understanding the history of our neighboring planetary body can help scientists understand the likelihood of life coexisting inside of the solar system, as well as the sustainability of human life on another planet, especially as concerns due to the changing climate here on Earth continue to grow.

Sample return missions have the promise to significantly enhance the understanding of the Martian environment, except for the problem that we have never launched from the surface of Mars. The technology has yet to be developed and while Perseverance has been collecting samples for future sample return missions, there is currently no existing method to get them back home. We were tasked with designing a system that would launch from Earth and collect the samples on Mars from Perseverance and return safely to Earth for research.

It was important to make essential assumptions during the introductory phase of the project in order to simplify design decisions and calculations later on in the process. This process would also help to refine the scope of the system, basically what would the system need to be designed to accomplish and when we could assume pre-existing processes to fulfill mission objectives.

Listed below are the preliminary assumptions and their importance and impact on the overall design process:

- Sealing collections of samples is not a concern, as they are assumed to be securely sealed inside individual canisters and that Perseverance will handle the loading of samples (*Mars Sample Return Mission*, no date)
- Perseverance will be operational and running during the mission duration (*Mars Sample Return Mission,* no date)
- System will land in the perfect position and that Perseverance will be able to reach it.
    - Importance: These three assumptions decrease the scope of what the lander module is required to do on the surface of Mars as well as what the team is required to consider. There is no requirement to design a method of sample retrieval for the lander itself. Likewise, there is no need to design a method of sample loading as we can assume Perseverance will handle that. Assuming that Perseverance will be operational is highly important as the mission is designed around its capabilities. The technological and mathematical

intricacies of an interplanetary entry, descent, and landing at a specific point is neglected and assumed successful to simplify calculations. Therefore, the only design considerations for the project would be to design a lander that can hold samples and then launch from the surface.

- Our system is assumed to be successfully delivered to an Earth parking orbit by an existing launch vehicle.
  - Importance: This assumption once again refines the scope of the project to only focus on the sample retrieval system. Assuming the launch vehicle is successful means that we will not be required to design our own launch vehicle or consider risk mitigation for the initial launch.

- Mars' and Earth's orbits are circular with an orbit radius equal to their respective semi-major axis and are in the same ecliptic plane.
  - Importance: This simplifies orbital calculations and allows for the use of the Patched Conics Method for interplanetary calculations.

These are only some of the preliminary assumptions made prior to the technical calculations. Throughout the report, there will be more assumptions related to specific systems that would not have been appropriate to include within the high-level discussion of the introduction.

The remainder of this report is laid out as follows. Section 2 will detail the needs, stakeholders, and requirements of the system. Section 3 will dive into the design parameters including estimating ΔV requirements, estimating the mission timeline, and finally estimating system mass. Section 4 will discuss the concept generation, selection, and refinement with 3D models of the finalized system design. Finally, Section 5 will conclude the report with a design evaluation, and a discussion of next steps and lessons learned.

Throughout the remainder of the report, all of our work has been to fulfill the objectives and purpose of the mission. This is captured in the mission statement, stated as follows:

To return samples of regolith and rock from Mars safely back to Earth in order to study the chemical composition of the planet to further the understanding of NASA and global researchers and determine if Mars has ever or could ever host life.

2 Needs, Requirements, and Risk Analysis

Beginning any project requires a clear analysis on the needs, requirements, and potential risks to serve as the guidelines for the decision-making process throughout its lifecycle. For the Mars Sample Retrieval mission, we first outlined the important stakeholders that would be impacted the most by the mission outcomes. Using these stakeholders, we analyzed their individual needs, taking their objectives, interests, and risks into consideration. These needs are then addressed by the established requirements, providing a more comprehensive layout on the important goals for this mission. Finally, we perform a preliminary risk analysis based on several identified factors, and investigate numerous methods to mitigate them.

2.1 Stakeholders and Their Needs

To identify the stakeholders for this project, we analyzed the main goals for the Mars Return Mission and who would be impacted the most by its outcome. In this mission, the main goal is to collect Martian samples to gain scientific knowledge about Mars' climate and topography. Additionally, we also analyzed the direct financial contributors of the project who have a monetary stake in the success of the mission. We then expanded upon these primary stakeholders by considering how they affect other individuals as well in a sort of cascading effect. For example, we had to consider who benefits when the scientists perform their research on the sample other than the scientists themselves.

To identify the individual needs of each of the stakeholders, we conducted a comprehensive stakeholder analysis, considering each of their goals, interests, and responsibilities for the mission. Taking ethical and societal factors into consideration, we formulated a list of the most relevant needs for each stakeholder, providing us with valuable insights on the necessary elements for the success of the mission.

- **NASA (Organization):** NASA is the United States government agency that is responsible for the design and technology related to aerospace. Since NASA is the public figure representing the Mars Retrieval Mission, it has several needs to achieve their objectives. The first need is for the rock samples to be safely returned to Earth without being damaged during transit. This means that the mission would be a success, improving NASA’s public reputation and providing higher incentives for future projects. NASA’s second need is for the mission to have a negligible environmental impact which aligns with their commitment to sustainability and resource management in space. NASA’s third need would be for the system to be launched on November 26, 2026, the scheduled date of launch.

- **NASA Planetary Scientists:** The scientists working at NASA are tasked with analyzing the Martian samples after they are returned to Earth. One of their needs is for the samples to be safely retrieved from the Martian surface and into their designated containers. Another need that the scientists would have is for the samples to be securely contained to prevent contamination from outside particles that could potentially alter their data.

- **Mission Control Team:** The mission control team is responsible for monitoring the system during certain phases of the operation. Their need is for the system to transmit and receive data from Mission Control.

- **United Launch Alliance:** The System uses the RL10-C-1-1 Centaur Engine for the Kick stage of the mission, which is designed and created by the United Launch Alliance. ULA’s need would be for their engine to be successful for their public image.

- **Aerojet Rocketdyne:** For the Mars Sample Retrieval Mission, the system uses four Aerojet Rocketdyne MR-107S engines for the Orbiter Stage Capture. Their need would be for their engine to successfully work as intended for the mission, improving their company reputation and public image as a prominent aerospace company.

- **US Congress:** This is referring to Congress who is providing funding for this mission. They need the samples to be transported safely to Earth during transit to justify the funding that is being put into this project. Their second need would be for the system to be launched on November 26, 2026 which is the predicted launch date.

- **Global Scientific Community:** The global scientific community is the international interest related to the Mars Retrieval Mission. Their needs are for the samples to travel from Mars to Earth safely. This allows NASA scientists to analyze the sample properties, which will provide insights into Mars' composition. Another need from the global scientific community would be the transparent sharing of information from the mission to keep the community updated on important discoveries gained from the findings.
- **American Public:** The Mars retrieval mission captures significant interest from the American public. One of their needs is for the system to successfully transport the samples from Mars to Earth. Another need is for there to be transparent communication about the mission, dates, progress, findings, etc. to increase public involvement in the project.

Requirements

Need: The system transports Martian samples safely to Earth without being damaged during transit.

1. System should have enough total ΔV for each scheduled burn during the mission in order to ensure that the samples can return to Earth. A detailed discussion of what the values of the ΔV requirements are can be found in Section 3.3.
    a. This requirement is verifiable with vacuum hotfire testing on Earth to ensure propellant mass selected can produce required ΔV. It is dependent because it does not specify the number or type of engines which will be decided upon after more research.

2. Mars material samples should have no loss of sample mass in order to ensure no contamination of the sample and loss of valuable research material.
    a. This requirement is verifiable as *Perseverance* has already conducted sample mass measurements (*Mars Sample Return Mission*, no date) and can be verified with onboard system sensors. It is dependent because it does not specify the methods to mitigate sample loss.

3. System should be able to house 15kg of samples, that are themselves contained within individual 6in length sample canisters, to ensure safe tolerances inside the capsule (*Mars Sample Return Mission*, no date).
    a. This requirement is verifiable as testing on Earth with mock canisters can determine the viability of the capsule. It is dependent because it makes no mention as to the methods of storage, only that it will be able to store them.

4. The sample capsule heat shield must withstand extreme temperatures up to 3,000 °F during Earth re-entry to protect samples from damage (Hathcox et. al, 2003).
    a. This requirement is verifiable as temperature and vibration testing can be conducted to determine if chosen heat shield materials can withstand the

hazards or atmospheric reentry. This is dependent because it does not list specific materials.

Need: The system reaches Earth orbit safely.

1. The system should have a diameter of less than 5m to fit within the payload fairing of any launch vehicle being used to ensure no damage during launch (*Atlas V*, no date).
    a. This is verifiable as engineers can measure the system throughout the design process. It is dependent as it does not specify the shape of the design, only overall diameter.

Need: The system's components are not damaged throughout the mission lifecycle.

1. Propellants must be able to withstand extreme temperatures as low as -270 C in space in order to ensure proper propellant burn and ΔV requirements are met (Libal, 2023).
    a. This is verifiable as temperature testing of chosen propellants and subsequent chemical breakdown or reactions can provide insight into whether it is a viable option for a long-duration mission. This is dependent as any propellant chosen has to withstand these extremes and does not specify what type of propellant to use.

2. Electronics should receive 100 krad or less for device total ionizing dose (TID) to ensure proper function of mission critical systems such as flight computers and comms systems (LaBel, 2004).
    a. This requirement is verifiable as testing can be done on Earth to determine how much radiation is absorbed by a given component before and after mitigation tactics. This is dependent as there is no mention as to how the electronics would be shielded or resistant to radiation.

Need: The system has robust communication technology for Mission Control.

1. System should use ultra-high frequency technologies of at least 400 Megahertz, as seen on *Perseverance*, to allow for communication to Earth from Mars (*Communications*, no date).
    a. This requirement is verifiable as frequency of selected electronics can be tested on Earth prior to launch. This is dependent as it does not specify the type of technology.

Need: The mission is completed in a timely manner.

1. System should leave LEO within a day of November 29, 2026 and Mars departure should occur within a day of November 12, 2028, in order to ensure the mission is not delayed by an entire synodic period.
    a. This requirement is verifiable as percent of mission success can be determined prior to transfer orbit insertion to ensure whether or not to delay the stage. This requirement is dependent because it provides what the mission is required to do and not how it will do it.

The set of requirements are complete as it specifies all aspects of the system's main functions and properties. The most difficult requirement for the system to meet would undoubtedly be the ΔV requirement to retrieve the samples and return them to Earth. Determining the ΔV required is not so complex using patched conics, but finding systems that will be able to provide that amount of ΔV with a reasonable system mass is what makes the requirement so meticulous. This conflicts with the sizing requirements of the system, since it was determined that the system should have a diameter less than 5 meters in order to fit in the payload fairing of a launch vehicle. Because of this, the possibilities for the system become an even smaller pool because the system must generate a high amount of ΔV while having its size constrained. If the size was not constrained, it would be simpler to find means of theoretically generating any amount of ΔV necessary. This added complexity was thoroughly considered during the stage development in section 3.4.

2.2 PRELIMINARY RISK ANALYSIS

Part of designing an acceptable system is to understand all the ways it could fail, all the risks associated with the mission, and figure out ways to minimize them. No mission is ever risk-free but through careful analysis and mitigation, risk probability can be brought down to an acceptable level to give the go-ahead for launch.

Each stage of the mission comes with risks that could be catastrophic to the success of the mission. It's important to note that one of the main assumptions discussed in Section 1 was that the launch to Low Earth Orbit would be successful, therefore this risk analysis will omit mitigation of a launch failure (i.e. rapid unscheduled disassembly).

Risk #1: Mars transfer orbit maneuver fails

Given fuel leaks or improper thruster burn/orientation of burn, the system might not properly enter the planned transfer orbit to rendezvous with Mars. The main consequence of this would be that the system would be lost in space and on a course to potentially miss Mars entirely. However, with the efficiency and capabilities of current technologies, it is relatively unlikely that the computers would fail unless there is an unpredicted physical anomaly with the system. Even then, it would be far more likely for human error to play a part in a failed orbital insertion. The probability of an event like this happening is extremely low as only one historic mission has completely missed its target, Luna 1 back in 1959, which burned for too long on a Lunar transfer orbit and missed the Moon (*Luna 1*, no date).

However, given how catastrophic this event could be to the success of the mission, mitigation is necessary regardless of how low probability it may be. One such mitigation is to carry extra propellant to correct for small changes in expected location. This extra propellant should be well within the required ΔV for the mission, however, will likely not be enough for extreme deviations in trajectory. Prior to mitigation, the risk of missing Mars has low but catastrophic consequences and after mitigation, while the consequence is still high, the probability especially with the accuracy of computers and extra fuel drops the probability of this event even further.

Risk #2: Mechanical parts that are close together in the vacuum of space can get stuck together through a process called "cold-welding".

Given the extreme environment and vacuum in space, mechanical parts that are meant to be separate may become fused together and unable to move. Within the mission profile, this could impact such milestones as lander detachment and sample capsule rendezvous. If the lander is unable to detach due to cold-welding, this would spell the end of the mission and would likely transform objectives to be a data-gathering system with the orbiter. Mitigation tactics include minimizing the total moving parts of the system and for adding lubricant between moving parts if they are necessary to the system. Prior to mitigation , the probability of cold-welding is low likely in any instances where moving parts are very close together, but there is a grace period a little bit after leaving Earth because the system's surfaces are still oxidized. After mitigation, probability will drop but the mission consequences will likely still be high.

Risk #3: Spacejunk or asteroid collision

Given that there are around 23,000 pieces of space debris larger than 10 cm and about 100 million pieces of space debris larger than 1 mm, collision with our system could mean complete destruction or damage to critical systems such as flight computers and solar panels (Impey, 2023). Space debris and micro asteroids pose immense risk as even millimeter-sized objects will be traveling at speeds excess of 15,000 mph. Consequences include catastrophic loss of data, money, samples, and time from all parties who helped make the mission possible. Mitigation tactics include debris mapping and tracking prior to launch, as well as robust design with system critical components like computers deeper into the orbiter/lander body to protect from surface collisions. The probability of collision is low but grows the longer the system stays in LEO where the majority of space junk can be found. It would be a much lower risk if we have an understanding of where potentially dangerous solids are located. However, this risk is never completely mitigated because small objects can go undetected until collision.

Risk #4: Space Radiation and Charged Particles

As the system travels through space, radiation from the Sun and charged particles from galactic cosmic rays and within the Van Allen belt poses significant risk to electronics on board and can degrade the effectiveness of paints and protective materials. Data within electronics can be lost or changed (i.e. bit flip) when the system encounters radiation. Which, depending on the process affected, can impact burn times and telemetry data. Mitigation includes shielding electronics with lead, which will block gamma rays from entering the enclosure. Also utilizing a bus-style design for the orbiter to house most of the mission critical electronics within protective materials can decrease the effects of radiation. Radiation hardening can also be used to design robust electronics that are resistant to radiation effects. The spacecraft will encounter radiation at all times but the probability to cause data loss is low but can create significant consequences.

Risk #5: Mars Collision

Given that there could be an anomaly in altitude measurements, unprecedented weather events, or unexpected terrain at the landing site, the lander module could collide with the Martian surface if it does not slow down enough for soft landing. Weather events could decrease the effectiveness of the parachute landing system and anomaly in altitude measurements could provide faulty data for parachute deployment. The system could sustain damage depending on how fast it is moving when landing on the planet. This damage could be as significant as destroying the entire system. Mitigation could be to use redundancy in telemetry data, data gathered from the lander itself as well as data from the orbiter above. This would mitigate any errors in the lander data as it can check distance from the orbiter. In terms of the parachute system, small thrusters can be added for course correction should winds push the descent off course. The design of the lander legs should be robust and tested to withstand high-impulse landings. While we assume the landing to be perfect, it is still important to discuss these risks. The risk of computer anomaly and unexpected weather would not be lowered after mitigation, but the probability of damage to the lander should these happen is greatly lowered.

Risk #6: Sample canister failure

While we assume that Perseverance can get the canisters into the system safely, there is still inherent risk with the canisters failing within the sample capsule module. The scope of this project means we must focus on our system in mitigating these effects rather than discussing the canisters themselves. Improper mitigation of sample canister failure could mean sample contamination and overall mission failure as researchers would not be able to properly analyze the samples as pure Martian specimens. Preventing contamination and loss would include robust design of the capsule to house broken canisters and a system of capsule-within-capsule to ensure samples still be protected inside the bigger capsule. The probability of canister failure is highly unlikely as years of research would have been conducted prior to the launch of *Perseverance*, however, making sure our capsule is robust and designed with potential failures in mind, the consequences can be greatly decreased.

There are more risks associated with this mission, however, these six describe six different ways the mission could fail. Within each risk there could be dozens of ways for the mission to fail, for example, we could make a risk analysis of each electronic component and their reaction to radiation. This, however, would likely be taken into consideration with a more thorough risk analysis and not in a preliminary proposal.

Table 1: Risk Matrix Before Mitigation

LIKELIHOOD	**5**			4		
	4					
	3				3	2, 5
	2				6	1
	1					
		1	**2**	**3**	**4**	**5**
	CONSEQUENCES					

Table 2: Risk Matrix After Mitigation

	5					
	4					
	3			3		
LIKELIHOOD	**2**		6	4	5	
	1			2	1	
		1	**2**	**3**	**4**	**5**
	CONSEQUENCES					

Table 1 is a risk matrix of the above risks plotted along probability and mission consequences should the risk occur. Consequences are on a 1-5 scale with 1 being minimal mission effects to 5 being mission failure. Likelihood is defined on a quantitative scale for our purposes of L1 referring to a less than 1/10000 chance in occurring to L5 being a near certainty. The only L5 is encountering radiation as there is no possible way to somehow dodge radiation in space.

Table 2 is the risk matrix after taking into account the mitigation tactics discussed in detail for each risk. Notice, there are still risks in the “caution” yellow sections of the table, which indicate that risks can be successfully minimized but will likely always pose a threat.

3 Estimating Design Parameters

This section will describe mission critical design parameters and the calculations made to estimate their values. It will start with overall mission design and describe the process to estimate the mission timeline. Finally it will describe the derivations to estimate ΔV requirements for different mission milestones which will lead into system staging, propellant selection, and initial mass estimates.

3.1 Mission Design

The mission only proceeds with specific payload estimates in mind regarding the dimensions and mass. It is given that the samples being returned from Mars have a mass of 15 kg. The sample return capsule would be 50 cm in diameter, which should be much smaller than the stage carrying the capsule because there are numerous other payload components to consider, such as entry, descent, and landing (EDL) gear, heat shields, and docking equipment as necessary. The EDL equipment was estimated to be around 235 kg based on documentation of the "aeroshell" from NASA's rover missions (*Mars Exploration Rover Mission: The Mission*, no date). On the other hand, any docking equipment the system may end up using would be at most 110 kg based on information from eoPortal (*IDSS*, 2022).

The sample return mission is described as follows:

1. Our system waits for the orbital paths of Earth and Mars to align prior to launch as determined by phase angle calculations described in section 3.2.
2. After alignment has been achieved, our system starts on a launchpad, attached to an Atlas V launch vehicle.
3. The Atlas V launches from the launchpad, carrying our system with it. The launch vehicle continues until the system has achieved a circular parking orbit around Earth.
4. From this point, our system will receive ΔV from the Centaur upper stage as it detaches in order to be placed on a transfer orbit to Mars.
5. Our system will then coast towards Mars.

6. Prior to reaching Mars, the lander will detach from the orbiter and continue on a path for direct entry to the Martian surface.
7. The orbiter will then reach Mars' orbit and burn into a highly eccentric capture orbit, utilizing aerobraking to circularize into a parking orbit around Mars.
8. During the process of aerobraking, the lander module will land on the equator of Mars and the Perseverance rover will meet the lander.
9. Perseverance will then load the samples from the rover onto the craft.
10. After all samples are collected, the sample capsule launches from the Martian surface, leaving the entry, descent, and landing gear behind, and rendezvous with the orbiter.
11. The now combined orbiter and sample capsule system will wait once again for the orbital paths of Earth and Mars to align so it may return to Earth.
12. The system departs Martian orbit and is placed on a transfer orbit back to Earth.
13. The sample capsule is detached from the orbiter and safely returned to the surface of Earth for research and study.

The system will launch from the equator at a latitude of 0 degrees. While there are no current launch stations exactly on the equator, the Guiana Space Center in Kourou, French Guiana will provide the best option for launch. In order to simplify calculations we assumed an equator launch for preliminary purposes, however, a launch at GSC would be the best option either way to enter a 0^{o}-inclination parking orbit to correspond with the ecliptic plane of Mars and the Earth. Launching with an azimuth of 90^{o} at GSC will also allow the launch vehicle to need less propellant mass as launching due east would maximize the Earth's help for ΔV required based on the Earth's angular velocity.

3.2 Mission Timeline

Table 3: Dates For Each Mission Phase

Phase	Date	Wait Time (days)
Mission Begins	01/16/2025	
Departure from Earth	11/29/2026	683.64
Arrival at Mars	08/15/2027	258.90
Departure from Mars	11/12/2028	454.02
Arrival at Earth	04/14/2030	517.87
Total Travel Time	**1914.56 days**	

Table 3 describes the important dates for the mission while also including the "wait time" in days, which represents how much time elapses in between each phase. These are preliminary estimates based on simplifying assumptions, but nonetheless, these estimates have a complex mathematical basis behind them.

3.2.1 Equations for Time Estimates

In order to calculate the time for each phase in the mission, there are four important values that need to be found: T_{wait}, T_{trans}, T_{syn}, and T_{backup}. These all depend on a few parameters, some of which are constants, and some of which have to be calculated.

Table 4: Constants Used For Time Estimate Calculations (Weber, no date)

Variable	Description	Value	Units
μ_S	Sun's Gravitational Parameter	$1.327 \cdot 10^{11}$	km^3/s^2
r_{SM}	Radius of Sun-Mars Orbit	227,990,400	kilometers
r_{SE}	Radius of Sun-Earth Orbit	149,600,000	kilometers
a_{trans}	Transfer Orbit Semi-Major Axis	188,795,200	kilometers
γ_r	Reference Phase Angle	0	rad

It is assumed that upon launch, Earth and Mars will be perfectly aligned in opposition, which is why γ_r in Table 4 is assumed to be 0 radians at this point.

Obtaining T_{trans} is singlefold. Its evaluation is as follows:

$$T_{trans} = 2\pi\sqrt{\frac{a_{trans}^3}{\mu_S}} = 2\pi\sqrt{\frac{(188795200\ km)^3}{1.327\cdot10^{11}\ km^3/s^2}} \approx 517.87\ days$$

Therefore, the system will spend 517.87 days in the transfer orbit. This value, and other time values, did not originally output in days; they were converted manually by dividing each value by 86400 seconds to convert to days.

Prior to calculating the wait time parameter, T_{wait}, the orbital period of both Mars (T_{SM}) and Earth (T_{SE}) need to be estimated. This can be done via the following equations:

$$T_{SM} = 2\pi\sqrt{\frac{r_{SM}^3}{\mu_S}}\ \&\ T_{SE} = 2\pi\sqrt{\frac{r_{SE}^3}{\mu_S}}$$

$$T_{SM} = 2\pi\sqrt{\frac{(227990400\ km)^3}{1.327\cdot10^{11}\ km^3/s^2}} \approx 687.24\ days$$

$$T_{SE} = 2\pi\sqrt{\frac{(149600000\ km)^3}{1.327\cdot10^{11}\ km^3/s^2}} \approx 365.28\ days$$

Although perhaps not flawlessly precise, the values do have merit. It can be observed that T_{SE} obtains around the same amount of days as the amount of days in an Earth calendar year, 365. With T_{SM} and T_{SE} obtained, their respective mean motions (the average rotational velocity around the orbit), n_M for Mars and n_E for Earth, need to be calculated next to proceed towards acquiring a T_{wait} value.

$$n_M = \frac{2\pi}{T_{SM}} = \frac{2\pi}{59377536\ s} \approx 1.058 \cdot 10^{-7}\ rad/s$$

$$n_E = \frac{2\pi}{T_{SE}} = \frac{2\pi}{31560192\ s} \approx 1.991 \cdot 10^{-7}\ rad/s$$

The mean motions can then help to identify the phase angle at departure (γ_d) and at arrival (γ_a). For the purpose of simplicity, Δt will refer to $\frac{T_{trans}}{2}$. The phase angles are defined accordingly:

$$\gamma_d = \pi - n_M \cdot \Delta t = \pi - 1.058 \cdot 10^{-7}\ rad/s \cdot \frac{44738784\ s}{2} \approx 0.7742\ rad$$

$$\gamma_a = \pi - n_E \cdot \Delta t = \pi - 1.991 \cdot 10^{-7}\ rad/s \cdot \frac{44738784\ s}{2} \approx 4.971\ rad$$

More readably speaking, γ_d would be a departure phase angle of 44.36° while γ_a would be an arrival phase angle of 284.82°.

At last, T_{wait} can be obtained with the help of the phase angle and mean motions. In succinct fashion, it expresses itself as:

$$T_{wait} = \frac{\gamma_d}{n_M - n_E} = \frac{0.7742\,rad}{1.058 \cdot 10^{-7}\,rad/s - 1.991 \cdot 10^{-7}\,rad/s} \approx -\ 96.08\ days$$

This means that the system would have had to depart around 96 days *before the mission* even started. If the mission's start date cannot move from January, then the system has no choice but to wait until a backup launch date that actually takes place after it reaches low Earth orbit, which is where T_{syn} and T_{backup} come into play.

T_{syn} refers to the "synodic period" between Earth and Mars. The synodic period determines how long it will take before these two planets are perfectly aligned again, i.e. the scenario where $\gamma = \gamma_r = 0$. Thus, T_{backup} is essentially just a formal expression of how many synodic periods must pass before the system is ready to launch again. The expression used for T_{syn} in this specific scenario was:

$$T_{syn} = \left|\frac{1}{T_{SE}} - \frac{1}{T_{SM}}\right|^{-1}$$

This is just a rewritten form of the equation $T_{syn} = \frac{2\pi}{|n_2 - n_1|}$ retrofitted to work in MATLAB, where the majority of calculations were done.

$$T_{syn} = \left|\frac{1}{31560192\,s} - \frac{1}{59377536\,s}\right|^{-1} \approx\ 779.72\ days$$

Following the logic established before, T_{backup} should just then be a product of T_{syn} and some variable, "n", determining how many times the system has waited, like so:

$$T_{backup}(n) = T_{wait} + (T_{syn} \cdot n)$$

In this scenario, $n \geq 1$ because it is known from the original T_{wait} that not waiting out the synodic period at least once is less ideal with a base T_{wait} value of -96.08 days, which would force a departure in October 2024; this is far too soon and unnecessarily rushes the mission. There are two instances accounted for in the design for this mission: $n = 1$ and

$n = 2$, although the system could theoretically wait as long as it needs to assuming that backup time is within its service life.

$$T_{backup}(1) = T_{wait} + T_{syn} =- 96.08\,days + 779.72\,days = 683.64\,days$$

$$T_{backup}(2) = T_{wait} + T_{syn} =- 96.08\,days + 779.72\,days \cdot 2 = 1463.36\,days$$

Ideally, the system would only utilize the value of $T_{backup}(1)$, meaning that it would depart from Earth at most 684 days from the start of the mission.

3.2.2 Supplemental Estimated Datesission Timeline

While the planned departure date is 11/29/2026, there are other possibilities, such as if ground control decides that the system should wait another synodic period before departing to Mars. In this instance, in the context of T_{backup}, its "n" variable would equal 2. If this were to happen, the mission dates would be adjusted significantly from the planned dates such that they would fall out of the date constraint of December 31, 2030.

Table 5: Dates For Each Mission Phase, Backup Scenario

Phase	Date	Wait Time (days)
Mission Begins	01/16/2025	
Departure from Earth	01/17/2029	1463.36
Arrival at Mars	10/04/2029	258.90
Departure from Mars	01/01/2031	454.02
Arrival at Earth	06/02/2032	517.87
Total Travel Time	**2694.28 days**	

Obviously, mission control probably would not want to wait this long, and so departing after the first synodic period should be a priority because the backup scenario would cause the mission to end nearly 2 years later than originally planned.

Although it is possible to not wait out the synodic period and launch in October 2024, it would lead to the development of the system being somewhat rushed. Because waiting out the synodic period would put the mission end in 2030, it still is within the date range constraint and it would make more sense to use as much time as possible before launching to be completely certain that every system works properly and to make any refinements necessary to the mission.

3.3 ΔV Estimation

To estimate the ΔV required for each phase of the mission, MATLAB was utilized to streamline calculations. Refer to appendix section "A.1" for further information about the program.

Table 6: ΔV Estimations For Each Mission Phase

Phase	ΔV Required (km/s)	Involved System(s)
To Low Earth Orbit	*8.80*	*Launch Vehicle*
Earth Departure	3.55	Kick stage
Mars Orbit Capture	0.85	Orbiter
To Low Mars Orbit	3.54	Lander
Mars Departure	2.11	Orbiter
Overall Mission	**10.05**	**Kick stage, Lander, Orbiter**

Table 6 details the ΔV requirements for each crucial portion of the mission, though there are various intermediate calculations that occurred in between to obtain those values.

3.3.1 ΔV Estimation of Launch to Low Earth Orbit (LEO)

Although not included in the overall ΔV required for the system being developed, it was important to understand the ΔV requirements of a launch vehicle to get the designed system to low Earth orbit in order to determine which launch vehicle should be used for this mission.

All calculations use the following standard parameters:

Table 7: Constants Used For ΔV Calculations (Weber, no date)

Variable	Description	Value	Units
r_E	Equatorial radius of Earth	6,378	kilometers
r_M	Radius of Mars	3,390	kilometers
μ_E	Earth's Gravitational Parameter	$3.986 \cdot 10^{5}$	km^3/s^2
μ_S	Sun's Gravitational Parameter	$1.327 \cdot 10^{11}$	km^3/s^2
μ_M	Mars' Gravitational Parameter	$0.428 \cdot 10^{5}$	km^3/s^2
ω_E	Earth's Angular Velocity	$7.270 \cdot 10^{-5}$	rad/s
ω_M	Mars' Angular Velocity	$7.058 \cdot 10^{-5}$	rad/s
r_{SM}	Radius of Sun-Mars Orbit	227,990,400	kilometers
r_{SE}	Radius of Sun-Earth Orbit	149,600,000	kilometers
La	Launch Latitude	0	degrees
Az	Launch Azimuth	90	degrees
$\Delta V_{loss,E}$	Assumed ΔV Losses Near Earth	1.65	km/s

$\Delta V_{loss,M}$	Assumed ΔV Losses Near Mars	0.30	km/s
$r_{park,E}$	Radius of Earth Parking Orbit	6,878 (500 + r_E)	kilometers
$r_{park,M}$	Radius of Mars Parking Orbit	3,540 (150 + r_M)	kilometers
$r_{cap,M}$	Radius of Mars Capture Orbit	3,490 (100 + r_M)	kilometers

To begin with, LEO ΔV calculations begin with the base amount of velocity required to get up to LEO, $\Delta V_{launch,E}$, retrieved from a simplified Vis-Viva equation.

$$\Delta V_{launch,E} = \sqrt{\frac{\mu_E}{r_{park,E}}} = \sqrt{\frac{3.986 \cdot 10^5\ km^3/s^2}{6{,}878\ km}} \approx 7.61\ km/s$$

Also included is the "Earth help term", ΔV_{EH}, which determines how much the launch latitude and azimuth will affect the efficiency of the launch.

$$\Delta V_{EH} = \omega_E \cdot r_E \cdot cos(La) \cdot sin(Az) = 7.270 \cdot 10^{-5} rad/s \cdot 6378\ km \cdot cos(0) \cdot sin(90)$$

$$\Delta V_{EH} \approx 0.463\ km/s$$

Of course, also considered is the estimated amount of ΔV losses assumed in table 7, $\Delta V_{loss,E}$, where $\Delta V_{loss,E} = 1.65\ km/s$. Subsequently, the total ΔV required to reach LEO can be expressed as follows:

$$\Delta V_{LEO} = \Delta V_{launch,E} + \Delta V_{loss,E} - \Delta V_{EH} = 7.61\ km/s + 1.65\ km/s - 0.463\ km/s$$

$$\Delta V_{LEO} \approx 8.80\ km/s$$

So, based on this estimation, the launch vehicle chosen to carry the system would have to be capable of meeting this ΔV requirement of 8.80 km/s.

3.3.2 ΔV Estimation of Earth Departure

Once the system is in low Earth orbit, the next step is for the kick stage to depart from Earth's orbit. The associated ΔV estimation is significantly more nuanced than it was for getting to LEO, with five intermediate terms involved: $V_{S,p}$, V_{SE}, $V_{excess,E}$, $V_{E,p}$, and $V_{park,E}$.

To start, the three parameters of $V_{excess,E}$, $V_{S,p}$ and V_{SE} need to be calculated. $V_{S,p}$ refers to the system's velocity at the Sun transfer orbit periapsis in km/s, while V_{SE} refers to the velocity of Earth in its assumed circular orbit around the Sun. Both utilize variations of the Vis-Viva equation such that:

$$V_{S,p} = \sqrt{\frac{2\mu_S \cdot r_{SM}}{r_{SE} \cdot (r_{SM} + r_{SE})}} = \sqrt{\frac{2 \cdot 1.327 \cdot 10^{11} km^3/s^2 \cdot 227990400\ km}{149600000\ km \cdot (227990400\ km + 149600000\ km)}} \approx 32.73\ km/s$$

$$V_{SE} = \sqrt{\frac{\mu_S}{r_{SE}}} = \sqrt{\frac{1.327 \cdot 10^{11} km^3/s^2}{149600000\ km}} \approx 29.78\ km/s$$

$V_{S,p}$ is a high value for a system to achieve, however what V_{SE} reveals is that the system is already traveling at Earth's velocity, so the additional velocity required to reach the transfer orbit would be the difference between the two, $V_{excess,E}$. Calculating $V_{excess,E}$ becomes simple once $V_{S,p}$ and V_{SE} have been calculated, since $V_{excess,E} = V_{S,p} - V_{SE}$ which yields that $V_{excess,E} \approx 2.95\ km/s$. All that is to say that the additional velocity necessary to get into the transfer orbit from Earth's orbit is 2.95 km/s.

Knowing $V_{excess,E}$, it can be used to calculate another parameter, $V_{E,p}$, which refers to the velocity of the system at Earth's periapsis. $V_{E,p}$ is calculated as such:

$$V_{E,p} = \sqrt{V_{excess,E}^{\ \ 2} + \frac{2\mu_E}{r_{park,E}}}$$

But where does this expression come from? The expression for $V_{E,p}$ can be derived from the equations relating to the energy possessed by the system. More specifically, the kinetic energy and potential energy of the system at this moment in time can be expressed as:

$$K = \frac{1}{2}mv^2 \ \& \ P = -\frac{\mu_E m}{r}$$

The specific energy of the system, which is the energy per unit mass, is just the sum of these two expressions excluding the mass such that:

$$E = \frac{1}{2}v^2 - \frac{\mu_E}{r}$$

But "r" needs to approach infinity in order for the orbit transfer to occur, leading to:

$$E = \frac{1}{2}v^2 - \frac{\mu_E}{\infty} = \frac{1}{2}V^2_{excess,E}$$

Substituting this value of E into the equation:

$$E = \frac{1}{2}v^2 - \frac{\mu_E}{r} \Rightarrow \frac{1}{2}V^2_{excess,E} = \frac{1}{2}V^2_{E,p} - \frac{\mu_E}{r_{park,E}}$$

Solving for $V_{E,p}$ gives the asserted equation, $V_{E,p} = \sqrt{V_{excess,E}{}^2 + \frac{2\mu_E}{r_{park,E}}}$ from above.

$$V_{E,p} = \sqrt{(2.95\ km/s)^2 + \frac{2 \cdot 3.986 \cdot 10^5 km^3/s^2}{6878\ km}} \approx 11.16\ km/s$$

Additionally, the velocity of the system in its parking orbit around Earth, $V_{park,E}$, must also be calculated in order to finally find the total ΔV required to depart from Earth entirely, $\Delta V_{dep,E}$.

$$V_{park,E} = \sqrt{\frac{\mu_E}{r_{park,E}}} = \sqrt{\frac{3.986 \cdot 10^5 km^3/s^2}{6878\ km}} \approx 7.61\ km/s$$

Using the obtained value of $V_{park,E}$, $\Delta V_{dep,E}$ is the difference between $V_{E,p}$ and $V_{park,E}$ such that:

$$\Delta V_{dep,E} = V_{E,p} - V_{park,E} = 11.16\ km/s - 7.61\ km/s = 3.55\ km/s$$

Therefore, from those five parameters, it takes 3.55 km/s of ΔV to depart from Earth.

3.3.3 ΔV Estimation of Mars Arrival, Entry, and Departure

After the system departs from Earth, its next phase would be a Mars orbit capture. From there, the lander will descend upon Mars, retrieve the samples, and then return to low Mars orbit (LMO) where the orbiter stage will rendezvous with it. Finally, the orbiter and samples will depart from Mars back to Earth.

Beginning with the ΔV required for the Mars orbit capture phase, $\Delta V_{cap,M}$, the estimation requires similar parameters to the Earth departure phase: $V_{S,a}$, V_{SM}, $V_{excess,M}$, $V_{cap,M}$, and $V_{M,p}$. The process will be the same as Earth's departure with different inputs: calculate $V_{S,a}$ and V_{SM} to obtain $V_{excess,M}$, use $V_{excess,M}$ to find $V_{cap,M}$ while then calculating $V_{M,p}$. $\Delta V_{cap,M}$ is equal to the difference between $V_{cap,M} - V_{M,p}$.

In this phase, a new constant, $a_{cap,M}$, will be introduced to represent the semi-major axis of the capture orbit, such that $a_{cap,M} = \frac{(r_{cap,M} + (r_M + 40000\ km))}{2} = 23440\ km$. Now, the intermediate calculations for $\Delta V_{cap,M}$ proceed as described:

$$V_{S,a} = \sqrt{\frac{2\mu_S \cdot r_{SE}}{r_{SM} \cdot (r_{SE} + r_{SM})}} = \sqrt{\frac{2 \cdot 1.327 \cdot 10^{11} km^3/s^2 \cdot 149600000\ km}{227990400\ km \cdot (149600000\ km + 227990400\ km)}} \approx 21.48\ km/s$$

$$V_{SM} = \sqrt{\frac{\mu_S}{r_{SM}}} = \sqrt{\frac{1.327 \cdot 10^{11} km^3/s^2}{227990400\ km}} \approx 24.13\ km/s$$

$$V_{excess,M} = |V_{S,a} - V_{SM}| = |21.48\ km/s - 24.13\ km/s| = 0.85\ km/s$$

$$V_{cap,M} = \sqrt{\mu_M (\frac{2}{r_{cap,M}} - \frac{1}{a_{cap,M}})} = \sqrt{0.428 \cdot 10^5 km^3/s^2 (\frac{2}{3490\ km} - \frac{1}{23440\ km})} \approx 4.77\ km/s$$

$$V_{M,p} = \sqrt{\frac{V_{excess,M}^2 + 2\mu_M}{r_{cap,M}}} = \sqrt{\frac{(0.85\ km/s)^2 + 2 \cdot 0.428 \cdot 10^5 km^3/s^2}{3490\ km}} \approx 5.62\ km/s$$

$$\Delta V_{cap,M} = |V_{cap,M} - V_{M,p}| = |4.77\ km/s - 5.62\ km/s| = 0.85\ km/s$$

The ΔV required to arrive at the Mars capture orbit is 0.85 km/s.

After the lander module obtains the samples from the Perseverance rover, it needs to return back to the orbiter, which awaits it in the Mars parking orbit. The ΔV required to rendezvous with the orbiter, referred to as ΔV_{LMO}, is predictably calculated similarly to ΔV_{LEO}. There are three variables involved: $\Delta V_{launch,M}$, ΔV_{MH}, $\Delta V_{loss,M}$. $\Delta V_{loss,M}$ is an assumed constant per table 7, (0.3 km/s) but $\Delta V_{launch,M}$, the ΔV required to launch from Mars, will need to be calculated with the Vis-Viva equation as shown:

$$\Delta V_{launch,M} = \sqrt{\frac{\mu_M}{r_{park,M}}} = \sqrt{\frac{0.428 \cdot 10^5 km^3/s^2}{3540\ km}} \approx 3.48\ km/s$$

Next, ΔV_{MH}, which is the "Mars help term", can be defined by these means:

$$\Delta V_{MH} = \omega_M \cdot r_M \cdot cos(La) \cdot sin(Az)$$

To maximize the Mars help term, it is important to note that both La and Az will be the same as on Earth; this is a slight oversimplification, but it is assumed (and used advantageously) that the Perseverance rover can meet the lander at any point. Therefore, in this simplified model, we operate under the assumption that the lander can be programmed to land at the Martian equator and launch with an azimuth of 90° to minimize ΔV requirements during ascent. Substituting the known constants into this equation produces:

$$\Delta V_{MH} = 7.058 \cdot 10^{-5} rad/s \cdot 3390\, km \cdot cos(0) \cdot sin(90) \approx 0.24\, km/s$$

Computing ΔV_{LMO} is trivial beyond these calculations considering the equation:

$$\Delta V_{LMO} = \Delta V_{launch,M} + \Delta V_{loss,M} - \Delta V_{MH}$$

$$\Delta V_{LMO} = 3.48\, km/s + 0.3\, km/s - 0.24\, km/s = 3.54\, km/s$$

The ΔV required for the lander to rendezvous with the orbiter is 3.54 km/s.

Finally, once the lander has met with the orbiter, it will need to depart from Mars' orbit back towards Earth. This ΔV estimation will follow the same process as the Earth departure ΔV estimation, having four values: V_{SM}, $V_{excess,M}$, $V_{M,p}$, $V_{park,M}$.

V_{SM}, the velocity of the system in the Sun-Mars orbit, is found simply using Vis-Viva:

$$V_{SM} = \sqrt{\frac{\mu_S}{r_{SM}}} = \sqrt{\frac{1.327 \cdot 10^{11}\, km^3/s^2}{149600000\, km}} \approx 24.13\, km/s$$

To find $V_{excess,M}$, V_{SM} will be used, but $V_{S,a}$ from the Mars orbit capture, which was equal to 21.48 km/s, will also be reused such that $V_{excess,M} = |\, V_{S,a} - V_{SM}|$.

$$V_{excess,M} = |\, 21.48\, km/s - 24.13\, km/s| = 2.65\, km/s$$

With $V_{excess,M}$ known, it can be utilized to calculate $V_{M,p}$. More specifically:

$$V_{M,p} = \sqrt{V^2_{excess,M} + \frac{2\mu_M}{r_{park,M}}}$$

$$V_{M,p} = \sqrt{(2.65\ km/s)^2 + \frac{2\cdot 0.428\cdot 10^5 km^3/s^2}{3540\ km}} \approx 5.59\ km/s$$

Lastly, $V_{park,M}$, the velocity of the Mars parking orbit, is needed before finally calculating $\Delta V_{dep,M}$. This is found with a simpler Vis-Viva than $V_{M,p}$ composed simply by:

$$V_{park,M} = \sqrt{\frac{\mu_M}{r_{park,M}}} = \sqrt{\frac{0.428\cdot 10^5 km^3/s^2}{3540\ km}} \approx 3.48\ km/s$$

To conclude, $\Delta V_{dep,M}$ is just the difference between $V_{M,p}$ and $V_{park,M}$.

$$\Delta V_{dep,M} = V_{M,p} - V_{park,M} = 5.59\ km/s - 3.48\ km/s = 2.11\ km/s$$

The ΔV required for the orbiter to depart from Mars' orbit is 2.11 km/s.

3.3.4 Total Mission ΔV Estimation and Conclusion

The total ΔV required for the mission is the sum of all of the ΔV estimations for the intermediate phases. Therefore:

$$\Delta V = \Delta V_{dep,E} + \Delta V_{cap,M} + \Delta V_{dep,M} + \Delta V_{LMO}$$

Substituting known values into the equation gives:

$$\Delta V = 3.55\ km/s + 0.85\ km/s + 3.54\ km/s + 2.11\ km/s = 10.05\ km/s$$

The mission overall has a ΔV requirement of 10.05 km/s from the system according to these calculations.

3.4 Stages, Propellants, and Initial Weight

The design features three stages overall: the kick stage, orbiter, and lander. The kick stage is primarily responsible for departing from Earth's orbit. The orbiter's purpose is mainly to transport the lander and samples. Upon entering the transfer orbit, the kick stage is shed by the orbiter. Right before the orbiter enters the Mars capture orbit., it will release the lander to descend (through direct entry) upon the Mars surface to rendezvous with the Perseverance rover. After gathering samples from the rover, the lander launches back up to LMO to transfer the samples to the orbiter. The orbiter finally departs with the samples, leaving the lander behind.

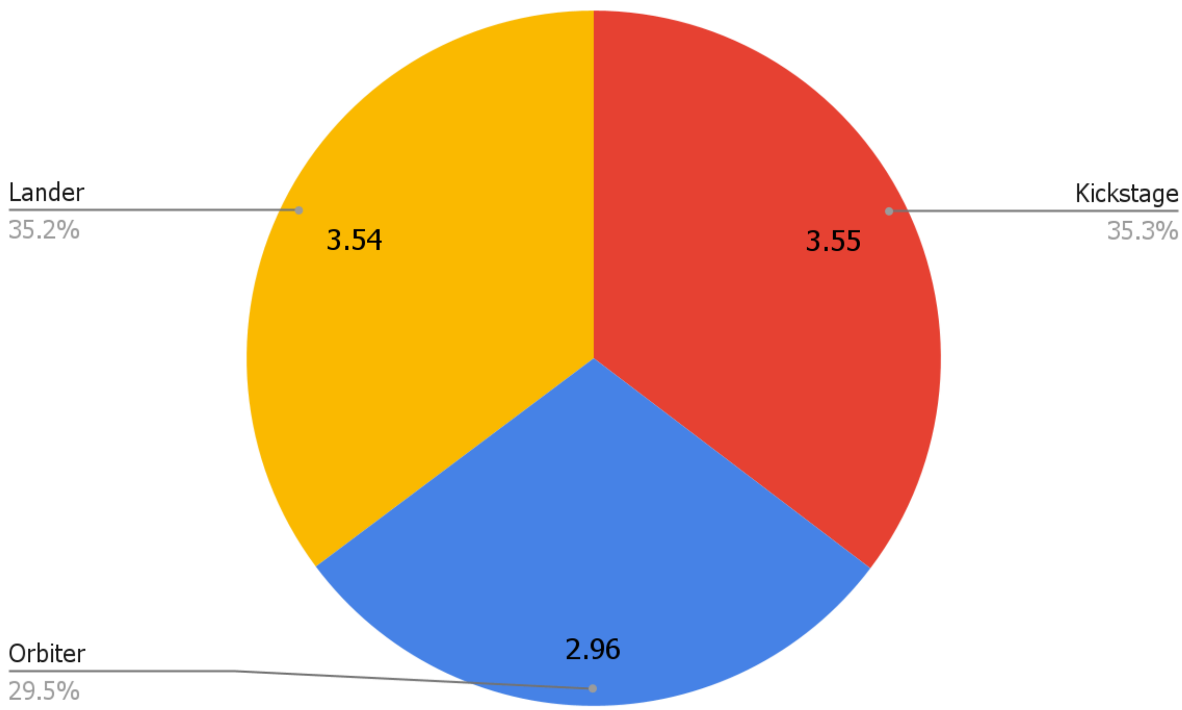

Figure 1: System Staging ΔV Distribution (in km/s)

Through referencing section 3.3, figure 1 describes the distribution of ΔV for the three stages. It is important to note that while the lander needs to launch to LMO, it is not necessarily displacing much (it simply returns back to the orbiter where it came from) despite having a considerable amount of ΔV required. The kick stage requires the majority of the ΔV at 35.3%, while the orbiter requires less ΔV at 29.5% since it is only responsible for the Mars capture orbit and Mars departure.

3.4.1 Propellant Analysis

Table 8: Propellant Choice For Each Stage

Stage	Propellant Type	Specific Impulse (s)
Kick stage	LOX/LH_2	448
Orbiter	Hydrazine monopropellant	328
Lander	MMH/NTO (MON-3)	293

Table 8 details the propellant used for each stage as well as the specific impulse (I_{sp}) for each stage. These values are crucial to the eventual mass estimations of the stages because the propellant mass is determined partially by the specific impulse. The propellants were chosen as a result of extensive research done on previous historical spacecraft. The research is explained and justified more thoroughly in section 4.2, but to summarize the findings:

- The kick stage utilizes LOX because it has been reliable for numerous Atlas launches. (*Atlas V Rocket* 2021)
- The orbiter utilizes a hydrazine monopropellant based on the selection of the Osiris-REx satellite because of its similar purpose to the orbiter being utilized for this mission. (*OSIRIS-REx Spacecraft Overview*, no date)
- The lander is mostly constrained to MMH/NTO (MON-3) due to the engine it uses. The engine choices are very limited because the lander itself has to be incredibly light but still capable of reaching low Mars orbit. (Wade, no date b)

3.4.2 Stages Mass Estimations

In order to estimate the mass of the different stages, MATLAB was utilized to streamline the calculations. The program used can be viewed in appendix section "A.2" if needed. The stage mass estimations are based on a few fundamental equations:

$$m_{prop} = m_{pay} \frac{[e^{\frac{\Delta v}{I_{sp} g_0}} - 1](1 - f_{inert})}{1 - f_{inert} e^{\frac{\Delta v}{I_{sp} g_0}}}$$

$$m_{inert} = m_{prop} \frac{f_{inert}}{1 - f_{inert}}$$

$$m_i = m_f + m_{prop}$$

$$m_f = m_{pay} + m_{inert}$$

$$f_{inert} = \frac{m_{inert}}{m_{inert} + m_{prop}}$$

Although the bottom four equations are standard, the m_{prop} equation is actually meticulously derived from $\Delta V = C \cdot ln(\frac{m_i}{m_f})$. The proof that upholds this crucial derivation is detailed in appendix section "A.3".

To begin with, f_{inert} needs to be known before calculating the other mass parameters. Of course, there is no definitive way to know what the components of f_{inert} are, so they will be estimated based on historical values. Starting with $f_{inert,1}$, the inert mass fraction of the kick stage, it is assumed that the kick stage itself, modeled after the Centaur upper stage on the Atlas V, will have an inert mass of around 2,250 kg, while the propellant mass is somewhere near 21,000 kg. (*Atlas V Rocket*, 2021) Subsequently,

$$f_{inert,1} = \frac{m_{inert,3}}{m_{inert,3} + m_{prop,3}} = \frac{2250\ kg}{2250\ kg + 21000\ kg} \approx 0.0968$$

Meanwhile, for $f_{inert,2}$, the inert mass fraction of the orbiter prior to sample return, the inert mass is approximated to 900 kg with a propellant mass of around 1100 kg based on the historical values of the similar OSIRIS-REx. (*OSIRIS-REx Spacecraft Overview*,, no date)

$$f_{inert,2} = \frac{m_{inert,2}}{m_{inert,2}+m_{prop,2}} = \frac{900\ kg}{900\ kg+1100\ kg} \approx 0.45$$

$f_{inert,3}$ represents the inert mass fraction of the orbiter right before it is about to depart from Mars. The inert mass is still 900 kg, but it is assumed that around half of the fuel will be used getting to the Mars capture orbit, so the propellant mass is decreased to only 550 kg.

$$f_{inert,3} = \frac{m_{inert,3}}{m_{inert,3}+m_{prop,3}} = \frac{900\ kg}{900\ kg+550\ kg} \approx 0.31$$

Somewhat separate from these calculations is $f_{inert,4}$, the inert mass fraction of the lander. The lander itself needs to be small and light so as not to burden the orbiter, so the inert mass was estimated to be 25 kg with only 100 kg of propellant.

$$f_{inert,4} = \frac{m_{inert,4}}{m_{inert,4}+m_{prop,4}} = \frac{25\ kg}{25\ kg+100\ kg} \approx 0.20$$

With the inert mass fractions determined, the specific impulses of each stage, as well as the ΔV required for each stage, all that is left is to iterate through the fundamental equations provided four times to find the gross liftoff weight. The constants will be listed in a table first for clarity.

Table 9: Constants for Stage Mass Calculations

Variable	Description	Value	Units
$m_{pay,4}$	Sample Mass (Payload of Lander)	15	kilograms
$m_{pay,3}$	Assumed Mass of Orbiter + Lander	1000	kilograms
g_0	Gravitational Constant	9.81	m/s^2
ΔV_4	Lander Ascent ΔV	3540	m/s
ΔV_1	Kick stage ΔV	3550	m/s
ΔV_2	Orbiter ΔV (pre-capture)	850	m/s
ΔV_3	Orbiter ΔV (departure)	2960	m/s
$I_{sp,4}$	Lander Vehicle Specific Impulse	293	seconds
$I_{sp,1}$	Kick stage Specific Impulse	448	seconds
$I_{sp,2}$ & $I_{sp,3}$	Orbiter Specific Impulse	328	seconds

For the lander, considered to be "stage 4":

$$m_{prop,4} = m_{pay,4} \frac{[e^{\frac{\Delta v}{I_{sp} g_0}} - 1](1 - f_{inert,4})}{1 - f_{inert,4} e^{\frac{\Delta v}{I_{sp} g_0}}} = \frac{15\ kg \cdot [e^{\frac{3540\ m/s}{293\ s \cdot 9.81\ m/s^2}} - 1](1 - 0.20)}{1 - 0.20 e^{\frac{3540\ m/s}{293\ s \cdot 9.81\ m/s^2}}} \approx 92.54\ kg$$

$$m_{inert,4} = m_{prop,4} \frac{f_{inert,4}}{1 - f_{inert,4}} = 92.54\ kg \cdot \frac{0.20}{1 - 0.20} \approx 23.14\ kg$$

$$m_{final,4} = m_{pay,4} + m_{inert,4} = 15\ kg + 23.14\ kg = 43.14\ kg$$

$$m_{initial,4} = m_{final,4} + m_{prop,4} = 43.14\,kg + 92.54\,kg = 135.68\,kg$$

For the orbiter in its departure form, considered to be “stage 3”:

$$m_{prop,3} = m_{pay,3} \frac{[e^{\frac{\Delta v}{I_{sp}g_0}} - 1](1 - f_{inert,3})}{1 - f_{inert,3} e^{\frac{\Delta v}{I_{sp}g_0}}} = \frac{1000\,kg \cdot [e^{\frac{3550\,m/s}{448\,s \cdot 9.81\,m/s^2}} - 1](1 - 0.31)}{1 - 0.31 e^{\frac{3550\,m/s}{448\,s \cdot 9.81\,m/s^2}}} \approx 1611.2\,kg$$

$$m_{inert,3} = m_{prop,3} \frac{f_{inert,3}}{1 - f_{inert,3}} = 1611.2\,kg \cdot \frac{0.31}{1 - 0.31} \approx 723.89\,kg$$

$$m_{final,3} = m_{pay,4} + m_{inert,3} = 15\,kg + 723.89\,kg = 738.89\,kg$$

$$m_{initial,3} = m_{final,3} + m_{prop,3} = 738.89\,kg + 1611.2\,kg = 2350.09\,kg$$

For the orbiter in its pre-capture form, considered to be “stage 2”:

$$m_{prop,2} = m_{initial,3} \frac{[e^{\frac{\Delta v}{I_{sp}g_0}} - 1](1 - f_{inert,2})}{1 - f_{inert,2} e^{\frac{\Delta v}{I_{sp}g_0}}} = \frac{2350.09\,kg \cdot [e^{\frac{850\,m/s}{328\,s \cdot 9.81\,m/s^2}} - 1](1 - 0.45)}{1 - 0.45 e^{\frac{850\,m/s}{328\,s \cdot 9.81\,m/s^2}}} \approx 944.09\,kg$$

$$m_{inert,2} = m_{prop,2} \frac{f_{inert,2}}{1 - f_{inert,2}} = 944.09\,kg \cdot \frac{0.45}{1 - 0.45} \approx 772.44\,kg$$

$$m_{final,2} = m_{initial,3} + m_{inert,2} = 2350.09\,kg + 772.44\,kg = 3122.53\,kg$$

$$m_{initial,2} = m_{final,2} + m_{prop,2} = 3122.53\,kg + 944.09\,kg = 4066.68\,kg$$

Finally, for the kick stage, considered to be “stage 1”:

$$m_{prop,1} = m_{initial,2} \frac{[e^{\frac{\Delta v}{I_{sp}g_0}} - 1](1 - f_{inert,1})}{1 - f_{inert,1} e^{\frac{\Delta v}{I_{sp}g_0}}} = \frac{2814.25\,kg \cdot [e^{\frac{2960\,m/s}{328\,s \cdot 9.81\,m/s^2}} - 1](1 - 0.13)}{1 - 0.13 e^{\frac{2960\,m/s}{328\,s \cdot 9.81\,m/s^2}}} \approx 5747.78\,kg$$

$$m_{inert,1} = m_{prop,1} \frac{f_{inert,1}}{1 - f_{inert,1}} = 5747.78\,kg \cdot \frac{0.13}{1 - 0.13} \approx 858.86\,kg$$

$$m_{final,1} = m_{initial,2} + m_{inert,1} = 2814.25\ kg + 858.86\ kg = 3673.11\ kg$$

$$m_{initial,1} = m_{final,1} + m_{prop,1} = 3673.11\ kg + 5747.78\ kg = 10738.05\ kg$$

So, the gross liftoff mass of the system only is 10,738.05 kg. A breakdown of all of the stage masses is as follows:

Table 10: Stage Masses

Stage	Propellant Mass (kg)	Inert Mass (kg)	Final Mass (kg)	Initial Mass (kg)
Lander	92.54	23.14	43.14	135.68
Orbiter (D)	1611.2	723.89	738.89	2350.09
Orbiter (C)	944.09	772.44	3122.53	4066.68
Kick stage	5747.78	858.86	3673.11	**10738.05**

This is not the gross liftoff mass of the launch vehicle. The launch vehicle's gross liftoff mass would be found by adding the initial mass of the system to the overall initial mass of the Atlas V rocket. According to NASA's analysis on the Atlas V, its usual gross liftoff weight can be expressed as:

$$m_{launch} \approx 333000\ kg\ (\textit{Atlas V Rocket,}\ 2021)$$

Based on this mass estimation by NASA, the gross liftoff weight for this mission would be:

$$GLOW = m_{launch} + m_{initial,1} = 333000\ kg + 10738.05\ kg = 343738.05\ kg$$

The gross liftoff weight of the launch vehicle would be 342,420.89 kg when the system is attached.

4 Detailed Concept

The final mission design consists of a 3 stage system containing an orbiter for Low Mars Orbit (LMO), a lander to collect the samples prepared by the Perseverance rover, and the thruster systems to aid the spacecraft in reaching required orbits. Using historical mission data from spacecraft such as Hayabusa for the orbiter and lander design, Atlas V for the Centaur kick stage, and propellant types for both systems, our team chose compatible elements from each to create an optimal design for the spacecraft.

4.1 Concept Generation

I. Kick Stage and Launch Vehicle

Researching historical mission data from heavy-lift launch vehicles was the first objective in finding a kick stage which would suit our sample retrieval mission's needs. With the trajectory having a long range in addition to a relatively heavy payload system, we knew that a reliable and powerful kick stage would be required. We began to research various upper stages for renowned heavyweight launch vehicles such as Saturn V, Atlas V, and Falcon Heavy. The primary areas of focus were vacuum thrust, payload capabilities, and cost efficiency. It was important to our team to find a launch vehicle and corresponding kick stage which was capable of getting our payload of 4200 kg into Low Earth Orbit (LEO), without choosing a vehicle too large or powerful for the sake of cost efficiency.

II. Orbiter

When designing the orbiter, there was found to be ample room to modify existing designs to be most compatible for the mission's objectives. The first step was to cultivate ideas as to how the orbiter would be captured into orbit, as well as its compatibility with the lander / EDL system, thrusters, and power supply.

As a given parameter of the project, our team knew that the orbiter's orbit capture into Low Mars Orbit (LMO) would be completed via aerobraking to use the least thrust possible, and reserve ΔV for LMO departure. Thus, our team decided that an important aspect of our design would be a component with a large surface area to induce drag for the aerobraking maneuver.

Additionally, the orbiter system would need a power supply, which undoubtedly would require some form of solar power to be implemented. By using solar power, our system could focus on carrying the mass of only the most crucial payload components, such as the lander, EDL docking system, and onboard computer systems.

III. Lander and EDL Systems

The most vital component of our lander is the ability to capture and retrieve the Mars samples collected by the Perseverance rover; therefore, all other design components of the lander must be compatible with the retrieval method. Due to the preexisting robotic arm attached to the Perseverance rover, our team decided that the most efficient method of retrieval would be to mitigate the need for any additional attachments to the lander, and simply use the rover's robotic arm to place the samples inside of the capsule.

Our team considered the use of historic systems such as a parachute or thrusters as a mode of descent onto the surface of Mars. Given the orbital velocity which the lander would initially be traveling at, it was crucial that our system would be able to withstand the high temperatures of atmospheric entry. This called for research on heat shields, which could be modeled by the heat shield used for Perseverance's landing as our lander would be experiencing the same ΔV and atmospheric drag.

Another important design consideration of the lander is its ability to provide thrust to propel itself back up to the orbiter. This would require additional thrusters to be placed on the lander.

Finally, our lander must be able to re-dock with the orbiter in order to capture the samples to be returned to Earth. This maneuver has been modeled by other missions such as ISS payload capture or Apollo lunar sample capture, which we gathered inspiration from.

IV. Engine(s)

There are multiple engine systems needed by our full system from the low earth parking orbit to the Low Mars Orbit (LMO) departure. With the previously noted staging being the kick stage, orbiter capture stage, and orbiter and sample departure stage, our system will require an engine at each of these.

Our kick stage choice, which will be discussed in the next section, is already equipped with appropriately compatible engines which are capable of producing enough thrust for both the 3.55 km/s ΔV to depart from Low Earth Orbit (LEO), as well as the 0.85 km/s ΔV for Mars orbit capture. Thus, we could ignore this engine selection.

Next, our team considered engine types for the lander ascent stage. This stage would need to be capable of producing enough thrust for our payload to establish 3.54 km/s of ΔV to launch from the surface of Mars into Low Mars Orbit (LMO). This stage was the most difficult given that this portion of the mission will include an unprecedented launch from the surface of Mars. The primary concern for this stage was to keep the mass of the engine low, as more mass will require more thrust to reach orbit.

The final stage which would require an engine is the LMO departure stage, containing the sample recovery E capsule. The calculated ΔV necessary for this maneuver is 2.11 km/s, thus making an engine capable of producing sufficient thrust the main focus for this stage.

V. Propellant Type(s)

Propellant choices were primarily determined by the choice of engine, so our team decided to focus on finding the best engines for the mission, and allow the propellant type to fall into place consequently. We kept in mind the need to keep mass low and achieve a total ΔV of 10.05 km/s, but focused on compatibility with our engines chosen which are elaborated in the next section.

4.2 Concept Selection

I. Kick Stage and Launch Vehicle

Table 11: Launch Vehicle Statistics for Comparison

System Name	Saturn V	Falcon Heavy	Atlas V
# of Stages	3	2	2
# of Strap on Boosters	N/A	2 (Falcon 9 Boosters)	5
Sea Level Thrust	7,600,000 lbs	5,130,000 lbs	860,300 lbs
Vacuum Thrust	1,155,800 lbs	5,953,500 lbs	380,000 lbs
Upper Stage Vacuum Thrust	231,913 lbf	hill 220,500 lbs	22,900 lbs
Gross Liftoff Weight (GLOW)	6,537,000 lbs	3,125,735 lbs	730,000 lbs
Payload Capabilities	LEO: 117,930 kg / 260,000 lbs Moon: 45,360 kg / 100,000 lbs	LEO: 63,800 kg / 140,660 lb GTO: 26,700 kg / 58,860 lb Mars: 16,800 kg / 37,040 lb	LEO: 18,940 kg / 41,750 lbs GTO: 8,900 kg / 19,620 lbs GEO: 3,855 kg / 8,500 lbs
Dimensions	Height: 363 ft Diameter: 33 ft	Height: 230 ft Diameter: 12 ft (1st and 2nd Stages)	Height: 188 ft Diameter: 16 ft
Propellant Used	Liquid Hydrogen Liquid Oxygen Liquid Kerosene	Liquid Oxygen RP-1	Liquid Oxygen Liquid Kerosene Liquid Hydrogen

Sources	(Lea, R., 2022) Wade, M. (no date c) (Uri, J., 2022)	*(Falcon Heavy*, no date)	*(Atlas V Rocket*, 2021) *(Atlas V*, no date) *(Atlas V*, 2014) (Wade, M., no date)

Table 11 details the specifications of various heavyweight launch vehicles. After conducting research on the Saturn V, Falcon Heavy, and Atlas V launch vehicles, our team found that the Centaur upper stage of Atlas V was the most compatible for the mission objectives. It has a maximum thrust in a vacuum of 22,900 lbf (*Atlas V*, no date), while Saturn V and Falcon Heavy were capable of 231,913 lbf (Lea, R., 2022) and 220,500 lbf *(Falcon Heavy*, no date), respectively. We deemed the thrust and payload capabilities of these two upper stages to be much more than necessary for our objective payload of merely 4200 kg. For this reason, we chose Atlas V for the launch vehicle of the mission, also using its Centaur upper stage.

II. Orbiter

With our objectives in mind, we began idea generation by researching other sample return missions, such as Hayabusa and OSIRIS-REx, which have performed similar sample capture and departure maneuvers. Both of these designs contain similar components, such as the cubic center structure containing the payload with long rectangular solar arrays attached to either side (Beshore, E. *et al.*, no date). The solar panel components were found to be an optimal design feature as it efficiently achieves the need to induce drag in the Martian atmosphere for orbit capture, as well as the need for solar energy to provide power to the system.

Unlike OSIRIS-REx and Hayabusa, our system design also needed to include a docking system which would be compatible with the lander and sample return capsule system after ascent from the surface of Mars. For this, our design was inspired by the probe and drogue docking system used by both the International Space Station and the Apollo Lunar Module. According to our research, the main components of this subsystem is a hatch/tunnel system on both the orbiter and lander, a probe, drogue, and docking ring. We considered the size difference between our ½ meter diameter sample return capsule, and the 2.4 meter diameter of the International Docking Adapter (IDA) (*International Docking System Standard (IDSS)*, 2022) used by the International Space Station (ISS), and considered this ratio when calculating the mass of our docking system.

III. Lander and EDL Systems

Because EDL into the Martian atmosphere has successfully been completed before by other systems such as the Curiosity and Perseverance rovers (*Mars Exploration Rover Mission: The Mission*, no date), we decided that the safest manner of EDL would be to use the same design as what has been done historically. The Perseverance rover used a combination of both parachutes and thrusters to provide sufficient ΔV to slow the lander to a safe velocity to not crash upon landing (*Mars 2020 Perseverance Rover*, no date). However, with our system, given the need for ample thrust and propellant to provide the ΔV for LMO departure, our team decided that excess ΔV and propellant used by thrusters to slow the lander during Mars descent would not be an efficient use of mass or fuel. Thus, we refined this down to deciding to only use a parachute for landing, as well as a powerful heat shield.

VI. Engine(s)

As stated above, the engines for the kick stage were predetermined by the Centaur upper stage of the Atlas V which was chosen as our launch vehicle. The Centaur kick stage uses one RL10C-1-1 engine, and is capable of producing 22,900 lbs of thrust, which we found to be appropriate for our small payload of 4200 kg.

Next, for the lander ascent stage, we were very careful in choosing a low-mass engine which would not add a considerable amount of additional mass to lift into LMO, given that the ΔV requirement for our ~15 kg payload is already large at 3.54 km/s. After conducting research on various engine types, we chose to use one Aerojet R-40B engine which has a mass of 6.8 kg, and can provide 900 lbs of thrust (Wade, M., no date b). We were most drawn to this engine due to its low mass. The low amount of thrust this engine is capable of is sufficient for this mission as the total mass necessary to reach LMO is only the 15 kg mass of the sample return capsule, in addition to the 6.8 kg mass of the engine itself.

The final design of the third stage engine was inspired by the engines equipped on OSIRIS-REx. This consisted of 4 MR-107S thrusters, capable of producing a combined 228 lbs of thrust in a vacuum (Wade, M., no date a). The ΔV requirement for the 1015 kg payload's Mars departure is calculated to be 2.11 km/s, which these thrusters will be capable of accomplishing based on OSIRIS-REx's similar ΔV of 2.0 km/s with a payload of 2100 kg (Sutter, B. *et al.,* no date). We made the decision to not differ from this system because this mission was greatly successful in recent years, and given the unprecedented nature of this sample return mission, it is better to limit risk factors and opt for reliability.

VII. Propellant Type(s)

Each engine chosen above will use the customary propellant type used by each respective thruster in previous missions.

1. Kick Stage
    a. Thruster Type
        i. RL10C-1-1
    b. Propellant
        i. Liquid oxygen and liquid nitrogen bipropellant
2. Lander Ascent Stage
    a. Thruster Type
        i. Aerojet R-40B
    b. Propellant

i. MMH/NTO(MON-3) (Wade, M., no date b)

3. Mars Capture and Departure Stage

   a. Thruster Type

      i. MR-107S

   b. Propellant

      i. Monopropellant hydrazine fuel system (Wade, M., no date a)

4.3 Concept Refinement

The final design of the system features a 3 stage spacecraft, equipped with one substage containing a lander ascent vehicle. The orbiter is equipped with a low gain antenna that would transmit and receive data from Mission Control. Each solar panel has an area of 22.3 square feet of surface area, providing a total of 3.75 kW of power for the avionics (Solar Electricity, no date). Additionally, the solar panels are built to fold vertically onto the spacecraft, changing its total width from 12 meters to approximately 2.5 meters. This is to allow the orbiter to fit into the payload fairing which requires the diameter of the system to be less than 5 meters. Furthermore, we also used hinges in the lander legs to fold inward, decreasing the lander diameter from 4.66 meters to 2.9 meters.

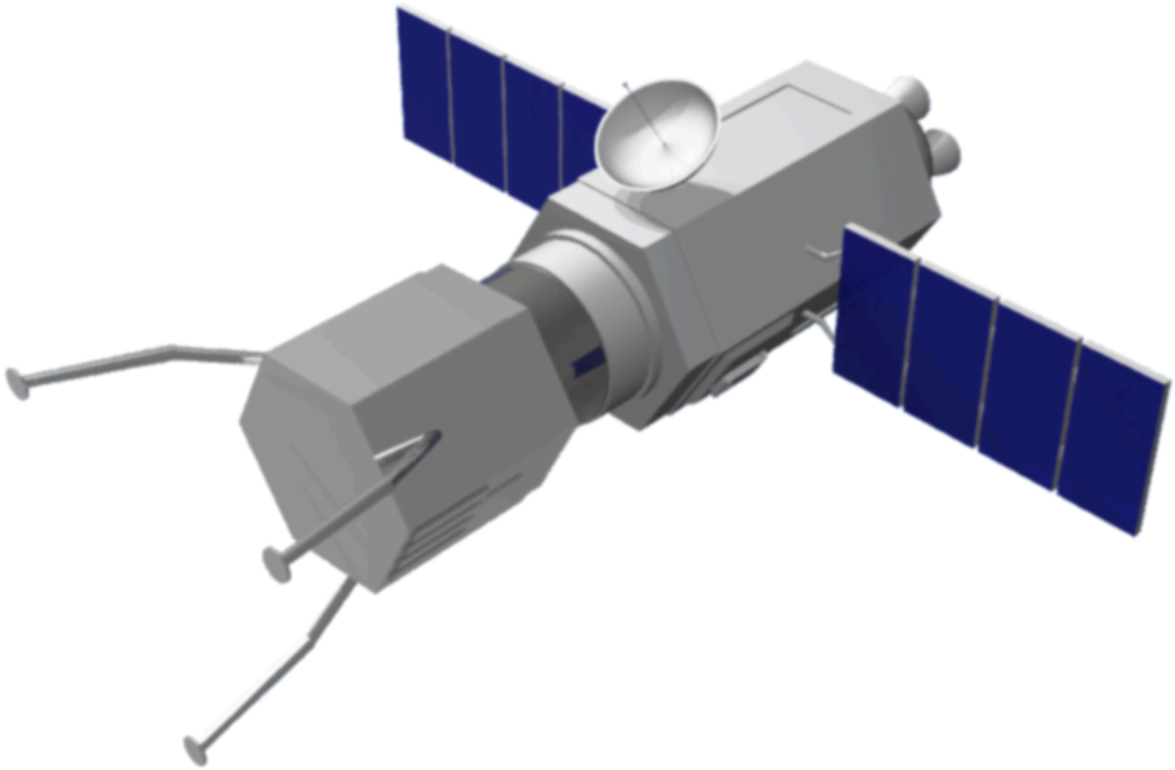

Figure 2: Full Assembly Rendered

Figure 2 shows the rendered final assembly of the Mars Orbiter Mission. This includes the Mars ascent module, lander, and orbiter interconnected using the IDA docking mechanism.

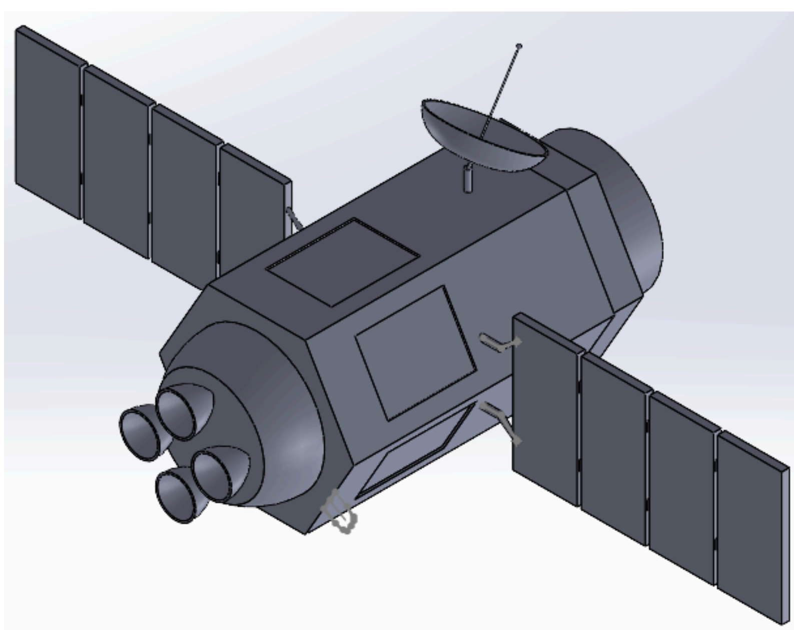

Figure 3: Orbiter

The orbiter consists of two 22 square ft solar panels, a satellite dish and antenna, and four Rocketdyne MR-107S engines as shown in detail in Figure 3.

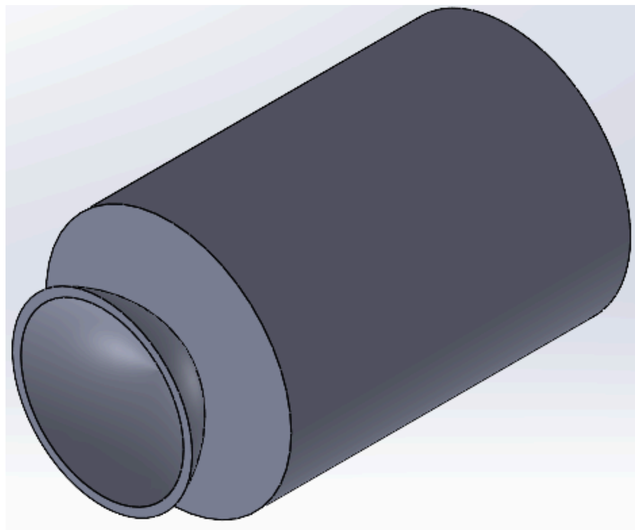

Figure 4: Lander Ascent Module

The lander ascent module consists of a carbon phenolic heat shield and one RL10-C-1-1 Centaur Engine as seen in Figure 4.

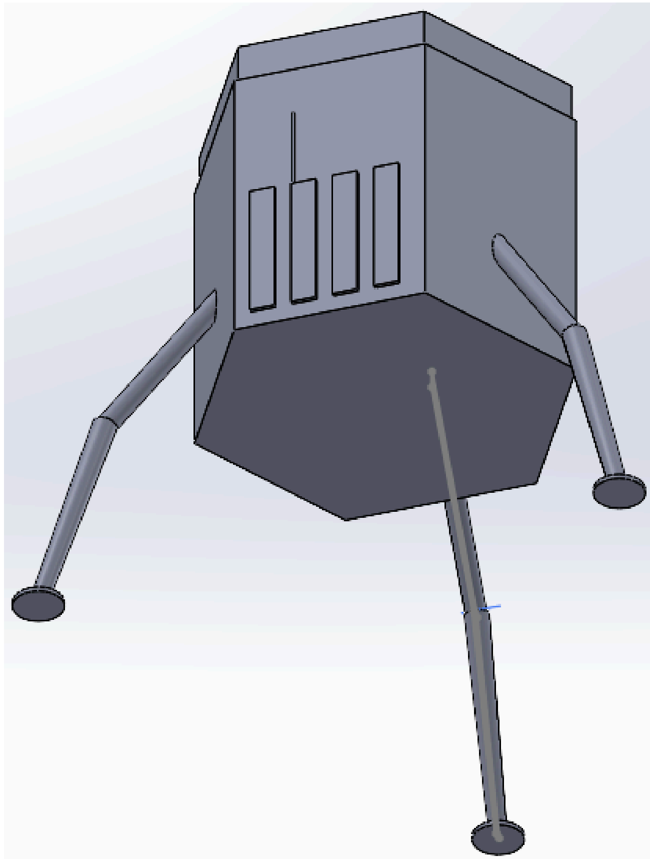

Figure 5: Lander

The landing vehicle is made up of 6.75 ft foldable landing legs and a compartment to hold the mars ascent lander as depicted in Figure 5.

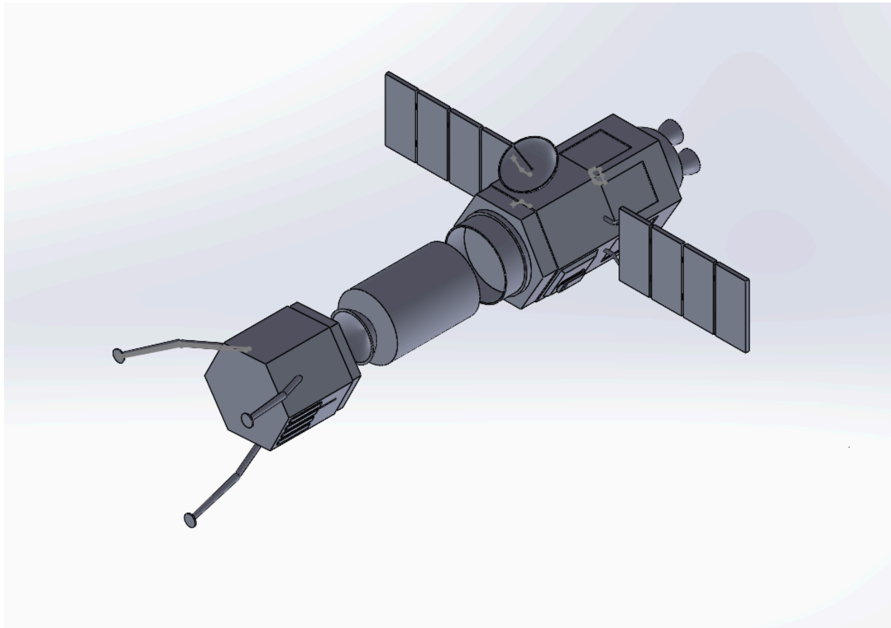

Figure 6: Full Assembly Exploded View

Figure 6 depicts the full assembly of the design in an exploded view to showcase the different stages and components of the system.

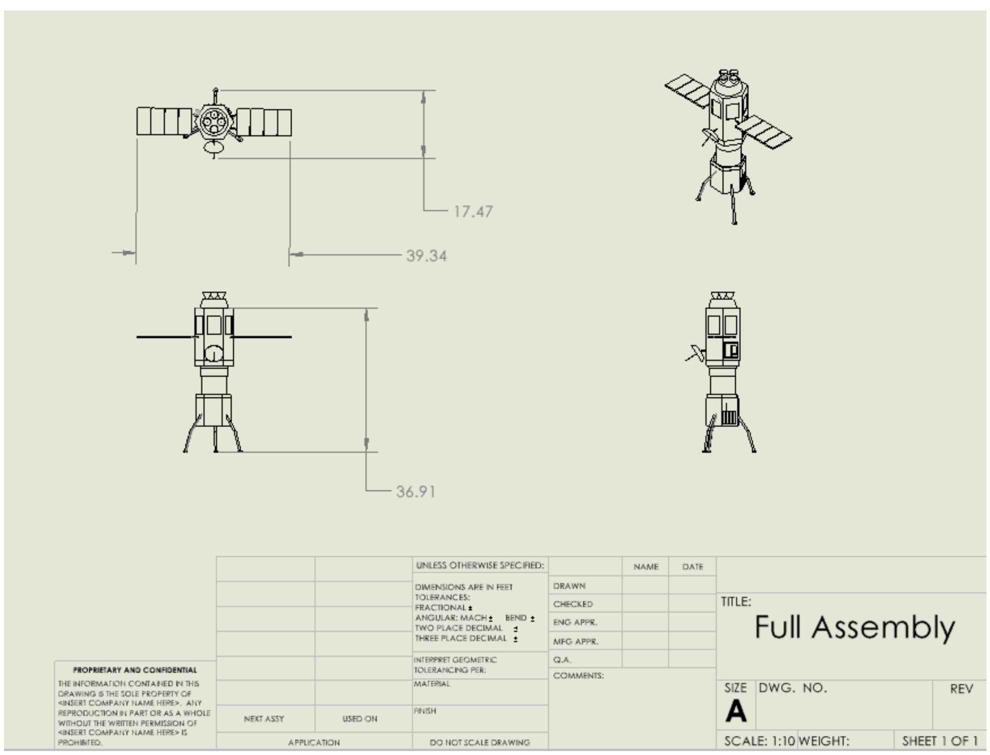

Figure 7: Full Assembly Drawing [ft]

(Refer to A.4 for Individual Component CAD Drawings)

5 Conclusions

With such an extensive design project, there is much to be evaluated and many lessons learned along the way. This section will elaborate on the potential for success with this design, where there is room for improvement, and what was learned as the project was being designed.

5.1 Design Evaluation

The outcome of this design will obviously stay unknown until it is tested, but based on the integrity of the calculations done and assumptions made, a fair assessment could be made on which aspects of the design would work and which would not. Because of the sheer complexity of the mission, it is not enough to say that the system overall works or does not work at all, and so the system must be evaluated component-by-component.

To start with the mission timeline, the phase angles and launch dates seem to have merit, which is especially proven by the moderate accuracy of the orbital period for both Earth and Mars. Both orbital periods were calculated to be within a fraction of a day of the actual scientific values of orbital periods, and since most calculations after that require the orbital periods, these calculations should yield relatively accurate results. At most, these dates and times would be at most a day or two off, so the mission timeline is not problematic towards the system's design and could be reasonable to use because it fits within the time requirement of the mission.

The one aspect of the final concept that could have gone through more development is the lander module. Because there is not much to historically justify the values describing that stage, it would be difficult to call the Mars ascent vehicle a completely valid design. The biggest concern from analyzing the mission requirements of the lander was the massive ΔV requirement needed for such a small stage. Even though the lander is only supposed to have a mass of around 135 kg, it is expected to exert around as much ΔV as the very large, but proven Centaur upper stage. This becomes even more conspicuous when considering that the engine chosen for ascent only has an I_{sp} of 293 seconds, much less than the orbiter or kickstage. And, unfortunately, the lander is mostly constrained to having that I_{sp} in order to keep the stage as small and light as possible. So, immediately, the biggest obstacle by far to the success of the design would have to be the ascent from Mars.

Besides the unprecedented nature of the Mars ascent vehicle, the other stages of the mission could be considerably reliable for a preliminary design because all of them could be justified with historical values. Very few aspects of this mission actually end up being outstanding; they are mostly based off of systems that are known to work in the past. However, we wanted to prioritize reliability of the system more than trying to innovate space travel because the mission is already high stakes enough and needs a proven methodology. Of course, it should be noted that since the ΔV is approximated with the patched conics method, this design can only ever be preliminary until more work is done on the astrodynamics of the mission, since the patched conics method cannot fully describe the interactions that occur in low-energy scenarios when there are three bodies involved. (Liu et. al, 2021) Otherwise, each stage theoretically can meet the ΔV requirements determined in section 3.3.

Some risks do still remain and were not able to be mitigated entirely. For example, we were able to mitigate the risk of the Mars orbit transfer failure by designing systems that can generate much more ΔV than is actually necessary like we had planned in the preliminary risk analysis. However, in the process of getting to and from Mars, we had no way of mitigating the possibility of a collision with space debris or asteroids. Although the internals of the system were not discussed in detail, it would very much be feasible to add protection against radiation to prevent the control systems from being damaged. This in turn also decreases the odds of a collision with Mars, since one of the causes of a Mars collision could be a computing failure.

5.2 Next Steps

The next step in refining and improving the accuracy of our design would be testing our inert mass fraction values. Because both our inert mass fractions and propellant masses were unknown when calculating our staging mass values, it was nearly impossible to know exactly if we had an exact value for either propellant mass or inert mass fraction. Our team provided the most accurate values possible given our knowledge of staging mass calculations and historic inert mass fractions for similar missions with nearly equivalent payload masses, thruster and propellant systems, and ΔV requirements.

Also another semester of work can help to refine calculations and remove some of the simplifying assumptions to make the mission as real-world applicable as possible. These include removing the equator launch assumptions on Earth and Mars and to take into account the plane change ΔV required to achieve a 0^{o} inclination orbit from a non-equatorial launch site.

More research and development will also need to be done regarding the lander module and capsule ascent system, given that this is the most innovative feature of the design. More knowledge surrounding the feasibility and logistics of the Mars launch will allow the system to be better controlled and understood.

5.3 Lessons Learned

After almost 4 months of this project, there has been an incredible amount of work done to be able to compile this report. So much of what we have been able to document and complete would have shocked all of us back in August. Learning how to properly estimate ΔV requirements during interplanetary transfer was certainly a challenge at first but it was also rewarding when concepts and equations started to make sense the more we worked on it. With that, calculating the mission timeline was also challenging but incredibly cool at that same time. We learned how to accurately model planetary motion based on orbital periods and figuring out optimal launch times for the mission based on the calculated planetary positions.

But not everything we learned was technical. A project of this size required a lot of planning on our part and we had to learn the best ways to be efficient during meetings. We learned fairly early on that working together would be the best course of action rather than split up tasks individually. That way, all members of the team would understand all aspects of the project to avoid assigning roles to members such as “coder”. With that did come the added risks of straying off-topic, but we learned the importance of setting personal deadlines to ensure timely completion of the project.

Lastly, the biggest takeaway from this project and class, is that aerospace engineering is our future. This project was by no means easy and there were times that it seemed we were falling too far behind schedule, but not once did we question our commitment to being an aero-student. Throughout all the challenges, there was always a level of awe and amazement at the sheer complexity of the calculations that we were doing that always kept us moving forward.

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Appendix: Matlab Code And Derivations

A.1 ΔV and Launch Date Calculations

```
%% AAE 25100 Project Milestone 5-6

% ΔV and Launch Date Calculations

% By Jaden Hernandez, Alayna Miller, Shreesh Nalatwad, Nischal Suhas

%% Initialization

clc %clears the output window

clear %clears the workspace

%% Constants

r_e = 6378; %eq. radius of earth in kilometers

r_m = 3390; %radius of mars in kilometers

au = 1.496 * 10^8; %astronomical units converted to kilometers

mu_e = 3.986*10^5; %gravitational parameter of earth in km^3s^-2

mu_s = 1.327 * 10^11; %gravitational parameter of sun in km^3s^-2

mu_m = 0.42828 * 10^5; %gravitational parameter of mars in km^3s^-2

w_e = 7.27 * 10^-5; %angular velocity of earth in rad/s

w_m = w_e / 1.03; %angular velocity of mars in rad/s

%a mars day is 1.03 earth days, can use to get w_m

r_sm = 1.524 * au; %radius of the sun-mars orbit

r_se = au; %radius of the sun-earth orbit

a_transfer = (r_sm + r_se) / 2; %semi-major axis of transfer orbit in km

a_se = (r_sm + r_se + 130) / 2; %semi-major axis of sun-earth orbit in km

latitude = 0; %launch latitude in degrees

v_loss = 1.65; %assumed ΔV loss in km/s

alt_park = 500; %earth parking altitude of system in km
```

```
alt_mcap = 100; %mars capture altitude in km

alt_park_m = 150; %mars parking altitude in km

r_park = r_e + alt_park; %radius of the parking orbit in km

%% ΔV Calculations

%%

% Launch to LEO

azimuth = 90; %azimuth for launch to LEO in degrees

v_leo = sqrt(mu_e / r_park); %standard ΔV for getting to LEO

v_eh = w_e * r_e * cosd(latitude) * sind(azimuth); %earth help term

deltaV_leo = v_leo + v_loss - v_eh; %LEO ΔV with earth help and loss

%%

% Earth Departure

v_sp = sqrt(mu_s * 2 * r_sm / (r_se * (r_se + r_sm))); ...

      %velocity of system at sun's periapsis (km/s)

v_se = sqrt(mu_s / r_se); %velocity of system in sun-earth orbit (km/s)

v_earth_excess = v_sp - v_se; %velocity left from orbit transfer (km/s)

v_ep = sqrt(v_earth_excess^2 + 2 * mu_e / r_park); ...

      %velocity of system at earth's periapsis

v_epark = sqrt(mu_e / r_park); %earth parking velocity in km/s

deltaV_dep = v_ep - v_epark; %ΔV needed to depart from earth's orbit

%%

% Mars Arrival

r_mcap = r_m + alt_mcap; %radius of the mars capture orbit in km

v_sa = sqrt(mu_s * 2 * r_se / (r_sm * (r_se + r_sm))); ...

      %velocity of system at sun's apoapsis (km/s)
```

```
v_sm = sqrt(mu_s / r_sm); %velocity of system in sun-mars orbit

v_mars_excess = v_sm - v_sa; %velocity left from orbit transfer (km/s)

a_mcap = (r_mcap + r_m + 40000) / 2; %altitude at mars capture

v_mcap = sqrt(mu_m * ((2 / r_mcap) - (1 / a_mcap))); ...

        %velocity upon mars capture in km/s

v_mp = sqrt(v_mars_excess^2 + 2 * mu_m / r_mcap); ...

      %velocity of system at mars' periapsis (km/s)

deltaV_cap = abs(v_mcap - v_mp); %mars capture ΔV

% Mars Departure

r_park_m = alt_park_m + r_m; %radius of the mars parking orbit in km

v_sm = sqrt(mu_s / r_sm); %velocity of system in sun-mars orbit

v_mars_excess = v_sa - v_sm; %velocity left from orbit transfer (km/s)

v_m_dep = sqrt(v_mars_excess^2 + 2 * mu_m / r_park_m); ...

         %velocity to leave mars orbit

v_mpark = sqrt(mu_m / r_park_m); %mars parking velocity in km/s

deltaV_dep_Mars = v_m_dep - v_mpark; %ΔV needed to leave mars orbit

%%

% Velocity of System at Earth's Periapsis

v_ep_reentry = sqrt(((2 * mu_e) / (r_e + 130)) - (mu_e) / (a_se)); ...

%how fast the system is traveling upon re-entering earth's orbit

%%

% Sample Retrieval Lander Launch and Rendezvous

v_lmo = sqrt(mu_m / (r_m + 150)); %velocity to get to low mars orbit (km/s)

vm_loss = 0.3; %assumed velocity losses for mars in km/s

v_mh = w_m * r_m * cosd(latitude) * sind(azimuth); %mars help term (km/s)
```

```
deltaV_lmo = v_lmo + vm_loss - v_mh; %ΔV to get to low mars orbit (km/s)

% Total ΔV

deltaV_total1 = abs(deltaV_dep); %earth departure phase

deltaV_total2 = abs(deltaV_cap + deltaV_lmo + deltaV_dep_Mars); ...

               %mars entry/depart phase

deltaV_overall = deltaV_total1 + deltaV_total2; %total mission ΔV

%% Transfer Phasing

T_sm = 2 * pi * sqrt((r_sm ^ 3) / mu_s); % period of mars sun orbit

T_se = 2 * pi * sqrt((r_se ^ 3) / mu_s); % period of earth sun orbit

T_trans = 2 * pi * sqrt(a_transfer ^ 3 / mu_s); %calculated transfer period

delta_t = T_trans / 2; %the Δt term for phase angle calculations

nM = (2 * pi) / T_sm; % Mean motion of mars sun orbit

nE = (2 * pi) / T_se; % Mean motion of earth sun orbit

pa_d = pi - nM * delta_t; % phase angle at departure

pa_a = pi - nE  * delta_t; %phase angle at arrival

%% Launch Date

ta_r = 0;

tests = 0;

T_wait = [pa_d / (nM - nE), (-2 * pa_a - 2 * pi * tests) / (nM - nE)]; ...

   % time it takes after Synodic Period Date (January 16th, 2025)

while T_wait(2) < 0

   tests = tests + 1;

   T_wait(2) = (-2 * pa_a - 2 * pi * tests) / (nM - nE);

end

pa_a = abs(pa_a);
```

```
%% Backup Date

N = [0, 1, 2]; % cycles of waiting for earth and mars to line up

Tsyn = abs(1 / T_se - 1 / T_sm) ^ -1; % Synodic Period

TBackup = T_wait(1) + (Tsyn * N); % time it takes for backup date from Synodic Period Date
(January 16th 2025)

Ttotal = TBackup(2) + T_wait(2) + delta_t; %finds the total time

%% Mass Calculations

massEarthHeatShield = (2.8 * 10^7) * (1.96 * 10^(-7)) * ...

                     (v_ep_reentry ^ 2) / ((r_e + 130) ^ 2); ...

%calculates the mass of system's heat shield

%% Outputs

fprintf("ΔV (earth) total: %.2f km/s\n", abs(deltaV_total1))

fprintf("ΔV LEO: %.2f km/s\n", abs(deltaV_leo))

fprintf("ΔV departure: %.2f km/s\n", abs(deltaV_dep))

fprintf("T_wait: %.2f days\n", T_wait(1) / 60 / 60 / 24)

fprintf("T_trans: %.2f days\n", T_trans / 60 / 60 / 24)

fprintf("T_synodic: %.2f days\n", Tsyn / 60 / 60 / 24)

fprintf("T_backup, N = 1: %.2f days\n", TBackup(2) / 60 / 60 / 24)

fprintf("T_backup, N = 2: %.2f days\n", TBackup(3) / 60 / 60 / 24)

fprintf("mars orbital period: %.2f days\n", T_sm / 60 / 60 / 24)

fprintf("earth orbital period: %.2f days\n", T_se / 60 / 60 / 24)

fprintf("phase angle @ departure: %.2f degrees\n", rad2deg(pa_d))

fprintf("phase angle @ arrival: %.2f degrees\n", rad2deg(pa_a))

fprintf("arrival date: 8/15/2027\n")

fprintf("departure wait time: %.2f days\n", T_wait(2) / 60 / 60 / 24)

fprintf("departure date: 11/12/2028\n")
```

```
fprintf("re-entry periapsis velocity: %.2f km/s\n", v_ep_reentry)

fprintf("mars departure burn ΔV: %.2f km/s\n", deltaV_dep_Mars)

fprintf("ΔV LMO: %.2f km/s\n", deltaV_lmo)

fprintf("ΔV mars capture: %.2f km/s\n", abs(deltaV_cap))

fprintf("total time: %.2f days\n", Ttotal / 60 / 60 / 24)

fprintf("ΔV (mars) total: %.2f km/s\n", abs(deltaV_total2))

fprintf("ΔV overall: %.2f km/s\n", deltaV_overall)
```

A.2 Staging Calculations

```
%AAE 251 Fall 2023

%Project Milestone 9

%Stage Mass

%Authors: Jaden Hernandez, Alayna Miller, Shreesh Nalatwad, Nischal Suhas

%% Initialization

clc %clears the output window

clear %clears the variables in the workspace

payload = 20; %payload mass in kg

sampleMass = 15; %kg

satMass = 1000; %kg

finert1 = .0968;

finert2 = .45;

finert3 = .31;

finert4 = .2;

isp1 = 448; %isp in seconds for stage 3

isp3 = 328; %isp in seconds for stage 1

isp2 = 328; %isp in seconds for stage 2

isp4 = 293; %isp in seconds for the mars ascent

g0 = 9.81; %gravitational acceleration in m/s^2

deltaV1 = 3.55 * 1000; %kick stage delta V

deltaV2 = 0.85 * 1000; %orbiter delta V (capture)

deltaV3 = 2.11 * 1000; %orbiter departure delta V

deltaV4 = 3.54 * 1000; %ascent capture delta V

%% Calculations
```

```
expTerm1 = exp(deltaV1 ./ (g0 * isp1));

expTerm2 = exp(deltaV2 ./ (g0 * isp2));

expTerm3 = exp(deltaV3 ./ (g0 * isp3));

expTerm4 = exp(deltaV4 ./ (g0 * isp4));

mProp4 = (sampleMass * (1 - finert4) * (expTerm4 - 1)) ...

        ./ (1 - (finert4 .* expTerm4)); ...

        %propellant mass calculation for ascent lander

mInert4 = mProp4 * (finert4 / (1 - finert4)); %finds inert stage mass

mFinal4 = payload + mInert4; %finds final mass of the stage

ascentMass = mFinal4 + mProp4; %finds initial mass of the stage

mProp3 =  ((sampleMass + satMass) * (1 - finert3) * (expTerm3 - 1)) ...

        ./ (1 - (finert3 .* expTerm3)); ...

        %propellant mass calculation for stage 2

mInert3 = mProp3 * (finert3 / (1 - finert3)); %finds inert stage mass

mFinal3 = sampleMass + mInert3; %finds final mass of the stage

mInitial3 = mFinal3 + mProp3; %finds initial mass of the stage

mProp2 = mInitial3 .* ((1 - finert2) * (expTerm2 - 1)) ...

       ./ (1 - (finert2 * expTerm2)); ...

       %propellant mass calculation for stage 1

mInert2 = mProp2 * (finert2 / (1 - finert2)); %finds inert stage mass

mFinal2 = mInitial3 + mInert2; %finds final mass of the stage

mInitial2 = mFinal2 + mProp2; %finds initial mass of the stage

mProp1 = (mInitial2 + ascentMass) .* ((1 - finert1) * (expTerm1 - 1)) ...

       ./ (1 - (finert1 * expTerm1)); ...

       %propellant mass calculation for stage 1
```

```
mInert1 = mProp1 * (finert1 / (1 - finert1)); %finds inert stage mass

mFinal1 = mInitial2 + mInert1; %finds final mass of the stage

mInitial1 = mFinal1 + mProp1; %finds initial mass of the stage

glow = mFinal1 + mProp1; %finds gross liftoff weight from initial mass 1

%% Outputs

fprintf("Gross Lift-Off Weight: %.2f kg\n", glow)

fprintf("Kick Stage Mass: %.2f kg\n", mInitial1)

fprintf("Orbiter Mass (Capture): %.2f kg\n", mInitial2)

fprintf("Orbiter Mass (Departure): %.2f kg\n", mInitial3)

fprintf("Lander Ascent Mass: %.2f kg\n", ascentMass)
```

A.3 Propellant Mass Equation Proof

Beginning with…

$$\Delta V = C \cdot ln(\frac{m_i}{m_f})$$

assuming that "C" represents the exhaust velocity, equalling $g_0 \cdot I_{sp}$:

$$m_i = m_{pay} + m_{inert} + m_{prop}$$

$$m_f = m_{pay} + m_{inert}$$

$$\Delta V = C \cdot ln(\frac{m_{pay}+m_{inert}+m_{prop}}{m_{pay}+m_{inert}})$$

$$\frac{\Delta V}{C} = ln(\frac{m_{pay}+m_{inert}+m_{prop}}{m_{pay}+m_{inert}})$$

If we exponentiate, then the equation would become…

$$e^{\frac{\Delta V}{C}} = \frac{m_{pay}+m_{inert}+m_{prop}}{m_{pay}+m_{inert}}$$

$$m_{prop} + m_{pay} + m_{inert} = e^{\frac{\Delta V}{C}}(m_{pay} + m_{inert})$$

$$m_{prop} = e^{\frac{\Delta V}{C}}(m_{pay} + m_{inert}) - m_{pay} - m_{inert}$$

$$f_{inert} = \frac{m_{inert}}{m_{inert}+m_{prop}}$$

$$m_{inert} = \frac{f_{inert \cdot m_{prop}}}{1-f_{inert}}$$

$$m_{prop} = e^{\frac{\Delta V}{C}}(m_{pay} + \frac{f_{inert \cdot m_{prop}}}{1-f_{inert}}) - m_{pay} - \frac{f_{inert \cdot m_{prop}}}{1-f_{inert}}$$

$$m_{prop} = e^{\frac{\Delta V}{C}} m_{pay} + \frac{e^{\frac{\Delta V}{C}} f_{inert \cdot m_{prop}}}{1-f_{inert}} - m_{pay} - \frac{f_{inert \cdot m_{prop}}}{1-f_{inert}}$$

$$m_{prop}(1 - f_{inert}) = (e^{\frac{\Delta V}{C}} m_{pay})(1 - f_{inert}) + (e^{\frac{\Delta V}{C}} \cdot f_{inert} \cdot m_{prop}) - (f_{inert} \cdot m_{prop}) - m_{pay}(1 - f_{inert}))$$

$$m_{prop} - m_{prop} f_{inert} = e^{\frac{\Delta V}{C}} m_{pay} - f_{inert} m_{pay} + e^{\frac{\Delta V}{C}} f_{inert} m_{prop} - f_{inert} m_{prop} - m_{pay} + m_{pay} f_{inert}$$

$$m_{prop} - m_{prop} f_{inert} - (e^{\frac{\Delta V}{C}} f_{inert} m_{prop} - f_{inert} m_{prop}) = e^{\frac{\Delta V}{C}} m_{pay} - f_{inert} e^{\frac{\Delta V}{C}} m_{pay} - m_{pay} + m_{pay} f_{inert}$$

$$m_{prop} - e^{\frac{\Delta V}{C}} f_{inert} m_{prop} = m_{pay}(e^{\frac{\Delta V}{C}} - f_{inert} e^{\frac{\Delta V}{C}} + f_{inert})$$

$$m_{prop}(- e^{\frac{\Delta V}{C}} f_{inert} + 1) = m_{pay}(e^{\frac{\Delta V}{C}} - f_{inert} e^{\frac{\Delta V}{C}} + f_{inert})$$

$$m_{prop} = \frac{m_{pay}(e^{\frac{\Delta V}{C}} - f_{inert} e^{\frac{\Delta V}{C}} + f_{inert})}{1 - f_{inert} \cdot e^{\frac{\Delta V}{C}}}$$

$$m_{prop} = m_{pay} \frac{(e^{\frac{\Delta V}{C}} - 1)\left(1 - f_{inert}\right)}{1 - f_{inert} e^{\frac{\Delta V}{C}}}$$

A.4 System Component Technical Drawings

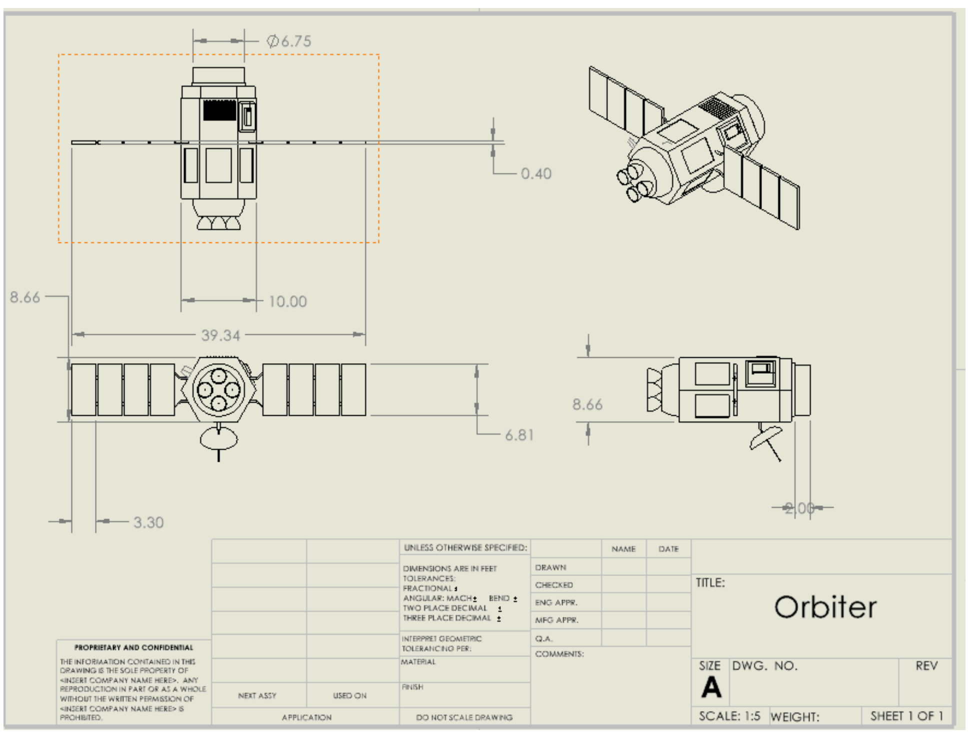

Figure 8: Orbiter Drawing [ft]

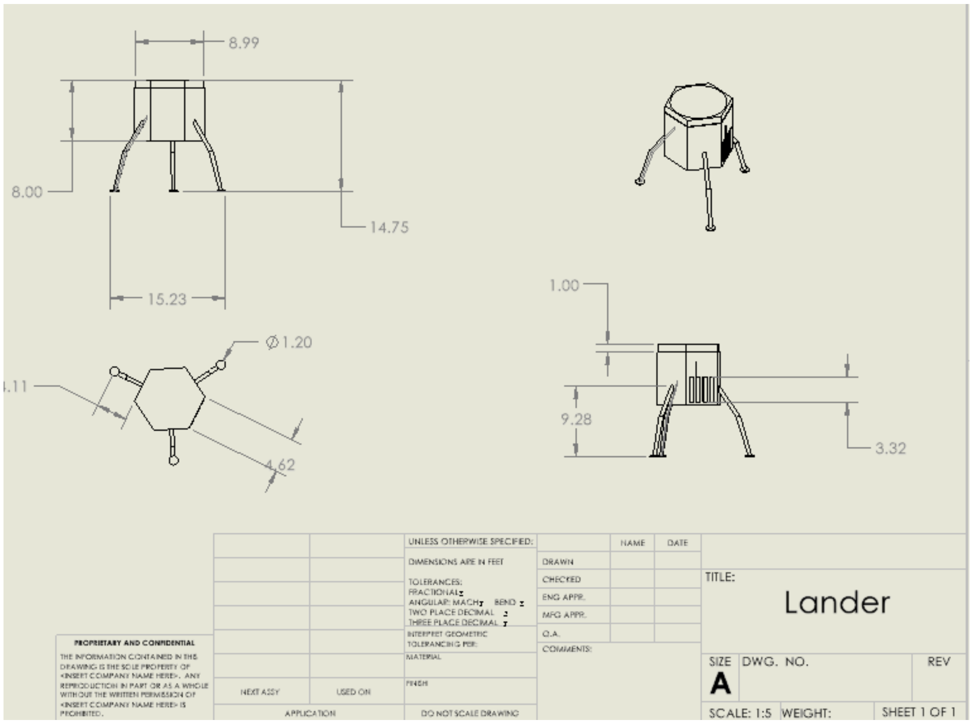

Figure 9: Lander Drawing [ft]

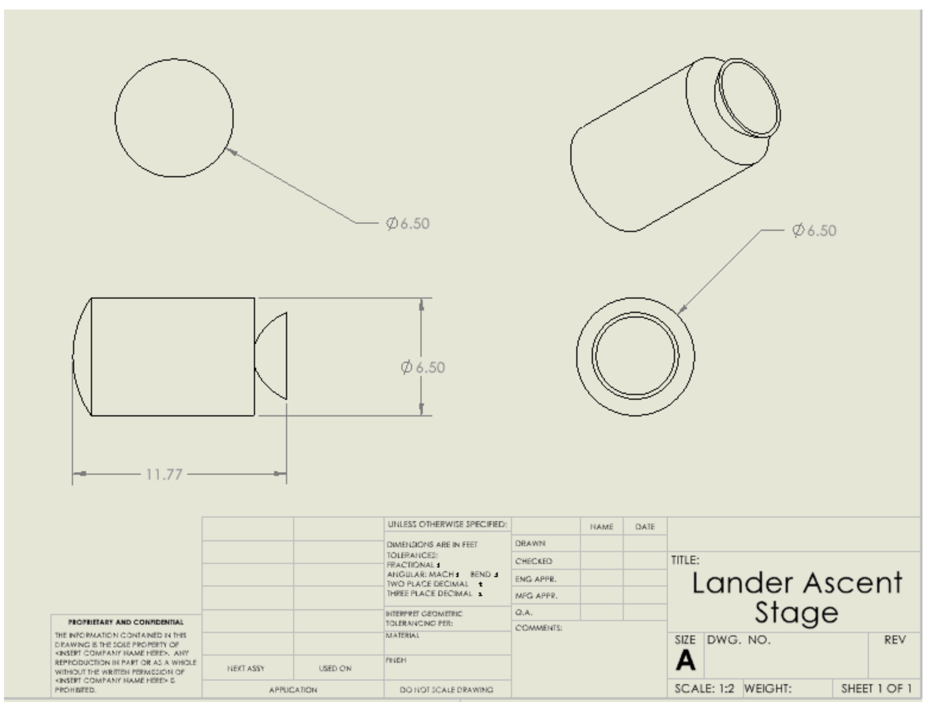

Figure 10: Lander Ascent Stage Drawing [ft]