Mars Ascent Vehicle Design Proposal

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Abstract

This report details the development of a Mars Ascent Vehicle (MAV) to assist in the sample return of Martian soil to Earth. The mission of the MAV is to launch from the Jezero Crater on Mars and rendezvous with an Earth Return Orbiter (ERO) in a circular orbit at an altitude of 382 km above the surface of Mars. As the major stakeholders, NASA and the ESA approached the team about developing an alternative design for the MAV as they are both over budget and regretting their initial design decisions. The clients provided the team with sizing, mass, and performance requirements that the design had to obtain. To address these requirements, the risks and the corresponding mitigations that were required to complete the mission requirements safely were analyzed. Examples of these risks include inner atmosphere storms, design errors, and moisture in propellants. Due to these risks, propellants were highly considered with respect to their operation and storage temperatures beyond just their I_{sp}. After deciding on possible propellants, many existing propulsion systems were analyzed with respect to their inert mass, size, and operability with the chosen propellants. Using three combinations of the motors and propellants, the team varied the number of stages and calculated the initial masses for a simple MAV design. After analyzing the results, the design was determined to be two stages as it decreased the mass of the design significantly from a one stage design but also did not overcomplicate the design like a three-stage rocket. From this information, three concepts were proposed. Concept 1 utilized solid ALITEC fuel in a Northrop Grumman Star 17A motor for the first stage and in a Star 12GV motor for the second stage. Concept 2 used White Lightning solid fuel in approximately nine parallel AEROTECH N1000 high powered motors for the first stage and used the same second stage configuration as concept 1. The third concept utilized ten AEROTECH N1000 high powered rocketry motors for the first stage and approximately four for the second stage. Additionally, an avionics bay, MRC 106F RCS thrusters powered by hydrazine, insulation, and clamshell cargo fairings were implemented in the design to provide guidance and safe cargo containment on the mission. After analyzing the initial mass of the concepts, only the concept 1 met the specified requirements. This design was optimized with an initial mass of 204.8867 kg, could obtain a ΔV of 3341.95691 m/s, and had a maximum diameter of 0.46 m. The design was determined to be feasible for operation on Mars but could benefit from further research and testing.

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1 INTRODUCTION

The National Aeronautics and Space Administration (NASA) and the European Space Agency (ESA) are interested in returning Martian rock and soil samples to Earth. To complete this mission, NASA plans on launching a mobile robot to fetch the soil samples that were deposited by NASA's Mars 2020 Perseverance Rover, the Earth Return Orbiter (ERO), and a stationary landing platform containing a small Mars Ascent Vehicle (MAV) (Hautaloma, 2020). The mission of the MAV is to deliver a payload of Martian soil samples to the Earth Return Orbiter in a circular, Low Mars Orbit. From the delivery of the samples, the Earth Return Orbiter will then travel back to Earth containing the MAV and crash land in a Utah desert. NASA is investigating the possibility of launching the systems required for the Mars Sample Return launch in 2026 and plan on receiving the samples in 2031 (NASA, 2020). However, as NASA and ESA are suffering buyer's remorse on many of the MAV's systems and propellant choices and have exceeded their project budget, they have approached our team to design an alternative Mars Ascent Vehicle.

Figure 1: Mars Ascent Vehicle Concept Drawing (NASA/JPL-Caltech, 2010)

The overall problem that is being addressed in this report is the generation of a Mars Ascent Vehicle design that can launch from Mars' Jezero Crater into an altitude in Low Mars Orbit. The Jezero Crater was the chosen launch point for the MAV as it is the location of the deposited samples from the Perseverance Rover (Tesmanian, n.d.). In order to be at the same orbital altitude as the Earth Return Orbiter for a successful rendezvous, ESA and Airbus, the developers of the ERO, have provided the team with a target circular orbit that the Mars Ascent Vehicle must obtain. To accurately analyze this capability, the team is focusing highly on the propellants and propulsion systems required to complete this mission. Additionally, this was a focus as NASA is currently searching for alternative propellants for the MAV. However, as the choice of propulsion system greatly affects the other systems of a rocket's design, this report also provides the corresponding considerations related to the systems designed to store of propellants and the systems provided to guide, navigate, and control of the rocket.

The initial assumptions made regarding the new design can be broken down into design factors, mission factors, and risk factors. The first major design-based assumption was that there are no budgetary restrictions for the development of the MAV. However, as NASA is overbudget for the project, the team only considered existing propellants, propulsion systems, and other components to limit the overall cost of the project. Furthermore, the mass of the structural components is neglected in the analysis of the design. This is justifiable as NASA and the ESA typically produce rockets and spacecrafts using thin sheets of lightweight metals, such as titanium and aluminum, lightweight alloys, or composites (Materials and Manufacturing, n.d.). It is also assumed that through using similar materials and components from operational launch vehicles, the Mars Ascent Vehicle is structurally capable of withstanding the forces that it will experience during launch. When analyzing the design's ability to complete the mission, the MAV's fuel is also assumed to operate at the ideal values listed by their supplier and will be operable if it is stored within the correct conditions. According to the United Kingdom's Ministry of Defense, solid fuels can endure chemical and physical changes during storage and when exposed to the atmosphere (Stenson, n.d.). Similarly, a report from the U.S. Naval Air Rocket Test Station details problems that result from storing liquid fuels (Terlizzi and Streim, 1956). When considering mission factor assumptions, it is assumed that the Perseverance Rover will load the samples onto the MAV autonomously, will not contaminate the samples, and will cause no structural or balance problems. This is important because if the cargo is not installed properly and shifts in storage, the rocket will become unstable and will not be able to ideally operate. Another important assumption is that the ERO is responsible for the delta-V required to perform any necessary Hohmann Transfers that are required to rendezvous with the MAV. As the ERO has solar arrays with a span of over 40 meters to generate electricity for electric propulsion along with large chemical propellant tanks, the orbiter can generate the delta-V for the transfer while the MAV relies solely on a predetermined quantity of chemical propellant (Airbus, 2020). Finally, the space related assumptions included that the MAV will not collide with any space debris, satellites, or astronautical objects. This is a valid assumption as there have only been 45 missions to Mars and only a small fraction of those have reached Mars orbit (mars.nasa.gov, 2019). This means that there is very little space debris in Low Mars Orbit, and while the risk is still presented in the analysis, shielding from space debris is neglected in the design of the MAV.

Considering the problem and the assumptions, the team developed the following mission statement:

"Our mission is to design a reliable, size efficient spacecraft that successfully contains gathered Martian samples and safely delivers them in Low Mars Orbit to the Earth Return Orbiter."

Inspired by this mission statement, the following report will provide our customers and shareholders with an analysis of the optimal alternative design for the Mars Ascent Vehicle. The remainder of this report is laid out as follows. Section 2 expands on the stakeholders and their specific needs and requirements for the mission along with an analysis of the risks that the design might face on the mission. Section 3 focuses on the initial breakdown of the mission and the required design parameters, such as delta-V, rocket stages, propellant choice, and initial weight estimates of the MAV. Section 4 explores the concept generation for Mars Ascent Vehicle systems, such as the propulsion system, payload bay, navigation and control systems, and insulation techniques. Section 4 also analyzes the advantages of each design and details the selection process and optimization of a final concept. Section 5 concludes the report with a design evaluation, future improvements on the MAV, and provide the stakeholders with the information the team learned through the design process.

2 NEEDS, REQUIREMENTS, AND RISK ANALYSIS

In order to understand the project fully, the team first analyzed the stakeholders and their needs. The stakeholders discussed for this project include the owners, users, current contractor, competitors, advocates, and critics. These are all important to consider as they all impact the needs and design process of the Mars Ascent Vehicle. From these needs, customer requirements are generated for the design, Additionally, this section discusses the inherent risks that must be considered and mitigated when providing the new Mars Ascent Vehicle concept.

2.1 STAKEHOLDERS AND THEIR NEEDS

A stakeholder is identified by anyone or any organization having a vested interest in a system, its outcomes and success of the system, its mission, products, services, or activities. The stakeholders are broken into six different categories: owners and users, advocates, current contractor, competitors, and critics.

The primary stakeholders for this project are the owners and users of the Mars Ascent Vehicle. The main stakeholder is the National Aeronautics and Space Administration since they are the main owners of the MAV and are the client that approached our team for the redesign of the system. NASA laid out most of the specifications for the project as they are the future operators of the rocket. Additionally, as the MAV is launched with the Sample Retrieval Orbiter, NASA's investment in this mission extends beyond the ascent vehicle (NASA, 2020). As the European Space Agency is partnered with NASA on this mission, they are also a major stakeholder as an owner and user. Beyond being involved in the design of the Mars Ascent Vehicle, the ESA is also the owner of the Earth Return Orbiter which depends on the feasibility and success of the MAV design (www.esa.int, n.d.). Additionally, as private space companies are becoming competitors, both NASA and the ESA have a high interest in the MAV safely delivering the 14 to 16 kg payload of Martian rocks to the ERO so they can maintain their funding for future missions. They also require that the rocket provide a set of flight data so that it can be used in future flights. Therefore, both corporations provide the team with the mission requirements as their financial, political, and future investments depend on the success of the design.

Similarly, a large supporter for the mission is Airbus. The ESA has provided Airbus with over $650 million to develop the Earth Return Orbiter (Airbus, 2020). As this system is dependent on the MAV delivering the cargo to a specific orbit, Airbus is a large supporter of a working design. For the rocket's design, Airbus requires the design to have enough delta-V to obtain the orbit of the ERO and have a guidance, navigation, and control system to ensure that the MAV will enter in the correct circular orbit for a rendezvous.

Another highly invested stakeholder is the current contracted developer for the MAV's propulsion system. Northrop Grumman Corporation (NGC) is the current contractor as their recent acquisition of Orbital ATK has contributed to them maintaining their reputation as an industry leader in supplying solid rocket propellant and propulsion systems (SpaceNews, 2019). However, despite NGC providing historically high performing rockets, NASA is exploring other contractors to fulfill the MAV's propellant needs. NGC still is a large stakeholder as they are still a strong competitor for supplying propulsion systems for the design. Due to this, NGC strongly needs our team to select a solid fuel to maintain the contract for the Mars Ascent Vehicle.

Finally, the critics of the design are the United States Government and the European Union. As these governmental entities approve and provide all the necessary funds for the operation of the owners and users, they are very critical of the feasibility and reliability of the design (NASA, 2010). As this mission is incredibly expensive, the governments require a complete list and analysis of risks and corresponding mitigations for the MAV to determine the worthiness of the investment. Additionally, the US Government and EU need the samples to be uncontaminated and clear of hazardous materials in order to protect their investment (Younse, 2014).

2.2 REQUIREMENTS

Need Being Evaluated: The MAV fits the measurable design constraints given by NASA.

1. A payload of Martian rocks that is 14-16 kilograms is transported back to Earth because they are needed for research (Clark, 2020). This requirement is verifiable since the payload can be weighed. The dependent variable in this requirement is the weight of the rock payload.
2. The size of the rocket cannot exceed 2.804 meters tall and 0.579 meters wide because the MAV is constrained by the size of the initial rocket that will deliver the MAV to Mars (Clark, 2020). This requirement is verifiable since the height and width of the MAV can be measured. The dependent variable in this requirement is the height and width of the rocket which can vary if the dimensions do not exceed the constraints.
3. The weight of the MAV is a maximum of 400 kilograms; the rocket will require more fuel for takeoff since it is not designed to carry more than 400 kilograms. This requirement is verifiable since the weight can be measured. The dependent variable in this requirement is the weight of the fuel and rocket staging since this will impact the overall weight of the MAV significantly.

Need Being Evaluated: The MAV can safely and efficiently travel from Mars to the Earth Return Orbiter with the rock payload.

4. The rocket holds enough propellant to travel from Mars and enter Low Mars Orbit to rendezvous with the Earth Return Orbiter at an altitude of 382 kilometers, as instructed by the clients. This requirement is verifiable since the amount of propellant needed can be calculated based on the staging design of the MAV. The dependent variable in this requirement is the amount of propellant being used.
5. The MAV lands safely and efficiently on the Earth Return Orbiter without damaging any of the Martian rock payload. This requirement is verifiable since the payload tubes and container can be tested and inspected in order to prevent damage on

impact. The dependent variable in this requirement is the percentage of tubes returned without any damage.

6. The MAV will have the cargo loaded and securely contained on Mars autonomously before liftoff. This requirement is verifiable since the payload tubes can be tested and inspected in order to see if they are properly secured within the MAV and take no damage after liftoff. The dependent variable in this requirement is the percentage of tubes returned without any damage.
7. The MAV can contain 14-16 kilograms of Martian rock samples inside the volume of the MAV within tubes that have a 12.5 cm radius and 18 cm height. This requirement is verifiable since the weight of the payload can be measured. The dependent variable in this requirement is the weight of the payload brought back from Mars.
8. The reusable parts of the system will include the tubes containing the samples as well as their container because these parts can withstand a high-speed landing, allowing them to be used in future sample-retrieving missions. This requirement is verifiable since the tubes and containers can go through multiple tests to see if they can be reused. The dependent variable in this requirement is the percentage of parts that can be reused.
9. The MAV components will be able to withstand tests and be able to be fully refurbished in a time frame to sufficiently test the rocket before launching to Mars. This requirement is verifiable since the MAV can go through multiple tests way before launch date to see if it takes any damage and if it does it can be fixed. The dependent variable in this requirement is the time it takes to fix any damage placed on the MAV.
10. The MAV will limit the disposable pieces of debris in orbit. This requirement is justified as to limit the space debris and not affect the risks of any future missions to Mars.

This set of requirements is complete for the mission. The requirements analyze the major sizing specifications, the mission requirements, and some additional requirements such as limiting space debris. The team will use these requirements throughout the remainder of the report to analyze the design's success.

2.3 PRELIMINARY RISK ANALYSIS

While the design of the Mars Ascent Vehicle is mainly analyzed with respect to its ability to achieve mission performance requirements, understanding the risks is also a critical aspect in the design of the rocket. Risk analysis is also important as it is required to satisfy the critic stakeholder's needs. The following analysis presents the identified risks associated with the operation of the vehicle.

The first risk that the MAV may experience is cold welding. In the vacuum of space, there is a possibility that the small pockets of air between moving mechanical parts is diminished and the components fuse together (Seller, 2007). This impacts the proper movement of parts on the spacecraft and leads to loss in mechanical functionality. While the probability of the MAV cold-welding components together is low, if it does occur the results can be moderately severe as it causes the loss of proper mechanical operation. To mitigate cold welding, the design will minimize the number of moving parts in the spacecraft. Additionally, adding specialized lubricates that will not evaporate or outgas are necessary for the full operation of the mechanical parts. After mitigation, spacecraft parts can move or operate with mitigation, but adjustments may make it less efficient to complete mission criteria when reducing moving parts and considering lubricants.

The next risk to consider is out-gassing. As some materials, such as composites and plastics, naturally have trapped gas bubbles in them when they are manufactured, the limited pressure of the "vacuum" of space causes these gas bubbles to expand and release from the material. These gasses can coat onboard sensors and electronics and limit their effectiveness (Seller, 2007). However, if the correct precautions are followed, the probability of this is quite low. However, if it does occur, a loss of electronic systems can have a major impact on the operation of the vehicle. To mitigate outgassing, one precaution that can be taken is to "bake" the MAV in thermal-vacuum chambers before launch. This should eliminate the gas bubbles in the material. As an additional mitigation technique, MAV can be tested in a vacuum chamber to confirm that all the gas has been eliminated. After following these precautions to eliminate the gas, MAV should be safe from out-gassing.

Another risk results from atmospheric molecules providing a drag force on the ascent vehicle's structure. As these forces can be substantial, there is a probability that drag can greatly impact the maximum ΔV, damage the structure, or alter the flight path of the MAV during launch. To mitigate the magnitude of the drag force impacting the MAV, we will design the MAV to maximize the vertical component of the launch to escape the atmosphere quicker and adjust the spacecraft's speed, shape, size, and orientation to reduce drag. Another mitigation technique is to include drag in flight path calculations, so the results are predictable. After mitigation, the MAV's flight path will be minimally impacted by drag.

The MAV's course can also be altered due to solar pressure. Solar pressure is the resulting pressure that occurs when photons from the sun's light hit a surface. While solar pressure is very small, approximately 5 Newtons of force per square kilometer of a surface, it can still have small effect on the forces on the rocket in orbit (Seller, 2007). However, the effect is so small that it is a very small risk. In order to mitigate the small effects, the surface area of the MAV will be minimized. However, the MAV will always have a small effect from solar pressure.

Additionally, the propellant in the MAV can be affected in storage and pose a large risk to the success of the MAV. Liquid fuels can freeze or evaporate, and solid fuels can gain moisture. If these problems occur, the MAV may not be able to obtain the required ΔV for orbit which would ultimately make the mission a failure. However, simple system considerations can be made to keep the fuels in the correct storage environment. These include installing heaters in the MAV's storage system and providing the launch vehicle with insulation. While these precautions will not eliminate the risk, they significantly improve the storage life and performance of the MAV.

As was discussed in the assumptions for this project, the Mars orbit contains small space debris along with micrometeoroids. Given that the MAV will be going into the Low Mars Orbit there is a chance that the debris can hit and damage the MAV. This can result in large amounts of damage which can harm our samples or knock the craft off the intended trajectory (Seller, 2007). For mitigation, debris tracking can be used to provide the MAV with a trajectory that avoid all major debris in the Low Mars Orbit. This mitigation should be enough to eliminate most of this risk as Low Mars Orbit has significantly less debris than Low Earth Orbit.

On Mars, there are strong inner atmosphere dust storms near the planet's surface. The dust storms have covered the entire planet blocking sunlight for solar panels and providing a harsh environment for a launch vehicle (Mars One, n.d.). This risk is severe as solar panels are required to provide the energy required to generate the heat that was previously discussed as a mitigation to keep the propellant operable. This can be mitigated by NASA installing backup battery systems within the storage system for the MAV. Additionally, it is a large risk to the MAV as strong storms can impact the trajectory, ΔV, and electronic systems of the MAV. To minimize the impact of the Martian weather on the rocket, the launch of the MAV may need to have a delayed launch date to have the best weather conditions for launch so the effects of weather on the MAV are minimized.

Another risk that we found is the impact of ultraviolet radiation on the MAV when it is on Mars. Since Mars doesn't have a magnetosphere and has a thinner atmosphere than Earth, the ultraviolet radiation levels are higher, and the MAV needs to be protected from being exposed to these levels for extended periods of time (Williams, 2016). The Mars Odyssey calculated that the surface of Mars has radiation levels 2.5 times higher than the radiation levels on the International Space Station (Williams, 2016). Two ways to prevent damage on the MAV from the radiation levels are the usage of solar cells and emphasizing thermal control.

The final risk is related to design errors that might occur. While this may not initially seem like a large risk after an in-depth consideration, in 1999 NASA lost contact with the Mars Climate Orbiter due to signals being sent in the incorrect units (NASA Solar System Exploration, n.d.). Due to this, it is vital that all calculations, units, and considerations from all contractors are checked and simulated before the mission as the chance of at least one design error is quite high and can have a severe impact. Therefore, it is important that all calculations are checked by all team members and are reviewed by an independent source. These efforts could greatly limit the chances of a severe effect resulting from a design overlook.

Figure 2 below is a risk matrix that was generated to compare the likelihoods and consequences of each risk. This risk matrix provided a foundation of understanding what mitigation practices are the most important to perform in NASA and ESA's limited budget.

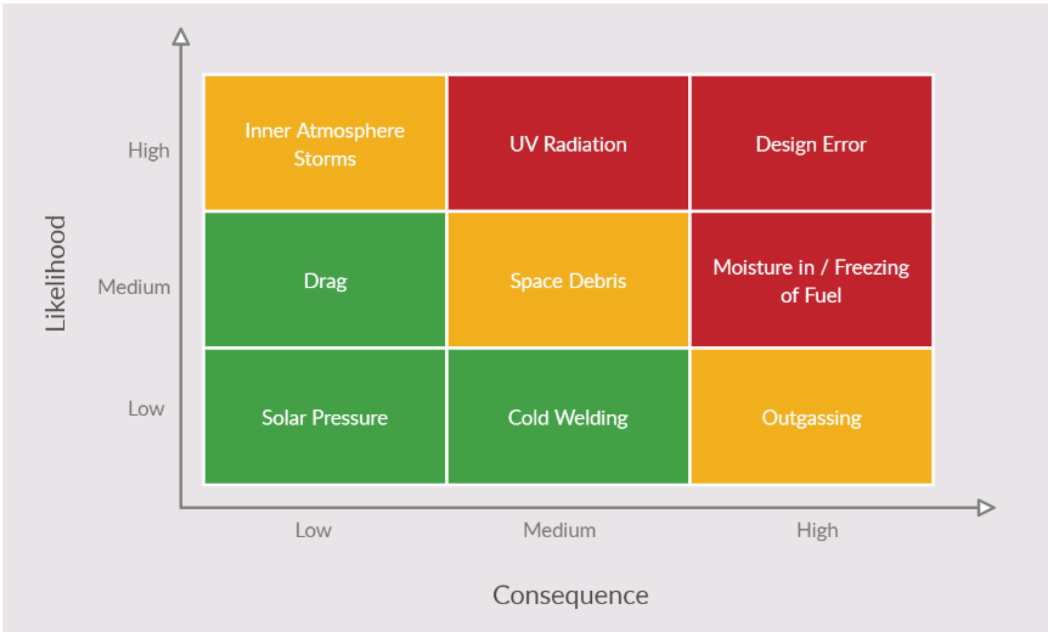


Figure 2: Risk Matrix for the Mars Ascent Vehicle

3 Estimating Design Parameters

To estimate the parameters of our design, we will take the mission requirements and begin calculations of the payload mass, ΔV, and the number of stages required for the MAV. Additionally, propellants and motors are proposed for the design to perform initial mass estimations. The calculations and considerations are shown in the following section to create a basis for selecting design concepts for the MAV.

3.1 MISSION DESIGN

To achieve the mission statement that was presented in the introduction of the report, the mission had to be broken down into smaller components. First, the Martian samples are loaded into the cargo bay autonomously by the Sample Retrieval Rover. The samples would be stored in a cargo container and are secured as cargo in the MAV. The launch of the MAV will then begin when the Earth Return Orbiter enters Mars orbit. Since the MAV has multiple stages, first will be the deployment and burnout of the first stage. After that will be the deployment and burnout of the second stage. After this it will be in the Low Mars Orbit until the Earth Return Orbiter comes and picks up the payload. This process is detailed in Figure 3 below.

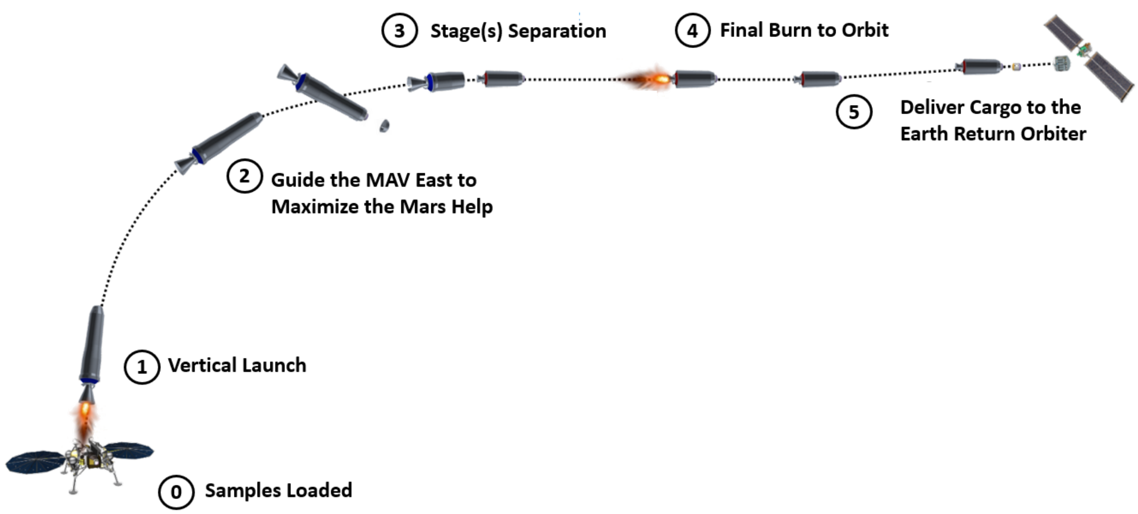


Figure 3: Mission Diagram for the Mars Ascent Vehicle

When comparing a launch from Earth to a launch from Mars, there are a few different factors to consider for a successful liftoff. Due to the different size and mass of Mars compared to Earth, the gravity constant is also different. Instead of $9.81\frac{m}{s^2}$, the gravity constant on Mars is $3.711\frac{m}{s^2}$, resulting in 62.5% decrease in gravity from Earth to Mars (NASA, 2019). Another key difference between the two planets is the atmosphere. The air on

Mars is “a hundredth of the thickness on Earth’s” which means that Mars has a thin atmosphere. This affects the MAV in two major ways. The first is that upon arrival to Mars with the Retrieval Rover, the atmosphere is capable of burning the MAV and all the other technology, but, since the atmosphere is so thin, there’s not enough thickness in the air to slow it down before impact upon entry to Mars (Strauss, 2015). The other aspect is that the thinner atmosphere provides less drag on the MAV during launch. The next difference is that the Low Mars Orbit is at 250 km above Mars’s surface while the Low Earth Orbit is 2000 km above Earth’s surface. This means we must travel a shorter distance to enter the Low Mars Orbit than the Low Earth Orbit. The last comparison between the two is Mars rotates at a different rate and axis than Earth which should be taken in consideration when planning the launch of a rocket on Mars.

As noted in Section 2.2, the mass of the payload will be around 16 kg or about 35 pounds. This payload will be delivered by a rover in 43 tubes where 5 of the tubes will be blank in order to eliminate any contamination or testing issues (Clark, 2020). We estimated that the volume of the tubes that will hold the payload is between 0.00921 m^3 and 0.0105 m^3. An image of the payload capsule with the 43 tubes is shown below in Figure 4.

Figure 4: Cargo Container that the MAV must Deliver to Orbit (NASA/JPL-Caltech, 2020)

The MAV needs to be in a location that the Retrieval Rover can easily access and transfer the payload samples over. NASA has specified that the best location for the MAV to obtain the samples and launch is from the Jezero crater. The planned launch Azimuth for the MAV will be 90 degrees since we are launching due East, giving us added help from the rotation of Mars itself that will increase our total velocity. Since we don't want to ruin the environment on Mars, one of our design goals is to reduce the debris left behind from the mission.

3.2 ΔV ESTIMATION

We know that the estimated ΔV is 3370 meters per second. Below it can be seen that our calculations to obtain an actual ΔV of 3,341.95691 meters per second. We were able to find the total ΔV by using the equation:

$$\Delta V_{tot} = \Delta V_{MO} + \Delta V_{loss} - \Delta V_{MH}$$

Table 1: Reasonable Delta V Loss Estimates for a 2 Stage Spacecraft (Project Clients)

	Stage 1	Stage 2	Total
Gravity Loss [m/s]	165	0	165
Drag Loss [m/s]	12	0	13
Steering Loss [m/s]	12	11	23
Actual Delta V [m/s]	1685	1685	3370
Ideal Delta V [m/s]	1495	1673	3169

Based on Table 2, we can then calculate the value for ΔV_{Loss} by adding up the individual loss components giving us the following total:

$$\Delta V_{loss} = 165\frac{m}{s} + 13\frac{m}{s} + 23\frac{m}{s} = 201\frac{m}{s}$$

With ΔV_{Loss} calculated, we then need to calculate ΔV_{MO} and ΔV_{MH}. We use the following equations to complete these calculations were $GM = 4.297 * 10^{13} \frac{m^3}{s^2}$, $r = 3.39 * 10^6\ m$, $\omega_M = 868.22 \frac{km}{h} = 0.0000711527429 \frac{rad}{sec}$, $Lat = 18.3628\ degrees$, $Az = 90\ degrees$

Since we are using the equation to calculate the velocity of Mars orbit, r_M is equal to the radius of Mars plus the altitude of Mars Orbit where the MAV will meet the ERO for rendezvous. This means that the value for $r_M = 3.39 * 10^6\ meters + 382000\ meters$.

$$\Delta V_{MO} = \sqrt{\frac{GM}{r_M}} = \sqrt{\frac{4.297 * 10^{13}\ \frac{m^3}{s^2}}{3.39 * 10^6\ m + 382000\ m}} = 3369.84 \frac{m}{s}$$

The MAV will be launching due East, which yields an Azimuth value of 90 degrees. The radius of Mars and the latitude of launch are listed above, along with the rotational velocity of Mars converted into radians per second.

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

$$\Delta V_{MH} = 0.0000711527429\ \frac{rad}{sec} * 3.39 * 10^6\ meters * \cos(18.3628) \sin(90)$$

$$\Delta V_{MH} = 228.88309 \frac{\mathrm{m}}{\mathrm{s}}$$

After calculating all the different components of ΔV, we added them up to give us our final ΔV_{total} which is about the actual ΔV given in Table 2. This

$$\Delta V_{tot} = 3369.84 \frac{m}{s} + 201 \frac{m}{s} - 228.88309 \frac{\mathrm{m}}{\mathrm{s}} = 3341.957 \frac{\mathrm{m}}{\mathrm{s}}$$

3.3 STAGES, PROPELLANTS, AND INITIAL WEIGHT

3.3.1 Rocket Propellant and Motor Considerations

When considering the propellant for this mission, NASA originally investigated hybrid motors with a solid-based wax fuel with a liquid oxidizer. However, engineers were concerned with "cold soak," where the solid propellant grains are exposed to low temperatures for prolonged periods of time. Because of this, NASA only considered solid rockets for the MAV design. According to Jim Watzin, the director of NASA's Mars Exploration Program, storing solid propellant on Mars is not a big concern, so there is not a large risk (Clark, 2020). However, the team still considered liquid, hybrid, and solid propellants for a full analysis of the possibilities they present. NASA then approached Northrop Grumman, who recently acquired the solid motor companies Orbital ATK and Thiokol Propulsion, to make the new solid fuel for the MAV. In the original design that NASA is reconsidering, Northrop developed a proprietary solid propellant formulation for the MAV. In the past, Northrop developed a solid propellant for the Magellan spacecraft that fired after more than 15 months in space (Astronautix.com, n.d.). From this, we assume that Northrop's propellant for the Magellan spacecraft's STAR 48 motor is like the propellant they developed for the MAV. After further investigation, we can confidentially say that Northrop intends to use TP-H-3062 and TP-H-3340A propellants for the MAV (Dankanich, Rousseau and Williams, 2019). When evaluating the reasons why NASA may not want to use this solid fuel, we determined that both solid propellants are difficult to control. Similarly, the TP-H-3062 did not meet the initial stretch tests which could also cause NASA to sway on their decision to use this propellant (Dankanich, Rousseau and Williams, 2019).

One of the other solid propellants we considered is the Adranos ALITEC solid fuel. Adranos is a young rocket propellant company located in West Lafayette. The ALITEC fuel has a high I_{sp} value of 270.7 seconds. Additionally, this propellant reduces the hydrogen chloride production by 95% compared to other propellants (Terry, Son and Gunduz, 2016). This makes the ALITEC propellant more environmentally friendly and safer for the MAV since hydrogen chloride cause corrosion within the motor. The density of this propellant is $1910 \frac{kg}{m^3}$ (Terry, Son and Gunduz, 2016). Additionally, the location of Adranos is beneficial for our team to work with them since they are located near the Purdue campus.

We also considered using high power rocketry motors in parallel for some of our staging ideas. In order to reach the total ΔV required, we chose six motor candidates to examine: The Animal Motor Works N4000, AEROTECH N1000, Animal Motor Works N2800, Animal Motor Works N2700, Animal Motor Works N2600, and Animal Motor Works N2020. These motors were chosen because they were the six largest and most powerful on the official NAR certified motor list (www.nar.org, 2020).

After analysis using MATLAB Script #2, found in appendix A, the team found the following graphs depicted below by using the stage 2 ideal ΔV value of 1673 $\frac{\mathrm{m}}{\mathrm{s}}$ from Table 1.

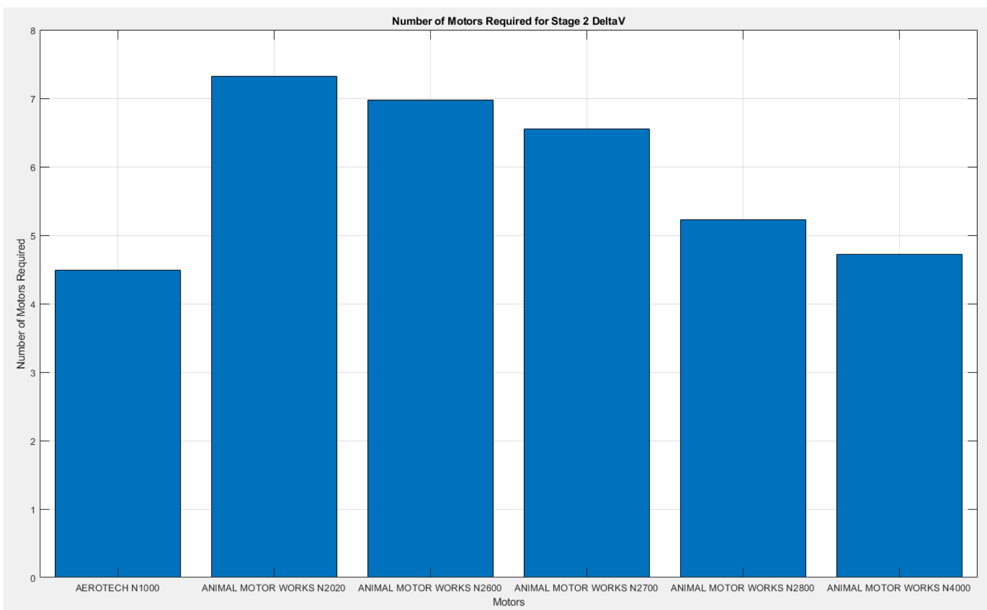


Figure 5: Number of Motors in Parallel Required for Stage 2 Delta-V

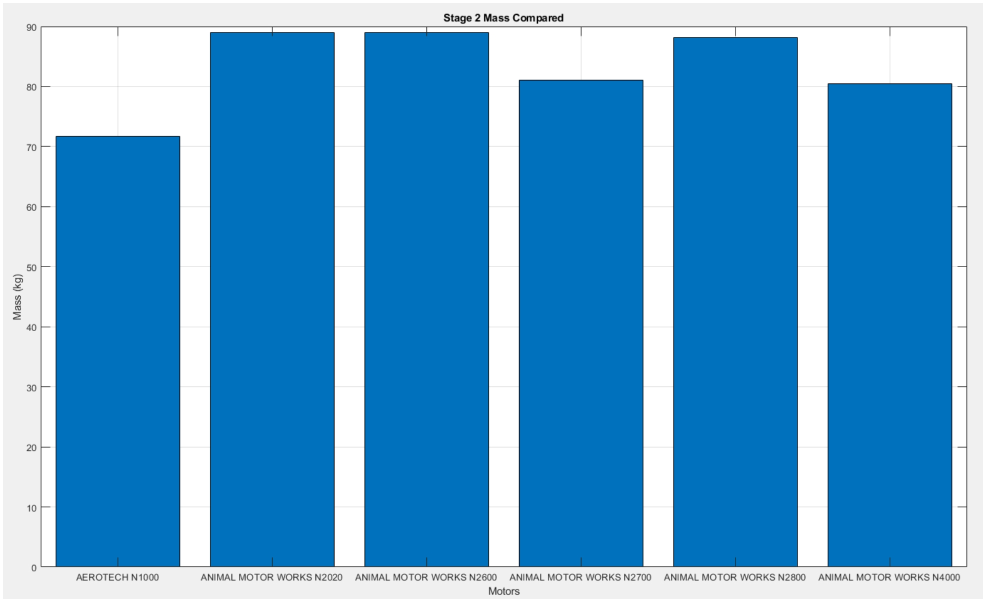


Figure 6: Mass of Motors Required for Stage 2 Compared

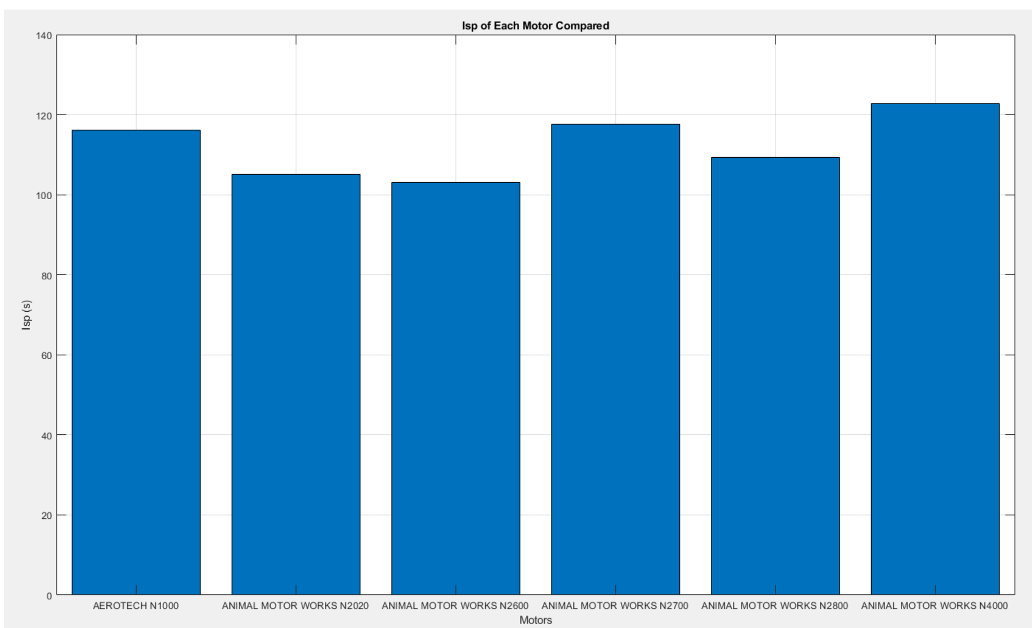


Figure 7: I_{sp} of Each Motor Compared

Based on the results depicted above, the team concluded that the AEROTECH N1000 was the best choice of the six high-powered rocketry motors. This is because it has the second highest Isp, the lowest number of motors required per stage, and the lowest total stage two mass. Therefore, the AEROTECH N1000 will be the only high-powered rocketry motor considered in the remainder of the report.

3.3.2 *Rocket Staging*

In order to determine the number of stages needed for the MAV, we took three of the motors off our previous list that met our needs the best. We can solve for the initial weight of the MAV by using the following equation:

$$m_i = m_{pay} * \prod_{k=1}^{n} \frac{e^{\frac{\Delta V_k}{g_0 * I_{sp,k}}} * (1 - f_{inert,k})}{1 - (f_{inert,k} * e^{\frac{\Delta V_k}{g_0 * I_{sp,k}}})}$$

We know that since the design requirements of the MAV have a maximum mass constraint of 440 kilograms, therefore any initial mass higher than that must be eliminated. The total payload mass used in our initial calculations was estimated to be 37 kg. This is an increase from the Martian rock payload to account for the other components on the MAV that add some mass.

We will have to use the following equation to calculate the inert mass fractions for the rocket motors and their propellants below:

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

We started by plotting the initial mass versus the number of stages for the NGC STAR 12GV with the Adranos ALITEC solid propellant. The NGC STAR 12GV has an inert mass of approximately 8.981129 kg and a propellant mass of approximately 32.93081 kg. Using our f_{inert} equation given above, we find an inert mass fraction of approximately 0.214. The ALITEC propellant has a specific impulse value of 270.7 seconds. By using these values, along with $\Delta V = 3341.957\,\frac{m}{s}$ and $g_0 = 9.81\,\frac{m}{s^2}$. The plot outputted from the following inputs is shown below.

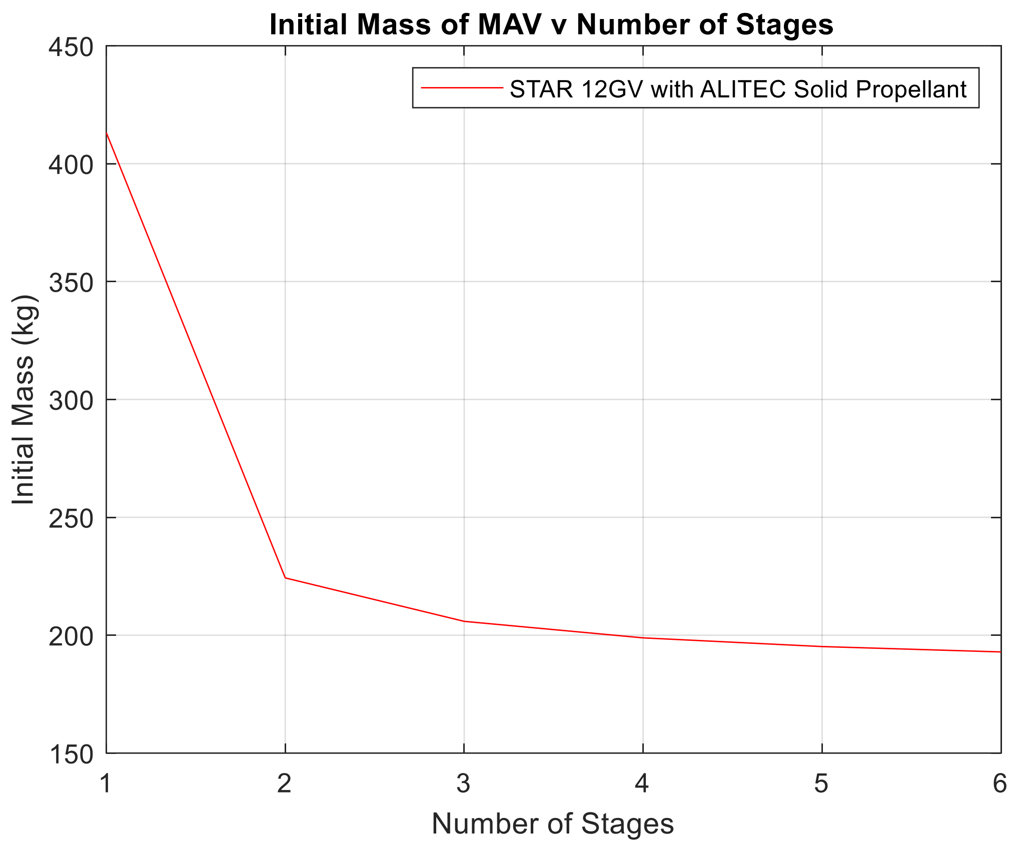


Figure 8: Initial Mass v Stages for STAR 12GV (MATLAB Script #4)

From this plot, we can see that a single stage MAV with the STAR 12GV and Adranos ALITEC propellant will exceed the initial mass requirement since it is greater than 400 kilograms. However, a two stage MAV with these specifications is at an appropriate initial mass of ~225 kilograms. Beyond the two-stage configuration, the initial masses of the MAV are relatively like the two-stage initial MAV. Since they are all relatively similar after two stages, the complications brought by a three-stage design would not be worth it since the initial mass being spared is miniscule.

We then plotted the initial mass versus the number of stages for the NGC STAR 17A solid motor with the Adranos ALITEC solid propellant. The NGC STAR 17A has an inert mass of approximately 13.38097 kg and a propellant mass of approximately 112.26411 kg. Using our f_{inert} equation, we find a inert mass fraction of approximately 0.106. The ALITEC propellant has a specific impulse value of 270.7 seconds. By using these values, along with $\Delta V = 3341.957 \frac{m}{s}$ and $g_0 = 9.81 \frac{m}{s^2}$. The plot outputted from the following inputs is shown below.

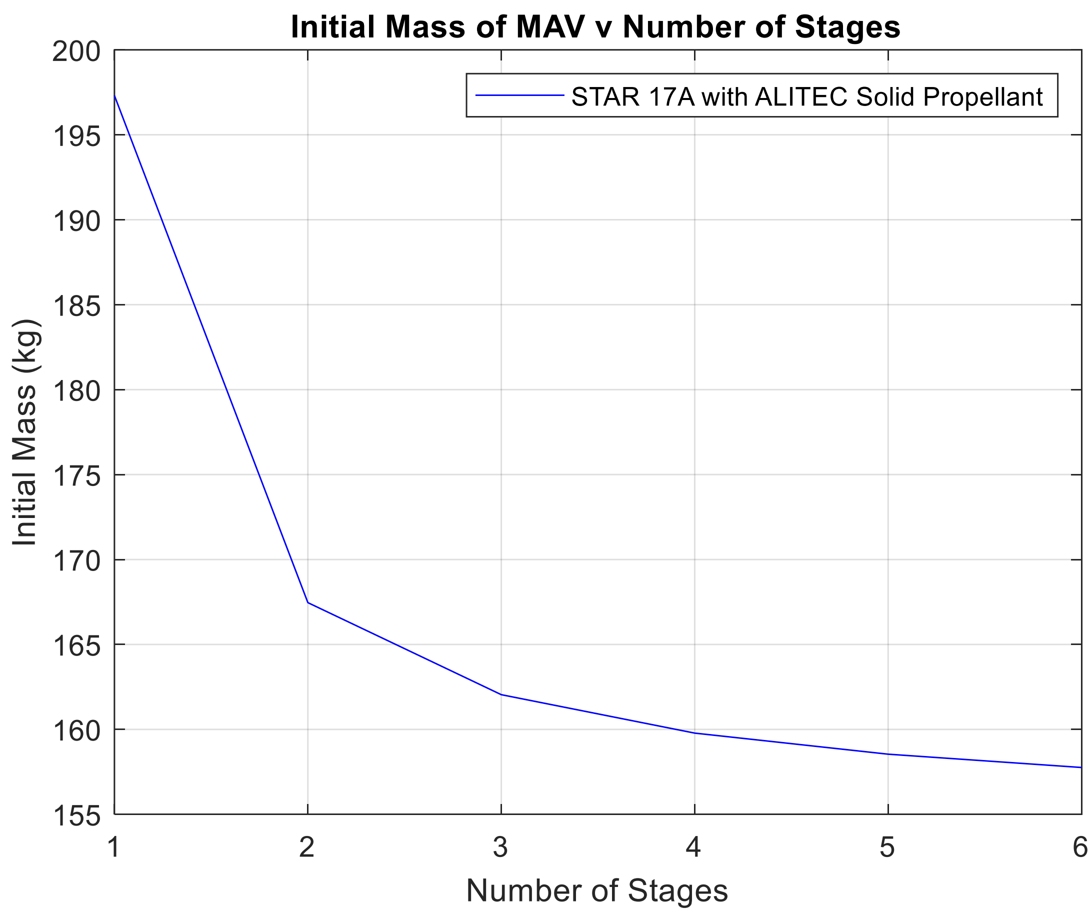


Figure 9: Initial Mass v Stages STAR17A (MATLAB Script #4)

In comparison to Figure 8, the initial mass versus number of stages for the STAR 17A is similar. The initial mass for a single stage MAV is less than our maximum mass; however, since we know the smaller the initial mass the better in terms of efficiency, we know we should use more than a single stage. Following the pattern from Figure 8, we can determine that a two-stage design for the MAV is the best since it has a low initial mass but isn't as complicated as a MAV design with three or more stages.

Lastly, we plotted the initial mass versus the number of stages for the AEROTECH N1000 solid motor with the AEROTECH White Lightning solid propellant. This motor would need to be set up in parallel since it is significantly smaller than the STAR 12GV and the STAR 17A. The N1000 has an inert mass of approximately 4.478 kg and a propellant mass of approximately 7.925 kg. Using our f_{inert} equation, we find that the N1000 has an inert mass fraction of approximately 0.361. The White Lightning propellant has a specific impulse value of 116.1 seconds. By using these values, along with $\Delta V = 3341.957 \frac{m}{s}$ and $g_0 = 9.81 \frac{m}{s^2}$. The plot outputted from the following inputs is shown below.

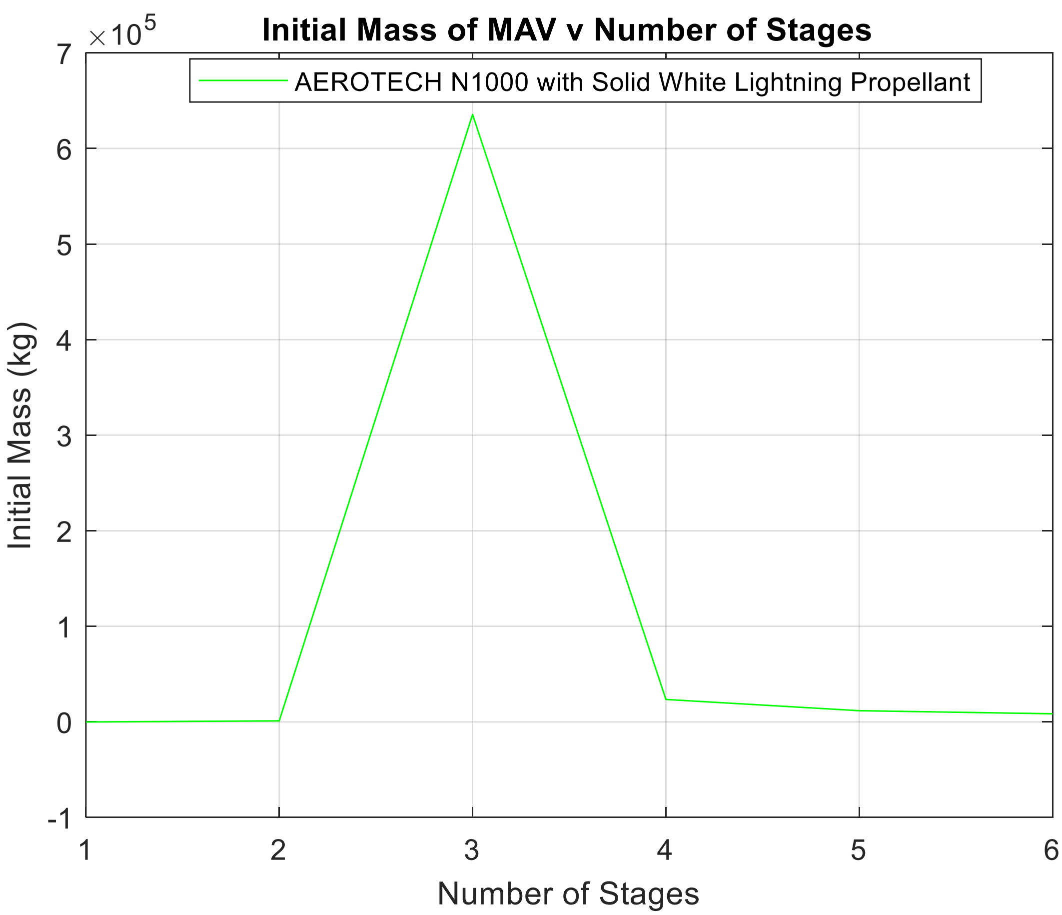


Figure 10: Initial Mass vs Stages AEROTECH N1000 (MATLAB Appendix Script #4)

This plot is different than the plots shown in Figure 8 and Figure 9 which means it needs to be analyzed differently. We can see that the initial mass of the MAV would be slightly negative in this plot at one stage and every stage following the first would have an initial mass value that exceeds our maximum 400 kilograms. This plot is a good example of a configuration that will not work in the MAV without the pairing of another stage with a different motor.

After observing Figure 8, Figure 9, and Figure 10, we determined that using a three-stage design is too complex to make an accurate decision on the velocity split. We can gather the same results in terms of initial mass and velocity by using two stages without needing to add another separating mechanism. Therefore, based on our observations from these three concepts, we settled on 2 stages for the MAV.

3.3.3 *Initial Weight Estimates*

Given the number of stages we have determined (n = 2), we can calculate the initial weight of the launch vehicle using this equation:

$$m_i = m_{pay} * \frac{e^{\frac{\Delta V_1}{g_0 * I_{sp,1}}} * \left(1 - f_{inert,1}\right)}{1 - \left(f_{inert,1} * e^{\frac{\Delta V_1}{g_0 * I_{sp,1}}}\right)} * \frac{e^{\frac{\Delta V_2}{g_0 * I_{sp,2}}} * \left(1 - f_{inert,2}\right)}{1 - \left(f_{inert,2} * e^{\frac{\Delta V_2}{g_0 * I_{sp,2}}}\right)}$$

We will use the example motors and propellant combinations from Section 3.3.2, the NGC STAR 12GV, the NGC STAR 17A, and the AEROTECH N1000. The initial mass calculations for each will be shown below where $\mathrm{m_{pay}} = 37\ \mathrm{kg}$, $\mathrm{g_0} = 9.81\frac{\mathrm{m}}{\mathrm{s^2}}$, and $\Delta\mathrm{V} = 3341.96\frac{\mathrm{m}}{\mathrm{s}}$.

For the ΔV split between the two stages, we will be using the actual ΔV split values found in Table 1 to get an estimate of where our initial mass will be. This means that $\Delta\mathrm{V_1} = 1495\frac{\mathrm{m}}{\mathrm{s}}$ and $\Delta\mathrm{V_2} = 1673\frac{\mathrm{m}}{\mathrm{s}}$. The calculations with the initial weight equation are shown below for each example motor.

NGC STAR 12GV: $f_{inert,1} = f_{inert,2} = 0.214;\ I_{sp,1} = I_{sp,2} = 270.7\ s$

$$m_i = 37\ kg * \frac{e^{\frac{1495}{9.81*270.7}} * (1 - 0.214)}{1 - \left(0.214 * e^{\frac{1495}{9.81*270.7}}\right)} * \frac{e^{\frac{1673}{9.81*270.7}} * (1 - 0.214)}{1 - \left(0.214 * e^{\frac{1495}{9.81*270.7}}\right)} = 193.3921\ kg$$

NGC STAR 17A: $f_{inert,1} = f_{inert,2} = 0.106;\ I_{sp,1} = I_{sp,2} = 270.7\ s$

$$m_i = 37\ kg * \frac{e^{\frac{1495}{9.81*270.7}} * (1 - 0.106)}{1 - \left(0.106 * e^{\frac{1495}{9.81*270.7}}\right)} * \frac{e^{\frac{1673}{9.81*270.7}} * (1 - 0.106)}{1 - \left(0.106 * e^{\frac{1495}{9.81*270.7}}\right)} = 147.1829495\ kg$$

AEROTECH N1000: $f_{inert,1} = f_{inert,2} = 0.361;\ I_{sp,1} = I_{sp,2} = 116.1\ s$

$$m_i = 37\ kg * \frac{e^{\frac{1495}{9.81*116.1}} * (1 - 0.361)}{1 - \left(0.361 * e^{\frac{1495}{9.81*116.1}}\right)} * \frac{e^{\frac{1673}{9.81*116.1}} * (1 - 0.361)}{1 - \left(0.361 * e^{\frac{1495}{9.81*116.1}}\right)} = 2092.049242\ kg$$

From these calculations the NGC STAR 12GV and NGC STAR 17A produce reasonable initial mass values at two stages with initial masses of 193.4 kilograms and 147.1829495 kilograms respectively. The AEROTECH N1000 exceeds are maximum mass of 400 kilograms significantly, making it unreasonable to use by itself in this staging configuration. Therefore, all future considerations of the motor will be in parallel.

4 Detailed Concept

In this next section, the team will detail different propulsion, payload, guidance and control, and integrated design concepts generated and discussed while determining the best possible MAV design. This section will detail the reasoning as to why each concept was considered and, ultimately, what concepts were used in the final design.

4.1 Concept Generation

4.1.1 *Propulsion System Concepts*

Due to the results of the analysis performed on the staging of the Mars Ascent Vehicle in section 3.3.2, the team would consider only a two-stage rocket. Additionally, from the analysis of the feasibility of rocket motors and propellants in section 3.3.1, we used the fuel characteristics provided in Table 2 to determine potential propellants for the system. Additionally, in Table 3, the table shows the features in the corresponding propulsion systems configurations for the Mars Ascent Vehicle.

Table 2: The Considered Propellants and their Properties

Supplier	Propellant	Propellant State	I_{sp} (s)	Operation Temperature (°C)	Storability Characteristics	Other Characteristics
Aerojet Rocketdyne	Monomethylhydrazine / Dinitrogen Tetroxide	Liquid Bipropellant	336.0	$-52 \leq T_{MMH} \leq 87.5$ $-11.2 \leq T_{N_2O_4} \leq 21$	MMH and N_2O_4 are hypergolic, so they are unsafe to store	As a bipropellant, it would require two large storage tanks
Aerojet Rocketdyne	Hydrazine	Liquid Monopropellant	228	$1.4 \leq T \leq 113.5$	Requires heaters to operate on Mars	Used on Mars Reconnaissance Orbiter
NASA	Nitrous Oxide / Paraffin	Hybrid	N/A	Experimental, but is designed to operate in cold climates on Mars	NASA is hopeful this combination is the future of Mars Propellant	Experimental, it is not mission ready
Northrop Grumman	TP-H-3340A	Solid	281.8	$4.44 \leq T \leq 35$	Requires heaters to operate on Mars	NASA is looking for alternative fuels
Adranos	ALITEC	Solid	270.7	~ 20 *	Requires heaters to operate on Mars	95% Less Hydrogen Chloride Production
Aerotech Consumer Aerospace	White Lightning	Solid	116.1	~ 20 *	Must be stored at 4.44 – 37.77°C	Limited testing beyond high power rocketry

*Optimal testing temperature

Table 3: Considered Motors and their Properties

Supplier	Motor	Motor Stage	Propellant	Operation Temperature (°C)	Diameter (m)	Length (m)	f_{inert}
Northrop Grumman	Star 17A	First	Solid	$-17.78 \leq T \leq 43.33$	0.44196	0.9815	0.119
Northrop Grumman	Star 24	First	Solid	$-17.78 \leq T \leq 43.33$	0.62230	1.0287	0.08399
Northrop Grumman	Star 12GV	Upper	Solid	$4.44 \leq T \leq 35$	0.31090	0.5715	0.2138
Animal Motor Works, Inc.	ANIMAL MOTOR WORKS N4000	First/Upper	Solid	~ 20 *	0.09800	1.213	0.446
Aerotech Consumer Aerospace	AEROTECH N1000	First/Upper	Solid (White Lightning)	~ 20 *	0.09800	1.046	0.351
Aerojet Rocketdyne	MRM 106-F RCS	Thruster	Liquid (Hydrazine)	$8.00 \leq T \leq 53.00$	0.06400	0.178	0.41334
Aerojet Rocketdyne	AJ10-220 RCS	Thruster	Liquid (MMH/NTO)	$4.44 \leq T \leq 43.33$	0.07300	0.19000	0.59768

*Optimal testing temperature

In Table 2, there were six propellant heavily considered for the fuel for the MAV. The first two liquid propellants listed are intended for use in RCS thrusters. Monomethylhydrazine and dinitrogen tetroxide is a liquid bipropellant with a high I_{sp} of 336 seconds. Our main concern with this propellant is the required two storage tanks for the bipropellant. Another major concern is that because the fuels are hypergolic, they will ignite spontaneously if they interact. Therefore, if there is a leak in a fuel tank of the MAV, the entire upper stage fuel source would explode. This would destroy the structure of the MAV, so this propellant choice was avoided for the thrusters. However, hydrazine is a liquid monopropellant that has been proven for long storage missions when it was used on the Mars Reconnaissance Orbiter. Additionally, as a monopropellant, it requires only one tank which would save space and limit the inert mass. Due to these reasons, hydrazine was selected as the fuel source for the MAV's RCS system. The remaining four propellants in Table 2 were considered for the main two stages of the Mars Ascent Vehicle. The team considered a hybrid nitrous oxide and paraffin hybrid propellant for the mission. This hybrid fuel is in development at NASA for future Mars missions as it supposedly has an extremely low operating temperature. However, NASA's research is still experimental, and no substantial data was available. Additionally, we do not believe that the fuel has enough time to get properly researched and tested before the launch date from Earth in 2026. Due to these reasons, we did not highly consider this propellant combination as an effective fuel for the MAV because we could not provide definitive conclusions for it. Northrop Grumman's TP-H-3340A is a solid propellant with a high I_{sp} that is currently NASA's propellant for the mission. The fuel also exhibits a reasonable operating temperature that can easily be obtained through using heaters. While this fuel seems optimal for the mission, NASA wants to consider alternate designs as this fuel can become difficult to control when burning. Adranos ALITEC is a solid propellant which has a high I_{sp} of 270.7 seconds. This fuel also has an added benefit of being cleaner than most alternatives by producing 95% less hydrogen chloride by mass during launch (Adranos, 2020). Due to this, the fuel is less harsh on the Martian environment. Furthermore, White Lightning is a commercial high-powered rocket propellant intended for hobby rocketry. Its energy density allows the MAV to use multiple rockets in parallel with this fuel which could be ideal. However, it does exhibit a low I_{sp} of 116.1 seconds. After examining these propellants, the team determined that the best propellants for our mission would be the ALITEC fuel for its high power, controllability and clean products, as well as the White lightning fuel because of its ability to work in parallel on this rocket.

The first five motors on Table 3 were considered for the main stages of the MAV. The STAR 17A. STAR 24, and STAR 12GV are all produced by Northrup Grumman Corporation and are solid rocket motors. The STAR 17A was considered for our first stage since it is small enough to fit our design constraints and uses solid propellant which prevents fuel freezing on Mars. The STAR 24 was also considered for our first stage; the primary difference between the two is that the STAR 24 is slightly bigger than the STAR 17A, but it is still within our design constraints. The STAR 12GV was considered for an upper stage portion of the MAV since it smaller than both the STAR 17A and the STAR 24 (www.northropgrumman.com, 2016). The AMW N4000 and AEROTECH N1000 were considered as commercial high-powered rocketry motors that could be used in parallel on the same stage. For our main propulsion systems, we selected three motors to analyze further throughout the course of this report. We selected the STAR 17A, as it was smaller than the STAR 24, but could still adequately contain the required amounts of propellant, the STAR 12GV, as it operated as an efficient and proven small upper stage motor, and the AEROTECH N1000 because of its lower mass compared to the AMW N4000. The AEROTECH N1000 also gave use flexibility to operate the motors in parallel if we found this was a more efficient alternative to a single motor like the STAR 17A. The last two motors were considered for the RCS system. The Aerojet Rocketdyne MRM 106-F RCS thruster has small dimensions that can fit within our rocket and uses liquid hydrazine as a propellant. The Aerojet Rocketdyne AJ10-220 RCS thruster is a small motor like the MRM 106-F that utilizes monomethylhydrazine dinitrogen tetroxide as a bipropellant (In-Space Propulsion Datasheets, n.d). For our RCS system, we selected the MRM 106-F RCS thruster since it worked well in our MAV design.

To concentrate on the performance of the propulsion systems and not external factors, the analysis excluded additional complex considerations that we will consider in our detailed design (such as the weights of reaction control system thrusters, avionics, and insulation. For this simplified analysis, we considered an ideal rocket carrying only the Martian rock samples, which has a payload weight of 16 kg. The analysis also used the corresponding inert masses to calculate the propellant masses for each of the two stages in every design. The equation used to calculate these masses using the following equation.

$$m_{prop} = e^{\frac{\Delta V}{g0*Isp}} * \left(m_{pay} - m_{inert}\right) - m_{pay} - m_{inert}$$

We also used estimated ΔV split values for each stage (table 1). These include a ΔV of 1495 m/s for stage 1 and 1673 m/s for stage 2. The initial mass of the MAV was then calculated using equation # shown below.

$$m_i = m_{prop1} + m_{prop2} + m_{inert1} + m_{inert2} + m_{pay}$$

These calculations were analytically performed in MATLAB. The corresponding MATLAB code for this section can be found in Appendix A Script 3.

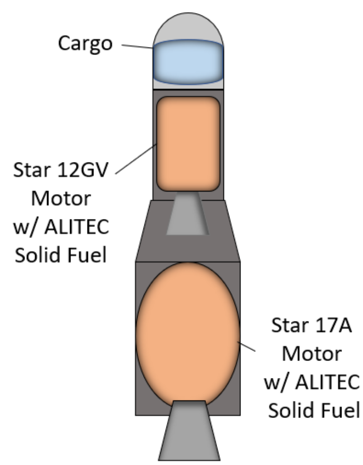


Figure 11: Concept 1 Design

The first two-stage concept design utilized two proven Northrop Grumman Corporation solid motors with Adranos ALITEC fuel. This concept was composed of a STAR 17A motors for the first stage and a STAR 12GV motor as the second stage motor, as shown in Figure 11. Both motors will be fueled by ALITEC solid rocket propellant due to its high but controllable I_{sp} of 270.7s, as well as its reduced hydrogen chloride production. Through using the previously described equations and MATLAB script, the team determined that the STAR 17A would require 45.6322 kilograms of propellant and the STAR 12GV would require 21.9634 kilograms of propellant. The total rocket mass would be approximately 106.0021 kilograms. The full calculated results are shown below in Table 4.

Table 4: Concept 1 Calculated Stage and Total Mass

	Stage 1	Stage 2
	NGC STAR 17A	NGC STAR 12GV
Inert Mass (kg)	13.38	9.0265
Propellant Mass (kg)	45.6322	21.9634
Propellant I_{sp} (s)	270.7	270.7
Stage Mass (kg)	59.0122	30.9899
Total MAV Mass (kg)	106.0021	

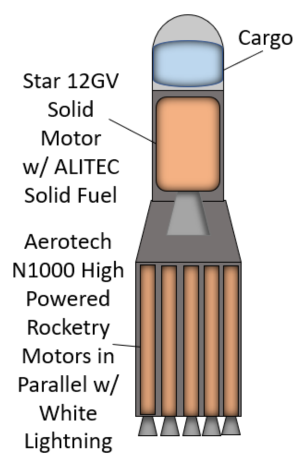


Figure 12: Concept 2 Design Concept

Our second design consisted of two stages. The first stage would be powered by AEROTECH N1000 commercial high-powered rocketry motors, and the second stage would be propelled by a STAR 12GV motor. The STAR 12GV would be fueled by ALITEC solid rocket propellant again for the reasons stated in the first design. The AEROTECH motors are power with white lightning solid rocket propellant. Through ideal rocket equation calculations, the team determined that the STAR 12GV would require 21.9634 kilograms of ALITEC propellant and 9.1655 AEROTECH N1000 motors in parallel. The total rocket mass would be about 160.6695 kilograms. Other important calculations concluded the results shown in Table 5.

Table 5: Concept 2 Calculated Stage and Initial Mass Values

	Stage 1	Stage 2
	9.1655 x AEROTECH N1000 High Powered Rocketry Motors in Parallel	NGC STAR 12GV
Inert Mass (kg)	41.0431	9.0265
Propellant Mass (kg)	72.6365	21.9634
Propellant I_{sp} (s)	116.1	270.7
Stage Mass (kg)	113.6796	30.9899
Total MAV Mass (kg)	160.6695	

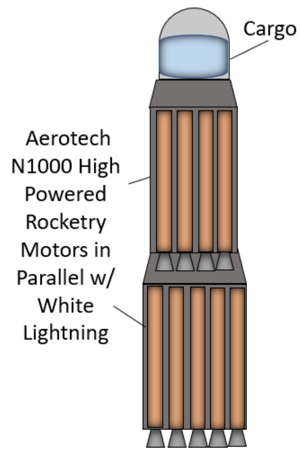


Figure 13: Concept 3 Design Concept

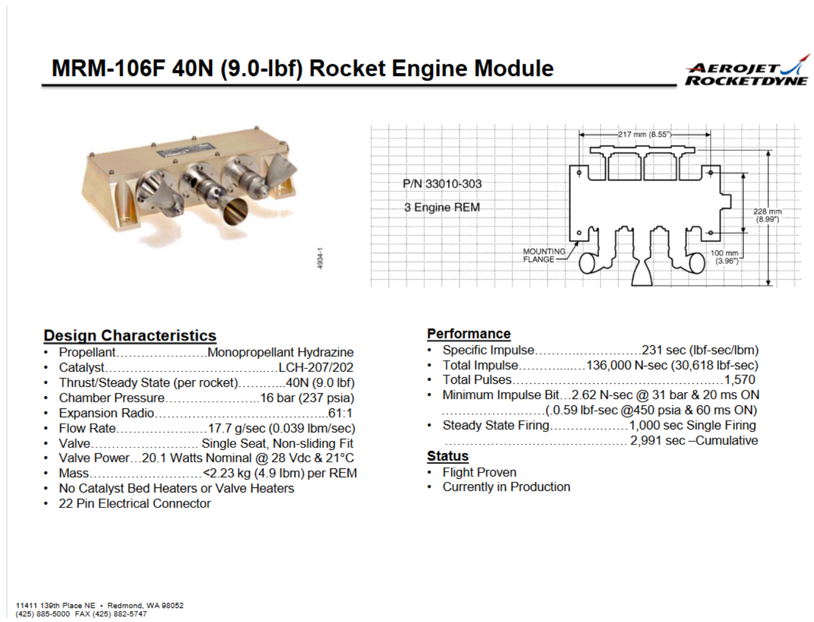
Our third design consisted of two stages, both powered by AEROTECH N1000 commercial high-powered rocketry motors. These motors contain the White Lightning solid rocket propellant. Through ideal rocket equation calculations, it was determined that the first stage would consist of 9.9278 AEROTECH motors, and the second stage would consist of approximately 4.4902. The total rocket mass would be about 178.826 kilograms. Other calculations concluded:

Table 6: Concept 3 Calculated Stage and Initial Mass

	Stage 1	Stage 2
	9.9278x AEROTECH N1000 High Powered Rocketry Motors in Parallel	4.4902x AEROTECH N1000 High Powered Rocketry Motors in Parallel
Inert Mass (kg)	44.4565	20.1071
Propellant Mass (kg)	78.6776	35.5848
Propellant I_{sp} (s)	116.1	116.1

Stage Mass (kg)	123.1341	55.6919
Total MAV Mass (kg)	178.826	

Additionally, the team determined to mitigate the risk of veering off course due to unforeseen circumstances, a reaction control system was required. When selecting our reaction control system (RCS), we considered two designs. One would be to utilize a similar system to that proposed by Darius Yaghoubi and Andrew Schnell of NASA. This system would have two MRM-106F 40N rocket motor modules, depicted below in Figure 14, facing opposite lateral directions (Wilson, 2020). It would be powered by liquid hydrazine stored in a near-spherical 12-inch diameter aluminum alloy tank located below the thrusters in the rocket (Diaphragm propellant tanks, 2013). The tank proposed would be a MOOG LEO Satellite tank, as depicted below in Figure 15, with a diameter of 12 inches. We estimate that the tank would be able to hold up to 10.9 kilograms of hydrazine, which the team believes will be more than enough for our intended mission. One final note on this design would be the concern of keeping the hydrazine liquid. Hydrazine freezes at 2 degrees Celsius (Price and Evans, 1968). Therefore, heating elements and insulation will be required to keep the hydrazine liquid during its flight.


MRM-106F 40N (9.0-lbf) Rocket Engine Module

AEROJET ROCKETDYNE

P/N 33010-303

3 Engine REM

217 mm (8.55")

228 mm (8.99")

100 mm (3.96")

MOUNTING FLANGE

Design Characteristics
- Propellant......................Monopropellant Hydrazine
- Catalyst..LCH-207/202
- Thrust/Steady State (per rocket)...........40N (9.0 lbf)
- Chamber Pressure......................16 bar (237 psia)
- Expansion Radio...61:1
- Flow Rate......................17.7 g/sec (0.039 lbm/sec)
- Valve.......................... Single Seat, Non-sliding Fit
- Valve Power…20.1 Watts Nominal @ 28 Vdc & 21°C
- Mass..........................<2.23 kg (4.9 lbm) per REM
- No Catalyst Bed Heaters or Valve Heaters
- 22 Pin Electrical Connector

Performance
- Specific Impulse..........................231 sec (lbf-sec/lbm)
- Total Impulse................136,000 N-sec (30,618 lbf-sec)
- Total Pulses..1,570
- Minimum Impulse Bit…2.62 N-sec @ 31 bar & 20 ms ON ....(.0.59 lbf-sec @450 psia & 60 ms ON)
- Steady State Firing..................1,000 sec Single Firing .... 2,991 sec –Cumulative

Status
- Flight Proven
- Currently in Production

11411 139th Place NE • Redmond, WA 98052
(425) 885-5000 FAX (425) 882-5747

Figure 14: MRM-106F 40N Rocket Engine Module (In-Space Propulsion Datasheets, n.d.)

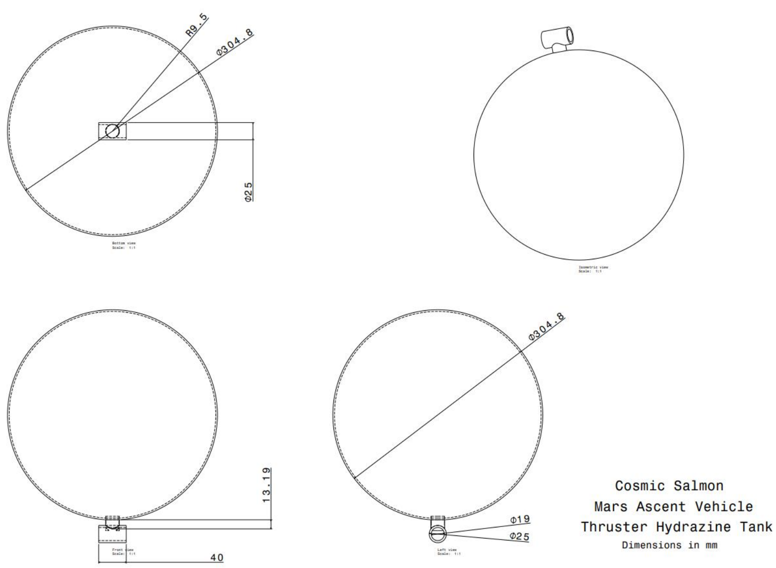


Figure 15: MOOG LEO Satellite Hydrazine Tank (Appendix B Figure 31)

The second RCS design that we considered involved AJ10-220 Reaction Control Thruster, shown below in Figure 16, powered by monomethylhydrazine/dinitrogen tetroxide liquid bipropellant (Wilson, 2020). The main concern with this design would be the required two tanks to hold the bipropellant. Fitting two tanks inside of our rocket would be a challenge, considering our height and width constraints. The tanks we selected were the MOOG Hit to Kill Monomethylhydrazine, with a diameter of 0.1397 m, and MOOG Hit to Kill Dinitrogen Tetroxide, with a diameter of 0.12446 m (Diaphragm propellant tanks, 2013).

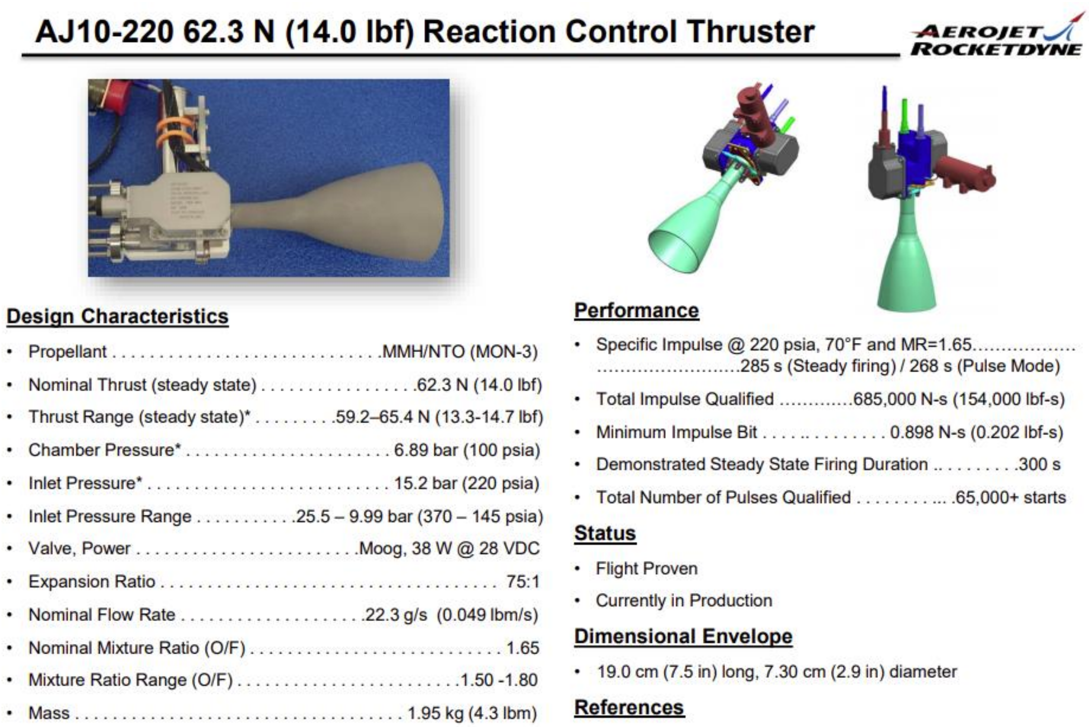


Figure 16: AJ10-220 Reaction Control Thruster (Aerojet Rocketdyne, 2020)

Finally, the team investigated insulation materials for our propulsion systems. The insulation will be able to help keep the chosen propellants within their operable temperatures during launch. We determined two distinct possibilities for the insultation: silica/NBR and asbestos/SBR. Both insulations have been approved by NASA within a 1976 analysis on insulating propellant storage systems, so the team assumes that both insulations are appropriate options for the final design (Solid Rocket Motor Internal Insulation, 1976).

4.1.2 *Payload System Concepts*

As the MAV's mission revolves around the ability to transport cargo to orbit, significant design considerations must be considered in the safe storage and deployment of this cargo. To decrease the complexity of the design, the MAV will store the cargo in a payload system at the top of the rocket structure. This would allow the cargo to easily be released and transported to the Earth Return Orbiter during rendezvous. Two main design options were considered in the payload system for the MAV.

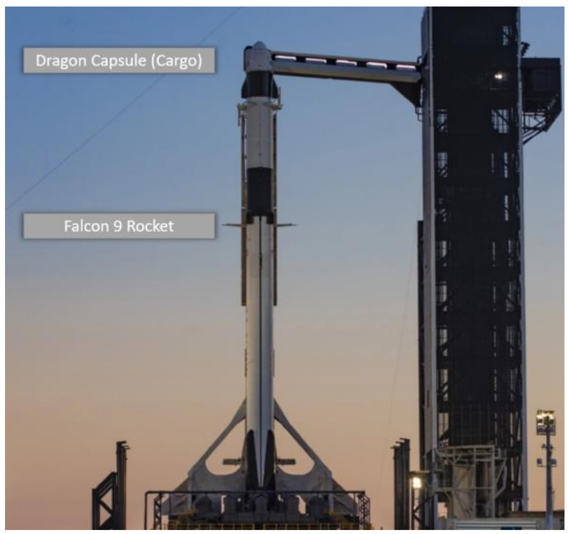


Figure 17: The Dragon Capsule Sits on Top of the Falcon 9 and is Used as the Nose of the Rocket (SpaceX, n.d.)

The first cargo storage design would use the cargo container as the nose of the MAV during launch. A similar concept to this is SpaceX's use of the Dragon capsule positioned on the top of the Falcon 9 rocket, as seen in Figure 17. This design would enable the MAV to be constructed with less structural weight as the cargo container would serve as the nose and not require any additional materials to be used. However, if the cargo container was unprotected at the top of the MAV, it would experience extreme temperatures, large aerodynamic forces, and weathering during the ascent through the Martian atmosphere.

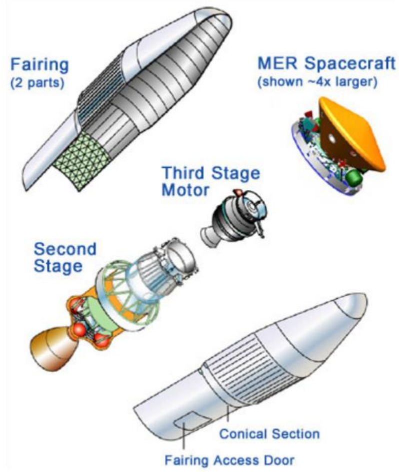


Figure 18: Example of Fairings Used on the Mars Exploration Rover (mars.nasa.gov, n.d.)

The main alternative concept that was considered to combat this challenge was the use of rocket fairings. According to RUAG, the world's leader in manufacturing rocket fairings, fairings make up the upper part of the rocket and protect the cargo from extreme temperatures, dust, humidity or rain, noise, aerodynamic forces, and mechanical loads (www.ruag.com, n.d.). The visualization of the use of fairings are represented in Figure 18 above.

4.1.3 Guidance, Navigation, and Control System Concepts

The Mars Ascent Vehicle has a mission that requires a specific orbit to rendezvous with the Earth Return Orbiter, so maintaining the mission trajectory is vital. As the vehicle is autonomous, it must maintain the route to the correct orbit by itself. As previously discussed, RCS thrusters are considered for all designs to provide the trajectory adjustments. Additionally, for Concepts 1 and 2, the STAR 12GV motor can supply thrust vector control capabilities to guide the MAV along its trajectories (www.northropgrumman.com, 2016). However, as the MAV is required to reach a ΔV, the second stage motor provides vertical thrust while the guidance is mainly allocated to the thrusters

As controlling the MAV requires sensors and GN&C techniques, an avionics bay is required. As the avionics bay serves a very specific purpose for the MAV, the team determined that the proposed design for the avionics bay by Darius Yaghoubi and Andrew Schnell of NASA would be the only concept for the avionics bay. This is because the team is limited in expertise on avionics systems and does not want to require NASA to provide more funding to research alternatives. This proposed design would be like Figure 19 below. It would use the Sphinx flight computer (~1 kg), Honeywell HG5700 IMU (1.36 kg) (HG5700 Inertial Measurement Unit, 2020), Blue Canyon Technologies Nano Star Tracker (0.35 kg) (https://www.bluecanyontech.com/, 2020), ISIS TRXVU transmitter (0.085 kg) (TRXUV VHF/UHF Transceiver, 2016), and approximately 7 SAFT 176065 cells (1.05 kg) (MP 176065 XLR Rechargeable Li-ion Cell, 2018) mounted on a PCB and sled (~0.5 kg) (Yaghoubi and Schnell, 2020). These parts were all chosen based on their mass and ability to operate effectively at Martian temperatures. In total, the team estimates the mass of the avionics bay to be approximately 4.5 kilograms.

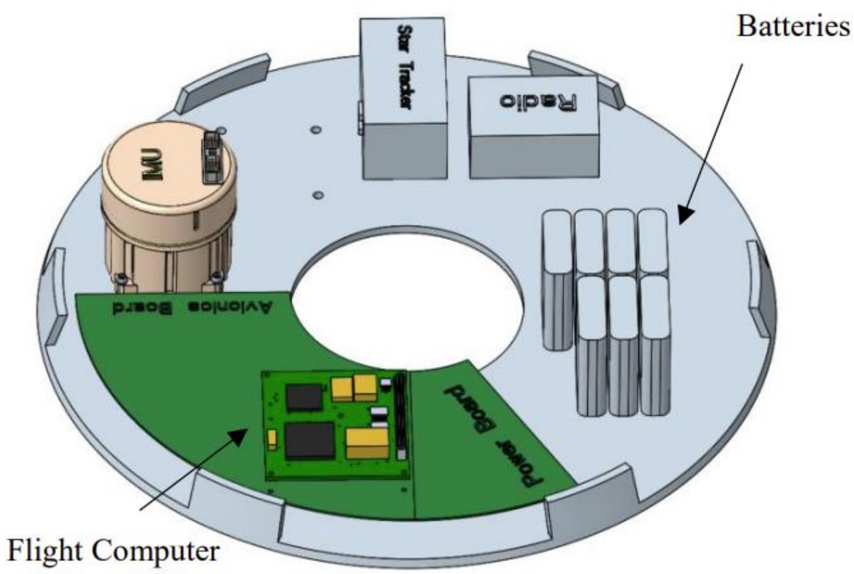


Figure 19: The conceptual avionics bay designed by Darius Yaghoubi and Andrew Schnell of NASA (Yaghoubi and Schnell, 2020)

4.1.4 *Integrated MAV System Design Concepts*

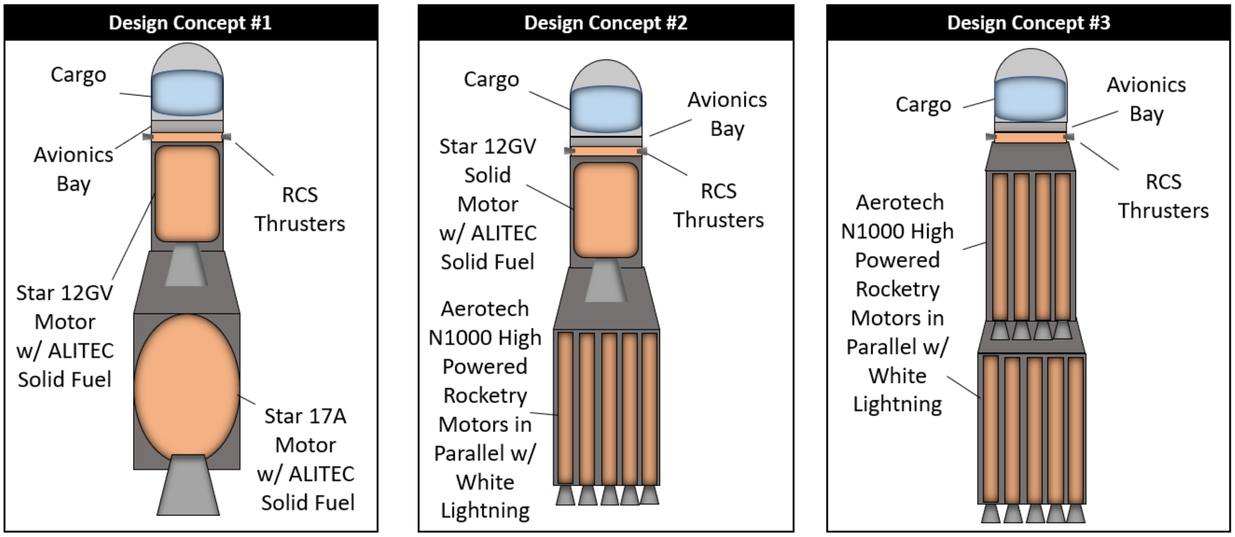


Figure 20: Design Concept Configurations

As the analysis of all the simplified basic propulsion systems yielded MAV concepts with initial masses below the specified 400 kilograms, all three concepts were expanded upon for the final design concept selections. The team then added RCS thrusters, an avionics bay, a cargo bay, and insulation in the MAV design. The final generated concepts are displayed in Figure 20 above.

4.2 Concept Selection

In order to select the best propulsion design concept, we plotted the initial mass of the conceptual designs from 4.1 versus an F1 value that will give us the designated ΔV split and corresponding initial mass. We show our plots to determine the optimized initial mass configuration below.

For staging design 1, we are planning on using the Adranos ALITEC Solid Propellant (refer to 3.3.2 for more information) for both stages of the MAV, so $I_{sp,1} = I_{sp,2} = 270.7$ seconds. Our first stage is using the NGC STAR 17A Solid Motor where $f_{inert,1} = 0.106$. Our second stage is using the NGC STAR 12GV Solid Motor where $f_{inert,2} = 0.214$.

We then had to calculate the ΔV division for our design. In order to do this, we used MATLAB Script #6 in Appendix A to plot the values of F1 and ultimately find the split between the velocities for stages one and two. The following plots were outputted from the code for each concept design:

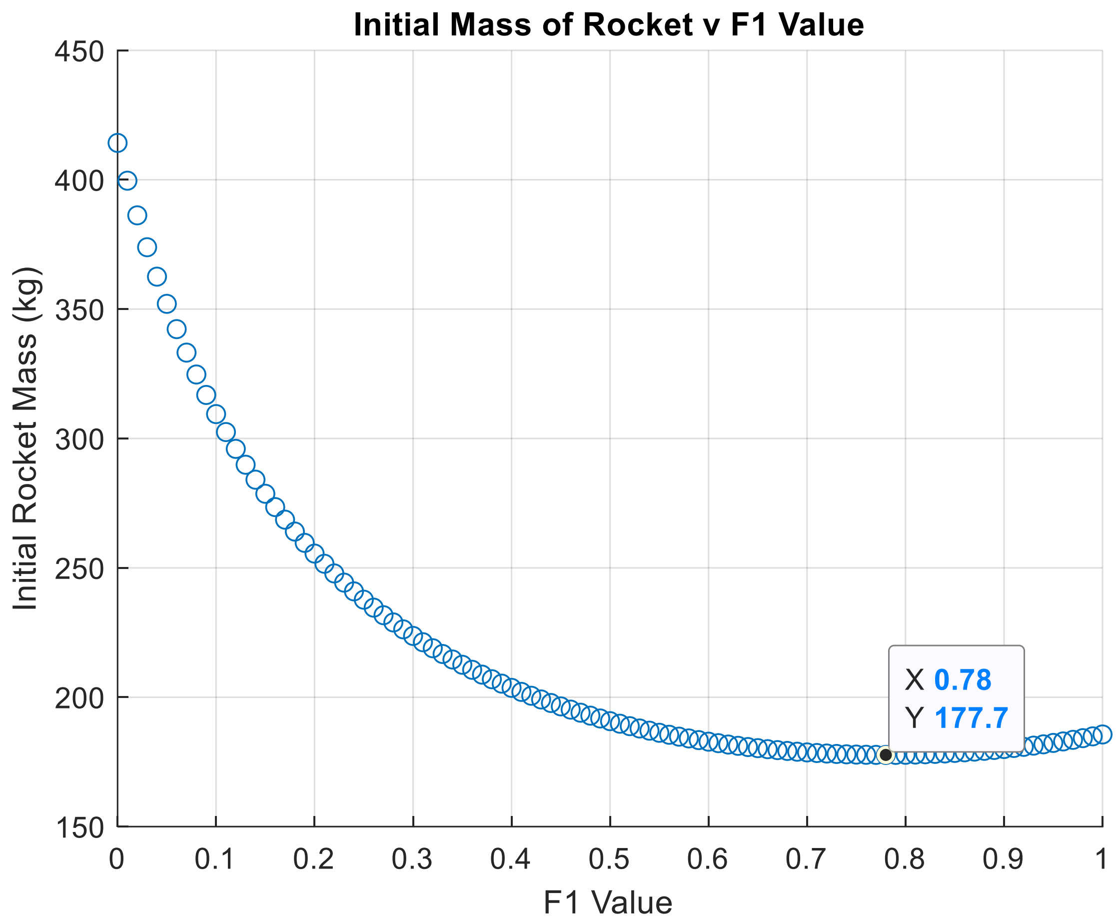


Figure 21: Initial Mass Concept 1 v First Stage Delta-V Fraction (Appendix A Script #6)

As seen in Figure 21, the NGC STAR 12GV and NGC STAR 17A combination gives us a F1 value of 0.78 and an initial rocket mass of 177.7 kilogram. Once we use this F1 value for the ΔV division, we can then calculate the two separate velocities for each stage, giving us a ΔV_1 value of 2606.726877 m/s and a ΔV_2 value of 735.2306575 m/s for the second stage.

We then moved on to staging design 2 where we will be using the AEROTECH N1000 with the AEROTECH White Lightning solid propellant for stage 1 and the NGC STAR 12GV with the Adranos ALITEC Solid Propellant for stage 2. Section 3.3.2 provides the propellant details. The N1000 has an inert mass fraction value of 0.361 and the STAR 12GV has an inert mass fraction value of 0.214. The White Lightning propellant has an I_{sp} value of 116.1 seconds and the ALITEC propellant has an I_{sp} value of 270.7 seconds. We plotted the initial rocket mass v F1 resulting in the following plot:

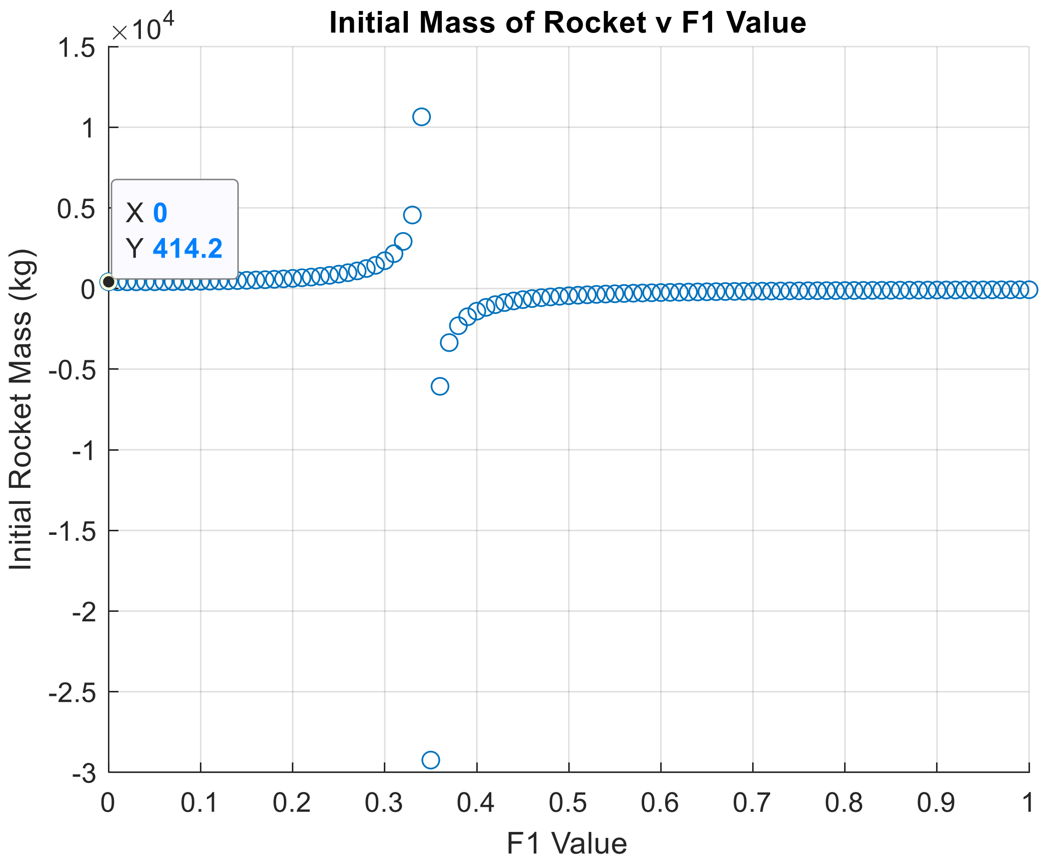


Figure 22: Initial Mass Concept 2 v First Stage Delta-V Fraction (Appendix A Script #7)

As seen in the Figure 22, the stage one 12 AEROTECH N1000 motors in parallel and the stage two NGC STAR 12GV gives us a F1 value of 0 and an initial mass of 414.2 kilograms. This initial mass is higher than the maximum 400-kilogram mass. Unfortunately, that makes this propulsion design impossible to pursue.

We finally moved on to staging design 3 where we will be using the AEROTECH N1000 with the AEROTECH White Lightning solid propellant for stage 1 and stage 2 (see 3.3.2 for propellant details). The N1000 has an inert mass fraction value of 0.361 and the White Lightning propellant has an I_{sp} value of 116.1 seconds. We plotted the initial rocket mass v F1 resulting in the following plot:

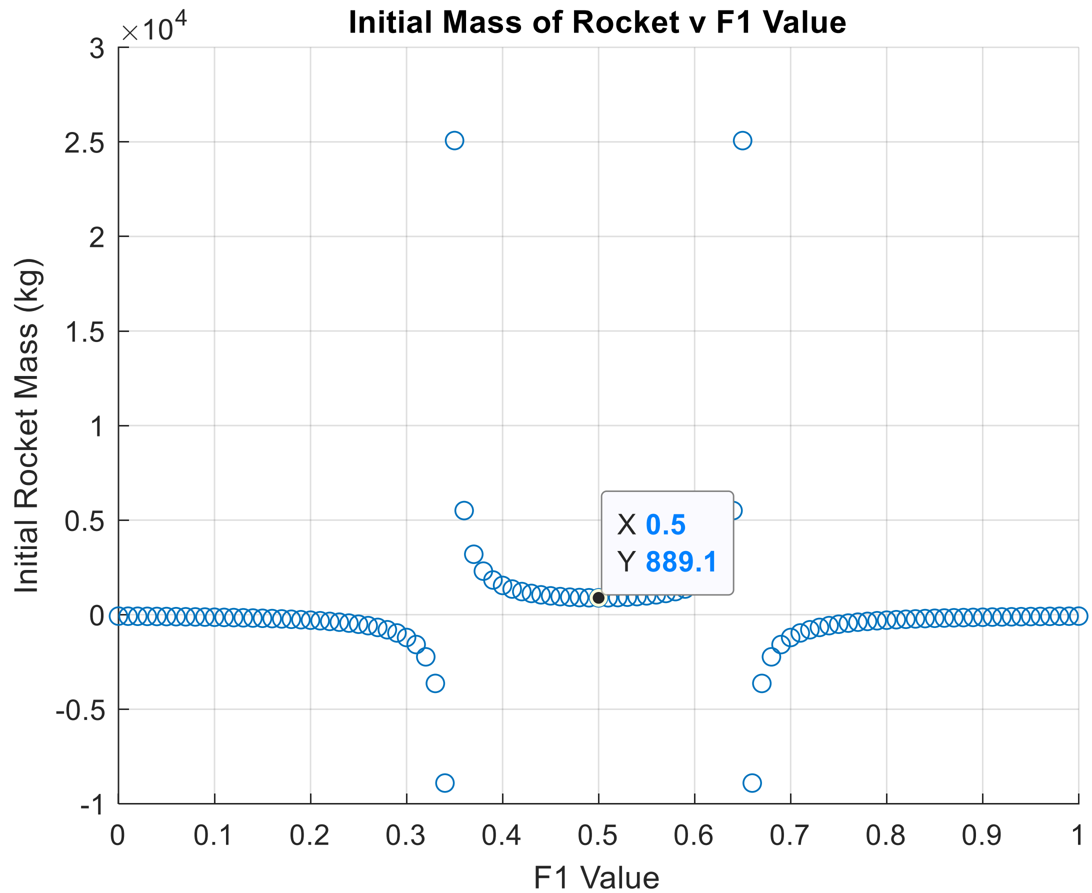


Figure 23: Initial Mass Concept 3 v First Stage Delta-V Fraction (Appendix A Script #8)

As seen in the Figure 23, the stage one 10 AEROTECH N1000 motors in parallel and the stage two 4 AEROTECH N1000 motors in parallel gives us a F1 value of 0.25 and an initial mass of 889.1 kilograms. This initial mass is far higher than the maximum 400-kilogram mass, making this propulsion design impossible to pursue.

With regards to the main propulsion system on the rocket, only our first concept falls within our required initial mass. Therefore, the team will select that design to move forward with.

Plugging all these values into the equation:

$$m_i = m_{pay} * \frac{e^{\frac{\Delta V_1}{g_0 * I_{sp,1}}} * \left(1 - f_{inert,1}\right)}{1 - \left(f_{inert,1} * e^{\frac{\Delta V_1}{g_0 * I_{sp,1}}}\right)} * \frac{e^{\frac{\Delta V_2}{g_0 * I_{sp,2}}} * \left(1 - f_{inert,2}\right)}{1 - \left(f_{inert,2} * e^{\frac{\Delta V_2}{g_0 * I_{sp,2}}}\right)}$$

$$m_i = 37 * \frac{e^{\frac{2606.726877\frac{\mathrm{m}}{\mathrm{s}}}{9.81\frac{m}{s^2} * 270.7s}} * (1 - 0.106)}{1 - \left(0.106 * e^{\frac{2606.726877\frac{\mathrm{m}}{\mathrm{s}}}{9.81\frac{m}{s^2} * 270.7s}}\right)} * \frac{e^{\frac{735.2306575\frac{\mathrm{m}}{\mathrm{s}}}{9.81\frac{m}{s^2} * 270.7s}} * (1 - 0.214)}{1 - \left(0.214 * e^{\frac{735.2306575\frac{\mathrm{m}}{\mathrm{s}}}{9.81\frac{m}{s^2} * 270.7s}}\right)} = 177.7\ kg$$

Considering insulation of the rocket motors, we investigated two distinct possibilities: silica/NBR and asbestos/SBR. Both insulations were approved by NASA within a 1976 analysis on insulating propellant storage systems (Solid Rocket Motor Internal Insulation, 1976). The team decided that both are acceptable for the rocket.

When determining our reaction control system (RCS), the team decided that the best path forward would be to utilize the similar system to that proposed by Darius Yaghoubi and Andrew Schnell of NASA. This system utilized the two MRM-106F 40N rocket engine modules and a MOOG LEO Satellite tank. We selected this design over the alternative bipropellant design for a few reasons. For one, the bipropellant required two tanks which concerned the team as these tanks would have to be very small to fit into the rocket. The monopropellant tank can take up more room and contain more fuel. Even though the monopropellant design has a lower I_{sp}, 231 seconds versus 285 seconds, this increased propellant mass is more ideal. Additionally, the AJ10-220 Reaction Control Thruster has only one outlet in one direction. The MRM-106F has three outlets per module. This conserves more space and allows for better control of the MAV.

When considering the payload system concepts, we wished to limit the risk of damaging the cargo on the ascent from Mars. Therefore, we chose fairings as our method of cargo containment. While it does add more mass to the design, it will supply the owners, supporters, and the critics of the mission with more confidence that the samples will not be damaged.

With regards to our guidance, navigation, and controls (GNC) systems, we only considered one option which we will select as our final design. This design was also proposed by Darius Yaghoubi and Andrew Schnell of NASA. The avionics bay we selected was composed of a Sphinx flight computer, Honeywell HG5700 IMU, Blue Canyon Technologies Nano Star Tracker, ISIS TRXVU transmitter, and approximately 7 SAFT 176065 cells. This design is both lightweight (4.5kg) and optimized to work on Mars.

4.3 Concept Refinement

Our final design for the Mars Ascent vehicle is a two-stage miniature rocket where the first stage is a single STAR 17A motor and the second stage is a single STAR 12GV motor. Both are fueled by Adranos ALITEC solid propellant with an I_{sp} value of 270.7 seconds. At the top of the rocket, we have our payload fairing where our 16-kilogram payload of Martian soil will be kept for travel. Directly underneath the payload fairing is the 4.5 kilograms avionics bay that is composed of a Sphinx flight computer, Honeywell HG5700 IMU, Blue Canyon Technologies Nano Star Tracker, ISIS TRXVU transmitter, and approximately 7 SAFT 176065 cells mounted on a PCB and sled. Underneath the avionics bay is the two MRM-106F 40N RCS thrusters which face away from each other. These thrusters will provide additional stability and velocity to get into stable orbit prior to the rendezvous with the Earth Return Orbiter. Attached underneath these thrusters is the MOOG LEO Satellite tank containing hydrazine which will fuel the MRM-106F RCS thrusters externally. Below this tank is our second stage STAR 12GV motor surrounded by insulation and heaters to keep it within its operating temperature range since the temperature on Mars is too cold for the propellant to operate efficiently without protection. Finally, below our second stage motor, is our first stage STAR 17A motor surrounded by insulation and heaters to keep it within its operating temperature since it has similar temperature constraints as the STAR 12GV and needs protection.

Using our optimal ΔV breakdown calculated in Section 3.3.3, ΔV_1 = 2439.628544 m/s and ΔV_2 = 902.3283657 m/s. From these values, our updated rocket weight, we can calculate accurate amounts of ALITEC propellant needed and find a final estimate for our proposed MAV weight. The equations we will use to find these are seen below.

$$m_{prop} = e^{\frac{\Delta V}{g0*Isp}} * \left(m_{pay} - m_{inert}\right) - m_{pay} - m_{inert}$$

$$m_i = m_{avionics} + m_{RCS} + m_{stage1} + m_{stage2}$$

After performing these calculations in MATLAB Script 5, which can be found in Appendix A, we find the following values as seen below in the table.

Table 5: Final Design Mass Calculations

Parts	Mass (kg)
Payload (Martian soil)	16
Avionics Bay	4.5
RCS Thrusters	4.46
Hydrazine Propellant	10.9
Hydrazine Tank	3.22051
Star 12GV Motor	8.981129
Star 12GV ALITEC Propellant	15.331

Star 17A Motor	13.38
Star 17A ALITEC Propellant	128.1140
Total Initial Mass	204.8867

The final design also had many qualitative features that were optimized in the final design. The final design of the fairings was inspired by the Rocket Lab Electron Rocket. The Electron's mission is to supply small payloads into Earth's orbit in a small, cost effective rocket (Rocket Lab, 2020). As this has a similar mission to the MAV, the general shape of the fairings was inspired by the design shown in Figure 24. However, as seen in Figure 25, the design for the MAV was optimized by creating a cargo hold that fit the cargo container exactly. This was important as it allows the cargo to be secure after it has been autonomously loaded.

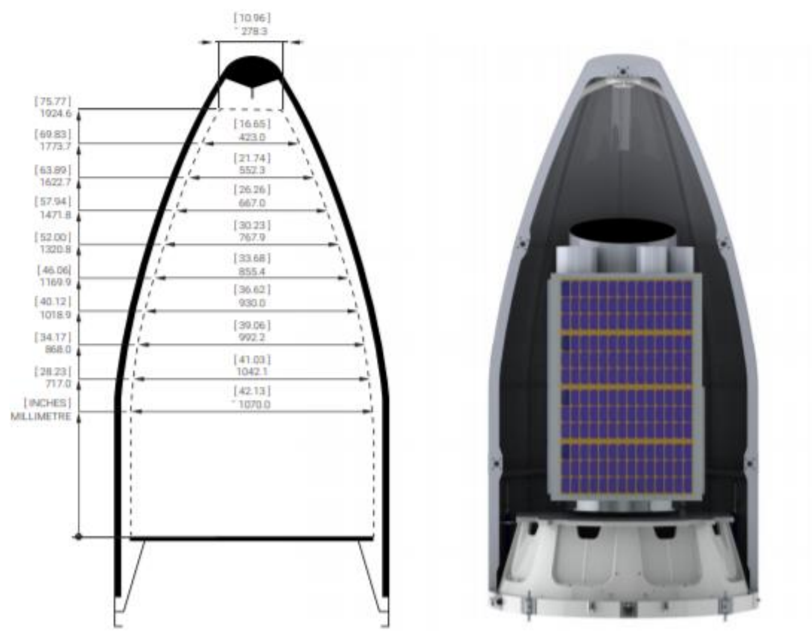


Figure 24: Rocket Lab Electron Rocket Fairing (Rocket Lab, 2020)

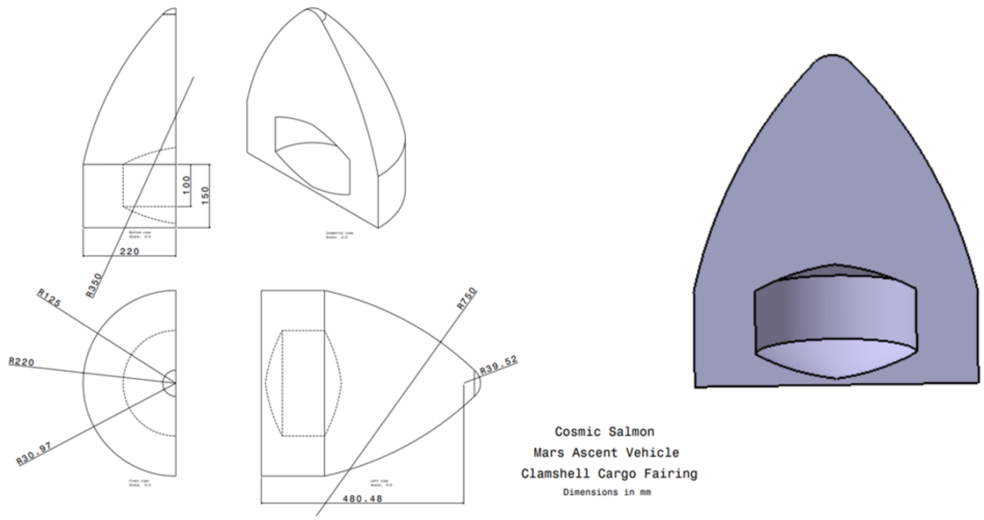


Figure 25: Generated Design for the MAV Payload Storage System (Appendix B Figure 37)

The propulsion systems were also designed for the MAV in the same dimensions as are listed by Northrop Grumman in their Propulsion Catalog (www.northropgrumman.com, 2016). The Star 12GV and Star 17A motors presented in Figures 26 and 27 demonstrate the detail that the team put into accurately defining and optimizing the model before supplying it to the clients.

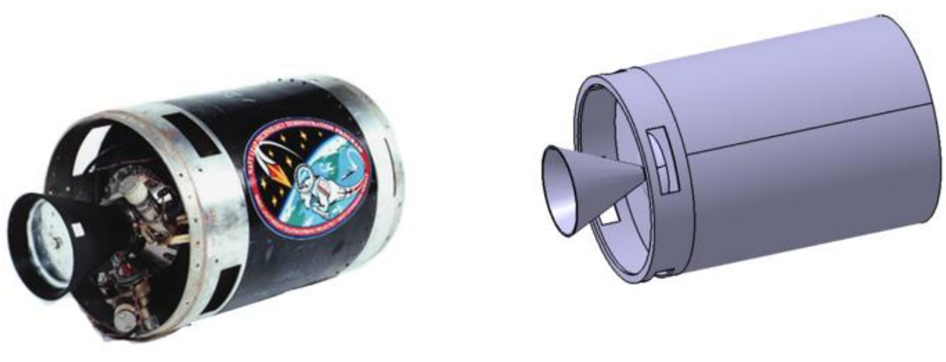


Figure 26: NGC Star 12GV Motor and CAD Representation (www.northropgrumman.com, 2016)

Figure 27: NGC Star 17A Motor and CAD Representation (www.northropgrumman.com, 2016)

Shown below in Figure 28 is our CAD model of the Aerojet Rocketdyne MRM-106F RCS thruster. On our final MAV design there will be two facing away from each other with an external hydrazine fuel tank. While this CAD has a few assumptions and isn't an exact replica of the product itself, this demonstrates the thought the team went to perfect our final design since RCS thrusters were not a requirement. We felt that the thrusters were an important addition since they contribute to the overall velocity of the MAV and provide increased stability.

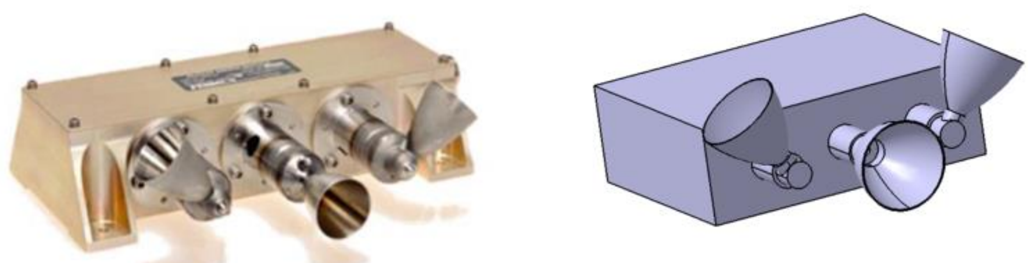


Figure 28: Aerojet Rocketdyne MRM-106F RCS Thrusters and CAD Representation (In-Space Propulsion Datasheets, n.d.)

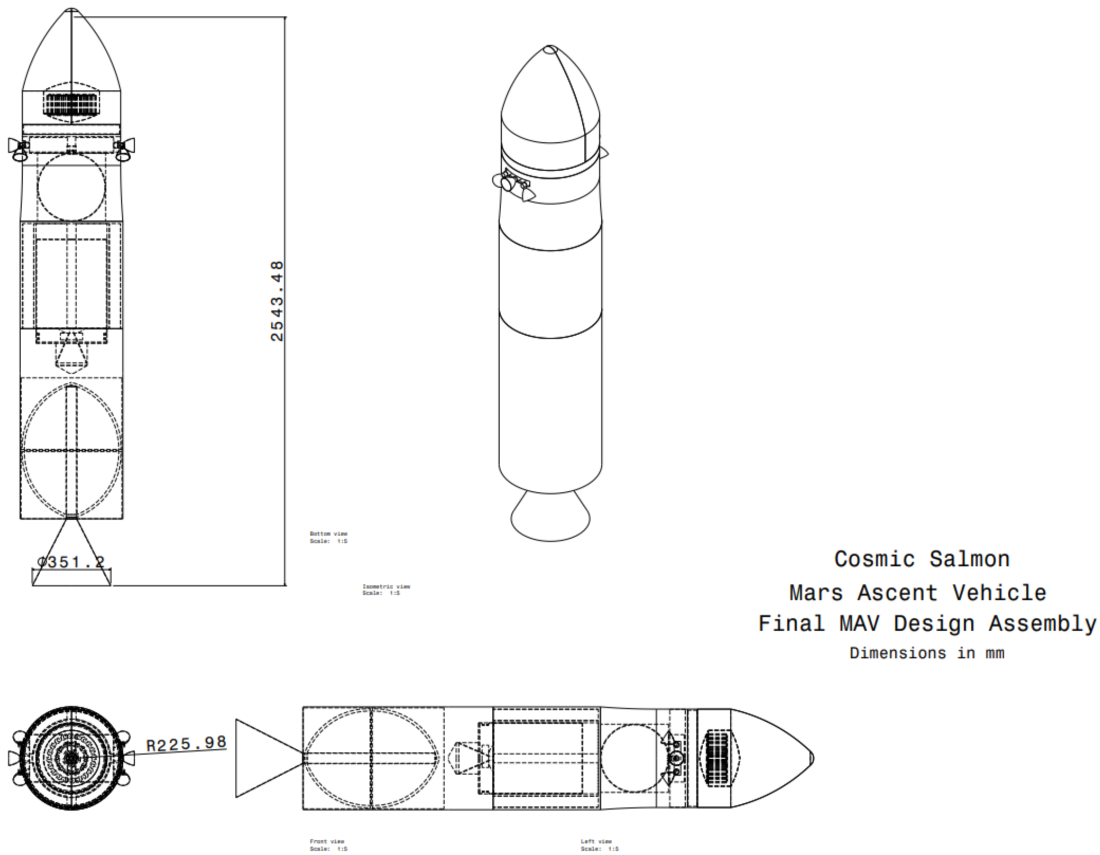


Figure 29: Final Dimensioned MAV Assembly (Appendix B Figure 44)

Our final design for the MAV was less than all the maximum dimensions defined in Section 2.2. In terms of sizing, the MAV has a height of 2.54348 meters, a width of 4.5 meters, and has a final mass of 204.89 kilograms. The major components of the MAV include the first stage STAR 12GV and second stage STAR 17A solid motors, followed by the MRM-106F thrusters with their corresponding external hydrazine fuel tank, finished off at the top with the avionics bay and the payload fairing. With all the requirements given by NASA, the MAV can successfully rendezvous with the Earth Return Orbiter and bring the Martian soil payload samples back to Earth safe and efficiently. Shown below in Figure 30 is our 3D CAD model of the final MAV design with dimensioned parts featured in Appendix B.

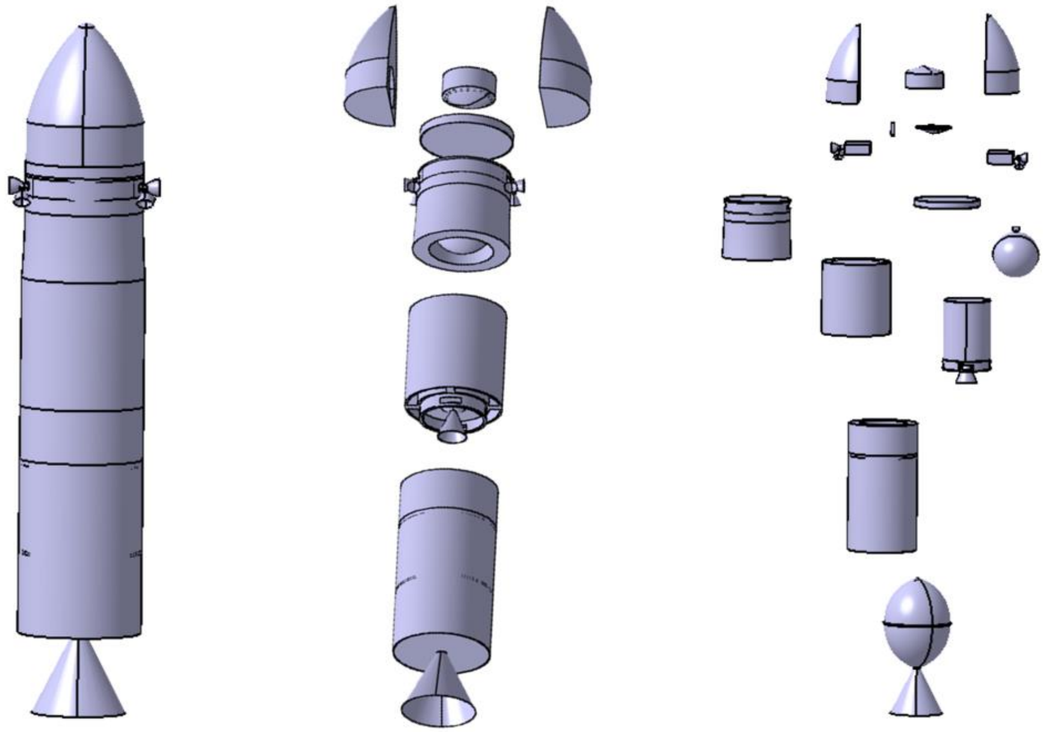


Figure 30: 3D View of MAV Design as Assembled, Major Components, Parts

5 Conclusions

While the design of the Mars Ascent Vehicle presented in this report meets all the specifications provided by the client, there are still many assumptions and neglected aspects that could call the design's feasibility into question. This analysis proves that the mission of the MAV with our design may be possible, but there are multiple necessary next steps required to fully confirm the feasibility of this design. Through this project, we were able to learn about the process of evaluating stakeholder's needs, selecting propulsion systems, and designing a rocket.

5.1 DESIGN EVALUATION

The team developed a design that met all the client requirements. However, there are still uncertainties about the feasibility of our design. The team believes that the risks can be fully mitigated through our listed mitigations and the MAV design, theoretically, would not be largely affected by any of the risks listed. Additionally, the calculations for the design show that it can, in theory, obtain the necessary delta-v. This is also important as the model does not even consider the additional delta-v source that the thrusters can provide beyond the guidance corrections. This additional source provides the team with confidence that the design can obtain the required delta-v in a real launch. However, our model does make a few assumptions that can greatly affect the design's feasibility. The first major assumption is that the mass of the structure is neglected in the calculations. While the rocket would use lightweight metals and composites, the added mass could greatly increase the amount of propellant required for this mission. This could make the MAV design exceed the specified mass from our clients. Another assumption is that the MAV would be able to follow the correct trajectory to orbit. This assumption would be a large concern for the design's feasibility as deviating from the trajectory would require more delta-v or result in a failure to rendezvous with the ERO. However, as the design has an avionics bay that controls both RCS thrusters from Aerojet Rocketdyne and a Northrop Grumman motor with thrust vector control, the team is confident that this assumption is relatively reasonable and further proves the design concept. Finally, careful considerations with respect to the operational temperatures of the propellants and propulsion systems provides the team with hope that the fuels would operate on Mars. The team did not obtain a full range of operational temperatures for the ALITEC fuel which does danger the feasibility of the design. However, we feel that installing insulation into the MAV and heaters into the MAV's storage unit on Mars would allow for the fuel to work at the known values.

In summary, our design is believed to satisfy the mission criteria despite making some assumptions. The team acknowledges that if an assumption was incorrectly addressed, the design may have minor flaws that could affect the mission outcome. However, we believe that even in that scenario, the clients would only have to make minor adjustments in the design to keep it operational.

5.2 Next Steps

The next step in this project would be to undergo a design review by experienced members of the aerospace field. We would present our design and collect feedback from our reviewers. From this, we would hope to learn what design flaws we overlooked. From there, the team would make the necessary changes to the concept and validate this new design meets our previous requirements.

Following this, our next steps would involve obtaining more detailed information about each system, then testing and integrating the proposed systems. For example, the team attempted to contact Adranos about the performance and storability of the ALITEC fuel but did not receive a response. Moving forward with this project, the team would continue to establish contact with the manufacturers and suppliers of the propellants, propulsion systems, mechanical systems, electrical systems, fairings, and insulation to gather more precise values. Additionally, the computer aided design model would be altered to allow for computer simulations, such as computational fluid dynamics simulations to test the performance of the fluid systems. Furthermore, computer simulations would allow for a more precise analysis of the drag forces that the new design experiences as it has a different aerodynamic shape than the design that generated the values given to the team by our clients. Finally, an advanced simulation could provide a model to test the design to see how high-probability risks could impact the mission.

Following these advanced simulation methods, the team would contact the clients and stakeholders to evaluate how the design meets their needs and adjust the design accordingly. If the design still meets the requirements and stakeholder needs, the team would, again, host a design review. From this, we hope to learn if there were issues with our testing, if more testing is required, and if there are more necessary design changes based upon our test results. If the concept is approved by our design reviewers and professors, the team would send our final proposed design to NASA and ESA. This will include a detailed report, model, and presentation of our design.

5.3 LESSONS LEARNT

Through generating the design concept for the Mars Ascent Vehicle, the team gained very valuable technical and teamwork experience and knowledge.

The first major area of growth for the team involves the design of rocket systems. Through learning how to calculate delta-v using the ideal rocket equation, and calculating the initial mass of the rocket, the team grew in our understanding of how all the systems are co-dependent. We learned that mass is an important aspect of rocket design and how choosing propellants with different specific impulses and inert mass fractions affects the design. Additionally, the team learned important differences between solid, liquid, and hybrid propellants. We also discovered that the operating temperature of a propellant is vital to consider when used in conditions like those of Mars.

Beyond learning the process of selecting propulsion motors and designing a rocket, the team was very enthusiastic about the possibility of designing the MAV with parallel high-powered rocketry motors. We initially theorized that this design was a unique concept that would provide the clients with an alternative that extended beyond small changes to the propellant choice. However, the team ultimately learned that the technology developed by industry leaders in propulsion systems, such as Northrop Grumman, are superior to these proposed alternative designs. The current high-powered rocketry motors available have a high inert mass fraction and low specific impulse. Due to this, they were not a viable alternative. However, the team learned about both conventional design while exploring alternative routes which helped teach us to always be looking outside the box in a design.

Finally, from this project, the team learned a lot about teamwork and communication. As our team was so excited about the unique design of the high-powered rocketry motors in parallel, we developed a sense of group think during a period of the design process. Everyone was so concentrated on trying to make our new innovative design work that we did not always consider the better alternatives. This is evident, as the team made a full computer aided design assembly for both concept two and our final design. However, in the end, the team recognized our mistake and pursued the best possible design for our clients.

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APPENDIX A: MATLAB CODE

% Appendix A: MATLAB SCRIPT #1

```
% PM 7 Q 2
% Initial Mass v F1 Value

deltaVMO = sqrt((4.28*10^13)/3771500); %m/s
deltaVLoss = 201;
deltaVMH = 0.00007115*3389500*cosd(18.3628)*sind(90);
deltaV = deltaVMO + deltaVLoss - deltaVMH;

rocketStagePlot = 1:2;
f1 = 0:.01:1;
f2 = 1.0 - f1;

conc2stage2 = 12;
conc3stage1 = 11;
conc3stage2 = 11;

inertMassFrac1 = 0.119; % star 17A
inertMassFrac2 = 0.2138; % star 12GV

specificImpulse1 = 270.7;
specificImpulse2 = 270.7;

c2inertMassFrac1 = 0.351;
c2inertMassFrac2 = 0.2138; % star 12GV

c2specificImpulse1 = 116.1;
c2specificImpulse2 = 270.7;

c3inertMassFrac1 = 0.351;
c3inertMassFrac2 = 0.351;

c3specificImpulse1 = 116.1;
c3specificImpulse2 = 116.1;

initialMassCalc1 = initialMassCalc(inertMassFrac1,inertMassFrac2,
specificImpulse1, specificImpulse2,deltaV,f1,f2);
initialMassCalc2 =
initialMassCalc(c2inertMassFrac1,c2inertMassFrac2,c2specificImpulse1,c2speci
ficImpulse2,deltaV,f1,f2);
initialMassCalc3 =
initialMassCalc(c3inertMassFrac1,c3inertMassFrac2,c3specificImpulse1,c3speci
ficImpulse2,deltaV,f1,f2);

[minimumMass1, index1] = min(initialMassCalc1);
```

```
[minimumMass2, index2] = min(initialMassCalc2);
[minimumMass3, index3] = min(initialMassCalc3);

scatter(f1,initialMassCalc1);
hold on;
scatter(f1(index1),minimumMass1, 'r', 'filled');
hold on;
scatter(f1,initialMassCalc2);
hold on;
scatter(f1(index2),minimumMass2, 'b', 'filled');
hold on;
scatter(f1,initialMassCalc3);
hold on;
scatter(f1(index3),minimumMass3, 'g', 'filled');
hold off;
%ylim([0 400]);

%chart organization
title("Inital Mass of Rocket v F1 Value");
xlabel("F1 Value");
ylabel("Initial Rocket Mass (kg)");

function initialMass = initialMassCalc(mfrac1, mfrac2, Isp1,
Isp2,deltaV,f1,f2)
    gravityConstant = 9.81; %m/s^2
    payloadMass = 37;

    stage1eqn = ((exp((deltaV .* f1) ./(gravityConstant .* Isp1)).*(1-
mfrac1))...
    ./(1-(mfrac1 .* exp(((deltaV .* f1)) ./(gravityConstant .* Isp1)))));

    stage2eqn = ((exp((deltaV .* f2) ./(gravityConstant .* Isp2)).*(1-
mfrac2))...
    ./(1-(mfrac2 .* exp(((deltaV .* f2)) ./(gravityConstant .* Isp2)))));

    initialMass = payloadMass .* stage1eqn .* stage2eqn;
    %[minimumMass, index] = min(initialMass);
    %fprintf('Minimum initial rocket mass of %f kg at an f1 value of
%f',minimumMass,f1(index));

end
```

% Appendix A: MATLAB SCRIPT #2

```
%AAE 251 Fall 2020
%Final Report
%Commercial High-Powered Rocketry Motors Comparison
%Authors: Mark Paral

close all;
clear all;
clc

x = 6; %number of types of motors
mpay = 16; %estimated payload mass (kg)
mprop = [7.5657,7.925,7.6947,5.1471,5.5914,5.1609]; %propellant masses of 1
motor (kg)
```

```
minert = [6.1026,4.478,6.1147,4.7837,4.8812,4.8084]; %inert masses of 1
motor (kg)
Impulse = [16460.94,14126,14801.70,11452.32,10590.80,10280.73]; %impulse
values of 1 motor (Ns)
weight = zeros(x,1); %weight array (N)
Isp = zeros(x,1); %Isp array (s)
V2 = 1673; %DeltaV of stage 2 (m/s)
Vs = zeros(x,1); %DeltaV of 1 motor array (m/s)
Ns = zeros(x,1); %Number of motors required for DeltaV of stage 2 array
Stagem = zeros(x,1); %Total mass of stage 2 (kg)
Motor = categorical({'ANIMAL MOTOR WORKS N4000','AEROTECH N1000','ANIMAL
MOTOR WORKS N2800','ANIMAL MOTOR WORKS N2700','ANIMAL MOTOR WORKS
N2600','ANIMAL MOTOR WORKS N2020'}); %motor names

%calc weights, Isp, Velocities, number of motors, and stage masses
for i=1:1:x
    weight(i) = (mprop(i)+minert(i))*9.81;
    Isp(i) = Impulse(i)/weight(i);
    Vs(i) = 9.81*Isp(i)*log((mprop(i)+mpay+minert(i))/(mpay+minert(i)));
    Ns(i) = V2/Vs(i);
    Stagem(i) = Ns(i)*(mprop(i)+minert(i))+mpay;
end

%plot of masses
figure(1);
bar(Motor,Stagem);
title('Stage 2 Mass Compared');
xlabel('Motors');
ylabel('Mass (kg)');
grid on;

%plot of number of motors required
figure(2);
bar(Motor,Ns);
title('Number of Motors Required for Stage 2 DeltaV');
xlabel('Motors');
ylabel('Number of Motors Required');
grid on;

%plot of Isp
figure(3);
bar(Motor,Isp);
title('Isp of Each Motor Compared');
xlabel('Motors');
ylabel('Isp (s)');
grid on;
```

% Appendix A: MATLAB SCRIPT #3

```
%AAE 251 Fall 2020
%Final Report
%Conceptual Design Calculations
%Authors: Mark Paral

close all;
clear all;
```

```
clc

%overall design constraints
V1 = 1495;
V2 = 1673;
g0 = 9.81; %gravitational accel const (m/s^2)

%Aerotech N1000 Motor specifications
mpropN1000 = 7.925; %mass of propellant in 1 N1000 motor (kg)
minertN1000 = 4.478; %inert mass of 1 N1000 motor (kg)
ImpulseN1000 = 14126; %total impulse of 1 N1000 motor (Ns)
weightN1000 = 0; %weight of 1 N1000 motor (N)
IspN1000 = 0; %Isp of 1 N1000 motor (s)
%Aerotech N1000 Motor calculations
weightN1000 = (mpropN1000+minertN1000)*g0;
IspN1000= ImpulseN1000/weightN1000;

%design 1: First Stage Motor Star 17A, Second Stage Motor Star 12GV
mpay12 = 16; %mass of martian soil payload (kg)
mpay11 = 0; %mass of 1st stage payload (kg)
mprop11 = 0; %mass of stage 1 prop (kg)
mprop12 = 0; %mass of stage 2 prop (kg)
minert11 = 13.38; %mass of stage 1 inert (kg)
minert12 = 9.026488; %mass of stage 2 inert (kg)
Isp11 = 270.7; %Isp of stage 1 (s)
Isp12 = 270.7; %Isp of stage 2 (s)
Stagem11 = 0; %stage 1 total mass (kg)
Stagem12 = 0; %stage 2 total mass (kg)
%design 1: calculations
mprop12 = exp(V2/(g0*Isp12))*(mpay12+minert12)-mpay12-minert12;
mpay11 = mpay12+mprop12+minert12;
mprop11 = exp(V1/(g0*Isp11))*(mpay11+minert11)-mpay11-minert11;
Stagem11 = mprop11+minert11;
Stagem12 = mprop12+minert12;
%design 1: results
display(mprop11);
display(mprop12);
display(Stagem11);
display(Stagem12);

%design 2: First Stage AEROTECH N1000 Commercial High-Powered Rocketry
Motors, Second Stage Motor Star 12GV
mpay22 = 16; %mass of martian soil payload (kg)
mpay21 = 0; %mass of 1st stage payload (kg)
mprop21 = 0; %mass of stage 1 prop (kg)
mprop22 = 0; %mass of stage 2 prop (kg)
minert21 = 0; %mass of stage 1 inert (kg)
minert22 = 9.026488; %mass of stage 2 inert (kg)
Isp21 = IspN1000; %Isp of stage 1 (s)
Isp22 = 270.7; %Isp of stage 2 (s)
Vs21 = 0; %velocity of 1 motor of stage 1 (m/s)
Ns21 = 0; %number of required motors for stage 1
Stagem21 = 0; %stage 1 total mass (kg)
Stagem22 = 0; %stage 2 total mass (kg)
%design 2: calculations
mprop22 = exp(V2/(g0*Isp22))*(mpay22+minert22)-mpay22-minert22;
mpay21 = mpay22+mprop22+minert22;
```

```
Vs21 = g0*Isp21*log((mpropN1000+mpay21+minertN1000)/(mpay21+minertN1000));
Ns21 = V1/Vs21;
mprop21 = Ns21*mpropN1000;
minert21 = Ns21*minertN1000;
Stagem21 = (mprop21+minert21);
Stagem22 = mprop22+minert22;
%design 2: results
display(mprop21);
display(mprop22);
display(Stagem21);
display(Stagem22);
display(Ns21);
display(minert21);

%design 3: First Stage AEROTECH N1000 Commercial High-Powered Rocketry
Motors, Second Stage AEROTECH N1000 Commercial High-Powered Rocketry Motors
mpay32 = 16; %mass of martian soil payload (kg)
mpay31 = 0; %mass of 1st stage payload (kg)
mprop31 = 0; %mass of stage 1 prop (kg)
mprop32 = 0; %mass of stage 2 prop (kg)
minert31 = 0; %mass of stage 1 inert (kg)
minert32 = 0; %mass of stage 2 inert (kg)
Isp31 = IspN1000; %Isp of stage 1 (s)
Isp32 = IspN1000; %Isp of stage 2 (s)
Vs31 = 0; %velocity of 1 motor of stage 1 (m/s)
Vs32 = 0; %velocity of 1 motor of stage 2 (m/s)
Ns31 = 0; %number of required motors for stage 1
Ns32 = 0; %number of required motors for stage 2
Stagem31 = 0; %stage 1 total mass (kg)
Stagem32 = 0; %stage 2 total mass (kg)
%design 3: calculations
Vs32 = g0*Isp32*log((mpropN1000+mpay32+minertN1000)/(mpay32+minertN1000));
Ns32 = V2/Vs32;
mprop32 = Ns32*mpropN1000;
mpay31 = mpay32+mprop32+minert32;
Vs31 = g0*Isp31*log((mpropN1000+mpay31+minertN1000)/(mpay31+minertN1000));
Ns31 = V1/Vs31;
mprop31 = Ns31*mpropN1000;
minert31 = Ns31*minertN1000;
minert32 = Ns32*minertN1000;
Stagem31 = (mprop31+minert31);
Stagem32 = (mprop32+minert32);
%design 3: results
display(mprop31);
display(mprop32);
display(Stagem31);
display(Stagem32);
display(Ns31);
display(Ns32);
display(minert31);
display(minert32);
```

% Appendix A: MATLAB SCRIPT #4

```
%Initial mass varying based on amount of stages
%Authors: Ryan Horvath
%% ____________________
```

```
%% INITIALIZATION
Pl = 37; %payload mass constant
Fi1 = .2138;  %inert mass fraction
Fi2 = .119;
Fi3 = .351;
Isp1 = 270.7; %specific impulse
Isp2 = 270.7;
Isp3 = 116.1;
g = 9.81;   %gravity constant
Vr = 0.00007115*3389500*cosd(18.3628)*sind(90); %speed of rotation
Vl = 201; %velocity lost
GM = 3.986 * 10^5; %GM constant
stages = linspace(1,6,6);    %gets all the stages in a vector
Im1 = zeros(1,6);             %vector to store initial mass in each stage
Im2 = zeros(1,6);
Im3 = zeros(1,6);
%% ____________________
%% CALCULATIONS
C1 = Isp1 * g;      %gets the exhaust velocity
C2 = Isp2 * g;
C3 = Isp3 * g;
V = sqrt((4.28*10^13)/3771500) + Vl - Vr;   %change in velocity
k = 1;              %runner variable
while k < 7         %while going through the possible amount of stages
    Im1(1,k) = Pl * ((exp(V / (k * C1)) * (1 - Fi1)) / (1 - Fi1 * exp(V / (k
* C1))))^k;
    k = k + 1;     %updates runner variable
end
r = 1;
while r < 7         %while going through the possible amount of stages
    Im2(1,r) = Pl * ((exp(V / (r * C2)) * (1 - Fi2)) / (1 - Fi2 * exp(V / (r
* C2))))^r;
    r = r + 1;     %updates runner variable
end
j = 1;
while j < 7         %while going through the possible amount of stages
    Im3(1,j) = Pl * ((exp(V / (j * C3)) * (1 - Fi3)) / (1 - Fi3 * exp(V / (j
* C3))))^j;
    j = j + 1;     %updates runner variable
end
%% ____________________
%% OUTPUTS
plot(stages,Im1(1,:),'r');        %plots initial mass versus amount of
stages
grid on
xlabel("Number of Stages");  %labels x axis
ylabel("Initial Mass (kg)"); %labels y axis
title("Initial Mass of MAV v Number of Stages");  %titles the  plot

figure
plot(stages,Im2(1,:),'b');
grid on
xlabel("Number of Stages");  %labels x axis
ylabel("Initial Mass (kg)"); %labels y axis
title("Initial Mass of MAV v Number of Stages");  %titles the  plot

figure
```

```
plot(stages,Im3(1,:),'g');
grid on
xlabel("Number of Stages");  %labels x axis
ylabel("Initial Mass (kg)"); %labels y axis
title("Initial Mass of MAV v Number of Stages");  %titles the  plot
```

% Appendix A: MATLAB SCRIPT #5

```
%AAE 251 Fall 2020
%Final Report
%Concept Refinement
%Authors: Mark Paral

close all;
clear all;
clc

%Variables
g0 = 9.81; %grav accel const (m/s^2)
mtot = 0; %total rocket initial mass (kg)
mpay = 16; %mass of Martian rocks (kg)
mpay1 = 0; %total payload mass of stage 1 (kg)
mpay2 = 0; %total payload mass of stage 2 (kg)
mav = 4.5; %mass of avionics bay (kg)
minert1 = 13.38; %mass of stage 1 inert (kg)
minert2 = 8.981129; %mass of stage 2 inert (kg)
Isp1 = 270.7; %Isp of stage 2 (s)
Isp2 = 270.7; %Isp of stage 2 (s)
minertRCS = 4.46; %inert mass of RCS motors (kg)
mpropRCS = 10.9; %mass of RCS propellant (kg)
mtankRCS = 3.22051; %mass of RCS propellant tank (kg)
V1 = 2606.7268; %estimated deltaV of stage 1 (m/s)
V2 = 735.23066; %estimated deltaV of stage 2 (m/s)
mprop1 = 0; %mass of propellant stage 1 (kg)
mprop2 = 0; %mass of propellant stage 2 (kg)
finert1 = 0.1065; %finert of stage 1
finert2 = 0.2151; %finert of stage 2

%stage 2 calc
mpay2 = mpay+minertRCS+mpropRCS+mtankRCS+mav;
mprop2 = exp(V2/(g0*Isp2))*(minert2+mpay2)-minert2-mpay2;
%mprop2 = (mpay2*(exp(V2/(g0*Isp2))-1)*(1-finert2))/(1-
finert2*exp(V2/(g0*Isp2)));

%stage 1 calc
mpay1 = mpay2+minert2+mprop2;
mprop1 = exp(V1/(g0*Isp1))*(minert1+mpay1)-minert1-mpay1;
%mprop1 = (mpay1*(exp(V1/(g0*Isp1))-1)*(1-finert1))/(1-
finert1*exp(V1/(g0*Isp1)));

%total rocket
mtot = mpay1+mprop1+minert1;

%display results
display(mpay2);
display(mprop2);
```

```
display(mpay1);
display(mprop1);
display(mtot);
```

% Appendix A: MATLAB SCRIPT #6

```
%Concept 1
%Delta V Values

deltaVMO = sqrt((4.28*10^13)/3771500); %m/s

deltaVLoss = 201; %m/s

deltaVMH = 0.00007115*3389500*cosd(18.3628)*sind(90); %m/s

deltaV = deltaVMO + deltaVLoss - deltaVMH; %m/s

%Rocket staging

rocketStagePlot = 1:2;

gravityConstant = 9.81; %m/s^2

payloadMass = 37; %kg

f1 = 0:.01:1;

f2 = 1.0 - f1;


inertMassFrac1 = 0.106;

inertMassFrac2 = 0.214;


specificImpulse1 = 270.7;

specificImpulse2 = 270.7;


stage1eqn = ((exp((deltaV .* f1) ./(gravityConstant .*
specificImpulse1)).*(1-inertMassFrac1))./(1-(inertMassFrac1 .* exp(((deltaV
.* f1)) ./(gravityConstant .* specificImpulse1)))));


stage2eqn = ((exp((deltaV .* f2) ./(gravityConstant .*
specificImpulse2)).*(1-inertMassFrac2))./(1-(inertMassFrac2 .* exp(((deltaV
.* f2)) ./(gravityConstant .* specificImpulse2)))));


initialMass = payloadMass .* stage1eqn .* stage2eqn;
```

```
[minimumMass, index] = min(initialMass);

fprintf('Minimum initial rocket mass of %f kg at an f1 value of
%f.\n',minimumMass,f1(index));



scatter(f1,initialMass);

hold on;
grid on;
scatter(f1(index),minimumMass, 'r', 'filled');

hold off;



%chart organization

title("Initial Mass of Rocket v F1 Value");

xlabel("F1 Value");

ylabel("Initial Rocket Mass (kg)");
```

% Appendix A: MATLAB SCRIPT #7

```
%Concept 2
%Delta V Values

deltaVMO = sqrt((4.28*10^13)/3771500); %m/s

deltaVLoss = 201; %m/s

deltaVMH = 0.00007115*3389500*cosd(18.3628)*sind(90); %m/s

deltaV = deltaVMO + deltaVLoss - deltaVMH; %m/s

%Rocket staging

rocketStagePlot = 1:2;

gravityConstant = 9.81; %m/s^2

payloadMass = 37; %kg

f1 = 0:.01:1;

f2 = 1.0 - f1;



inertMassFrac1 = 0.361;

inertMassFrac2 = 0.214;
```

```
specificImpulse1 = 116.1;

specificImpulse2 = 270.7;



stage1eqn = ((exp((deltaV .* f1) ./(gravityConstant .* specificImpulse1)).*(1-inertMassFrac1))./(1-(inertMassFrac1 .* exp(((deltaV .* f1)) ./(gravityConstant .* specificImpulse1)))));



stage2eqn = ((exp((deltaV .* f2) ./(gravityConstant .* specificImpulse2)).*(1-inertMassFrac2))./(1-(inertMassFrac2 .* exp(((deltaV .* f2)) ./(gravityConstant .* specificImpulse2)))));



initialMass = payloadMass .* stage1eqn .* stage2eqn;

[minimumMass, index] = min(initialMass);

fprintf('Minimum initial rocket mass of %f kg at an f1 value of %f.\n',minimumMass,f1(index));



scatter(f1,initialMass);

hold on;
grid on;


hold off;



%chart organization

title("Initial Mass of Rocket v F1 Value");

xlabel("F1 Value");

ylabel("Initial Rocket Mass (kg)");
```

% Appendix A: MATLAB SCRIPT #8

```
%Concept 3
%Delta V Values

deltaVMO = sqrt((4.28*10^13)/3771500); %m/s

deltaVLoss = 201; %m/s

deltaVMH = 0.00007115*3389500*cosd(18.3628)*sind(90); %m/s
```

```
deltaV = deltaVMO + deltaVLoss - deltaVMH; %m/s

%Rocket staging

rocketStagePlot = 1:2;

gravityConstant = 9.81; %m/s^2

payloadMass = 37; %kg

f1 = 0:.01:1;

f2 = 1.0 - f1;



inertMassFrac1 = 0.3610413362;

inertMassFrac2 = 0.3610417314;



specificImpulse1 = 116.1;

specificImpulse2 = 116.1;



stage1eqn = ((exp((deltaV .* f1) ./(gravityConstant .*
specificImpulse1)).*(1-inertMassFrac1))./(1-(inertMassFrac1 .* exp(((deltaV
.* f1)) ./(gravityConstant .* specificImpulse1)))));



stage2eqn = ((exp((deltaV .* f2) ./(gravityConstant .*
specificImpulse2)).*(1-inertMassFrac2))./(1-(inertMassFrac2 .* exp(((deltaV
.* f2)) ./(gravityConstant .* specificImpulse2)))));



initialMass = payloadMass .* stage1eqn .* stage2eqn;

[minimumMass, index] = min(initialMass);

fprintf('Minimum initial rocket mass of %f kg at an f1 value of
%f.\n',minimumMass,f1(index));

scatter(f1,initialMass);

hold on;
grid on;


hold off;

%chart organization
```

```
title("Initial Mass of Rocket v F1 Value");

xlabel("F1 Value");

ylabel("Initial Rocket Mass (kg)");
```

APPENDIX B: DIMENSIONED CAD DRAWINGS

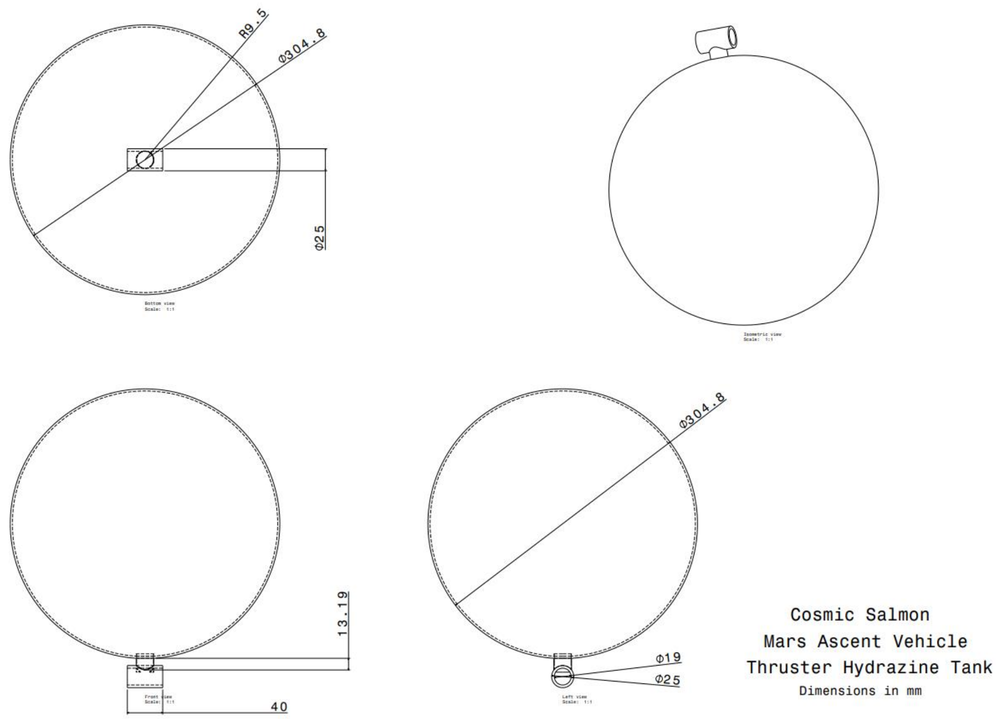

Figure 31: Diaphragm Hydrazine Storage Tank Drawing

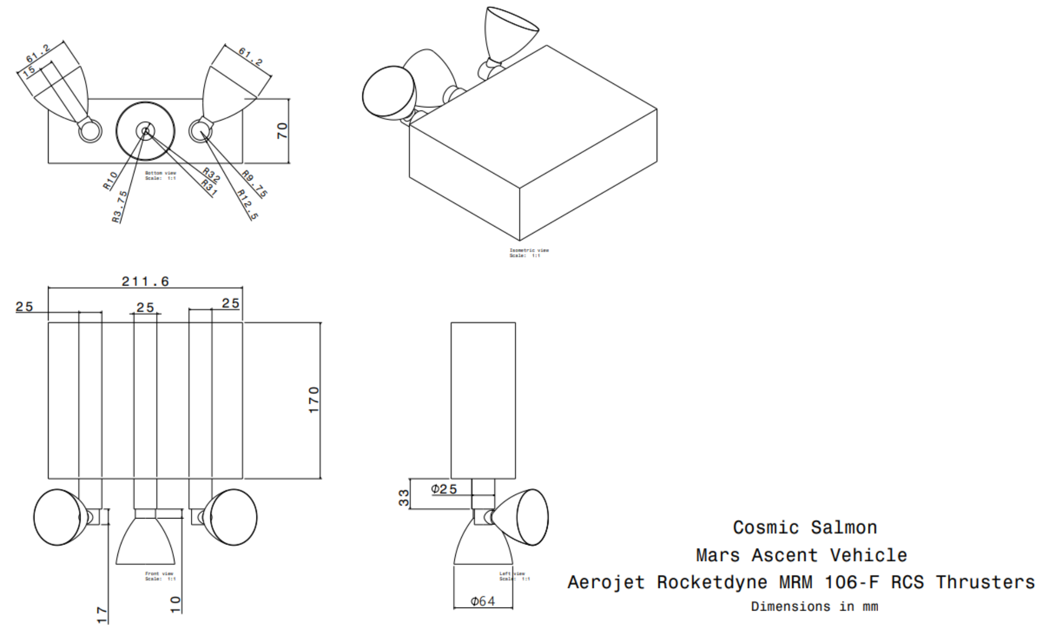

Figure 32: Aerojet Rocketdyne MRM 106-F RCS Thrusters Drawing

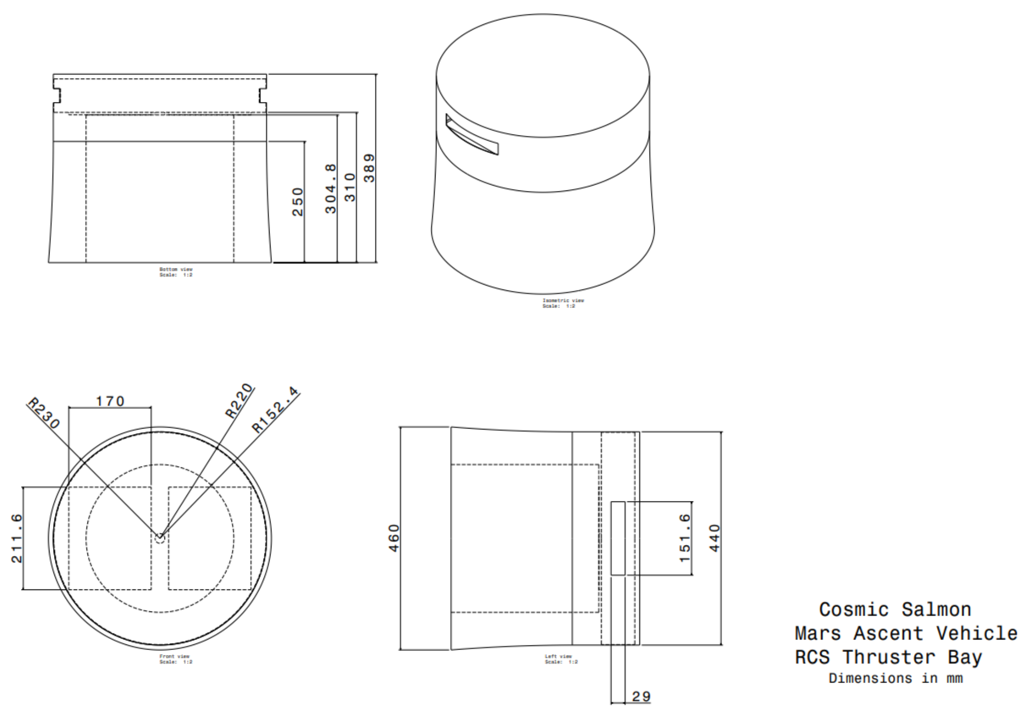

Figure 33: RCS Thruster Bay Drawing

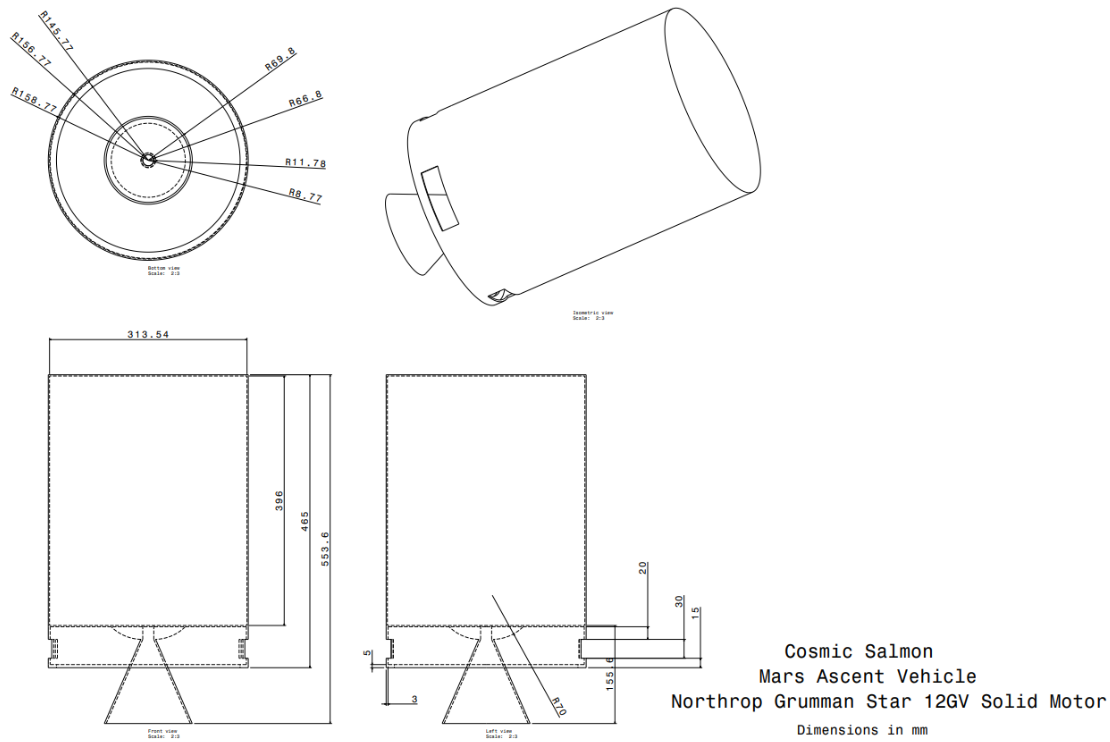

Figure 34: Northrop Grumman Star 12GV Solid Motor Drawing

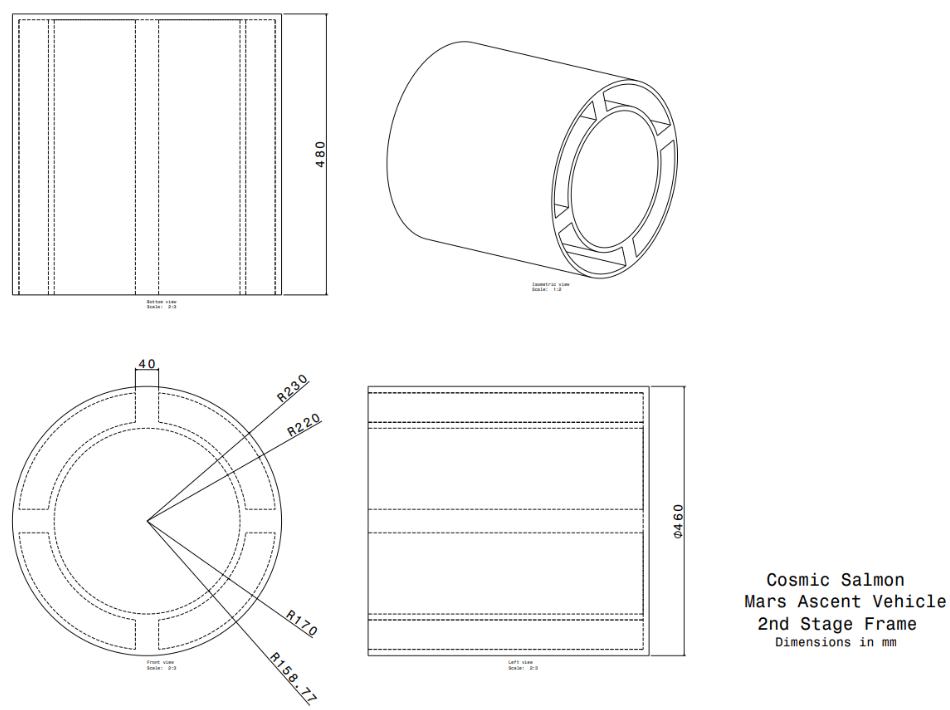

Figure 35: Second Stage Frame with Room for Insulation Drawing

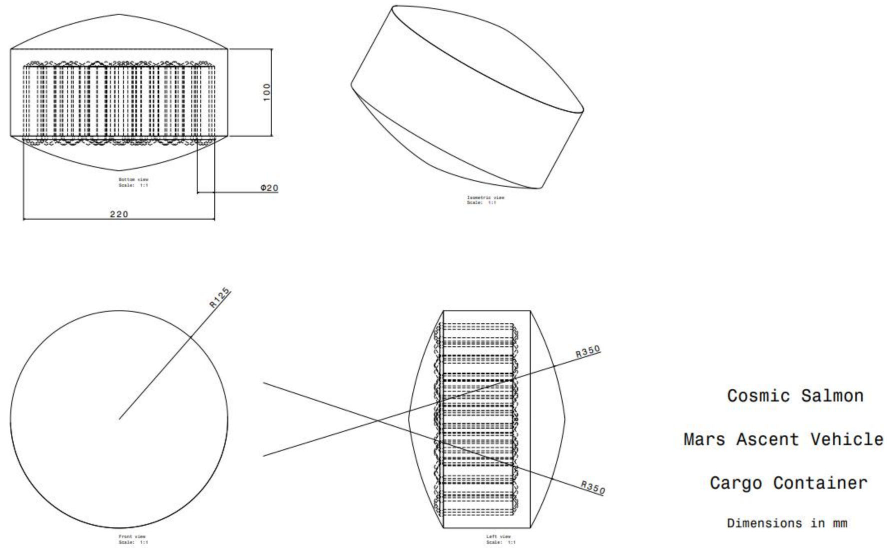

Figure 36: Cargo Container Drawing

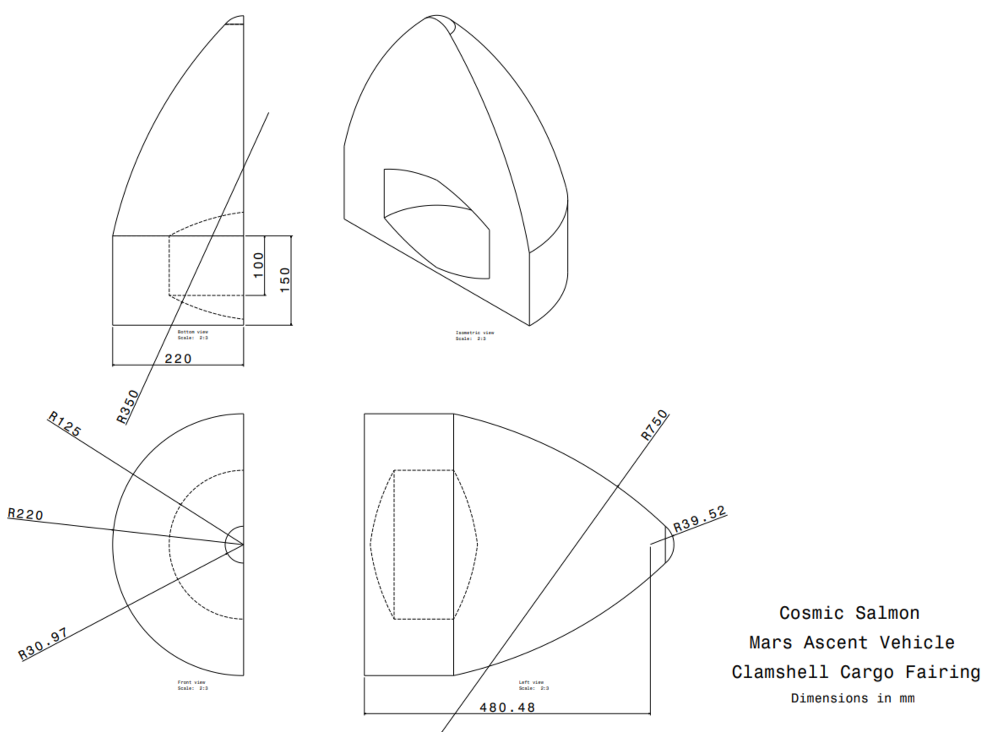

Figure 37: Clamshell Cargo Fairing Drawing

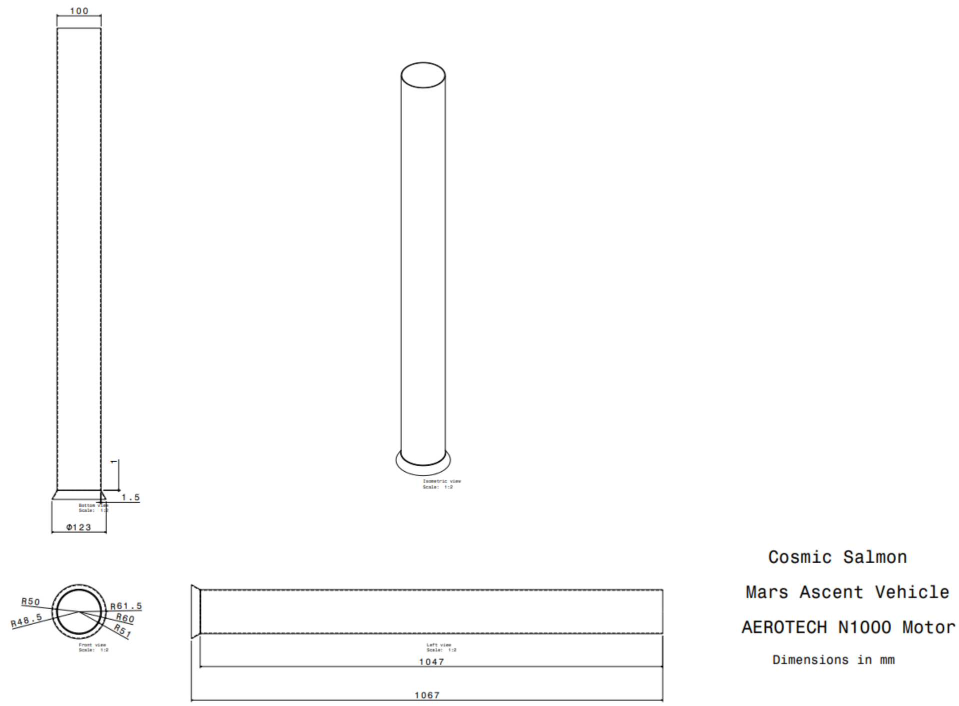

Figure 38: AEROTECH N1000 High Powered Rocketry Motors Drawing

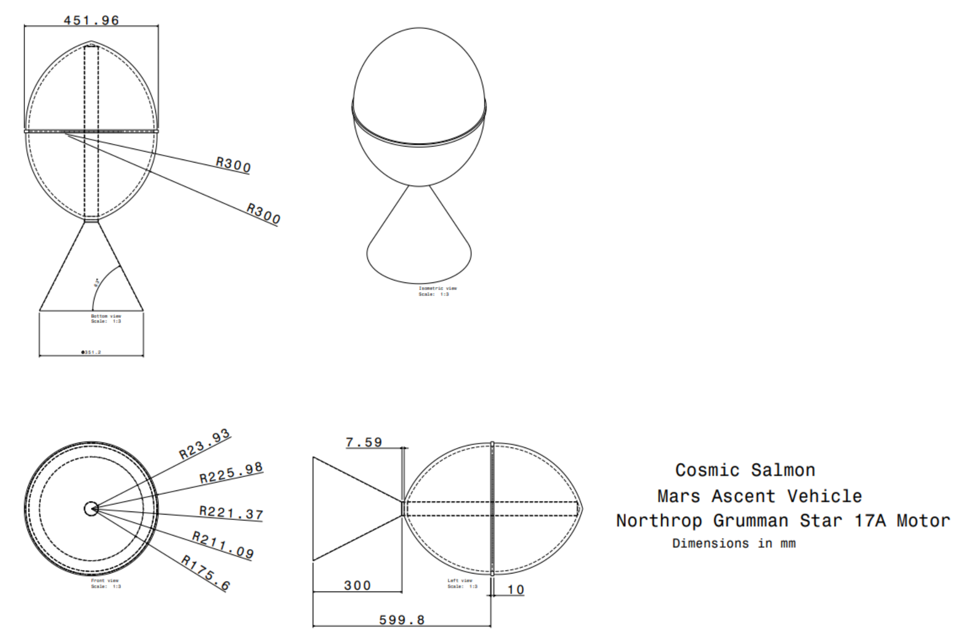

Figure 39: Northrop Grumman Star 17A Motor Drawing

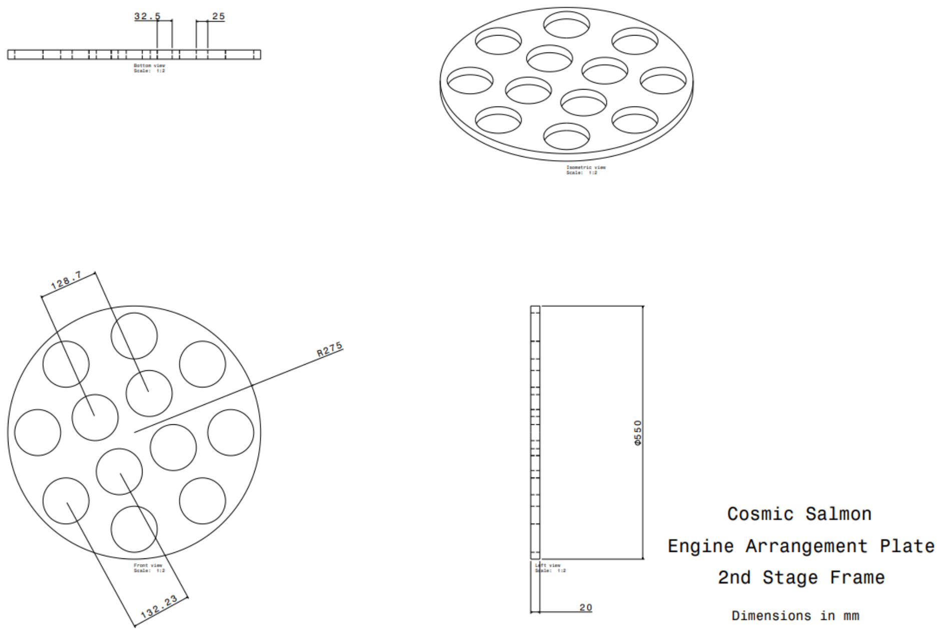

Figure 40: Concept 2 Parallel Motor Arrangement Plate Drawing

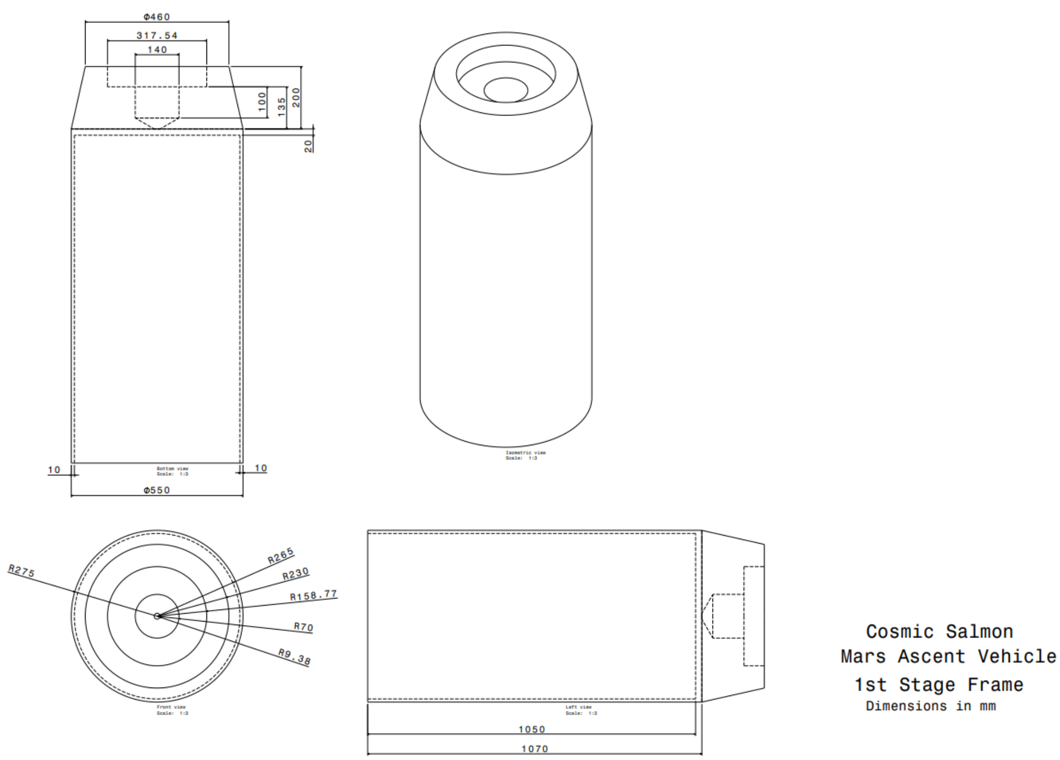

Figure 41: 1st Stage Frame Drawing for Concept 2

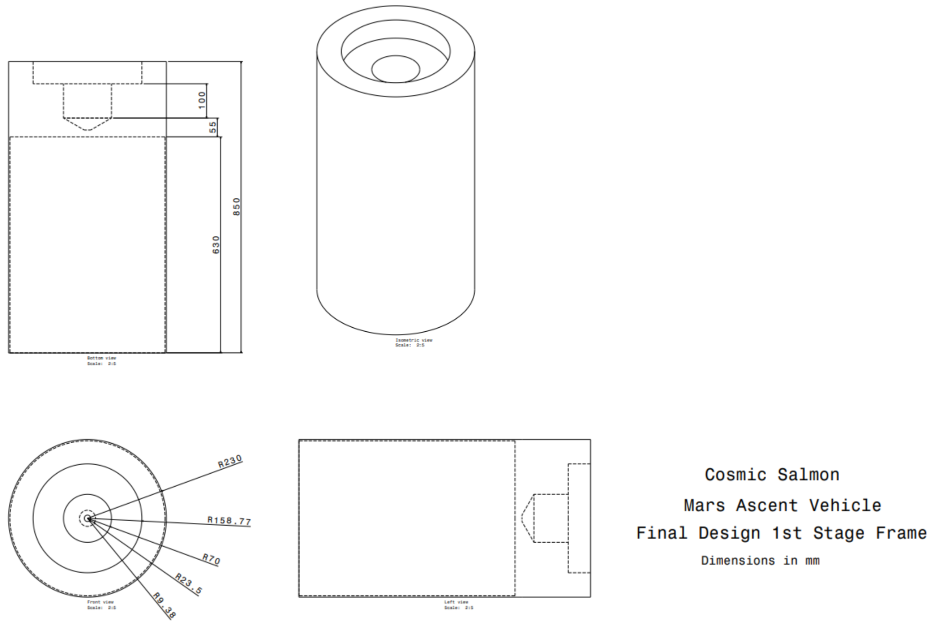

Figure 42: Final Design First Stage Frame Drawing

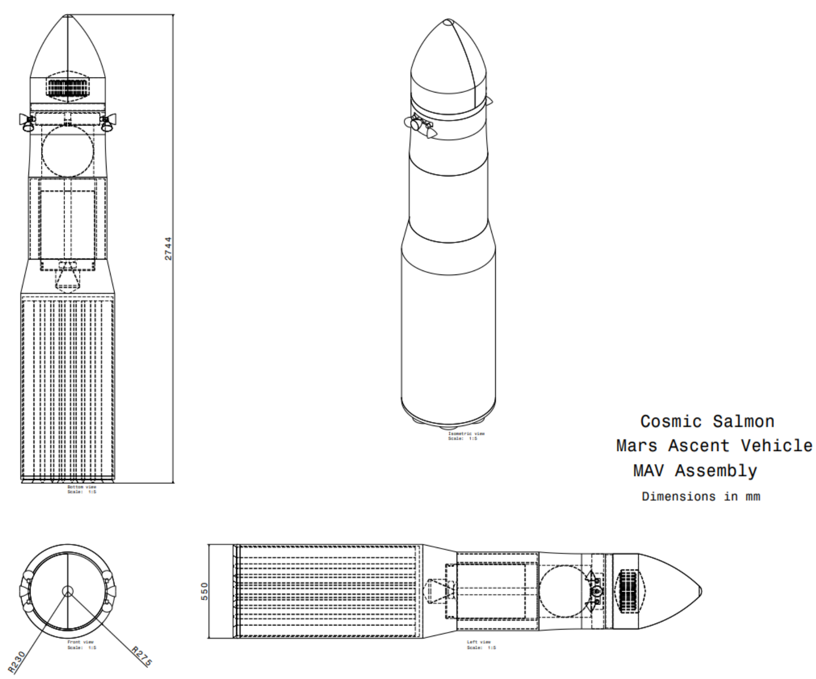

Figure 43: Dimensioned Drawing of Concept 2 Assembly

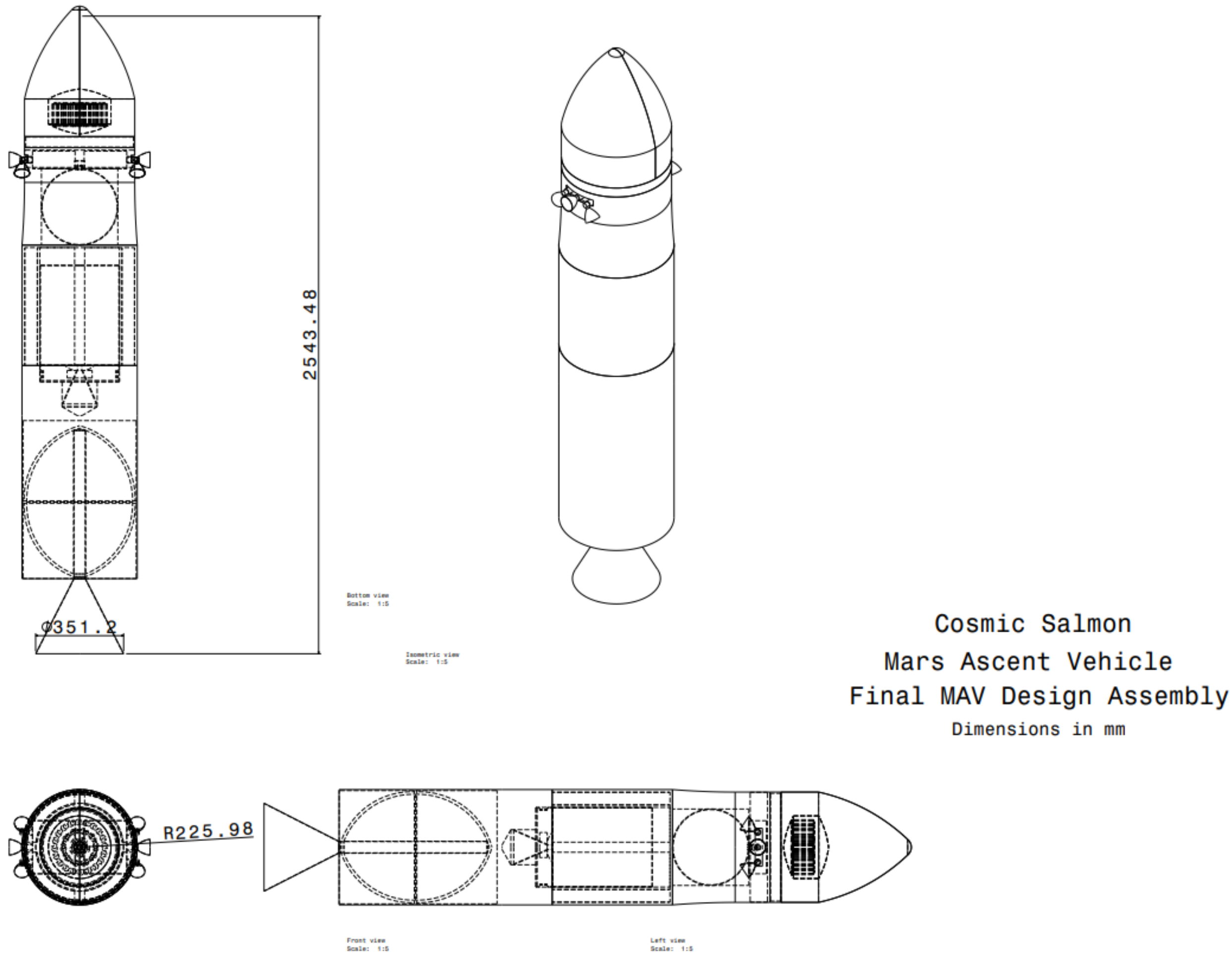

Figure 44: Final MAV Concept Dimensioned Drawing