

Project Argo

A Proposal for a Mars Sample Return System

J. Cunningham, S. Thompson, C. Ayon, W. Gallagher, E. Gomez, A. Kechejian, A. Miller

Pyxis Aerospace, Cal Poly Pomona, Pomona, CA, 91768, USA

AIAA Undergraduate Team Space Transportation Design Competition, 2016

Abstract

Pyxis Aerospace at Cal Poly Pomona is pleased to present *Project Argo*, its proposal for a Mars Sample Return System as requested by the American Institute of Aeronautics and Astronautics.

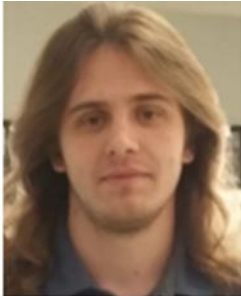
The goal of Project Argo is to travel to Mars, retrieve a sample collected by a rover on the Martian surface, and return it to Earth. To accomplish this goal, we have designed three vehicles: the Earth Return Vehicle (ERV), the Mars Ascent Vehicle (MAV), and the Mars Landing Platform (MLP). The three vehicles will launch on July 24, 2020 and travel together as one unit. The MAV and the MLP will be enclosed in a backshell/heat shield similar to previous Mars missions, and the ERV will act as the cruise stage. The vehicles will arrive at Mars Feb 17, 2021. After Mars orbit capture, the MAV/MLP will separate and enter the Martian atmosphere. The MLP, carrying the MAV, will make a precision landing within 5 km of the rover location. After the sample cache has been received and secured, the MAV will launch from the Martian surface and rendezvous with the ERV, which will have been left in a parking orbit. The sample cache will be transferred to the ERV, and the ERV will return to Earth, arriving in May 2025. The ERV will capture into Earth orbit using a combination of propulsive maneuvers and aerobraking to bring it to a suitable orbit, and rendezvous with the ISS. ISS crew will retrieve the sample and the ERV will be secured, thereby completing Project Argo. The total mission estimated cost is FY2020\$1.7B.

**Pyxis Aerospace of Cal Poly Pomona:
Project Argo Design Team**



John Cunningham
(Team Leader, Trajectory)
AIAA ID# 544291

John Cunningham



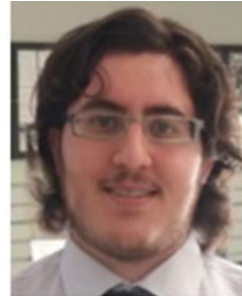
Sean Thompson
(Deputy Leader, Power,
Aerodynamics)
AIAA ID# 665541

Sean Thompson



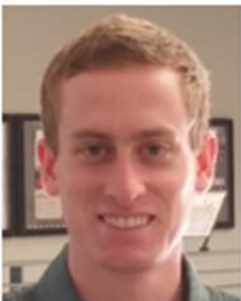
William Gallagher
(Structures, CDS)
AIAA ID# 688478

William Gallagher



Alexander Kechejian
(Propulsion, ACS)
AIAA ID# 495825

Alexander Kechejian



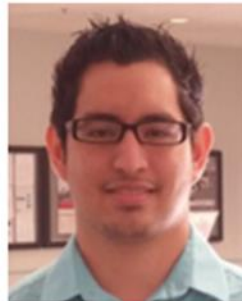
Austin Miller
(Rendezvous Operations,
Telecom, Vehicle Health
Monitoring)
AIAA ID# 605959

Austin Miller



Eddie Gomez
(Configuration, CAD
Drawings)
AIAA ID# 578871

Eddie Gomez



Carlos Ayon
(Thermal Control, Economics)
AIAA ID# 688481

Carlos Ayon

Space Design Advisor
Donald Edberg, Ph.D.
AIAA ID# 022972-00

Donald Edberg

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Nomenclature/Acronyms

AIAA – American Institute of Aeronautics and Astronautics

ACS - Attitude Control System

CAD – Computer Aided Design

CDS - Command and Data System

CEP – Circular Error Probability

EDL – Entry, Decent, and Landing

ERV – Earth Return Vehicle

EVA – Extravehicular Activity

FOM – Figure of Merit

FOV – Field of View

FRGF – Flight Releasable Grapple Fixture

G-FOLD – Guidance for Fuel Optimal Large Diverts

GMAT – General Mission Analysis Tool

GNC – Guidance, Navigation, and Control

IR – Infrared

ISRU – In Situ Resource Utilization

ISS – International Space Station

JPL – Jet Propulsion Laboratory

Ksps – Kilo Symbols per Second

MAV – Mars Ascent Vehicle

MLP – Mars Launch Platform

MOI – Moments of Inertia

MMH – Monomethylhydrazine

Nms – Newton * meter * seconds

NPR – NASA Procedural Requirements

NTO - Nitrogen Tetroxide

RFP – Request for Proposal

SOI – Sphere of Influence

SSTO – Single Stage to Orbit

TOF – Time of Flight

TRN – Terrain Relative Navigation

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Project Argo Mission Objectives

A. Scientific background and goals

Retrieving a sample from Mars for studying on Earth has been a NASA goal for decades. Compared to in-situ experiments conducted by a rover, performing such studies in person provides the chance to design new experiments based on new discoveries, and allows for use of the most up-to-date instruments. Much could be learned about Martian geologic and climatic history, and about whether the Red Planet ever supported life. In addition, a sample return mission would demonstrate much of the same technology and techniques which could be used for future human exploration.

The objective of Project Argo is to fulfill this NASA goal by landing on Mars, receiving a sample cache from a rover, and returning it to Earth to a location where it can be retrieved by NASA personnel.

B. AIAA Request For Proposal (RFP)

This proposal answers the 2015-16 AIAA Undergraduate Team Space Transportation Design Competition RFP for a Mars Sample Return System. The critical requirements are broken down as follows:

Project Argo: AIAA Critical Requirements			
Reference	Req. Ref #	Title	Requirement
3, 1	T 1.1	MAV Landing Location	The MSRS shall land the Mars Ascent Vehicle (MAV) at “any designated location” on Mars.
3, 1	T 1.2	MAV Mission	The MAV shall receive a sample cache (payload) from a pre-positioned rover.
3, 1	T 1.3	Mars Sample Return and Earth Retrieval	The MSRS shall return the payload to a location where it can be collected by NASA personnel.
3, 3	T 1.4	MAV Landing Accuracy	The MAV shall land no more than 5 km from the rover’s position.
3, 4	T 1.5	MAV Surface Loiter	The MAV shall be capable of remaining on the surface of Mars for no less than 50 Earth days.
3, 5-6	T 1.6	Sample Cache Characteristics	The MAV shall be capable of receiving, securing, and transporting to Earth a payload with the following characteristics:
	T 1.6.1	Cache Rover Lift Height	- Lifted by the rover to a height no more than 2 meters above the Martian surface
	T 1.6.2	Cache Mass	- No more than 3 kg total mass
	T 1.6.3	Cache Outer Dimensions	- Cylindrical container 15 cm in diameter and 15 cm long.

3, 2	T 1.8	MAV Timeline	The MAV timeline shall accommodate the timeline of a rover that lands on Mars Feb 2021 and operates for no less than two years.
	T 1.8.1	MAV Mars Landing Timeframe	The MAV shall land on Mars no later than Dec 2022
	T 1.8.2	MAV Mars Launch Timeframe	The MAV shall depart from Mars no sooner than Feb 2023

Critical RFP Requirements

In addition, the RFP suggests that the sample be delivered to the International Space Station (ISS) in order to remove the risk of sample contamination associated with entering Earth’s atmosphere and possibly contacting the ground or ocean before being collected.

Project Argo: AIAA Desired Characteristics			
Reference	Req. Ref #	Title	Goals / Customer Desired
3, 1	G 1.3.1	ISS Rendezvous	It is desired that the MAV return the payload to an orbital location where it can be retrieved by ISS or other spacecraft crew

The RFP recommends rendezvous with the ISS

One of our first decisions for Project Argo was to fulfill the customer desire of an ISS rendezvous for sample transfer. The reason behind this was planetary protection. If contamination of the sample is a concern, how much more is the risk of contaminating the Earth? According to NASA planetary protection provisions, a Mars sample return mission is Restricted Class V, the highest level. Meaning, any vehicle that came into contact with Mars, whether direct or indirect, is required to comply with specified sterilization levels.¹ By delivering the sample to the ISS, the station acts as line of defense, protecting the Earth against possible Mars pathogens.

Project Argo Overview

A. Selecting the Overall Mission Architecture

Early in the design process, the Pyxis Aerospace team members independently analyzed several possible mission architectures. These concepts were compared in a trade study with the goal of down-selecting to two architectures that would be analyzed in more detail. The initial architectures investigated were:

1. Single stage Mars Ascent Vehicle (MAV) + Mars orbit rendezvous with an Earth Return Vehicle (ERV)
2. 2-stage MAV + rendezvous with ERV
3. 3-stage MAV + rendezvous with ERV
4. Direct Return: A single vehicle lands on Mars, then launches and returns to Earth
5. Direct Return using In Situ Resource Utilization (ISRU): Similar to #4, but an ISRU system is used to produce propellant using Martian materials
6. Single stage MAV + rendezvous with ERV, using ISRU

The criteria used to judge the architectures were derived from the RFP requirements:

1. Mars landing mass: The total mass that must be brought safely to the Mars surface, with lower being more desirable. This was our first criteria because it reflected both the increasing risk of trying to land a heavier vehicle, and the difficulty of achieving a precision landing per the RFP requirement.
2. Earth launch mass: The total wet mass that must be lifted by an Earth launch vehicle. This criteria was a key determining factor because it indicated whether the mission could be performed with an existing launch vehicle, or if it would be dependent on a launch vehicle still in development. This was particularly important because the MAV is required to land on Mars by December 2022, in the not-too-distant future.
3. Lowest mission critical Technology Readiness Level (TRL): The element unique to the architecture that would require the most development before being ready for the mission. If the mission were to rely on an element that still required significant development work, there would be a high risk that the spacecraft would not be ready in time for launch. This criteria was driven by the December 2022 timeline requirement.
4. Number of TRLs \leq 5: NASA TRL 5 indicates that the element's components have been tested in a relevant environment. This criteria captures how many of the elements unique to the architecture are still in a fairly theoretical state of development, indicating that a significant amount of development time is still required. This criteria was driven by the RFP timeline requirement.
5. Number of separating parts: This includes propulsion system staging, separation from a cruise stage, etc. We used this as an indicator of the complexity of the vehicle, and therefore the associated risk.
6. Number of launches required: How many Earth launch vehicles would be required by the architecture. This criteria reflected how the inherent risk of Earth launch increases with multiple launches. It also

reflected that the mission cost would increase significantly with multiple launches, both due to the launches themselves and because mission operations would now be required to monitor and control multiple vehicles.

7. Number of rendezvous required: How many orbital rendezvous operations were unique to the mission (ISS rendezvous was assumed for all). We added this criteria to capture the inherent difficulty and risk associated with autonomous rendezvous and docking.

We used this criteria to compare these architectures in a Figure of Merit (FOM) analysis. The architectures using ISRU scored the lowest due to their low TRL values. Because although ISRU shows much promise for the future, it is still in the early stages of development. The Direct Return system scored the next lowest due to its high mass both for Mars landing and Earth launch. The three architectures utilizing an Earth Return Vehicle scored the highest, despite the added complexity of requiring a rendezvous.

In the end, the single stage MAV and the 3-stage MAV were selected for further analysis. The single stage MAV was chosen because it is less complex than the others, and the 3-stage MAV was chosen because it is the least massive. The 2-stage MAV had enough added complexity and mass to earn a lower score.

Down-selecting from these two architectures to the final design will be discussed in a later section.

B. Project Argo Concept of Operations Overview

1. Major Mission Phases

We have split the mission into four phases, as follows:

Phase I: Earth launch, transit, Mars capture

Phase 2: MAV separation, EDL, sample transfer

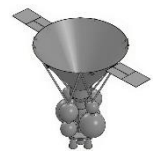
Phase 3: MAV Mars launch, ascent, ERV rendezvous and sample transfer

Phase 4: Mars escape, transit, Earth capture and aerobraking, ISS rendezvous

C. Vehicle Description

As previously mentioned, Project Argo utilizes three vehicles to carry out its mission.

1. The **Earth Return Vehicle (ERV)** acts as the cruise stage for the MAV and MLP during Earth-Mars transit. It establishes a parking orbit around Mars and acts as a communication relay during

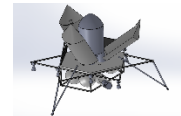


ground operations. It delivers the Mars sample cache back to Earth and conducts a rendezvous with the ISS for sample delivery.

2. The **Mars Ascent Vehicle (MAV)** descends to Mars surface via the MLP, where it receives the sample cache from the rover. It ascends to Mars orbit for rendezvous with the ERV and passes on the sample cache. It is a Single Stage to Orbit (SSTO) launch vehicle using a bi-propellant hydrazine/NTO propulsion system.



3. The **Mars Landing Platform (MLP)** is used to carry the MAV to the Mars surface for a precision soft landing. It will remain on Mars after MAV launch.



D. Overview of Key Vehicle Design Features

Project Argo utilizes a unique architecture to carry out its mission. These are some of the mission-enabling design features of the vehicles:

1. Earth Launch Configuration

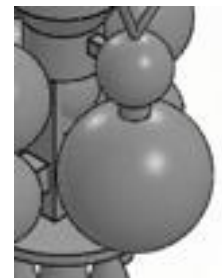
Project Argo launches from Earth as a single vehicle. The MAV/MLP are enclosed in a backshell/heatshield assembly similar to those used on previous Mars missions. This assembly is secured within the ERV capture cone. Power and data connections between the two allow the ERV to function as the cruise stage for the MAV/MLP.



2. ERV drop tanks

The ERV is equipped with 4 pairs of propellant drop tanks, which function in much the same way as stages do for a launch vehicle.

Firstly, the drop tanks reduce mass after maneuvers. Each set of drop tanks is sized to perform a specific set of maneuvers; when depleted, they are jettisoned in order to reduce the spacecraft mass for the subsequent maneuvers.



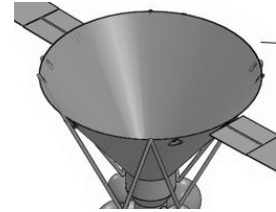
Secondly, the drop tanks serve as structural support. At Earth launch, the MAV/MLP are a large mass sitting on top of the ERV. This mass must be supported structurally in order for the spacecraft to survive the high acceleration loads during launch. But, the structural loading for the ERV is dramatically reduced once the MAV/MLP separates for Mars landing. Therefore, the drop tanks are arranged such that they act as the primary load bearing structure for

the lander. This design greatly reduces the required primary structural mass of the ERV. No tanks are jettisoned until after the MAV/MLP separate and their structural role is no longer needed.

But most importantly, utilizing drop tanks made it possible to perform the entire mission using a single launch vehicle, which reduces the cost and complexity of the mission.

3. *ERV capture cone*

The forward end of the ERV is a concave cone. It is used to assist in the sample transfer between the ERV and the MAV. The cone truss system acts as the supporting structure for the MAV /MLP during Earth launch and Earth-Mars transit, where the ERV acts as the spacecraft cruise stage. The cone also acts as a stabilizing drag area during Earth aerobraking.



4. *MAV sample storage and deploy system*

After retrieval from the rover, the MAV stores the sample in a cavity along its center line. Upon rendezvous with the ERV, the sample will extend out from the vehicle on a coil boom. The ERV capture cone will guide the sample canister into its receptacle during the ERV / MAV rendezvous.

The concept of using a capture cone on one vehicle and an extending boom on the other is taken from a 1974 JPL Mars docking feasibility study.ⁱⁱ

5. *MLP Terrain Relative Navigation system*

One of the challenging requirements was to land within 5 km of the rover's position. The MLP will use the Terrain Relative Navigation system currently being developed by JPL for the Mars 2020 mission. This system uses optics and image recognition to identify its location relative to images stored in the vehicle computer. It will then use monopropellant thrusters to divert to its pre-programmed landing site as it descends.

E. Design and Analysis Tools

Several software tools were used to design the mission trajectories and spacecraft maneuvers.

The preliminary trajectory dates, C3 values, and arrival velocities were identified using Trajectory Optimization Tool ver. 2.1.1.ⁱⁱⁱ This tool is a MATLAB-based program that uses patched-conic approximations to solve Lambert's

problem in the DE421 ephemeris. Its solutions can be controlled to optimize for C3 departure energy or for hyperbolic arrival velocity. This tool was also invaluable in providing the trajectory orbital elements that were used as the initial conditions for the GMAT simulations, discussed below.

The Earth launch opportunity porkchop plot was generated using a MATLAB tool retrieved from the MathWorks file exchange.^{iv} This was used to find the specific target launch date and to determining the length of the launch opportunity.

Early in the design process, MS Excel spreadsheets were set up to calculate maneuver Δv values for all mission maneuvers. Even after a higher fidelity mission model was produced, these spreadsheets were very useful for performing quick checks of different scenarios. MS Excel was also used to build simulations for analyzing Mars landing and launch. These will be discussed in more detail in their respective sections.

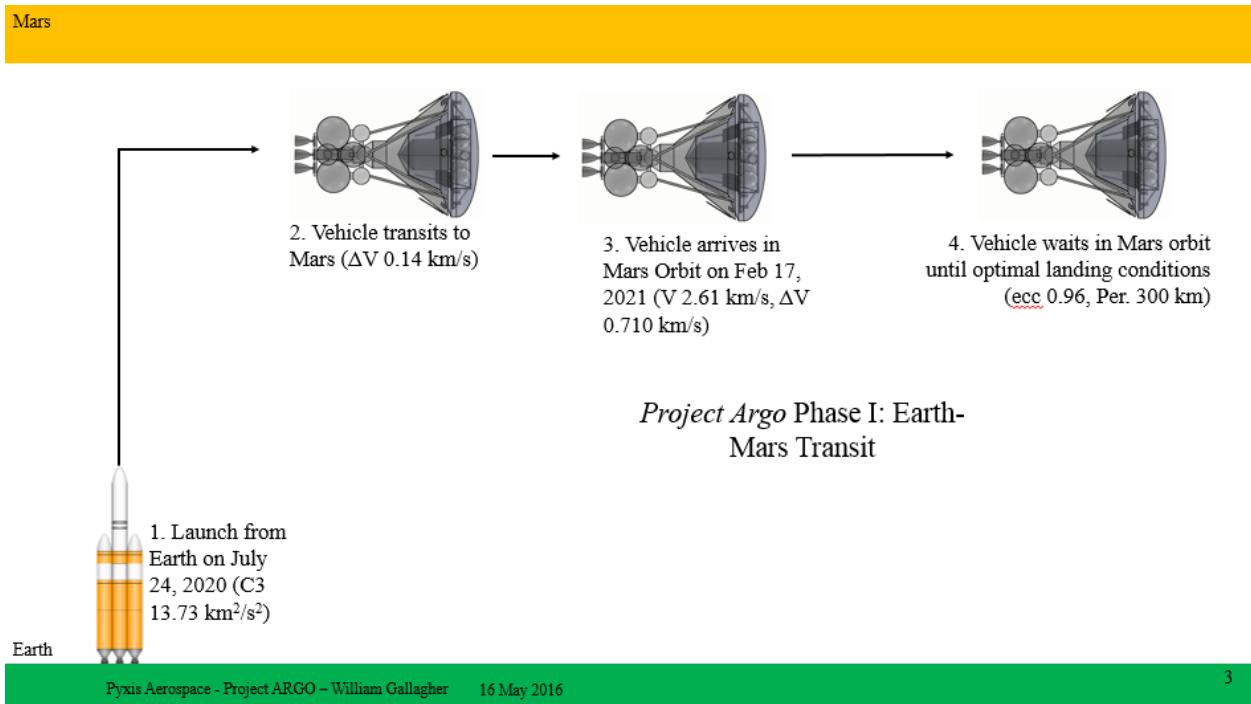
Later in the design process, the primary tool used to analyze the mission trajectories and maneuvers was the General Mission Analysis Tool, or GMAT. GMAT is an open source space mission design tool developed by a team of NASA personnel, private industry, and individual contributors. It is used for both real-world engineering analysis and educational purposes. A GMAT simulation was created for each mission phase, which was used for validating and refining the initial trajectory calculations.

MS Excel was used extensively throughout Project Argo, both for analysis of each subsystem and for organization of mass tables, etc.

All CAD drawings were created using SOLIDWORKS. SOLIDWORKS Simulation was also used to model both static loading for vehicle structural analysis, and transient heat transfer for thermal analysis

Project Argo Concept of Operations

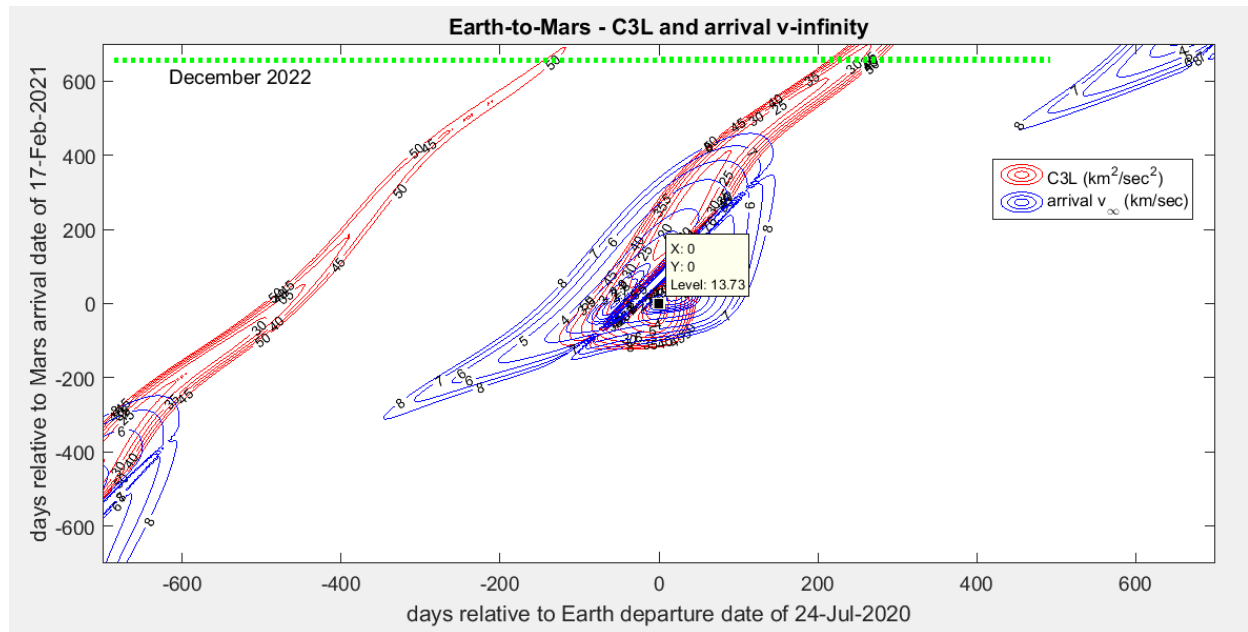
A. ConOp Phase I: Earth launch, transit, Mars capture



1. Earth to Mars Trajectory

Project Argo's Earth-Mars trajectory was selected based on the timeline given in the RFP. The RFP states that the rover lands on Mars in February 2021 and has a lifespan of no less than two years. The RFP also states that our system shall give the rover 50 days to reach the MAV and deliver the sample cache. We therefore derived the requirement to land the MAV by December of 2022.

One important way that our trajectory is different from Earth orbit trajectories is that launch latitude was not a constraint. A hyperbolic escape trajectory can be reached from any inclination, so there is no need to choose a launch location based on latitude. Cape Canaveral is therefore the most economic and accessible launch facility.

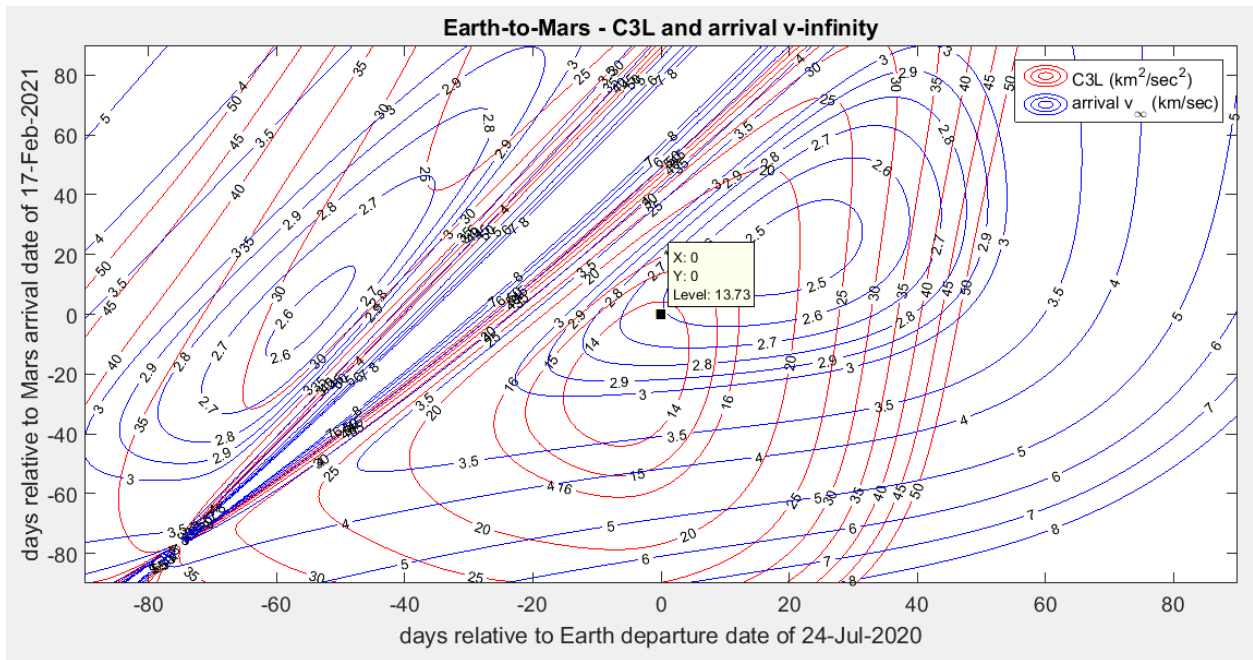


Porkchop Plot for Launch Opportunities
 The plot shows that the only feasible launch opportunity for an arrival prior to December 2022 exists near July 2020

To arrive by this date, there is only one opportunity with feasible launch energy and arrival velocity. This opportunity occurs June-August of 2020, with a Mars arrival of Jan-March 2021. This is the same launch opportunity that will be used by the rover; the risk associated by this cross-over will be addressed in the risk section.

We selected Project Argo’s target launch date to be July 24, 2020, with a margin \pm two weeks. Our analysis showed this date to have an optimal balance between launch C3 and arrival velocity, resulting in completing the mission with

the highest propellant margin possible. Launching at another date in the \pm two weeks margin allows us to complete the mission with an acceptable propellant margin. Propellant margins will be discussed in a later section.



Porkchop Plot for Earth Departure
 Earth launch date: July 24, 2020 Launch C3: 13.73 km²/s²
 Mars arrival date: February 17, 2021 Arrival velocity: 2.61 km/s

2. Earth Launch Vehicle

We decided early in the design process that Project Argo would be designed around existing, flight proven launch vehicles. The mission is required to launch in the next five years, and we did not want to assume the risk of the mission failing because a promised higher performance launch vehicle failed to materialize. To find what existing flight vehicles could be used, we used NASA’s Launch Vehicle Performance Website to find the launch mass capability of different launch vehicles for a given C3. The Delta IV Heavy was by far the most capable.



Project Argo will utilize a single Delta IV Heavy for Earth launch. Image source: <http://www.americaspace.com/?p=22180>

As part of investigating multiple mission architectures, we analyzed the cost and risk factors of using a single launch vs. two or more launches. We found that the higher price tag of the Delta IV Heavy compared to other launch vehicles was offset by the lower operational cost of monitoring and controlling one

spacecraft vs. many. The Delta IV Heavy also has a perfect flight record (after its first test flight), lowering the risk associated with launch.

We therefore selected a single Delta IV Heavy as the Project Argo Earth launch vehicle. It is capable of lifting 8321 kg with our launch C3 of 13.73 km²/s². The Payload Attach Fitting (PAF) has a mass of 418 kg, which means that the do-not-exceed mass for Project Argo is 7903 kg. The payload fairing has a usable diameter of 4.572 m and a usable cylindrical height of about 13 m.

The opportunities associated with future launch vehicles will be discussed in the Opportunities section.

3. Earth-Mars transit

The Project Argo spacecraft launch from Cape Canaveral on a Delta IV Heavy on July 24, 2020 and inject into a Mars transfer orbit. Time of flight for the Earth-Mars journey is 208 days. During this time, calibration burns of the main engines and of the ACS thrusters are conducted in order to obtain actual values for engine thrust and specific impulse. During transit the position of the ERV will be tracked by the Deep Space Network (DSN), similar to other spacecraft.

Similar to other missions, three to five Trajectory Correction Maneuvers (TCMs) are expected.^v Our GMAT simulation needed one maneuver of 140 m/s at 45 days from Earth in order to reach the desired Mars periapsis. TCMs will be used to target the Mars B-plane in order to capture into the inclination that will place it over the rover's position.

4. Mars Capture

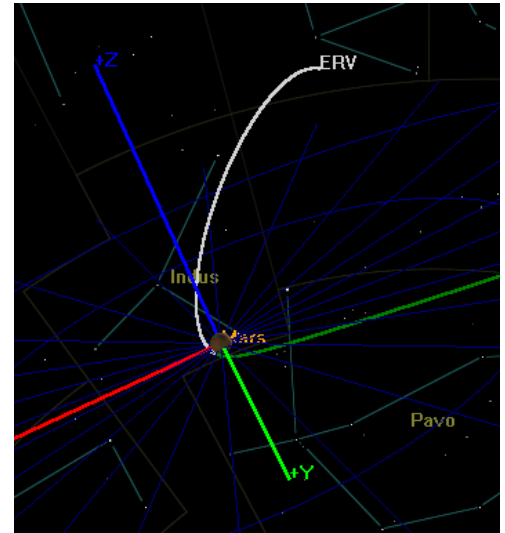
The spacecraft encounters the Mars Sphere of Influence (SOI) on February 15, 2021 and captures into Mars orbit on February 17. The Martian B-plane is targeted in order to bring the spacecraft to a 300 km periapsis. This altitude minimizes the Δv needed for capture while staying above the atmospheric interface, estimated by the Mars Science Laboratory mission to be at 125 km. This is the same capture periapsis used by the Mars Reconnaissance Orbiter (MRO).

The capture orbit inclination will be determined by the rover location. The rover should arrive at Mars before our spacecraft; we will capture into the inclination that is planned for the rover. This risk is addressed in the Risk section.

At this phase the spacecraft is at its most massive, with a full propellant load and the ERV still carrying the MAV and MLP in their entry configuration. In order to minimize Δv , and therefore the propellant needed, the initial capture orbit will be highly elliptical, with an eccentricity of 0.96. Compared with capturing into a circular orbit, this shrinks the Δv needed from 2.07 km/s to 0.71 km/s.

The orbital insertion maneuver is performed using propellant from the ERV's first (and largest) set of drop tanks. This first set of drop tanks will not be jettisoned until after the MAV and MLP separate; this means that the tanks can use their left over propellant for any maneuvers needed to optimize the MAV landing, or even to assist with the next phase of maneuvers allocated to the second set of tanks. This will also be discussed later.

The Mars orbit insertion burn time is calculated to be 1493 seconds, about 25 minutes. MRO had a similar orbital insertion profile, with the same 300 km periapsis and a highly elliptical capture orbit. The MRO capture burn also lasted 25 minutes. Based on publically available information of wet mass, dry mass, total thrust, and insertion Δv , we calculated that the MRO used approximately 4% more propellant during its finite burn than the propellant needed for an ideal impulsive burn. We assume that we will have approximately the same kinematic efficiency. With this taken into account, we designed propellant margin for the orbital insertion maneuver to be 25%.



<p>Mars Capture Orbit Periapsis altitude: 300 km Inclination: Determined by rover mission Eccentricity: 0.96 Δv: 0.71 km/s</p>
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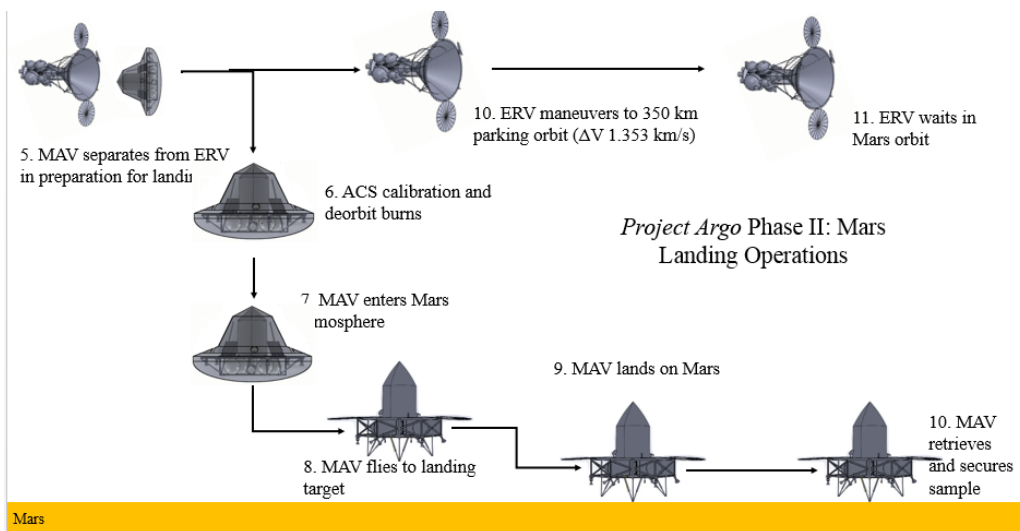
One of the design decisions we made was between capturing into Mars with all three vehicles vs. having the MAV/MLP separate from the ERV on approach and enter Mars atmosphere directly while the ERV captures into Mars orbit on its own. Capturing into Mars orbit with the MAV/MLP still attached to the ERV increases the required propellant mass significantly. However, this strategy was ultimately selected because of several benefits to the mission, as described below.

First, the ERV and MAV arrive at Mars at about the same time as the rover; but, the rover is not required to link up with the MAV until the end of its (the rover's) two year life span. By capturing into orbit, the MAV has almost two years before it needs to land. This gives the rover ample time to travel freely, without being required to stay within range of the MAV. Mission Control can select a landing zone based on the rover's position at end of life, or at some other time determined to be optimal.

Second, capturing into orbit allows precise tracking of the spacecraft's position. It will be loitering in orbit for anywhere from several days to several months, giving ample time to pinpoint its orbital elements using a variety of methods. Precise knowledge of the MAV/MLP's position greatly reduces the uncertainty associated with direct hyperbolic entry, thereby improving the landing accuracy.

Third, entering Mars atmosphere from orbit decreases the entry velocity by approximately 1 km/s compared to entering from a hyperbolic trajectory. This decreases the heat load during atmospheric deceleration, which in turn decreases both entry risk and the necessary heat shield mass.

B. ConOp Phase II: MAV separation, EDL, sample transfer



1. MAV/MLP Detach and De-Orbit

As discussed previously, the ability to loiter in orbit gives Mission Control a great deal of flexibility in determining when to initiate landing for the MAV/MLP. Expected decision factors include: 1) availability of DSN resources; 2) close approach of Mars and Earth in order to minimize communication lag; 3) rover mission, including updates to the rover lifespan and proximity to ideal landing sites; and 4) local Mars weather.

An additional factor is the Terrain Relative Navigation system used by the MLP to navigate to a precision landing. This system uses image recognition in the visible light spectrum, comparing images it takes with its camera with images stored in its memory. For this reason the precise deorbit window depends on visibility being high (no dust storms) and is timed such that the current lighting conditions match those of the stored images.

When it comes time for separation and landing, the MAV/MLP will first come out of standby and perform systems checks. Mission Control will give the go ahead for separation. The physical and electric connections with the ERV will be severed, and the ERV will use its forward-facing ACS thrusters to perform a separation maneuver. These thrusters are oriented parallel to the cone edge, so this will keep the exhaust plume from impacting either of the vehicles.

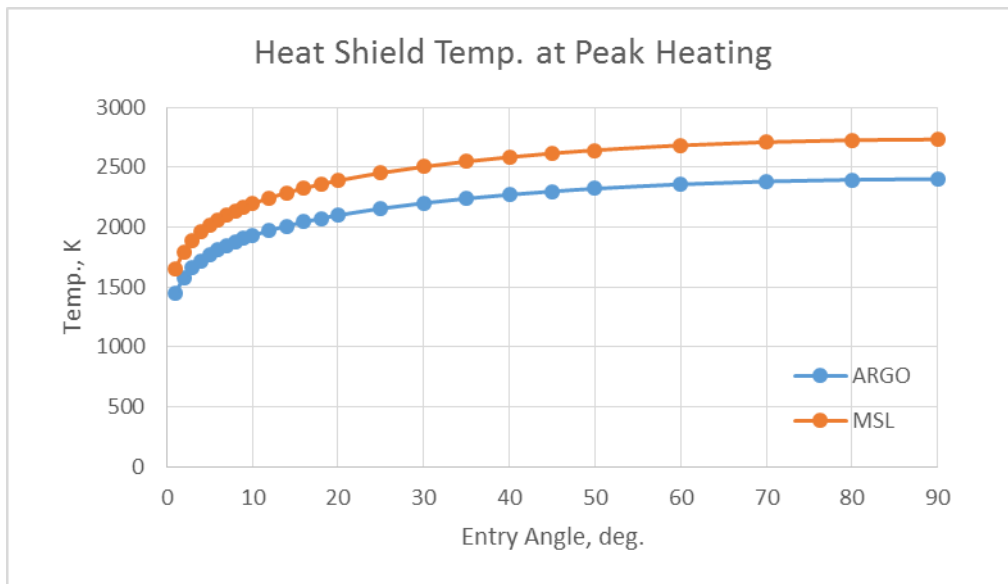
Once a safe standoff distance has been established between the ERV and MAV, the MAV will perform small maneuvers with its ACS thrusters for calibration. Finally, it will perform a de-orbit maneuver that targets the landing site. The approximate Δv for this maneuver, performed at apoapsis, is 8 m/s. This and all subsequent maneuvers will be performed by thrusters on the MAV/MLP landing backshell, similar to that of MSL.

2. Mars Entry Aerothermodynamics

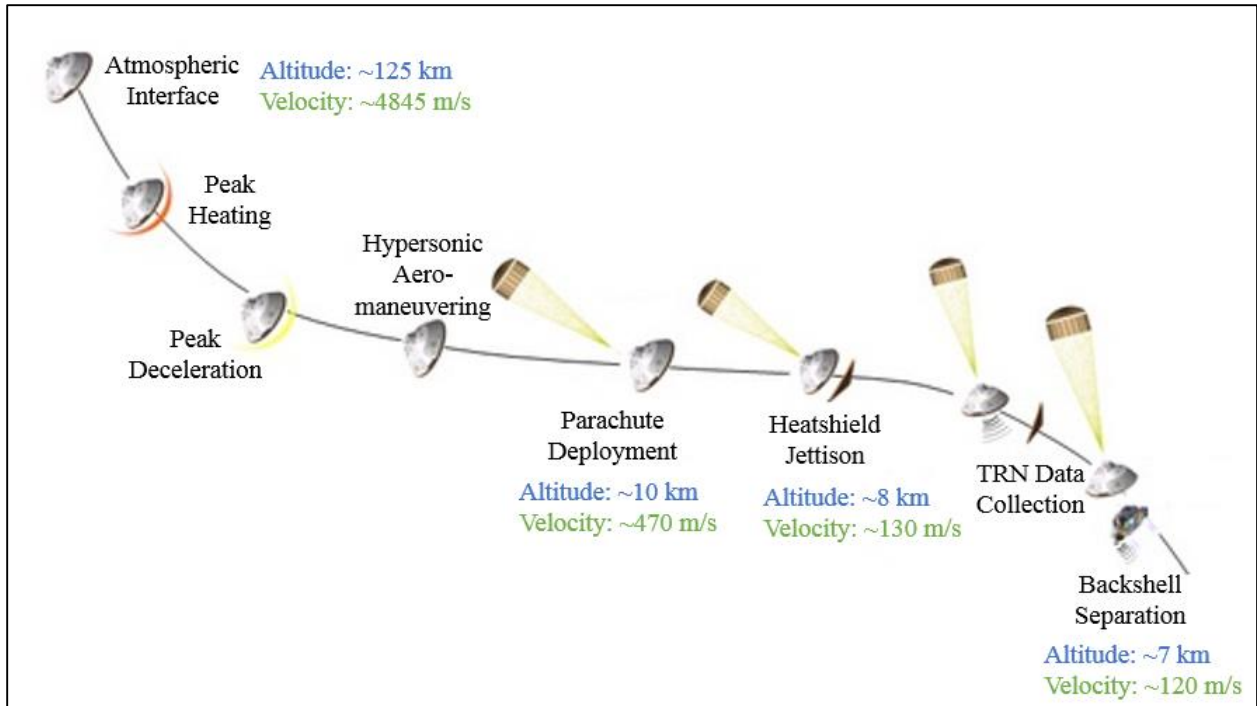
The goal of our aerothermodynamic analysis was to determine the size of the heat shield required for atmospheric entry at Mars. To do this we needed to calculate the total heat load, peak heating rate and peak wall temperature experienced by the heatshield during entry. In order to do this we use Allen-Eggers trajectory equations to determine the density, altitude and velocity at peak heating. Then these values were used in the Sutton-Graves equation to calculate the peak heating rate. We then use this peak heating rate in the Stefan-Boltzmann law to determine the wall temperature of the heatshield. Then, assuming an exponential atmosphere we can obtain an

approximate value for the total heat load.^{vi} We can also calculate the peak deceleration and the altitude and velocity at which it occurs.

In order to do these calculations we assume a value for the vehicle emissivity and drag coefficient of 0.8 and 1.4 respectively. Also we assume a Viking heritage 70 degree blunted nose cone shape for our heatshield with a diameter of 4.5 meters, thus allowing us to directly compare with MSL. The input for the equations are then simply the entry mass, entry velocity, and entry angle. The calculations were then completed for a range of entry angles and then graphed along with those completed with data obtained for MSL.^{vii} The resulting graphs can be seen below.



As shown in the graphs above the peak heating and total heat load expected for our vehicle are approximately 30% less than those calculated for MSL for the same entry angle. Therefore we have estimated that our heat shield will be 15% less massive than that of MSL. Design of the backshell was beyond the scope of this proposal, but because our landing mass and volume are close, we assume that its design as well as the EDL sequence will be similar to MSL. The expected entry sequence for Project Argo can be seen in the following figure.



As shown in the figure above, atmospheric interface occurs at approximately 125 km. At atmospheric interface our entry vehicle will be traveling at 4845 m/s which is about 1000 m/s less than MSL, which was traveling at 5800 m/s. The altitude and speed at peak heating and peak deceleration will depend upon the exact entry angle, which will be determined by mission control at the time of entry. But, based on structural analysis, the maximum allowable g-load is 5.5 g. This limits our entry angle to values smaller than 8° . This is acceptable because smaller entry angles result in more accurate landing areas. The parachute deployment, heat shield jettison, and backshell separation are assumed to occur at the same altitude and speed they did for MSL.

The Monte Carlo analysis necessary to determine the landing Circular Error Probability (CEP) of probability for descending from orbit was beyond the scope of this project. However, the size of the CEP is a function of the uncertainties of the entry profile. Capturing into Mars orbit allows for precise tracking and validation of its orbital position, thereby decreasing the uncertainty associated with a hyperbolic entry. We are therefore confident that the CEP for Project Argo will be less than that of MSL, which entered hyperbolically. For comparison, MSL had a CEP of approximately 20 km x 7 km.

3. MAV Landing: Terrain Relative Navigation

The RFP required that the MAV land within 5 km of the rover's position. This was one of the more challenging aspects of the RFP, as no current Mars lander has been able to guarantee a landing ellipse more accurate than 20 km x 7 km. In order to accomplish this, the MLP will use Terrain Relative Navigation.

TRN is a concept being developed by JPL in conjunction with Masten Space Systems in Mojave, CA. The test bed is called ADAPT and utilizes two key pieces of TRN technology. The first is the Lander Vision System (LVS). As the vehicle descends, the LVS takes a series of pictures – lower resolution at first, at high altitude; then with increased resolution as it gets nearer to the ground. The pictures are compared with images stored in the vehicle's computer. The system uses object recognition to identify first the vehicle's location, and second the vehicle's target.

The second key piece of technology is an algorithm called Guidance for Fuel Optimal Large Diverts (G-FOLD). Developed by JPL, this software calculates the trajectory needed to reach the target site while minimizing propellant burn. In September 2013, a Masten vehicle used G-FOLD to abort its original trajectory, calculate a new trajectory in 1 second, and immediately maneuver the vehicle to a new landing site half a mile away.

The ADAPT test bed integrated both LVS and G-FOLD and was successfully flown twice in December 2014. JPL plans to use this technology for the Mars 2020 mission, so we assume that an operational TRN system will be ready in time for our launch as well.

The TRN system kicks in while the MAV/MLP are descending on the parachute, still within the backshell, and continues to operate during freeflight. The LVS begins imaging immediately after the heat shield falls away and operates continually through the flight. G-FOLD calculates the optimal trajectory in real time.

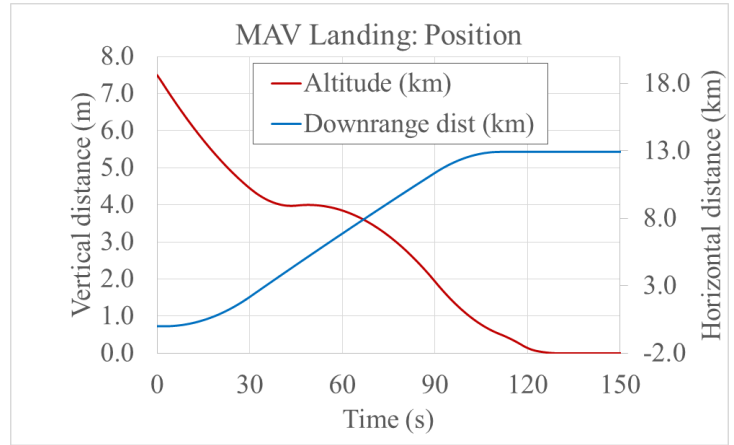
In addition to the LVS system, the MLP carries a radar ranging system with 2 antenna, based on the radar used by MSL. The radar acts as a backup system in order to determine altitude, particularly on final approach when dust could potentially disrupt the optics of the primary system.



Masten Space Systems' Xombie technology demonstrator lifts off during a flight test of JPL's G-FOLD precision flight control algorithm, autonomously diverts to an alternate landing site, and descends for landing at the alternate site.
Credits: NASA / Tom Tschida

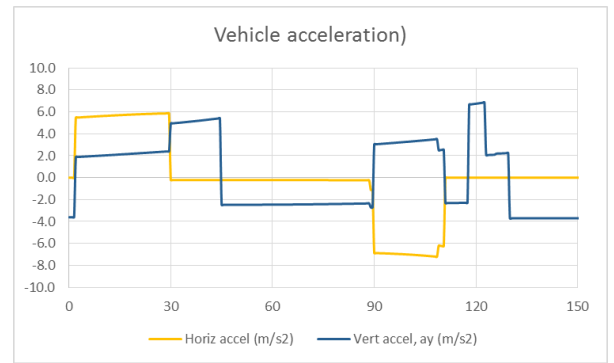
4. Mars Landing: Mars Landing Platform Propulsion System

In order to show that the MLP would be able to use TRN to land within 5 km of the rover, it was necessary to design a propulsion system that allows the vehicle to catch itself in free fall, turn and fly horizontally, then arrest its horizontal velocity and descend to a soft landing. We reasoned that the farther it is able to fly horizontally, the further its initial descent could be from the rover before flying to the landing site. For example, if the maximum dimension of the CEP

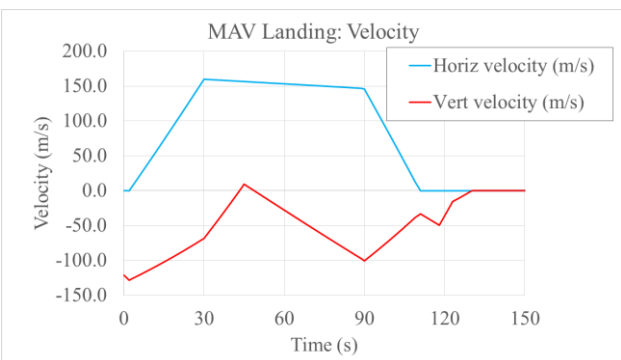


was 15 km, the MLP would have to be able to fly 10 km horizontally in order to meet the 5 km accuracy requirement.

To design this system, we developed a 2-degree-of-freedom (2 DOF) MLP flight simulator using Microsoft Excel. It allows the user to change the vehicle flight path angle and thrust as a function of time. Propellant mass is decremented according to the value of thrust at each second. The simulator also takes into account local gravity as a function of altitude, and horizontal/vertical drag as a function of velocity and altitude. An exponential atmospheric model is used with a scale height of 11.1 km and a surface density of 0.015 kg/m^3 .

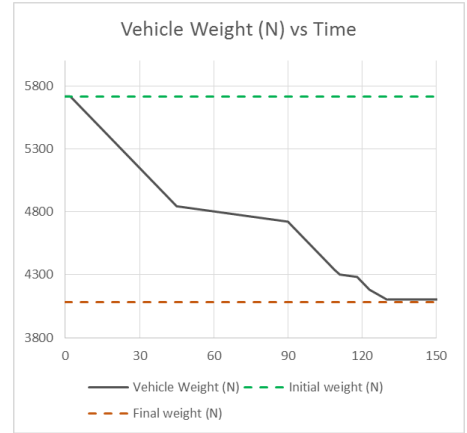


The results of the simulated flight are shown. As soon as the heat shield is clear, the MAV/MLP drop from the backshell and begin free flight at approximately 7.5 km altitude with an initial descent velocity of 121 m/s. The vehicle free falls for 2 seconds in order to clear the backshell. It then performs a hard lateral maneuver to build up 150 m/s horizontal velocity toward the landing target. It levels off and continues to descend while coasting horizontally. A



minimum thrust of 1600 N is assumed for maintaining control. Approximately 90 seconds into the flight, the MLP performs another aggressive maneuver to arrest its horizontal velocity. It then descends to a soft landing with a touchdown velocity less than 0.5 m/s. This is accomplished with a propellant margin just over 15% remaining.

It is worth noting that this is a 3 DOF simulation and does not take into account all variables. Flight path angle and thrust changes are instantaneous, so actual propellant use will be higher. However, the flight path described is not optimized. We are therefore confident that the engine power and propellant mass that we designed are sufficient to carry the vehicle several kilometers from its initial landing trajectory.



It is also worth noting that 13 km is more than enough to land with 5 km accuracy, since the max CEP dimension will almost certainly be less than 18 km.

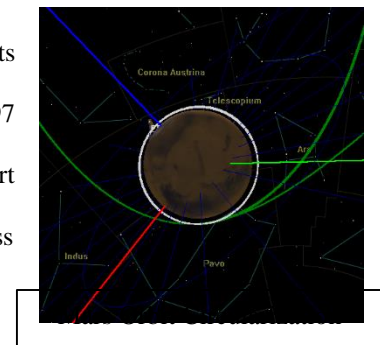
This leads to an opportunity to reduce the vehicle mass by reducing the mass of propellant. However, we chose to over-design the MLP propulsion system to leave plenty of room for contingencies, since this is the first time a powered flight and landing like this has been attempted.

5. ERV parking orbit

After the MAV has separated, the ERV is free to jettison its first set of drop tanks and maneuver to its 350 km altitude circular parking orbit. There is no need to jettison the tanks immediately; any propellant margin can be used to assist with the circularization maneuvers, thereby increasing the propellant margin of the second set. Another option, prior to circularization, is to lower the capture orbit periapsis such that the jettisoned tanks will skim Mars atmosphere. This will ensure that the orbit eventually degrades and the tanks impact Mars surface.

Deliberately deorbiting the drop tanks has the benefit of keeping the skies above Mars clear; however, it leads to planetary protection risks. These risks will be mitigated by decontamination measures during manufacture, assembly, and launch operations. Though we are not relying on it, the risk is further mitigated by the fact that the tanks are made of aluminum. During entry the tanks will experience high temperatures. The high thermal conductivity of aluminum will cause the entire surface to heat up, even if the tank is tumbling.

Assuming the ERV begins from its initial elliptical capture orbit, the Δv to raise its periapsis to 350 km is small, less than 1 m/s, and its total Δv to circularize is 1.297 km/s. This Δv is accomplished over several orbital periods, using a series of short burns at periapsis in order to simulate impulse burns and minimize the energy loss



associated with finite burns. With this strategy we assume that the additional propellant associated with finite burns is decreased from 4% to 1%.

The 350 km altitude was chosen in order to balance the propellant requirements between the ERV and the MAV. The lower the orbit, the more propellant required by the ERV to circularize. The higher the orbit, the more propellant required by the MAV to reach it. In addition, the orbit must be far enough from the influence of the atmosphere to be stable. Our analysis showed that this altitude was near optimal.

6. MAV Sample Stow and Transfer Subsystem

Once on the surface, the MAV and MLP wait for the rover to deliver the sample canister. This sample canister is the purpose of the entire mission, so we considered several ways to collect, stow, and transfer it to the ERV after launch.

The RFP states that the rover will be able to lift the sample canister no more than 2 meters off the surface, so this was our main design constraint. After this, our goal was to minimize the number and complexity of mechanisms required. We considered having the MAV nose cone pivot or flip open to receive the canister. This would be higher than two meters, so we considered equipping the MLP with either a robot arm or a gantry system to place the sample canister into its receptacle. However, this strategy required several moving parts; in particular, articulating the nose cone would be challenging. We also considered having a door open in the side of the MAV. This most likely would still require a robot arm or gantry to place it inside, in addition to the door mechanism. We also considered having the MAV mounted laying down on its side on the MLP. This would keep a door on the MAV below the two meter requirement. A gantry system would lift the MAV upright once the sample was stowed. However, we wished to avoid this because of the mechanisms required to lift the MAV; and because during Earth launch the vehicle would experience high acceleration loading across its side, meaning that the structural mass would most likely be higher.

In the end we hit on a design that accomplishes the required tasks with minimal mechanisms.

The first component of our design is the receptacle for the rover's sample canister. We call it the "sample bucket."

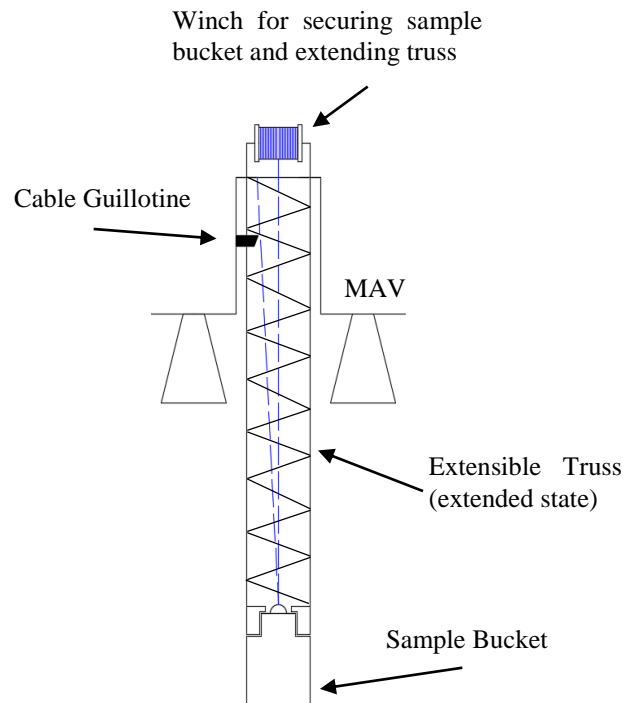
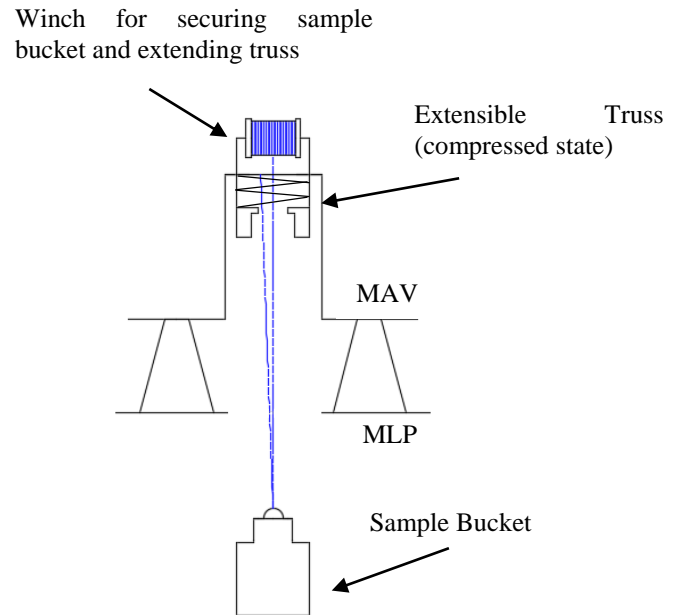
It is mounted to the bottom of the MLP at the end of a chute. The receptacle portion is pulled into a sleeve to close it after the sample canister is loaded, but it is mounted in the open position.

The bucket is connected to a cable which runs through the chute, up through an opening in the middle of the MLP, and up through a tube along the centerline of the MAV. Inside the MAV, the cable is connected to a small winch.

Once the rover places the sample canister into the sample bucket, the winch begins to wind the cable. This action first pulls the bucket door closed, then draws the bucket along the chute and up into the MAV. It fits into place securely but does not physically lock; it is held by the tension of the cable on the winch. The sample bucket handle fits into a key hole on the coil truss to hold it in place during launch. We are

With this strategy, the only powered moving part required to stow the Mars sample cache is a winch – a simple mechanism, with both low power and low mass.

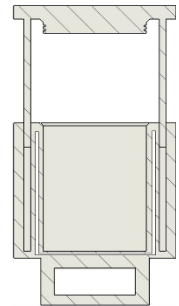
The other half of this subsystem is getting the sample from the MAV to the ERV during rendezvous. For this, we borrowed an idea from the JPL Mars autonomous docking feasibility study mentioned previously; namely, having the bucket on the end of an extending boom. For the JPL study the boom was powered. For Project Argo, we found that we could use a coil truss. This type of truss is arranged such that it can be folded axially down to 2% of its extended length; when unlocked, it extends naturally due to the tension of its members. For the MAV, the winch and cable can be used to control the extension rate. Using this system removed yet another need for a powered mechanism.



As stated before, the bucket is held to the truss by the tension of the cable. The cable is fed through the handle of the bucket and back inside the MAV, where it is fixed. Once the MAV has successfully inserted the bucket into the receptacle of the ERV and it is locked into place, a guillotine mechanism severs the fixed end of the cable. This way, as the MAV and ERV separate, the cable simply pulls through the bucket handle and stays with the MAV.

We recognized a risk that any physical lock securing the bucket inside the MAV could fail to disengage; this would result in mission failure, since the ERV would not be able to obtain the sample, and the MAV would not be able to leave Mars orbit on its own. Using the winch, cable, and guillotine system removes this risk. The risk of the cable failing is minimal, since the weight of the bucket is small even with the Mars launch loads.

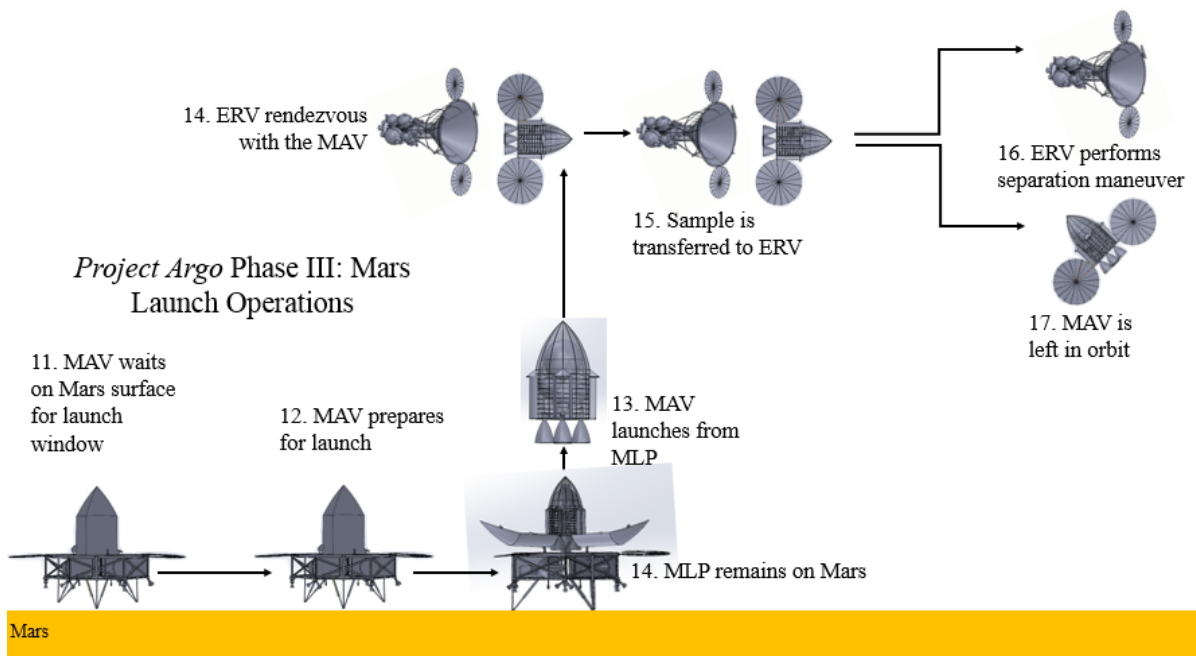
On the ERV side, the receptacle for the bucket is at the bottom of the capture cone. During sample capture, the MAV is thrusting against the more massive ERV; so as the MAV thrusts against the wall of the capture cone, the ERV will remain relatively unmoving while the MAV and bucket will be guided into the center of the cone. Once in place, the bucket will be locked by a simple spring loaded clip. At this time the cable will be severed.



Sample Bucket

Since the ERV is transferring the sample to the ISS, we considered that it must be made easily accessible by astronauts in space suits, if not by a robot. For this reason, the bucket handle protrudes from the receptacle toward the cone and is large enough to be gripped by an astronaut's gloves. The spring loaded lock that holds the bucket in place is itself held in place by a slide that is easily actuated.

C. ConOp Phase III: MAV Mars launch, ascent, ERV rendezvous and sample transfer



1. MAV Architecture Selection

Early on in the design process, two independent MAV architectures were considered: The first was a Single Stage to Orbit (SSTO) ascent vehicle and the second was a 3-stage ascent vehicle. Down-selecting to a single architecture was one of the major design decisions of Project Argo.

The SSTO MAV uses four NTO/Hydrazine engines to reach parking orbit. After an initial burn it coasts to 350 km altitude, then fires its engines again to circularize. The SSTO MAV is more massive than the 3-stage, but offers more on-orbit dry mass and volume, as well as greater control and flexibility on launch.

The 3-stage MAV uses two solid propellant stages to reach altitude, then a monopropellant 3rd stage to perform a Hohmann transfer to parking orbit. At first, the 3-stage seemed like the obvious choice due to its lower wet mass.

The trade study to choose between the two took into account wet mass, on orbit dry mass, on orbit volume, risks associated with stage separation, ability to control on ascent, launch g-loading, and other minor considerations. Despite its higher wet mass, the SSTO MAV won out on nearly all other points. Results indicate that the higher performance of a bipropellant engine makes more of a difference in the low drag/low gravity Mars environment.

SSTO	
Pros	Cons
More control on ascent	Heavier
Single stage - less risk	
More original	
Sample loading more simple	
Compact shape is better for aeroshell	
Less structural mass for clamshell	
3 Stage	
Pros	Cons
Less massive	Multi stages pose risk
More aerodynamic	Lower volume of final stage
In rendezvous, less mass to potentially crash into ERV	Less mass for final stage
	Higher g loads
	Final Hohmann transfer requires main engines/ large ACS thrusters

	SSTO	3 Stage
Wet mass	10	0
Available volume	9	1
Final stage dry mass	8	1
Staging risk/pyro shock	7	1
Ascent controls	5	1
Hohmann hardware	4	1
g-loading for launch	3	1
Ease of loading sample	2	1
Rendezvous crash mass	1	0
	38	11

MAV architecture selection Figure of Merit Analysis

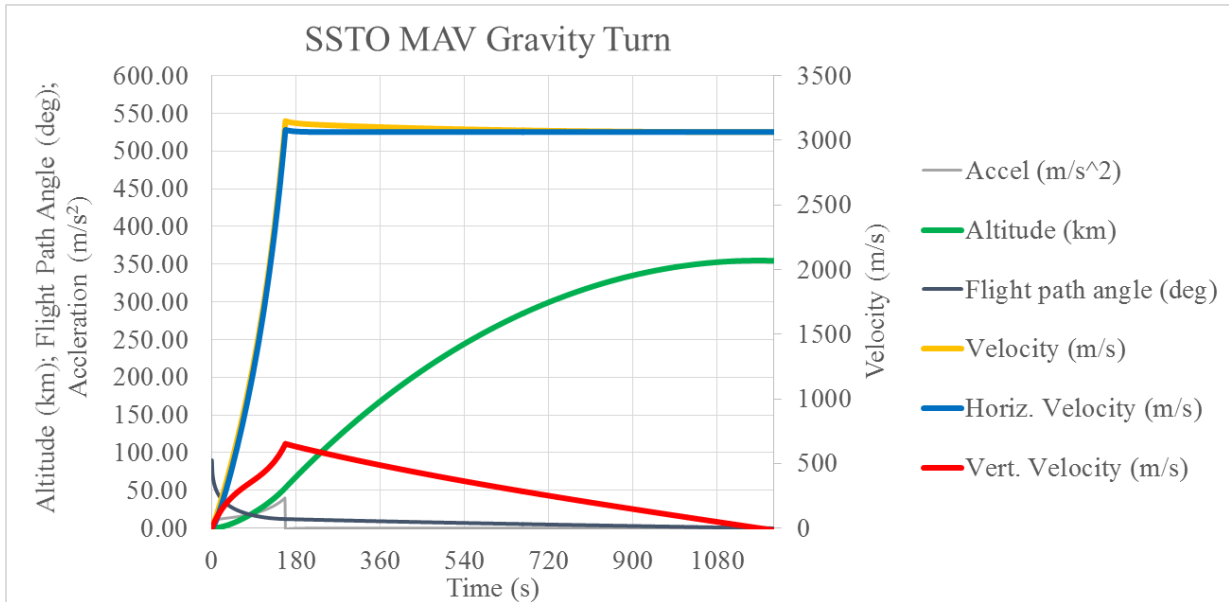
2. MAV Launch

According to the RFP, the rover will deliver the sample no later than February 2023. The Earth return trajectory takes place in July 2024. This means that the MAV has over 500 Earth days in which to launch and rendezvous with the ERV. Mission Control will determine the optimal launch and rendezvous date and time. Some considerations are: 1) vehicle status; 2) local weather; and 3) communications concerns such as alignment of Earth and Mars and availability of the DSN. The specific launch window will be timed to facilitate rendezvous with the ERV.

The pre-launch timeline will begin with MAV system checks. The MLP clamshell will then open. MLP cameras will confirm that the clamshell is open completely. If possible, the rover at this point will have navigated a safe distance away, at a position where it can view the launch.

A “method of shooting” launch simulation was built using MS Excel to simulate a gravity turn in the local horizon reference frame.^{viii} By giving the vehicle a “kick” near the start of launch, angular acceleration is established that eventually brings the vehicle to a horizontal trajectory at burnout. Like the landing simulator, the program takes into account drag and local gravity. This simulation allowed us to find the necessary propellant mass and thrust.

At launch, the MAV fires its four 2500 N thrusters and lifts off with an initial acceleration of 9.8 m/s^2 , almost exactly one Earth gravity. The thin atmosphere at Mars allows for a more aggressive gravity turn than would be possible on Earth; therefore, 1 second after launch, the vehicle kicks over by 16.2 degrees. This is accomplished by varying the thrust of the main engines combined with thrust from the ACS system. Maximum dynamic pressure occurs at 82 seconds with a drag force of 777 N. The MAV main engines burn for 156 seconds and reaches a peak acceleration of 40.0 m/s^2 , 4.1 gs. Burnout altitude is 53 km.



MAV Mars launch, gravity turn, initial burn, and coast. Orbit insertion is completed using a circularization burn that begins near apoapsis.

At burnout, the MAV is traveling over 3000 m/s horizontally, and is still climbing at rate of over 600 m/s. Its vertical velocity immediately begins decreasing, but it is able to coast to 350 km over the next 17 minutes. At this point it again fires its main engines for just over one minute for a Δv of 320 m/s in order to inject into a circular orbit. It completes its parking orbit insertion with 54% remaining of the propellant allocated to the circularization burn.

There are many unknowns for the MAV launch. This is the first time that a vehicle has lifted off from the surface of Mars, and the atmospheric model of Mars is not understood nearly as well as that of Earth. For this reason, we designed the propulsion system to give the vehicle as many options as possible. It targets a coasting altitude of 350 km with a velocity of 3.064 km/s, but it can miss this altitude by 100 km and velocity by 300 m/s and still have the propellant to reach its parking orbit using a Hohmann transfer.

To further mitigate the risk of launch unknowns, and to accommodate the RFP requirement that the MAV be able to land anywhere on Mars, our calculations assumed a polar launch. If the rover is at a latitude closer to the equator (as is expected), the planet's spin will provide extra velocity to the MAV, decreasing the required propellant.

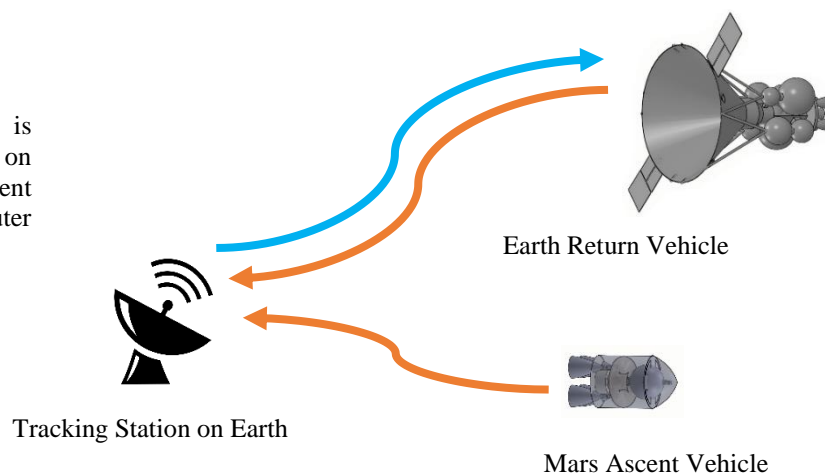
In addition, the MAV can launch with its nominal thrust of 10000 N cut in half to 5000 N and complete its parking orbit capture with 18% circularization propellant remaining. Even though it can reach orbit with this lower thrust, we chose to design for a maximum thrust of 10000 N in order to give the vehicle one engine out capability. For this worst case scenario, we cut the thrust in half because the opposite engine would have to throttle back in order to avoid a destabilizing moment.

At the end of the launch phase, the MAV is in a 350 km circular parking orbit. At this point it begins broadcasting its position to the ERV. Mission Control will work to establish a best guess approximation of the MAV's position to feed to the ERV's flight computer in preparation for rendezvous.

3. ERV/MAV Rendezvous

Since this mission operates with a large communication delay around Earth, the need exists for automatic rendezvous and docking. Our mission will utilize a version of the KURS system that is currently used on Soyuz spacecraft to automatically rendezvous and dock with the International Space Station. Other than initial positional data from ground control on Earth, the ERV will perform all calculations needed to arrive at the MAV. This allows for a completely autonomous rendezvous.

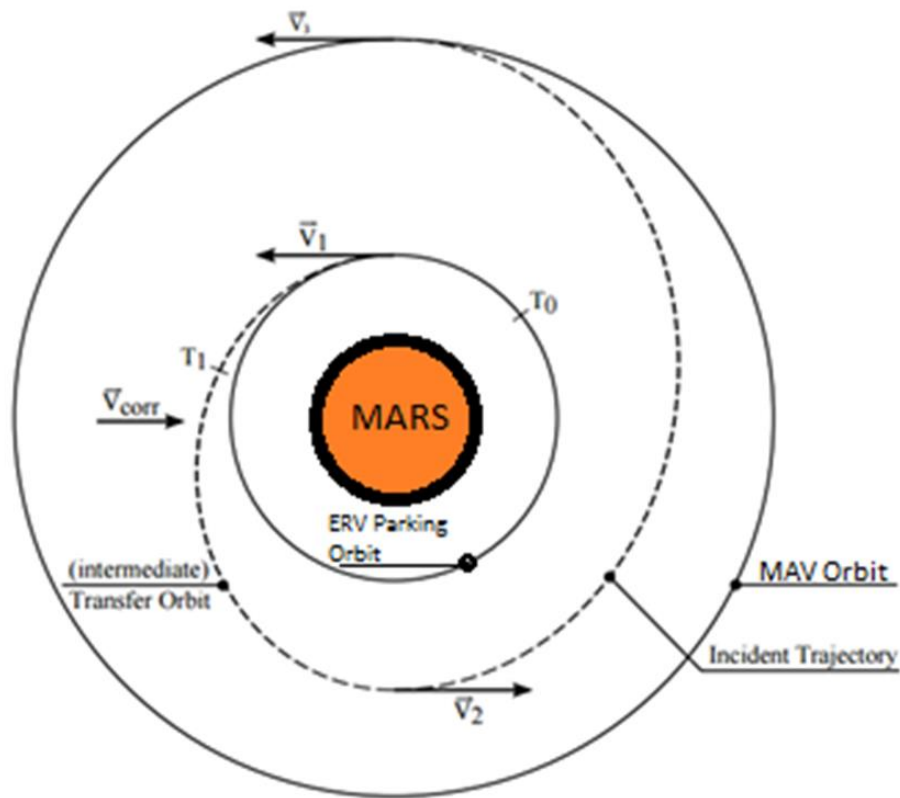
The ERV & MAV position is determined by a tracking station on Earth. This information is then sent to the ERV for the KURS computer to plan the rendezvous.



This phase of the mission begins once the MAV launches into orbit from Mars. Ground control will measure the position of the MAV and ERV. Upon receiving the measurements of each vehicle, the ERV will prepare its onboard

computer, propulsion system, and optical devices. The accelerometers and sensors used for measuring angular rate are activated. The Motion Control System (SUD) will fire thrusters in order to attain Local Vertical, Local Horizontal (LVLH) orientation. Once in LVLH, infrared horizon sensors are used to keep this orientation.

At time T_0 , the ERV onboard computer begins integrating the equations of motion in order to constantly attain the current state vector of itself (the ERV) and the MAV. Mission Control commands the first ΔV burn, placing the ERV into an intermediate transfer orbit. Our analysis shows that these rendezvous maneuvers are small, around 10 m/s or less.

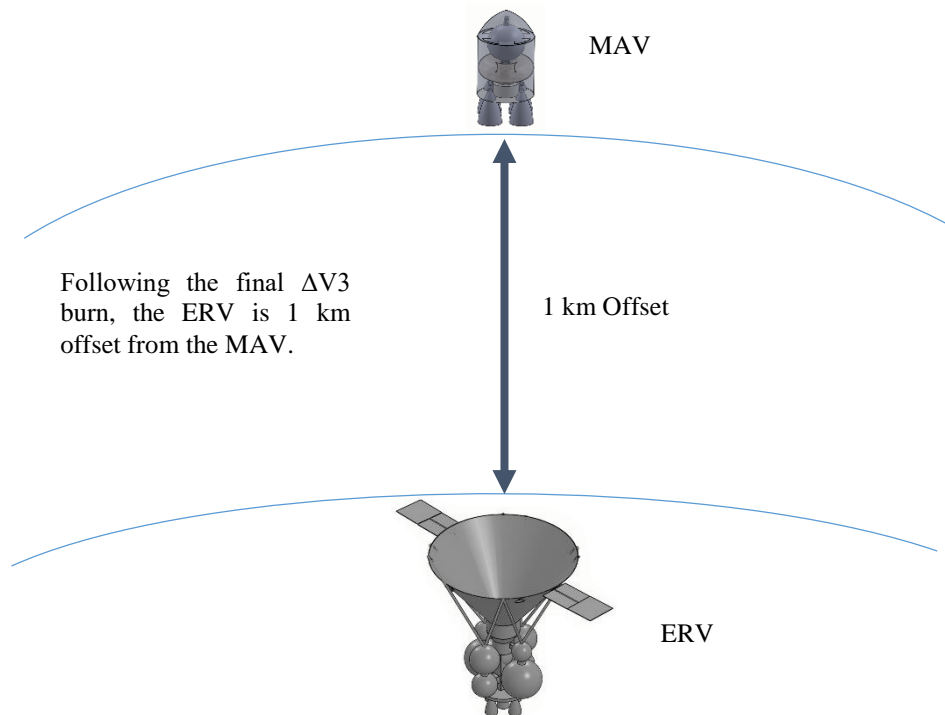


At time T_1 , between ΔV_1 and ΔV_2 , the KURS system is activated. A series of tests take place before the system switches into normal operating mode. The “omni-directional search” command is issued which alternately connects antenna AKR1 and AKR2 to the transmitter/receiver. This allows reception of a signal from the MAV KURS antennas as well as transmitting a homing beacon signal in any direction around the ERV.

As soon as a reliable signal is received by the ERV, the search mode is ended. The antenna which received the signal remains connected to the transmitter/receiver. Antenna 2AO, which measures heading and pitch angles, is now also activated and outputs its measurements to the motion control computer. When angular misalignment is less than

5°, “auto-tracking” command is issued. This deactivates the AKR and 2AO antennas and instead connects the ASF1 antenna to the transmitter/receiver. ASF1 is now measuring the heading and pitch angles and outputting this measurement to the motion control computer. The motion control computer uses the KURS data for attitude control for alignment in line of sight (LOS) orientation, instead of the original computer predictions from the state vector integration.

Measurement of relative distance and closing rate begins. Once a measurement is received, “lock-on” command is issued. KURS now is measuring relative distance, closing rate, heading angle, and pitch angle. The onboard computer starts a Kalman filter on the original state vector to match it with the measured values and correct the prediction. The ΔV for a correction burn is calculated and performed. The ΔV_2 burn is now determined. Prior to time T_2 , the ERV changes its orientation for the ΔV_2 burn. At time T_2 , the ΔV_2 burn is executed putting the ERV into an incident trajectory. The ERV then re-establishes LVLH and continues along the orbit path. This orbit will take the ERV to a 1 km offset target from the MAV for safety reasons. If no “lock-on” has been issued by this point, the KURS system will switch to the secondary system to search for MAV.

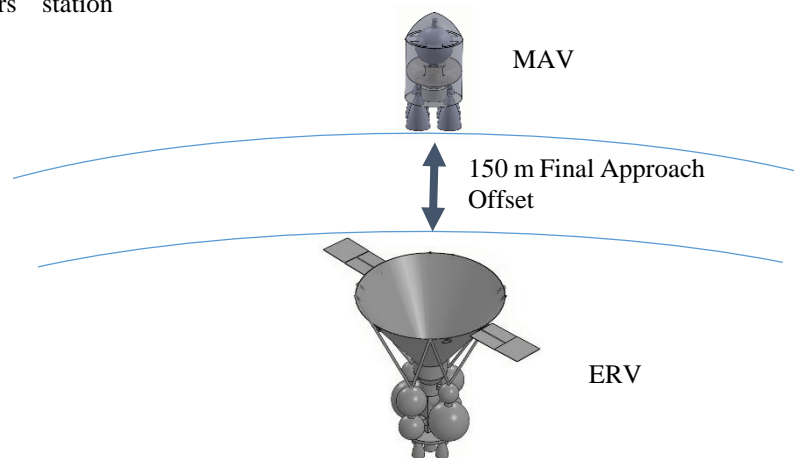


The computer calculates the firing time for the ΔV_3 burn. For this burn, the ERV needs to be rotated approximately 180° to cancel out the relative velocity between the ERV and MAV. At this point the ERV is at an offset target about 1 km from the MAV. The distance is decreased in steps until reaching about 300 m with a closing rate of less than 2

m/s and an angular misalignment of 0.3 degrees. The ERV flies around the MAV to align both vehicles axes. The relative distance continues to decrease until reaching about 150 m. The ERV targets the MAV antenna that the “lock-on” was generated by, and then enters station-keeping mode. The ERV now switches the antenna target from the homing beacon antenna on the MAV to the docking port antenna on the MAV. A new LOS orientation is acquired and a 2nd fly around conducted to align with the aft end of the MAV. After aligning at 150 m out with the docking port, the ERV enters station-keeping mode again. At this point, the ERV issues a command to the MAV to release the sample bucket locks. The MAV coil boom automatically extends with the sample bucket.

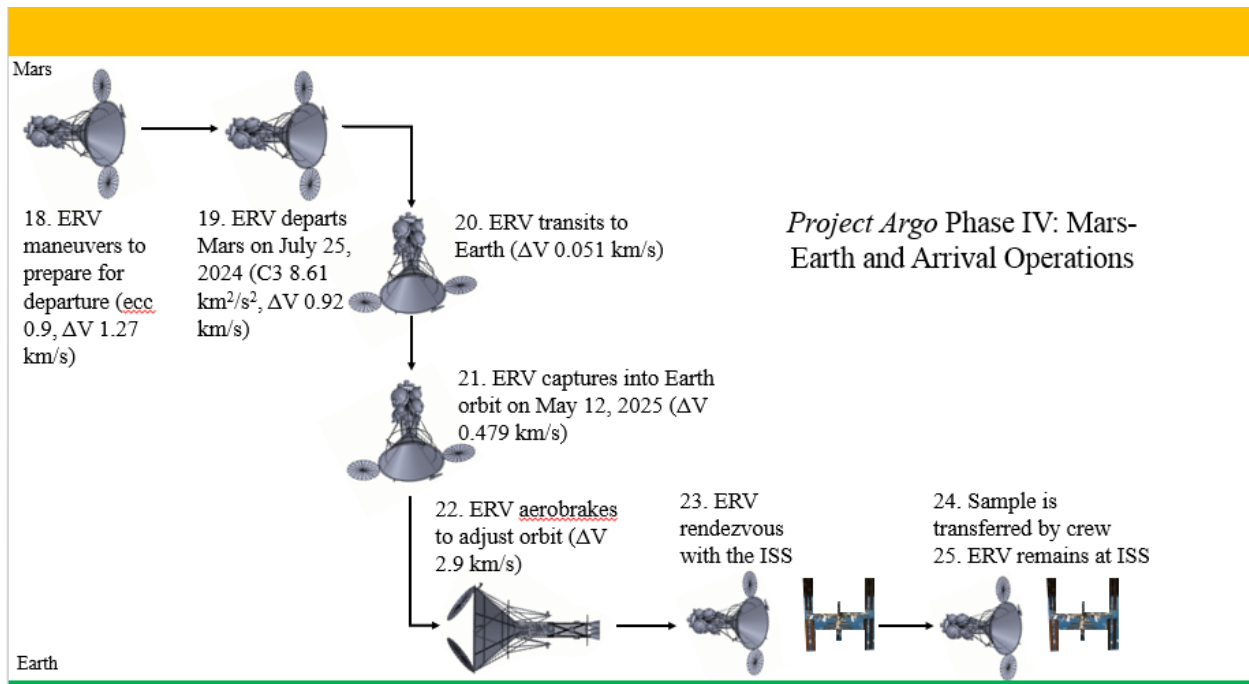
Upon approval from mission control, the ERV begins final approach to the MAV, maintaining LOS orientation relative to the docking port. At 50 m away, the 2AO antenna boom retracts on the ERV. The ERV assumes control of the MAV motion control system. Upon contact of the extended boom of the MAV with the ERV docking cone, the ERV motion control computer commands the MAV remotely to enter “touch-down mode”. In touch-down mode, the motion control system on the MAV immediately fires it’s thrusters to push the MAV into the ERV. Upon sample bucket capture in ERV socket, the “capture” command shuts down the KURS system and the motion control system enters “free-drift mode”. A guillotine system on the MAV severs the cable that secures the bucket to the boom, and the ERV maneuvers away from the MAV. This completes the automatic rendezvous and docking procedures.

The ERV approaches the MAV until reaching 150 m where it enters station keeping mode.



The MAV’s primary mission is now over. However, it has solar arrays for power and it should have propellant remaining from the design margins. It can communicate with Earth at a low data rate using its omnidirectional low-gain antenna. The opportunity therefore exists for follow-on missions to be designed.

D. ConOp Phase IV: Mars escape, transit, Earth capture and aerobraking, ISS rendezvous



1. ERV Maneuvers in Preparation for Escape

Prior to performing its escape burn, the ERV performs a series of maneuvers intended to minimize the required propellant. These can be started any time after the MAV has successfully delivered the sample cache, and must be completed in time for the Mars escape trajectory of July 2024.

The first step is to raise the apoapsis into an elliptical orbit with an eccentricity of 0.9. This increases the potential energy of the orbit and increases the velocity at periapsis, where the escape burn will be performed. The faster the spacecraft is traveling at periapsis, the smaller the difference with the required hyperbolic escape velocity.

Raising the apoapsis is accomplished in the same manner that the capture orbit was circularized, through a short burn at periapsis over several orbital periods. Propellant will first be pulled from any remaining in the second set of drop tanks. They will be jettisoned once depleted, at which point the third set of drop tanks takes over. Total Δv needed to achieve an eccentricity of 0.9 from a 350 km circular orbit is 1.279 km/s. The third set of drop tanks has the propellant to complete this on its own; any assistance provided by the propellant margin of the second set of drop tanks will allow the eccentricity to increase even more, thereby decreasing the Δv needed to escape.

Once its maximum eccentricity is reached, the ERV fires a short burst (~5 m/s) at apoapsis to lower its periapsis to 200 km, further increasing the orbit's energy. Lowering the periapsis to below 200 km would increase the velocity

even more, but the risk is encountering Mars atmosphere and inducing high drag loads in the middle of a high energy maneuver. Mission Control may desire to perform a number of passes to observe drag loads and determine whether the spacecraft can move lower. However, we designed to 200 km as a safe altitude: MSL registered the atmospheric interface at 125 km, and we calculate calculate the drag force for escape velocity at 200 km to be less than 1/3 N.

The third set of drop tanks will be jettisoned prior to the escape burn, thereby bringing the mass of the ERV to the lowest possible for the maneuver.

2. ERV Departs Mars

The ideal departure trajectory takes place on July 25, 2024. Using the fourth and final set of drop tanks, the ERV burns for approximately 4 minutes to accomplish a Δv of 920 m/s. Since the burn time is short, a 2% propellant increase over what is needed for an impulsive burn is assumed.

3. Mars-Earth Transit

The return journey takes 291 days. As with the outbound journey, three to five TCMs are expected and the spacecraft will be tracked by the DSN.

The final drop tanks are jettisoned during this time and will assume a heliocentric orbit. Planetary protection standards require that it can be shown that they will not encounter Earth or the Moon for at least 50 years. Analysis will be accomplished to show when the tanks should be jettisoned: most likely early in the transit.

4. Earth Capture

On May 9, 2025 the ERV encounters Earth's SOI with a hyperbolic excess velocity of 2.83 km/s. The final TCMs will have targeted the B-plane of Earth in order to achieve the inclination of the ISS orbit, 52 degrees (currently), and a periapsis of 900 km. This altitude avoids the zone of highest orbital debris, while mitigating the safety risk of trying to capture near to the ISS.^{ix} Capture takes place on May 12, 2025. The capture orbit eccentricity is 0.96, with a capture Δv of 0.479 km/s. Capturing into this highly elliptical orbit saves nearly 3 km/s Δv compared to a circular orbit.

The capture inclination of the ERV matches that of the ISS, but the orbit's right ascension will most likely not immediately be conducive to rendezvous. The higher altitude, highly elliptical orbit of the ERV will experience a much slower nodal progression than that of the ISS. The next phase of the mission is a sequence of aerobraking passes.

Before beginning this phase, the ERV will loiter in its initial orbit until the orbits have shifted such that the ERV and the ISS will be aligned by the time the ERV circularizes to its final orbit. Analysis and prediction of the final orbit will continue throughout aerobraking, and the ERV can loiter at elliptical orbits again as needed.

5. *Earth Aerobraking*

After capturing into Earth orbit, the next phase is the aerobraking sequence. Aerobraking is the process of slowing a spacecraft by allowing its periapsis to dip into the planet's atmosphere. This lowers the apoapsis and, over several periods, circularizes the orbit with minimal use of propellant.

Command and control during aerobraking is nontrivial. Since spacecraft tend not to be optimized for aerodynamics, the precise forces and moments produced by drag can be unpredictable. A spacecraft's attitude and trajectory must be quickly evaluated after each atmospheric pass, with corrective maneuvers applied at apoapsis. For Earth aerobraking, add to this the necessity of plotting a course through other satellites and orbital debris.



Plotting a course through the crowded skies above Earth is an acceptable risk because the ERV's position will be precisely tracked by Earth ground stations, and because communication will be in near real-time. Orbiting objects are precisely tracked; and while we think of it as crowded, collisions are rare and the ISS must perform evasive maneuvers only a couple of times per year. The ERV has sufficient propellant margin to maneuver as needed to shift its trajectory to avoid obstacles. Multiple vehicles have been successfully aerobraked around Mars with a significant margins of error for their known positions and a 25 minute round trip communication time delay.

At each pass, the ERV position will be precisely tracked and its current trajectory projected. Scenarios can be built up well before this stage of the mission in order to make decisions faster. At each apoapsis, a small maneuver is commanded to maintain the correct periapsis. According to our simulation, these maneuvers are small, less than 1 m/s. Communication will use the ERV's omnidirectional low-gain antenna.

The main consideration when planning an aerobraking profile is drag-induced heating. For the MRO, aerobraking peak heating was limited to about 200° C. If the ERV was required to stay at this limit, it would take over two years to complete aerobraking. Luckily, the ERV's design makes allowable peak heating much higher. The ERV skims the atmosphere engines first, which means that the engine nozzles and their mounting surface take the brunt of the heating.

Hydrazine and NTO burn with a combustion temperature over 3300 K. Engine nozzles are already able to withstand high temperatures and the engine mounting plate is the only place where insulation needs to be beefed up. At 108 km altitude and the velocity at the initial eccentricity, we calculate that the peak temperature will be 1500 K and that the drag experienced will be 709 N.

While the GMAT simulator does have the option of including an Earth atmospheric model above 100 km, it is not set up to readily accommodate an aerobraking simulation. Time and again, attempting to do so resulted in errors. So instead, we built a workaround by turning off the atmospheric model and creating a “thruster” for drag, which was tied to an empty propellant tank and did not decrement mass. We used a different simulation run to estimate how much time the ERV would spend at its lowest altitude – about 25 seconds. At first in our simulation we applied the expected drag force over the expected time. However, the only way to run the simulation was to begin applying the force at periapsis and continue it for 25 seconds, rather than applying the force equally on either side of periapsis. This would result in an asymmetrical drag profile. So instead, we multiplied the expected drag by the expected time and applied this total force for 1 second at periapsis. This more closely approximated the effect of drag as the ERV enters and exits the atmosphere.

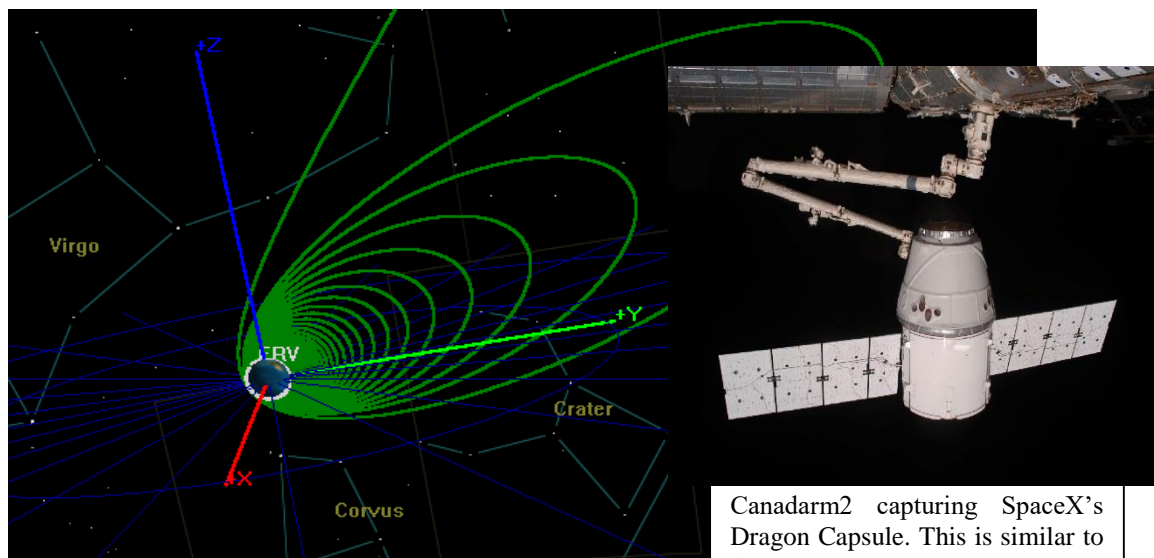


Figure: GMAT Earth Aerobraking Sim
 Periapsis altitude: ~ 100 km
 Duration: ~ 18 days
 Δv achieved: ~ 2.84 km/s

Canadarm2 capturing SpaceX's Dragon Capsule. This is similar to the ERV capture. (Courtesy: NASA)

In reality, the periapsis will be lowered incrementally in order to observe the effects of the thickening atmosphere. The altitude for each pass will be adjusted as needed. If excess heating is detected or if control is becoming unstable,

the solution is simply to raise the periapsis. The net result will be the same; all that matters is the final orbit achieved by aerobraking. Several lower-drag passes will take more time but accomplish the same thing as one high-drag pass. Using the given values we estimate that the aerobraking phase will take 30 passes over the course of 18 days.

The final goal is to reduce the Δv needed to achieve a 350 km circular orbit to less than 190 m/s. When the aerobraking profile is complete, the ERV fires its engines to raise its periapsis out of the atmosphere to 350 km, then fires again to circularize. As long as the Δv of the combined maneuvers is less than 190 m/s, the ERV will be left with a 30% propellant margin. We calculate that the final aerobrake pass to achieve this has a period of about 92 minutes – giving over 40 minutes to observe, track, and command corrections.

It should be noted that 350 km altitude is a value chosen for calculations, with the intent of being lower than that of the ISS in preparation for rendezvous. The actual final circular orbit will be chosen based on current conditions.

It should also be noted that we used GMAT to investigate the benefit of using chemical engines to use a bi-elliptical transfer after the capture, then use electric propulsion to spiral down to ISS altitude. As it turned out, a significant amount of chemical propellant was still needed, because the final orbit while using electric propulsion was elliptical and needed to be circularized. The mass savings were only slightly better than use of chemical propellant only. Therefore, aerobraking was selected instead.

6. *ERV/ISS Rendezvous*

For rendezvous and docking with the International Space Station (ISS) the same KURS automatic rendezvous used in Mars orbit by the ERV and MAV is also employed. The ERV will use an R-bar approach – approaching from below – in order to minimize thruster exhaust plume contamination of the ISS solar arrays. The ERV maintains the same active role of seeking the ISS around Earth as it did when seeking the MAV around Mars. No new hardware is needed for this final phase of the mission.

Following the aerobraking orbits, the KURS system will be turned on once the ERV orbit has been circularized at about 350 km. Ground control will be tracking vehicle position and uploading this data to the ERV. The rendezvous phase will perform the same as the ERV to the MAV. Variations in the KURS procedure vary slightly in the final transfer orbit, primarily being that the offset distance after the ΔV_3 will be much further from the ISS for safety reasons. Normally a Soyuz spacecraft would have an offset of 1km when approaching the station, but since the ERV is unmanned the offset will be increased to 5km. This is to mitigate the risk of a crash with the ISS. The final approach

of the ERV from the offset distance to the station will follow the same procedural steps to get closer to the ISS. Once the ERV is in range of the Canadarm 2, it will enter station keeping mode. KURS is not used for docking and will be shut down at this point. The Canadarm 2 will be attaching and keeping the ERV safely attached so that an astronaut can perform an extra-vehicular activity (EVA) to retrieve the sample canister.

7. Transfer of the Mars Sample to ISS

The ERV is equipped with the NASA standard Flight Releasable Grapple Fixture (FRGF), which interfaces with the Canadarm2. The arm captures the ERV and the ERV enters a safe mode with engines and ACS disabled.

Collection of the sample will be performed by an astronaut performing an EVA. The crew will determine the best way to accomplish this, but we expect that the Canadarm will bring the ERV close to an open airlock and that a tethered astronaut will float into the capture cone.

The sample cache container and its receptacle in the ERV is designed to make collection by an astronaut as easy as possible. The container has a handle large enough to be easily gripped by an astronaut's gloved hand. When the container is secured in the ERV capture cone, this handle is protruding, easily accessible. The latch that locks it in place is external and is disengaged by a slide that is easily operated. There are no hooks, latches, or wire connections inside the capture cone.

8. ERV disposal

Planetary protection concern was the design driver for determining how to dispose of the ERV. The vehicle carried a sample of Mars surface and came into contact with a Mars surface vehicle. Planetary protection protocol states that a vehicle returning from Earth either not encounter the planet, or be subject to the same sterilization standards as a planetary lander.

De-orbiting into Earth's atmosphere would therefore be an option only if the vehicle was designed to sterilize itself. Considering the risk and design complication involved with guaranteeing sterilization, we decided that this course was not worth pursuing.

Another option would be for the ERV to depart Earth orbit into an appropriate heliocentric orbit, or even better to de-orbit into the sun. However, the propellant cost for this maneuver is exorbitantly high and carrying it would have rendered the entire mission impossible. Launching a rocket upper stage, rendezvousing it with the ISS, and docking it

onto the ERV is a possibility; but the cost, mission complication, and risk again indicated that the idea was not worth pursuing.

The ERV could be left in Earth orbit, but this is undesirable since it would be one more piece of orbital debris, and there would still be the risk that collision with other debris could send it into a decaying orbit.

Luckily, another option presents itself. The ERV remains with the ISS. Its propulsion system is disabled and it is attached to an unoccupied section of truss. This way there is no risk of it de-orbiting and it is not added to the debris field. Its mass and volume are small relative to the station (less than 1/3 the mass of the Canadarm) and will not significantly affect its maneuvering capability. This solution has the added benefit that it allows the ERV to be studied to learn the effects of deep space travel. Scientists would also have the option of devising follow on experiments using the ERV's instruments.

If circumstances ever required that the ERV be detached and disposed of into Earth's atmosphere, it could be decontaminated by ISS crew. The de-orbit maneuver could be performed using small strap-on solid rockets.

1.1. Navigation, communication – add paragraph to Earth-Mars cruise

Vehicle Design

A. Earth Return Vehicle (ERV)

1. *ERV Design Driver: Propulsion Subsystem Design Logic*

The design of the ERV went through many evolutions. The final proposed design is the result of careful reasoning centered around the propulsion system. The propulsion system was the vehicle's major design driver - essentially the ERV is a just a pickup truck that travels to A and takes a single load back to B.

Early in the design process we ran into an issue while using the rocket equation to determine the required propellant mass. To illustrate the issue with an example:

Say that we have a payload mass of 10 kg. Our initial strategy was to assume that the payload was a certain percentage of the final dry mass – say, 10%. This gives an $m_{\text{final}} = 100$ kg. We use a specific impulse of 320 seconds for a hydrazine/NTO system, and a total mission Δv of 5000 m/s. Plugging this into the rocket equation

$$m_{propellant} = m_{final} \left(\exp \left(\frac{\Delta V}{g_c I_{sp}} \right) - 1 \right) = 392 \text{ kg}$$

However, best practice says to add a propellant margin. For example we add 20%, so our total propellant becomes 470 kg.

We then look at the tanks needed to carry 470 kg of propellant. We assumed titanium tanks and calculated the tank mass based on the internal volume required to carry the required propellant and the thickness required to maintain the required pressure without rupturing (including a factor of safety; tank mass estimates will be discussed in more detail later). We also include the mass of the diaphragm for a blowdown system, an estimate for the added mass of tank welds and structural attachments, and finally a mass estimate for a helium pressurant tank for each of the main tanks. The total mass is just under 42 kg. This still does not include plumbing and engine masses for the propulsion subsystem.

Now we go back to our expected spacecraft final mass of 100 kg. Subtracting the assumed payload mass and the tank masses just calculated, we have 48 kg remaining dry mass– not much for the rest of the spacecraft subsystems, but perhaps doable.

However, the key thing we realized is that this logic was incorrect. We added a propellant margin of about 80 kg – and the assumption is that this propellant will not be used. This means that in the rocket equation it must be part of m_{final} . So our final mass that was supposed to 100 kg already has 10 kg of payload, 42 kg of tank, and 80 kg of extra propellant. Obviously, it does not add up. Even using advanced, less massive composite-reinforced tanks would not help.

Our initial solution was to base the estimate of our final mass on the propulsion subsystem rather than the payload. Our goal was to have the propellant tanks be 35% of the final dry mass – but this final dry mass was not the same as the final mass in the rocket equation, which includes the additional propellant margin. In other words, we looked for

$$m_{final} = m_{dry} + m_{prop_margin} = \frac{m_{tanks}}{35\%} + m_{prop_margin}$$

This was an iterative process since increasing the final mass increased the required propellant non-linearly, which in turn increased both the mass of the tanks and the extra propellant; but it an easy process to perform using the MS Excel Goal Seek function.

However, it immediately became obvious that we had a problem. Early in the design process before we incorporated Earth aero braking, the total mission Δv was 7.89 km/s. We did not want to design our mission to depend

on an unproven launch vehicle. This meant staying within the launch capability of a Delta IV Heavy for a C3 of 13.13 km²/s², which was calculated to be 7992 kg after subtracting the mass of the PAF. The only way to accomplish this was by having zero propellant margin and allowing the propellant tank mass to be 83% of the total dry mass, with only 113 kg budgeted for all other components. Obviously, this was not feasible. Even by abandoning the Delta IV Heavy we had little luck. The mighty (and distantly future) SLS Block 2B could in theory launch 43000 kg with our trajectory's C3^x. Even with this limit, we found that we could manage only a 1% propellant margin and a dry mass of about 3500 kg, 88% of which was propellant tank.

Clearly we needed to reduce propellant mass. The first choice was to copy launch vehicles and utilize staging. But we quickly realized that there was no need for the ERV to have a separate set of engines for each stage. Thus the idea of drop tanks was born. The ERV would use the same main engine cluster for the entire mission but would jettison tanks as they became depleted. The mission maneuvers were separated according to when it would make sense to jettison tanks, and the tanks were sized accordingly.

(NOTE: Aerobraking around Earth reduced the total Δv by nearly 3 km/s, a significant savings. By the time aerobraking was incorporated into the mission, the ERV design with drop tanks had already been selected as a baseline. Revisiting the idea of using only fixed propellant tanks, we found that with a total propellant margin of 10% and allowing the tank mass to be 52% of the total dry mass, we could have an ERV with approximately the same wet mass as our final design. However, it would have approximately 100 kg less dry mass to allocate to other subsystems. In addition, we found that by combining aerobraking with drop tanks, the ERV mass was reduced enough to allow for capturing into Mars orbit with the MAV/MLP stowed. This opened up a tremendous opportunity to remove the risks associated with the MAV/MLP entering hyperbolically. Comparing the risk associated with drop tanks with the benefits of more dry mass, better propellant margin allocation, and of capturing the MAV/MLP into Mars orbit, we elected to stay with the baselined drop tank version.)

2. *ERV Drop Tanks*

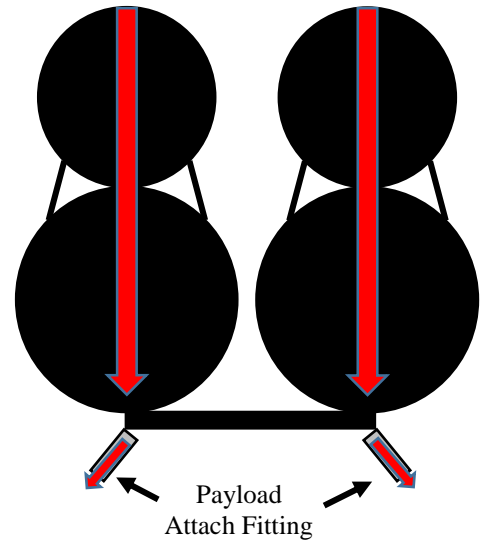
Utilizing drop tanks made it possible for the Project Argo to launch on a single launch vehicle, and to capture both the ERV and the MAV/MLP into Mars orbit. This resulted in cost savings and in reduced risk for multiple parts of the mission. However, incorporating drop tanks into the design was nontrivial.

The first challenge was to structurally support the relatively large propellant mass during Earth launch. To do this, we evaluated several concepts until we agreed upon the idea of stacking the tanks such that they act their own primary structure. The important feature of this concept is that the center of gravity of each of the tanks must be placed in line above the Payload Attach Fitting interface. This makes the load path pass through the tanks directly to the PAF without creating a bending moment on the engine mount plate.

The mission maneuvers cause less acceleration than Earth launch but are applied in the same direction. Therefore it was desirable to have the top tanks be those jettisoned first, so that in subsequent maneuvers the remaining tanks would still be supported by the engine mounting plate rather than putting a

shear load on the central column. However, the first drop tank to go is the largest, and stacking it on top of one of the smaller tanks would cause it to fail during Earth launch. We therefore arranged the tanks as shown. The first set is placed directly on the engine plate, and the smallest, fourth set of tanks is stacked on top so that the shear loading is minimized. The second set is stacked on top of the third set so that shear loading is avoided.

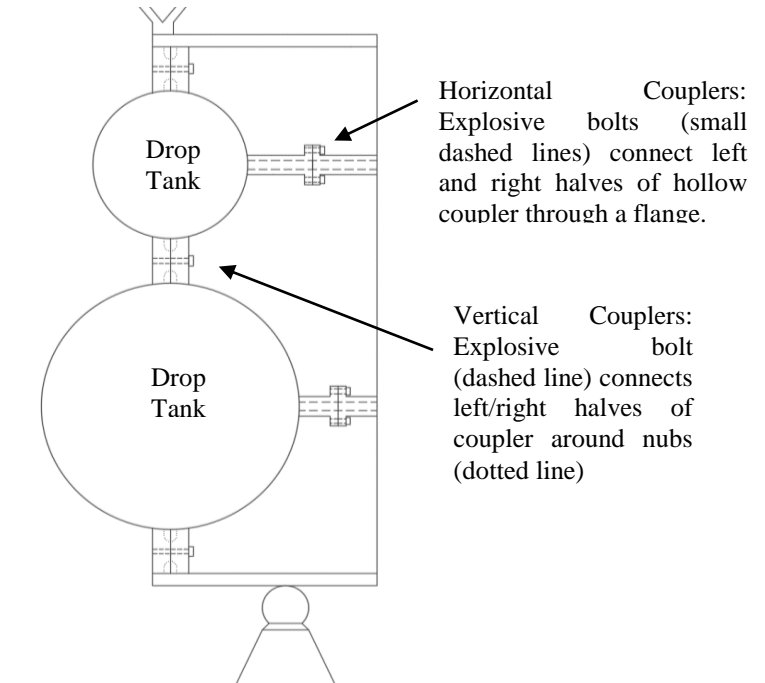
Appendix II shows a simplified schematic of the propulsion system. The purpose of this schematic is not to show every valve and redundancy, but rather to show the system for isolating, separating, and activating each subsequent set of tanks. Normally-open pyro valves are closed in order to isolate the tanks from the system, and to seal the tanks prior to separation. Normally-closed pyro valves are opened in order to switch to the next set of tanks. The schematic shows the firing order. Plumbing for the drop tank system is complex; however, it is simplified by using a standardized valve pattern that is duplicated for each tank.



Drop Tank Structural Loading
The load path passes through the centerline of the drop tanks and is transferred to the Payload Attach Fitting.

3. ERV Drop Tank Separation

The drop tanks on the Earth Return Vehicle are secured in place via vertical and horizontal couplers, as seen in the figure. The vertical couplers are each divided into two halves. The halves of the vertical couplers connect around nubs located on the drop tanks and ERV frame. The halves of the vertical couplers connect around nubs located on the drop tanks and ERV frame. The vertical couplers, when connected together, prevent the tanks from shifting out of the load path. A pyro bolt connects the two halves together. When the bolt is broken, the two halves will separate outwards causing the tanks to have clearance at that particular joint.



The horizontal couplers consist of a flange with two pyro bolts installed through it. The flange is attached to a hollow tube with one side connected to the drop tank and the other side connected to the ERV center support column. When the two bolts are broken, the two sides of the coupler separate allowing springs on the inside of the flange to push the drop tank away from the vehicle. The hollow area on the horizontal couplers allow fuel lines to pass from the tank to the ERV frame.

Project Argo will utilize a single Delta IV Heavy for Earth launch. Image source: <http://www.americaspace.com/?p=22180>

In order to eject a drop tank, the vertical couplers on the bottom and top of the tank are separated. Since there is no structural load while the tanks are being ejected, the vertical couplers are no longer necessary to keep the tank attached. After the vertical couplers are removed from the tank, the tank will be attached and supported only by the horizontal coupler. At this point a pyro valve located on the ERV halts the flow of fuel from tank to the ERV and halts the flow helium from the ERV to the tank. At the same time a pyro valve also closes on the fuel tank preventing fuel and helium from continuing to enter the fuel lines crossing the horizontal coupler. The fuel lines running through the horizontal coupler are then cut with a guillotine. The pyro bolts in the horizontal coupler are broken and the springs inside the coupler push the drop tank away from the vehicle.

In order to mitigate risk of a collision of the fuel tank with the vehicle during subsequent maneuvers, the fuel tank distance needs to be at least 10 meters away from the ERV before an ACS or main engine burn is commanded. In order to verify the clearance a small, low powered RF transmitter installed on the tank will activate at the time of jettison. This transmitter will have a battery with enough charge to operate continuously for 1 hour. Each tank will have its own transmit frequency assigned so that a receiver on the MAV can tell them apart and measure signal power. By knowing the transmitter power and comparing to the measured power by the receiver, distance can be verified by the computer.

4. ERV Propellant Margins

Each pair of drop tanks is sized for a particular set of maneuvers. The first set is used to capture into the initial elliptical orbit around Mars. The second set is used to circularize the orbit to set up for rendezvous with the MAV. The third set is used in preparation for Mars departure by increasing the orbit eccentricity and lowering the periapsis. The fourth set is used to escape from Mars orbit.

Best practice requires adding a propellant margin to the calculated required value in order to account for propellant loading errors, engine degradation, leaks, and other risk factors. However, propellant costs mass, which needs to be kept to a minimum, particularly for a mission like Project Argo that has several large maneuvers.

Fortunately the solution to this dilemma was found in the nature of the drop tanks themselves. The tanks are sized to perform a certain maneuver; but, once that maneuver was completed, there is no reason to jettison the tanks immediately. Instead, the extra propellant can be used to assist with the next set of maneuvers. In this way the propellant margin cascades from tank to tank. Therefore, even though each set of tanks has the capability to perform its designated maneuver on its own, there was no need to provide each with a large propellant margin, since that margin already existed on the vehicle.

<i>Cumulative Propellant Margins for Drop Tanks</i>	1st Set Drop Tanks	2nd Set Drop Tanks	3rd Set Drop Tanks	4th Set Drop Tanks
Total Tank Propellant (kg)	1969	849	515	254
Δv required (m/s)	710	1353	1284	920
Δv after previous stage extra Δv	n/a	1034	901	429
Stage Initial Mass (kg)	7286	2411	1485	919
Propellant used (kg)	1481	654	358	113
Propellant remaining (kg)	488	194	157	141
Individual propellant margin	25%	3%	5%	9%
Cascaded propellant margin	25%	23%	31%	55%
Extra Δv capability (m/s)	319	383	491	627

Drop Tank Propellant Margin Cascade

The propellant margin of each set of tanks is carried over into the subsequent set of maneuvers, such that the actual propellant margin is higher than that of the individual tanks.

5. ERV Propellant Type

As previously stated, the propulsion system was the main design driver for the ERV. With so many maneuvers, a high performance system was needed in order to keep mass to a minimum. Electric propulsion was deemed to not be feasible due to the time it would take – in order to be in position to meet the rover by the required date, chemical engines would be needed anyway in order to capture into orbit. A monopropellant system would be relatively simple, but would lack the necessary performance. For this reason, bi-propellant was selected. Cryogenics were deemed not feasible due to high risk of leaks. The two most promising candidates were hydrazine/NTO and MMH/NTO. Both have been used on multiple deep space missions.

In our trade study between hydrazine vs MMH, the factors we considered were 1) the overall wet mass of the vehicle under each system; 2) the propellant freezing point; 3) potential for dual mode; 4) shock risk; and 5) which had a tank size closer to that of the NTO tank it would be paired with.

A dual mode propulsion system is desirable because it offers the option for the ACS to utilize the propellant margin in the drop tanks. It also gives the main engines flexibility to use monopropellant if a smaller thrust maneuver is needed, and some redundancy if there is a problem with the oxidizer system.

Hydrazine and MMH have similar performance when combined with NTO, but the hydrazine system holds a slight edge. Actual specific impulse varies by engine; for our calculations we assumed 320 seconds for hydrazine/NTO vs 315.5 seconds for MMH/NTO. (For comparison, the Aerojet Rocketdyne R-42DM 890 M Hydrazine/NTO engine has

a rated thrust specific impulse of 327 seconds; the R-42 890 N MMH/NTO engine has an I_{sp} of 303 seconds.^{xi)} We found that using a hydrazine system decreased the overall wet mass of the ERV by 100 kg.

As to the other considerations: MMH was specifically designed for low temperatures and has a freezing point of -52 C, so it clearly beat out hydrazine, which freezes at +2 C. Hydrazine readily breaks down as a monopropellant and is commonly used in dual mode systems. Shock risk is higher for hydrazine, but we found a study that indicated that the risk is not particularly high after all.^{xii} Finally, two paired tanks of hydrazine and NTO are closer in size than those of MMH and NTO. As will be discussed later, the NTO tank was sized up to be equal to that of the fuel tank; since the hydrazine and NTO tanks were already close, this meant less wasted mass.

As can be seen in the figure of merit analysis, the hydrazine/NTO system beat out MMH/NTO. The final question was whether the thermal subsystem mass required to keep the hydrazine from freezing would be less than the 100 kg it saved. Our thermal analysis indicated that it would be.

		MMH		Hydrazine	
	Weight		Score	Score	
Overall Wet Mass (kg)	5	3934	0	1	3833
Freezing point (°C)	4	-52	1	0	2
Potential for dual mode	3	no	0	1	yes
Shock risk	2	lower	1	0	higher
Fuel/Ox tank size	1	less close	0	1	more close
			6	9	

ERV Propellant Figure of Merit Analysis

6. ERV Main Engines

We chose a five engine configuration for the ERV main propulsion system. This allowed one engine to be placed along the centerline of the vehicle, which was useful from an attitude control standpoint. It also allowed us to size the engines for one-engine-out capability without making the individual engines too large or too small.

We sized our main engines based on our highest mass flow maneuver, Mars capture. Though the Δv is small relative to other maneuvers, the vehicle is at its most massive. As a reference we used the capture burn of MRO, since it captured into a similar orbit. MRO's capture burn lasted 25 minutes.^{xiii} We also had the goal of using a COTS engine.

We selected the Aerojet Rocketdyne AMBR 623 N Dual Mode High Performance engine. With five engines, this gives a nominal thrust of 3115 N and a Mars capture burn time of approximately 28 minutes. With one engine out, the approximate burn time is 36 minutes. This is within each engine's rated steady state firing time of 45 minutes.^{xiv}

Another contender was the R-42DM with 890 N of thrust per engine. With 4450 N total thrust, it could have performed the capture burn in 20 minutes. However, the higher thrust would have been overkill for all subsequent maneuvers, so the AMBR was chosen for its lower mass and higher performance: 5.4 kg per engine and 333 seconds specific impulse vs. 7.3 kg and 327 seconds for the R-42DM.

7. ERV Propellant Tank Design

Propellant tanks for the ERV are aluminum and spherical with a gas blowdown system. We designed them with an internal pressure of 3300 kPa and 10% ullage. Thickness was calculated using a factor of safety of 2 times the material yield stress. Extra mass was added for a girth weld and a penetration weld, based on tank radius and thickness. Mass of the blowdown system diaphragm was calculated based on the propellant volume + ullage for each tank. An extra 10% of the tank membrane + diaphragm mass was added to account for structural attachments and plumbing. The volume and mass of a helium pressurant tank was calculated for each propellant tanks. We chose to give each tank of propellant its own pressurant system. This removed the risk of the hypergolic propellants leaking into the pressurant system, mixing, and igniting there.

We investigated using titanium tanks vs. aluminum. Though it is more dense, titanium's has a higher yield strength allows the tank walls to be made thinner, resulting in less mass. On the other hand, titanium is significantly more expensive than aluminum, both in material and machining cost. We also found the thermal properties of titanium would require much more electrical heater power to maintain the propellant temperature. Finally, there is the small but noteworthy observation that aluminum has better thermal conductivity and a lower melting point, which means that if a tank were to deorbit at Mars, there is a higher chance that it will burn up before contacting the surface; or at least, heat enough to kill any remaining Earth-originating microbes.

Ultimately we chose to design aluminum tanks. We considered cost to be an important factor due to the number and size of the tanks. For mass, we found that if we used an aluminum alloy with a high yield stress (503 MPa) we could cut the tank mass by more than half compared to aircraft aluminum. We decided to make the option of titanium tanks an 'ace in the hole' for future design iterations: using aluminum tanks saves costs; but if mass needs to be shaved then the switch can be made to titanium tanks with several layers of insulation.

8. ERV Instrumentation/Rendezvous Equipment

The KURS rendezvous system was described in detail previously. Instrument mass breakdown is shown in the table.

SUD is the vehicle computer that keeps the vehicle oriented the correct way and commands the ACS system and the engines to fire. The instruments necessary to provide input to the SUD computer are:

- Accelerometers
- Angular Rate Sensors
- Star Tracker
- Infrared Sensors

Cameras are installed for verification of alignment prior to docking of MAV and ERV.

9. ERV Thermal Control Subsystem

The ERV faces the widest range of thermal environments. It also has the task of holding propellant drop tanks. The propellant, hydrazine, has a relatively small range of storage temperatures. Analysis was done for worse case scenarios of extreme hot and cold phases throughout the mission. The ERV is a cold-bias design. The ERV is almost always kept at the lower range of the component temperature limits. Special care was taken for components with smaller temperature ranges. This includes propellant tanks (-2°C to 50°C) and batteries (5°C-40°C).

Payload		Budget	50.18
Item	Count	Mass (kg)	
Sample cache bucket - on return only (9 kg)	1		
Onboard Computer, KURS	1	30	
Cameras	2	0.21	
Accelerometers	3	0.09	
Angular Rate Sensors	3	0.75	
AKR1 antenna	1	0.5	
AKR2 antenna	1	0.5	
2AO antenna	1	0.5	
ASF1 antenna	1	3.5	
Infrared sensors (measure horizon)	4	0.033	
Star tracker	2	0.35	
Flight Releasable Grapple Fixture (FRGF)	1	2	
Total			40.772
Remaining budget			9.41

Components	Temperature Ranges °C	
	Operational	Survival
Batteries, Li-ion	5 to 40	-20 to 45
Solar Arrays	-150 to 110	200 to 130
Propellant, Hydrazine	7 to 35	
Structures	-100 to 100	
Antennas	-100 to 100	-120 to 120
Motors	-40 to 40	-50 to 155
Gyros/IMUs	0 to 40	-40 to 80
C&DS Box Base Plates	-20 to 60	-40 to 75

Hardware Thermal Limits

For our preliminary thermal analysis, we assumed thermal equilibrium of a spherical geometry configuration. A diameter of 7.2 m gave an equivalent surface area to the ERVs large surface area of 63 m². This simplified view factor calculations but still gives a close estimate of the heat transfer between the vehicle and the space environment.

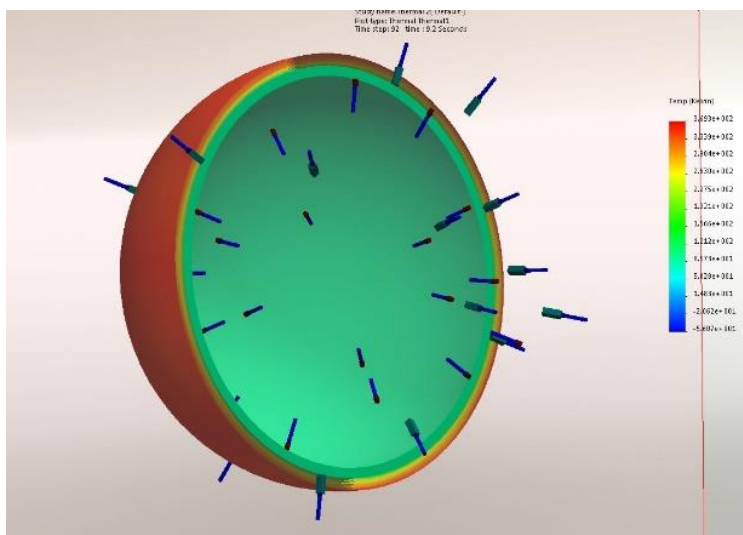
A maximum internal power dissipation of 150 watts will determine the maximum vehicle temperature. This happens during rendezvous with the ERV. 85 watts of this heat is dissipated from the ACS. Changing the vehicle surface properties was an iterative process to find acceptable surface

temperature ranges for all conditions. 25 μm Aluminized Kapton MLI was the best material. This also has a main purpose of insulating the interior of the vehicle that houses the electronics and other systems.

The hottest temperatures of around 31°C occur in earth orbit when rendezvousing with ISS. This takes into account degradation of 3 years to the surface. The coldest temperatures are when the vehicle is farthest from the sun in Mars orbit. When the planet blocks the sunlight, temperature drops to -108°C . When receiving sunlight and infrared radiation from the planet, the equilibrium temperature is -13°C . In both cases, a combination of heaters are thermostatically controlled to add up to 50 watts of heat. The maximum surface temperatures (33°C in Earth orbit) on solar arrays are well below their operating limits.

When considering a drop tank design for the ERV, safe propellant temperatures were a key factor in the evaluations. Since they will be mounted on the perimeter of the vehicle to be jettisoned, enclosing them in an insulated box is not the best approach. The trade study of using aluminum or titanium tanks also depended on the surface properties of these materials and the equilibrium temperatures. Analysis was done on Mars orbit conditions. Bare aluminum yielded surface temperatures of 125°C near Mars, higher than what is safe. Titanium yielded surface tank temperatures of 22°C which gives a 24° margin and 90°C near Earth. From our trade study, aluminum was chosen for other reasons (such as cost and manufacturing) as long as the temperature could be controlled through insulation and heaters.

The wide range of temperatures needed to be controlled, therefore, it was decided it is best to wrap them in Multilayer Insulation. We did analysis comparing this to fiberglass insulation. The fiberglass insulation would lose 14 watts per square meter of thermal heat near mars orbit. Linear steady state heat conduction was used in the analysis



for this using fiberglass 8cm thick and a surface temperature of the conditions in Mars orbit. The analysis for using MLI neglects conduction and uses pure radiation heat transfer through each layer. Ideally, 25 layers of a combination of aluminized Mylar and Kapton will keep the maximum heat transfer to 2 watts per tank. This is using MSFC laboratory data where effective emissivity of

0.001 are achieved (Brown, AIAA). Based on actual previous missions, specifically the Viking mission, an effective emissivity of 0.004 is assumed.

The heat absorbed per tank is about 4 watts beyond the lower propellant temperature margin. A 5 watt heater is used on each tank during solar eclipse in Mars orbit. This heater power was modeled using SolidWorks Simulation add on. The time dependent study modeled the effectiveness of using 25 μ m layers of vapor deposited aluminum on sheets of Mylar® and Kapton® (polyamide). The software iteratively calculated radiation view factors for each of the layers and the outside layer to space. An initial temperature of 5°C was assumed. The figure shows the visual results of this analysis.

10. ERV Integrated Vehicle Health Monitoring System

Since a mission to Mars is a very large distance from Earth, there are many failure risks that need to be solved remotely. As such, an integrated vehicle health monitoring (IVHM) system is very important. IVHM gives the opportunity to use diagnostics and prognostics to fix faults before they become mission threatening issue.

The IVHM control system will be divided into two modes. First, the system will automatically take corrective action on certain regulatory actions. Second, the system will collect and analyze sensor data for mission control to determine if an action needs to be taken.

Systems needing automatic regulation reside in the propulsion, thermal, and power subsystems. The control system will take care of this task without human intervention. In the propulsion system, for example, the fuel tank pressure needs to be kept in an operating range. This is accomplished by checking a pressure transducer attached to the tank and determining current tank pressure. If the pressure is too low, the helium pressurant valve needs to be opened to allow additional helium gas to flow into the fuel tank.

In the thermal system, tank temperatures, battery temperatures, and instrument temperatures need to be regulated. The system will analyze the temperature sensor to determine current temperatures and compare it to the needed operating range. If the temperature is out of range a command will be sent to turn on the heaters if it is too hot, or if it is too cold a command to turn off heaters will be sent.

In the power systems, IVHM will monitor and control battery charging, monitor current draw throughout electrical system, regulate battery temperatures, monitor solar output, and adjust solar panel pointing direction. An example of IVHM operation in the power system would be checking solar output and comparing to estimated output data based on range from the Sun and angle. In the event of a lower than expected output, IVHM could verify

spacecraft orientation by analyzing Sun and star sensor. The solar panels could be rotated to adjust for certain misalignments with the sun to bring production to an acceptable level. If an acceptable level is still unable to be reached, mission control can be notified. Mission control can request images of panel position to check against position sensor data.

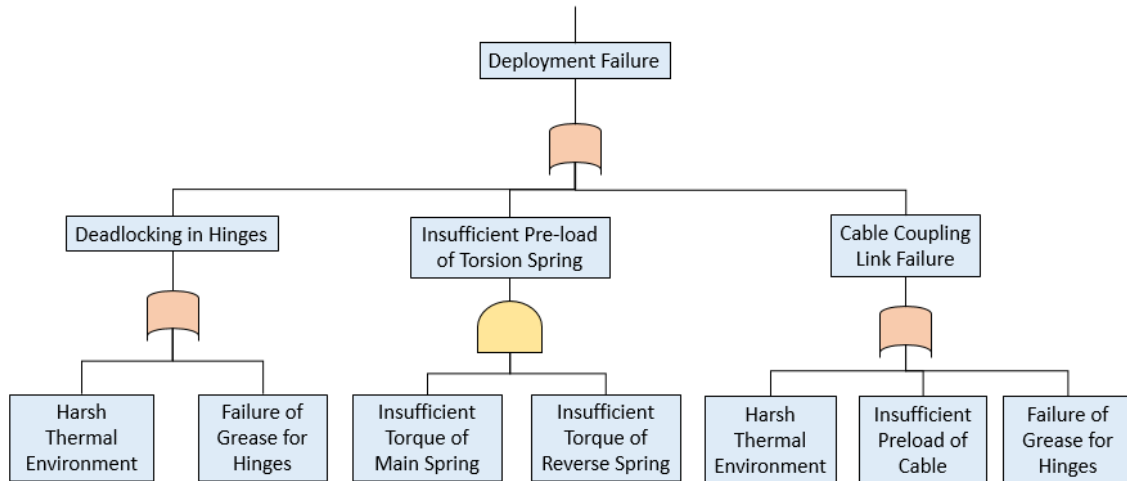
In addition, IVHM will check mechanical positions of moving components.

The Earth Return Vehicle has a few moving components that need to be monitored. There are multiple antennas mounted on arms that need to deploy. Some of the antennas are for the KURS automatic rendezvous and docking system; another antenna is for communication. If the KURS antennas aren't deployed correctly rendezvous and docking will be very difficult or impossible. If the communication antenna can't deploy the mission will be forced to utilize the low gain antennas which only allow a very low speed of communication. IVHM will check position sensors attached to each antenna boom to determine the alignment angle.

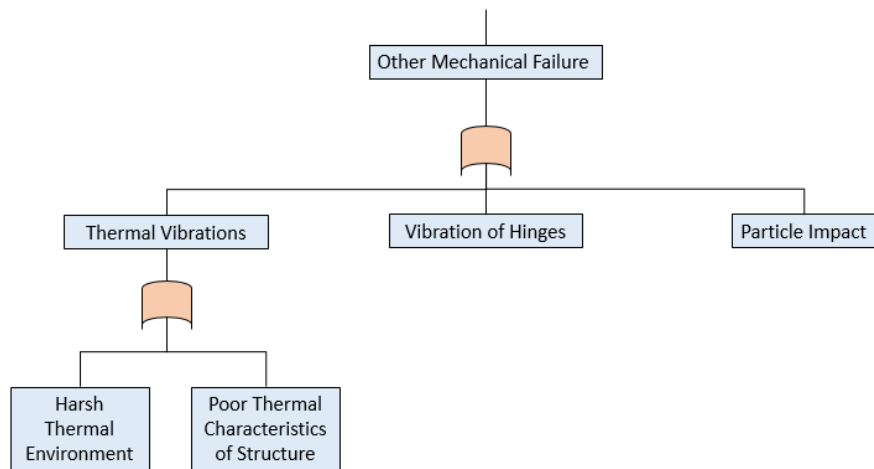
Apart from antennas, the most important part of the ERV is the sample bucket docking port. When the MAV extends its coil to place the sample bucket into the ERV, the docking port needs to successfully allow the sample bucket to enter and lock. The locking mechanism is simple and spring operated. When the sample bucket is pushed into place it will be locked in by the locking mechanism. A few small cameras will be used to verify the locks are fully in place before the MAV detaches from the sample bucket. If a complete lock is not detected, the ERV will command the MAV to fire its thrusters to push the sample bucket further into the docking port. The same process for checking the locks will again be performed. When a successful lock is verified, the MAV will detach from the sample bucket.

11. IVHM: Power System Fault Tree Analysis

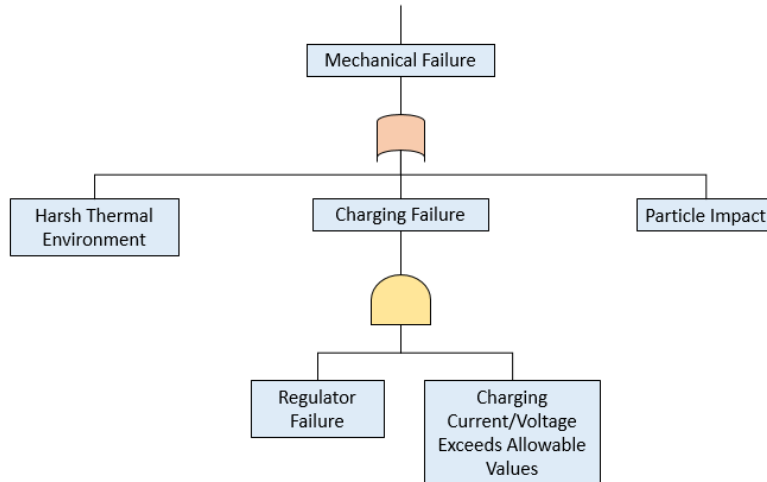
We also completed a fault tree analysis of the power system for the ERV, which as will be discussed later is applicable to the MLP and MAV as well. A power system failure is a result of either solar array failure or battery failure. A solar array failure could have been caused by an unlocking failure, deployment failure, a locking failure, an orientation failure, or some other mechanical failure. An unlocking failure is caused by either an electronic failure or a cutting failure. The deployment failure causes can be seen in the figure below.



A locking failure is either the result of inappropriate driving torque of locking spring or the harsh thermal environment. An orientation failure is caused by either a motor failure or transmission failure. The causes of other mechanical failures can be seen in the figure below.



A battery failure is the result of electronic failure or mechanical damage. The causes of mechanical damage can be seen in the figure below.



All critical components of the power system either have built-in redundancies or there are multiple of them within the system, which reduces the risk of failure. Testing, particularly thermal and vibrational testing, can also reduce the risk of failure.

12. ERV Attitude Control System

The RFP did not require us to address the Attitude Control Systems (ACS) of our spacecraft, but we nevertheless performed a preliminary analysis.

The first step was to determine what type of ACS was needed for the ERV. A necessary derived requirement for the system was that the ERV has to be able to rendezvous with the MAV, as well as be able to dock with the ISS during the Earth Capture stage. Precise adjustments the ERV were necessary to dock with the MAV and ISS. We therefore determined that the ACS should be three axis control; reaction wheels were determined to be a viable solution for precise adjustments.

The reaction wheel size was determined by the highest environmental torques being applied to the ERV. We calculated the solar, atmospheric, and gravity gradient torques for both Mars and Earth orbit scenarios. Once all the torques were calculated for the expected orbits, the highest torque was used to size the ACS components. The highest torque applied was determined to be 0.0001 Nm of torque from the gravity gradient at an orbit of 350 km orbit. The 0.0001 Nm was calculated with the mass moment of inertia when the first two sets of drop tanks were detached. The aerobraking phase was dealt with separately and was not used in this analysis, because the reaction wheels will not be used for control during aerobraking.

Three reaction wheels oriented orthogonally are necessary for full axis control. A fourth reaction wheel in an offset orientation provides redundancy, allowing it to compensate for any of the primary wheels. This way, the system can lose one wheel and still have full control. The reaction wheels chosen needed to be capable of overcoming the orbital torques that occurred at each stage of the trajectory as well as getting a reasonable time to saturation. We chose the Honeywell HR 2030, with an angular momentum capacity of 19.5 Nms and an output torque of 0.21 Nm. The amount of time calculated for saturation time was 188551 seconds, about 2.2 Earth days. In reality, the precise pointing ability provided by reaction wheels would only be needed during the rendezvous phase. However, for the sake of sizing the system, we assumed that the wheels would be used for the entire 42 months that the ERV is in Mars orbit.

Thrusters are needed to desaturate the reaction wheels. To size the thrusters, the desaturation time was calculated based on the amount of thrust, angular momentum of the reaction wheel, and the expected moment arm of the thrusters location. The amount of torque needed to be enough to overcome the max torque of the system. At least 12 thrusters are needed to provide control for three-axis control.

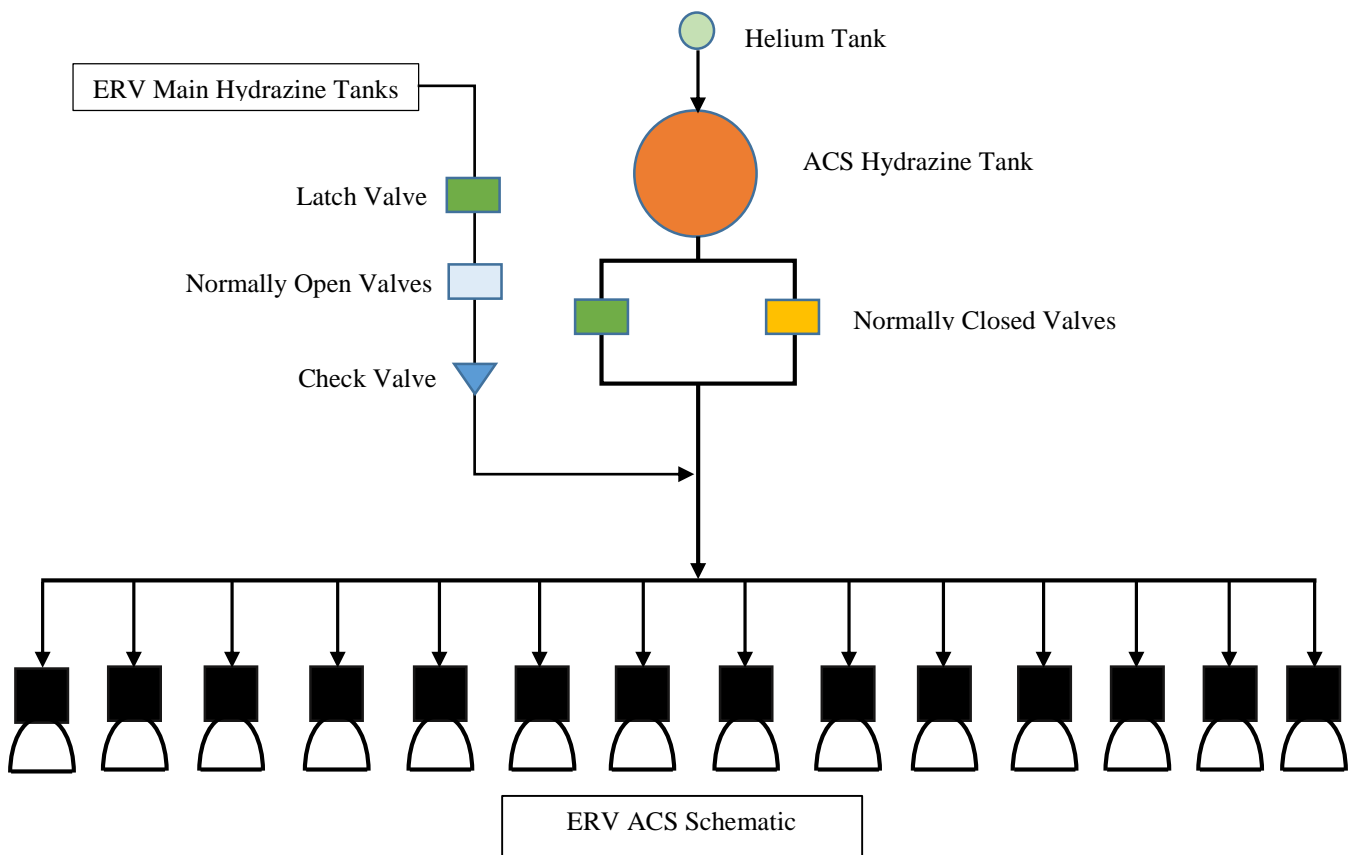
We chose a system similar to the MRO mission, with eight MR-103D 1 N engines and six MR-106E 22 N hydrazine monopropellant thrusters. The minimum impulse bit for the MR-103D engine is 0.027 Ns, while the minimum impulse bit for the MR-106E is 0.46 Ns. For preliminary analysis, thrusters were arranged in pairs with each having a moment arm of 0.5 m from the ERV center of the gravity. This means that the thrusters can provide 0.027 Nm and 0.46 Nm of torque, respectively. Since the torque provided by the 1 N thrusters is less than the maximum torque of the reaction wheels, they can desaturate the wheels without overcompensating.

The required propellant was estimated from the mass flow of the thruster and the amount of time necessary to desaturate the reaction wheels. For the 42 month period while the ERV is orbiting Mars, 11.3 kg of hydrazine propellant was required. For one month orbiting Earth (more time than is expected for ISS rendezvous) 0.118 kg is required. According to our GMAT simulation, the ERV used a total of 1.2 kg of propellant to maintain its orbit. This comes to an estimated total of about 12.6 kg for the mission.

Since the rendezvous sequences will require a high number of maneuvers, and because the aerobraking sequence carries a lot of uncertainties, we wanted to provide the ERV with as much maneuvering propellant as possible. For this reason, we provide more hydrazine to the ACS in two ways.

First, the ACS system has its own dedicated hydrazine tank with 18 kg of propellant and its own pressurant system. This provides a 30% margin over the calculated requirement.

Second, the ACS can draw propellant from the main hydrazine tanks. Plumbing connects the ACS to the main line that feeds from the main tanks. We chose this design because each drop tank needs to be jettisoned at some point, so they may as well be depleted as possible beforehand. For example, we anticipate that the first set of drop tanks will have a margin of 25% - nearly 500 kg, as was discussed previously. This margin will be used to assist with the subsequent set of maneuvers so that the propellant margin cascades through each mission phase. However, some of that extra propellant can be used for the ACS; and the amount needed is so small relative to the main maneuvers that it will hardly be missed. Ideally, the ACS thrusters will be able to draw from the drop tanks for nearly the entire mission, and will capture into Earth orbit with its dedicated 18 kg tank still full.



In order to use this system, Mission Control needs to have the ability to decide when to draw propellant from the main tanks and when from the ACS tanks. The shows a simplified schematic of our plumbing design. Powered latch valves can be commanded open and closed to control the propellant source. If necessary, the ACS can be isolated from the main tanks using a normal-open pyro valve. If the latch valve on the ACS side fails closed, the normal-closed pyro valve in parallel can be opened to ensure propellant gets to the thrusters. In this event, the valves on the thrusters would provide redundancy and control flow.

13. ERV Command and Data System / Telecommunications

The RFP states that CDS and telecom were not the focus of the proposal. Nevertheless, we performed preliminary analysis to determine some of the characteristics of these subsystems.

The SUD used for the KURS system functions as the ERV's flight computer.

For its transmitter/receiver, the ERV carries a Small Deep Space Transponder (SDST). This spacecraft terminal transmits and receives on both the X-band and the Ka-band. Because these communication bandwidths are standard to all spacecraft, the ERV will be able to communicate with Earth, with the MAV and MLP, and with any other operational Mars spacecraft.

We predict that the telecommunication data rates for the ERV will be relatively low, compared to other Mars missions. It will relay images sent by the MLP, and it will transmit images recorded during the MAV rendezvous. But its purpose is transportation, not data collection; so the bulk of its transmissions will be telemetry. Telemetry data rates are small; nevertheless, we wanted to give the vehicle a decent transmission ability, so we assumed a data rate of 5900 bps, about half that the Mars Exploration Rovers. This translated to a symbol rate of 6800 sps using Reed-Solomon encoding. To transmit at maximum range from Mars to Earth at this rate, we calculated that the ERV needs a high-gain parabolic antenna with a 1 meter diameter.

In addition, the ERV will need both a primary and a backup low-gain omnidirectional antenna. These will be used to receive commands and transmit telemetry. In particular, they will be used during the rendezvous phase and during the aerobraking phase, when it is not feasible to precisely point the high-gain antenna.

14. ERV Power Subsystem

The ERV power system must be able to supply power throughout the entire mission, including transit to Mars, orbit around Mars, rendezvous and docking with the MAV, transit back to Earth, orbit around Earth, and rendezvous and docking with the ISS. Our options for power sources are reduced to either an RTG or solar arrays due to the timeline of our mission. An RTG would be able to provide continuous power throughout the mission, but with the small amount of power they provide we would need multiple RTGs, which is not feasible. Therefore we have decided to use solar arrays.

During transit to Mars the solar arrays must provide power to two star trackers (1.5 W), heaters for all four sets of drop tanks (5 W), the thermal control system for the rest of the spacecraft (35 W), the five main engines (45 W), two 22N ACS thrusters (25.3 W), four 1N ACS thrusters (7.1 W), a low gain antenna (7 W), a motion control system (15 W), and two bi-axis solar array drive mechanisms (3.6 W).

During orbit maneuvers around Mars power must be supplied to four horizon sensors (0.132 W), two star trackers (1.5 W), heaters for all four sets of drop tanks (5 W), the thermal control system for the rest of the spacecraft (35 W), the five main engines (45 W), two 22N ACS thrusters (25.3 W), four 1N ACS thrusters (7.1 W), a low gain antenna (7 W), and a motion control system (15 W). Also during the day we need to power two bi-axis solar array drive mechanisms (3.6 W).

During rendezvous with the MAV, the ERV needs to power the KURS onboard computer (40 W), two cameras (3.2 W), three accelerometers (0.45 W), three angular rate sensors (12 W), four horizon sensors (0.132 W), heaters for two sets of drop tanks (5 W), the thermal control system for the rest of the spacecraft (35 W), the one of the main engines (45 W), three reaction wheels (20 W), two 22N ACS thrusters (25.3 W), four 1N ACS thrusters (7.1 W), a low gain antenna (7 W), and a motion control system (15 W). Also during daylight we need to power two bi-axis solar array drive mechanisms (3.6 W).

The Earth return transit, Earth orbit maneuvers, and rendezvous with the ISS will have similar power requirements as the transit to Mars, Mars orbit maneuvers, and rendezvous with the MAV, respectively. The only difference is that heaters for the drop tanks are no longer required.

The subsystem breakdown of required power for the ERV during launch and rendezvous can be seen in the table below. The total power required includes a 20% margin, which will decrease as the power requirements are refined and the design matures.

Characteristic	Transit to Mars	Mars Orbit Maneuver - Day	Mars Orbit Maneuver - Night	Rendezvous w/ MAV - Day	Rendezvous w/ MAV - Night
Time in Mode, h	–	1.177	0.752	1.177	0.752
Payload/Scientific Instruments	3	3.5	3.5	84.3	84.3
Thermal Control System	75	75	75	65	65
Propulsion System	225	225	225	45	45
Attitude Control System	79	79	79	139	139

Communication Systems	7	7	7	7	7
Command and Data System	15	15	15	15	15
Power Systems	7.2	7.2	0	7.2	0
Subtotal	411.2	411.7	404.5	362.5	355.3
Margin	20%	20%	20%	20%	20%
Total	493	494	485	435	426

The required power and operation time, seen in the table above, will be used to determine the size of the solar arrays and batteries. For the power system we assume an array to battery efficiency of 0.7, a battery to load efficiency of 0.96, and an array to load efficiency of 0.95. With these efficiencies and the day and night requirements for rendezvous, the solar arrays must provide approximately 408 W. The solar array will be composed of SolAero ZTJ triple-junction solar cells. These cells are 29.5% efficient and have an area of 26.62 cm².^{xv} Then, we assumed a 92% packing factor, 10% loss due to radiation damage, 2% loss due to UV discoloration of cells, 1% loss due to thermal cycling, 2.5% loss due to cell mismatch, 2% loss due to resistance in cell interconnects, and 1% loss due to contamination. This results in a required solar array area of 7.49 m². This will be achieved by using two Orbital ATK UltraFlex solar arrays, similar to those for Phoenix, each with a roughly circular area of 3.75 m² and mass of 5.62 kg.^{xvi}

The batteries are required to supply 485 W for 0.75 hours. For the battery system we assume a 3% power loss, 55% depth of discharge, and required bus voltage of 28 V. this results in a required capacity of 24.4 Ah. We will achieve this through use of Quallion QL015KA lithium-ion cells. These cells have a capacity of 15 Ah and mass of 0.36 kg and can provide 3.6 V.^{xvii} We have designed the power system to be able to function with one string out, therefore we will need 3 strings of 8 cell each. Thus the batteries can supply 28.8 V with a capacity of 30 Ah and mass of 8.64 kg.

The power system will also require two deployment/drive mechanisms with a mass of 1.15 kg each and two panel attachments with a mass of 0.84 kg each.

15. ERV Structure

For structural analysis on Project Argo all structural components were assumed to be rigidly bonded together and no strength losses at the connections. The connection method between the components and the structure was not

detailed, the resultant forces on the structure were modeled where the configuration called them to be. The analysis was done using hand calculations, Microsoft Excel, and Solidworks Simulation module. Models of the structural components were created and loaded with the expected forces and acceleration at various stages of the mission. Using the stress and factor-of-safety data generated by Simulation, the diameter, thickness, and length of the various structural supports were optimized to reduce mass and still withstand the expected forces at each mission stage.

Carbon-fiber, titanium alloys, and aluminum alloys were considered during the design of the structures. The aluminum alloys investigated were 7075-T651, 7075-T6, and 7068-T6511, and the titanium alloys investigated were Ti-6Al-4V, Ti-6Al-2Sn-2Zr-2Mo-2Cr-0.15Si, Ti-4.5Al-3V-2Mo-2Fe, Ti-5Al-2.5Sn, Ti-6Al-4V ELI, and Ti-3Al-8V-6Cr-4Zr-4Mo. Carbon fiber was also considered, but its material properties had to be approximated due to the various methods of creating carbon fiber components.

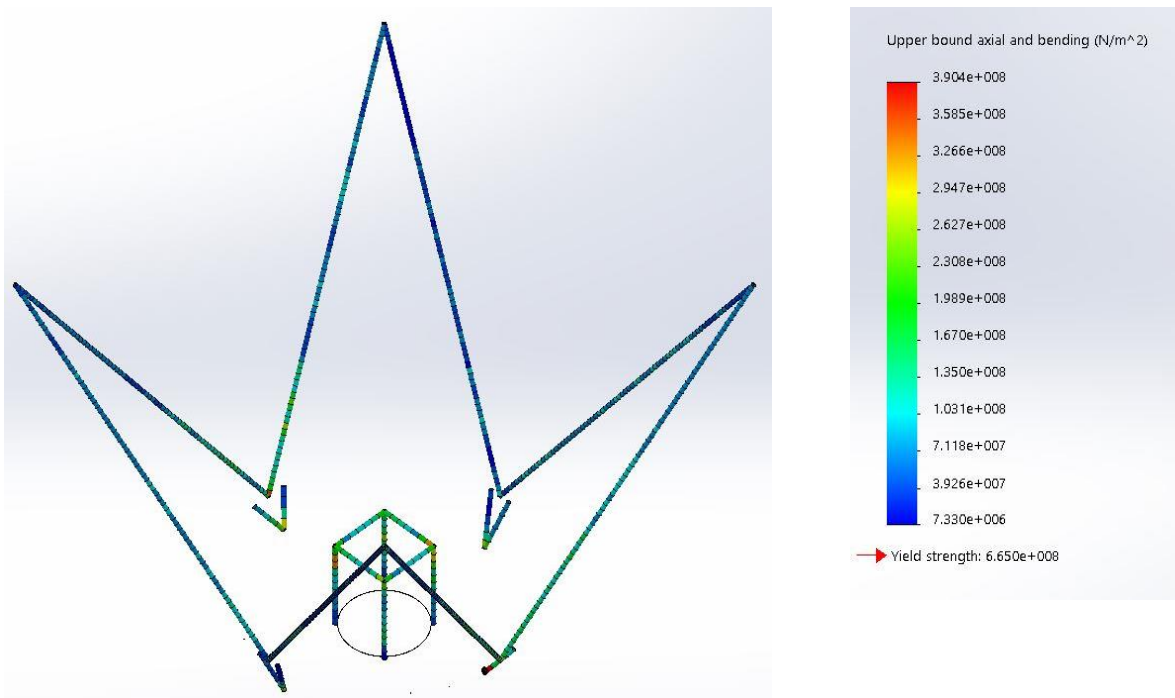
The ERV design is derived from other Martian sample return systems, mainly a proposal authored by JPL in the 1980s, which included using a cone as a guide to make transferring the sample easier for an automated docking system. The cone is held in place by a ring around its top edge, and at the bottom by a box truss. The cone and top ring is held in place by four triangular supports, which connect to a second ring at the base of the truss, which is in turn connected to the fuel tanks. The fuel tanks are supported by cylindrical inserts that will be friction fitted into place. Supporting the central truss and the internal components of the ERV is a single column, which leads down to a circular frame at the bottom of the ERV. This frame consists of an outer ring and spokes running to the middle, supporting the fuel tanks and central support. This frame's edge is also where the PAF from the Delta IV will connect with the ERV.

The central box truss and internal support is designed to accept the entire load due to the cone, MLP, and MAV, with the external supports accepting a third of the load, along with the solar panels and antennas. The original design had the external supports accepting the entire load from the cone, MLP, and MAV, with the third of the load being accepted by the central box truss. Analysis on the fuel tanks accepting this load showed they would be unable to withstand the liftoff forces caused by the Delta IV, and thickening the fuel tanks would result in an unacceptable mass increase. In addition, the truss was originally aluminum, but it, the central column, and the lower truss were changed to carbon fiber to save mass. The lower truss was redesigned from a honeycomb plate to a truss system to save weight, which will need thermal insulation to survive aerobraking around Earth.

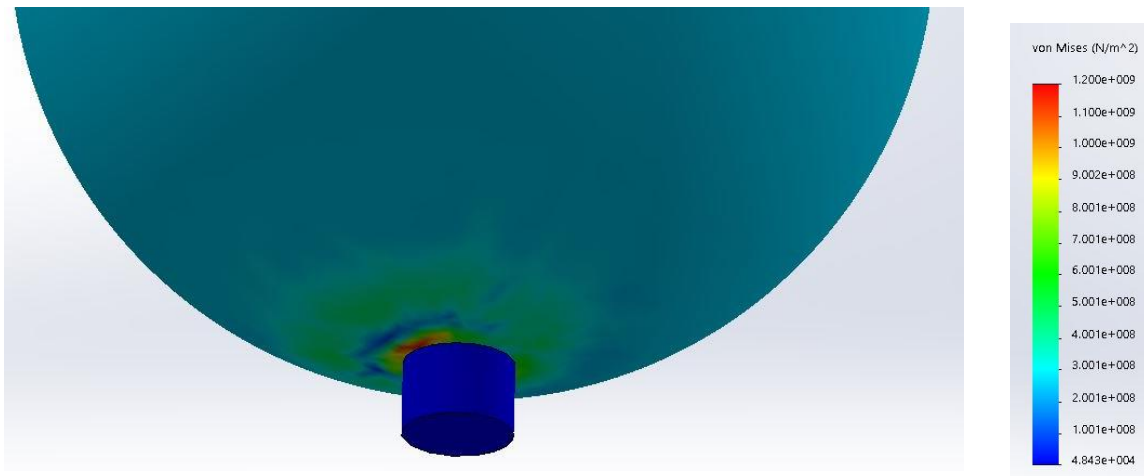
Buckling analysis was also conducted on the upper truss system, increasing mass but staying within budget. Including the buckling mass increase, the final mass of the upper truss is 37 kg, the internal support is 36 kg, the lower

truss is 9 kg, and each of the tank inserts are 2.3 kgs, resulting in a total mass of 100 kg. This is 44% under the allocated mass budget of 177 kg, though this model does not include several components, most notably the tank ejection system supports. While not intending to load-bearing structures, they will require a part of the mass budget. Similarly, the capture cone is estimated to be made out of fiberglass or a similar lightweight material, though at the current size, it has a mass of 76 kg. With a more precise design, such as using stringers instead of a solid shell, the mass of the cone can be fit within the mass budget.

However, the most pressing issue to solve is the resulting stress on the tanks due to the loading of the upper truss. The point where the tanks meet the separators is causing an unacceptable level of stress to occur. A more precise design will show what kind of tank support will reduce stress in the tanks while allowing them to be jettisoned easily. There are similar stress risers in the lower truss, which can be eliminated by reinforcing the joints during manufacturing. The yield strength of carbon fiber is $6.65 \times 10^8 \text{ N/m}^2$, and $5.05 \times 10^8 \text{ N/m}^2$ for the 7075-T6 aluminum fuel tanks, and the validity of the design can be seen in Figures 2-10



ERV Upper Truss Beam Stress during Delta IV Stage 2 Launch (carbon fiber with >1.5 margin of safety)



ERV Tall Tank stack during Delta IV Stage 1 Launch (Aluminum tank with carbon fiber support)

B. Mars Ascent Vehicle (MAV)

1. MAV Propulsion Subsystem

The MAV propulsion subsystem uses a spherical titanium gas blowdown tank for its fuel, and a gravity fed toroidal titanium tank for the NTO oxidizer. This spherical + toroidal architecture is used on a Russian rocket.

Hydrazine was chosen because of its dual mode ability – for the MAV, mass and volume are at a premium, so the ACS system pulls from the same tank rather than have its own dedicated tank. The tank is sized up to include an extra 4 kg of propellant for attitude control.

The spherical tank was designed using the same process described in the ERV Propulsion section. It is placed in the nose of the vehicle to bring the CG forward and improve aerodynamic stability during climb.

We decided to use a toroidal tank for the oxidizer in order to make the vehicle more compact and to facilitate the stowing of the sample cache: the sample container is drawn through the hole in the center of the toroid. A bladder would not work inside a toroidal tank, so instead it is mounted at a 15 degree angle to facilitate gravity feeding. In zero gravity, the main engines are fired in monopropellant mode from the pressurized hydrazine tank for a settling burn. This forces the oxidizer into the feed tube and the vehicle can immediately switch to bipropellant mode.

We designed the toroid based on the required inner diameter needed for the sample stow and the required NTO volume + ullage. For ease of analysis we assumed the thickness of the tank of the tank was 1.2 times the thickness of the spherical tank. This is a conservative estimate since the tube diameter of the toroid is less than that of the sphere, indicating that the hoop stress would be less; further analysis will likely decrease the indicated tank mass.

Our Mars launch mass indicated that a nominal thrust of 2500 N per engine was optimal. We could not find a COTS dual mode engine available with this spec. However, we are confident that it can be developed within the available timeframe by leveraging the designs of the many engines that are available. For mass, power, and dimension estimation we used the Aerojet Rocketdyne R-40B 4000 N engine. We assumed a specific impulse of 320 seconds, typical of hydrazine/NTO.

2. MAV Instrumentation/rendezvous equipment

The KURS Automatic Rendezvous and Docking System on the MAV is the passive side which is sought out by ERV. KURS on the MAV works by emitting an initial homing beacon for the ERV to lock onto. The system on the MAV is considered passive because it doesn't have to maneuver the vehicle. Because KURS has been used for many years successfully on the Soyuz and ISS and is an off-the-shelf system, it was attractive for our mission. Listed below are the components of the KURS system on the MAV:

- 4AO Antenna
- AP Antenna
- 2AP Antenna
- Transmitter
- Receiver

Payload			Budgeted:	17.10
Item	Count	Mass (kg)		
Can (launch only, 3 kg)	1			
Coil boom	1			0.7
Truss lock/release mechanism	1			0.3
Bucket	1			6
Winch	1			0.3
Cable	1			0.1
4AO antenna	1			0.5
AP antenna	1			0.5
2AP antenna	1			0.5
antenna arms	2			0.5
transmitter	1			1
receiver	1			1.4
Total				12.3
Remaining budget				4.80

3. MAV Thermal Control Subsystem

The MAV will be under protection of the aeroshield and backshell during most of its journey to Mars. The majority of its thermal events will take place on the surface of Mars. Therefore, the main concern is the wide range of temperatures throughout the Martian seasons. The steady state temperature analysis used for the ERV is also used for the MAV using a surface area of 16 m².

The heatshield will be similar to the heatshield used in the MSL program. The backshell will be composed of the same cork/silicone super light ablator (SLA) 561V (Lockheed Martin). The heat shield will also carry over the tiled Phenolic Impregnated Carbon Ablator (PICA) thermal protection system. Electronics will be enclosed in an insulated compartment. Heat pipes will transfer excess heat to propellant lines and tank linings. These tanks will be insulated with 3cm thick fiberglass. MLI loses its effectiveness in the presence of pressure from the Martian atmosphere.

In orbit while awaiting rendezvous with ERV, skin equilibrium surface temperatures will reach a maximum of negative 30°C. This will require 3 15 watt heaters to keep the batteries, sample, and navigation system around 10°C. Solid state controllers and type-T thermocouples are utilized for these relatively more sensitive components.

Most active systems will be on sleep mode for this vehicle while on the surface of Mars. The warmest temperature near the newton crater is around 33°C during the summer day. Excess internal heat will be dissipated to a deployable finned radiator with a surface area of 0.35m². The radiator is connected by heat pipes to the electronics box. The radiator was chosen to be deployable since the added surface area from the fins will be ineffective in the vacuum of space. The other side of the radiator will be covered in vapor deposited aluminum on Teflon®. The coldest surface temperature is -120°C during winter nights.

4. MAV Integrated Vehicle Health Monitoring System

On the Mars Ascent Vehicle the primary mechanical component are the coil truss and the sample canister winch. Both of these devices need to function correctly in order to transfer the sample bucket from the rover to the MAV and then from the MAV to the ERV. If the winch used to hoist the sample bucket into the MAV becomes jammed, the sample won't be loaded into the MAV and the MAV cannot launch to orbit. The IVHM system can monitor current draw from the winch motor as well as check sensor data to know the length of cable that has been pulled in. In the event of a jam, the system can reverse the winch a little bit to try to clear the jam.

5. MAV Attitude Control System

The MAV ACS system was analyzed similarly to the ERV ACS system. Solar, atmospheric, gravity gradient, and magnetic torques were calculated for a Mars 350 km orbit, where the rendezvous will take place. The highest torque came from the solar torque at $1.45e-5$ Nm. Reaction wheels were necessary to accommodate the pointing precision needed for ERV rendezvous and payload transfer.

Based on the calculated orbital torques, the Blue Canyon RWP500 reaction wheel was chosen, providing 0.5 Nms of torque necessary to overcome solar torque. The RWP500 also had to have a fairly long saturation period to maintain control long enough to transfer the payload to the ERV. A calculated 34,488 seconds to saturation was found, about 9.6 hours. The MAV will accommodate 4 reaction wheels: 1 reaction wheel for all axes, as well as 1 extra reaction wheel oriented off-axis for redundancy.

The MAV will also have three-axis control, provided by 12 thrusters. To size the thrusters, we considered first that they had to be small enough to desaturate the reaction wheels. Second, we wanted to provide as much thrust as possible to assist with control during launch. Third, we wanted to minimize mass and valve electrical power.

We chose the flight proven MR-111C thrusters built by Aerojet Rocketdyne. The minimum impulse bit of the MR-111C is 0.08 Ns; a pair of these acting at a distance of 1 meter will provide a couple of 0.08 Nms. This is less than the maximum reaction wheel torque of 0.5 Nms, which means that the thruster can desaturate the wheel without overcompensating. We calculate that it requires only 0.0005 kg of propellant to desaturate a wheel.

During launch, primary control will be provided by varying the thrust of the main engines. But we wanted to make the ACS thrusters as powerful as possible to assist, since the MAV shape means that moment arms will be relatively small. The MR-111C has a nominal thrust of 4 N and a max thrust of 5.3 N. Compared to other thrusters with an acceptably small minimum impulse bit, this is the most thrust for the same amount of mass and electrical power. For comparison, the Delta IV Heavy upper stage ACS system thrusters are rated at 41 N and 21 N.

Desaturating the reaction wheels takes a negligible amount of propellant, considering that the MAV will have completed its mission within a couple days after launch. More propellant is expected to be used during launch. For a worst case scenario of one engine out for Mars launch, where the opposite engine will be forced to throttle down to near zero in order to minimize the moment on the vehicle, the burn time is 313 seconds. In this worst case, four thrusters could conceivably be required to fire continuously in order to maintain control. At a max thrust of 5.3 N for the MR-111C, this would mean that 3.1 kg of propellant is required. The ACS pulls from the same hydrazine tank as

the main bi-propellant engines. We sized up the hydrazine tank to include an extra 4 kg to account for the worst case launch, as well as to provide extra maneuvering ability for rendezvous.

6. MAV CDS/Telecom

The RFP states that for the MAV, the total mass for avionics, guidance/navigation/control (GNC), and communications may be assumed to be 15 kg. For Project Argo, we designed the MAV attitude control subsystem and calculated its total mass to be approximately 8.5 kg (including horizon sensors used for guidance and navigation). This leaves 6.5 kg for the flight computer and for telecommunications.

The MAV will carry a Small Deep Space Transponder; similar to the other vehicles, but with the X-band exciter only in order to save power. It will not have the power to communicate directly with Earth; but during launch and rendezvous it will be broadcasting telemetry that will be received and relayed by other vehicles. The SDST also allows for 2-way communication with the ERV during rendezvous. The MAV will require an omnidirectional low-gain antenna.

The MAV will require a flight computer/avionics package for command and data. It will need to have a fast processor in order to make real time decisions during Mars launch. The command and control software will be nontrivial to write. However, for this proposal we need only provide a budget for mass and power.

The SDST has a mass of 3.2 kg, leaving 3.3 of the assumed 15 kg. But in fact, our design allows for additional mass to be allocated. We budget an additional 2.5 kg for telecom, and 3.8 kg for CDS. Once further analysis is complete and hardware is selected, anything remaining from the mass budget may be allocated to other subsystems.

7. MAV Power Subsystem

The MAV power system must be able to supply power throughout launch and rendezvous with the ERV. For this short time frame we have many options for a primary power source, including batteries, a solar array, or an RTG.

Primary batteries have the benefit of having low mass and volume, but they are single use only which has inherent risks if issues arise that change the power requirements or cause the mission to last longer than expected.

RTGs provide a constant power source for a long period of time, but it would need to be supplemented with batteries to provide the large power necessary during launch. An RTG also limits the landing location of the MAV due to planetary protection protocol – an RTG cannot be brought to a location that is believed to have ice at a depth

of less than 5 meters.^{xviii} Since we are required to have the ability to launch from anywhere on the surface of Mars, we cannot use an RTG.

Solar arrays can provide power during daytime operations, but need rechargeable batteries for nighttime operations. Also the solar arrays they must be stowed and protected during launch. We have determined the risks associated with using primary batteries outweigh the benefits of lower mass and volume. Therefore the power source for the MAV will be solar arrays and rechargeable batteries.

During launch the batteries will need to provide power to the thermal control system (35 W), four 2500 N thrusters (70 W), four infrared sensors (0.132 W), two thruster clusters (20.1 W), three reaction wheels at peak torque (23 W), a small deep space transponder (15.8 W), and the command and data system (15W).

During rendezvous the solar array will provide power and charge batteries during the day and the batteries will provide power at night. For rendezvous power must be supplied to a transmitter (11 W), a receiver (9.2 W), the thermal control system (35 W), four infrared sensors (0.132 W), two thruster clusters (20.1 W), one reaction wheels at peak torque (23 W), two reaction wheels at steady state (6 W), a small deep space transponder (15.8 W), the command and data system (15W), and, during the day, two bi-axis solar array drive mechanisms (3.6 W).

The subsystem breakdown of required power for the MAV during launch and rendezvous can be seen in the table below. The total power required includes a 30% margin, which will decrease as the power requirements are refined and the design matures.

Characteristic	MAV Launch	Rendezvous w/ ERV - Day	Rendezvous w/ ERV - Night
Time in Mode, h	0.133	1.235	0.699
Payload/Scientific Instruments	0.0	20.2	20.2
Thermal Control System	35.0	35.0	35.0
Propulsion System	280.0	0.0	0.0
Attitude Control System	109.7	75.7	75.7
Communication Systems	15.8	15.8	15.8
Command and Data System	15.0	15.0	15.0
Power Systems	0.0	7.2	0.0
Subtotal (W)	455.5	168.9	161.7
Margin	30%	30%	30%
Total (W)	592	220	210

The required power and operation time, seen in the table above, will be used to determine the size of the solar arrays and batteries. For the power system we assume an array to battery efficiency of 0.7, a battery to load efficiency of 0.96, and an array to load efficiency of 0.95. With these efficiencies and the day and night requirements for rendezvous, the solar arrays must provide approximately 408 W. The solar array will be composed of SolAero ZTJ triple-junction solar cells. These cells are 29.5% efficient and have an area of 26.62 cm². Then, we assumed a 92% packing factor, 10% loss due to radiation damage, 2% loss due to UV discoloration of cells, 1% loss due to thermal cycling, 2.5% loss due to cell mismatch, 2% loss due to resistance in cell interconnects, and 1% loss due to contamination. This results in a required solar array area of 3.29 m². This will be achieved by using two Orbital ATK UltraFlex solar arrays, similar to those for Orion, each with a roughly circular area of 1.64 m² and mass of 2.47 kg.

The batteries are required to supply 210 W for 0.7 hours. For the battery system we assume a 3% power loss, 80% depth of discharge, and required bus voltage of 28 V. this results in a required capacity of 6.8 Ah. We will achieve this through use of Quallion QL015KA lithium-ion cells. These cells have a capacity of 15 Ah and mass of 0.36 kg and can provide 3.6 V. We have designed the power system to be able to function with one string out, therefore we will need two strings of 8 cell each. Thus the batteries can supply 28.8 V with a capacity of 15 Ah and mass of 5.76 kg.

The power system will also require two deployment/drive mechanisms with a mass of 0.51 kg each and two panel attachments with a mass of 0.37 kg each.

8. *MAV Structure*

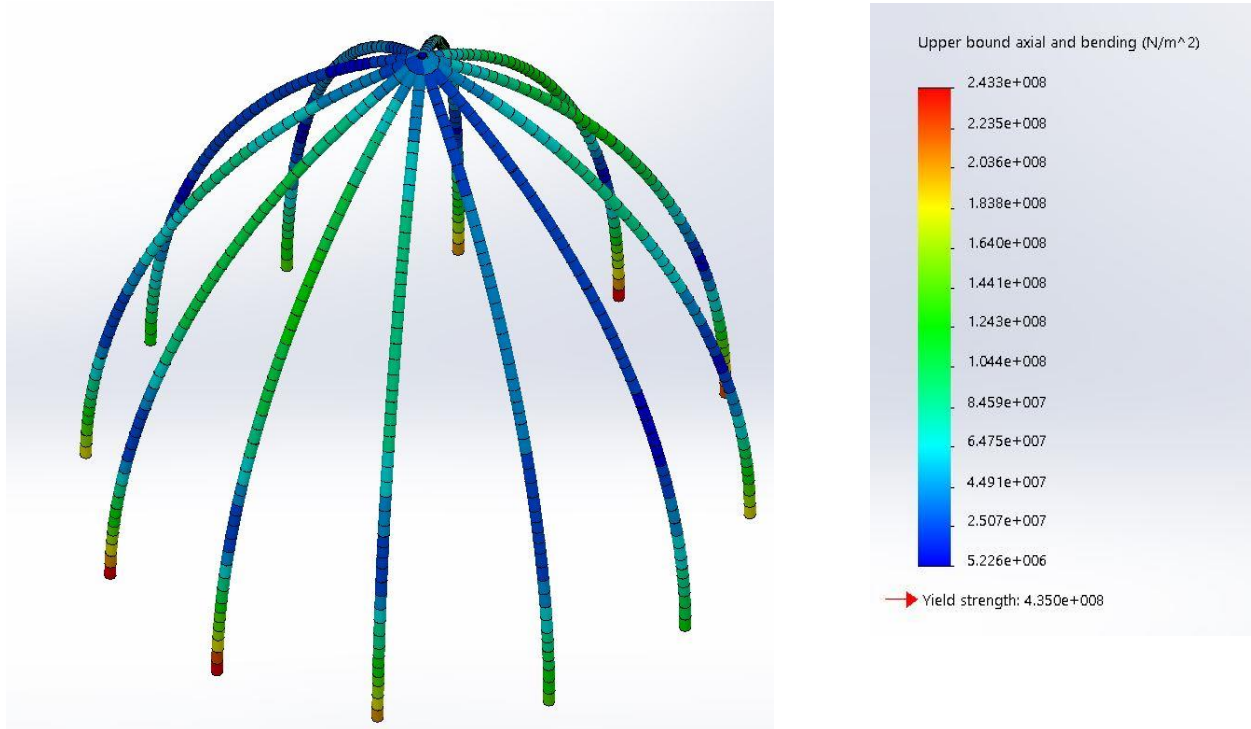
The MAV is a single-stage rocket, and is modeled structurally off of Earth-launch rockets. The MAV structure is a series of twelve internal stringers supported by ten internal rings to provide support to the internal components and to support an external aerodynamic shell. The two largest components to support are the two internal fuel tanks, with the circular fuel tank in the nose and the toroidal tank towards the middle of the MAV. Internal supports connect the upper circular tank to the toroidal tank, and transfer the load to the stronger middle section of the MAV. The load is then transferred to engine mount points on the lower plate of the MAV.



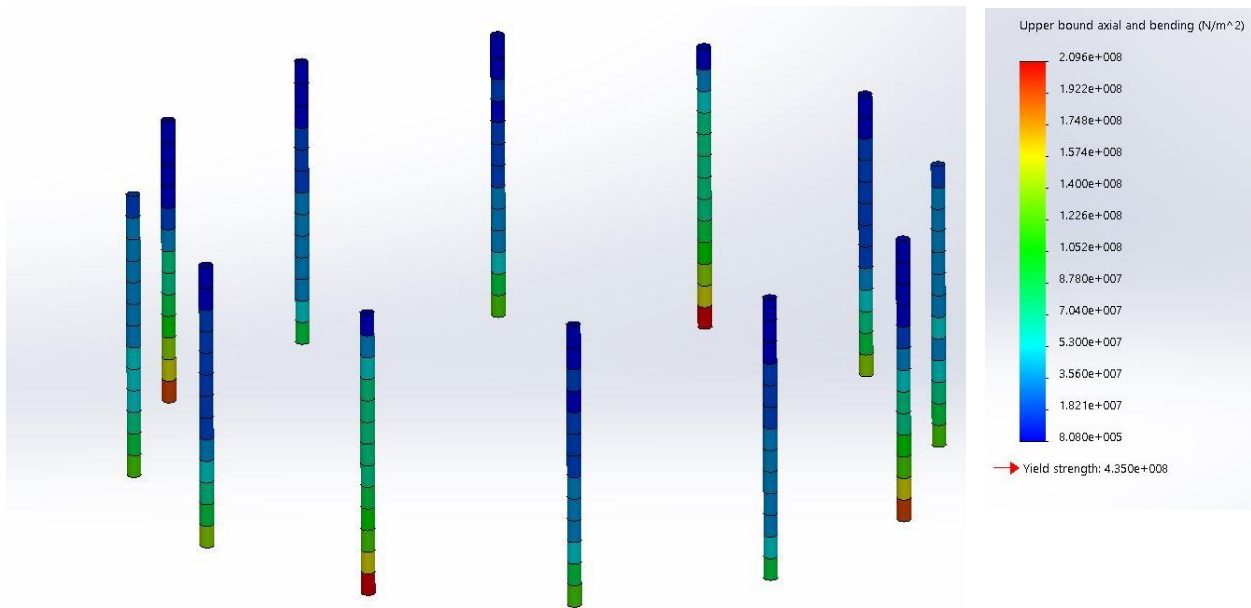
Figure 14: MAV Structural Components

The internal supports, rings, and plate are all 7075-T7 aluminum, and the outer shell is carbon fiber. The internal supports total 11.3 kg, the upper tank supports are 1.9 kg, the lower tank supports are 1.3 kg, the bottom plate is 11 kg, and the shell is 10 kg. With a total mass of 35.5 kg, the MAV is 18 % below the mass budget of 43.7 kg. This mass amount can be further reduced, as the margin of safety for this vehicle is currently larger than the ERV and MLP, due to limitations with Solidworks Simulation

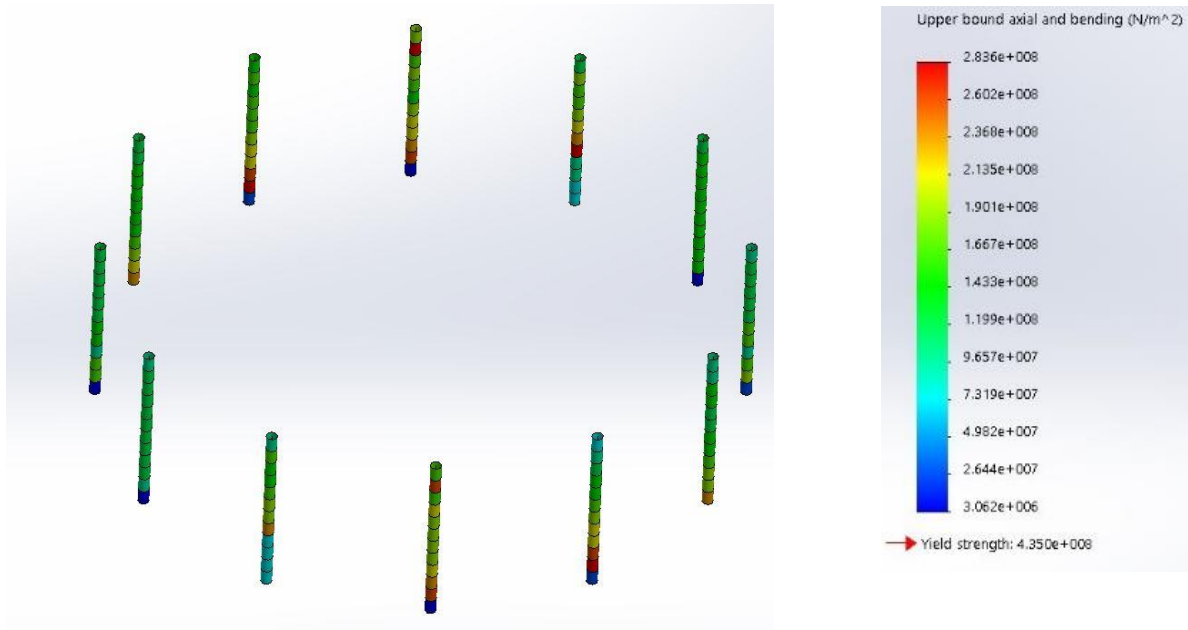
The weakest part of this analysis is the estimation of where all of the components attach, the resultant forces are applied to all internal supports near where they will be attached, rather than exactly where each component will be. The toroidal tank is also at an angle to the interior of the MAV, and thus will not load each internal ring equally, and how much of the fuel tank load is distributed between the internal supports and the outer supports of the MAV is not exact. The aerodynamic forces during Martian liftoff are also not modeled yet. Subsequent redesigns will account for these issues, though with the current high factor of safety, redesigning can be kept to a minimum.



Upper MAV Frame Stress during Delta IV Stage 2 Launch (>1.5 margin of safety)



Middle MAV Frame Stress during Delta IV Stage 2 Launch (>1.5 margin of safety)



Lower MAV Frame Stress during Delta IV Stage 2 Launch (>1.5 margin of safety)

C. Mars Landing Platform

1. Design drivers

The purpose of the MLP is to bring the MAV to the surface safely, within range of the rover. This means that as with the other vehicles, the propulsion subsystem was the main design driver. The engines had to be capable of maneuvering both the MAV and the MLP's mass, and the vehicle had to carry sufficient propellant to fly to the designated landing site.

2. MLP Propulsion Subsystem

The MLP using a hydrazine monopropellant propulsion system. Historically, bipropellant propulsion systems using a fuel and an oxidizer have not been used for Mars landers, because one of the products of the reaction is water. This would contaminate the landing site and possibly lead to incorrect experimental results. Hydrazine was the obvious and reliable choice for a high energy, proven monopropellant.

The propellant is carried in three spherical gas blowdown titanium tanks, designed in the same way as discussed previously. Each tank has its own helium pressurant tank and each tank is cross fed with each nozzle. The nozzles will draw from each tank simultaneously in nominal mode, but in a contingency mode one or more tanks can be separated from the system.

The MLP has nine engines. The three main engines are throttleable, 3100 N, model MR-80B from Aerojet Rocketdyne. These engines provide the bulk of the thrust and are powerful enough to land the MAV/MLP on their own. The six secondary engines are pulse controlled, 440 N, model MR-104A/C, also from Aerojet Rocketdyne. They are arranged in equidistant pairs at the ends of the MLP platform. They are pulsed to provide control for the MLP as it maneuvers and hovers, and are fired in steady state to assist with the major maneuvers.

Combining multiple more powerful and less powerful thrusters gave our design redundancy while keeping engine mass low. Using the less powerful engines to control the vehicle meant that we could save mass by not using a gimbal system.

3. MLP Instrumentation/rendezvous equipment

Terrain Relative Navigation, TRN, provides the ability to estimate location of the descending MLP. This works by comparing stored images of the Mars terrain with photos taken during descent. This location information is used by a flight divert algorithm called G-FOLD to plot the most fuel efficient route to the landing site. This system was utilized because it has been tested successfully on a demonstration flight on Earth and was able to land very accurately.

- Camera: PhotoFocus with a Kowa 6mm Lens
- Computer: Virtex 5 FPGA & LEON 3 Flight Processor

The Landing Radar System provides an accurate measurement of altitude during descent. This information is necessary to for the G-FOLD algorithm to function correctly, since the TRN system only provides coordinates and not altitude information.

- Digital Electronics Assembly (DEA)

Payload		Budgeted:	20.58
Item	Count	Mass (kg)	
(MAV wet; not included in this mass budget)			
PhotonFocus DS1-D1024 CMOS Camera w/ global shutter	2		0.21
Kowa 6M6HC 6mm Lens	2		0.215
Structural attachment	1		0.02
Antenna	2		
Digital Electronics Assembly (DEA)	1		
Digital Power Distribution Unit (DPDU)	1		
RF Power Distribution Unit (RPDU)	1		
Frequency Synthensizer (FSA)	1		
Upconversion / Downconversion Assembly (UDA)	1		
Frontend Filters (FFA)	1		
Transmit/ receive modules (TRM)	1		
Harness	1		
Radar system estimate	1		14
Science package	1		3
Total			17.87
Remaining budget			2.71

- Digital Power Distribution Unit (DPDU)
- Frequency Synthesizer (FSA)
- Upconvert/Downconvert Assembly (UDA)
- Frontend Filters (FFA)
- Transmit/ Receive Modules (TRM)
- Antennas

4. MLP Thermal Control Subsystem

As with the MAV, the MLP will be protected by the aeroshield during interplanetary travel. Most of the components are used during the entry, descent, and landing phase of the mission. Batteries and communication boxes will be mounted on the top surface relative to landing configuration. A white oxide paint will cover the majority of non-structural components to reflect excess sunlight while still emitting infrared heat. Radiation heat transfer will be small compared to convection to the air. This will keep surface temperatures around Mars atmospheric conditions. During EDL, a thin layer of carbon-carbon will keep radiant heat from the nozzles to the MLP. The heat dissipated from communications keeps components above -20°C. The batteries will be insulated and have a self-powered solid-state controlled in the form of flexible Kapton® heater.

5. MLP Integrated Vehicle Health Monitoring System

On the Mars Landing Platform there are three petals that are closed over the Mars Ascent Vehicle to protect it during flight. Prior to launch of the MAV from the MLP, these protective petals need to open fully to allow the MAV clearance to take off. The health management system will check position sensors attached to each petal to verify them to be open all the way. Without the petals being open fully, there is a possibility of the MAV clipping the edge and crashing. In order to rule out a sensor failure, a camera onboard the MLP can be used to verify the petals are opened.

6. CDS/Telecom

As a baseline for the MLP's flight computer, we chose the Airbus OSCAR. This computer is similar to that used on the Masten vehicle testing JPL's Terrain Relative Navigation system.

The MLP carries a number of cameras and a small science package, but it will not need to transmit large amounts of scientific data. We predict that a telecom system with a small high-gain dish antenna and two to three low gain antennas will be appropriate. The dish antenna could be the same used by the Mars Exploration Rovers – smaller than that of the Mars Science Laboratory. Like the other vehicles of Project Argo, the MLP will also be equipped with a Small Deep Space Transponder.

7. MLP Power Subsystem

The MLP power system must be able to supply power for entry, descent and landing and throughout surface operations. The MLP may have to survive for a long period of time on the surface of Mars. This limits our options for a primary power source to either an RTG or solar arrays. While an RTG can provide constant power for a long period of time, it also limits the landing location because of planetary protection protocol. While a solar array's landing location is not limited by planetary protection protocol, the time of year will limit the landing location and surface operation timeline. We have determined the risks associated with using solar arrays are more acceptable than those associated with an RTG.

For the solar array, we assume the worst case scenario on landing will be at winter/summer solstice at a latitude of $\pm 60^\circ$, respectively.^{xix} Using these inputs and the Daylight Hours Explorer online tool provide by UNL^{xx}, we found the expected sunlight hours on Earth and, since Earth and Mars have similar rotation axis tilt relative to their orbital plane, we calculated the expected sunlight hours received at Mars to be 5.7 hours. Thus the worst case nighttime expected on Mars is 19 hours. From EDL analysis, the MLP must perform powered landing for 3 minutes.

During landing the batteries must provide power to the radar system (50 W), the photon focus camera (3.2 W), three Northrop Grumman LN-200 fiber optic gyro (12 W), three MR-80B engines (168 W), six MR-104A/C engines (30 W), a flight processor (15 W), and a small deep space transponder (15.8 W).

For daytime surface operations the system can supply power to three ECAM-C50-NFOV cameras (full – 2.5 W, idle – 1.75 W), one ECAM-IR1 camera (full – 8.75 W, idle – 6.5 W), one ECAM-DVR4 recorder (full – 13.5 W, idle – 9.75 W), two bi-axis solar array drive mechanisms (3.6 W), and a small deep space transponder (15.8 W). By turning the cameras and DVR into idle mode there are 8.25 W available for a science package.

For nighttime surface operations the system must supply power to three ECAM-C50-NFOV cameras (idle – 1.75 W), one ECAM-IR1 camera (idle – 6.5 W), one ECAM-DVR4 recorder (idle – 9.75 W), and a small deep space transponder (15.8 W).

The subsystem breakdown of required power for the MLP during launch and rendezvous can be seen in the table below. The total power required includes a 30% margin, which will decrease as the power requirements are refined and the design matures.

Characteristic	Mars EDL Sequence	Mars Nighttime Operations	Mars Daytime Operations
Time in Mode, h	0.05	19.0	5.7
Payload/Scientific Instruments	53.2	21.5	29.8
Thermal Control System	0.0	0.0	0.0
Propulsion System	684.0	0.0	0.0
Attitude Control System	36.0	0.0	0.0
Communication Systems	15.8	15.8	15.8
Command and Data System	15.0	0.0	0.0
Power Systems	0.0	0.0	7.2
Subtotal	804.0	37.3	52.8
Margin	30%	30%	30%
Total	1045	48.5	68.6

The solar array for the MLP will be the same UltraFlex solar arrays and composed of the same triple-junction solar cells used for the MAV. We also assume the same efficiencies and losses assumed for the MAV, which results in a required solar array total area of 4.45 m². This will be split between two roughly circular arrays each with an area of 2.23 m² and mass of 3.34 kg.

The batteries will make use of the same lithium-ion cells used for the MAV and we assume the same power loss, depth of discharge, required bus voltage, and one string out capability. Form the nighttime operational requirements, presented in the table above, the system has a required capacity of 42.4 Ah. Therefore the MLP will need four strings of 8 cells each. Meaning the batteries can supply 28.8 V with a capacity of 45 Ah and mass of 11.52 kg.

The power system will also require two deployment/drive mechanisms with a mass of 0.68 kg each and two panel attachments with a mass of 0.5 kg each.

8. MLP ACS Subsystem

As discussed previously, attitude control for the MLP will be provided by pulsing the six 440 N secondary engines. Further analysis will determine the maximum controllable tipping angle, but we expect that the MLP should not exceed 45 degrees from upright.

Part of the avionics package in the TRN system is a LN-200 fiber optic gyroscope built by Northrop Grumman. For the MLP we incorporate three of them. This way the flight computer can utilize 2-out-of-3 voting logic for the instrument readings, providing redundancy if one of the gyros fails.

9. MLP Structure

The MLP was designed to be similar to the Mars Phoenix Lander, a stationary platform to remain on Mars for an extended period, though while supporting a second vehicle. The MLP is two hexagon frames with cross-bracing between them, and the upper hexagon has six spokes to support an upper plate, a honeycomb structure filed on one side. A single hole is in the top of the plate to act as a path for the sample to be transferred from the Mars rover to the MAV. Three legs extend away from the MLP when the system is deployed on the Martian surface, and is protected during the descent by an EDL system. The MAV is protected by a cone attached to the plate that will retract before.

Initially, the entire MLP was made out of Aluminum 7068-T6511, with a yield strength of 6.8×10^8 N/m². During development, the struts where the engines are attached were changed to type 203 Stainless Steel, yield stress of 5.52×10^8 N/m², to provide additional thermal insulation around the engines. The total mass of the MLP sums to 67 kg, with the frame at 39 kg, each leg at 3 kg, the support plate at 9 kg, and the MAV containment cone is 10 kg. This totals to 44% below the 120 kg mass budget, allowing for redesigns if needed.

One potential problem area is the landing legs' articulating ability, which are not modeled here due to the immaturity of the design. The simulation also hold the support plate in place during flight, which may not be true in the final configuration of the EDL-MLP interface. The mass margin will allow both components to be redesigned or made out of a different material. Further, the exact connection method between the MAV and the MLP is not precise yet, and thus a more exact system will adjust the loading and the mass accordingly.

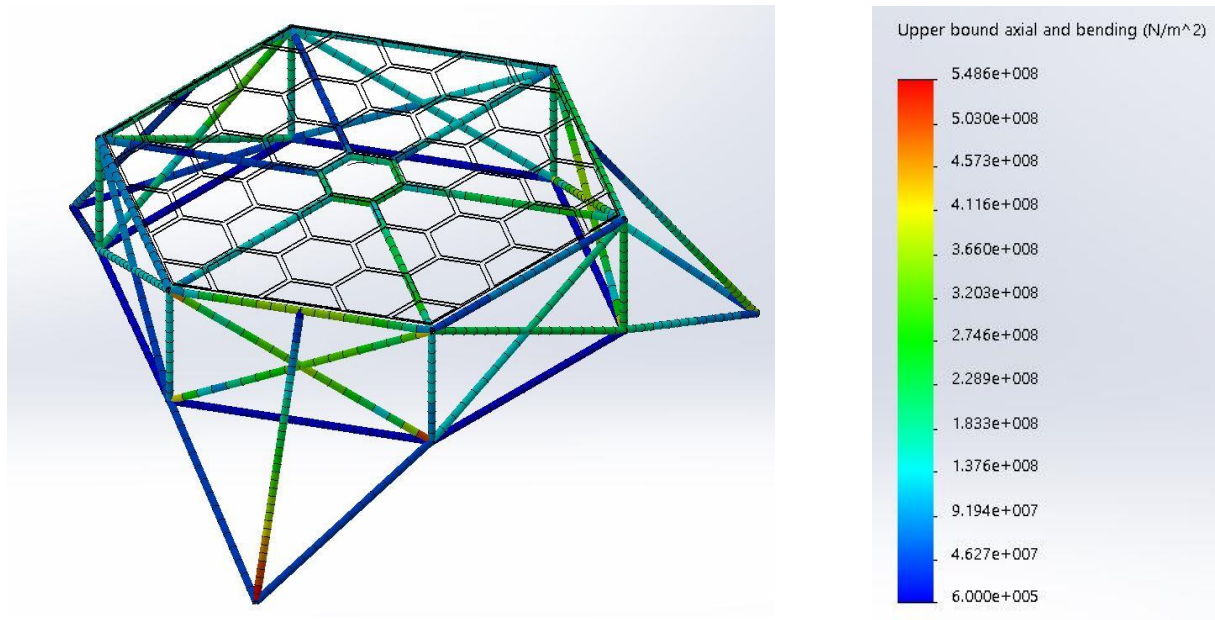


Figure 12: MLP Beam Stress during Martian descent (>1.5 margin of safety)

D. Vehicle Mass Summary

Our vehicles were designed with two tiers of margins. Because the propulsion subsystem was the major design driver of each vehicle, we had to start by choosing an initial mass as a baseline to determine the propellant required for the planned maneuvers. This initial mass had a margin built into it – that way, the mass of the vehicle could increase without having to resize the propulsion system.

On top of the planned masses we designed for a launch mass margin based on the capability of the Delta IV Heavy. This ensured that even if vehicles increase from the planned mass, they can still be launched on schedule. Early in the design process we gave the vehicles a 20% weight contingency, according to the AIAA recommendations for PDR Class 1, Category BW for the MAV/MLP and Category CW for the ERV. As our design matured, these contingencies decreased to 15% for the MAV/MLP and 10% for the ERV.

We used the two tiered margin system so that a mass increase of one subsystem does not cause a redesign of the vehicle’s propulsion system. Additionally, if we had used only the launch margin, then the propulsion subsystem would have had to be designed to according to the full mass after the margin. Then any mass increase to the propulsion subsystem would have immediately caused the spacecraft to go over its launch limit.

For a launch energy C3 of 13.73 km²/s², and subtracting the mass of the 418 kg PAF, the Delta IV Heavy is capable of launching 7903 kg. The planned total launch mass for Project Argo is 7488 kg, giving a margin of 833 kg:

379 kg (15%) for the MAV/MLP and its backshell/heatshield, and 454 kg (10%) for the ERV. The vehicle mass breakdown and its relation to the launch mass is shown in Appendix I.

E. Mission Critical Technologies and TRLs

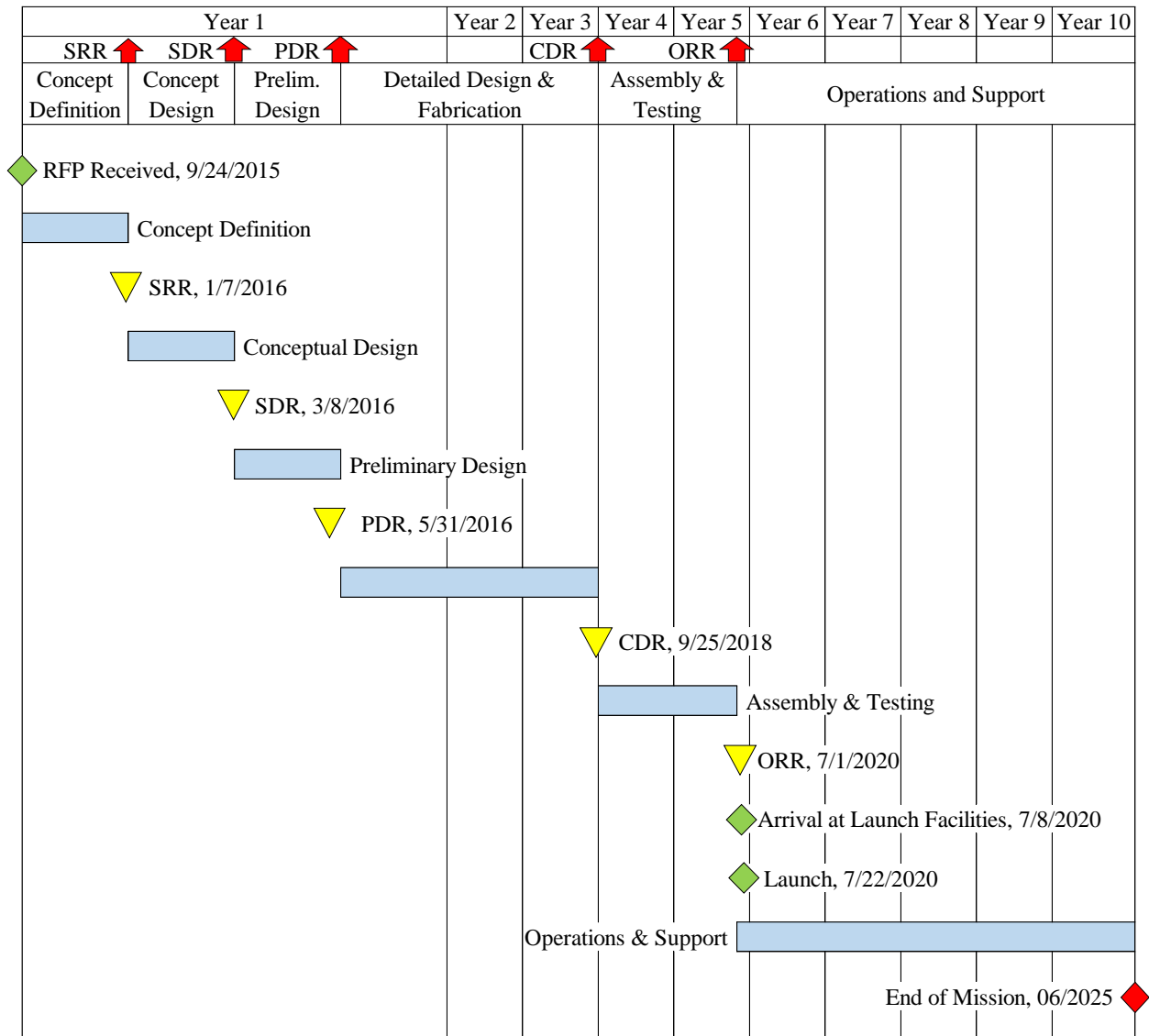
For a mission as unprecedented as Project ARGO, we decided it was best to rely on proven technologies, those with higher TRLs, as opposed to having mission critical technologies with low TRLs. Most of the critical system components have been flown on previous missions. For example, the KURS rendezvous and docking system on the ERV has been used on multiple occasions onboard the Soyuz spacecraft. Also, the solar array deployment planned for the MLP was used for the Phoenix lander. A coil truss similar to that used for sample transfer by the MAV was used to deploy solar arrays for the ISS. All of these technologies are at TRL 9.

However, some of the technologies planned for this mission have lower TRLs. The main engines on the ERV are AMBR dual-mode bipropellant engines, which are currently at TRL 6. The G-FOLD terrain relative navigation, which will be used on the MLP during landing, is also at TRL 6. The drop tank separation planned for the ERV is a new concept that will be developed alongside Project ARGO. These technologies will need to mature to TRL 7 or higher before mission operations. The required development and testing necessary to accomplish this has been accounted for in the mission timeline and will occur during the detailed design/fabrication and assembly/testing phases.

Project Management

A. Project Schedule

Project ARGO responds to an RFP received on 24 September 2015. Since that time we have completed both the concept definition and conceptual design phases of the project. We are currently in the preliminary design phase, with a PDR upcoming on 31 May 2016. We estimate detailed design and fabrication will take approximately two years. This will be followed by assembly and testing, which is also assumed will take two years. The operations and support phase is planned to be about five years long, with the end of mission occurring in June 2025. The timeline for Project ARGO from receipt of RFP to end of mission and important dates can be seen in the figure below.



B. Risk management

While space does not allow an extensive discussion of the risks met by Project Argo, a few key risks will be addressed.

1. Narrow launch window

Risk: If the launch of July 24, 2020 is scrubbed due to weather or other unforeseen circumstances, then the launch vehicle will not be able to deliver the required C3 and the spacecraft will not encounter Mars on a trajectory that will allow it to capture into orbit, resulting in loss of mission.

Explanation / Details: The launch C3 is a measure of the energy that the launch vehicle must deliver in order for a spacecraft to achieve its desired trajectory. C3 and payload mass are inversely related. For our launch opportunity, the C3 is at a minimum starting July 20 and increases with each passing day – meaning that the payload mass must decrease in order to achieve the required trajectory. Assuming that the spacecraft is at its maximum weight, a scrubbed launch means that the payload sitting on top of the rocket is now too heavy to complete its mission.

Launch date	C3 (km ² /s ²)	ERV max wet mass (kg)	Arrival date	ArrV (km/s)	Capt Δv1 (km/s)	Max prop 1st drop tanks (kg)	Max 1st drop tanks prop contin	Min prop for 17% contin (kg)
12-Jul-20	13.84	4529	2-Feb-21	2.816	0.812	1954	15%	n/a
18-Jul-20	13.44	4584	9-Feb-21	2.708	0.758	2008	21%	1864
22-Jul-20	13.48	4578	13-Feb-21	2.647	0.728	2002	24%	1775
24-Jul-20	13.73	4544	17-Feb-21	2.610	0.71	1969	25%	1724
26-Jul-20	13.86	4526	17-Feb-21	2.592	0.702	1951	25%	1701
28-Jul-20	14.41	4450	21-Feb-21	2.555	0.684	1875	25%	1650
3-Aug-20	15.51	4299	26-Feb-21	2.501	0.659	1725	22%	1580
11-Aug-20	17.8	3984	5-Mar-21	2.459	0.640	1451	14%	n/a

Launch Opportunity Propellant Masses

Project Argo can adjust to a higher launch energy while on the launch pad by removing propellant from the first set of drop tanks.

Risk Mitigation Strategy: After July 20, the trajectory C3 increases; but for about the next 40 days, the Mars arrival velocity decreases. This means less Δv to capture into Mars orbit, and therefore less propellant. We analyzed the launch opportunities over several days, comparing required C3 and the resulting maximum spacecraft mass against the new arrival velocity and the required propellant needed to capture the adjusted spacecraft mass. The results show that by removing propellant from the first set of drop tanks, the mission can continue as planned, with the only change being a later arrival date. This strategy indicates a derived requirement, that the maintenance valves of the first set of drop tanks shall be reachable and can be operated using the access hatches of the Delta IV fairing. This will be a non-trivial procedure and will require specialized equipment and techniques to handle the propellant in this environment.

The risk is also mitigated if, by time of launch, the Delta IV Heavy has undergone upgrades that allow it to launch more mass for a given C3. These upgrades are expected to be completed by the launch date, as discussed in Opportunities.

2. Conflicts with the rover mission

Risk: If Project Argo's trajectory conflicts with that of the rover, such that they are required to use the same launch facilities, then one or the other could miss the launch opportunity, resulting in mission failure.

Explanation / Details: Since there is the one viable launch opportunity in 2020, we assume that the rover will be launched at about the same time as Project Argo. Depending on the rover's launch vehicle and trajectory, it might not be possible to launch both the rover and Project Argo at the same time.

Risk Mitigation Strategy: This risk will be mitigated by communicating with the rover design team, early and often. The Delta IV Heavy is Project Argo's only launch vehicle option, and this vehicle launches from either Cape Canaveral or Vandenberg. On the other hand, assuming that the rover is close to the launch mass of Mars Science Laboratory, it is significantly less massive than our spacecraft. This means that it has more options for launch vehicles and launch facilities. Communication will ensure that the rover team does not take a facility that Project Argo requires.

3. Drop tank separation

Risk: If space environment hazards lead to a failure of the drop tank jettison system, the ERV will be too massive to complete its required maneuver with the propellant it carries, resulting in mission failure.

Explanation / Details: Micrometeoroid impacts or other space hazards could damage one of the elements of the tank jettison system, causing it to remain attached to the ERV.

Risk Mitigation Strategy: The drop tank separation and jettison system will undergo extensive development and testing during the design phase. Still, we wanted to see if the mission could be completed if jettison failed. The first set drop tanks are the largest, so if they fail to jettison this is the worst case scenario. We analyzed two scenarios, where either one or both of them fail to jettison after being depleted, forcing them to be carried by the ERV for the rest of the mission.

We found that the ERV can still complete its mission if one tank fails to jettison. The extra mass causes the individual propellant margins of the subsequent tanks to drop to between 0% and 2%; however, the cascading margin effect described previously indicates that the total propellant margin is still 20%. But because there is just enough propellant in the individual tanks, it will still be possible to complete the mission without Δv help from the previous drop tanks.

If both tanks fail to jettison, the margin of the subsequent tanks become negative. This means that if they did not receive assistance from previous tanks, they would be forced to draw propellant from the subsequent tanks, and the vehicle would run out of propellant before completing its mission.

To mitigate the risk of two tanks failing to separate, the command and actuation of the jettison mechanisms of each of the tanks will be completely independent from all other tanks. This will keep a single failure from resulting in multiple stuck tanks.

Design and testing will ensure that the likelihood of a single jettison failure is low, making a double failure even more unlikely. The fact that we can still complete our mission even if one of the largest tanks fails to separate makes this an acceptable risk.

C. Opportunity management


As with the Risk section, space does not allow for a comprehensive discussion of available opportunities; however, we will mention some of them.

1. Advanced Launch Vehicles

As previously discussed, we chose to design Project Argo based on a flight proven launch vehicle. Launch vehicles are notoriously difficult and expensive to develop, so we determined that it would be an unacceptable risk to force our design to depend on a vehicle that might fail to become flight worth in time. However, there are launch vehicles with advanced capabilities that are scheduled to be completed before the 2020 launch opportunity. If they are successful, they could be a benefit to Project Argo.

The SLS Block 1, the Falcon Heavy, and an upgraded version of the Delta IV Heavy are all scheduled to fly within the next couple years. As a reference, the following table shows the mass that each could lift to Low Earth Orbit, compared to the current Delta IV Heavy.

The main benefit that these vehicles would afford is that they would mitigate the risk of a short launch opportunity without requiring propellant to be removed from the tanks on the launch pad, as was discussed previously. The ERV would simply loaded with its full complement of propellant, and the new launch vehicles would be able to send the spacecraft on its required trajectory with launch energy to spare.

Launch Vehicle	<p>Delta IV Heavy</p>  <p>Credit: Boeing photo by Carleton Bailie</p>	<p>Delta IV Heavy with upgrades</p>  <p>Credit: ULA</p>	<p>SLS Block 1</p>  <p>VCR</p> <p>Credit: NASA</p>	<p>Falcon Heavy</p>  <p>Credit: Space X</p>	
	Planned Operational Date	In Operation	2018 ^{xxi}	2018 ^{xxii}	Late 2016 ^{xxiii}
	Payload mass to LEO (kg)	28,790 ^{xxiv}	50,000 ^{xxv}	70,000 ^{xxvi}	54,400 ^{xxvii}

2. Aerobraking at Mars

The ERV is designed to be able to aerobrake around Earth. This means that it will also be capable of aerobraking around Mars. Aerobraking would take place after MAV/MLP separation and would be used to circularize the initial elliptical orbit. The Martian atmosphere is about 100 times thinner than that of Earth, so aerobraking takes a long time. For comparison, the MRO aerobraking phase took over six months.^{xxviii} Because the ERV mission timeline is tied to that of the rover and to unknown environmental factors for the MAV on the ground, we chose not to limit the ERV's maneuvering capabilities by forcing it to rely on aerobraking. However, if the timeline allows for it, aerobraking would increase the propellant margin of the second set of drop tanks, and therefore that of the subsequent tanks.

3. KURS system upgrade

We designed our system to use the older KURS system which has been used successfully on over one hundred missions. A newer version of KURS is being developed that uses less massive hardware and fewer antenna. We chose

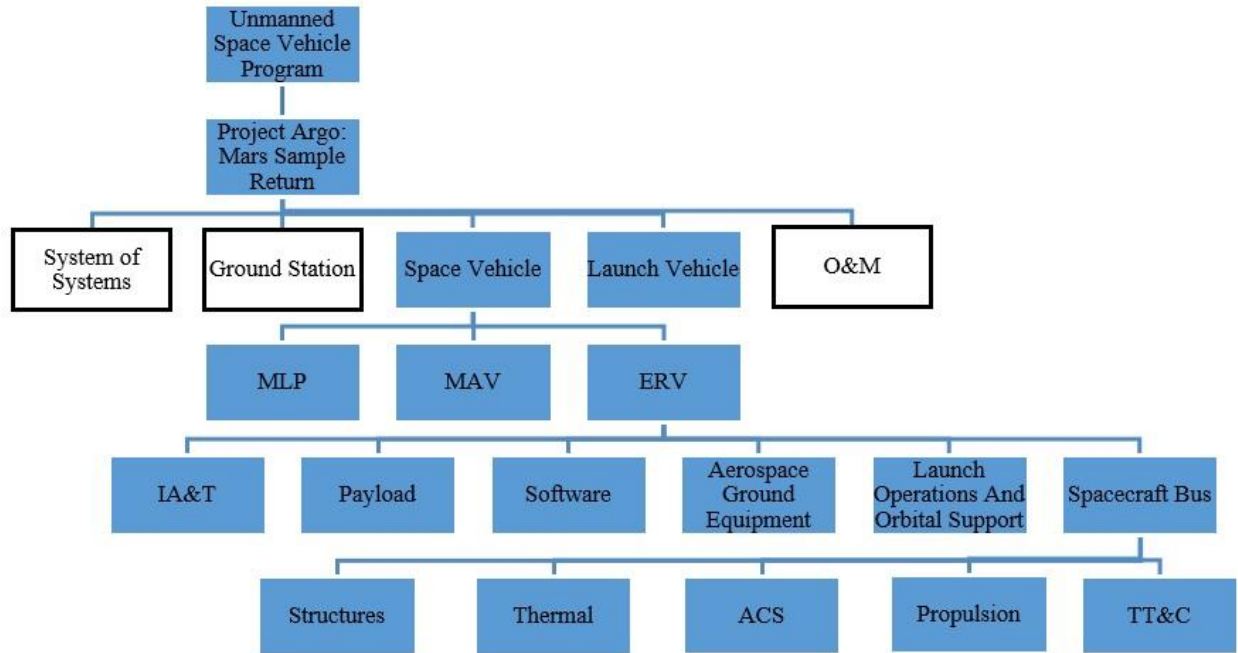
not to use this version because it has had a number of failures in the times it has been used. However, if it becomes flight proven in time to incorporate it into the ERV and MAV designs, the mass savings can be reallocated to other subsystems.

Cost estimation

Two cost models were used to estimate the cost of the mission. Both these estimates are in 2010 US dollars and have been adjusted to 2020 US dollars with a 3% inflation rate. The first was the NASA programmatic QuickCost Model. QuickCost is a top level parametric cost model using design drivers such as dry mass, SC power, design life, and the percentage of new technology used. This was done to each vehicle of the three vehicles. The summary of this estimation is shown below.

		Millions\$
Mars Ascent Vehicle	Cost 1	\$ 184.9
MLP	Cost 2	\$ 97.5
ERV	Cost 3	\$ 178.1
Mission Ops	18.25M/Year	\$ 54.75
	Total Cost FY2010\$M	\$ 1,235.3
2020 EOM	After Inflation	\$ 1,676.37
	Std Error	\$ 687.3

The second method of cost estimation is the Unmanned Space Vehicle Cost Model 2008. USCM provides cost estimating relationships (CER) and covers the highlighted portion of the WBS shown below.



USCM WBS for project Argo

Cost Model	ERV K\$	MLP K\$	MAV K\$	TOTAL (2020M\$)
USCM8	830,441	279,048	328,313	\$ 1,438
Quick Cost	178,116	97,535	184,931	\$ 1,676

To the left is a summary of both cost models. Below is a chart showing the distribution of cost using USCM8. The launch vehicle is the single highest

expense compared to the rest of the mission.

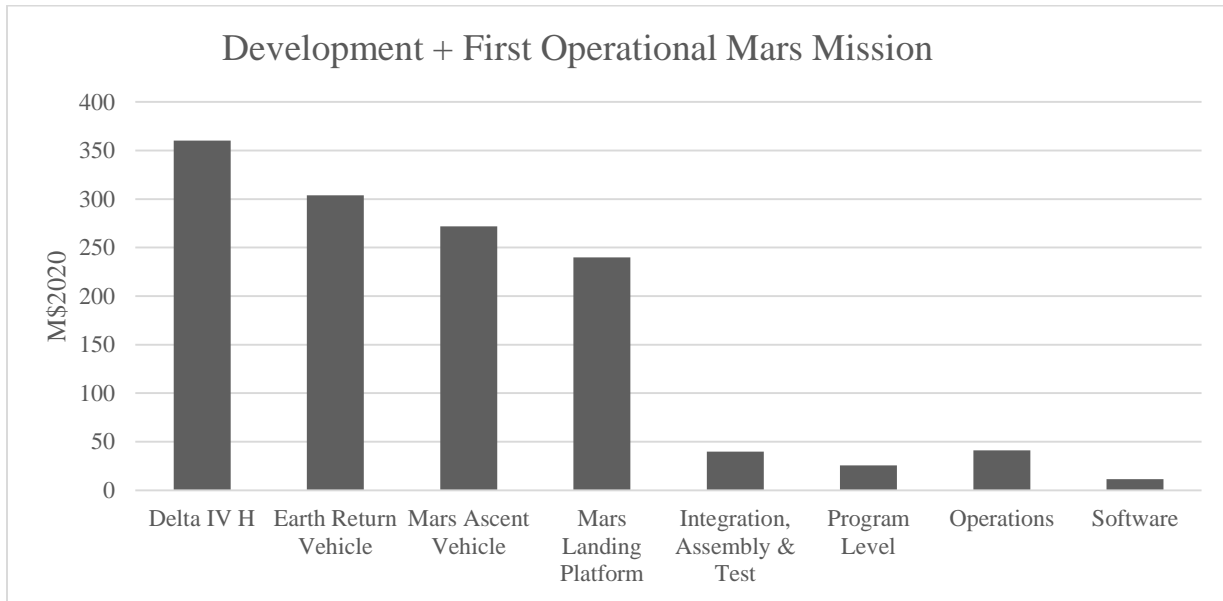


Figure x.x – Non-reoccurring cost estimate for first mission

Development + First Operational Mission	
Element	Cost estimate M\$2020
Delta IV H	360
Earth Return Vehicle	303.7
Mars Ascent Vehicle	271.8
Mars Landing Platform	239.7
Integration, Assembly & Test	39.86
Program Level	25.73
Operations	41.03
Software	11.54

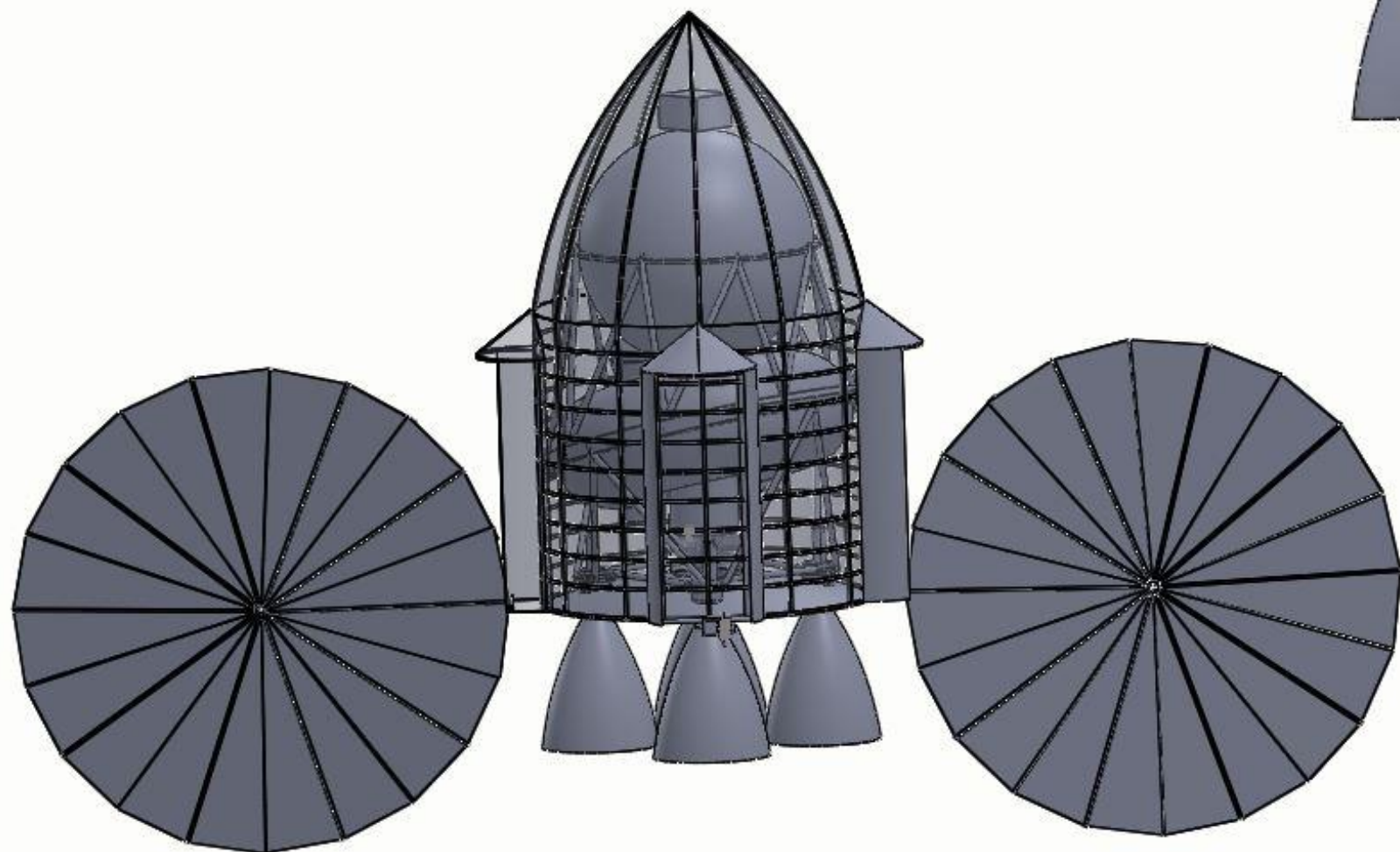
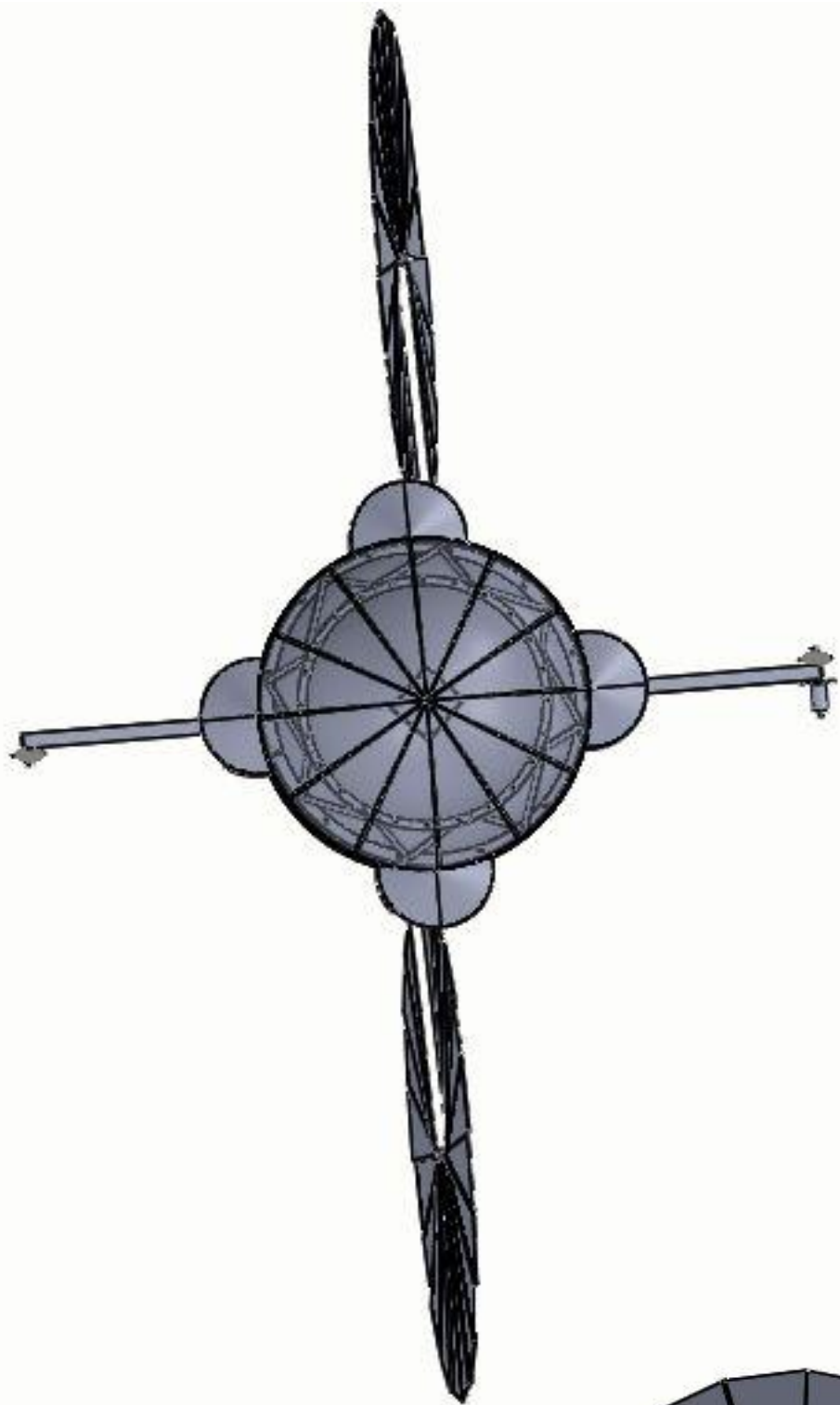
Software cost was based on total Lines Of Code and used an average of \$20 per line. This cost was in US 2008 dollars and adjusted for inflation to 2020 US dollars. The summary of the cost model for the ERV is shown on the next page. The same process was done for the MLP and the MAV.

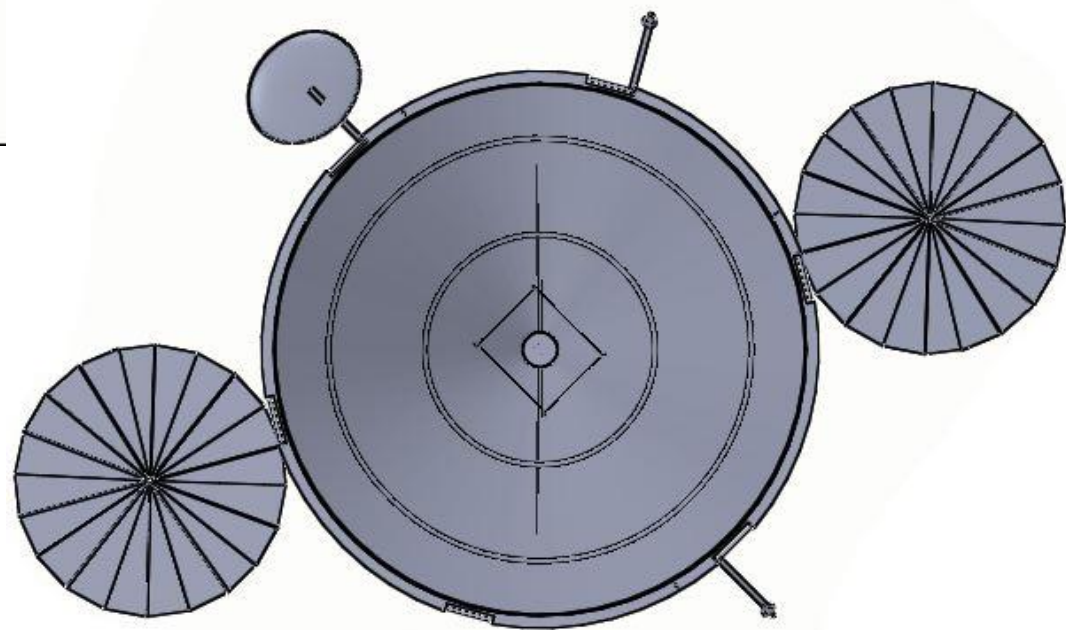
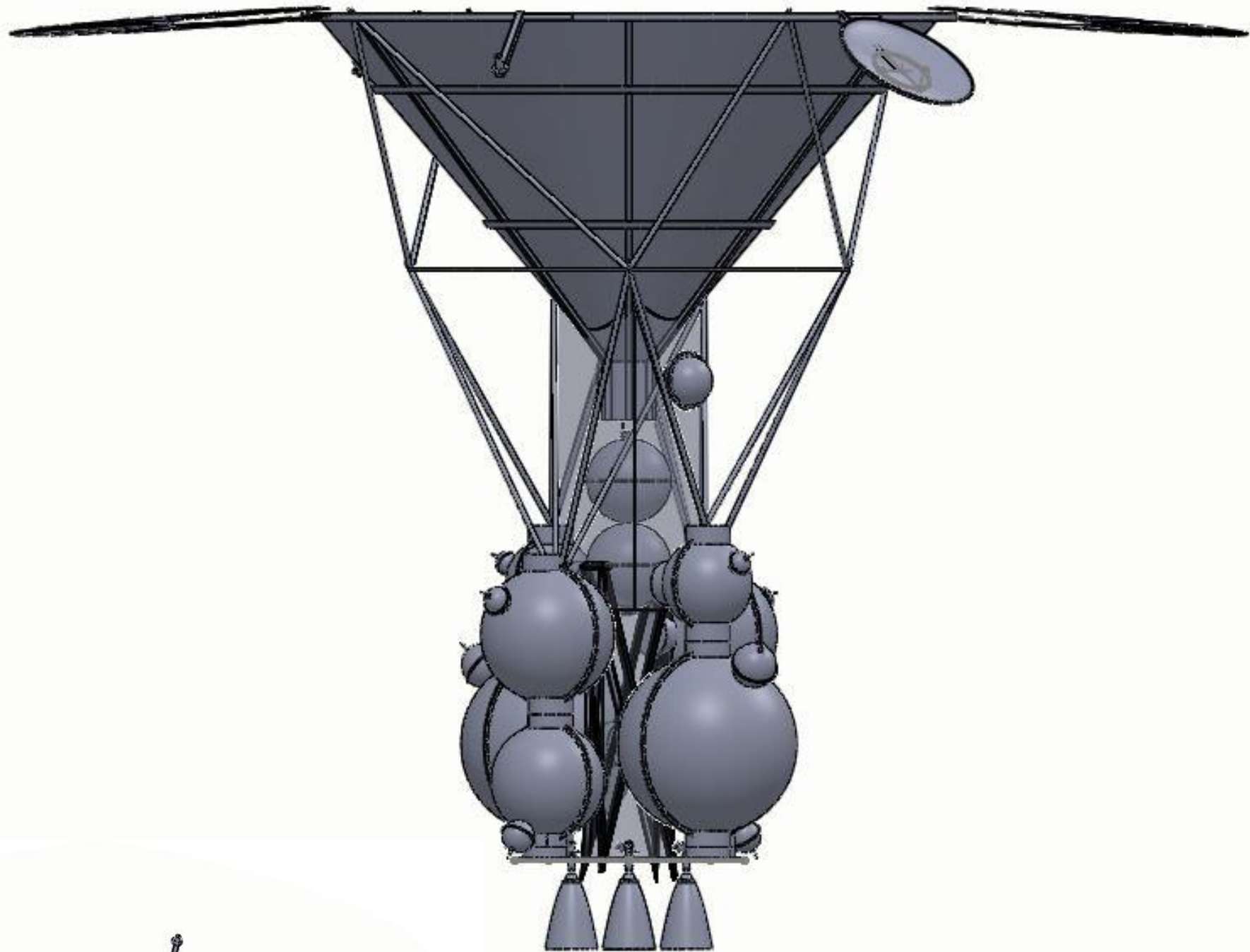
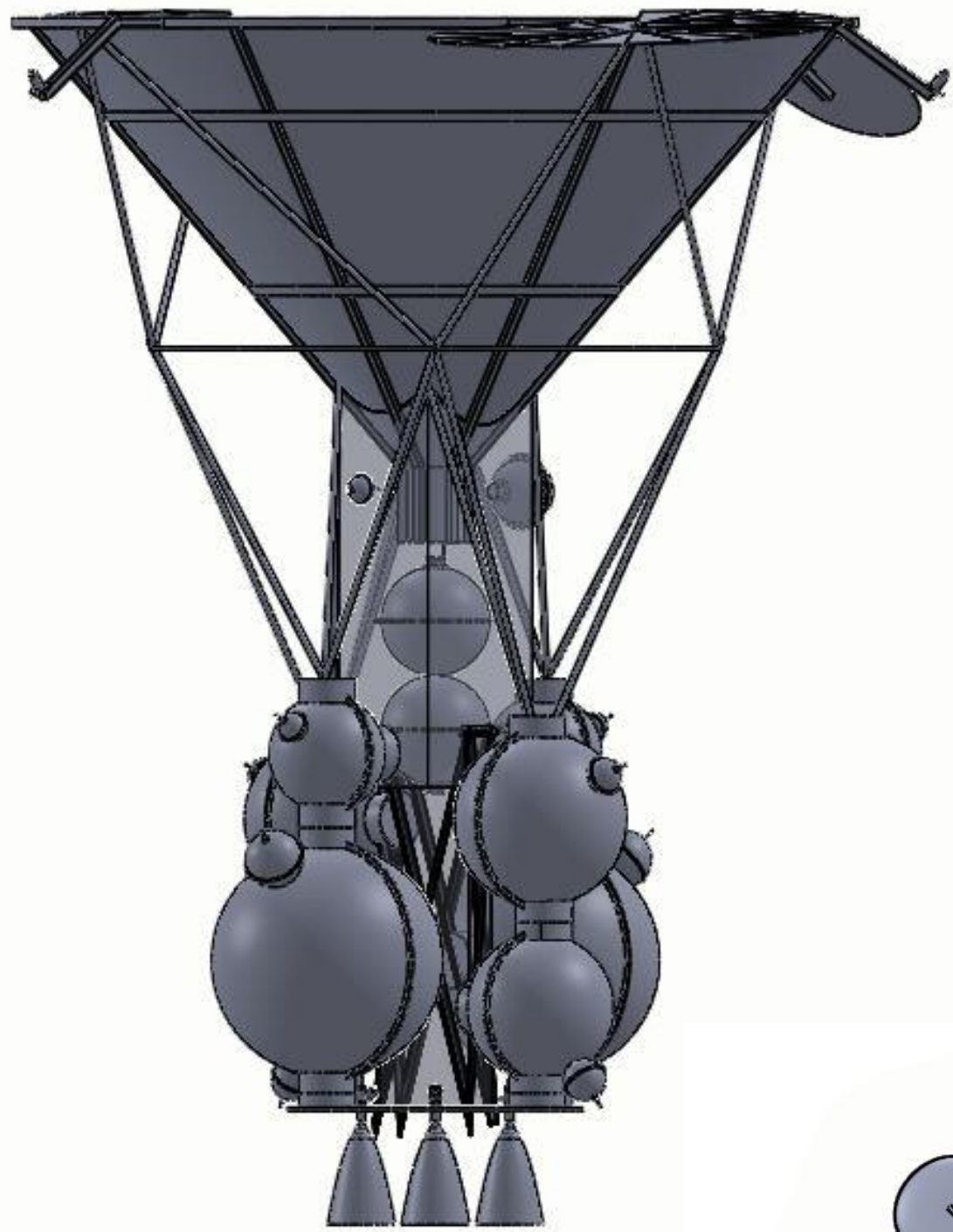
Table x.x – First mission cost breakdown

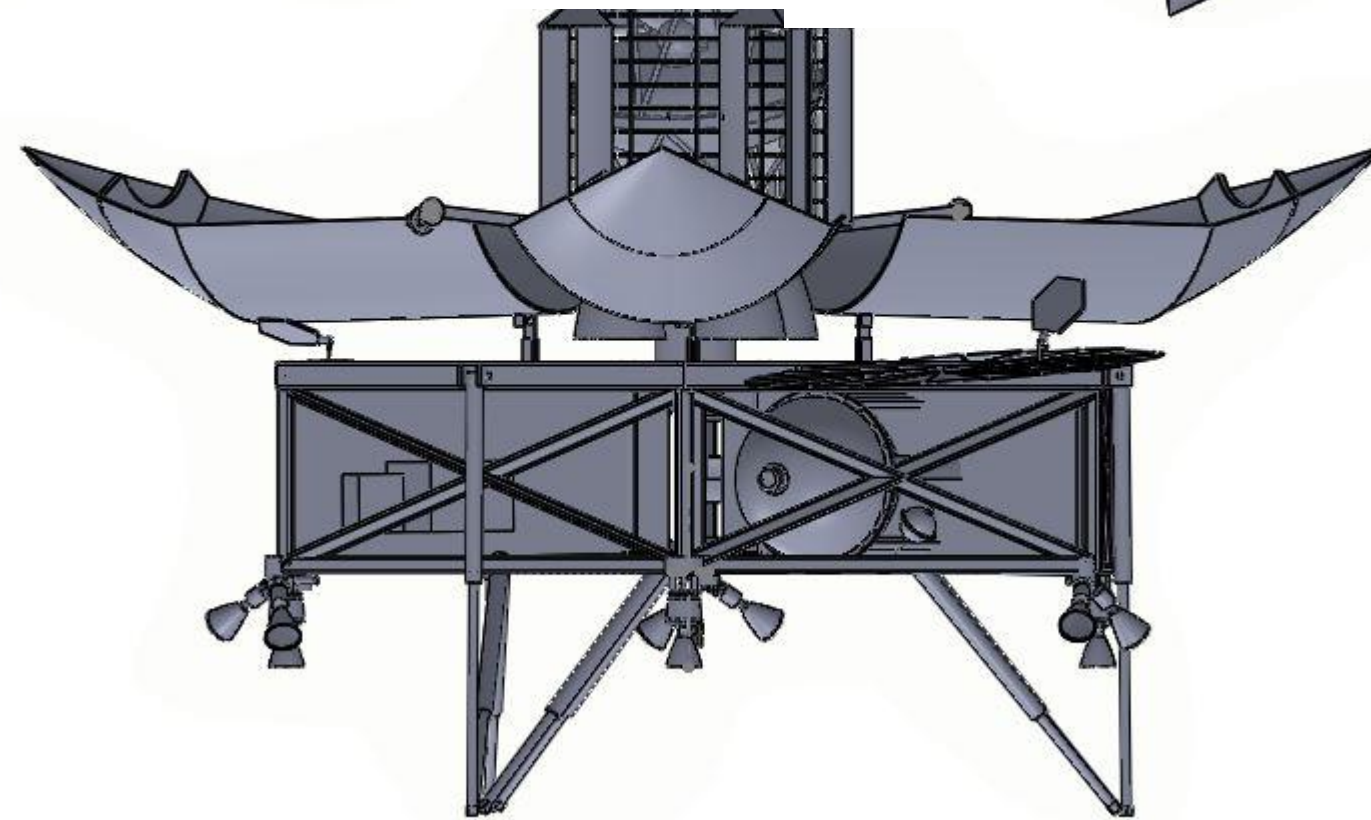
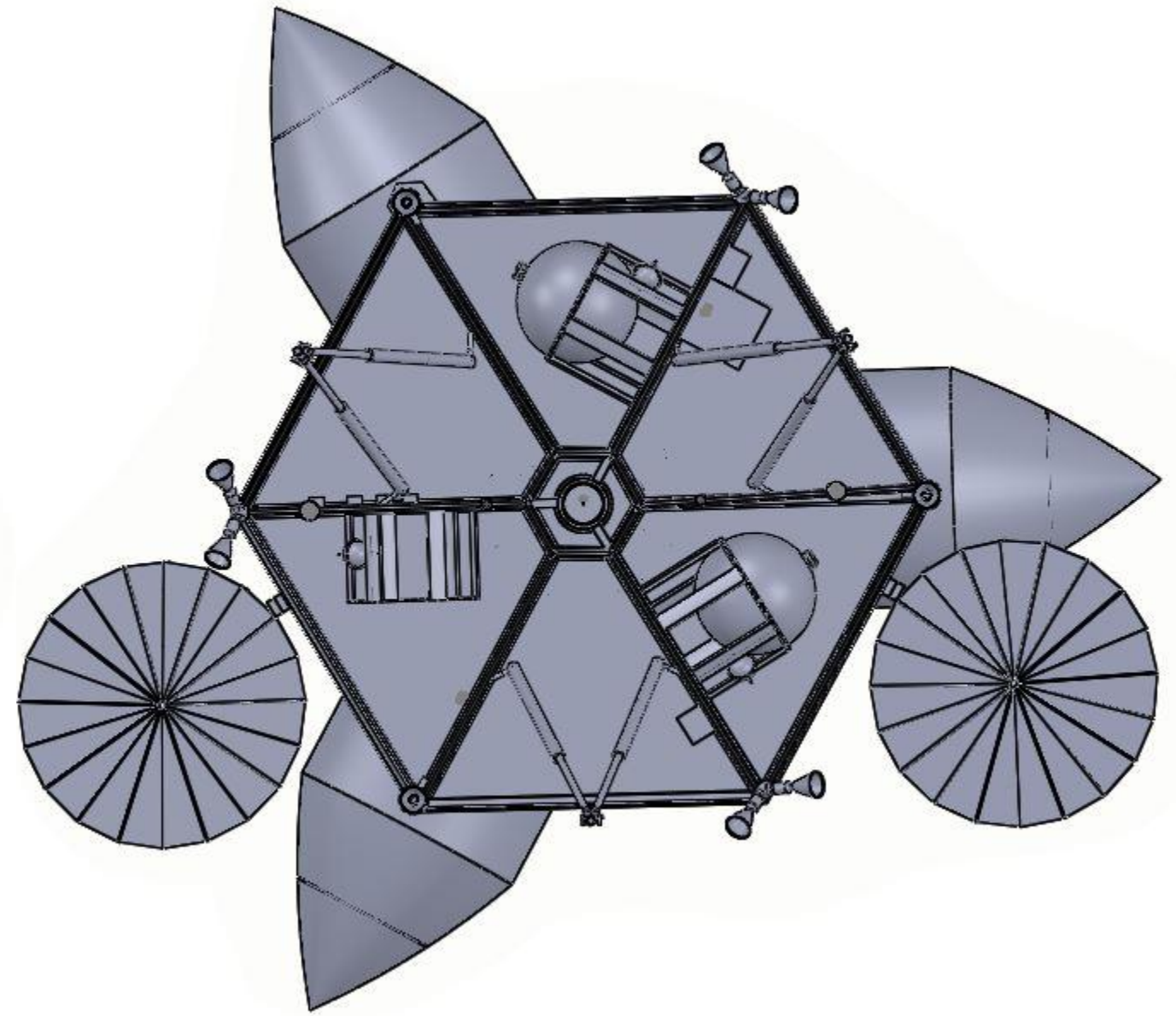
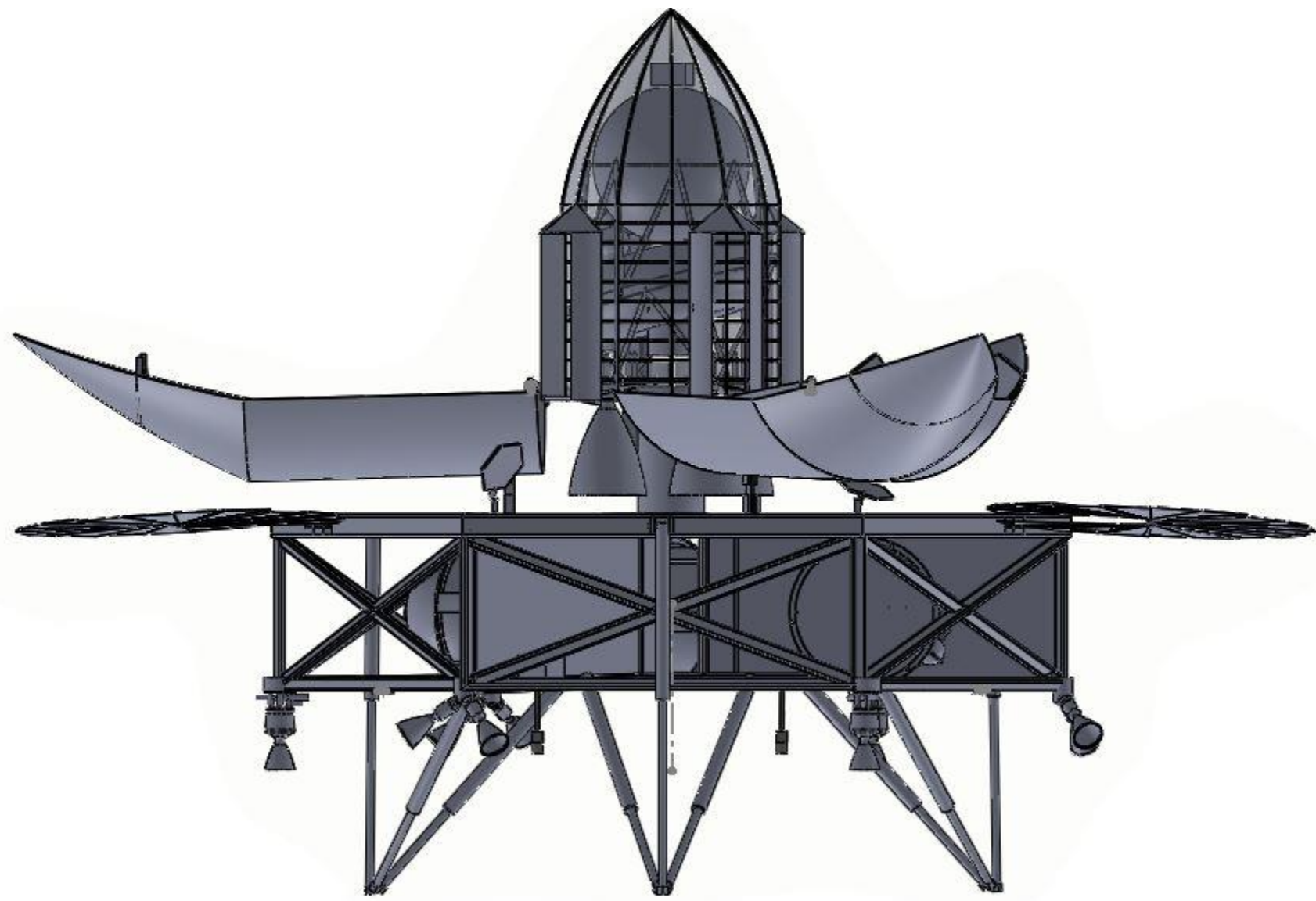
ERV				
Non Recurring Subsystems				
CER Category	Cost Drivers	Notes	CER 2010K\$	CER 2020K\$
SC Bus	750	kg	82500	111954
Structure & Thermal Control	115	kg	16586	22508
ADCS	75	kg	24300	32976
Electrical Power System	68	kg	4372	5933
Propulsion	3500	cc	1047	1421
Telemetry, Tracking, Command	-	-	26916	36526
Communications	7	kg	4326	5870
Integration, Assembly & Test	82500	\$K	16088	21831
Program Level	16088	\$K	6660	9038
Aerospace Ground Equipment	82500	\$K	416	565
TOTAL			183,212	248,622
Recurring Subsystems				
CER Category	Cost Drivers	Notes	CER 2010K\$	CER 2020K\$
SC Bus	675	kg-SC Mass	30,721	41,689
Structure & Thermal Control	115	kg-Structure+Thermo Mass	2,599	3,527
ADCS	75	kg-ACS Mass	10,287	13,959
Electrical Power System	68	kg E-Power Mass	2,203	2,990
Propulsion	378	kg- Kick Motor Mass	11,048	14,993
	3600	s-Burn time		
Telemetry, Tracking, Command	42	kg-TT&C mass	5,538	7,515
Communications	7	kg-Comm Payload Mass	1,323	1,795
Integration, Assembly & Test	30721	\$K-	3,809	5,169
Program Level	3809	\$K	1,219	1,654
Launch Operations & orbital Support	360,000	\$K	360,000	488,528
TOTAL			\$ 428,747	\$ 581,819
			ERV Total \$K	\$ 830,441

Appendix I: Vehicle Mass Summary

Mars Ascent Vehicle (MAV)	Mass (kg)		Mars Launch Platform (MLP)	Mass (kg)	
Payload	12.3		Payload	18	
Structure	35.5		Structure	67	
Thermal	3.1		Thermal	0	
ACS	7.092		ACS	2	
Power	12.46		Power	21	
Cabling	0		Cabling	0	
Propulsion	77.0		Propulsion	91	
Telecomm	3.2		Telecomm	3	
CDS	1.0		CDS	5	
Current dry mass	151.6		Current dry mass	206.8	
Planned dry mass	190.0		Planned dry mass	294.0	
Vehicle mass margin	38.4	20%	Vehicle mass margin	87.2	30%
Total propellant	550		Total propellant	506	
MAV current wet mass	701.6		MLP current wet mass	712.8	
Planned wet mass	740.0		Planned wet mass	800.0	
Subtotal landing mass	1540				
Heat shield	390				
Backshell, parachute	596				
Entry Descent Landing (EDL)	986				
Planned wet mass	2526				
Earth Return Vehicle (ERV)			ERV + MAV	Mass (kg)	
Payload	40.8		C3 (km^2/s^2) =	13.73	
Structure	118.32		Delta IV (Heavy) capability	8321	
Thermal	10.6		ERV + MAV/MLP/EDL	7070	
ACS	60.5		Delta IV (Heavy) PAF mass	418	
Power	23.86		Planned total launch mass	7488	
Cabling	38.6				
Propulsion	371.52				
Telecomm	7.2				
CDS	5.2		Total launch mass margin	833	
Current dry mass	677				
Planned dry mass	772				
Vehicle mass margin	95	12%	MAV/MLP Launch mass margin	379	15%
Total propellant	3772		ERV Launch mass margin	454	10%
Planned wet mass	4544				







Acknowledgments

Pyxis Aerospace would like to thank Dr. Don Edberg for his continuous patience, deadpan humor, and valuable insights as he shared his extensive knowledge of all things spaceworthy.

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n.d.

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