Interplanetary Spacecraft

Team 12

Alliance: Foxtrot

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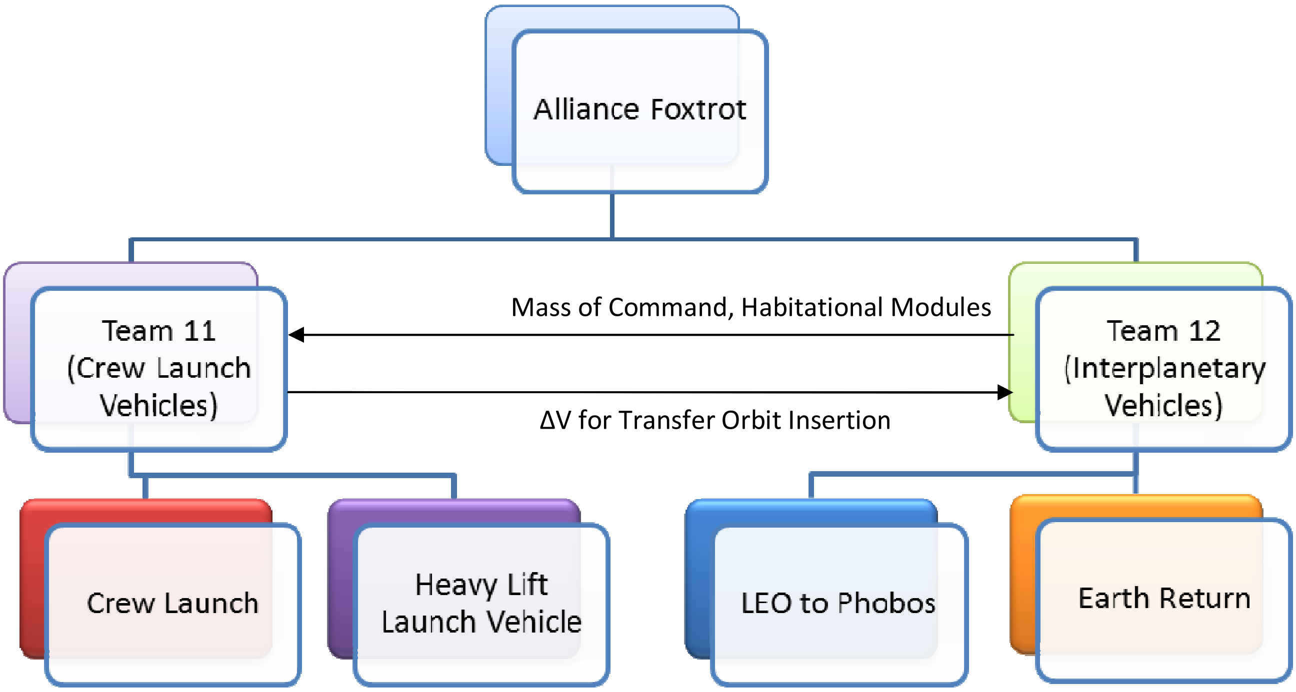
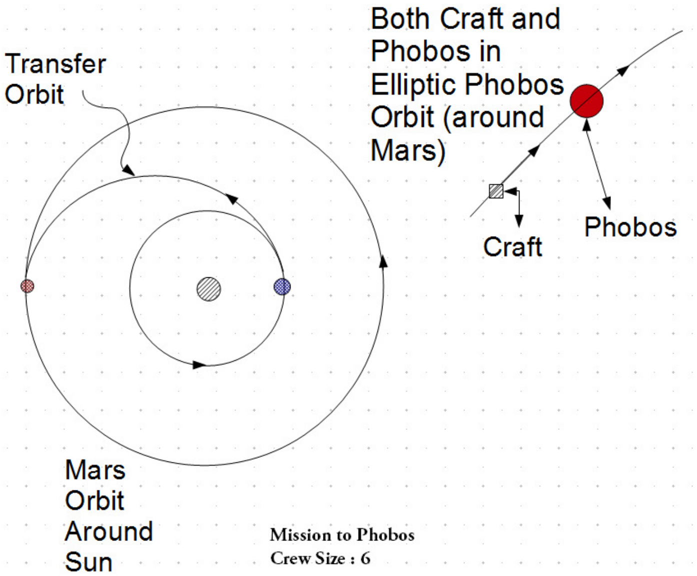
Alliance Level Breakdown



As depicted in the hierarchical diagram of the Alliance above, we (Team12) are providing Team 11 with our mass calculations of the Command Module and Habitational Module. The gross mass of both of these shall be the "payload" mass for Team 11 in our alliance. As far as we are concerned, Team 11 is providing us with the initial ΔV that is required to insert our craft into the Mars transfer orbit.

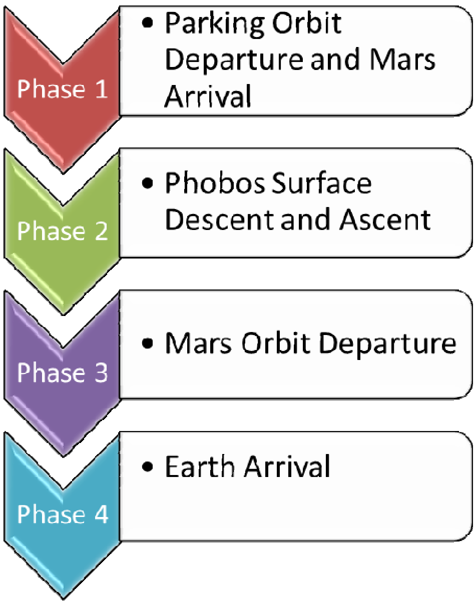
The overall schematic of our transfer orbit, is shown below:



[Figure 1]

Mission Overview

Our mission design will comprise of the second part of an integrated concept for a human mission to Mars's satellite; Phobos. With the proposed number of crew members of 6, this high-cost, high-risk expedition that routes from Mars Transfer Orbit, to Phobos's surface, to Lower Mars Orbit (LMO), and back to Earth will have approximated halt duration of 500 days. Because of the scale, there is no room for mistakes and to minimize errors we have decided to split the expedition into several phases (Figure 2) for easy monitoring and execution. (Refer to Figure 1, previous page)



[Figure 2]

Phase One (Parking Orbit Departure & Mars arrival)

The first phase of the mission consists of the cruising phase which takes the vehicle from its initial point at the earth periapsis to its terminal location at Mars proximity and Mars Orbit insertion phase. Mars Orbit insertion for this particular mission will be carried out at an altitude similar to Phobos's orbit around Mars by utilizing the aerocapture procedure. This is done so that the vehicle can rendezvous with Phobos solely by moving around the trajectory of its new orbit and thus eliminating the need for any propulsive maneuver. Clearly this will be beneficial as it can save mass needed for propulsion and in turn reduce the cost to build the vehicle.

Phase Two (Phobos surface Descent, Ascent)

Upon getting in line with Phobos, the spacecraft will now be presented with the opportunity to perform Entry, Descent and Landing (EDL) to the surface of Phobos, the crew will manually detach the first phase of the vehicle, which is the Habitational Module that consists of everything required to sustain human life during the trans orbit flight, from the rest of the vehicle. At the same time, the Command Module will descend and land on Phobos's surface. Assuming that the gravity is very small and negligible we would not need any heat-shield and therefore a direct angle landing will suffice and will be more cost-efficient.

Phase Three (Mass Orbit Departure)

After the completion of the research mission on the surface of Phobos for the particular time length, astronauts will return to Earth by using the Hohmann transfer. The Command Module will rendezvous back with the Habitational Module. Next, a propulsive maneuver will place the craft into the Earth return trajectory.

Phase Four (Earth Return)

Finally, as the vehicle arrives in the proximity of Earth the same orbit insertion will be performed as what was done to enter Mars's orbit. This will get the astronauts into Lower Earth Orbit (LEO). From there, the craft will make its way to the surface of the Earth. For this second EDL we will need a heat-shield, as the atmosphere is thick enough to burn the Command Module if not adequately protected. As the capsule achieves subsonic speeds, the heat shield will be jettisoned and parachute will be used to permit soft-landing.

Trajectory Analysis

(Also See Appendix B)

The trajectory is modeled as a patched conic approximation with three segments, reflecting the orbits around the three bodies: The escape from Earth, the transfer about the Sun, and the entry into Martian orbit (Figure 4). The ranges of each segment of the approximation is determined by the gravitational sphere of influence of each body in the system, with the assumption that the exit from and entrance to the sphere of influence is parallel to the velocity of the gravitational body.

To calculate the changes in velocity we made a few necessary assumptions. We assumed that the Earth and Mars are in a coplanar orbit around the Sun. This assumption made possible a transfer orbit via Hohmann Transfer (Figure 1). This interplanetary trajectory design may change as we research more into the orbital inclinations of the Earth and Mars. We also assumed that our propulsive maneuvers are instantaneous. Without further research into the specifics of our engines we are unable to qualify any further data.

We assumed that all propulsive maneuvers are parallel to the current velocity vector, with no re-direction, which reduces the problem to one dimension. We also assumed there will be no course correction maneuvers en route to Mars. We also neglected the gravity of Phobos at this time, assuming that it would be more like a docking maneuver, since Phobos's gravity is only 0.0008g.

Variables Used:	Subscripts:
a - semi-major axis	SoI – Sphere of Influence
M – Mass	E – Earth
R - Orbital radius/ radius	M – Mars
V - Orbital velocity	PH - Phobos
μ - Gravitational parameter, M*G	S – Sun

1) Solar Transfer Orbit

First, calculate the periapsis velocity of the elliptical transfer orbit from Earth to Mars.

$$a_s = \frac{(R_{SE} + R_{SM})}{2} \text{ (1)}; \quad V_4 = \sqrt{\mu_S \left(\frac{2}{R_{ES}} - \frac{1}{a_S} \right)} \quad \text{(2)} \; V_4 = 32.7318 \text{ km/s}$$

2) <u>Earth Orbit Escape</u>

The hyperbolic orbit required to escape Earth and reach Mars is that which will provide this velocity (V_4) relative to the Sun when the spacecraft leaves Earth's sphere of influence. The radius of this sphere of influence is given by:

$$R_{SoI_E} = R_{SE} \cdot \left(\frac{M_E}{M_s} \right)^{\left(\frac{2}{5}\right)} \text{ (3)}$$

At this radius, the velocity of the escaping spacecraft (relative to Earth) must be:

$$V_3 = V_4 - V_E \text{ (4)} \qquad V_3 = 2.9267 \text{ km/s}$$

This requires a hyperbolic trajectory with a velocity at infinity (V_∞) given by:

$$V_\infty = \sqrt{V_3^2 - \frac{2\mu_E}{R_{LEO}}} \text{ (5)} \; V_\infty = 2.7754$$

Using V_∞, we can find the periapsis velocity of the hyperbolic escape orbit, V_2:

$$V_2 = \sqrt{\frac{2\mu_E}{R_{LEO}} + V_\infty^2} \quad \text{(6)} \; V_2 = 11.3534$$

Subtracting the circular Earth orbit velocity, V_{LEO}, we obtain the ΔV_1.

$$V_{LEO} = \sqrt{\frac{\mu_E}{R_{SE}}} \qquad \text{(7)}; \; \Delta V_1 = V_2 - V_{LEO} \quad \text{(8)} \quad \Delta V_1 = 3.5689$$

This is provided by our alliance partner, and so is not included in the final mass calculations for our spacecraft.

3) <u>Mars Orbit Capture</u>

Knowing the geometry of the transfer orbit, we can calculate the velocity relative to the sun (V_5) of the spacecraft when it encounters Mars, as well as its velocity relative to Mars at that same time (V_6).

$$V_5 = \sqrt{\mu_S \left(\frac{2}{R_{SE}} - \frac{1}{a_S} \right)} \qquad \text{(9)}; \; V_6 = V_5 - V_M \qquad \text{(10)} \; V_6 = 2.26630$$

Considering V_6 as the velocity of the spacecraft upon entering Mars's sphere of influence, we can obtain the hyperbolic encounter orbit parameter, V_∞, and use it to calculate the periapsis speed of the hyperbolic encounter orbit (V_7) at the atmospheric entry radius (R_{ent}).

$$V_\infty = \sqrt{V_6^2 - \frac{2\mu_M}{R_{SoI_M}}} \quad (11); \quad V_7 = \sqrt{\frac{2\mu_M}{R_{ent}} + V_\infty^2} \quad (12) \quad \Delta V_7 = 5.5915 \text{ km/s}$$

If we assume that the entirety of the mars capture ΔV is provided by one impulsive aerobraking maneuver, there are two ΔV required to insert into Phobos's orbit: ΔV_{1a}, provided by aerobraking at the height of the atmosphere, and ΔV_{1b}, provided by a burn at R_{LMO} to circularize the orbit.
In fact, even if the approximation of ΔV_{1a} as impulsive is false and several braking passes are necessary, the sum of the ΔV necessarily equals ΔV_{1a}.

First, find the semi-major axis of the transfer orbit from entry to LMO, and find the periapsis velocity of this orbit (V_8).

$$a_{hyp-LMO} = \frac{(R_{ent} + R_{LMO})}{2} \quad (13); \quad V_8 = \sqrt{\mu_M \left(\frac{2}{R_{ent}} - \frac{1}{a_{hyp-LMO}}\right)} \quad (14) \quad V_8 = 4.1961 \text{ km/s}$$

The total ΔV provided by the aerobraking maneuver (ΔV_{2a}) is given by

$$\Delta V_{1b} = V_7 - V_8 \quad (15) \quad \Delta V_{1b} = 1.3954$$

The apoapsis velocity of the entry-LMO transfer orbit (V_9) is given by:

$$V_9 = \sqrt{\mu_M \left(\frac{2}{R_{LMO}} - \frac{1}{a_{ent-LMO}}\right)} \quad (16) \quad V_9 = 1.6002 \text{ km/s}$$

Subtracting V_9 from the orbital velocity at R_{LMO} (V_{10}) gives ΔV_{2b}.

$$V_{10} = \sqrt{\frac{\mu_M}{R_{LMO}}} \quad (17); \quad \Delta V_{2b} = V_{10} - V_9 \quad (18) \quad \Delta V_{2b} = .5533 \text{ km/s}$$

For the return orbit (Figure 4), we again performed the Earth escape calculations, substituting Mars for Earth in all cases. The EDL at Earth is a direct EDL, going straight from the solar transfer orbit to entry into Earth's atmosphere, using Equation 15 with a final velocity of 0 (matching Earth.)

When these calculations are performed, we get a total round-trip ΔV of 3.822 km/s, where 2.427 km/s is provided propulsively. This is probably a slightly lower than realistic value, since it does not include any course-correction or orbital inclination adjustment maneuvers.

If aerobraking is used, approx. 20% of the mass of the craft being braked is added as a heatshield, but since the planned trajectory requires a huge delta-V upon Earth aerocapture anyway, this 20% number is added to the dry mass of the spacecraft in both scenarios, and Martian aerobraking does not change the dry mass of the spacecraft. It does, however, lengthen the amount of time spent in travel by about 55 days (2 passes, see Appendix B), which increases the amount of food required. Even with this, the mass of the aerobraking spacecraft is about 81% that of the thrust-braked craft.

The total time required to reach Mars from Earth is about 260 days, as is the time to return. In addition to a stayover time of 500 days and the 55 day aerobraking period, this amounts to a 1075 day trip, almost 3 years.

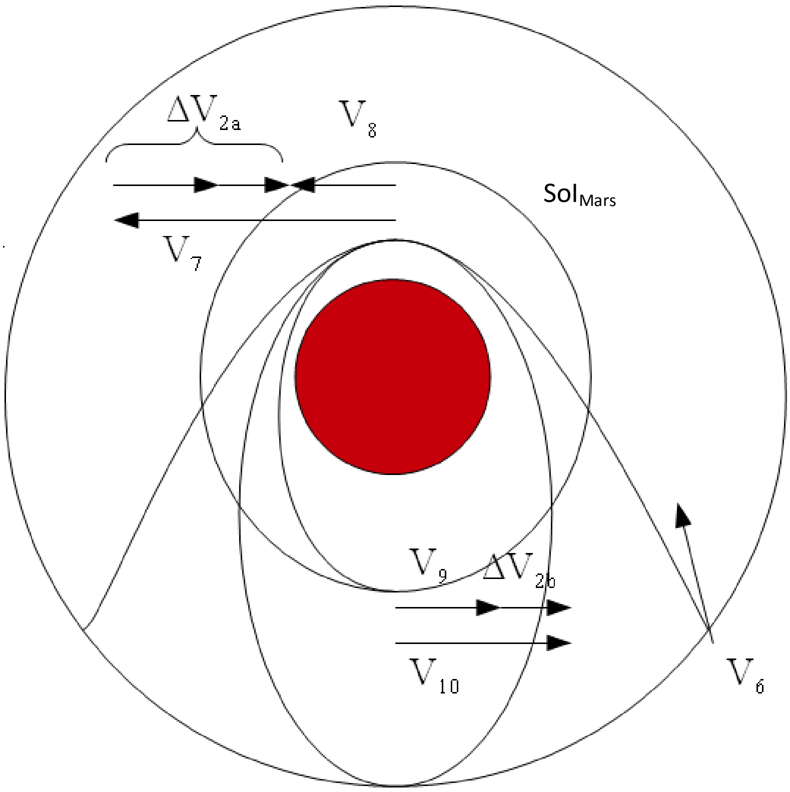
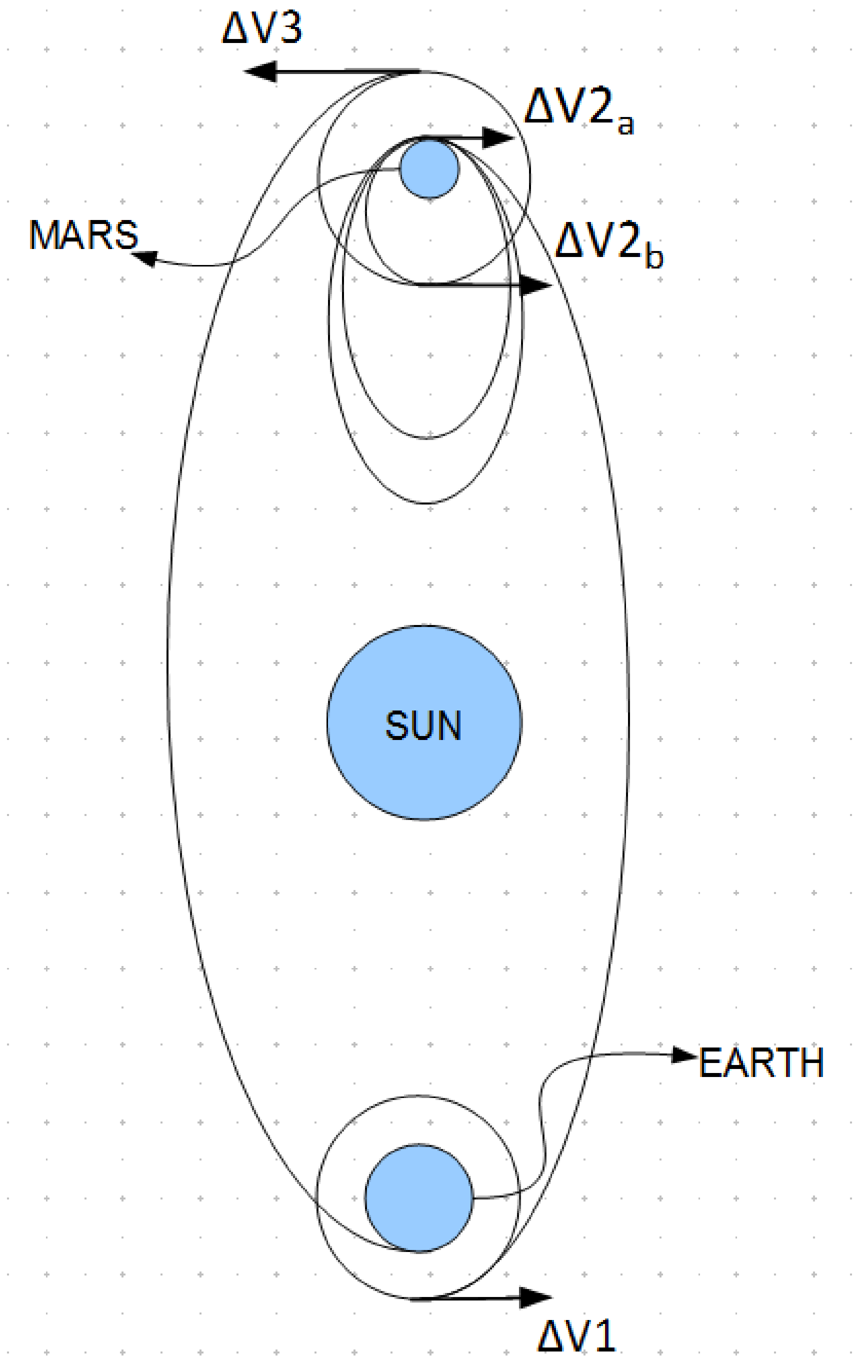


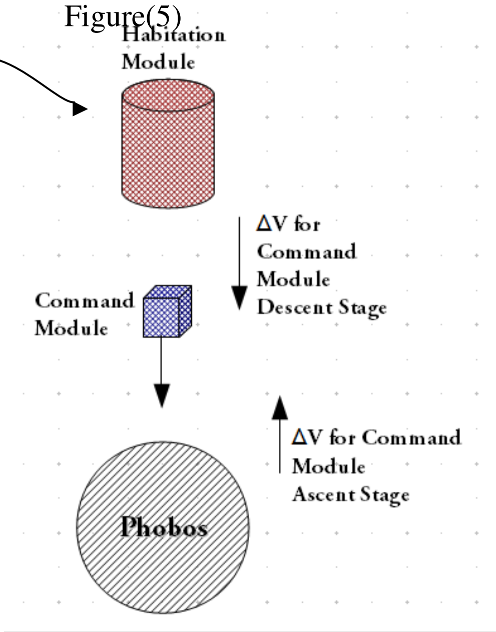
Figure 3



[Figure 4]

This is a schematic that represents the EDL stage, where we land on Phobos and then Depart to return to the Habitation Module.

Figure(5)



Vehicle Configuration & Detail

Habitation Module (Fig. 6)

Living Compartment

- *Sleeping Quarters*
- *Hygiene Facilities*
- *Exercise Area*

Working Compartment

- *Small Laboratory*

Command Module (Fig. 7)

Crew Module (Command Cente

Service Module

- *Engine*
- *Propulsion*

Launch Abort System (LAS)

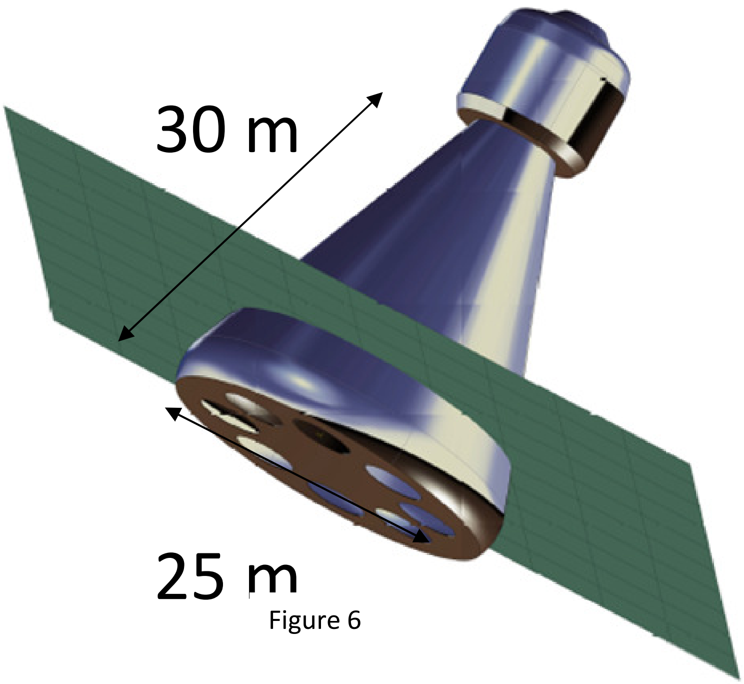
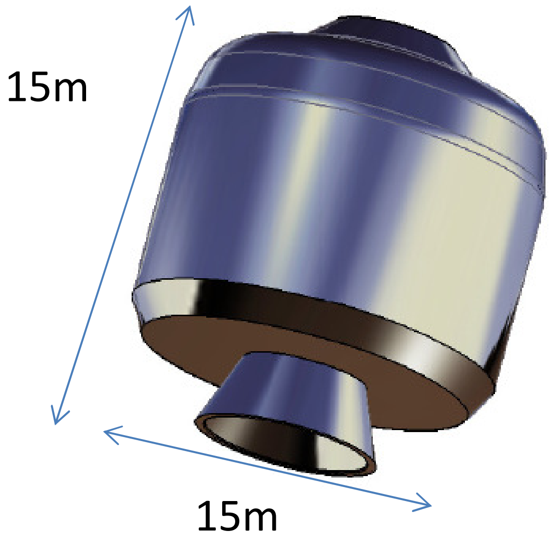


Figure 6



CATIA v5 Models

Figure 7

Proposed Rocket Engines for Interplanetary Travel

NERVA – Nuclear Engine for Rocket Vehicle Application. The NERVA engine is a nuclear thermal propulsion system that is capable of a specific impulse greater than twice most chemical rocket engines at 825 seconds (Wikipedia). It also provides a high amount of thrust, making it desirable for a Hohmann transfer maneuver to reach Mars. The drawback of such a rocket engine lies in the high cost; using nuclear thermal propulsion for the interplanetary travel costs one billion dollars more than conventional chemical rockets.

Merlin – the Merlin rocket engine was originally developed by SpaceX. It utilizes liquid oxygen and a highly refined form of kerosene known as RP-1 to produce an average amount of specific impulse at 304.8 seconds (Wikipedia). Although this rocket is significantly cheaper than the NERVA rocket, the mass would have to be significantly larger due to the lower ΔV value. The following graph presents this disparity in initial mass versus ΔV for both the Merlin and NERVA rocket engines:

Choice of Propulsion : NERVA (Justification below)

Mass Calculations

Estimating the mass of the Command Module and habitation Module was provided by a simple equation based on the number of astronauts included in the mission. The estimated mass of the Command Module is approximately equal to one ton per astronaut (6000 kg). This number includes the mass of any EDL systems. The habitation Module is estimated at a standard amount of eight tons plus an additional two tons per astronaut (20,000 kg). Included in the total mass is the mass of the six astronauts. Assuming the average weight of an astronaut to be about 140 lbs on Earth, this translates to a mass of 64 kg, adding an additional mass of 384 kg. To provide sustenance for the astronauts on their journey, an estimated five kg of consumable mass per astronaut per day was included. This is dependent on the time of the journey. The journey will last 259 days one way with 55 days of aero braking and a 500 day stopover at Mars. This means that there needs to be a mass of consumables for 1073 days which translates to a mass of 47190 kg. Further calculations are based on the assumption that all mass in the crew Modules remains contained within and is not expended into space, keeping the mass of these Modules constant throughout the journey.

The mission overview requires aero braking and aero capture maneuvers. This type of maneuver requires the use of a heat shield to protect the astronauts and equipment from the high heat that these types of deceleration necessarily demand. An estimated 20% of the crew Modules and dry mass of the rocket is given to be the mass of the heat shield.

All of the fore mentioned masses constitute the payload mass of our rocket. To find the initial mass of the space craft the mass must be related to the change in velocity. For this we used Tsiolkovski's rocket equation, modified for our use.

Initial Mass = Final Mass * exp(deltaV/exhaust velocity) [Table M1]

The final mass is given by the inert mass and the payload mass. The exhaust velocity is given by the following equation that is dependent on the specific impulse of the rocket.

Exhaust Velocity = Specific Impulse * the standard free fall acceleration

Subtracting the final mass from the initial mass gives the mass of the propellant. With the propellant mass and the inert mass the inert mass fraction can be found.

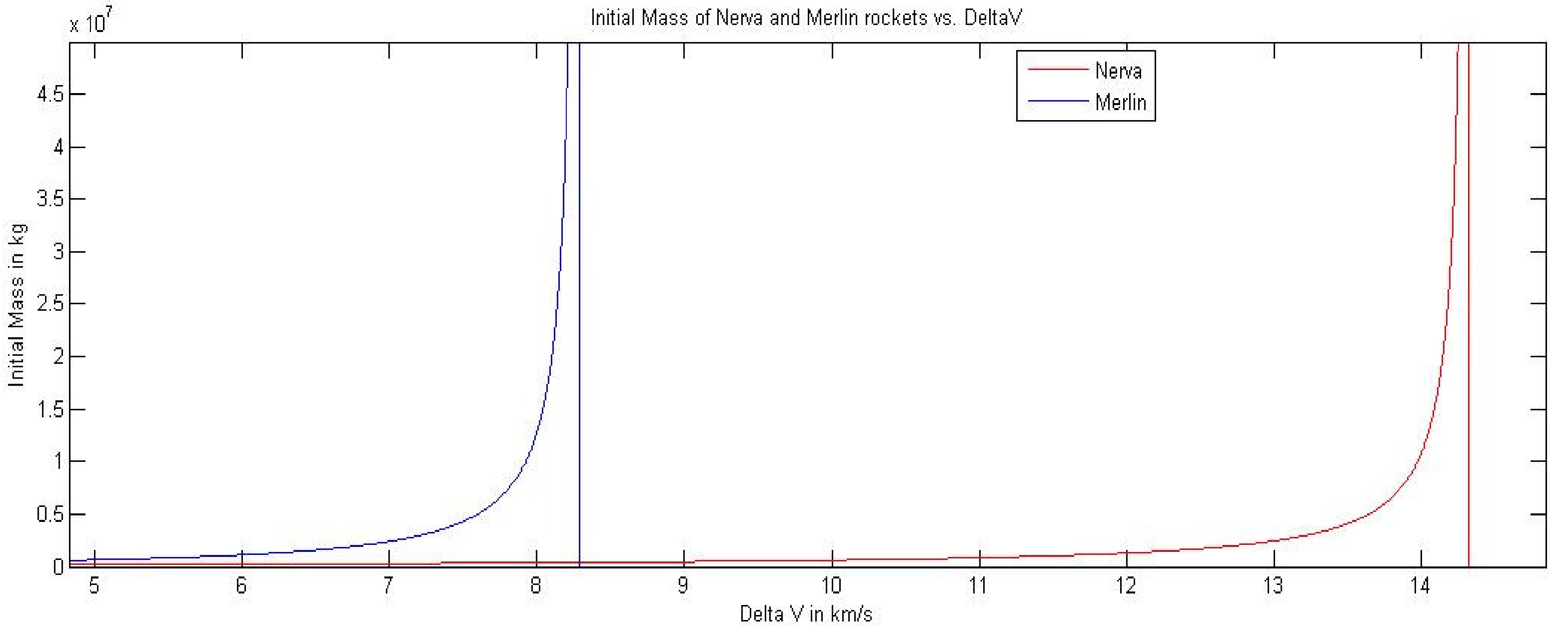
F inert = M inert / (Mprop + Minert) (Values from Astronautix.com, Reference #11)

Calculating the two previously discussed rockets simultaneously provides the following values.

Table of Masses [Table M1]

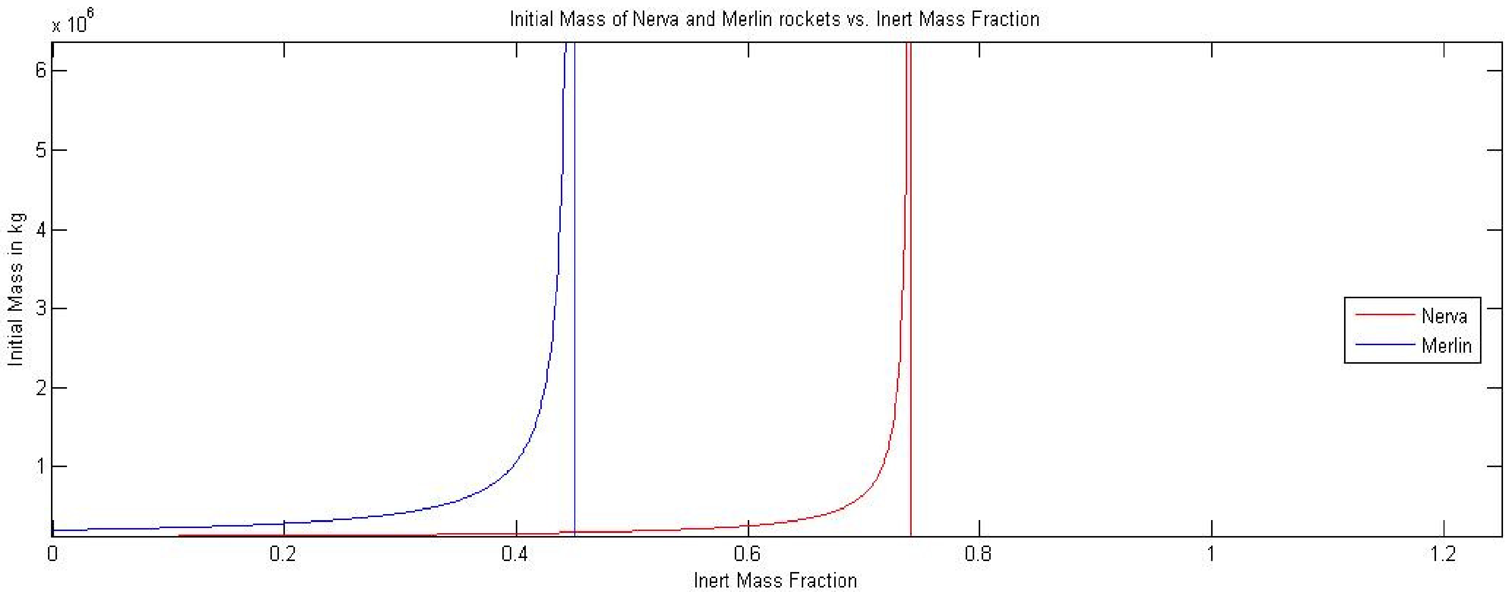
	NERVA – Isp 825 sec	Merlin - Isp 310 sec
Inert Mass [kg]	27000	1505
Heat shield mass [kg]	14715	14715
Initial Mass [kg]	129148.086	214895.024
Propellant Mass [kg]	107134.208	200829.134
Inert Mass Fraction [kg]	.171	.0655

The NERVA rocket has a higher specific impulse and therefore requires less propellant mass. The initial mass of the NERVA rocket is also less than the Merlin rocket. This translates to a lower launch cost for the other team in our alliance. The following graph demonstrates the relationship between delta V and initial mass. The rocket with the higher Isp can achieve a lower initial mass for a higher delta V (Figure 9).



[Figure 8]

Similar comparisons can be done with the consideration of specific impulse and inert mass fraction. In either case a similar graph(Figure 8) is obtained.



[Figure 9]

Basically, this shows us the relationship between initial mass and inert mass fraction. Clearly, the NERVA wins in this respect with a higher inert mass fraction.

Our mission is dedicated to the safety of our crew and systems. To do that, we depend on the reliability of our equipment and design. The NERVA rocket will add an extra billion dollars to our development cost. However, we want to ensure the reliability of the equipment by not embarking on a mission with rockets that are strained. It is more important to have a successful mission in which the astronauts and equipment return to Earth safely. The NERVA rocket is also built for reuse after sea recovery. The extra billion dollars can therefore be considered an investment. The total cost of the mission will be about 1.5 billion dollars.

Cost Efficiency of Aerobraking/Aerocapture

Aerobraking and aerocapture are two similar techniques used to achieve orbit around a celestial body (when approaching from a larger elliptical orbit) that do not require the same magnitude of propulsive thrusts, thus saving fuel, as conventional ΔV maneuvers.

Aerobraking is the less extreme of the two, and our Team decided to rely on it in order to achieve orbit around Mars. Our interplanetary vehicle will approach Mars from its hyperbolic orbit, perform a small burn to transfer into an elongated elliptical orbit, and utilize aerobraking to subsequently slow the velocity of the vehicle through numerous loops. This will be cheaper for our Team, because performing a burn large enough to transfer directly into our final orbital destination around Mars would require a significantly larger mass of fuel than the mass of our reusable heat shield, thus would be a lot more expensive.

Aerocapture is very similar to aerobraking, and will be used for our Teams approach back to Earth. Instead of performing a small burn to initially transfer into an elliptical orbit, the atmosphere of Earth will slow our vehicle down to arrive safely back on the surface, with the help of the same heat shield used on Mars as well as a parachute. Once again, this tactic is much cheaper than carrying a fuel load that would be used to change the vehicle's trajectory through a propulsive maneuver.

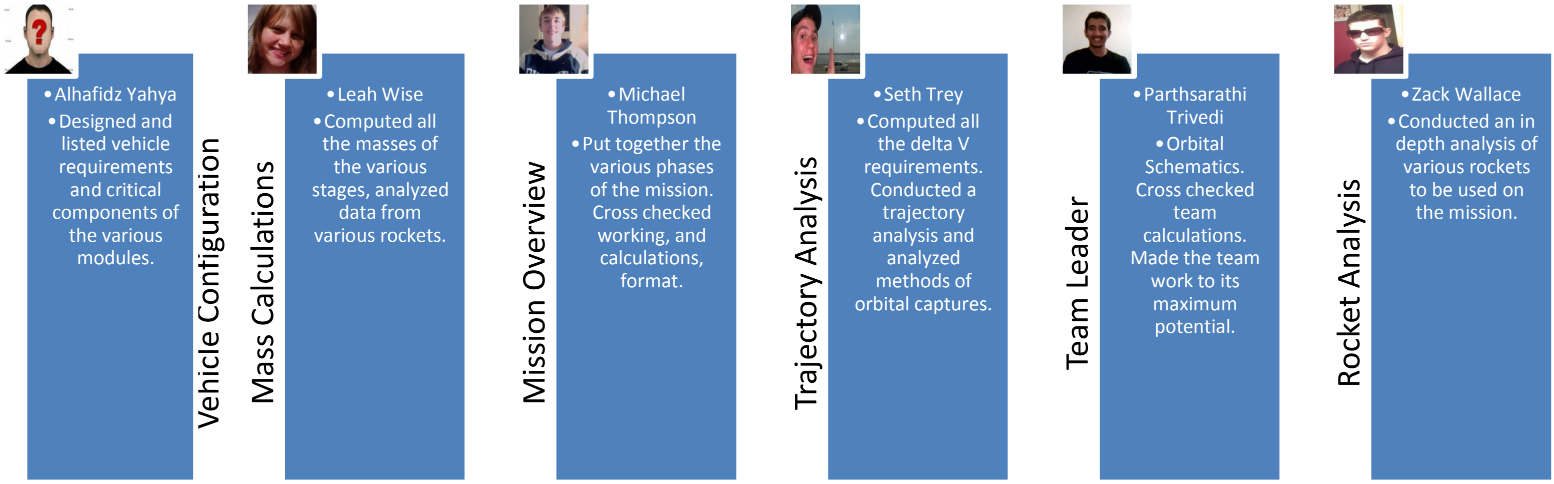
Summary and Open Issues

In summary our vehicle design is broken down into two main parts. The first being the Habitation Module and the second being the Command Module. These two Module parts give the entire spacecraft a total weight of 129148.086 kg. The Command Module makes up for about 6000 kg of the total weight, giving the majority of the weight to the Habitation Module. Note that when the heat shield is added it does increase the amount of weight. The weight of the heat shield is 14715 kg. The entire trip consists of 8 months to Mars, 500 day stay on Phobos and another 8 months back to Earth. The total delta V for the trip will be 2.427039 km/sec.

The advantages of the vehicle are the safety and durability of the ship. This spacecraft is constructed well and made with the finest of materials to ensure that crew inside stays safe for the entire trip. However the spacecraft does come with disadvantages. That disadvantage is the cost. Because of the size of the crew, the spacecraft will be bigger than most to accommodate the entire crew. Also because of the increase in size more propellant will be need to launch the craft into space thus increasing the total cost of the spacecraft.

We made several assumptions to help with our mission. One assumption was that Mars and Earths orbits are coplanar to make the calculations easier for the mission. Another issue that we made assumptions on was the fact that Mars was in fact going to be there when we launched from Earth and that Earth was going to be at that arrival point when we returned from Phobos. A third issue that we did not take a look at the total price of the mission (with crew training etc). We assumed we were giving an unlimited budget for all our expenses. Other Assumptions that were made can be found throughout in the report when dealing with approximations.

Our Team



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Appendix A

r_earth = 1.496e8 ; (Distance between center of Earth and Sun, km)

r_mars = 2.27936640*10^8 (Distance between center of Mars and Sun, km)

r_atmosphere = 6378 +175 (Atmosphere, distance between center of Earth and tip of atmosphere, km)

r_LEO = 6378 + 200 ; (Distance between center of Earth and LEO, km)

r_phobos = 9235.6 (Distance between center of Mars and Phobos orbit, km)

$$v_{circular\ orbit} = \sqrt{\mu/r_c}$$

$$v_{transfer_p} = sqrt\left(mu_{sun} * \left(\left(\frac{2}{r_{earth} + r_{LEO}} - \left(\frac{1}{a_{transfer}}\right)\right)\right)\right)$$

$$v_{transfer_a} = sqrt\left(mu_{sun} * \left(\left(\frac{2}{r_{mars} + r_{phobos}} - \left(\frac{1}{a_{transfer}}\right)\right)\right)\right)$$

Appendix B

To calculate the number of passes required to slow the spacecraft to Phobos transfer orbit more realistically, we decided to model the drag of the spacecraft realistically. We modeled the trajectory as a circular arc through the Martian atmosphere in order to neglect the gravity of the planet (which should about cancel out, because if the atmosphere wasn't present the spacecraft would continue in the same elliptical orbit forever)and only model the drag forces.

We made the approximation that each braking pass through the atmosphere was the same length, and we guessed the length as the chord of the atmospheric sphere tangent to the planet sphere.

$$l = 2\sqrt{R_{atm}^2 - R_{Mars}^2} \tag{1}$$

To simplify the drag equations we approximated the density of the Martian atmosphere as constant, a reasonable simplification, as the atmospheric density of Mars is actually chaotically variable, due to the dust storms and other weather patterns, which would render a more detailed approximation less useful. This led us to the differential equation for drag,

$$\frac{d}{dt} V(t) = -C\, V(t)^2 \tag{2}$$

where C accounts for the density, drag coefficient and reference area terms of the drag equation (all constant.) The solution to this equation is fairly straight forward, with the initial condition that V(0) (first entry velocity) is equal to the hyperbolic periapsis velocity.

$$V(t) = \frac{Vhyp}{1 + C\, t\, Vhyp} \tag{3}$$

Since velocity is the time derivative of position (assumed to be along the circular arc path),

$$\frac{d}{dt} x(t) = \frac{Vhyp}{1 + C\, t\, Vhyp} \tag{4}$$

Which, when given the initial condition of x(0) = 0, solves to

$$x(t) = \frac{\ln(1 + C\, t\, Vhyp)}{C} \tag{5}$$

Solving this equation for t when x(t) = l, we obtain

$$t = \frac{e^{l\,C} - 1}{C\, Vhyp} \tag{6}$$

We started with the assumption that the spacecraft MUST first be slowed into an orbit within the Martian sphere of influence, so that our two-body model remains valid. This gives us a semi-major axis and from that, a periapsis velocity.

$$a = \frac{(R_{ent} + R_{SoI,\,M})}{2} \tag{7}$$

$$V_1 = \sqrt{\mu_M \left(\frac{2}{R_{ent}} - \frac{1}{a}\right)} \tag{8}$$

Substituting this t into the velocity equation and solving for C, we obtain:

$$C = \frac{\ln\left(\frac{V_{hyp}}{V_1}\right)}{l} \qquad (9)$$

This gives us the drag coefficient*density*reference area number that we can use to compute the time and distance through the atmosphere it would require to slow down with equation (3). By setting the left-hand side of equation (3) equal to the periapsis velocity of the transfer orbit between the entry radius and Phobos's orbital radius and solving for *t,* we determine the time it would take to slow down. Inserting this *t* into equation (5) gives us the distance through the atmosphere the spacecraft would have to travel to slow down to this velocity.

Using our assumption that each pass is the same length, we simply divide *x(t)* by *l* to obtain the approximate number of passes through the
atmosphere required to slow down to the appropriate speed.