Design of an Artificial Gravity Mars Mission

Mission Design Considering Human Factors, Structures, and Cost

16.851 Satellite Engineering

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Motivation

Mars exploration is one of the main directions of NASA. One overarching goal is to someday have a manned mission to Mars. This is the next major step for space exploration beyond the moon and out into the solar system.

A manned mission to Mars poses several significant technological challenges for engineers. One such challenge is to minimize the physiological impact on the astronauts during prolonged spaceflight. A possible solution to this is using artificial gravity. Once the crew arrives on Mars, they will almost immediately be able to begin useful scientific research, rather than spending significant time rehabilitating due to problems like bone decalcification. Using current or planned technology, artificial gravity almost certainly requires some sort of spinning spacecraft.

The extended mission to Mars also poses psychological challenges for the crew. The psychological well-being of the crew may depend on the number of astronauts, the gender makeup of the crew, the ages of the crew members, and the "free volume" available per astronaut. Many of these human factors will contribute to the design of the spacecraft used to transport humans to Mars.

Problem Statement

Create a tool to evaluate the feasibility of an artificial gravity Mars mission. The tool should output the cost for four designs: a large monolithic station, a tethered multi-spacecraft station, a tethered two-spacecraft system, and an EMFF system (see Figure 1). In addition, the tool will determine how Mars mission inputs such as number of crew members affect the design of each system. A systems engineer using the tool will be able to vary these parameters to fit a launch or cost constraint.

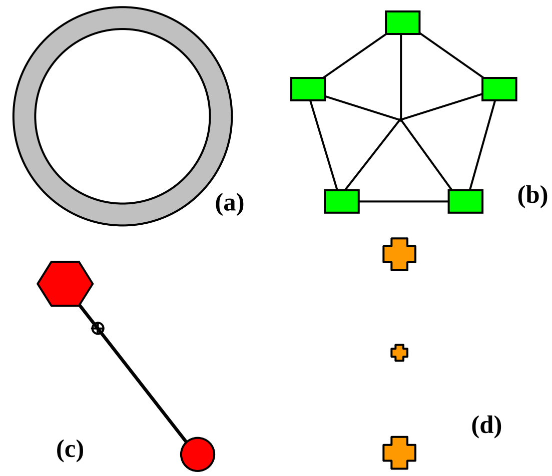

Figure 1 Four designs: (a) monolithic station (Toroid), (b) multi-spacecraft tethered, (c) two-spacecraft tethered, and (d) EMFF.

Introduction

This study analyzes human, structural, and cost aspects of the various spacecraft types in order to determine their feasibility. In addition, the power and propulsion systems of these spacecraft are modeled. The number of crew members is treated as a variable in order to analyze the effects of this parameter on the designs.

Some of the human factors that are considered, in addition to the area issues, are requirements for reducing motion sickness (since the spacecraft will be spinning) and support systems and maintenance (such as food, waste management, thermal and power needs etc.).

The structural aspects are dictated by the design configuration and human factors. For instance, in order to prevent motion sickness, a minimum distance to the spin axis is required. Similarly, the space/area needs for the crew are imposed requirements on the habitable volume of the spacecraft. In addition, volume must be allocated for the Mars science payload, equipment, and spacecraft subsystems. This tool assumes an Earth return vehicle already exists on Mars for the astronauts to use. The design of each particular system has unique structural requirements.

Total cost of each type of spacecraft is evaluated based on the structural and mass requirements of the design. Factors such as launch and operations are included.

Figure 2 summarizes the map of how the various subsystems relate to each other.

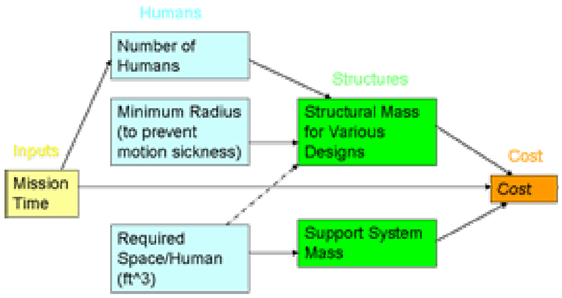

Figure 2. Relationships between the various subsystems of the tool.

Crew Size and Composition

A long journey such as the one to and from Mars would put any crew under extreme stress. The size and composition of the crew for a manned Mars mission are factors that can be controlled in such a way as to minimize the stress during such a mission.

The human-human interface is the most important with respect to the psychological and sociological aspects of the extreme environment of a manned Mars mission. The success of the mission depends on the ability of the crew to effectively work together to accomplish their mission objectives.

Based on an ongoing study of this human-human interface in extreme environments, several important observations have been documented. First, larger crews tend to have lower rates of deviance and conflict. Second, deviance and conflict tend to decline with increasing length of mission. Third, heterogeneous crews have lower rates of deviance and conflict.[1]

Although it was found that a larger crew had fewer incidents of deviance and conflict, a maximum value for crew size needs to be set. In the study previously mentioned, it was found that the least dysfunction of any crew studied was a crew of nine people.[2] This favorable crew size of 9 and the fact that a manned mission to Mars could take as long as nine months, a crew size of nine was set for a mission length of nine months.

The other end of the spectrum, a shorter mission, needs to have a limit for crew size as well. As the duration of the mission gets shorter, the “extremeness” of the environment decreases. This is because the crew knows that they will not be as far from home as they might be on a longer mission and they are closer to reality than a nine-month expedition to Mars. This lessening of the “extremeness” of the trip makes it plausible for a crew of two members to run a mission for duration of approximately one month. Several manned missions to Mars even suggest using a crew of two.[3] Therefore, it is reasonable to assume a crew of two could handle a month-long space mission.

Based on these two limits, linear interpolation is used to estimate the crew sizes for mission durations between one and nine months. However, since a worst-case scenario is assumed in which the crew must stay in the vehicle and return to Earth without landing on Mars, the mission durations are doubled for the same estimated crew size. This effectively places a cap of 9 as the crew size for a mission to Mars using a Hohmann transfer (roughly 9 months transit time each way). These crew size estimates are shown below in Figure 3.

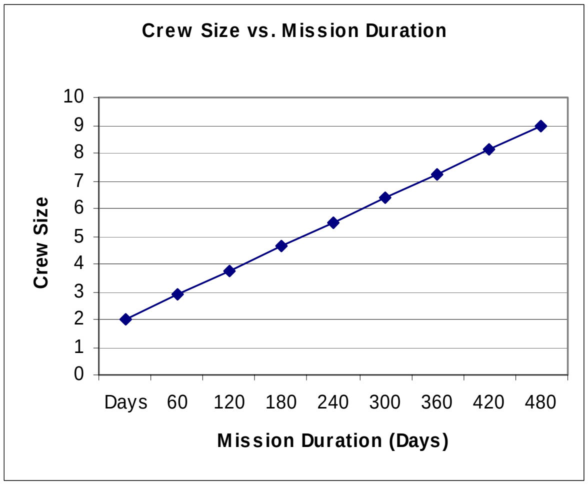

Figure 3 Crew size vs. mission duration

In addition to the crew size, the gender, ethnic, and cultural makeup of the crew plays a large role in the performance of the crew during the mission. It was found that more heterogeneous crews begin a mission with some deviance, conflict, and dysfunction, but these problems seem to decline as the mission progresses. On the other hand, a more homogeneous crew tends to begin a mission without much, if any, deviance, conflict, or dysfunction, but these problems tend to increase throughout the duration of the mission.[4] Therefore, a heterogeneous crew, most likely half men and half women, with a mix of various ethnicities and cultures, would tend to produce a more effective crew for an extreme mission such as a manned mission to Mars.

Human Factors

Interior "Free" Space for Crew

The long duration of a mission to Mars requires that extra comfort be given to the crew than that given to astronauts on a one or two week mission to low Earth orbit. Significant comfort can be given to the crew in the form of increased interior volume to use for work and leisure activities. This would result in improved mental health of the crew at the time of their arrival at Mars.

Breeze (1961) estimated that a crew on a space mission would require a minimum volume of 600 ft^3 per crew member for space missions longer than two months.[5] Sloan,[6] on the other hand, estimates the minimum volume per crew member for life on a space station to be approximately 700 ft^3. Being conservative, a value of 700 ft^3 is assumed for the free volume required per crew member for a manned Mars exploration mission.

Life Support System Equipment Volume and Mass

Crew Systems

The crew systems onboard the spacecraft for a manned mission to Mars contain equipment such as galley and food system, waste collection system, personal hygiene, clothing, recreational equipment, housekeeping, operational supplies, maintenance, sleep provisions, and health care. HSMAD contains a detailed breakdown on the mass and volume requirements of crew systems specifically designed for a manned Mars mission.[7] By dividing the numbers provided in HSMAD by the estimated mission duration and specified crew size, a normalized crew systems mass and volume per crew member per day can be determined. These values are shown below in Table 1.

Table 1 Crew systems normalized volume and mass.

Crew Systems Mass (kg/CM-d)	7.55
Crew Systems Vol. (ft^3/CM-d)	1.51

ECLSS Atmosphere Management

The Environmental Control Life Support System (ECLSS) manages the air, water, waste, and other systems onboard the spacecraft which support human life in space. The portion of the ECLSS which manages the atmosphere onboard the spacecraft utilizes physio-chemical (P/C) technology in order to remove carbon dioxide from the air, control trace contaminants, and provide oxygen to the crew. An atmosphere management system suggested by HSMAD is used for the purposes of this study. This suggestion is a triple-redundant system of three different types of P/C atmospheric management systems.

The three types of P/C systems used in this manned Mars mission spacecraft are 4BMS (4-bed molecular sieve), TCCS, and Sabatier P/C atmosphere management systems.[8] A basic flow chart of the method used to manage the atmosphere on board the spacecraft is shown in the figure below.

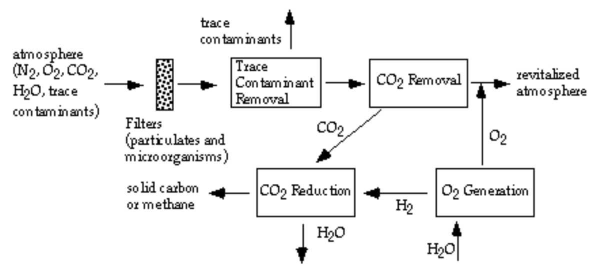

Figure 4 Atmosphere control and supply[9]

Based on the mass and volume requirements provided in HSMAD, the mass per crew member of these environmental support systems could be estimated. The three types of atmospheric management systems were summed and multiplied by a factor of three for redundancy. These values are shown below in Table 2.

Table 2 ECLSS atmosphere management mass and volume per crew member[10]

ECLSS Atm. Mass (kg/CM)	255
ECLSS Atm. Vol. (ft^3/CM)	35.3

ECLSS Water Management

Based on the manned Mars mission design example in HSMAD, the ECLSS water management system design for this project was estimated. HSMAD assumes a P/C water management system of vapor compression distillation (VCD) for use on the spacecraft. A basic flow chart detailing the process of water recovery and management is shown in the figure below.

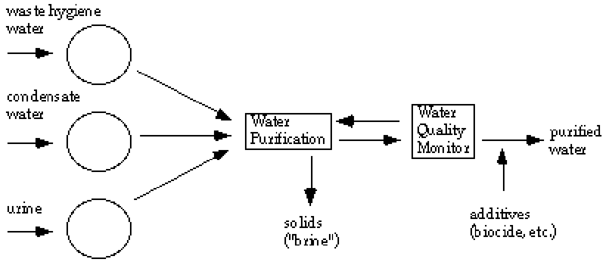

Figure 5 Water recovery and management[11]

This technology requires a mass of approximately 25 kg per crew member and a volume of 3.53 ft^3 per crew member.[12] A redundancy of two water management systems is assumed,[13] which brings the total mass per crew member to 50 kg and total volume per crew member to 7.1 ft^3.

Artificial Gravity

A manned mission to Mars requires that the crew be subjected to the space environment for a significant period of time. A travel time of nearly one year would result in significant musculoskeletal deterioration of the crew members if the transit period were completely zero-g.[14] This would result in the crew members being physically incapable of performing much work, if any, when they arrive at Mars.

The downtime as a result of the crew's required physical rehabilitation would dramatically reduce the available time on Mars for the crew to perform valuable research activities.

In order for the crew to be productive when they reach Mars, an artificial gravity of 0.38g, the magnitude of gravity on Mars, is created on board the spacecraft during the transit to Mars. The artificial gravity is set to Mars gravity because it is unnecessary to provide artificial gravity of 1g if the crew will need to adjust to Mars gravity of 0.38g when they arrive. Also, a smaller artificial gravity requirement reduces the propellant required to spin-up and spin-down the spacecraft, as well as the structural requirements on the spinning spacecraft (and tethers).

Gravity Gradient, Coriolis, and Cross-coupled Acceleration Effects

Due to the fact that the centrifugal acceleration resulting from the spin of the spacecraft varies with radial distance from the spin center, a different level of gravity will exist between various levels of the structure as well as throughout the human body. If this gravity gradient is too large, it could become uncomfortable for the crew members.[15]

In addition, crew members will experience pseudo weight changes depending on their direction of motion due to radial and tangential Coriolis effects. When walking parallel to the spacecraft spin axis, crew members will feel heavier when walking in the direction of the spin and lighter when walking in the opposite direction. Tangential Coriolis effects will be felt by crew members walking moving radially about the spacecraft (possible in the Toroid spacecraft). They will feel a push in the direction of the spacecraft spin when climbing towards or away from the spacecraft's center of motion.

Another potential uncomfortable result of the spinning spacecraft, cross-coupled angular acceleration effects, can be felt by crew members. This occurs when a crew member moves his/her head in directions transverse to the axis of rotation and the direction of flight of the spacecraft. Interior design of the spacecraft may help to alleviate this problem. In addition, researchers at Slow Rotating Room in Pensacola, Florida, found that human test subjects in a room rotating at speeds up to 10 rpm could be trained to adapt to the rotating environment.

These potential impacts to the human crew for the manned Mars mission result in design requirements in order to minimize the impacts of these potential problems and create a safer, healthier, and more enjoyable environment for the crew during their long journey to Mars. The two requirements imposed on the spacecraft design are a maximum spin rate and a minimum spin radius.

Stone (1970) and Thompson (1965) recommend a rotation radius greater than 14.6 meters and spin rate less than 6 rpm, while Shipov (1997) thinks a minimum radius of 20 meters is appropriate. In order to be conservative, a minimum spin radius of 30 meters and a rotation rate of 6 rpm were used for the purposes of this project.

Radiation Design Considerations

During the journey from Earth to Mars, the crew will not enjoy the protection of the Earth's atmosphere from high energy particles from the Sun. Solar particle events (SPEs) cause large numbers of these high energy particles to emanate from the Sun. These particles may impact the spacecraft and could result in harmful health effects for the crew.

Background radiation in space, such as galactic cosmic rays, may also affect the crew during transit to Mars.

In order to design a spacecraft to provide reasonable protection for the crew from radiation, the thickness of the aluminum hull of the spacecraft must be designed with a minimum thickness. This thickness is determined from the maximum allowable radiation dose for crews. This is given to be 1 Gy.[16]

This allowable radiation exposure for crews is compared to the dose the crew would receive based on the aluminum hull thickness to obtain the minimum thickness. The data table used to make this decision is shown below.

Table 3 Radiation dose from an unusually large solar particle event

Shielding Depth (cm Al)	Dose (Gy)
0.5	4.68
1.0	1.95
1.5	1.02
2.0	0.59
2.5	0.37

Since an acceptable dose for the crew is 1 Gy, a minimum hull thickness of 1.5cm of aluminum is chosen for this spacecraft.

Spacecraft Power

In order to obtain an estimate for the power system for the Earth-Mars cruise spacecraft, a rough approximation of spacecraft power per crew member was required. Several opinions exist as to exactly how much power per crew member is required for the Earth-Mars cruise phase of a manned Mars mission.

HSMAD assumes 20kW for a six-crew member mission to mars. This normalizes to 3.33kW per crew member.[17] In addition, Sloan notes that 2kW per crew member is required purely for life support.[18]

It is realistic to assume that more power will be required than the minimum for life support. Research and other activities will require additional power beyond life support. Therefore, it is assumed for the purposes of this project that 3.3kW is required per crew member for the Earth-Mars cruise phase of a manned Mars mission.

Structures

Cylinder

A cylindrical pressure vessel is used as the structure for the tethered multiple spacecraft, two tethered spacecraft and EMFF spacecraft designs. A cylindrical habitat module was chosen because they have a high TRL and can fit easily into a launch vehicle. The diameter of the launch vehicle was used as the diameter of the cylinder. Given the required volume and the number of spacecraft in the array, the length of the cylinder was then determined. The volume is equally distributed among the spacecraft. The material selected for the cylinder was Aluminum 606a-T6, based on the design for a habitat module in HSMAD (Chapter 21). The thickness of the cylinder can be determined by the Hoop stress (since the hoop stress is greater than the longitudinal stress)

$$f_h = \frac{pr}{t} \leq F_{tu} \qquad (0.1)$$

where f_h is the Hoop stress, p is the pressure, r is the radius of the cylinder, t is the thickness of the cylinder, and F_{tu} is the allowable tensile ultimate stress for aluminum. The thickness is set as 0.015 cm if it is found to be less than that because of radiation shielding requirements. The maximum internal pressure of 0.1096 times a safety factor of 2 is used as the pressure inside the cylinder (based on HSMAD). Finally the mass of the cylinder including the two ends is found by

$$mass = 2 \cdot \pi r^2 t \rho_{AL} + 2rlt\rho_{AL} \qquad (0.2)$$

The dry mass of each spacecraft includes the structural mass plus the solar array mass (see Power Module section) and the life support equipment mass.

Toroid

The toroid for the monolithic system is found in a similar fashion as the cylindrical case. The inner radius of the toroid, r_t, is found by the following

$$V = 2\pi^2 r_t^2 R \qquad (0.3)$$

where V is the required volume and R is the radius of rotation. The minimum radius is set as 3 feet (0.9144 m) if the radius, r_t, is found to be less than that. The thickness is found using the hoop stress requirement. The mass of the toroid is found by the following

$$mass_{toroid} = (2\pi r_t)(2\pi R)t\rho_{AL} \qquad (0.4)$$

The dry mass for the monolithic system includes spacecraft includes the toroid mass plus the solar array mass (see Power Module section) and the life support equipment mass.

EMFF Coil Mass

The superconducting EMFF coils are used to rotate the two habitat modules for the EMFF design by creating torque at a distance[17]. The EMFF system assumes that spin-up of the array has occurred and the reaction wheels will not saturate during the steady state spin by rephrasing the array to dump momentum.[19]

To determine the mass of the coils, the force generated by the coils must equal the centripetal force from steady state rotation as seen in Equation (0.5) where R is the coil radius, I_t is the total current, S is the array baseline, ω is the rotation rate, and m_{tot} is the total mass of a habitat module.

$$F = \frac{3}{2}\mu_o \pi I_t^2 R^4 \left(\frac{1}{\left(\frac{S}{2}\right)^4} + \frac{1}{S^4} \right) = \frac{m_{tot} S \omega^2}{2} \quad (0.5)$$

For further clarification, Equation (0.5) uses a three identical satellite system, where the two outer spacecraft are the habitat modules, and the center spacecraft contains only the EMFF coil as shown in Figure 6. The reason for this design is to increase the amount of electromagnetic force in the system. The center spacecraft increases the electromagnetic force by 17 times the force produced by the outer spacecraft. The result is that the three spacecraft design contains EMFF coils that are $17^{-0.5}$ times lighter than those in a two spacecraft design.

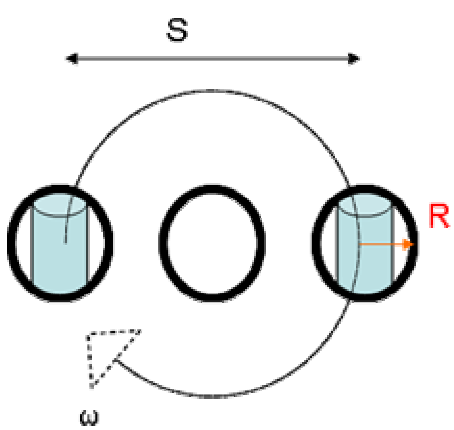

Figure 6 EMFF System Layout

To find the mass of the EMFF coils, the left hand side of Equation (0.5) can be rearranged to include the mass of the coil, M_c, and the wire current density over the wire density, I_c/p_c

$$M_c = \frac{\omega}{R} \frac{1}{\frac{I_c}{p_c}} \sqrt{\frac{m_{tot} S^5}{3 \cdot 17 \cdot 10^{-7}}} \quad (0.6)$$

For a high temperature superconducting coil, the I_c/p_c from the EMFF lecture is 16250 A m/kg.

Tether Sizing

Tethers are required for two of the system designs, so their mass is calculated based on the requirements of each design. For the single tether setup, the radius from the habitation module to the center of rotation is calculated according to the following equation.

$$r_{des} = \frac{g_{des}}{\omega_{\max}^2} \quad (0.7)$$

Here r_{des} is the radius that is 'desired' by the system given the desired force, g_{des}, and the maximum allowable rotation rate, ω_{max} (ω_{max} is determined by human factors). If the calculated radius is larger than the minimum radius allowed by human factors, then r_{des} can be used to calculate the tether length. If not, the minimum radius is used and the rotation rate must be slowed accordingly. Assuming that the two payloads are equal mass, the tether length for this system is then twice the desired radius. The tension in the tether may be calculated as:

$$T = m r_{des} \omega^2 \quad (0.8)$$

Here T is the tension in the tether, m is the mass of one payload, and r_{des} and ω are as above. This is simply an expression of Newton's Second Law, where the radial acceleration is calculated as the radius times the square of the angular velocity. The axial stress equation can then be used to calculate the required cross-sectional area to support the tension T, given the ultimate tensile strength of the chosen material.

$$A = \frac{T}{\sigma_{uts}} \quad (0.9)$$

For this analysis, two materials were considered, as shown in Table 4.[20]

Table 4 Tether material properties

Material	*σ_{uts} (GPa)*	*P (kg/m^3)*
Kevlar	3.6	1440
Spectra	2.6	970

Since the area, length, and density of the tether are now known, the mass can be easily calculated as follows.

$$m = lA\rho \quad (0.10)$$

The multiple tether design requires a different calculation of tether length, but similar techniques are used to compute the final mass. From Figure 1 it is clear that the multiple tether system is overconstrained.

For a system of tethers in tension this is not necessarily a bad thing, it simply makes analysis more tedious. For example, the multiple tether system would maintain its shape (while spinning) if it consisted of *either* the spokes or the rim tethers alone. However, if only the spoke tethers were used, there would be a small risk of collision between the pods during spin up, etc, so it might be wise to include the rim tethers. The solution to this problem is to analyze these systems separately and add the results.

Allowing the angle between two spokes to be α, and the angle between any spoke and its adjacent rim tether to be β, it is clear that:

$$\alpha = \tfrac{2\pi}{n} \qquad (0.11)$$
$$\beta = \tfrac{1}{2}(\pi - \alpha)$$

Here n is the number of spokes. The pods are each at distance r_{des} (or r_{min}, whichever is larger), as calculated above, so the total length of the spokes, l_s, is n times r. The total length of the rim tether, l_r, can be calculated as:

$$l_r = 2nr\sin(\tfrac{\alpha}{2}) \qquad (0.12)$$

The tension in the spokes is calculated without the rim tethers in place as:

$$T_s = mr\omega^2 \qquad (0.13)$$

The tension in the rim tethers is calculated without the spokes as:

$$T_r = \frac{T_s}{2}\cos\beta \qquad (0.14)$$

The area and mass of the rim and spoke tethers can then be calculated as before for each type of tether, and the results added for total tether mass. The following table gives examples of total tether mass for various systems. Systems labeled 'tether' are single tether systems, while systems labeled 'mult-*n*' are multiple tether systems with *n* pods. The subscript *r* represents the rim tethers, and the subscript *s* represents the spoke tethers. To get the total tether mass for a mult-*n* system, add the spoke mass to the rim mass. For single tether systems, the mass of both pods is 70,000 kg, and for mult-*n* systems the individual pod mass is taken as 40,000 kg.

Table 5 Example tether properties

System	Material	Force kN	Area (mm^2)	Length (m)	Mass (kg)
Tether	Kevlar	261	72.5	75.5	39.4
Tether	Spectra	261	100.4	75.5	36.8
Mult-5_r	Kevlar	127	35.2	222	56.3
Mult-5_s	Kevlar	149	41.4	189	56.3
Mult-5_r	Spectra	127	48.8	222	52.5
Mult-5_s	Spectra	149	57.4	189	52.5

In general, it is not wise to base a design on the ultimate tensile strength of a material, so a factor of safety of five is included in the implementation of these equations. It was found that the tether is a small portion of the total system mass, so this factor does not have a large impact. In addition, it helps to account for other tether properties that have been ignored, such as coatings against atomic oxygen, connecting hardware, etc.

Spacecraft Propulsion

A major contributor to total system mass is the propulsion system. While propulsion is not the main focus of this analysis, it is recognized that the fuel required by these spacecraft will be a significant portion of total system mass. Several parts of the required propulsion are treated in some detail, and others are left for a future study. The current analysis is of a high-risk, one-shot Mars approach. Enough fuel is provided to initiate the Mars transfer and the spin required for artificial gravity. It is assumed that the mission will succeed spectacularly. That is, the landing craft will reach Mars interface with hyperbolic velocity, and perform an aerocapture-assisted entry and descent phase. The astronauts will return on a vehicle that is already in place at their landing site. There is no provision for deceleration on Mars approach, for Earth-entry in case of an aborted mission, or other margin of any kind. This is obviously no way to design a manned mission to Mars, but since the primary thrusts of this analysis are cost, structures, and human factors, this greatly simplified propulsion model has been used.

Mars Transfer

In terms of fuel, the cheapest trip to Mars on high-impulse chemical thrusters is a Hohmann transfer. The Hohmann transfer assumes that the transfer orbit is tangent to both the initial and final circular orbits, so it is very efficient and easy to analyze.[21] Knowing the radii of the initial and final orbits, r_1 and r_2 respective, the semi-major axis, *a,* of the transfer orbit can be calculated as follows.

$$a = \tfrac{r_1 + r_2}{2} \qquad (0.15)$$

The velocity of the spacecraft in the transfer orbit, but at the point of making the impulsive injection maneuver, can then be calculated from the energy integral as:

$$v = \sqrt{\mu(\tfrac{2}{r_1} - \tfrac{1}{a})} \tag{0.16}$$

Here v is the required velocity of the spacecraft in the transfer orbit, μ is the gravitational parameter for the central body (the Sun), and r_1 and a are as above. The required change in velocity can then be calculated as:

$$\Delta V = v - v_c \tag{0.17}$$

Where ΔV is the required change in velocity, v is the required spacecraft velocity above, and v_c is the circular velocity that the spacecraft already has due to the orbital motion of the Earth, given by:

$$v_c = \sqrt{\frac{\mu}{r_1}} \tag{0.18}$$

The required ΔV in the solar frame is only half of the calculation, however, for it takes extra fuel to escape from the Earth's sphere of influence. The ΔV required in the solar frame can be considered the "hyperbolic excess velocity" that is required in the Earth-centered frame, or the speed that the spacecraft has with respect to Earth as it leaves the sphere of influence. The change in velocity required from a low Earth parking orbit can be found by first calculating:

$$v_i = \sqrt{v_\infty + \tfrac{2\mu}{r_c}} \tag{0.19}$$

Here v_i is the insertion velocity that is required from LEO, v_∞ is the hyperbolic excess speed that is required in the heliocentric frame (the ΔV solved for above), and r_c is the radius of the parking orbit where the spacecraft is holding until departure. For this study, r_c was taken to be 200km. The required ΔV is then calculated as above, where v_c is recalculated for the parking orbit in the Earth frame.

For the Hohmann transfer from Earth to Mars studied here, the required change in velocity in the solar frame (v_∞) is calculated as 2.942 km/s. Using a circular parking orbit at 200 km, the burn required for the spacecraft is calculated to be 3.61 km/s.

The required ΔV can be used to calculate the fuel required to get the spacecraft to Mars. Using the classic rocket equation, the fuel mass is seen to be a function of required change in velocity, spacecraft dry mass, and the efficiency of the thruster (or specific impulse, I_{sp}).

$$m_p = m_o \left[1 - e^{-(\Delta V / I_{sp} g)}\right] \tag{0.20}$$

Here m_p is the mass of the propellant, m_o is the dry mass of the vehicle, I_{sp} is the specific impulse of the chosen thruster, and g is the acceleration due to gravity at the Earth's surface (where I_{sp} is defined). For this study, a value of 350 seconds is assumed for the specific impulse, a typical value for a bipropellant chemical thruster [see Ref 23, pg 692]. Thus, for a given spacecraft dry mass, the required fuel mass can be determined for each structural design. Note that the required fuel masses to insert the desired payloads into Mars orbit are quite large, so it is a reasonable assumption to ignore the thruster hardware at this stage of analysis. Additionally, for potentially massive components such as fuel tanks, all of the systems under consideration will have similarly scaled components so the relative error here is not significant. Finally, the question of thruster location is not specifically addressed here. It is assumed that the Mars transfer burn will be performed *before* the various designs have initiated their spin. Thus, all tethers will be retracted, and the EMFF system will be docked into a single unit. This way, the whole system can be started on the transfer orbit as a unit, and then the rotations can be initiated en route.

Spacecraft Rotation

A unique feature of the EMFF system is that it does not require fuel to initiate and maintain the nominal spin rate. All other designs, however, will require some manner of external thrust to start spinning. It will be assumed that the monolith structure has a pair of thrusters on opposite sides of the wheel (i.e. at the ends of a line of diameter) to create a couple. The single and multiple tether systems will have a single thruster on each individual pod, oriented to create a pure moment with no net force. The selected thruster has the same specific impulse, 350 s, as the primary thruster.

Under these assumptions, the fuel required to spin up can be calculated from the rocket equation above, noting that:

$$\Delta V = \Delta \omega r \tag{0.21}$$

Here $\Delta\omega$ is the change in angular velocity, and r is the radius from the center of rotation to the thruster. When calculating the fuel requirement from the rocket equation, the total fuel requirement is the dry mass of each individual spacecraft times the number of spacecraft. The following table shows example fuel requirements for spin up for the monolith, single tether, and multiple tether systems. In this table, both 'mass' and 'fuel mass' represent the individual spacecraft masses, and should be multiplied by the total number of spacecraft if total system mass is desired.

Table 6 Propellant mass for system designs

System	Dry Mass (kg)	Fuel Mass (kg)
Monolith	86,774	56,704
Tether	44,344	28,971
Multiple	43,027	28,117

Cost Estimation

A first order model was developed to estimate approximate costs of the manned Mars mission. A detailed work breakdown structure (WBS) was not created since this study considers high-level concept designs that focus only on certain aspects of the mission, *i.e.* human factors, and structural configurations. The cost model utilized a mix of *analogy based estimation*, and *parametric estimation* in determining the costs of the various segments.

The cost model determines the mission cost in FY03$ by evaluating the required expenses in the following categories:

Space Segment

This is driven by the space segment cost factor (S_{cf}), the program level cost factor (P_{cf}), the heritage cost factor (H_{cf}), and the space system mass, M.

The space segment cost, S_c, in dollars is calculated, after adjusting the relationship given by Reynerson, as:

$$S_c = \left(S_{cf} H_{cf} M\right) + P_{cf} \qquad (0.22)$$

The S_{cf} ($/kg) is the price per kg of facility on orbit. For government run, manned space programs it ranges from 38 to 157 $K/kg, with the mean being 104 $K/kg.[22] The maximum value of 157 $K/kg is used in the model in order to get a conservative estimate.

The P_{cf} ($) accounts for the program level costs such as contractor costs for system engineering, management, quality assurance, and other costs that cannot be directly assigned to individual hardware or software components. The P_{cf} was determined from the parametric cost estimation data provided in table 20-4 in SMAD.

The heritage cost factor, H_{cf}, is a dimensionless quantity and accounts for the technology readiness level (TRL) costs. SMAD discusses the development heritage factor in space segment cost (pg 798) as a multiplicative factor. It defines heritage as the percentage of a subsystem that is identical to one or more previous spacecraft, by mass. This idea is applied in the cost model by assuming that the TRL can be considered as the system's heritage. A TRL of 3 is thus considered to have a heritage of only 30%, and the basic RDT&E cost estimate is increased by 70% to account for additional costs that will be accrued due to the development required for the new technology. This assumption provides a means to roughly estimate effects of different design TRLs on the cost. Note that the heritage factors are more appropriate to consider at the subsystem level, and it would be more accurate if they were considered when determining costs of specific subsystems. However, in this study only structures and human life support systems were considered in detail. Therefore, a blanket 'heritage factor' to the whole system cost estimate in this model really means an application to only these two subsystems.

The mass, M (kg), of the system is the total mass of the facility in space. The mass is often the primary cost driver of space systems.[23] The model used in the study also uses the facility's mass as a chief factor in the cost.

Launch Segment

The launch segment costs are determined by using the launch cost factor, L_{cf}, the insurance cost factor, I_{cf}, and the mass of the system, M, to be placed in orbit.

The launch cost is determined as:

$$L_c = L_{cf} I_{cf} M \qquad (0.23)$$

The launch cost factor, L_{cf}, is based on historical data and planned future cost goals. It is the cost per kilogram of placing a payload in LEO orbit. Table 20-14 in SMAD lists the cost per kg to LEO for various launch vehicles in FY00$. The average value for US launch vehicles comes out to be 14.66 $K/kg. Only US launch vehicles were considered since it is assumed that the mars mission will be a government run program. L_{cf} was taken as 15.4 $K/kg (after converting the dollar value from FY00 to FY03).

The insurance cost factor, I_{cf}, was used to account for insurance related expenses associated with launch. For commercial launches, the insurance is a third of the launch costs and I_{cf} is typically 1.33. The cost model in this study assumes a value of 1.5 to account for somewhat higher insurance costs that would probably be involved for a new type of mission. Furthermore, a higher factor would give a conservative estimate.

The mass, M (kg) used in this cost portion is the same facility mass that was used in determining the space segment cost.

Ground Operations and Support

The ground operations and support cost is usually much smaller than the space segment and launch cost. For missions that extend over long periods of time however, this cost can become quite significant. The ground segment costs are normally evaluated by considering the requirements for the ground station facilities such as square footage, equipment, personnel, *etc*. However since such details are not available at concept level studies, an analogy-based estimation was done to determine the operations and support cost for the manned mars mission. The International Space Station has a $13 billion operations budget for its ten-year life. The yearly operations costs are therefore earmarked as $1.3 billion. The cost model in this study uses a value of $1.5 billion per year for operations cost.

The total cost is obtained by summing the space segment, launch and operations cost of the mission.

Software Modules

Volume and Equipment Mass Module

Requirements

The MATLAB module *constants.m* determines the number of crew required for the mission as well as the volume and mass of the vehicle, life support equipment, as well as the required power for the spacecraft.

Description of Code

The code uses the input of the mission duration to calculate the number of crew members required for the mission. The number of crew members combined with the mission duration is used to size the free volume of the vehicle along with the life support system and power requirements.

Constants

The constants used in this module are the values for spacecraft volume, mass, and power which are given and explained in the "human factors" and "spacecraft power" sections of this document.

Inputs

duration (days): This input is the mission duration from Earth to Mars.

Outputs

crew: This output is the total number of crew required for the mission to Mars.

free_vol (ft^3): This output is the total required "free volume" in the spacecraft.

cs_vol (ft^3): This output is the total required volume for the crew systems equipment.

cs_mass (ft^3): This output is the total required mass of the crew systems equipment.

ls_air_vol (ft^3): This output is the total required volume for the atmosphere management equipment.

ls_air_mass (kg): This output is the total mass of the required atmosphere management equipment.

ls_water_vol (ft^3): This output is the total required volume of the water recovery and management equipment.

ls_water_mass (kg): This output is the total mass of the required water recovery and management equipment.

power (W): This output is the total power required for the spacecraft during the Earth-Mars transit. This is purely based on the number of crew members in the spacecraft.

Power Module

Requirements

The MATLAB module *SolarArrays.m* determines mass of the solar array and the area of the solar array. This module was used by Kwon, Vaughan, and Siddiqi in Problem Set 5.

Description of Code

The code uses the required power to determine the mass of the solar array. Multijunction arrays with no ellipse periods were used in the calculation

Constants

The constants used in this module are the specific powers for each type of solar array design.

Inputs

Average_power (W): This input is the required power that the solar arrays need to deliver.

Ellipse_fraction (0-1): This input is the fraction of the orbit spent in eclipse.

type (number): This input is selects the solar array type (1, 2, or 3 for Si, GaAs, or multijunction respectively).

Mission_duration (years): This input is the mission duration in years.

Outputs

Mass_solarArray (kg): This output is the mass of the solar array..

Area_solarArray (m^3): This output is the area of the solar array.

Structures Module

Requirements

The MATLAB module *structures.m* determines mass of the structure given the total volume required, the number of vehicles, and the type of architecture. The architecture options are a toroidal monolithic spinning spacecraft, a tethered multiple spacecraft, two tethered spacecraft, or two EMFF spacecraft.

Description of Code

The code uses the required volume and calculates the dimensions of a cylindrical pressure vessel. The diameter of the cylinder is constrained by the launch vehicle diameter. For the toroidal monolith system, it is assumed that the toroid is cut into sections while it is in the launch vehicle. The total volume is divided equally between the number of spacecraft for the design. Additionally the radius of rotation is used to determine the length of the cylinder. Once the dimensions of the structure are determined, its mass is calculated and outputted.

Constants

The constants used in this module are the values for density and allowable tensile ultimate stress for Aluminum 6061-T6 and the maximum internal pressure for design of the pressure vessel. These values are given and explained in the "structures" section of this document.

Inputs

V (m^3): This input is the total volume required for the structure to contain.

D (m): This input is the launch vehicle diameter.

N (number): This input is the number of vehicles in the array.

R (m): This input is the radius of rotation.

w (rad/s): This input is the rotation rate of the system.

design ('text'): This input is the desired design, options include 'monolith', 'multiple', 'tether', and 'emff'.

Outputs

Mass (kg): This output is the total mass of the structure.

Tether Mass Module

Requirements

The MATLAB module *tether_mass.m* determines mass of the tether given the type of architecture, number of vehicles, dry mass, tether material, and desired acceleration.

Description of Code

The code takes the system architecture and decides how to calculate the tether length and tension. For the monolith and EMFF, there is no tether. For the single and multiple tether systems, the values are computed appropriately as described above. Mass of the tether is then calculated from the material properties of the tether and the required length and tension.

Constants

The constants used in this module are the values for maximum allowable spin rate and minimum allowable radius, as defined by human factors.

Inputs

AG_type ('text'): This input is the desired design, options include 'monolith', 'multiple', 'tether', and 'emff'.

n (number): This input is the number of vehicles in the array.

w (rad/s): This input is the rotation rate of the system.

dry_mass (kg): This is the mass of the spacecraft. For the single tether, an array of 2 masses (can be unique). For the multiple tether, one mass is provided and the modules are assumed to be identical.

tether_mat ('text'): Input describes what material to use for the tether. Current options are 'spectra' and 'kevlar'.

g_des (m/s^2): This desired acceleration at the rim.

Outputs

Mass (kg): This output is the total mass of the tether(s).

Propulsion Module

Requirements

The MATLAB module *propulsion.m* determines mass of the required fuel for orbit transfer and spin-up, given the type of architecture, number of vehicles, dry mass, tether material, and the moment arm to the thruster.

Description of Code

The code calculates the change in velocity required for a Hohmann transfer to Mars, and then solves the Earth-centered problem for ΔV required from a LEO parking orbit. The rocket equation is then used for an assumed thruster to find the fuel mass for the transfer. Given the type of system and the thruster moment arm, the rocket equation is used again to find the propellant required for spin-up. This function calls several auxiliary functions that are included and commented in Appendix A, namely: *ic_circ.m, hohmann.m, p_conic.m,* and *r_equation.m*.

Constants

The constants used in this module are the gravitational constants for the Earth and Sun, the radius of the Earth, the Earth-Sun distance, the parking orbit radius, and the Mars-Sun distance.

Inputs

AG_type ('text'): This input is the desired design, options include 'monolith', 'multiple', 'tether', and 'emff'.

n (number): This input is the number of vehicles in the array.

w (rad/s): This input is the rotation rate of the system.

dry_mass (kg): This is the mass of the spacecraft. For the single tether, an array of 2 masses (can be unique). For the multiple-tethered spacecraft, one mass is provided and the modules are assumed to be identical.

r_outer (m): The moment arm for the thruster.

Outputs

Mass (kg): This output is the mass of the propulsion system per pod.

EMFF Module

Requirements

The MATLAB module *emff.m* determines mass of the superconducting EMFF coils needed to rotation rate for a given amount of artificial gravity.

Description of Code

The code uses the size of the cylinder as the size of the coils, the total dry mass each satellite, the radius of rotation, and the rotation rate to determine the mass of the coils for a three spacecraft collinear array. The equation used for this is explained in the "emff coil mass" section.

Constants

The constant used in this module is the Superconducting coil current density divided by the wire density as given in the EMFF Lecture.

Inputs

V (m^3): This input is the total volume required for the structure to contain.

D (m): This input is the launch vehicle diameter.

R (m): This input is the radius of rotation.

w (rad/s): This input is the rotation rate of the system.

Mass_tot (kg): This input is the total dry mass of one of the satellites.

Outputs

Mass_coil (kg): This output is the total mass of the EMFF coil.

Cost Module

The MATLAB module *cost.m* calculates the total cost of a manned mission based on the system mass, technology readiness level of the system, and mission duration.

Inputs

mass (kg): This is the total mass of the system /facility in space.

TRL: The Technology Readiness Level of the system

duration (days): The mission duration from Earth to Mars.

Outputs

TotalCost ($): This is the total cost of the mission in $FY03. It is the sum of all the cost segments that are also given out by the module.

SpaceSegCost ($): The space segment cost in $FY03

LVCost ($): Launch cost in $FY03

OpSupCost ($): Operations support cost in $FY03

Results

The total program costs for a 1.5 year mission for the different designs are shown in the figure below. It is seen that the cheapest design option is the monolith while the multiple tether configuration is the most expensive.

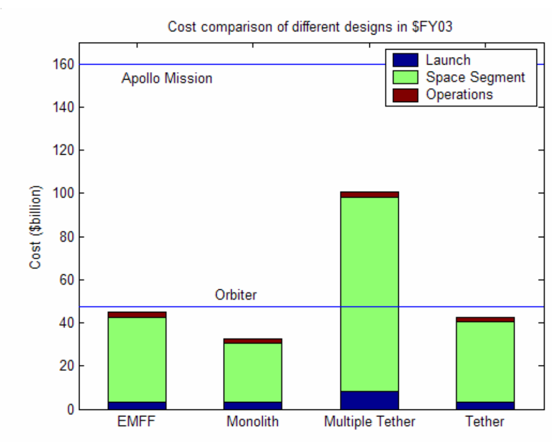

Figure 7 Cost of different designs

The cost breakdown shows that the space segment cost is by far the largest portion as compared to launch and operation costs. A comparison with Apollo and Orbiter costs show that the model estimates lie within a reasonable range.

Since the cost model is driven primarily by the system mass, an analysis of the mass of the different designs shows a trend that matches with the cost results. The figure below illustrates the total mass estimates obtained for the different designs.

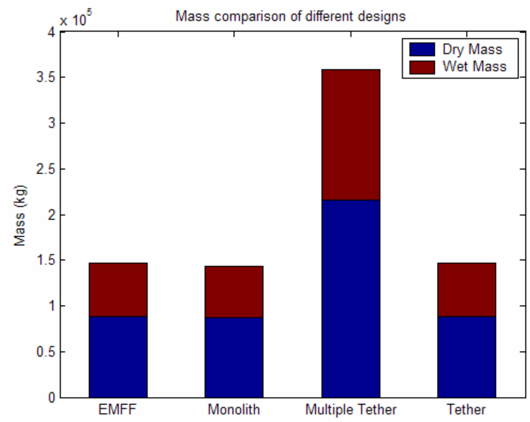

Figure 8 Mass comparison of different designs

Although the mass of the EMFF, monolith and tether designs are in the same range, the monolith is cheaper than the other two designs due to a higher TRL value. EMFF and tether designs have lower TRLs (the model assumed 3 and 4 respectively), therefore they cost more than the monolith. The dry mass in each design included the power subsystem, the structural mass, and crew and life support equipment mass. It also included mass of subsystems that were unique to each particular configuration, for instance the dry mass of the EMFF design includes the mass of coils while in the multiple tether and tether options it includes the mass of tethers. From these results it appears that the monolith design offers the lightest and cheapest option.

Varying Crew Size

One interesting plot is the change in total program cost versus the number of crew used in the mission to Mars. As the number of crew increases, the required structure volume increases, which in turn increases the mass and cost. The results for the four designs considered are shown in the figure below.

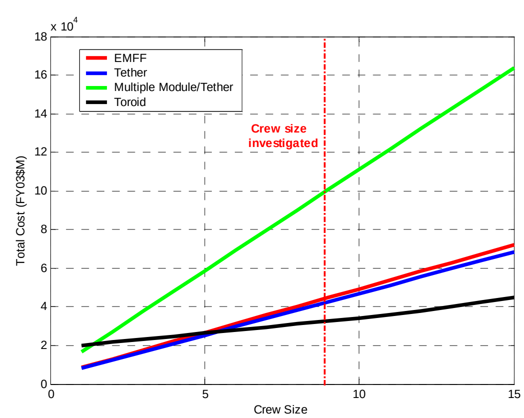

Figure 9 Cost vs. crew size

The above figure shows significant differences in the rate of change of cost as the crew size changes for the various spacecraft designs. The single tether and EMFF designs have nearly identical curves in the figure as well. This is due to the fact that they both have the same basic structural design: each design has two main modules and little or no mass connecting the two modules. Also, these two designs have nearly identical TRL values.

The other two designs, the toroid and the multiple-tethered module, are radically different designs than the previous two. It can be seen that the cost of the monolith increases much less dramatically than the cost of the multiple-tethered module vehicle. This difference is mainly the result of the fact that the TRL of the toroid is much higher than that of the other three designs, especially the multiple-tethered module.

Finally, it can be seen in Figure 9 that the cost of the Toroid spacecraft becomes the most cost effective design for crew sizes greater than five. Based on this information, a Toroid may be the most cost effective design for a large crew of approximately nine members for a manned mission to Mars.

Effect of Varying Artificial Gravity

The artificial gravity is created by rotation of the vehicle(s). A higher artificial gravity results in a higher rotation rate, given a fixed radius of rotation. For the tethered two spacecraft, multiple spacecraft, and monolith systems, a higher rotation rate results in a larger ΔV needed for spin-up and results in more propellant. For the EMFF system, the EMFF coil mass is directly related to the rotation rate as seen in equation (0.6). Figure 10 illustrates these results for the four different systems. Each of the systems shown an increase as the Earth's gravity is approached. None of the curves overlap and the multiple-tethered spacecraft shows the highest mass while the two tethered spacecraft is the least massive option.

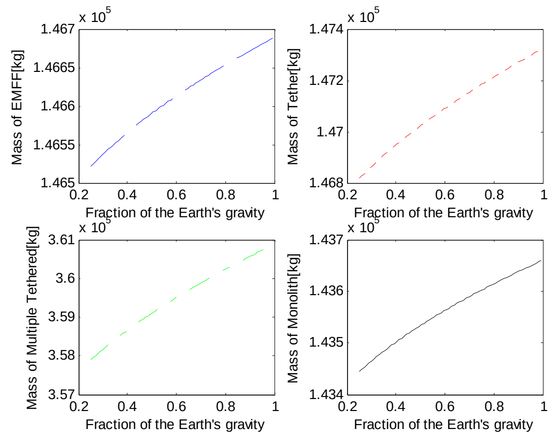

Figure 10 Effect of varying the fraction of Earth's gravity on total system mass

Figure 11 illustrates the effect of varying the fraction of Earth's gravity on the total system cost. Since the cost varies directly with the mass, these results show an expected trend; the cost shows an increase as the Earth's gravity is approached.

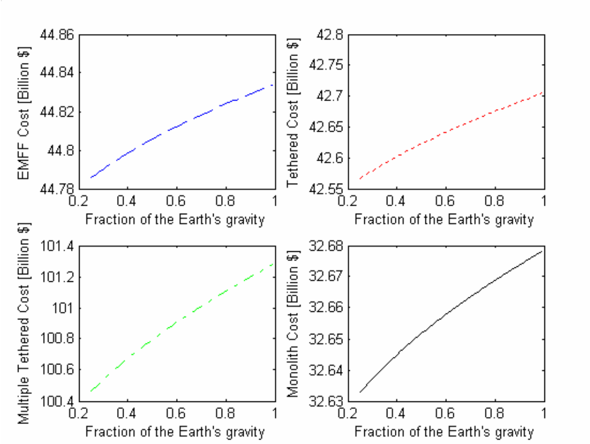

Figure 11 Effect of varying the fraction of Earth's gravity on cost

Conclusion

The preceding design of an artificial gravity Mars mission demonstrates that the mission has feasibility in terms of cost since the cost is less than the Apollo program. The mass of the systems are high mainly due to the significant propellant mass, but more advanced propulsion systems could help decrease this. The monolith system is currently the most favorable design for cost and mass, and for large crew sizes. The tether and EMFF designs may become more favorable with further development of their technology boosting their TRL.

Future Work

This is an exciting project with much potential for future work. An improvement that is immediately obvious is to allow different mission durations and to evaluate the effect of that change on mission cost and mass. Currently, a Hohmann transfer from Earth to Mars is specified, but other orbits should be examined such as faster one-tangent burns, or perhaps longer orbits with free-return trajectories. Changing the mission duration will impact the number of desired crew-members as well as the required propellant for transfer, and so could have a large impact on mass and cost.

Another improvement would be to add a detailed propulsion system model to this analysis. Current all propulsion system hardware is neglected, along with any propellant margin, corrective maneuvers, terminal rendezvous burn, or mission-abort scenarios. All of these things could be added to increase the fidelity of the overall model. Certainly, including these things will increase the total mass and cost of the systems.

There are many other systems that could be added as well. Power, while mentioned in this study, could be investigated in a much more thorough fashion. Issues could be addressed relating to human needs such as thermal controls, debris and radiation mitigation, and communications. Each of these improvements could greatly enhance the quality of the analysis and make this an even more valuable tool for future use.

Appendix A: MATLAB source code

main.m

```
%Constants
g_des = 1/3 * 9.81;
rmin = 30; %meters
wmax = 6; %rpm
w = wmax * pi/30;

D = 5;  %m, Launch vehicle width

mission_duration = 1.5;   %years

%first call constants
%Get the required volume and average_power
[crew, free_vol, cs_vol, cs_mass, ls_air_vol, ls_air_mass, ls_water_vol, ...
        ls_water_mass, power] = constants(mission_duration*365);
V_ft = free_vol + cs_vol + ls_air_vol + ls_water_vol; %Total Volume, ft^3
V = V_ft * 2.83168*10^-2;  %Total Volume, conversion from ft^3 to m^3

M_systems = cs_mass + ls_air_mass + ls_water_mass; %Mass of crew and support systems

%Find Power Mass
[M_power, Area_power] = SolarArrays (power,0,3,mission_duration);

r_des = g_des/w/w;   % calculate 'desired' radius to get desired acceleration
if (r_des < rmin);
    r_des = rmin;
    w = sqrt(g_des/r_des);
end

%Now For Each Design
%EMFF%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
N=2;
%Find structural mass
%mass of each cylinder (note that there are two for emff, tether)
M_struct_emff = structure(V, D, N, r_des, w,'emff');
%Find Propulsion system Mass
Mass_dry_emff = M_struct_emff + M_power + M_systems;
M_coil_emff = emff(V, D,r_des, w, Mass_dry_emff);
M_wet_emff=propulsion('emff', N, w, Mass_dry_emff+M_coil_emff, r_des);
%Compute the total mass
M_total_emff = Mass_dry_emff + M_coil_emff+M_wet_emff;
%Computer system mass
M_system_emff = N * M_total_emff;
%Find cost
[TotalCost_emff, SpaceSegCost_emff, LVCost_emff,
OpSupCost_emff]=cost(M_system_emff,3,mission_duration);
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


%Tether%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
N=2;
%Find structural mass
%mass of each cylinder (note that there are two for emff, tether)
M_struct_tether = structure(V, D, N, r_des, w,'tether');
%Find Propulsion system Mass for each spacecraft
Mass_dry_tether = M_struct_tether + M_power + M_systems;
M_wet_tether=propulsion('tether', N, w, Mass_dry_tether, r_des);
M_tether = tether_mass('tether', N, w, [(Mass_dry_tether+M_wet_tether)
(Mass_dry_tether+M_wet_tether)], 'kevlar', g_des);
%Compute the total mass
M_total_tether = Mass_dry_tether + M_wet_tether;
M_system_tether = N * M_total_tether + M_tether;
%Find cost
[TotalCost_tether, SpaceSegCost_tether, LVCost_tether,
OpSupCost_tether]=cost(M_system_tether,4,mission_duration);
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


%Multiple Tethered 5 spacecraft%%%%%%%%%%%%%%%%%%%%%%%%%%%%
N=5;
%Find structural mass
%mass of each cylinder (note that there are two for emff, tether)
M_struct_multiple = structure(V, D, 5, r_des, w,'multiple');
%Find Propulsion system Mass
Mass_dry_multiple = M_struct_multiple + M_power + M_systems;
M_wet_multiple=propulsion('multiple', N, w, Mass_dry_multiple, r_des);
```

```
M_tether_multiple = tether_mass('multiple', 5, w, Mass_dry_multiple+M_wet_multiple, 'kevlar',
g_des);
%Compute the total for each spacecraft
M_total_multiple = Mass_dry_multiple + M_wet_multiple;
%Computer system mass
M_system_multiple = N * M_total_multiple + M_tether_multiple;
%Find cost
[TotalCost_multiple, SpaceSegCost_multiple, LVCost_multiple,
OpSupCost_multiple]=cost(M_system_multiple,4,mission_duration);
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


%Monolithic Spacecraft%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Find structural mass
%mass of each cylinder (note that there are two for emff, tether)
M_struct_monolith = structure(V, D, 1, r_des, w,'monolith');
%Find Propulsion system Mass
Mass_dry_monolith = M_struct_monolith + M_power + M_systems;
M_wet_monolith=propulsion('monolith', N, w, Mass_dry_monolith, r_des);
%Compute the total mass
M_total_monolith = Mass_dry_monolith + M_wet_monolith;
%Find cost
[TotalCost_monolith, SpaceSegCost_monolith, LVCost_monolith,
OpSupCost_monolith]=cost(M_total_monolith,8,mission_duration);
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

constants.m

```
% William Nadir
% 16.851 Satellite Enginnering
% Module to estimate Mars mission crew size and vehicle volume and mass requirements
%
% INPUTS
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%
% duration      = Mission duration (days)
%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%
% OUTPUTS
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%
% crew                    = Crew size (number of people)
% free_vol               = Amount of total "free" volume required for crew (ft^3)
% cs_vol                  = Amount of total volume required for crew systems (ft^3)
% cs_mass             = Mass of crew systems (kg)
% ls_air_vol            = Life support equpment (air) total volume (ft^3)
% ls_air_mass        = Mass of life support equipment (air) (kg)
% ls_water_vol        = Life support equpment (water) total volume (ft^3)
% ls_water_mass    = Mass of life support equipment (water) (kg)
% power                     = Required total spacecraft power (W)
%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

function [crew,free_vol, cs_vol, cs_mass, ls_air_vol, ls_air_mass, ls_water_vol, ...
        ls_water_mass, power] = constants(duration)

% Here the required crew size is determined based on the duration of the
% mission to Mars
crew = ceil((.0292 * duration + (9/8))/2); % (No. of crew members)

free_vol = 700 * crew; % Total free volume required (ft^3)

cs_vol = 1.512* crew * duration; % Total crew systems volume required (ft^3)
cs_mass = 7.55 * crew * duration; % Total crew systems mass (kg)

ls_air_vol = 35.3 * crew; % Total ECLSS air control system volume (ft^3)
ls_air_mass = 255 * crew; % Total ECLSS air control system mass (kg)

ls_water_vol = 7.1 * crew; % Total ECLSS water control system volume (ft^3)
ls_water_mass = 50 * crew; % Total ECLSS water control system mass (kg)

power = 3300 * crew; % required S/C power (W)
```

SolarArrays.m

```
%This function calculates the mass, cost, and size of a given type of solar array.
%The inputs are eclipse fraction, average power, shadow fraction, mission duration, and cell
type.


function [Mass_solarArray,Area_solarArray]=SolarArrays
(average_power,eclipse_fraction,type,mission_duration)
```

```
%P_solarArray: power produced by solar arrays
%Pe          :power required during eclipse period
%Pd          : power required during daylight period
%Xe          :path efficiency from solar array, through battery to loads
%Xd          :path efficiency from solar array to loads
%Xcell       :cell efficiency
%Isolar      :solar illumination intensity
%Id          :Inherent degredation
%DSi etc     : degredation/yr
%Ld          :lifetime degredation

%********************************
% Constants
Xe = 0.65;
Xd = 0.85;
XSi = 0.148;
XGaAs= 0.185;
Xmulti= 0.22;
Isolar = 1367;
Id = 0.77;
DSi = 0.0375;
DGaAs = 0.0275;
Dmulti = 0.005;
SpSi = 0.55; %16.89 design doc kg/m^2
SpGaAs = 0.85; %kg/m^2
Spmulti= 0.85; %kg/m^2
%******************************

Pe = average_power;
Pd = average_power;

%Te = orbit_period*eclipse_fraction; %eclipse time
%Td = orbit_period-Te;       %daylight time
%P_solarArray = ((Pe*Te )/(Xe*Td)) + (Pd/Xd) %power produced by solar arrays

P_solarArray = Pd/Xd;

%Silicon is type 1, GaAs is type2, and multijunction is 3

if type == 1
    Xcell = XSi;
    Degredation = DSi;
MassPerArea = SpSi;
SpecificPower = 25;
%SpecificCost = SpCostSi;
end

if type == 2
    Xcell = XGaAs;
    Degredation = DGaAs;
MassPerArea = SpGaAs;
SpecificPower = 60; %ref:
http://lheawww.gsfc.nasa.gov/docs/balloon/2nd_tech_workshop/Loyselle.pdf
%SpecificCost = SpCostGaAs;
end

if type ==3

    Xcell = Xmulti;
 Degredation = Dmulti;
MassPerArea = Spmulti;
SpecificPower = 66; %assumed based on info in SMAD
%SpecificCost = SpCostmulti;
end

%Power out of solar cell assuming sun rays are normal to solar panels
Pout = Xcell * Isolar;


%Power at begining of life

Pbol = Pout *Id;
Ld = (1-Degredation)^mission_duration;

%Power at end of life

Peol = Pbol*Ld;

Area_solarArray = P_solarArray/Peol;

Mass_solarArray = P_solarArray/SpecificPower;

%Cost = SpecificCost * Mass_solarArray;
```

```
%Cost = NaN;
```

structure.m

```
function [mass] = structure(V, D, N, R, w,design)

% V = 85;             %m^3
% D = 5;              % Launch vehicle diameter
% N = 2;              %number of vehicles
% R = 20;             % distance from center of rotation to floor(really ceiling)
% design = 'monolith';    %choices: monolith, multiple, tether, emff
% w = 6;              %rotation rate in rpm
%Material Selection: AL 6061-T6
rho = 2.85 * 10^3;  %kg/m^3, density
Ftu = 290 * 10^6;   %Pa, Allowable Tensile Ultimate Stress

% Design Factors
Pmax = 0.1096 * 10^6;   %Pa, Maximum Internal Pressure
SF = 2.0;           %Safety Factor

Pu = SF * Pmax;     %Design Ultimate Internal Pressure

switch lower(design)
    case {'emff', 'tether'}
        r = D/2;            %m, Radius of the cylinder
        l = (V/N)/(pi*r^2);     %m, Length of the Cylinder
        % Thin-Walled pressure cylinder thickness
        t = Pu * r / (Ftu); %m

        if t < 0.015
            t = 0.015;      %minimum thickness necessary for radiation dosage
        end

        %Calculating the mass of the cylinder structure
        Mcyl = 2 * r * l * t * rho;
        Mends = pi * r^2 * t * rho;
        mass = 2 * Mends + Mcyl;

    case {'multiple'}
        Vi = V/N;
        r = D/2;            %m, Radius of the cylinder
        l = Vi/(pi*r^2);     %m, Length of the Cylinder

        % Thin-Walled pressure cylinder thickness
        t = Pu * r / (Ftu); %m

        if t < 0.015
            t = 0.015;      %minimum thickness necessary for radiation dosage
        end

        %Calculating the mass of the cylinder structure
        Mcyl = 2 * r * l * t * rho;
        Mends = pi * r^2 * t * rho;
        mass = (2 * Mends + Mcyl);
    case {'monolith'}
        %Given V
        %R is set from minimum radius needed for artifical gravity

        rt = 1/pi * sqrt(V/(2*R));    %inner toroid radius
        %if rt < some height, then rt = minimum height
        if rt < 1.8288/2 % if height is less than six feet
            rt = 1.8288/2;    %meters
        end

        t = Pu * rt / Ftu;  %m
        if t < 0.015
            t = 0.015;      %minimum thickness necessary for radiation dosage
        end

        %Calculating the mass of the toroid structure
        Mass_toroid = rho * t * (2*pi*rt) * (2*pi*R);

        %Calculating the mass of the center spherical shell
        %         rs = 1/10 * R;
        %         t_shell = 0.015;
        %         Mass_shell = 4*pi*rs^2 * t_shell*rho;
        %
        %         %Calculating the mass of the beams
        %         Num_beams = 4;
        %         rb = R - rs;
        %         h = 0.1;    %beam width
        %         Mass_beam = rb * h^2 *rho;

        %Calculating total monolith mass
```

```
        %         mass = Mass_toroid + Mass_shell + Mass_beam * Num_beams;
        mass = Mass_toroid;
    otherwise
        disp('Unknown method.')
end
```

tether_mass.m

```
function [mass]=tether_mass(AG_type, n, w, dry_mass, tether_mat, g_des)

% Function "tether_mass.m" takes parameters of the system and returns the
% calculated mass of the tether.
%
% Inputs:
%    AG_type [] - The type of system in question (monolith, multiple, tether, emff)
%    n [] - The number of spacecraft, only applicable for the pinwheel design
%    w [rad/s] - The desired rotational rate of the system [to be hardcoded?]
%    dry_mass [kg] - The dry mass of the spacecraft.  This should be a row
%       vector with appropriate dimensions as follows:
%           * Monolith: N/A
%           * EMFF: N/A
%           * Single Tether: [(habitation module)  (other mass)]    ###[1x2]
%           * Pinwheel: (mass of nodes, assumed uniform)     ###[1x1]
%    tether_mat [] - Indicates what tether material should be used
%    g_des [m/s^2] - The desired acceleration at the habitation module
%
% Outputs:
%    mass [kg] - The mass of the tether

% Constants

rmin = 30; %meters
wmax = 6; %rpm
verbose = 1;

% /Constants

% Tether

% source for this stuff 'http://callisto.my.mtu.edu/my4150/props.html' is ok

if strcmp(tether_mat, 'kevlar')
    sig_uts = 3.6e9;     % [Pa]
    density = 1440;    % [kg/m^3]
elseif strcmp(tether_mat, 'spectra')
    sig_uts = 2.6e9;     % [Pa]
    density = 970;    % [kg/m^3]
else
    fprintf('Error :: tether_mass :: Unknown tether material!\r')
end


% /Tether

wmax = wmax*pi/30;  % rad/s
if (w > wmax)
    fprintf('Warning :: tether_mass :: Spacecraft spinning too fast!\r')
end

if strcmp(AG_type, 'monolith')
    mass=0;
elseif strcmp(AG_type, 'emff')
    mass=0;
elseif strcmp(AG_type, 'tether')
    m1=dry_mass(1);
    m2=dry_mass(2);
    r_des = g_des/w/w;   % calculate 'desired' radius to get desired acceleration
    if (r_des < rmin);
        r_des = rmin;
        w = sqrt(g_des/r_des);
        fprintf(['Warning :: tether_mass :: Calculated radius is below minimum, used minimum
radius of ' num2str(rmin) ' [m].\r'])
        fprintf(['                          Angular rate should be less than ' num2str(w*30/pi) '
[rpm].\r'])
    end
    tlength=r_des*(m1+m2)/m2;    % calculate tether length based on desired radius and relative
masses
    F=m1*r_des*w*w;   % calculate tension in the tether
    A=F/sig_uts;   % required tether area is tension/ultimate tensile strength

    mass=density*tlength*A * 5;    % calculate tether mass using factor of safety of 5

    if (verbose == 1)
```

```
        fprintf(' \r')
        fprintf(['Designed a ''' AG_type ''' tether of ' tether_mat '.\r'])
        fprintf(['Desired radius: ' num2str(r_des) ' [m].\r'])
        fprintf(['Tether length:  ' num2str(tlength) ' [m].\r'])
        fprintf(['Tether tension: ' num2str(F) ' [N].\r'])
        fprintf(['Tether area:    ' num2str(A) ' [m^2].\r'])
        fprintf(['Tether mass:    ' num2str(mass) ' [kg].\r'])
        fprintf(' \r')
    end

elseif strcmp(AG_type, 'multiple')
    m=dry_mass(1);
    r_des = g_des/w/w;   % calculate 'desired' radius to get desired acceleration
    if (r_des < rmin);
        r_des = rmin;
        w = sqrt(g_des/r_des);
        fprintf(['Warning :: tether_mass :: Calculated radius is below minimum, used minimum
radius of ' num2str(rmin) ' [m].\r'])
        fprintf(['                           Angular rate should be less than ' num2str(w*30/pi) '
[rpm].\r'])
    end
    alpha=2*pi/n;  % angle between the spokes
    beta=.5*(pi-alpha);  % angle betwen spokes and outer strands
    tlength1=n*r_des;    % calculate length for the spokes
    tlength2=n*2*r_des*sin(alpha/2);  % calculate length for outer strands

    F1=m*r_des*w*w;   % calculate tension in the spoke tethers
    F2=F1/2/cos(beta);   % calculate tension in the rim tethers

    A1=F1/sig_uts;    % required tether area is tension/ultimate tensile strength
    A2=F2/sig_uts;

    mass1=density*tlength1*A1 * 5;   % calculate spoke tether masses using factor of safety of 5
    mass2=density*tlength2*A2 * 5;   % calculate rim tether masses using factor of safety of 5

    mass=mass1+mass2;

    if (verbose == 1)
        fprintf(' \r')
        fprintf(['Designed a ''' AG_type ''' tether of ' tether_mat '.\r'])
        fprintf(['Desired radius: ' num2str(r_des) ' [m].\r'])
        fprintf(['Spoke tether length (total):  ' num2str(tlength1) ' [m].\r'])
        fprintf(['Outer tether length (total):  ' num2str(tlength2) ' [m].\r'])
        fprintf(['Spoke tether tension: ' num2str(F1) ' [N].\r'])
        fprintf(['Outer tether tension: ' num2str(F2) ' [N].\r'])
        fprintf(['Spoke tether area:    ' num2str(A1) ' [m^2].\r'])
        fprintf(['Outer tether area:    ' num2str(A2) ' [m^2].\r'])
        fprintf(['Spoke tether mass:    ' num2str(mass1) ' [kg].\r'])
        fprintf(['Outer tether mass:    ' num2str(mass2) ' [kg].\r'])
        fprintf(['Total tether mass:    ' num2str(mass) ' [kg].\r'])
        fprintf(' \r')
    end

else
    fprintf('Error :: tether_mass :: Unknown spacecraft type!\r')
end
```

propulsion.m

```
function mass=propulsion(AG_type, n, w, dry_mass, r_outer)

% function 'proplusion.m' calculates the required fuel mass for both
% "spin-up" and initiation of the interplanetary transfer orbit.

% dry_mass should be the dry mass of a single 'pod'

% Inputs:
%    AG_type [] - The type of system in question (monolith, multiple, tether, emff)
%    n [] - The number of spacecraft, only applicable for the pinwheel design
%    w [rad/s] - The desired rotational rate of the system [to be hardcoded?]
%    dry_mass [kg] - The dry mass of the spacecraft.  This should be a row
%       vector with appropriate dimensions as follows:
%           * Monolith: N/A
%           * EMFF: N/A
%           * Single Tether: [(habitation module)  (other mass)]    ###[1x2]
%           * Pinwheel: (mass of nodes, assumed uniform)     ###[1x1]
%    r_outer [m] - moment arm to the thruster
%
% Outputs:
%    mass [kg] - The mass of the propulsion system per pod

MU_s=1.327e20;  %m^3/s^2, gravitational constant for the sun
MU_e=3.986e14;   %m^3/s^2, gravitational constant for the earth
```

```
res=1.5e11;    %m, earth-sun distance
re_mag=6.38e6;   %m, earth radius

parking=200e3;  % parking orbit ALTITUDE in km

planet1 = 3;  % choose Earth as the origin
planet2 = 4;  % choose Mars as the destination

planet(1)=0.3871;    % define planetary radii for future use
planet(2)=0.7233;
planet(3)=1;
planet(4)=1.524;
planet(5)=5.203;
planet(6)=9.519;
planet(7)=19.28;
planet(8)=30.17;
planet(9)=39.76;

planet = planet * res;   % put planet distances in [m]

[r, v] = ic_circ(MU_s, planet(planet1));   % get circular initial conditions for the Earth
[dv, t_trans]=hohmann(MU_s, r, v, planet(planet2));  % calculate dV and transfer time for hohmann
to mars

[re, ve] = ic_circ(MU_e, re_mag+parking);   % get circular initial conditions for circular 200km
parking orbit
[eta, dv_earth, t_soi] = p_conic(MU_e, MU_s, norm(dv), re, ve, res);  % find the dV required to
Mars from the parking orbit

i_mass=r_equation(dry_mass, dv_earth);   % find the mass required for insertion

% s_mass is the spin-up propulsion system mass

if strcmp(AG_type, 'monolith')
    % assumes a pair of thrusters at the rim of the craft, to set up a couple
    dv_req=w*r_outer;
    s_mass=r_equation(dry_mass, dv_req);
elseif strcmp(AG_type, 'emff')
    s_mass=0;
elseif strcmp(AG_type, 'tether')
    dv_req=w*r_outer;
    s_mass=r_equation(dry_mass, dv_req);
elseif strcmp(AG_type, 'multiple')
    dv_req=w*r_outer;
    s_mass=r_equation(dry_mass, dv_req);
end

mass=i_mass + s_mass;
```

ic_circ.m

```
function [r, v] = ic_circ(MU, r_init)

% calculate circular initial conditions (position and velocity) given
%   a central body and an initial radius

r = [r_init 0 0];
vc = sqrt(MU/r_init);
v = [0 vc 0];
```

hohmann.m

```
function [dv, t_trans]=hohmann(MU, r, v, r_target)

% Function ip_hohmann takes the current (sun-centered inertial) position
%    and velocity vectors, verifies an initial circular orbit, and then
%    calculates the delta-V required at that instant to enter a
%    minimum-energy (Hohmann) transfer to a given radius (scalar).  The
%    function also returns the time of transfer, which is half the period
%    of the transfer orbit.

% MU is the gravitational parameter of the central body (m^3/s^2)
% r is the radius vector to the spacecraft (sun-centered)
% v is the velocity vector of the spacecraft (sun-centered)
% r_target is the orbital radius of the target planet

h = cross(r,v);       % angular momentum vector
h_mag = norm(h);
p = h_mag*h_mag/MU;     % orbit parameter
a = -1/((norm(v)^2)/MU-2/norm(r));     % semimajor axis
e = sqrt(1-p/a);
```

```
if ( e > .01)
    fprintf('Error :: IC :: Initial orbit is not circular!\r')
    fprintf(['Orbital eccentricity is ' num2str(e) '!\r'])
end

r_mag = norm(r);

a_trans = (r_mag + r_target)/2;
v_trans_i = sqrt(MU*(2/r_mag - 1/a_trans));

dv_trans = v_trans_i - norm(v);
dv = dv_trans*v/norm(v);

t_trans=2*pi*sqrt((a_trans^3)/MU)/2;
```

p_conic.m

```
function [eta, dv, t_soi] = p_conic(MU1, MU2, v_inf, r, v, rps)

% inputs:
%   MU1: MU for the primary body, i.e. escaping from Earth orbit
%   MU2: MU for the contending body in the SOI problem, typically the sun
%   v_inf: the scalar velocity required at r_inf to enter the desired
%      helicentric transfer.  Found by solving heliocentric problem.
%   r: the radius vector to the spacecraft
%   v: the velocity vector of the spacecraft
%   rps: the distance between the two SOI bodies, i.e. the Earth and Sun

% returns:
%   eta: the angle between the velocity vector of primary body and radius vector to s/c
%   dv: the scalar change in velocity required to get the desired v_inf
%   t_soi: the time required to reach the sphere of influence (SOI)

rc = norm(r);
vc = norm(v);

v1 = sqrt(v_inf^2 + 2*MU1/rc);    % required velocity at the parking orbit radius

energy = (v_inf^2)/2;
h = rc*v1;
ei = sqrt(1 + 2*energy*h^2/MU1^2);   % eccentricity of the transfer orbit


eta = acos(-1/ei);    % angle between the velocity vector of primary body and radius vector to
s/c
dv = v1 - vc;   % the required delta-v

p = 2*rc;
r_soi=(rps)*(MU1/MU2)^(2/5);    % the radius of the sphere of influence

f = acos((p/r_soi-1)/ei);   % the true anomaly there

H = 2*atanh(sqrt((ei-1)/(ei+1))*tan(.5*f));   % the hyperbolic anomaly there

N = ei*sinh(H) - H;   % kepler's equation for hyperbolas, N ~~ Mean anomaly


a = p/(1-ei^2);   % the 'semi-major axis' of the hyperbola (<0!)

t_soi = N/sqrt(MU1/(-a)^3);   % the time to reach the sphere of influence (r = r_inf, v = v_inf)
```

r_equation.m

```
function mass=r_equation(dry_mass, dV)

% !!!The Rocket Equation!!!

% function 'r_equation.m' takes the dry mass of the vehicle and the
%   required dV to gain the transfer orbit, and computes the mass of
%   fuel required for injection (given a particular thrusting system)
%
% Inputs:
%   dry_mass [kg] - The mass of the spacecraft w/o fuel
%   dV [m/s] - The change in velocity required to gain the transfer orbit
%
% Outputs:
%   mass [kg] - The mass of the fuel required
```

```
% Assume typical bipropellant chemical thruster w/ Isp ~ 350s

Isp = 350;
g = 9.81;

mass = dry_mass * (1-exp(-(dV/(Isp*g))));
```

emff.m

```
function [mass_coil] = emff(V, D,R, w, Mass_tot)
%The code uses the size of the cylinder as the size of the coils,
%the total dry mass each satellite, the radius of rotation,
%and the rotation rate to determine the mass of the coils
%for a three spacecraft collinear array.

r = D/2;              %m, Radius of the cylinder
l = V/(pi*r^2);       %m, Length of the Cylinder
Ic_pc = 16250;        %Superconducting coil current density divided by wire density.
w_rad = w * 2*pi/60;%converting rpm to rad/sec

mass_coil = w_rad/(l*Ic_pc)*sqrt(Mass_tot * (R + r)^5 /(3 * 17 * 10^-7));
```

cost.m

```
%This module determines cost of a mars transfer vehicle based on vehicle
%weight.

%References:
% 1. Reynerson, C. "Human Space System Modeling: A tool for designing
%inexpensive Moon and Mars exploration missions"

% 2. SMAD

function [TotalCost, SpaceSegCost, LVCost, OpSupCost]=cost(mass,TRL,duration)

%mass: is total mass of vehicle in kg
%TRL: is technology readiness level and should range from 1 to 9.
%duration: is duration of mission in years

Infl = 1.052; %inflation factor to convert from FY00$ to FY03$ [2]

%Space segment costs

%Space Segment Cost Factor
Scf = 157e3; % ($/Kg) we use maximum value to obtain conservative estimate [1]

%Program level Cost Factor
Pcf = Infl * 1.963*(523e6)^0.841; %($) Program level cost estimated from table 20-4 [2]

%RTDECF of 1 means program based on existing hardware,
%3 means new development program, 2 is somewhere in between [1]
%Rcf = 2;

%Heritage Cost Factor: assume that a TRL of 9 means 90% heritage,
%therefore 10% extra needs to be spent in RDTE, [2] pg 798.
Hcf = 2-TRL/10;

%SpaceSegCost = Scf*Rcf*Hcf*Mass+Pcf;
SpaceSegCost = Scf*Hcf*mass+Pcf;
%*****************************************************************

%Launch Vehicle Cost

%Launch Vehicle cost factor
Lcf = 15.4e3; %($/Kg) [1]

%Insurance cost factor
Icf = 1.5; %for commercial launches it is 1.33, for govt. we are assuming a bit higher number [1]

LVCost = Lcf*Icf*mass;

%*********************************************************************

%Ground Operations and Support Costs

OpSupCost = 1.5e9*(duration); %($) ISS operational budget is $13 billion for 10 years [1]


%**************************************************************************

TotalCost = SpaceSegCost+LVCost+OpSupCost;
```

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