Asteroid Rescue Mission

Georgia Institute of Technology

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Abstract

This paper is in response to a request for papers from the Lunar and Planetary Institute in Houston, Texas as part of a National University Competition. A human rescue mission to the asteroid 16 Psyche was designed based around a failed Mars mission scenario. The scenario assumed the second human Mars mission, based on the Mars Design Reference Mission 3.0, failed to propulsively capture into Mars orbit, resulting in a higher energy trajectory headed towards the asteroid belt on an intercept trajectory with 16 Psyche. The task was to design a mission that could rescue the astronauts using existing Mars mission hardware prior to the failure of their life support system.

Analysis tools were created in the following six disciplines for the design of the mission: trajectory, propulsion, habitat and life support, space rescue vehicle and earth reentry vehicle, space transfer vehicle, and operations. The disciplinary analysis tools were integrated into a computational framework in order to aid the design process. The problem was solved using a traditional fixed-point iteration method with user controlled design variables. Additionally, two other methods of optimization were implemented: design of experiments and collaborative optimization. These were looked at in order to evaluate their ease of implementation and use at solving a complex, multidisciplinary problem. The design of experiments methodology was used to create a central composite design array and a non-linear response surface equation. The response surface equation allows rapid system level optimization. Collaborative optimization is a true multidisciplinary optimization technique which benefits from disciplinary level optimization in conjunction with system level optimization. By reformatting the objective functions of the disciplinary optimizers, collaborative optimization guides the discipline optimizers toward the system optimum.

The size and complexity of this design led to severe problems for the advanced optimization methods. The design space was non-smooth, multi-modal, and highly non-linear. Gradient based optimizers could not dependably gather gradient information or find their way out of local minima. Response surface methods produced poor results due to the non-quadratic nature of the design space. Therefore, the traditional fixed-point iteration method proved to be the most easily implemented and produced the best results.

1 Introduction

1.1 Purpose

This paper, in conjunction with the presentation given at the Lunar and Planetary Institute (LPI) in Houston, Texas in addition to several out-reach programs, is part of a National University Competition sponsored by LPI. The competition is an open-ended competition in which the competing universities are allowed to choose their project to design. The only constraint placed on the project is that it must be applicable to the Human Exploration and Development of Space.

The design proposed by the Spacecraft and Launch Vehicle Design class at the Georgia Institute of Technology is a human mission to a near Earth asteroid.

This paper out-lines the process that was used by the Georgia Institution of Technology design team in the design of a manned vehicle to travel to an asteroid in the main asteroid belt. This paper has two main focuses. The first major focus is on the individual tools that were developed specifically for this project in the following areas: trajectory, propulsion, habitat, space rescue vehicle /Earth return vehicle (SRV/ERV), space transfer vehicle (STV), and operations. The second focus is on the method of optimization that was employed by the design team in order to develop a converged design that would meet the mission success and time constraints while maximizing safety and minimizing cost.

1.2 Disaster Scenario

It is the year 2020 and the second manned space vehicle has been sent to Mars, carrying an international crew of six. A malfunction occurs during the vehicle's voyage, causing the vehicle to swing by Mars, hurtling through space towards the asteroid belt. They have no way of returning home using available hardware, and the length of time their food supply and life support system will last is limited. The design presented herein is in response to these events, aimed at a mission intended to rescue the crew from the asteroid belt and return them safely to earth. The ill-fated mission is part of the Mars Design Reference Mission so a brief description of the this is given in the next section, followed by a detailed description of the actual disaster.

1.2.1 Design Reference Mission

The Mars Design Reference Mission (DRM) is a product of the NASA Exploration Study Team. The DRM was last amended in June of 1998. The DRM has two primary roles. First, it serves as a template to which alternative approaches to the human exploration of Mars can be compared and contrasted. Second, it is intended to stimulate additional ideas and further progress in exploration. For this design, it is assumed that the DRM has been implemented into NASA's space program and that actual manned missions to Mars are taking place.

The DRM can be broken down into a single mission architecture, which is comprised of the following sequences of events:

1. In the first launch opportunity, two cargo missions are launched to Mars; one cargo vehicle carries a lander with a propellant production plant and ascent vehicle, the other carries an earth return vehicle. Each cargo mission requires the use of two Magnum launch vehicles.
2. The cargo lander lands on the surface of Mars where the propellant production plant produces and stores methane and liquid oxygen from the Martian environment. The Earth return vehicle enters into Mars orbit.
3. In the second launch opportunity, which is generally about twenty-six months after the first launch opportunity, the crew transit vehicle is launched. This vehicle carries along with it the crew lander. This also requires two Magnum launch vehicles. The crew reaches Mars in 130-180 days on a fast transit trajectory.
4. The crew performs various scientific activities on the surface of Mars for approximately 520-580 days.
5. The crew then uses the ascent vehicle from the previous launch opportunity (which now has stored enough locally produced methane and LOx for ascent) to rendezvous with the earth return vehicle that is waiting in Mars orbit.
6. The crew returns to earth on a 130-180 day fast transit trajectory

According to the DRM, two cargo missions would also be launched in the same year as the second launch opportunity of the previous mission, and another crew transit vehicle would launch in the following launch opportunity. One complete mission architecture is comprised of two cargo vehicles launched in one opportunity and one crew transit habitat launched in the following opportunity. Each mission architecture requires a total of 6 Magnum launch vehicles spread over two launch opportunities.

1.2.2 Disaster Events Timeline

December, 2020

Two cargo vehicles were launched on May 11, 2018 as support for the second manned mission to Mars. A crew transit vehicle departed for Mars on July 29, 2020. The crew vehicle is scheduled to perform a propulsive capture into Mars orbit and descend to the surface in the crew lander on December 6, 2020. However, as the crew vehicle approached Mars, one of the reaction control thrusters on the transfer vehicle failed and the crew was unable to achieve the correct orientation for a propulsive capture into Mars orbit.

Since the main engine could not perform the deceleration ΔV at Mars, the spacecraft completed a Mars swing-by and hurtled towards the asteroid belt. The swing-by at Mars provided an increase in velocity of 2200 m/s, creating a higher energy heliocentric trajectory that does not encounter the Earth at any time. As a result, NASA mission control decided to have the spacecraft prepare for a safe landing on an asteroid. This is done in order to minimize the difficulty of rendezvous and rescue.

Table 1-1: Disaster Trajectory Characteristics

Variable	Value	Units
$\Delta V_{Gravity\text{-}Assist}$	2242	m/s
Eccentricity - ε	0.419	
Aphelion Radius - r_a	3.190	AU
Perihelion Radius - r_p	1.307	AU
Semi-Major Axis - a	2.249	AU
Period - P	1232	Days
$TOF_{Mars\text{->}Psyche}$	532	Days

The best candidate asteroid in terms of location and relative velocity to the spacecraft is asteroid 16 Psyche. Psyche was chosen since it will lie near the spacecraft's current trajectory while also having a velocity vector close enough to that of the spacecraft. The spacecraft will then use the Mars lander's propulsion system to perform a soft landing on Psyche. The spacecraft will travel for 372 days until encountering Psyche and performing a propulsive ΔV to land.

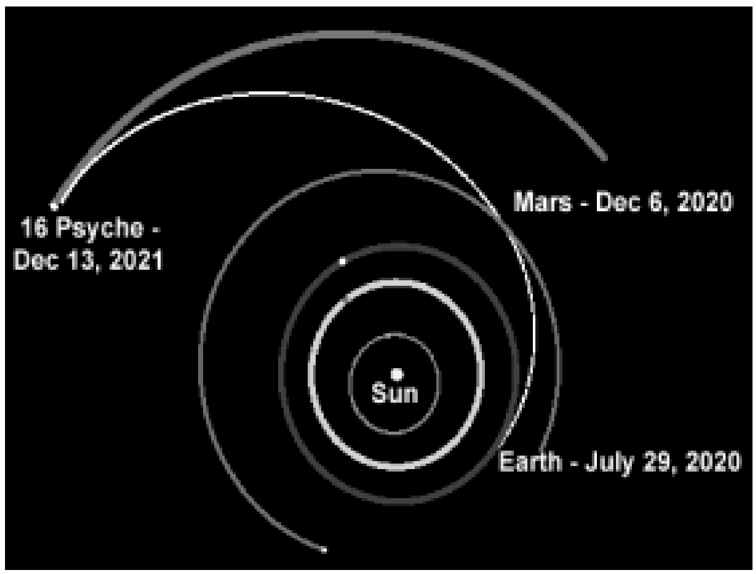

Figure 1-1: Disaster Trajectory Picture

1.2.3 Disaster Trajectory

After the swing-by of Mars, the crew gains velocity and begins an eccentric orbit through the asteroid belt. Table 1-1 provides the characteristics of the new disaster trajectory after the gravity-assist around Mars. Figure 1-1, is a picture of the disaster trajectory from launch at Earth, through the gravity-assist around Mars, to intercept of the asteroid Psyche.

1.2.4 Asteroid Physical and Orbital Characteristics

Psyche is an M-class asteroid composed primarily of a nickel-iron compound. The semi-major axis of Psyche is nearly twice that of Mars, and the orbital period is 5.0 years. Psyche is chosen as a good asteroid to intercept because it has a relatively small eccentricity and inclination as compared to other candidate asteroids. This means that a rescue mission to Psyche will require a smaller total ΔV than one in which a large motion outside of the ecliptic plane is required. Table 1–2, below, provides the physical and orbital parameters of Psyche.

Table 1–2 Physical and Orbital Parameters of Psyche

Physical Parameters of Psyche

Parameter	Value	Units
Diameter	253.2	km
Rotational Period	4.2	hr
Density - ρ	8000	kg/m^3
Mass - m	$6.8*10^{19}$	kg

Orbital Parameters of Psyche

Parameter	Value	Units
Semi-Major Axis - a	2.923	AU
Period - P	5.00	Years
Eccentricity - ε	0.139	
Inclination - i	3.09	Degrees

1.3 Constraints

The goal of the mission is to safely rescue the six astronauts stranded on Psyche. At first glance this appears to be simple, but under close scrutiny there are many constraints the problem imposes. First, the crew's survival time on the asteroid is limited. The resulting time constraint is a result of life support system durability and food supply.

The Mars DRM gives a trip time of 120-180 days to reach Mars with a planet stay time of 520-580 days. This yields a nominal LSS operation span of 760 days. It was assumed that the nominal operating time would be designed to a 3σ (approximately 99.97%) probability that the LSS will fail within 760 days. A conservative estimate is that there is a 50% probability that the LSS not fail with 1.5 times the nominal 760 days, or 1170 days. From these two estimates a crew survival probability plot (see Figure 1-2**)** was generated based on the assumption that once the LSS fails, the crew will die. It is assumed for this design that an 80% or greater probability of crew survival means mission success..

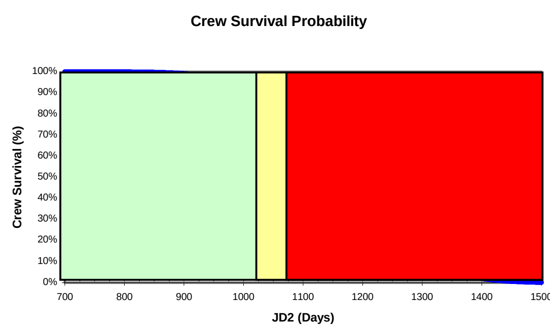

Figure 1-2: Crew survival probability.

Given that the Mars mission was considered to last a maximum of 760 days, food supplies would have to be stored for this duration. Once the Mars vehicle passes Mars, the crew would have 620 days of food remaining. They will be stranded in space for an unknown amount of time prior to rescue, so the logical course of action would be to put the crew on half-rations. This would allow remaining food to last 1240 days. The 80% crew survival probability based on LSS longevity is the dominating factor in the mission's success, rather than potential food shortage.

Given this time constraint, it is important that the required hardware be produced rapidly. Assuming the subsequent Mars mission has been forfeited, existing hardware is available for modification and use, as described in section 1.2.1. Therefore, the rescue mission design will primarily be a modification of existing DRM hardware in order to create a solution to the design problem.

2 Solution Process

A mission architecture was chosen based on preliminary work showing that a direct route to and from Psyche requires the lowest ΔV transfer within the time constraint. A ballistic reentry was also chosen due to lower overall mass. In order to get a better handle on the problem it is divided into sub-disciplines. This allows a more detailed approach be taken as each group member can become an expert in a specific field related to the problem. However, breaking up the design process has the disadvantage that the team knowledge becomes spread out, and thus a communication system must now be used to pass the necessary information between disciplines. More importantly, it is desirable to derive an optimal solution not only at the discipline level, but also at the system level. Two organized methods of doing this, namely Fixed Point Iteration (FPI) and Collaborative Optimization (CO), are described below along with the individual sub-disciplines.

2.1 Optimization

Numerical optimization methods were used to assist in the determination of a design solution that maximizes the chance of survival of the Mars astronauts while meeting all mission constraints. Several different system level optimization methods were used: fixed-point iteration (FPI), design of experiments (DOE), and collaborative optimization (CO). Additionally, several discipline level optimizers were used, to include, sequential quadratic programming and genetic algorithms.

All three system level design methods were implemented using a commercial framework integration code, ModelCenter. ModelCenter is capable of integrating multiple computer codes in several different formats, on different types of computers, running on separate workstations. ModelCenter automates the passing of information between the different disciplinary analysis codes, includes built in system optimizers, and can automatically iterate a design until it is converged. ModelCenter is run through an intuitive, graphic interface that allows the problem to be set up in a reasonable short amount of time. The chief drawback found with ModelCenter is its inability to parallel process even when the design is structured to allow it.

2.1.1 Fixed-Point Iteration

At the system level, the design of a space vehicle contains a feedback loop of several variables. A consistent design requires a method in which input variable values used by one discipline are the same as those output by previous disciplines. The most straightforward way of converging a design of this type is to use fixed-point iteration (FPI). This method starts with intelligent guesses of the variable values that have feedback loops. Each discipline then runs its own analysis, passing the results to the next discipline in line. When the last discipline is finished, the output is compared to the initial input and a new guess is made. This process is repeated until the output variables are the same as the input variables within a desired tolerance. Using a design structure matrix (DSM) as depicted in Figure 2-1 illustrates this process. In a DSM, lines above the disciplines represent feed-forward loops, while lines below the disciplines represent feedback loops. Solid dots represent a link between disciplines. Often the order of the disciplines can be changed to reduce, or even eliminate, feedback loops. The FPI solution has three feedback paths that require convergence.

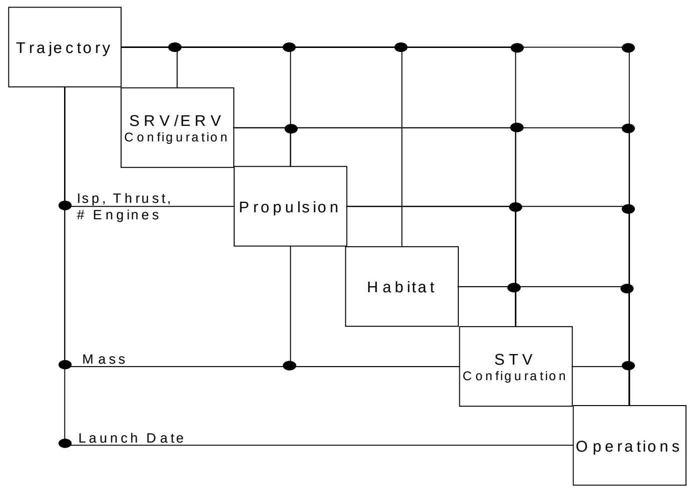

Figure 2-1: Design structure matrix configured for fixed-point iteration.

This method has the advantages of being simple to implement and leaving each discipline expert in control of his own area of expertise. Interestingly, the process can converge quickly if an intelligent scheme is used for guessing the next input in the loop. This scheme can even be made to rely on the knowledge of an experienced designer. Since a system level optimum requires a compromise between the desires of each discipline with a system level goal in mind, FPI can result in discipline level optima at the expense of system level optimum. FPI does not have the capability of reaching a compromise between disciplines, but merely ensures these disciplines are using consistent input and output values.

2.1.2 Design Of Experiments

Design of experiments (DOE) is a system level optimizer that finds an optimum value using a curve fit of the design space. A different optimization scheme is then used on the resulting response surface. To create the response surface an experimental array is used to define specific combinations of design variable values for each run. This design problem was pared down to seven design variables leaving the remainder fixed. Using a full factorial array all interactions between variables can be captured, but would require 2187 runs. The computation time required for this many runs is prohibitive. To reduce the number of runs a central composite design (CCD) was chosen. CCD uses fewer points while still capturing some variable interactions, and only requires 143 runs. If the interaction terms in the response surface are chosen well, the model will still accurately represent a quadratic design space. Knowledge of several outputs is desired which requires a separate surface for each output. With multiple outputs, optimization must be run using an overall evaluation criterion with weighting factors on each output. DOE is used in conjunction with FPI to find a converged solution for each run in the experimental array.

2.1.3 Collaborative Optimization

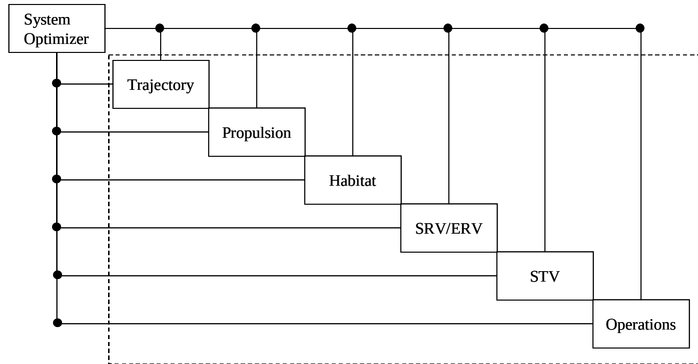

Figure 2-3: Design structure matrix configured for collaborative optimization.

Collaborative optimization (CO) is a true system level optimization scheme that is capable of forcing the disciplines to work toward a common goal while finding a consistent solution. The principle behind CO is that each discipline maintains control over any variables that do not affect other disciplines. The scheme leaves optimization of discipline level metrics with the discipline experts. Conflict between discipline and system level optimizers is resolved by changing the discipline level objective function. For example, instead of trying to minimize the I_{sp} in the propulsion module, the discipline optimizer will minimize the error, or J-value, between a system provided target I_{sp} and the local I_{sp}. For every variable that links disciplines, there must be a system level variable and an associated J-value. In order to achieve a consistent solution, all J-values must be identically zero. This allows the system level optimizer to control the design within feasibility limits. The discipline is still able to freely control any variables that do not couple with other disciplines; therefore control of details is left to the disciplinary expert. This also reduces the size of the problem for the system level optimizer and hence minimizes computation time.

CO has the distinct advantage of keeping the discipline expert in control of his or her own discipline and hence confidence in the final design is maintained. Furthermore, a true system level optimum is achievable since the system optimizer can find a balance between conflicting disciplinary interests. Since each discipline now only depends on the system level optimizer for input as shown in Figure 2-2, analyses can be run in parallel, reducing the overall computation time. Despite fixing many of the problems of FPI, this system is not without its disadvantages. Setup is significantly more difficult as legacy codes often require extensive modification in order to accept a new objective function. In addition, the use of a gradient-based system level optimizer necessitates high quality output from the disciplines. This means that the discipline level optimizer must consistently find the best solution such that the system level optimizer receives accurate gradients.

2.1.4 Discipline Level Optimization

In order for the system level optimizer in the collaborative optimization scheme to accurately find gradients of all the coupling variables, it is necessary that each disciplinary code return the true optimum solution every time. For example, if the system level optimizer perturbs one coupling variable a small amount in a 'good' direction, but the disciplinary level optimizer fails to find the better solution (or finds a worse solution) the system level optimizer will conclude that this coupling variable has no effect (or a 'bad' effect) on the overall objective function. The proper selection of discipline level optimizers is therefore critical to the performance of the overall optimization.

Discipline level optimization was initially attempted using a sequential quadratic programming (SQP) method. Although SQP is an extremely effective optimization method, it requires that the design space be unimodal and smooth. Most of the disciplinary codes have design spaces that are non-smooth and have several local minima. This is caused not only by the complex interactions of the variables, but by the fact that several design variables in the codes are either integer valued or behave as step functions. When testing the SQP optimizer on the remaining disciplines, occasional instances were found in which a global optimum was not reached. Thus even in these disciplines the design space was not well enough behaved to instill confidence that SQP returns the best solution.

Optimization methods suitable for use in non-smooth, multi-modal problems fall into the category of heuristic methods. Both a genetic algorithm (GA) and a tabu search optimizer were tested on each discipline. The GA method proved to converge to better solutions and did so in less time, although it still took much longer than SQP. For example, most of the codes would take less than 10 seconds to converge using SQP but take several minutes with GA. It was observed, however, that after the first minute only minor improvements were found, and so the optimization time was capped at 2 minutes. The GA optimizer also employed a local SQP search on the best designs in its population to improve its performance. A separate GA optimizer was linked to each disciplinary code, with the population size, mutation rate, and convergence criteria set for best performance on that particular code.

The GA optimizer used for the spreadsheet-based discipline codes is a commercial optimizer made by Frontline. Frontline's GA optimizer, Evolutionary Solver, integrates well into Microsoft Excel and can be automated using a Visual Basic (VB) routine. The VB code attached to every spreadsheet loaded the constraints and design variables into the Evolutionary Solver, which then returned the solution. The STV Configuration code had difficulties due to its iterative solution technique for finding vehicle mass - an occasional individual design selected by the GA optimizer would fail to iteratively converge, crashing the spreadsheet and optimizer. Thus, additional VB coding was needed for this discipline in order to reset the inputs and outputs whenever this happened.

For Matlab based codes, both SQP and GA were used. The SQP optimizer is Matlab's built in optimization function, fmincon, available in the optimization toolbox. The GA is taken from North Carolina State University's Meta-Heuristic Research and Applications Group. Specifically, the Genetic Algorithm Optimization Toolbox written in Matlab was used. These two methods were used together to reduce the time required to find the minimum. Each time a Matlab code is called it first runs SQP to see if it can find a minimum within the desired tolerance close to the starting position. If it cannot, then the GA is called with a fixed number of iterations. This ensures that the entire design space is searched and the global minimum is found. After the GA terminates, SQP is run to refine the best solution. This method permits the code to run fast when it is close to the global minimum (which is typically the case when finite difference derivatives are taken), but not get trapped in local minima.

2.2 Tools

Disciplinary design codes were developed for each box on the DSM in Figure 2.1. Cost and safety calculations have been moved into each discipline to lower the number of coupling variables necessary for CO. The space rescue vehicle and earth reentry vehicle have also been combined since preliminary work showed that the vehicles had many similar requirements and could share a platform. Because of the tight coupling between launch vehicle scheduling and ground manufacturing and operations, the launch vehicle and operations modules were combined. This reduced the number of disciplines from ten to six, significantly reducing the number of coupling variables, and hence the load on the optimizer. The application of both FPI and CO was done using ModelCenter as a framework to automate the process. Microsoft Excel and Matlab were chosen as the programming languages for their ease of integration with ModelCenter. Following are descriptions of each disciplinary module.

2.2.1 Cost

Since system cost is of prime importance to the design, cost models were carefully chosen from historical data when possible and modified when necessary. To determine the cost of the rescue mission, the following key assumptions were made:

- All hardware for the next Mars launch (one complete mission architecture) is available for the rescue mission, including six Magnum launch vehicles along with all hardware associated with the crew lander, cargo lander, and earth return vehicle.
- Additional costs arise from modifications to existing hardware and required manufacturing time.
- The design does not include any calculations for profit (the goal is simply to rescue the astronauts).

Cost was integrated into each discipline's design worksheet. This eliminated the need for a separate cost worksheet, thereby reducing the number of design variables required for the CO process. So, cost was calculated within each discipline and then sent as an output to the optimizer where the total cost is calculated.

Cost estimating relationships (CERs) were used to determine the costs incurred by each discipline. These cost estimating relationships are basically mathematical equations relating the cost of a specific piece of hardware to some characteristic performance parameter, usually based on mass. The CERs had the following form, where c is a complexity factor, and a and b are constants:

$$Cost = c\left\{a(mass)^b\right\}$$

CERs are generally based upon historical data, where costs of previous missions are analyzed and curve fits are applied to the data. However, since the rescue mission is a manned space mission, there is limited historical data upon which to base the CERs. Therefore, the CERs were based upon the data provided by NAFCOM96, a CER database for launch vehicles. Since the CERs in the NAFCOM96 database were based upon launch vehicle data, the CERs had to be slightly modified to account for both differences between launch vehicles and manned space missions and the accelerated schedule of hardware design, development, testing, and integration. These modifications were usually made to the complexity factors.

2.2.2 Safety

Successful completion of the mission requires two things – the stranded astronauts must be recovered alive, and the mission must not incur any further loss of life. To this end it is important to design a vehicle that does not simply meet the functional requirement of the mission, but attains the mission objective while minimizing the risk to the rescue personnel and the recovered crew. Since the Mars DRM hardware is to be used in a modified form, it makes sense to compare the safety of the rescue mission with that of the original DRM. A scale was defined where a value of 1.0 represents a safety factor equal to that of the discipline's DRM counterpart. Values less than 1.0 represent a decrease in safety, whereas values greater than 1.0 reflect improved safety. A lower bound exists at zero since this represents certain mission failure.

Specific key factors are used to calculate the safety of each discipline. These factors are further described in each discipline's sub-section. Once these key factors have been identified based on the advice of the individual disciplinary designer, a normal distribution of reliability is placed on each factor. Weighting the relative importance of the contributing factors within each discipline is accomplished by means of exponents. All the weighted contributing factors are then multiplied together and output as the safety of that discipline. Figure 2-3 shows a sample calculation for the propulsion module. All six discipline safety values are multiplied together yielding the overall mission safety factor.

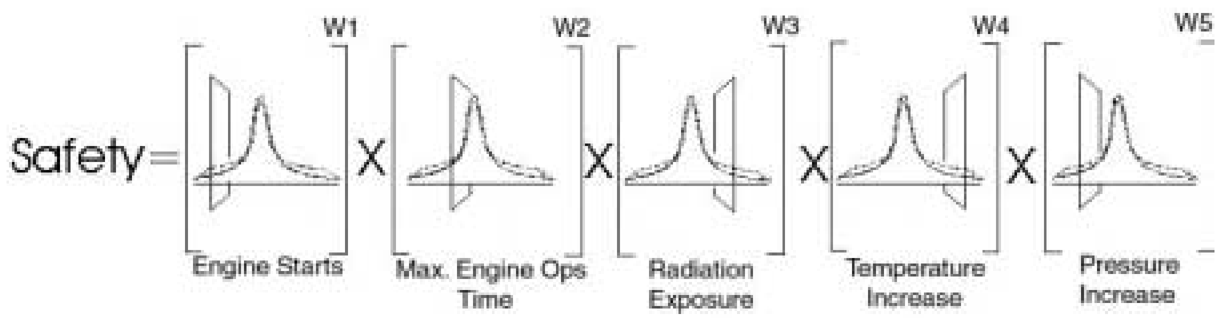

Figure 2-3: Sample safety calculation for the propulsion module.

2.2.3 Trajectory

A new trajectory must be found for the rescue mission that intercepts the stranded astronauts in a reasonable time and with a reasonable ΔV requirement. The ΔV required at each burn and the earth entry velocity are calculated using a patched conic method. Due to the computation time required to execute the analysis, and the large number of function calls typical of system level optimization, a program was written to create tabular data of the patched conic method output parameters. Data was obtained for all four ΔVs, the departure inclination angle, and the minimum distance to the sun on each transfer. Only direct transfers from the Earth to Psyche were considered since early sweeps indicated that inner planet swing-bys are not advantageous. Gravity losses at each burn are approximated and added to the ΔV found using the table data. Additional ΔV is also added if the desired heliocentric inclination angle is outside the range achievable from a 28.5° earth inclination through departure timing. The following are parameters that can be changed in the trajectory module.

- Julian dates: the absolute departure date and trip time determine the alignment of the planets and the energy required to move between them. These are the primary influences on the magnitudes of the required ΔVs.
- Wait time: this is the time that the rescue vehicle spends on Psyche before departing. This allows the return trip time and departure date to be varied.

There are eight system level inputs to the trajectory module: the mass of the STV at each of four thrust maneuvers, the thrust of an engine, the number of engines at the initial burn, the number of engines for the three remaining burns, and the specific impulse of the engines. The trajectory code outputs the Julian dates for Earth departure, asteroid arrival, and earth arrival, the ΔV required at each of four maneuver points, the Earth arrival entry velocity, the Earth departure inclination, the minimum distance to the Sun, and the safety relative to the original Mars mission based on number of thrust maneuvers.

2.2.4 Propulsion

The baseline engine for this mission is the Mars DRM 3.0 tri-carbide bi-modal NTR using liquid hydrogen as a propellant. The propulsion code uses the analysis approach in Space Propulsion Analysis and Design by Humble. An efficiency coefficient was applied to the generic equations in Humble in order to reproduce the design specification for the NTR as listed in the Mars DRM 3.0. Changes could be made to optimize the engine performance for this mission. Since time is the major constraint for this mission, only minor changes to the engine were considered. The following are parameters that could be reasonably changed within the mission time requirements.

Table 2-1: Baseline NTR Specifications.

Thrust	kN
Isp	960 s
Expansion Ratio	300
Percent bell nozzle	110%
Mass	1830 kg
Length	3.2 m
Diameter	1 m
Power	25 kWe

- Propellant Temperature and Reactor Pressure: it was assumed that the baseline engine would be designed to operate below the true maximum temperature and pressure to increase the safety of the engine; thus, increased performance can be traded for decreased safety in the manner that the SSME is operated at 109%.
- Nozzle Geometry: Increasing the nozzle's expansion ratio and length increase the engine's performance, but at the cost of added mass and size. Additionally, designing, manufacturing, and integrating a new nozzle requires a large amount of time.
- Number of Engines: Since the thrust level of the engines can only be varied over a small range, large increases in total thrust must be accomplished by adding more engines. If too few engines are used, gravity losses become significant.

There are ten system level inputs to the propulsion code: the mass of the STV at each of the four trajectory thrust maneuvers and the change in velocity associated with each of these maneuvers, the required total thrust, and the required I_{sp}. The propulsion code provides the following parameters as output to the system level: engine I_{sp}, thrust per engine, size (length and diameter) of an engine, the mass of an engine, the total time to prepare the propulsion system for the rescue mission, the cost of the modifications to the propulsion system, and a safety factor of the propulsion system relative to the Mars mission.

2.2.5 Habitat

One of the important components of the rescue vehicle is the habitat. Due to the time constraint and higher required crew capacity, the only suitable habitat is a modified version of the one designed and used for the Mars mission. The life support system (LSS) needs to be modified in order to make it adequate for the rescue mission, which necessitates a longer operational time and higher load than for which originally designed.

The habitat used is the Transhab, the same habitat as in the Mars DRM. The Transhab is an inflatable structure that weighs approximately 13,200 kg, has a length of 11 m, an inflated diameter of 8.2 m and an inflated volume of 339.8 cubic meters. Figure 2-4 shows a cut-away view of the Transhab.

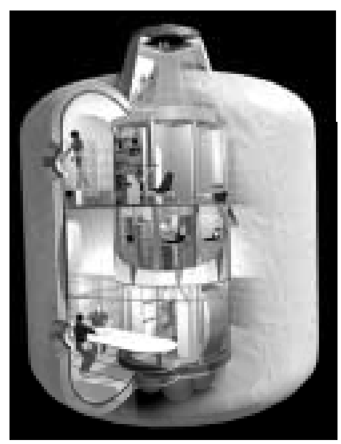

Figure 2-4: Cutaway view of the Transhab.

The first decision to be made is the number of crew assigned to the rescue mission. The Transhab was designed to support a crew of six. It was assumed in the initial problem that all six astronauts going on the Mars mission would be alive and rescued. That means that if a large crew were sent out to the asteroid there would be a problem accommodating everyone on the return trip. Therefore it was decided that the rescue crew would include three people, the minimum number of people required to operate the rescue vehicle. This would mean that a total of nine people would be on the return trip. This is a larger number than what the Transhab was designed for but it was determined to be acceptable due to cost and time constraints.

The second area that must be examined is the Life Support System. The LSS system designed for the Mars mission would require modifications to support a crew of 9 instead of 6 for an operating time greater than the 760 days it was designed for. The Advanced Life Support Research and Technology Development Metric developed by NASA was used as the basis for a detailed breakdown of the LSS. The Advanced Life Support System (ALSS) was designed specifically for use with the Transhab and is a derivative of the LSS used aboard the International Space Station (ISS). A summary of the ALSS is as follows: Mass=3,900 Kg, Power=14.5 kWe, and a single person's food intake is 1.7 kg/day. The ALSS was designed for an operating time of 400 days and a crew of six. When modifying the ALSS the dry mass was scaled by multiplying it by a ratio of the operation time of the rescue mission (Earth to the Asteroid and back to Earth) divided by the number of days the ALSS was originally designed for. Everything else about the ALSS remains the same.

Another component of the LSS that required some attention is radiation shielding for the crew. While the Transhab is designed to protect the crew from in-space radiation and solar radiation, it is not designed to travel closer to the sun then the Earth's orbit. The possibility existed that a transfer orbit to or from 16 Psyche would cross inside the Earth's orbit. In this case additional shielding would have to be added to the Transhab to provide protections against radiation.

The above assumptions and decisions made about the Transhab were incorporated into an Excel spreadsheet. The required inputs are the departure date from Earth, the arrival date at the asteroid, the date of arrival back at Earth, and the closest distance the vehicle will get to the Sun. The dates are used to scale the LSS mass and power to the mission, along with the amount of food that would have to be taken. The closest distance the vehicle gets to the sun is used to determine the amount of additional radiation shielding needed. The only internal variable is the safety margin on the LSS. This is included in order to give the module's optimizer a way to change the safety and hence match system level goals. Increasing the safety margin increases mass, cost, and safety of the habitat.

The outputs calculated by the habitat spreadsheet are the overall mass of the modified Transhab, the time required to make any needed modifications, the cost of those modifications and the safety of the modified Transhab compared to the one used on the Mars mission.

2.2.6 Space Rescue Vehicle/Earth Reentry Vehicle

The selection of a baseline vehicle comprises the majority of the work completed for this module. Presented is a brief description of the baseline vehicle followed by an explanation of the resulting module.

After exploring several high cost/high risk rescue options, the following method, which combines the forecasted benefits of hardware availability (i.e. low cost and minimal preparation time), common hardware design, fabrication, and installation procedures, mature technology, and reduced operator training, was adopted. This yields the best low-cost, reliability alternative for the mission. The resulting SRV/ERV design uses the latest DRM Mars Ascent Vehicle (MAV) capsule with the following modifications.

- Replace the LOX/CH_4 tanks and RL-10 engines in favor of a more easily stored, commercially available MMH/N_20_4 system coupled with an R-40B engine and matching nozzle based on the Space Shuttle design.
- Add a bolstered heat shield almost identical to the DRM but with more mass. Bolstering the heat shield is necessary due to the planned high ballistic reentry velocity at Earth.
- Add a touchdown system for asteroid landing / lift-off support via Apollo lunar module style collapsible landing legs (i.e. collapsed for launch vehicle packaging).

Once on the surface, the 2-man rescue crew performs a surface EVA in order to extract the downed space crew using extra tooling brought along to aid the ground rescue operations and add mission redundancy. Next, the 8-man crew departs the asteroid surface in the SRV capsule and mates with the remaining portion of the STV. After the earth return portion is completed, the landing legs and ascent/descent propellant tanks are jettisoned, a propulsive braking ΔV is performed, and the capsule is used for ballistic reentry.

Several key assumptions about the mission were made in developing the SRV/ERV design spreadsheet. Only significant assumptions are listed. First, the circular parking orbit at the asteroid was assumed to be 10 km. This is important for determining the asteroid ascent and descent ΔV along with the associated propellant required for mission completion. The gravitational constant μ was estimated for the asteroid. Next, it is assumed that the downed crew is stranded on the asteroid surface and thus requires rescue vehicle touchdown on the surface. Note that several constant design parameters are taken directly from the Mars DRM. Other parameters not explicitly stated in the DRM were estimated. Finally, the packaged SRV diameter is assumed to be 7.5m. Earth reentry calculations for maximum heat flux rate use the Allen-Eggers approximation, which linearizes the initial high acceleration portion of the trajectory. The resulting heat flux rate places an upper limit on the reentry angle at the fusion point of carbon. The entry corridor is also limited on the low end by the angle at which the vehicle skips out of the atmosphere.

This baseline vehicle was modeled parametrically. Only the following variables were changed in the model.

- Rescuer tooling mass: this provides a redundant operation benefit, yielding a safer rescue process. Increased safety is based on a utility per unit mass calculation.
- Propellant tank radius: this variable is primarily affected by both the required descent and required ascent ΔV, which drives the required propellant mass. This variable primarily affects overall vehicle dimensions and mass.
- Propulsive brake: part of the Earth approach velocity can be reduced by performing a combination of two propulsive ΔVs on Earth approach, one done by the STV at the Earth sphere of influence and one by the SRV/ERV near atmospheric reentry. The latter of these increases the mass of the SRV/ERV as the heat shield design mass increases but this improves safety, which in turn helps keep g loading within limits.
- Landing leg diameter: the diameter of the landing legs when unfolded can be modified for increased stability on landing. This increases the vehicle weight, and cost, and safety.
- Earth reentry angle: the reentry velocity has the largest impact on the value of this variable. Changing it can modify the heat shield mass and crew g load, and hence the cost and safety.

The SRV/ERV design uses the velocity at reentry and desired Earth return ΔV (in the case where a slightly propulsive ΔV is required when entering the Earth's sphere of influence in order to ensure that a safe g loading will be applied to the crew upon reentry). The process outputs are vehicle mass, length, preparation time, cost, and safety relative to the Mars DRM.

2.2.7 Space Transfer Vehicle

The baseline configuration for this mission is outlined in the Addendum to the Mars DRM 3.0 as the "all propulsive bimodal NTR carrying Transhab" vehicle. The baseline vehicle is the earth return version outlined in the DRM. It was assumed that the option to use a Transhab as an in-space habitat was chosen. Minor modifications could be made to one of these vehicles, and parts of identical vehicles built for other missions could be added on quickly and with little cost penalty.

The Space Transfer Vehicle (STV) spreadsheet uses the ideal rocket equation to calculate total mass at four distinct points throughout the mission. The spreadsheet uses the ΔVs from the system level optimizer, along with the I_{sp}, to calculate the amount of propellant required for each burn. When propellant mass is added to dry mass, total mass increases. More propellant is then required for the same ΔV. Additional propellant mass increases total mass and again more propellant is needed. Numerical iteration is used to find the points at which there is enough propellant at each of the four mission stages to provide for the required ΔVs. Several factors contribute significantly to the additional mass of the STV. These factors and the input and output values they affect are summarized as follows:

- Specific Impulse: the specific impulse of the engines has a large affect on the amount of propellant required, and thus the overall mass of the vehicle. Even small increases in I_{sp} reduce total mass at each mission point significantly.
- Engine Mass, Habitat Mass, SRV Mass: each of these components is present for most of the mission. The amount of propellant required to move them all decreases rapidly as their masses decrease.
- Number of engines: the number of engines at trans asteroid injection (TAI) is important because it dictates the number of baseline 'core' modules that are added to the baseline vehicle. The baseline vehicle has three engines. By adding additional, pre-existing 'core' modules taken from other baseline vehicles, the number of engines at TAI can be increased by factors of 3 with little cost penalty. Additional engines require the modification of the baseline vehicle and incur cost and time penalties. Since the subsequent burns require only three engines, additional core modules are jettisoned after the TAI burn to reduce mass.
- Time in LEO, trip time: propellant boil-off is calculated based on these times. As time increases, the amount of propellant lost increases, so more must be taken.
- Reinforcing structure, connecting hardware, fuel pumps & lines, structural redundancy factor: propellant that cannot be contained in onboard tanks is stored in external tanks attached to the STV. Reinforcing structure mass, connecting hardware mass, and fuel pumps & lines mass all depend on the amount of propellant carried externally. Propellant stored in external tanks is burned first, empty tanks are jettisoned immediately to decrease mass.
- Safety: several factors are used to calculate the safety of the STV relative to the DRM vehicle. Safety increases as the structural redundancy percentage and propellant safety margin increases. However, increases in both these factors add mass and therefore cost.

The STV sheet has sixteen system level input variables: the length, mass, and diameter of an engine, the number of engines at TAI and asteroid orbital insertion, the first four ΔVs, specific impulse, earth departure date, asteroid arrival date, and Earth arrival date, assembly orbit inclination, habitat (Transhab) mass, SRV mass, and SRV length. These inputs are all used to determine the size of the vehicle to be launched, and the mass of the vehicle at each mission point. The STV sheet provides the following output parameters: STV cost, STV safety, mass of propellant not launched with STV, number of Magnum launch vehicles required to launch the vehicle, STV integration and preparation times, and the STV mass at each of the four mission points.

2.2.8 Operations

The Operations discipline encompasses three primary areas: ground manufacturing and integration of the rescue vehicle, launch of the rescue vehicle components into Low Earth Orbit (LEO), and assembly of the vehicle in LEO. The goal of operations is to place an assembled rescue vehicle in LEO as quickly as possible at a minimum cost while not compromising safety.

In order to analyze the three areas, several assumptions are made. First, each worker assigned to ground manufacturing and integration of the rescue vehicle is assumed to cost $110,000 per year/worker. Second, the modification of rescue vehicle components is done in parallel, but the integration must wait for all components to be modified. The orbital assembly is assumed to take place at an altitude of 407 km and at an inclination of 28.5° in a circular orbit. Lastly, seven crewmembers are assumed to be working on the in-space assembly. The crew is delivered to the rescue vehicle's assembly orbit via the Space Shuttle. The following variables have a large effect on the output.

- Modification and integration time rush factors: rush factors increase the man hours put into work on a specific component of the rescue vehicle and hence lower the time required to complete the task. However, lowering a rush factor increases cost and hurts the safety of the operation.
- Launch vehicle selection and rush factors: two launch vehicles are available to the rescue mission, the Magnum heavy lift expendable, and a high flight rate reusable launch vehicle (RLV). The Magnum has a capacity of 80 metric tons, and costs $2,200 per kg. Only six Magnum vehicles are available. The RLV can lift 18 metric tons into the assembly orbit and only costs $1,600 per kg. Rush factors can be placed on the launch preparation time in much the same way as described above. Launch vehicles are chosen based on cost per kg to the assembly orbit with available volume as a constraint.
- In-space assembly time: in-space assembly time is modified much the same way as the modification and ground integration rush factors. Again, safety is compromised and cost increased for a faster than normal assembly.

The operations module takes, as input, the estimated modification and integration times for each of the rescue vehicle sub-systems, the mass of propellant not launched with the STV hardware, the launch inclination, the number of Magnum launch vehicles that are to be used for STV hardware launch, and the target launch date. The target launch date is treated as a limit on the time available for departure preparation.

3 Results

3.1 FPI Solution

The first design convergence was done using fixed-point iteration by manually picking suitable values for all system level variables. This intelligent selection of design variables is the traditional method of converging a design. It was selected as the starting point of this design process due to its simplicity for comparison to designs converged using other methods. It became apparent during this process that the feasible design space of the project was smaller than initially believed due to the exponential growth of mass with I_{sp} at the ΔVs required for this mission.

Table 3-1. FPI Design Results

Design Variables		Objectives		Design Specifications	
Isp	1010 s	Cost	$3.21 Billion	Initial mass	939,221 kg
Thrust	700 kN	Safety	0.795	Dry mass	79,977 kg
Launch Date	14-Sep-21	Survival	0.989	ΔV total	20.8 km/s
Asteroid Arrival Date	14-Jan-23			Re-entry g-load	8.85 g
Earth Return Date	12-Dec-24				

3.2 DOE Solution

The seven design variables with the most influence on the design are specific impulse (Isp), total thrust at trans-asteroid injection (thrust), launch date (LD), asteroid arrival date (AAD), earth return date (ERD), velocity change by the NTR prior to re-entry (ΔV_4), and velocity change by the ERV prior to re-entry (ΔV_5). A baseline value and ranges were set for each of the variables.

Table 3-2. DOE Design Variable Ranges

Variables	Units	-▶ 2	- 1	0	+ 1	+▶ 2
Isp	[s]	960 s	975 s	1010 s	1045 s	1060 s
Thrust	[kN]	441 kN	508 kN	669 kN	830 kN	897 kN
LD		8-Nov-21	31-Oct-21	11-Oct-21	21-Sep-21	12-Sep-21
AAD		31-May-23	13-Sep-23	3-May-23	22-Dec-22	5-Apr-23
ERD		22-Oct-24	13-Oct-24	24-Sep-24	5-Sep-24	26-Aug-24
ΔV4	[m/s]	0 m/s	439 m/s	1500 m/s	2561 m/s	3000 m/s
ΔV5	[m/s]	0 m/s	439 m/s	1500 m/s	2561 m/s	3000 m/s

The CCD array for these seven variables results in 143 design runs, enough data to facilitate quadratic curve fits to the objective functions and constraints. Cost, rescue mission safety (safety), and original crew survival (survival) are the three objective functions considered for the design. Two constraint Response Surface Equations (RSEs) were also created: 1) the number of days after JD1 that operations was ready to launch (late) which must be less then zero, and 2) the deceleration during re-entry (gload) must be less than 10 g.

The RSE generated by the CCD array was found to be highly inaccurate. Although several variables were accurately modeled by the RSE, others (such as cost) showed large discrepancies, indicating that the quadratic assumption was poor. By limiting the range of the variables to a small region of the RSE design space, a few solutions were obtained. The three solutions below were optimized for cost, safety, and survival, respectively.

Table 3-3. DOE Results

Isp	Thrust	JD1	JD2	JD3	Cost	Safety	Survival	Late	gload
[s]	[kN]				$BIL			[day]	[g]
960	454	10-Oct-21	2-Feb-23	16-Oct-24	6.73	0.594	0.999	0	12.1
971	516	18-Oct-21	25-Mar-23	22-Oct-24	12.2	0.796	1.000	0	9.61
982	675	1-Oct-21	10-Mar-23	22-Oct-24	0.516	0.769	1.000	-96	9.99

3.3 CO Solution

In order to solve the CO version of this design, two different system level optimization methods were attempted: sequential quadratic programming (SQP) and the Method of Feasible Directions (MoFD). Both methods are gradient based which allows them to converge to a solution much more efficiently than non-gradient or heuristic optimization methods. However, in order for gradient-based methods to work, accurate gradient information must be gathered. Since many of the discipline level modules had non-smooth, multi-modal design surfaces requiring heuristic optimizers, the gradient information returned to the system level optimizer was often poor. In order to improve the gradient information, large gradient steps were programmed into the system level optimizer and more time was given to the discipline level optimizers. Despite these improvements, the system level optimizer still had difficulty reaching the feasible region of the design space, which necessitated relaxation in the convergence criteria of the target values. An example of the convergence history using the SQP optimizer is shown below, along with the optimum solution it found.

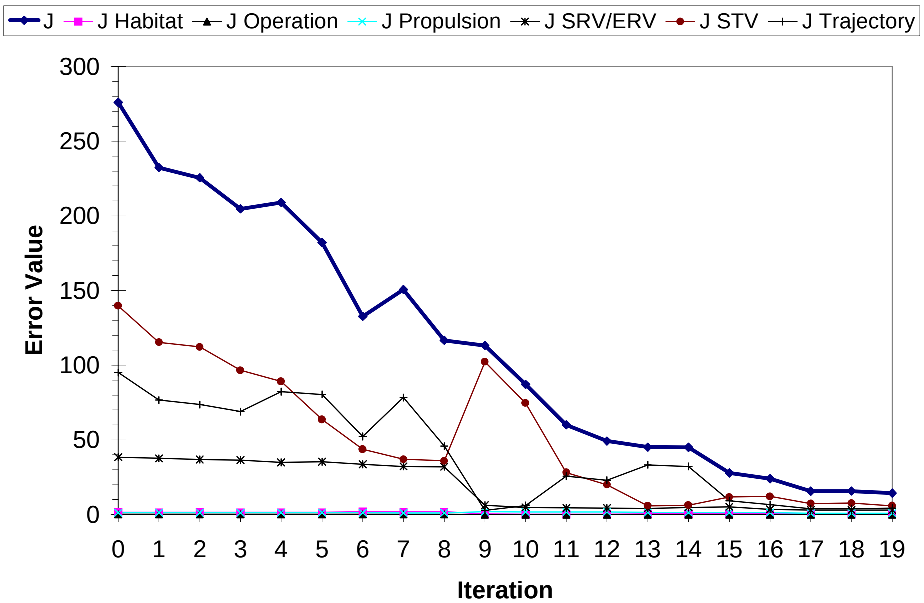

Figure 3-3. Plot of error values (J's) for each discipline and total showing the gradual convergence of target and calculated values

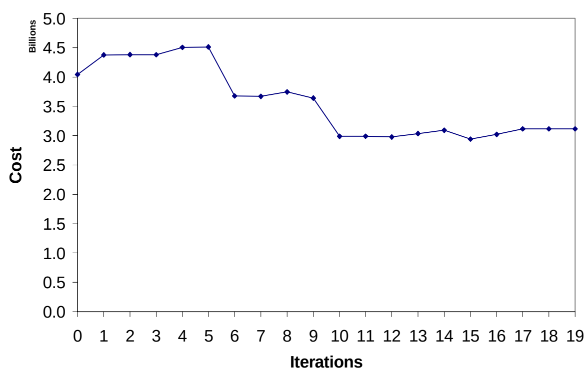

Figure 3-2. Plot of gradual improvement (decrease) in objective function cost

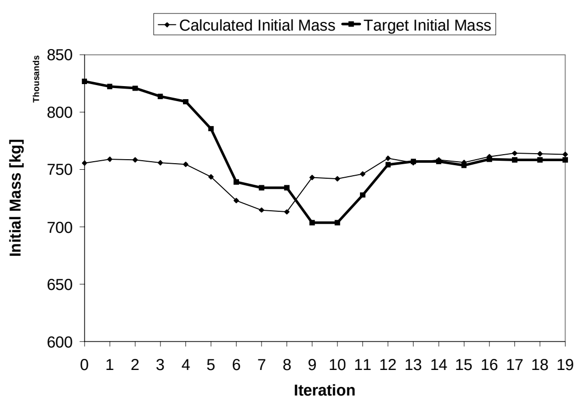

Figure 3-1. Plot of target and calculated initial mass showing gradual convergence of the two values

Table 3-4. CO Design Results

Design Variables		Objectives		Design Specifications	
Isp	1077 s	Cost	$ 3.11 Billion	Initial mass	763,107 kg
Thrust	634 kN	Safety	0.413	Dry mass	88,827 kg
Launch Date	12-Sep-21	Survival	0.993	ΔV total	19.9 km/s
Asteroid Arrival Date	12-Dec-22			Re-entry g-load	< 10 g
Earth Return Date	27-Dec-24				

4 Conclusion

This design has attempted several optimization schemes to solve this design problem. FPI produced the best results with a human controlling the design variables. Theoretically CO should produce a better solution, implying that work needs to be done on the implementation of CO for this problem. A penalty function at the discipline level as suggested in [Braun] may help consistency of gradients for the system level optimizer. The DOE method may benefit from a higher order curve fit, requiring more runs since the design space does not appear to be quadratic. For this problem, the traditional method of a human controlling the design variables and using FPI to converge a solution proved to be the fastest solution and the best results.

5 Outreach

Throughout the course of the Spring 2001 Semester, the Georgia Tech team led several projects aimed at educating the public and building community interest in the space program. For example, two team members presented typical aerospace engineering projects, tasks, and applications to an all-day, Society of Women Engineers-sponsored, program aimed at local high school girls. These two students gave an accurate picture of what aerospace engineering is about. In another project, two more team members discussed the world of aerospace engineering with a local cub scout troop. The GT students answered countless questions about space related issues, even questions about the team's involvement in the upcoming LPI design competition at Johnson Space Center. Finally, in perhaps its largest outreach project , the entire Georgia Tech design team organized the AIAA-sponsored model rocket contest open to all undergraduates – tomorrow's technical future. For more information, these outreach activities are further described on the website:

http://atlas.cad.gatech.edu/~ae6322a/outreach.html

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