Chimera- A Low Cost Solution to Small Satellite Space Access

John Crowley, Virgil Hutchinson, Alex Keisner, Dave Young
Mike Curry, Jonathan Jackson, John Maatsch, Pavel Piletsky
Faculty Advisor: Dr. John Olds
Georgia Institute of Technology
Atlanta, GA 30332

Abstract

The Chimera rocket was designed to enter the small satellite market by offering an affordable and flexible alternative to the Pegasus launch vehicle. A number of design concepts were evaluated, and one was selected to undergo detailed analysis. This included disciplinary analyses in aerodynamics, propulsion, trajectory, aeroheating, structures, weights, operations, and cost. The baseline vehicle, consisting of a Minuteman 2-2 first stage, a PAM-S second stage, and a new third stage carries a 100 and 50 kg payload to a 700 km altitude, at inclinations of 60° and 110° respectively. At this point a Monte Carlo Simulation was performed to determine how well the system met its price goals. The baseline vehicle fails to meet the desired launch price of $5 million to a reasonable confidence level. However, either the implementation of a cost reduction in the cost of the first stage, or the infusion of appropriate structural and propellant technologies in the design of the third stage, help to make the desired launch price viable.

Nomenclature

APAS	Aerodynamic Preliminary Analysis System
DoD	Department of Defense
DDT&E	Design, Development, Testing & Evaluation
HABP	Hypersonic Arbitrary Body Program
NPV	Net Present Value
POST	Program to Optimize Simulated Trajectories
RCS	Reaction Control System
RFP	Request for Proposal
ROSETTA	Reduced Order Simulation for Evaluating Technologies and Transportation Architectures
TFU	Theoretical First Unit
UDP	Unified Distributed Panel

Introduction

The small satellite market has been growing in recent years. Interest ranges from the DoD to universities wishing to launch scientific payloads. Current launch vehicles can provide services for these organizations, but at a high cost. The Pegasus launch vehicle, made by Orbital Sciences, can cost $12M or higher[1].

In many cases, universities cannot afford to pay for a launch if it costs more than twice the cost of the satellite that they built. Therefore they have sent out an RFP to build a new low cost launcher that is particularly suited for their tastes. Launch altitudes and inclinations are based on a survey of all the previous launches made by universities.

The RFP details a business case. The item of interest is the price per launch paid for by the universities. The RFP calls for the launch costs to be $5M. Additionally, the notional start up company must be able to show an internal rate of return of 10%. The company is also granted a loan from the DoD of $500M to cover non-recurring costs. The company may use any US launch facility for a nominal fee of $50,000 per launch.

Two design reference missions were detailed within the RFP. The first was to send a 100 kg payload to a 700 km, 60° orbit. The second mission was to send a 50 kg payload to a 700 km, 110° orbit. The constraints for the payloads were a 6 g axial load and 2 g lateral load, and the payload could not be exposed to a dynamic pressure of greater than 30 Pa.

As a final note, the RFP said that US or foreign parts could be utilized in the construction of the launch vehicle. It was decided early in the project to purchase most parts in order to reduce costs.

Design Methodology

The design team chose a methodology to explore as many concepts as possible while maintaining creativity and technical feasibility. A brainstorming technique, known as a morphological matrix, was used to look at all the possible characteristics of the vehicle. The first matrix was created to look at the subsystem components and all the possible parts that could fulfill them. The second matrix then combined, through a structured selection process, these sub systems into eleven different concepts. Table 1 shows the different types of concepts that were created through the morphological matrix.

Table 1- System Concepts.

Type	Number
Balloon Assist	2
Air Assist	3
Cannon Assist	1
MagLev Assist	1
Ground Launch	4

The design team then evaluated each of these qualitatively using TOPSIS. TOPSIS is an evaluation method used to show how close a design is to the ideal solution. The higher the closeness value of a design, the better it is. The team evaluated these 11 concepts using 19 criteria, each with a weight. These criteria and weights were developed through the use of a QFD.

The QFD maps the customer requirements and engineering characteristics through a relationship matrix. In this matrix each of the requirements and characteristics are rated as to how each affects the other based on a 1-3-9 scale. If they have a strong affect on each other (i.e. cost and weight), they are given a 9. The QFD then multiplies these numbers with the customer requirement importance values and determines a relative importance value. This importance value was used as the weighting, and the engineering characteristics were used as the criterion.

The TOPSIS analysis determined the ranking of each of these concepts. The top two designs were a 4-stage ground launch vehicle and a 3-stage air assist launch.

These vehicles were then designed and sized using the appropriate disciplines as seen in Figure 1. The data obtained from this analysis was then fed back into TOPSIS so the final two designs could be evaluated quantitatively. The initial cost estimate of the air assist launch was found to be approximately $9.5M and the ground launch cost estimate was approximately $10.9M. TOPSIS ranked the air assist launch as the best vehicle to use. This was confirmed by the launch price and an evaluation by the team of the designs. The air assist launch was chosen to be designed at a higher level of fidelity. The following sections outline the results of the disciplinary analyses. These analyses were performed as the following Design Structure Matrix indicates.

A probabilistic study was then performed on the design with the help of a ROSETTA model and a Monte Carlo simulation. This is detailed in the final section of the report.

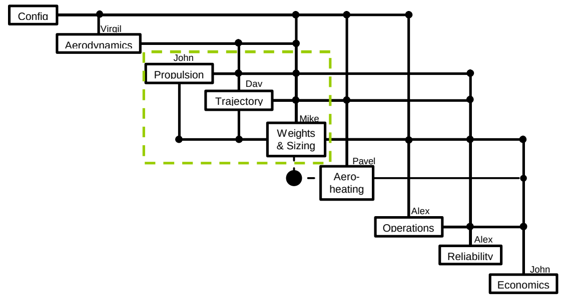

Figure 1. Design Structure Matrix.

Disciplinary Analyses

Aerodynamics

The stability and modicum of lift required during the atmospheric flight of the vehicle was provided by four fins attached to the first stage of the vehicle. Due to the rocket's release from the air-assist vehicle at a high flight path angle, it was assumed that the use of a wing to provide lift for pull-up was not required. Each of the four fins is 1.38 meters in length, has an area of 0.679 m^2, and is arranged at 35 degrees from the center neutral axis. This asymmetric arrangement of the fins allowed for a small increment of lift to be provided to the rocket along with providing stability through the first stage of flight.

The Aerodynamic Preliminary Analysis System (APAS), a combination of three individual programs, was utilized in performing the aerodynamic analysis of the vehicle. APAS was used to define the vehicle geometry at each stage of flight and initialize the analysis runs, which were based on altitude, velocity, and angle of attack. After defining the geometry and analyses, the Unified Distributed Panel (UDP) program performed the subsonic and supersonic aerodynamic analysis and the Hypersonic Arbitrary Body Program (HABP) was used to conduct the hypersonic analysis. UDP's analysis is based on slender body theory and source and vortex panel methods while HABP's analysis is based on impact theory. Aerodynamic analysis was only performed on the vehicle configuration from launch to payload fairing separation because all aerodynamic coefficients were constant above an altitude of 100 km[3].

The resulting data from the aerodynamic analyses showed that during subsonic and sonic flight, the fins have a lift coefficient of 0.175, which indicates their provision of a small increment of lift to aid in the pull-up of the rocket. The variation of zero-lift drag coefficient with Mach number for the aerodynamic performance of the vehicle provided by APAS is shown in Figure 2.

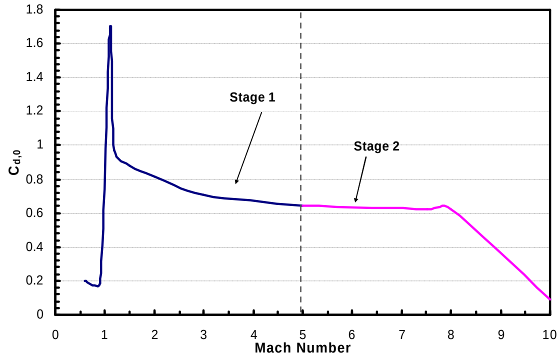

Figure 2. Variation of zero-lift drag coefficient with Mach number.

Propulsion

Even so, buying pre-existing stages and developing as little as possible is the surest way to reduce costs and uncertainty, and thus increase the chances of reaching the price goals set forth in the RFP. Therefore, the first two stages were set as existing solid rocket motor stages, and only the third stage was designed. Because all analyses indicated that the thrust and burn time necessary for the third stage were small, it would cost relatively little to design.

Cost remains the overriding factor in the design process, and the motor was designed accordingly. The third stage motor has a simple spherical casing made of affordable and readily available steel. For ease of manufacturing, the nozzle was set as a 15 degree half cone with carbon/carbon construction. As will be seen, this resulted in a relatively long nozzle; however, other aspects of design, such as the payload fairing design, were not significantly affected by this decision.

The mass of propellant for the third stage was used to size the spherical casing. The thrust and mass flow rate required determined the burn time, and combined with the burning rate the propellant thickness was calculated. To keep the thrust fairly constant, the casing was sized double the propellant thickness to leave a lot of empty space in the middle of the motor. The motor characteristics are summarized in Table 2.

Performance

The trajectory for Chimera was sized using the design reference missions. To model the trajectory of Chimera from air drop to orbit, POST was used. POST, the Program to Optimize Simulated Trajectories, is a three degree of freedom code written by Lockheed Martin and NASA[3]. As noted above, both the first two stages as well as the air launch aircraft will be existing flight hardware to limit costs.

Table 2 – Specification of Third Stage Motor.

Thrust	2669 N
Isp	295 sec
Burn Time	33.4 sec
Exit Area	0.152 m2
Expansion Ratio	50
Propellant Mass	30.8 kg
Motor Mass	6.63 kg
Motor Volume	0.0195 m3

A list of nine possible aircraft was compiled with the flight envelope of each aircraft. From this list three separate aircraft were chosen that seemed to cover the entire flight regime that is to be investigated. These aircraft are compiled in Table 3. From these three and an initial investigation of trajectories in POST it was determined that the B-52 resulted in an appropriately sized rocket (to fit beneath the aircraft). To choose the stages a comprehensive list of available US solid rocket engines was compiled and acceptable combinations (size and cost compatible) were run in POST.

Table 3- Aircraft Summary.

	Payload (kg)	Ceiling (m)	Velocity (m/s)
B-52	19320	15150	290
F-15E	11136	20000	840
SR-71	22250	26000	900

The final weight was set as a constraint so that each converged rocket would always meet the required payload mass. POST was also allowed to choose the duration of the ballistic coast between the 2nd and 3rd stages. Minimizing the size of the third stage minimizes the overall cost of the deigned rocket and therefore was the evaluation criteria for each design. From this analysis it was found that the Minuteman 2-2 1st stage with a PAM-S 2nd stage results in a very small third stage and therefore the cost of that stage would be minimized. Figure 3 shows the Chimera and Table 4 summarizes the Chimera's characteristics. The result of the trajectory profile is given in Figure 4. The figure depicts the different stages of the rockets trajectory.

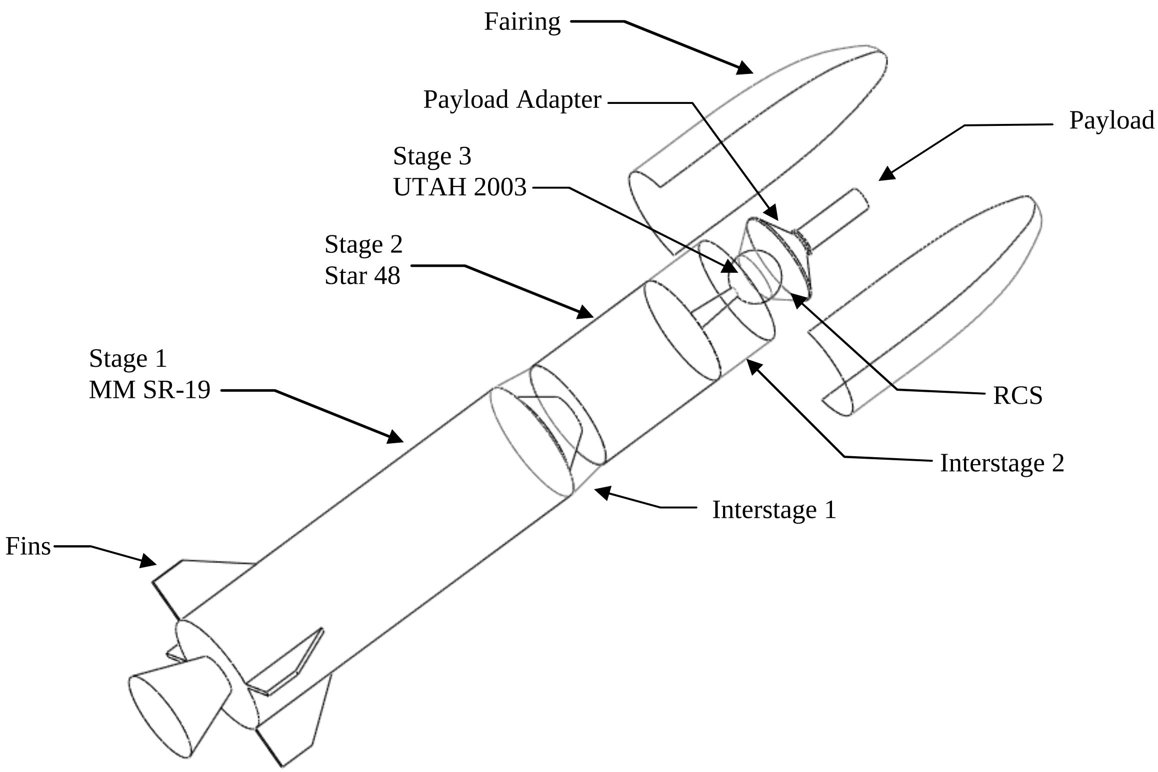

Figure 3- Vehicle Breakdown on Chimera.

Table 4- Performance Summary of Chimera.

	Stage 1 Minuteman 2-2 (MM SR-19)	Stage 2 PAM-S (Star 48)	Stage 3 UTAH 2003
Dimensions			
Length	4.12 m	2.00 m	1.64 m
Diameter	1.33 m	1.22 m	0.54 m
Mass			
Propellant Mass	6237 kg	1962 kg	30.86 kg
Gross Mass	7032 kg	2182 kg	37.50 kg
Structure			
Type	N/A	monocoque	N/A
Case Material	6Al-4V titanium	titanium	steel
Propulsion			
Propellant	ANB-3066	HTPB	HTPB
Average Thrust	268 kN	66.7 kN	2.67 kN
I_{sp}	287.5 sec (vac)	288 sec (vac)	295 sec (vac)
Chamber Pressure	N/A	39.7 bar	37.9 bar
Nozzle Expansion Ratio	N/A	54.8:1	50:1
Staging			
Nominal Burn Time	65.54 sec	87.1 sec	33.4 sec
Shutdown Process	burn to depletion	burn to depletion	burn to depletion
Staging Separation	spring ejection	spring ejection	spring ejection

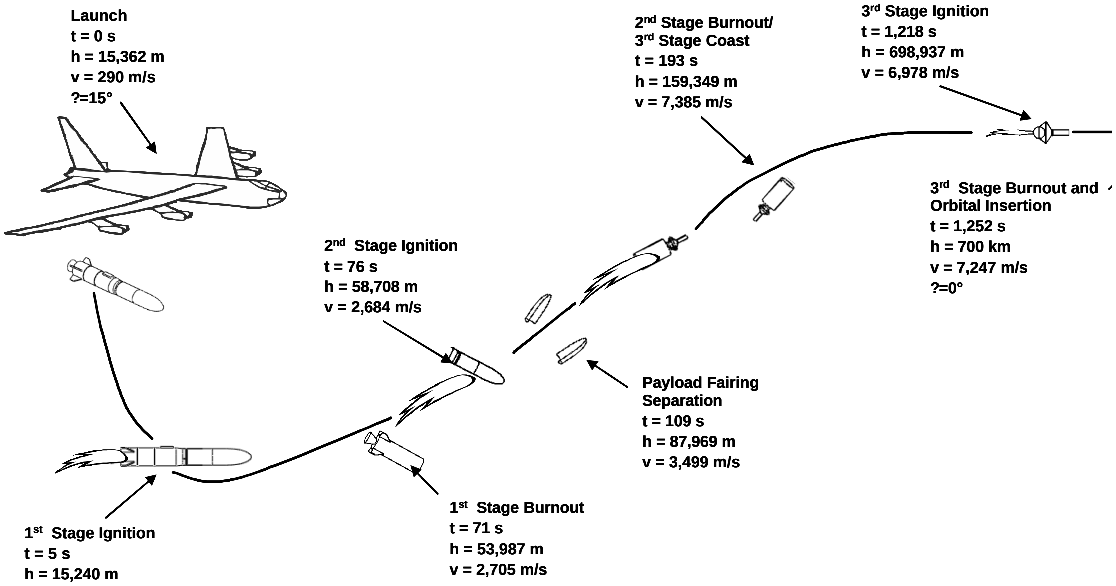

Figure 4- Mission Profile.

For economic reasons it does not make sense to build two different rockets for each of the DRMs, since both comprise similar missions. Therefore it is necessary to determine which mission requires the largest rocket and design Chimera for that mission. After both DRMs were simulated in POST the 100 kg to a 60 degree inclination resulted in the largest rocket, therefore it would be used as the reference mission. A second POST deck was then created to run the Chimera 60 degree inclination design to the 110 degree inclination DRM. Therefore the same rocket will fly a trajectory that gives the most payload weight. A summary of the results is included as Table 5.

Table5- Performance Summary of Chimera

Desired Inclination	60^o	110^o
Gross Mass (kg)	9507	9457
Payload (kg)	100	50.19

Figures 5 and 6 show graphs of the rockets' velocity and altitude as a function of time. To achieve the ideal trajectory to the 110 degree inclination the drop was conducted at a latitude of 70 degrees, while the 60 degree rocket was dropped at a latitude of 55 degrees. These different launch latitudes account for the different inertial velocities of the drops. The coast between the second and third stage is a very long ballistic trajectory where Chimera is trading velocity for altitude. This continues for almost a thousand seconds until the altitude is almost to the correct orbit. The third stage then fires to achieve the proper velocity to maintain a circular orbit.

The velocity plot for Chimera is somewhat deceptive. It seems that the 110 degree inclination trajectory has a greater velocity than the 60 degree trajectory at launch. The actual airspeed of the B-52 is the same, but the latitude of the drop is different.

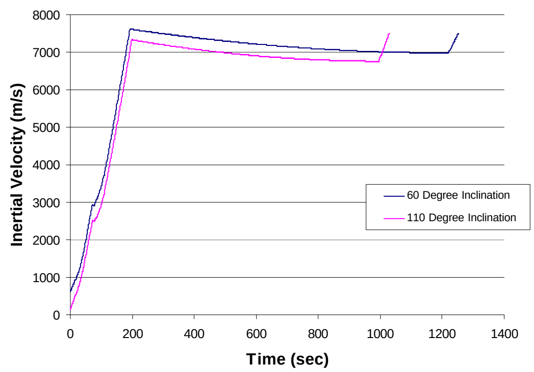

Figure 5. Inertial Velocity vs. Time for Chimera Trajectory.

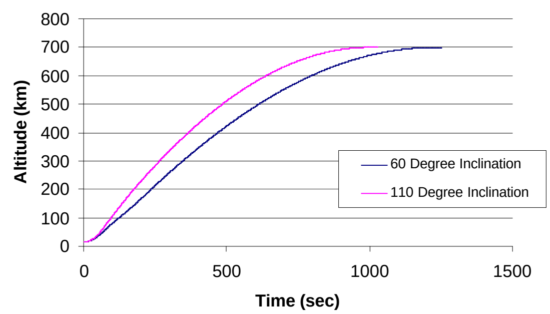

Figure 6. Altitude vs. Time for Chimera.

Interstage Design

The key requirement in the design of the interstages is that they must be able to withstand structural loads of up to 5 g's which represents the maximum theoretical acceleration of the vehicle. There are also several secondary requirements that were considered in the design process. Since the overall goal was to minimize the cost of the vehicle it was important that the interstages be lightweight. Any unnecessary weight would also take away from potential payload carrying capacity. It was also desirable to keep the design simple. Design simplicity results in reduced cost.

In the beginning of the design process three geometrical concepts were considered for the interstages. Those concepts were the straight wall cylinder, the I-beam reinforced cylinder and the corrugated cylinder. Several materials were also considered, namely, aluminum, titanium, graphite epoxy, and steel. To evaluate all the possible combinations of shapes and materials more rapidly, finite element analysis (FEA) was used to determine their structural rigidity.

By analyzing the stress in the parts it was possible to determine how thick the walls of each design would have to be in order to withstand the applied loading. This was used to determine the overall weight of each interstage concept. An example of a Von Mises stress contour plot for one of the concepts is shown in Figure 7 below.

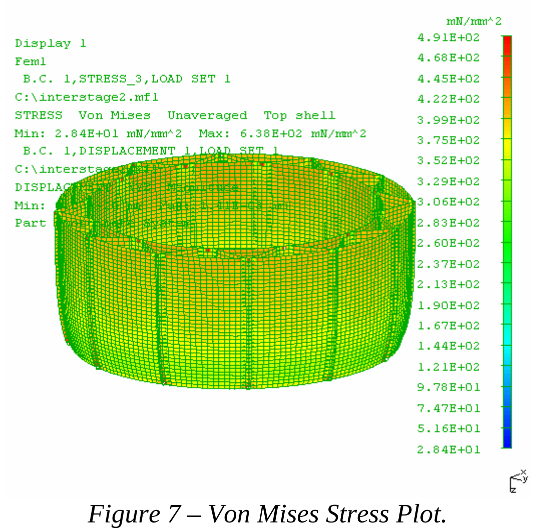

Figure 7 – Von Mises Stress Plot.

The final design was selected as the corrugated cylinder constructed with graphite/epoxy. This design maximized strength while minimizing weight. The total weight of the two interstages was 140 kg. The analysis also shows that the design will not fail at loads below 6 g's. It should also be noted that the structure might withstand much higher load because the points of high stress in this interstage design occur at very localized points. It is possible that under higher loads while there would be localized permanent deformation of the part but would not result in catastrophic failure.

Payload Adapter

The structure to mount the payload to the RCS atop the third stage booster is an aluminum monocoque conical shell. For the 60 degree inclination payload configuration, the adapter has a lower diameter of 1.00 m (so that it fits within the payload fairing), an upper diameter of 0.25 m, and a height of 0.25 m. The upper diameter is modified to 0.50 m for the 110 degree inclination payload configuration. The conical form of this structure is designed to withstand the high axial and lateral loads during the boost phase. Utilizing the properties of composites with this type of structure allows for a high-strength, weight efficient adapter design[4]. The thickness of the aluminum for the shell is calculated from a spreadsheet based upon the input of the payload mass, payload configuration, and the loads experienced during the boost phase.

The payload attaches to the separation plane atop the adapter with a Marmon clamp. The separation joint within the Marmon clamp is a continuous ring held together by an annular clamp[4]. The release of clamp tension allows the joint to separate, and springs then convey a small increment of velocity onto the payload. After the payload separates, the booster maneuvers to prevent accidental collision[4].

Payload Fairing

The payload fairing is made out of alternating layers of graphite/epoxy and aluminum. The material was selected by considering cost, weight, strength, and thermal properties. The materials examined were carbon composite (CC), aluminum and graphite/epoxy. Even though the CC fairing has better thermal properties and would not need any thermal protection system (TPS), the costs for design and production of CC fairings are very high relative to the other options considered.

The conventional aluminum structure would require a lot of TPS to withstand the thermal loads on the fairing and weight around three times more than CC or graphite/epoxy. The graphite/epoxy fairings cost as much as conventional aluminum ones and possesses better weight and strength properties.

The weight of the fairing was estimated by using payload fairing area to weight fraction established from the trade study done on current payload fairings for different vehicles made out of CC or graphite/epoxy. The mass of the fairing is estimated to be 28.35 kg. The diameter is 1.2 m with a height of 3 m.

Aeroheating Analysis and TPS

The aero and thermodynamic effects on the nose, fins leading edges and rocket body due to the vehicle's flight through the atmosphere were calculated using the MINIVER engineering methods aeroheating code[5]. This code is based on impact theory and Reynolds analogy, extrapolated skin friction point-to-point correlation.

The final Chimera launch vehicle design was analyzed using trajectories for the 60 and 110 degree orbits and launches from three different aircrafts: B-52, SR-71 and F-15. The results for all three airplanes showed that the max heat rates, pressure loads and temperatures occurred at altitudes of around 51.8 km and Mach numbers around 8. The peak temperatures at the nose ranged from 1144 to 1311 K; therefore the aeroheating scenario did not play a major role in the airplane selection. The results from MINIVER were verified with the heating rates calculated in POST and using Chapman's equations for redundancy.

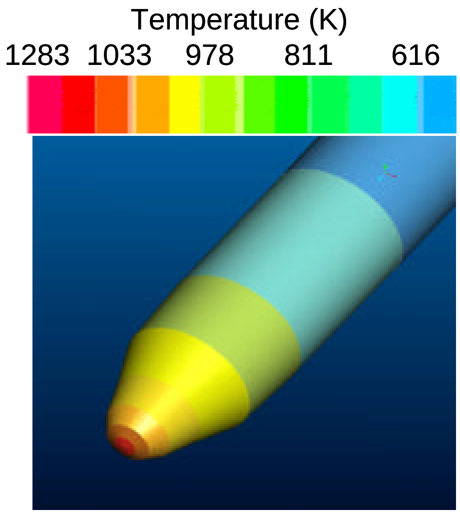

Figure 8 – Temperature Contour.

Figure 8 shows the distribution of peak temperatures around the fairing for a worst case scenario of 60 deg orbit launch from B-52 (final choice).

The distributions of pressure and heating on the vehicle were evaluated to establish TPS design guidelines. The 1st and 2nd stages of the Chimera did not necessitate any TPS application due to the titanium casings which retain their structural integrity at a temperature of 625 K. The nosetip and leading edges of the fins are subject to much higher magnitudes of the stagnation pressures and heat fluxes and must be thermally insulated from the frame.

Based on the aeroheating analysis, candidate TPS materials were selected to accommodate the maximum radiation equilibrium temperatures. Three concepts for the TPS were evaluated during the design process. These systems included three mature, present-day materials. The driving requirements for the TPS selection were the cost of the system and the weight limitations. From the trade study done on the three materials the conclusion was made that Lockheed Martin-produced Ma-25 sprayable ablator can fulfill the needs for the lowest cost. The total TPS weight for the 60 degree orbit launched from the B-52 is 57 kg.

Weights

Once the different contributing structures of Chimera were designed they were compiled into a complete weights sheet. This sheet was used to integrate the weights for the interstage FEA, the third stage propulsion design, the payload adapter, as well as the TPS and fairing analysis. The sheet also approximates such systems as avionics, subsystems, and propellant losses using Mass Estimating Relationships (MERs) for expendable launch vehicles.

These MERs are curve fits of existing subsystem weights that can be used to approximate the weights of the Chimera systems. A spin stabilization system is also approximated in the weights sheet. This system is a Nitrogen cold gas RCS system which is used to spin up the payload as well as the payload adapter for stability. This was approximated using historical data. A ten percent margin was also included into the design mass for the third stage. This is to accommodate any growth in the rocket due to unforeseen errors in the approximations or changes in the mission.

Operations

In order to analyze the operations costs associated with the ground and flight operations of Chimera, OCM-COMET was used[6]. OCM-COMET uses a series of user prompts to input the characteristics of a launch vehicle. Using the user inputs and historical data, labor crew sizes (or head counts) dependent upon flight rates were calculated and are outputted from the COMET model. The outputs from COMET, along with more user inputs go into OCM to result in final ground and flight operations costs, which are also dependent on flight rate.

After obtaining the characteristics of Chimera from preceding analyses, the many user prompts in COMET (e.g. General Information, Upper Stage(s) Description, Cross Training Effects, etc.) were completed, and head counts of 26 people for ground operations and 19 people for flight operations were found (assuming a baseline flight rate of 6 flights per year). Once the head counts were gained from COMET, they were inputted in the OCM model along with a standard encumbered salary per employee of $150,000 per year, and percentage factors for supplies and materials (10% for ground operations, 5% for flight operations). With the historically-based complexity factors for flight operations (resulting in a cost adjusted head count) built into OCM, this resulted in an operations cost of $1.45 M per flight, or $8.7 M per year was found at a rate of 6 flights per year.

However, since Chimera is an air-assisted launch vehicle, aircraft operations costs had to be added to the outputs of OCM-COMET to come up with a total operations cost for the launch vehicle. The cost of operations of a B-52 was estimated at $25,000 per flight for use in the launch of Chimera[7]. A breakdown of all of the operations costs can be seen in Table 6. A reliability study was also performed and it was found that the overall vehicle reliability was 96.0%.

*Table 6 – Operations Costs Breakdown.**

Flight Rate	4	6	8	12
Ground Ops	24	26	30	32
Flight Ops HC	19	19	20	21
Total HC	43	45	50	53
Ground Ops Supplies	$0.36	$0.39	$0.45	$0.48
Ground Ops Labor	$3.60	$3.90	$4.50	$4.80
Ground Ops Total	$3.96	$4.29	$4.95	$5.28
FlightOpsSupplies	$0.21	$0.21	$0.23	$0.24
Flight Ops Labor	$4.20	$4.20	$4.65	$4.80
Flight Ops Total	$4.41	$4.41	$4.88	$5.04
Aircraft Ops Cost	$0.10	$0.15	$0.20	$0.30
Total Annual Cost	$8.47	$8.85	$10.03	$10.62

* All dollars in US M$ 2003

Cost Estimation

Cost estimation of the manufactured parts was accomplished through the use of NAFCOM 99. NAFCOM 99 uses a historical database to estimate costs based on weight. Particularly, to calculate the cost of a part, a specific analogy to historically similar items was created. As an example, to calculate the costs of the interstages, the Saturn II and the Saturn IV-B interstages were chosen as data points. The program then creates a curve fit for the data points selected. The weight of the item is then entered as a parameter.

The cost data returned by NAFCOM includes several sub-costs. These contain manufacturing, materials acquisition, labor, overhead, and system integration costs. Additionally, items can have a DDT&E and TFU complexity factor applied, system test hardware can be added, and a learning curve rate can be applied.

The baseline costs for the parts had a TFU complexity factor of 1, and a DDT&E factor of 0.7. Table 7 shows the costs of the four items being manufactured. The payload adaptor was included as part of the last stage. The total system integration costs given by NAFCOM for all of these was $3 M 2003.

Table 7 – DDT&E and TFU Costs.

	Weight (kg)	DDT&E ($M)	TFU ($M)
Last Stage	57	29.49	3.25
Interstage 1	100	3.79	0.62
Interstage 2	40	2.29	0.33
Fairing	109	10.66	0.67

The costs of the PAM-S and Minuteman II-2 stages were estimated by using NAFCOM to determine a TFU cost. A learning curve rate was then applied and the cost after several thousand units was determined. The total came out to $1.25 M for the Minuteman II-2 and $1 M for the PAM-S.

Production Schedule

The customers of Chimera desire availability, which means a readiness to be launched within thirty days of notification or intent to launch. This requires some form of Just-In-Time service. To accommodate this, production would begin 2 years in advance, and the amount of production would depend on the estimated demand. A large finished goods inventory would be maintained and as items are needed they would be pulled from storage.

The flight rates that were explored coincided with the flight rates looked at by operations, that is from 4 to 12 flights per year. The design team felt that any fewer flights and the required price per flight would not be met, and any more than 12 flt./yr. would be more than maximum market demand.

The production/buying schedule was set up to handle the maximum number of units that would be manufactured or bought. That is up to a total of 360 units per part. A learning curve was applied to the production schedule at a rate of 80% as baseline.

The total cost per year was tabulated as a function of both flight rate and program year. This table was then summed up to create a total life cycle cost of production and procurement as a function of total program years and flight rate.

Business Case Analysis

The business case analysis was created as an Excel worksheet to solve the price per flight based on a required NPV. The total cost of procurement and production was taken from the production schedule and present worth factor was applied to place it in 2003 dollars. The present worth factor used a discount rate of 10% as that was what was called for in the RFP.

DDT&E costs were summed up also, the total coming out to $46.23 M. This was considered part of the non-recurring costs. Additionally, the cost of constructing facilities was included in this pricing structure. It was determined that the total non-recurring costs were consistently significantly lower than the $500 M given by the DoD. This allowed the team to ignore non-recurring costs within the data model as it is all paid for by the grant.

The operations cost was passed to the business case as a total cost per year dependent on flight rate. Since this cost is the same every year for the duration of the program, a uniform series present worth factor, as seen in Eq. 1, was applied that took into account program duration and discount rate to obtain the NPV for operations. These three items were summed up to obtain the NPV for costs.

$$T_{U,P,i,n} = \frac{[(1+i)^n - 1]}{i(1+i)^n} \qquad \text{Eq. 1}$$

The revenue per year was determined by multiplying together the cost per flight and the flights per year. A uniform series present worth factor was applied to this as well. The NPV was calculated by adding together the costs and the revenue present values.

The business case analysis was designed to determine the cost per flight in order to reach a particular NPV (required NPV). The baseline required NPV set by the design team was zero or break even. Excel's SOLVER method was utilized to optimize the price per flight until the difference between the real NPV and required NPV was zero.

A full factorial analysis of total program years and flights per year was conducted to see how the price reacted. The TFU complexity factor was set to 100%, the learning curve rate was set at 80%. Figure 9 shows the results of that study.

This study shows several different trends and facts. The first of course is that the highest price, associated with the lowest flight rate and fewest number of program years, is just over $13M. The lowest cost per flight, at maximum flight rate and program years, was $7.5M. This is $2.5M above the desired price.

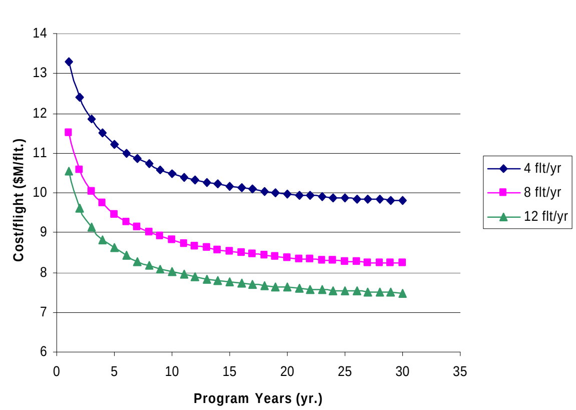

Figure 9. Business Case Analysis.

There are two trends of importance; these are the marginal cost improvements for both flight rate and program years. Marginal cost is simply the change in cost for an additional "unit" where unit here means flight rate or program year. After 15 years, the marginal cost improvement levels out, and not much price improvement is obtained, therefore trade offs can be made as to how long the company should continue. The same trend can be seen in flight rate as the delta in price between 8 and 12 flights per year is only half as much as between 4 and 8.

Probabilistic Design

Once the design reference missions have been fully designed and optimized it is necessary to probabilistically assess the viability of the Chimera rocket. To do this a design code had to be created that could be manipulated quickly to assess the effect of different noise variables and changing technologies on the performance and economics of the Chimera. To do this a Reduced-Order Simulation for Evaluating Technologies and Transportation Architectures model (ROSETTA model) was created. This ROSETTA model is a compilation of the different disciplinary analyses using metamodels of each analysis to achieve a fast approximation of the design disciplines using a reasonably available code (Microsoft Excel).

Two of the most difficult analyses to create a metamodel for were the high fidelity legacy codes such as POST and NAFCOM. To create a metamodel of POST a design of experiments (DOE) was conducted to analyze the effects of changing the thrust and altitude of launch on the design. These results were then fit into a Response Surface

Equation (RSE) that can be manipulated quickly. The RSE fit the data to at least an R^2 value of 0.9999. With this curve fit of altitude of launch and thrust, the modified rocket equation was used to manipulate the launch velocity (of the three different aircraft) and the Isp of the third stage. This analysis produces a mass ratio necessary for the third stage. This necessary mass ratio is then passed to the weights sheet.

The weights sheet is very similar to the compilation weights sheet used in the point design, where each sub-discipline of weights is calculated and then compiled on to one sheet. The main difference is that this sheet will manipulate the payload until the mass ratio calculated for the third stage is equal to the mass ratio required by the trajectory. When this is complete the performance aspects of the design are closed and an economic analysis is conducted based upon the operations, costing, and economics sheets created for the point design. A screen shot of the Input & Output page of the ROSETTA model is included as Figure 10.

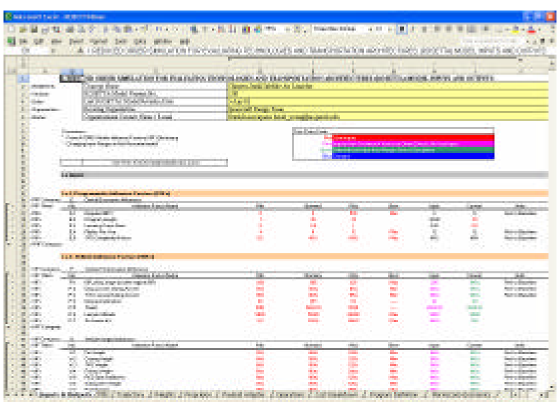

Figure 10: ROSETTA model.

Monte Carlo Simulation

Once the ROSETTA model was created, the capability to rapidly and parametrically explore the deterministic design space was available. In order to determine confidence in the results gained from the ROSETTA model, a Monte Carlo Simulation (MCS) was performed. Monte Carlo Simulation is a method for predicting the uncertainty in an output, given a set of inputs. Normally, MCS is very inefficient due to the large number of cases which must be run to produce an accurate result. The ROSETTA model is the ideal tool for conducting multiple cases in a relatively short period of time. In this case, the program Crystal Ball was used to facilitate the MCS[8]. Crystal Ball is a Microsoft Excel-based macro that provides MCS functionality. All simulations were performed on Intel Pentium 4 processor-based PCs.

The Monte Carlo Simulation process is described in Figure 11. Monte Carlo Simulation involves setting ranges over a variable, which can then be randomly varied over that range. Each variable is given a range and a type of probability distribution for the variable to be based upon. For the purposes of this analysis, only uniform and triangular distributions were used. By varying the appropriate number of variables and recording the resulting outputs from the ROSETTA model, a Probability Density Function (PDF) was created. This distribution describes the history of all cases run by the MCS. A Cumulative Distribution Function (CDF) can then be calculated from the PDF that clearly illustrates the confidence that exists for each output recorded from the ROSETTA model.

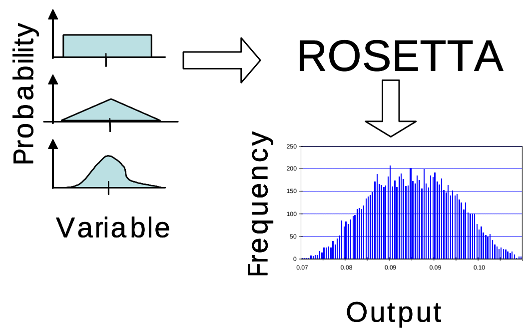

Figure 11. Mote Carlo Simulation Process.

The first simulation performed was to analyze the feasibility of the Chimera rocket. Although it was already determined that the Chimera is indeed able to carry the payloads to the proper orbits as specified in the DRMs, it is valuable to determine just how much of the design space can fulfill those requirements. Ranges were set on a number of design variables, all of which are control variables. In a feasibility investigation, all of the variables are control variables under the designer's discretion; therefore, a uniform distribution was used to model the behavior of each variable. After letting Crystal Ball run 10,000 cases (although fewer cases can be run, a large number is necessary to improve accuracy), it was determined that the Chimera's design space was over 90% feasible. This result is not surprising as the first two stages, as already explained, basically provide enough energy to reach the desired orbit.

The next analysis, and of much more interest, was the economic viability analysis. Because pricing considerations are of such concern in this project, the results of this analysis were key in determining whether or not the system could operate at the desired price per launch. A new set of inputs were varied; however, these variables were considered noise variables that a designer cannot exercise any control over. Such variables include the number of flights per year, the program length, and others. For these variables, a triangular distribution was applied to the run ranges. A summary of the ranges and the peak of the triangular distribution for all of the variables can be seen in Table 8.

Table 8-: ROSETTA Design Variables.

	Min	Peak	Max
Flights per Year	4	6	12
Program Length (yrs)	5	15	30
Learning Curve Rate	0.4	0.8	1
TFU Complexity Factor	0	0.45	1
Isp (secs)	280	295/318*	320
1st Stage Cost Factor	0	1/0.1**	1.3
*No technology/with technology			
**With or without Minuteman cost reductions			

The limiting case for the viability analysis was for the 60 degree, 100 kg payload rocket configuration, as the greater payload weight requires a more massive support structure and thus is more expensive to build. For the initial MCS, a required NPV of 0 was set, and all vehicle performance variables were set to their baseline values. The CDF produced by this case can be seen in Figure 12. This figure shows that there is only about a 34% chance that the goal of $5 million for the launch price is attainable if the Chimera rocket is going to break even. Generally, the desired confidence should be at 90% or above. The 90% confidence value for price per launch for the baseline case is $6.9 million dollars. Unfortunately, at this point it does not seem very likely that the Chimera will break even if the price per launch is $5 million.

Fortunately, there are a number of ways this result can be improved. In the baseline case the economic model assumed that the full price was being paid for the Minuteman 2-2 stage. Because the U.S. government is seeking to find alternative uses for Minuteman missiles and find ways to use them as regular rockets rather than ICBMs, it is not outrageous to assume that these rocket stages could be appropriated for a drastically reduced price. Therefore, the next MCS to be conducted evaluated the scenario in which the Minuteman stage was obtained at a 90% cost reduction. This MCS was run for the same variables, ranges, and distributions as before. The results, featured in the CDF of Figure 13, show that there is now about 90% confidence in reaching the $5 million launch price without losing money. Given that 90% is the desired confidence, it now seems very likely that a price of $5 million will be sufficient for the business case.

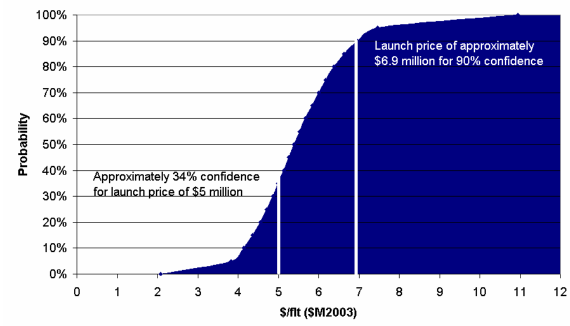

Figure 12. CDF for Price per Flight (Baseline Case).

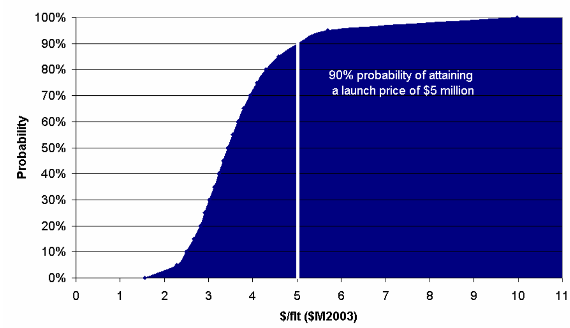

Figure 13. CDF for Price per Flight (Minuteman reduction applied).

However, it would be a mistake to rely on an unknown price reduction to make the case for this launch system. Therefore, another avenue was explored to increase the chances of attaining the desired launch price.

One hundred twenty-five structural materials were studied for use in the fairing, third stage, and interstages. All are commercially available, though many of them are not typically used in aerospace applications. For convenience, the materials have been grouped into ten classes, shown below.

The mechanical properties of each material were evaluated to determine the relative masses of each necessary to serve the same purpose. The 21 materials requiring the least relative mass, along with their costs per unit mass and maximum operating temperatures, were passed along for Monte Carlo analysis.

There are several different varieties of solid rocket propellants currently being studied that have performance characteristics superior to conventional aluminum/ammonium perchlorate propellants, including specific impulses as high as 318 sec. Advanced fuels include boron and advanced oxidizers include hydrazinium nitroformate and ammonium dinitramide. None of the advanced propellants have costs of less than $220/kg, however, and none of them are currently available in quantities of more than 100 kg/year. The need for further development was taken into account by increasing the complexity factors and therefore the cost of implementation of these technologies.

Table 9-: Technology Factors.

Types of Materials	Number
High Strength Metal Alloys	12
Discontinuous Reinforced Aluminum	8
Other Metal Matrix Composites	9
Boron Fiber Composites	2
Aramid Fiber Composites	8
Graphite Fiber Composites	53
Silicon Carbide Fiber Composites	13
Alumina Fiber Composites	13
High Performance Polyethylene	4
Other Advanced Materials	3

Ultimately a number of materials and a single propellant technology were selected to be evaluated. Each technology affected various system weights of the upper stage, and a technology cost factor which simulated the performance gains and cost penalties for implementing the technologies, respectively. The effect of the best technology combination is represented here, in Figure 14.

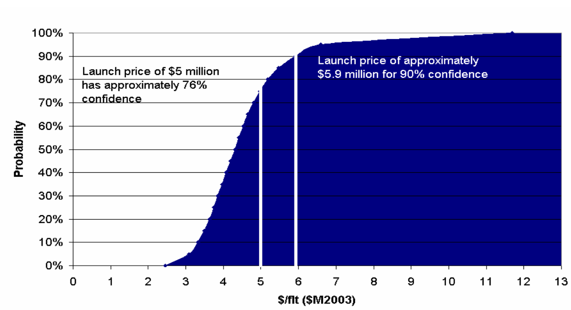

Figure 14. CDF for Price per Flight (Technology reductions applied).

This combination was Amoco T650-35 graphite composite with a phthlonitrile matrix, and utilizing advanced propellants. Although the introduction of technologies does not improve the system as much as the Minuteman cost reduction, there is now 75% confidence in attaining the desired launch price and 90% confidence if the launch price is set at $5.9 million. At this point the business case for the Chimera rocket is looking much improved.

Conclusion

Through the extensive analysis conducted on the Chimera launch system, it was determined that the baseline rocket design was fully capable of meeting the requirements set forth in the RFP. The Chimera rocket is a three stage, air-launched rocket consisting of a Minuteman 2-2 first stage, PAM-S second stage, and a custom designed UTAH2003 third stage. It is an 11 m , 9500 kg rocket with the capability to send a 100 or 50 kg payload to a 700 km altitude orbit, at a 60° or 110° inclination, respectively. Unfortunately, the Monte Carlo Simulation reveals that the baseline rocket does not have a high probability of meeting the desired launch price of $5 M per launch. However, if the Minuteman 2-2 first stage can be purchased from the government at a highly reduced price, there is 90% confidence that the desired price can be achieved. Even without this reduction, the infusion of structural and propellant technologies can increase confidence in the viability of a $5 M launch price to 75%. The Chimera can therefore be considered a worthwhile entrant into the small satellite launch market.

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