Responsive Access Small Cargo Affordable Launch (RASCAL) Independent Performance Evaluation

David A. Young* and John R. Olds†

Space Systems Design Lab
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA, 30332-0150
david_young@ae.gatech.edu

RASCAL is a United States Defense Department initiative that stands for Responsive Access, Small Cargo, Affordable Launch. The overall launch concept involves three stages. The first stage will consist of a reusable aircraft similar to a large scale Air Force fighter. The first stage will also utilize Mass Injection Pre-Compressor Cooling (MIPCC) turbojet engines that will propel the stage to approximately two hundred thousand feet before releasing the second and third rocket stages. The first stage will be similar to a large fighter of the F-22 class, although the turbofans will be that of the more available F100 class. The MIPCC system will be a plug-in addition to the engines to help high altitude performance. This stage will act as a first stage in the RASCAL architecture and will contribute significantly to the overall acceleration of the vehicle. The second and third stages of the RASCAL concept consist of expendable rockets. Releasing the upper stages outside the atmosphere will reduce the loads on the stages as well as the risk of staging. Also by relying on the reusable portion for all atmospheric flight, the expendable stages can be designed simpler and therefore cheaper.

The purpose of this project is to compare the published RASCAL numbers with those computed using a design methodology currently used in the Space System Design Laboratory (SSDL) at The Georgia Institute of Technology. When the initial Space Launch Corporation design was evaluated using the SSDL methodology it was found to fall short of the performance as well as the cost goals set by DARPA for the RASCAL program. The baseline vehicle was found to only carry 52 lbs to the 270 nmi sun synchronous orbit. Several alternatives were evaluated off of the baseline design. The best of these alternatives can meet DARPA's performance goals and reach the cost goals of $5,000 per pound of payload with eight first stage vehicles flying 46 times per year for a total of 363 flights per year. Different economic cases were also evaluated to try and meet the cost goals in a less ambitious number of flights per year. It was found that if the DDT&E was paid for by another party (NASA, DOD, etc.) the cost goals can be met with just three vehicles flying 42 times per year for a total of 125 flights per year.

Nomenclature

APAS	Aerodynamic Preliminary Analysis System
CAD	Computer Aided Design
CBO	Congressional Budget Office
CER	Cost Estimating Relationships
Cl	Coefficient of Lift
c_j	Specific Fuel Consumption
Cr	Cruise
DARPA	Defense Advanced Research Projects Agency
DDT&E	Design, Development, Testing and Evaluation

*Graduate Research Assistant, School of Aerospace Engineering, Student member AIAA.
† Associate Professor, School of Aerospace Engineering, Associate Fellow AIAA.

DOD	Department of Defense
DSM	Design Structure Matrix
Ff	Fuel Fraction
GTOW	Gross Takeoff Weight
ICBM	Intercontinental Ballistic Missile
H_2O_2	Hydrogen Peroxide
HABP	Hypersonic Arbitrary Body Program
HTPB	Hydroxyl Terminated Polybutadiene
I_{sp}	Specific Impulse
L/D	Lift to Drag Ratio
LOX	Liquid Oxygen
Ltr	Loiter
M	Mach
MER	Mass Estimating Relationships
MiniVer	Mini-Version
MIPCC	Mass Injection Pre-Compressor Cooling
MMC	Metal Matrix Composite
MTBF	Mean Time Between Failures
NAFCOM	NASA- Air Force Cost Model
NASA	National Aeronautics and Space Administration
O/F	Oxidizer Fuel Ratio
POST	Program to Optimize Simulated Trajectories
Pro E	Pro Engineer
Q	Dynamic Pressure
R	Range
RASCAL	Responsive Access Small Cargo Affordable Launch
SLS	Sea Level Static
SSDL	Space System Design Laboratory
TAT	Turn Around Time
TBEAT	Turbine Based Engine Analysis Tool
TFU	Theoretical First Unit
TPS	Thermal Protection System
T/W	Thrust to Weight Ratio
ΔV	Ideal Change in Velocity

I. Introduction

Due to the uncertainty in today's world and the reliance of the US military on space based assets there is a need for assured and timely access to space. One way to accomplish this assured access to space is to use a combination of reusable and expendable vehicles. This combination will involve the use of a completely reusable first stage that is very similar to today's fighter aircraft. The second and third stages will comprise of a low cost expendable rockets for exo-atmospheric flight. The goal of this project is to get 250 lbs to any inclination with a high flight rate and a low cost of less than $5,000 per pound of payload or $750,000 per flight.

RASCAL is a Defense Department initiative that stands for Responsive Access, Small Cargo, Affordable Launch. The first stage will consist of a reusable aircraft similar to a large scale Air Force fighter. The first stage will also utilize Mass Injection Pre-Compressor Cooling (MIPCC) turbojet engines that will propel the stage to approximately two hundred thousand feet before releasing the second and third rocket stages. The first stage will be similar to a large fighter of the F-22 class, although the turbofans will be that of the more available F100 class. The MIPCC system will be a plug-in addition to the engines to help high altitude performance. This stage will be not only a "Launch Platform", but more of a first stage in that it will contribute

Figure 1. RASCAL Concept.

significantly to the overall acceleration of the vehicle

The second and third stages will consist of simple expendable rockets. Releasing the upper stages outside the atmosphere will reduce the loads on the stages as well as the risk of staging. Also by relying on the reusable portion for all atmospheric flight, the expendable stages can be designed simpler and therefore cheaper.

II. Initial RASCAL Design

The initial RASCAL program was initiated by DARPA in March of 2002. Phase I of RASCAL was a nine-month study contracted six teams to evaluate the feasibility of launching small payloads at a significant cost reduction over current launch systems. After the initial nine-month study phase II was initiated. This phase was awarded to the Space Launch Corporation in January of 2003 [1]. This eighteen-month phase is intended to advance the RASCAL design and allow for risk reduction testing [1]. The final phase will be initiated in July of 2004 for construction, testing and demonstration of the RASCAL design for an initial operating capability of 2006.

The initial RASCAL design consists of a combination of reusable and expendable vehicles. The first stage is a “fighter like” design implementing MIPCC (Mass Injecting and PreCooling Compressors) engines for exo-atmospheric flight. This segment will allow for the faster TATs than conventional launch vehicles and the low operating costs necessary to reach the RASCAL goals. The upper stages will consist of mass produced low cost expendables. In the initial design this consists of two stages. A low cost, high performance hybrid engine will propel the upper stage when released from the first stage. This stage will then propel the vehicle until a third solid stage ignites to take 250 lb payload into orbit. These stages will be mass produced in large quantities to take advantage of learning curves to reduce the cost per vehicle. A summary of the RASCAL baseline system is shown as Figure 2..

This figure depicts some of the Space Launch Corporations specifications for the RASCAL design. This design utilizes an 89 ft long, 90,000 lb first stage and a 42 ft long, 16,000 lb upper stage.

The first stage of this RASCAL design is slightly larger and heavier than a typical USAF fighter. This first stage is powered by four F-100 turbofans. These are the same turbofans that power both the F-15 and the F-16 currently in the USAF inventory. This reliance on proven technology should drive down the initial DDT&E costs for the RASCAL design. Unlike other next generation launch vehicles the RASCAL design does carry a pilot. The pilot restricts the performance of the first stage by forcing the aircraft to maneuver with less than six g’s of acceleration. The pilot does help keep the DDT&E costs of the initial design low since there is no need for a complicated automatic flight system. Also the USAF is a major proponent of the RASCAL design and prefers to have a manned fighter aircraft as opposed to unmanned air vehicles.

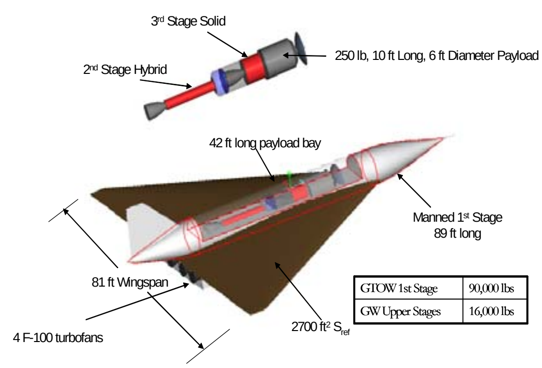

Figure 2. Launch Component Breakdown.

The RASCAL program requires the flight profile of the first stage aircraft to have a circular range of 250 nautical miles with a loiter capability of one half and hour. The flight profile involves a high speed acceleration and vertical rise segment which the RASCAL designers refer to as the “zoom maneuver”. This maneuver is accomplished by throttling up the MIPCC F-100s to full throttle and accelerating a high flight path angle to the operating limits of the MIPCC engines. The first stage engines shut down when this operating limit is reached and the entire vehicle coasts on momentum to over 200,000 ft. At 200,000 ft the first stage and second stage separate. This high altitude of separation allows the second stage to be released without high aerodynamic forces experiences at lower separation altitudes. This low dynamic pressure of separation allows the upper stages to operate without any added structure for aerodynamic fairings. This removal of structure increases the mass fraction of these upper stages and therefore the performance. The first stage then releases the upper stages which continue on to orbit. The first stage reenters the atmosphere unpowered along the glide slope determined by the trajectory. Once the dynamic pressure is within acceptable levels the MIPCC engines restart and the first stage returns to the airport as an aircraft. The zoom maneuver covers a downrange distance of over 180 nautical miles. This requires the first stage to fly a worst case of 430 nautical miles to the landing strip.

The flight profile is very similar to that which is flown by a typical aircraft. The launch vehicle segment of the flight profile is depicted in Figure 3. The first stage attains only 12% of the altitude obtained in the overall mission

(215,000 ft of the 1.6 million feet necessary to obtain a 270 nmi orbit). As Figure 4 shows the release of the upper stages occurs from the top of the first stage at approximately 220,000 ft with a flight path angle of 20 degrees with less than 10% of the velocity necessary to propel the payload into orbit. The remaining 90% of ΔV is provided by the upper stages. The main benefit of the first stage is not the ΔV provided but the release of the upper stages outside the atmosphere. This reduction in drag dramatically decreases the overall size of the upper stages. The hybrid propellant of the second stage was chosen over solid propellant for a number of reasons. First, the hybrid engines offer a higher performance than solids due to the higher I_{sp} that hybrids provide. A second reason hybrids were chosen over solids was that hybrids have the ability to shutdown if an anomaly in the firing of the engine is detected. This shutdown ability is not available in solid motors which could pose a danger to the manned first stage if a catastrophic failure were to occur soon after separation. The final stage was chosen to be solid for packaging as well as the cost considerations. These stages insert the payload directly into a 270 nautical mile circular orbit at a sun synchronous inclination of 98 degrees.

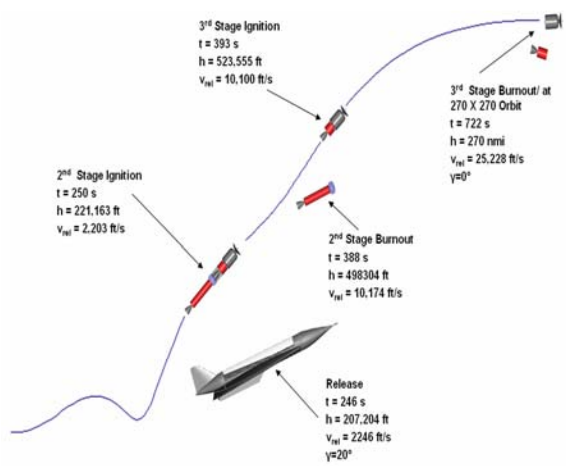

Figure 3. Launch Component Breakdown.

The RASCAL design is intended to launch a 250 lb payload into a 270 nautical mile circular orbit at a sun synchronous inclination of 98 degrees. The design goal is to also launch 400 lbs of payload into a 270 nautical mile, 28.5 degree inclination low earth orbit. Both of these mission profiles will be simulated with the RASCAL design being set for the more constraining mission. The secondary mission (the one which is not the driving constraint) will then be flown with the same design with the resulting payload exceeding the RASCAL design requirements.

III. Design Methodology:

The purpose of this project is to compare the published RASCAL numbers with those computed using a design methodology currently used in the Space System Design Laboratory at The Georgia Institute of Technology. To accomplish the RASCAL baseline design was analyzed in the following design structure matrix (DSM) (Figure 5). The DSM is a method of graphically interpreting the way that different contributing analyses (design disciplines) interact to create a design. This DSM is very similar to most launch vehicle designs. There is a strong iteration loop between the propulsion design, trajectory, and weights and sizing. This iteration loop does the major convergence of the vehicle. A smaller feedback occurs between aeroheating and weights and sizing (TPS weight). This is a smaller feedback since there are not radical departures in TPS design between similar trajectories. There is no feedback to the aerodynamics from the trajectory contributing analysis as one would expect. This is because the aerodynamic coefficients are non-dimensional and scale with the vehicle.

This design methodology is used for the RASCAL design. Due to the complexity of the design some contributing analyses are broken down into the two main constituents of the design (the first stage and the upper stage) to get converged solutions. These converged solutions are then recombined and reconverged in the contributing analysis before continuing through the DSM.

This method of simplifying the problem was mainly used in the trajectory contributing analysis. It was too difficult of a problem to run the entire trajectory from an initial guess so the first and upper stages were run independently matched at the separation point. Once this is achieved the two solutions are combined into one trajectory analysis with the solutions to the separate problems given as initial guesses to the combined problem. These "better guesses" allow the entire vehicle to be optimized and then closed in the main iteration loop.

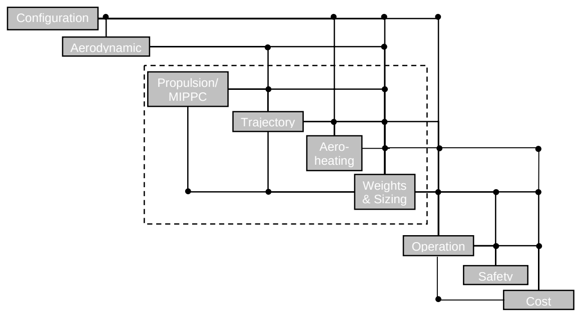

Figure 4. DSM for RASCAL Design.

Table 1. RASCAL Design Tools.

Discipline	*Analysis Tool*
Configuration	Pro/E
Aerodynamics	APAS (HABP)
Propulsion Design	TBEAT/MIPCC
Trajectory	POST 3-D
Weights & Sizing	MS Excel
Aeroheating	Miniver
Safety & Reliability	GT Safety
Operations	Historical Estimates
Cost	NAFCOM

As the DSM predicts many different design disciplines are combined to create the converged design. Each design discipline was executed to verify both the feasibility and viability of the RASCAL design. Each discipline will be presented with the tool used and results obtained from the analysis.

A. Aerodynamics:

The aerodynamic analysis for the RASCAL design was conducted using the Aerodynamic Preliminary Analysis System (APAS) computer code. APAS is a conceptual design aerodynamics tool used to obtain the lift, drag, and moment coefficients for a conceptual design. The APAS code is used to define the geometry of the conceptual design and then the analysis is conducted in one of two analysis codes. The geometry of the first stage, the total upper stage, as well as the third stage were all modeled in APAS. It should be noted that the second and third stages will be operating close to the APAS threshold of 350,000 ft and therefore the aerodynamic coefficients will be much less significant than that of the first stage.

Once the geometry is defined it must be analyzed to produce the aerodynamic coefficients necessary for the trajectory simulation. To analyze the aerodynamics historical data was used for the subsonic and transonic analysis Hypersonic Arbitrary Body Program (HABP) for was used for the hypersonic analysis. The historical data for subsonic and transonic analyses was obtained from F-14 design data [2]. This data contains the lift and drag coefficients as a function of Mach number and angle of attack. These coefficients were taken at a wing sweep of 55 degrees (that of the RASCAL design). The aerodynamic forces are then scaled by the coefficients and the RASCAL wing area of 2700 ft^2. This data was then combined with the HABP hypersonic analysis to create a complete the aerodynamic data for both the first stage and the upper stages of the RASCAL design.

B. Propulsion Design:

The design of the propulsion system for RASCAL involves four parts. The first part is the design of the turbofan engines. This will be conducted using historical data [3] as well as an airbreathing turbojet design tool, TBEAT [4]. The second part of the propulsion design involves the performance of the MIPCC engines. This will be evaluated via an AIAA paper written by Preston Carter and Vladimir Balepin [5]. The MIPCC design will then be combined with the TBEAT analysis to evaluate the performance of the first stage engines. The third part of the propulsion design will be the second stage hybrid design. This part will be design using historical data [6]. The final part of the propulsion design will be the design of the third stage solid propellant engine. This will again be designed from historical data [6]. The propulsion elements will then be combined to be used in the trajectory analysis.

1. Turbofan Design:

The turbofans used in the RASCAL design are the Pratt and Whitney F-100s. As noted earlier these engines are the same as those used on the F-15 and F-16 fighters. In fact these engines were not chosen because of their performance. In fact the Pratt and Whitney F-119, which powers the F-22, provides over 20% more thrust. The F-100 was chosen due to the availability, and therefore low cost, of the engines. The characteristics of the F-100 are provided in Table 2.

Table 2: F-100 Engine Characteristics [3].

Thrust	29,000 lbs
Weight	3,740 lbs
T/W Ratio	7.754
Length	191 in
Inlet Diameter	34.8 in
Max Diameter	46.5 in
Bypass Ratio	0.36
Pressure Ratio	32

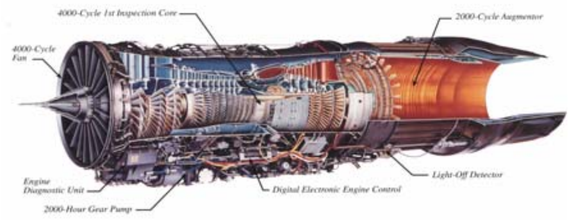

Figure 5: Pratt and Whitney F-100 Turbofan.

The F-100 is a low bypass turbofan which offers both high performance and efficiency. A diagram of the F-100 is included as Figure 5. These characteristics were then analyzed in TBEAT using an afterburning turbofan to obtain the dependence of thrust and I_{sp} on Mach number and altitude (Figure 6).

As expected as the altitude increases both the thrust and I_{sp} diminish. This is the main problem for using turbofans in space access systems. As the altitude increases the density of the incoming air decreases and the engines become inefficient and unable to produce the required thrust. Another problem is that in high speed flight the turbo machinery exceeds the maximum temperature of the materials. This causes the engine to melt itself. The solution is to pre-cool the incoming air to below the machine limits and to increase the density at high altitudes to retain the high thrust experiences at lower altitudes.

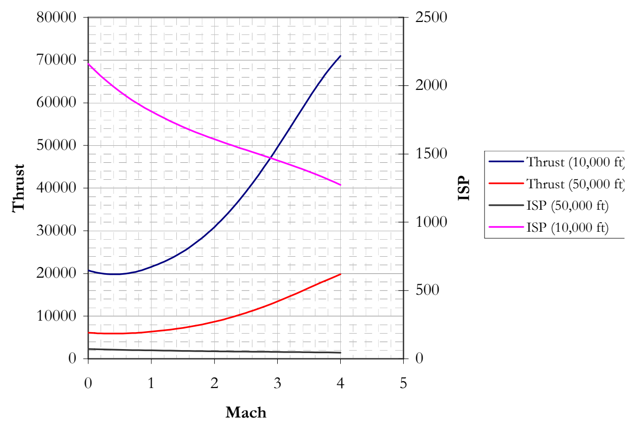

Figure 6: TBEAT data for F-100 Engine.

2. MIPCC Design:

MIPCC (Mass Injecting Pre-Cooling Compressor) technology dates back to the early 1950's. MIPCC is an engine enhancing technology initially designed to propel high speed fighters beyond Mach 3. These fighters were pursued by the USAF in the early 1950's to combat perceived cold war threat of high speed USSR bombers. As Intercontinental Ballistic Missile (ICBM) technologies improved the need to defend from high speed bombers was superceded by the need to protect from ICBMs. This put MIPCC on the technology shelf until recently with the advent of the RASCAL program.

The MIPCC bolt-on additions to the F-100 turbofans are the single most important enabling technologies for the RASCAL design. The MIPCC is what is used to attain the engine inlet conditions desired in the turbofan design section. MIPCC is a technology that introduces tanked water and LOX to the incoming air flow (pre-compressor) at high Mach numbers and at high altitudes. Typical aircraft engines are limited in altitude by the density of the incoming fluid (oxidizer). Typical turbofans are also limited in speed of the flow by the temperature limits of the combustor materials. MIPCC pushes out the altitude and speed boundaries by both cooling and adding density to the incoming flow of a turbofan. The result is that at high Mach numbers the incoming air is cooled by the water and the LOX. This allows the engine to operate at Mach numbers far exceeding

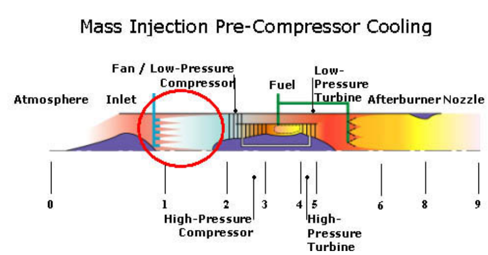

Figure 7: MIPCC Design Features [1].

the design limits for the nominal turbofans. Another benefit is that the incoming water and LOX add density to the incoming flow. This allows the engine to operate at higher altitudes that the design limits. A third benefit is that the LOX in the intake acts as a stabilizer in the oxidizer deprived combustion chamber at high altitudes.

To analyze the effect of MIPCC an AIAA paper written by Preston Carter was analyzed [5]. MIPCC engines are able to operate to altitudes greater than 85,000 ft and Mach numbers in excess of Mach 4. At these conditions the MIPCC system cools and increases the density of the flow so the engine appears to be operating at a lower altitude and Mach number. To model this, a translation of the TBEAT data was preformed using the apparent altitude and apparent Mach number at the flight altitude and Mach numbers. A summary of the performance of the MIPCC engines is included as Figure 8 and Figure 9.

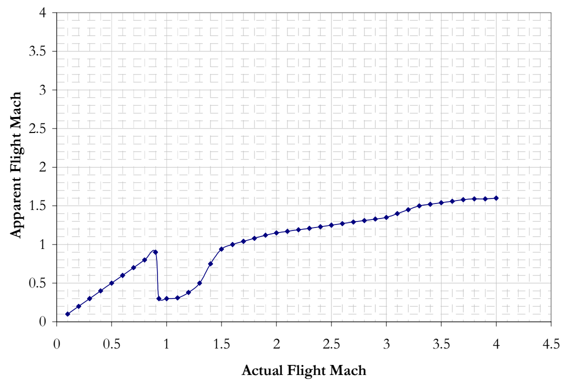

Figure 8: Apparent Mach Number Experienced by Turbo Machinery as a Function of Actual Flight Mach Number[5].

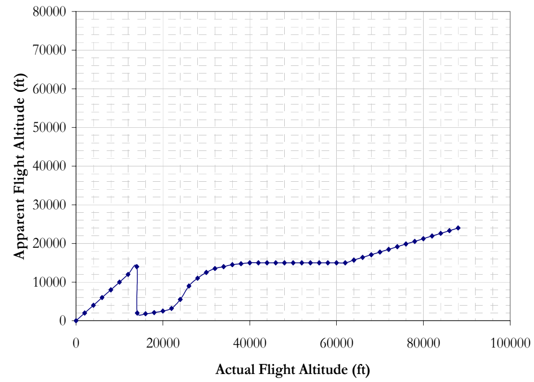

Figure 9: Apparent Flight Altitude Experienced by the Turbo Machinery as a Function of Actual Flight Altitude [5].

As these figures indicate the MIPCC system engages at Mach 0.9 and continues to operate throughout the trajectory. At Mach 4 and 88,000 ft the engine performs as if it is operating at an altitude of 24,000 ft and a Mach number of 1.6.

Unfortunately the MIPCC concept does have drawbacks. First the increased weight of the oxygen and water carried increases the vehicle size. Also these propellants have to be stored in separate tanks and therefore further increase the vehicle dry weight. Also the I_{sp} of the engines cannot be computed from a simple translation of the TBEAT data. As the tanked water and oxidizer flow rate increases, the I_{sp} of the engine must be adjusted to account for the additional mass flow. This was accomplished by first calculating the MIPCC propellant flow rate as a function of SLS fuel flow rate according to Figure 10. This data was then curve fit so the MIPCC propellant flow rate could be calculated at every Mach number and altitude. This curve fit was broken up into four regions two for each type of tanked MIPCC propellant. In three of the four cases a linear fit produced a good representation of the data, but for the initial water injection profile a cubic polynomial was used to fit the data. These curve fits are also provided in Figure 10. The result of this translation is a MIPCC thrust and I_{sp} as a function of flight Mach and altitude. It should be noted that the percentage of LOX in the MIPCC flow is determined by oxidation limits of the F-100 turbofans. Therefore an upper limit of 23% of the total flow of incoming air was set. This percentage of LOX by weight was used since that approximated the amount of oxygen in standard air.

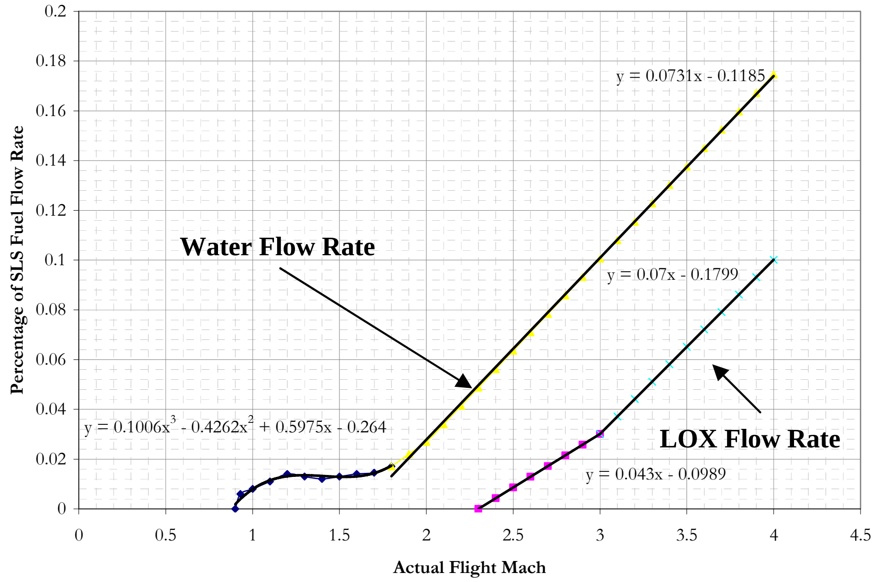

Figure 10: MIPCC Propellant Consumption Curve Fits as a Function of SLS Fuel Flow Rate [5].

3. Hybrid Design:

Hybrid engines are also a relatively new technology. Hybrid rocket engines combine both a solid fuel with a liquid oxidizer. The combination provides a performance greater than solid engines, but a cost and simplicity that can't be achieved by liquid engines. Another benefit of the hybrid engines over solid engines is that the hybrid engine can be shut down if a problem occurs by simply shutting off the flow of oxidizer to the solid fuel. This will allow a shutdown of the upper stage if a problem occurs near the manned first stage.

The hybrid engine for the second stage was designed using conceptual design methods[6]. Many different fuels were analyzed with HTPB/LOX combination providing the best I_{sp}. Unfortunately for the size of the second stage this results in a vehicle that is almost 10 feet longer than the 42 foot payload bay designated in the baseline. Therefore hydrogen peroxide (H_2O_2) was used the next best performance and a higher mixture ratio than the HTPB/LOX configuration which results in a smaller vehicle (Since the oxidizer is more dense than the fuel).

Once the propellant was determined the fuel chambers were designed to be as short as possible while still having the proper length to diameter ratio to support combustion. For hybrids this seemed to result in a seven port fuel chamber . The I_{sp} for this design can be estimated using conceptual design equations [6] and nozzle parameters that were set to be the maximum nozzles to fit within the diameter as well as the length constraints of the upper stage (approximated as 80% of a 15 degree half cone). This resulted in an I_{sp} of approximately 310 seconds. With this design and the propellant combination selected the overall weights can be calculated using MERs which will be described in subsequent sections.

4. Solid Design:

The solid third stage was designed in much the same way as the hybrid engine using conceptual design equations [6]. It was decided to use AL/HTPB propellant which would have commonality with the hybrid engine as well as providing an I_{sp} of approximately 293 seconds in vacuum. Once the third stage was then compared with existing rockets such as the Star 37 and Orion 38 to compare appropriate mass fractions.

Table 3: Propellant Mass Fraction Comparison of Solid Propellant Upper Stages [6].

	Mass Fraction
Star 37	0.915
Orion 38	0.859
3rd Stage	0.875

C. Trajectory:

The trajectory analysis for the RASCAL design involved two separate parts. For the airplane components of the trajectory airplane fuel fraction estimates were used [7]. This included segments 1-6 and 8-12 of the first stage mission profile. The second part of the trajectory analysis involved what is considered the launch vehicle segment of the trajectory. To analyze this segment of the trajectory POST was used. POST, the Program to Optimize Simulated Trajectories, is a three degree of freedom code written by Lockheed Martin and NASA [8]. POST was used to model the trajectory from Mach 0.8 through the zoom maneuver and the stage separations to third stage MECO.

1. Aircraft Trajectory:

The aircraft portion of this trajectory is very similar to the flight profiles flow by conventional aircraft. Because of the similarities between this profile and typical aircraft profiles a standard aircraft conceptual design method, fuel-fraction method [7], will be used to calculate the mission fuel in each segment of the mission profile. In this method the fuel fraction of each mission segment will be calculated from a combination of historical regressions as well as simple static values for similar aircraft types. Each fuel fraction is defined as the ratio of end weight to beginning weight. Fuel fractions for mission segments 1-3 and 9-12 were used as static historical values. While the remaining fuel fractions have been calculated using historical equations using aircraft characteristics. These equations are given below for the climb, cruise, and loiter portions of the mission profiles.

Table 4: Mission Fuel Fractions.

1 FF Engine Start	0.9900
2 FF Taxi	0.9900
3 FF Takeoff	0.9900
4 FF Climb	0.9714
5 FF Cruise out	0.9596
6 FF Loiter	0.9624
8 FF Cruise In	0.9295
9 FF Descent	0.9900
10-12 FF Landing	0.9950

With these fuel fractions the entire fuel consumed in the aircraft portions of the trajectory can be calculated.

2. Launch Vehicle Trajectory:

The second portion of the trajectory is the launch vehicle portion. This is the part of the trajectory which is unique to the RASCAL design. Because of the uniqueness of the trajectory POST was used to calculate the optimized trajectory. POST is a three dimensional trajectory optimization code which takes inputs from the propulsion, weights, and aerodynamics disciplines and simulates the trajectory of the spacecraft subject to the performance constraints listed in Table 5.

Table 5: POST Trajectory Constraints.

Max Axial Acceleration	6 g's
Max Dynamic Pressure	2000 psf
Max Angle of Attack	20 degrees
Max Dynamic Pressure at Release	1 psf
Final Orbit Apogee	270 nmi
Final Orbit Perigee	270 nmi
Final Orbit Inclination	98 degrees

Due to the complexity of the RASCAL trajectory the airbreathing, and rocket portions of the trajectories were calculated separately to get approximate “guesses” for the combined trajectories. These outputs with their appropriate initial conditions are then combined into one POST input file which is then optimized to minimize the propellant consumed by the stages.

The first stage input deck was set up to use the MIPCC engines as defined in the propulsion section. This input file starts with an initial weight, accelerates the aircraft until an appropriate time when the aircraft begins the zoom maneuver. This zoom maneuver involves increasing the altitude while still firing the MIPCC engines. Once the aircraft reaches the MIPCC limits of 88,000 ft and Mach 4 the engines shut down. The aircraft then continues to gain altitude by trading kinetic energy for potential energy. The first stage then coasts until the flight path angle drops to 20 degrees (other flight path angles were used, but 20 degrees results in the smallest vehicle). The first stage input deck actually tries to maximize the velocity of release to give the second stage as much energy as possible. The dynamic pressure constraint of release combined with this optimization scheme also results in an altitude in excess of 200,000 ft at release.

Once the first stage is optimized the second stage begins at the altitude, velocity, azimuth, latitude, longitude, and flight path angle of the end of the first stage. The second stage ignites after a coast of 5 seconds after release from the aircraft to get a significant distance between the stages. The second stage hybrid then ignites until the ideal ΔV provided by the second stage reaches 11,000 fps. This ΔV number was set in the RASCAL design, but it was traded and found to be close to the optimal point. After the second stage falls away the third stage ignites after a five second delay. The third stage then fires until the proper orbit is reached. The entire upper stage input file is designed to optimize the weight consumed. The final weight at the end of the run is then the total payload weight and the dry weight. The dry weight can then be subtracted off from the weights and sizing sheet to get the maximum payload. Once both stages are optimized they are combined into one deck to verify the results and to optimize the entire system. The optimized solution stages as soon as possible (while still meeting the dynamic pressure constraint) thereby firing the second stage with the highest relative velocity. The results of this trajectory are presented as Figure 11. As these figures show that the first stage provides only a small amount of the overall altitude and velocity that it necessary to achieve orbit. The first stage does release the second stage outside the drag of the atmosphere and that is where the majority of the benefit of the first stage is achieved.

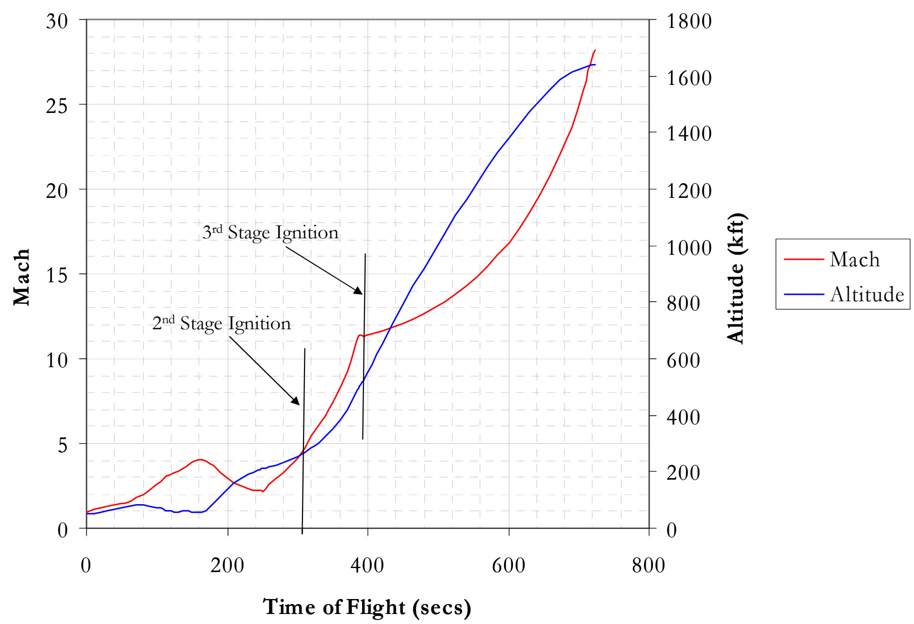

Figure 11: Baseline Trajectory (Altitude and Mach Number).

Once the baseline trajectory was set the same vehicle was flown from the Cape Canaveral, Fl flying due east to calculate the payload capability to that orbit. The payload capability of the baseline to both orbits is included in Table 6.

Table 6: Payload Comparisons for Baseline.

98 Degree Sun-Synchronous Orbit	52 lbs
28.5 Degree Low Earth Orbit	149 lbs

Both payload capacities are below the required RASCAL payloads, but from a comparison of the requirements to the obtained payload amounts the sun-synchronous mission drives the design of the rocket. The LEO orbit will exceed the required 400 lbs if the sun-synchronous orbit attains the 250 lbs requirement.

D. Weights and Sizing:

Once the trajectory analysis is complete the dry weights for each of the stages must be computed from the propellant weights calculated in POST. These weights are then converged to close the baseline vehicle. The dry weights are calculated using historical mass estimating relationships (MERs). Most of the first stage MERs were calculated from historical aircraft data [[9]]. Other components were taken directly from the actual flight hardware (ejection seat, turbofans, etc.). The only major exception is the TPS weight which was sized independently from the aeroheating data.

The major components of the first stage are the wing, body, main propulsion, and landing gear. The weight breakdown of the first stage is included as Figure 12.

Gross Weight	101,503 lb
Dry Weight	56,565 lb
Wing Group	7,936 lb
Tail Group	1,432 lb
Body Group	12,727 lb
TPS	865 lb
Landing Gear	3,034 lb
Main Propulsion	16,992 lb
Electrical Conversion and Distribution	1,138 lb
Hydraulic Systems	660 lb
Surface Control and Actuators	2,021 lb
Avionics	1,964 lb
Environmental Control	200 lb
Personnel Equipment	218 lb
Dry Weight Margin	7,378 lb

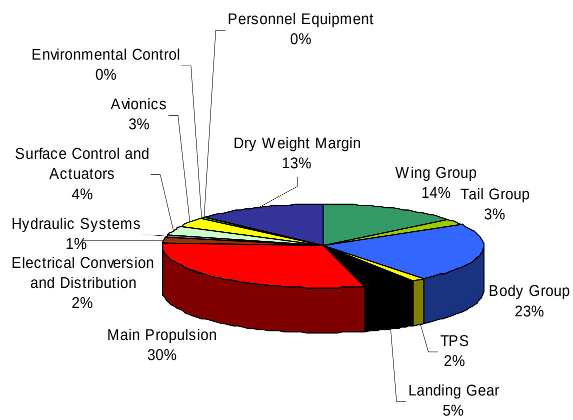

Figure 12: Weight Breakdown for 1st Stage Baseline.

From this weight breakdown it can be seen that the body and main propulsion (including MIPCC engines) are the main contributors to the dry weight. Weight growth margin is also a significant portion of the dry weight (15% of pre-margin dry weight).

The second stage was modeled from MERs from both expendable rocket data [9], as well as conceptual design equations [6]. The hybrid engine was modeled as a pressurized oxidizer tank, a solid fuel casing, a feed system for the oxidizer, and a nozzle. The fuel casing is sized as a seven port solid with a 48% volumetric efficiency [6]. A weight breakdown of the second stage is included as Figure 13. It should be noted that the structure includes both the fuel and oxidizer tanks, while the propulsion elements include the engine nozzle and the feed system. The gross weight includes both the payload and the gross mass of the third stage.

Gross Weight	16,000 lb
Dry Weight Margin	261 lb
Structure	999 lb
Propulsion	397 lb
Power	121 lb

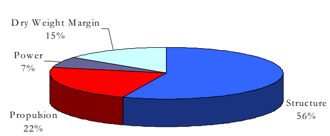

Figure 13: Weight Breakdown for 2nd Stage Hybrid.

Gross Weight	3,289 lb
Dry Weight Margin	419 lb
Structure	105 lb
Propulsion	126 lb
Power	25 lb
Avionics	183 lb

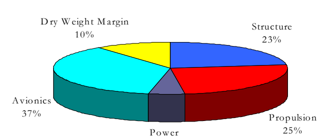

Figure 14: Weight Breakdown of Third Stage Solid.

The third stage of the RASCAL design was also modeled from MERs from both expendable rocket data [9] as well as conceptual design equations [6]. The solid engine was modeled as tank with a volumetric efficiency of 90%. The weight breakdown is very similar to that of the second stage except that the third stage has a more complicated avionics system. The third stage also has a larger percentage of the weight in the propulsion system since the nozzle is much larger percentage when compared to the overall third stage system. The third stage also carries the payload previsions. The gross weight contains the entire structure, propellant, and payload.

E. Aeroheating:

The aeroheating analysis for the RASCAL design was complete using Miniver. Miniver is an aeroheating code that predicts the radiative equilibrium temperature for a given cross section and trajectory. The Miniver analysis was conducted for the first stage (the upper stages are released outside the atmosphere), at both the centerline of the aircraft, as well as the quarter chord location of the wing. Miniver takes the trajectory outputs as well as the geometry defined in the configuration and calculates the temperature at each position. The trajectory (AOA, Sideslip angle, altitude, and velocity) of the first stage was inputted to Miniver using 35 points all for the first stage. To model the geometry, 19 points were used on both the windward and leeward sides of the fuselage, while 8 points were used for both sides of the wing. A temperature profile for both the wing and the fuselage are included in Figure 15.

As this plot shows the maximum stagnation temperature on both the wing and nose does not exceed 1300 degrees F. Unfortunately, Aluminum's reuse temperature is only 300 degrees F. Therefore TPS is needed on the first stage. MA-25 was chosen due to its availability, and will be used as a spray-on ablator which would be reapplied on every flight [10]. This ablator is used extensively in the shuttle program and can withstand one time uses exceeding 1,200 degrees F. The overall weight of the TPS system is 865 lbs and is included in the weights and sizing sheet.

The TPS for the second and third stages was neglected as well as the reentry of the first stage. The upper stages are released outside the sensible atmosphere so would not experience much heating. The reentry of the first stage may be significant, but to accommodate it, a generous safety factor of 1.5 was applied to the TPS weight on the first stage.

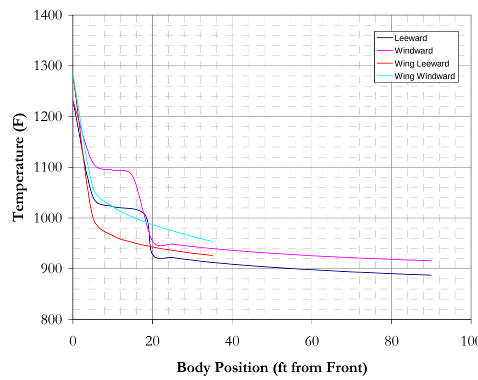

Figure 15: Radiative Equilibrium Temperature Calculated from Miniver.

F. Operations:

The operation model for the RASCAL design was analyzed using a manpower analysis derived from historical X-15 data [11]. The RASCAL program will be a reduced manpower program that resembles a small X-plane architecture rather than a massive Space Shuttle architecture. A tool was developed in Excel to model the manpower necessary for the RASCAL program based upon the number of flights per year and the premise that one first stage could fly no more than 50 flights per year (TAT greater than 1 week). It was assumed that it would take 12 ground operators working on each RASCAL first stage with 26 flight operations officers (X-15 heritage) and one pilot for each first stage. There would also be one manager for each first stage aircraft with a minimum of 3 managers. The number of people was then multiplied by the average man-year number given for 2004 in Transcost ($220,500 USD (FY'04)) [12]. The necessary supplies were taken to be ten percent of the labor costs for both the ground and flight ops.

The cost of operating the aircraft was calculated using a USAF average of operations cost per hour of flight [13]. This is then multiplied by a conservative 2.5 hours per flight (averaging 230 mph over the flight). This then computes the aircraft operating costs. New facilities were assumed to be the vehicle assembly building which was set at the cost of a typical hanger with equipment ($50 M USD (FY'04)).

G. Safety and Reliability:

The safety and reliability analysis was conducted using GT-Safety II v1.6. This safety program is an Excel model that uses failure rates, vehicle configuration, and weights and sizes to calculate vehicle failure rates and casualty rates. The RASCAL design was modeled as two stages with the upper stages combined into one stage with two engines with no engine out capability.

The first stage of the RASCAL design was analyzed using the operations data as well as the cost data for required flights per year. Aggressive options for abort options and windows as well as crew escape were taken since the first stage features a high speed ejection seat, as well as the ability to land while fully loaded. Other factors such as subsystem failure rates were taken at 1/10 of the ELV data since the fighter operation should be significantly more reliable that ELV launches. It was also assumed that the first stage could land safely with two engines out, but fail to make the mission. This would give the vehicle a thrust to weight of at worst 0.6 at takeoff, which is well within the flight regimes of most fighter aircraft.

The upper stages of the RASCAL design were modeled in GT Safety with much less aggressive numbers. The subsystem failure rates were taken at 80% of the ELV data since the second stage will act as an ELV, just operating outside the atmosphere. It was also assumed that the staging point would not be over a populate area. The resulting numbers for the reliability and safety analysis are included as Table 7.

Table 7: GT Safety Outputs.

1st Stage	
Loss of Mission MTBF	1 in 1094 Flights
Loss of Vehicle MTBF	1 in 6494 Flights
Casualty Rate	0.0014
Upper Stages	
Loss of Mission MTBF	1 in 149 Flights
Loss of Vehicle MTBF	1 in 186 Flights
Casualty Rate	0
Total Vehicle	
Loss of Mission MTBF	1 in 131 Flights
Loss of Vehicle MTBF	1 in 180 Flights
Casualty Rate	0.0015

As this table shows the MTBF for the first stage is very good with a mission failure only once in 1094 flights. A loss of vehicle is even rarer with one occurring in 6494 flights. The casualty rate of 0.0014 is also exceptional with only one accident occurring every 715 years. The upper stage is not nearly as reliable since it operates as a rocket rather than an aircraft. The loss of mission every 149 flights with a loss of vehicle every 186 flights (80% of LOM failures are consider LOV). This reliability analysis results in an overall launch system that will lose a mission every 131 flights, a vehicle ever 180 flights, and a man every 660 years. This loss of crew number is very high due to the abort capability and ejection system built into the manned components of the vehicle (first stage).

H. Cost Estimation:

The cost estimation for this project was conducted using the NASA-Air Force Cost Mode (NAFCOM), with some inputs from the Transcost model [12]. NAFCOM with some Transcost cost estimating relationships was used to calculate the DDT&E as well as the TFU for the RASCAL vehicle.

Each of the subsystems of the design has its own coefficients A and B. Therefore the user of NAFCOM can manipulate the cost of the components by adjusting the CF or complexity factor. For the RASCAL baseline complexity factor of close to one were used for all subsystems except the avionics which typically uses a CF of approximately 0.3. The resulting costs for the first stage are included as Figure 16.

In Millions of $	DDTE	TFU
Wing Group	$ 149.82	$ 21.47
Tail Group	$ 53.55	$ 7.28
Body Group	$ 198.70	$ 57.14
TPS Group	$ 96.18	$ 20.02
Landing Gear	$ 20.40	$ 0.66
Electrical Conversion & Dist.	$ 44.34	$ 14.69
Hydraulic Systems	$ 31.98	$ 5.08
Surface Control Actuation	$ 121.78	$ 81.58
Avionics	$ 126.67	$ 7.55
Environmental Control	$ 20.58	$ 2.77
Personal Equipment	$ 33.21	$ 1.30
Engines	$ 583.00	$ 6.00
Total	**$ 2,931.00**	**$ 446.00**

F-22 TFU ~ $345

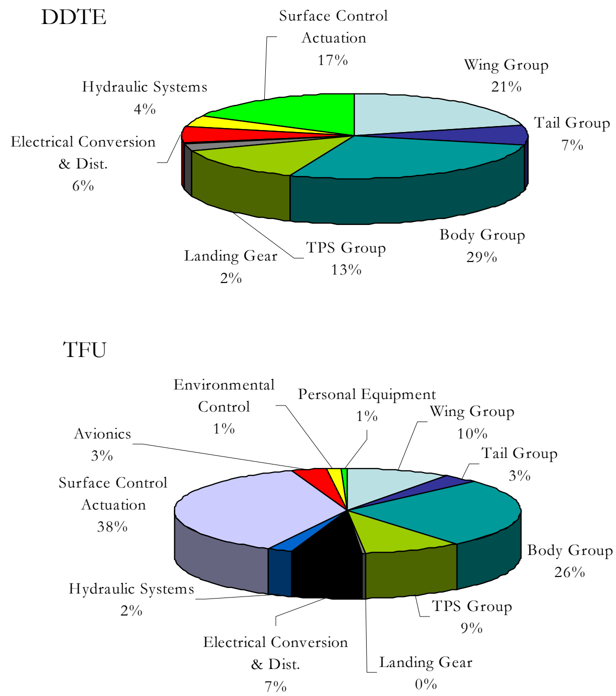

Figure 16: Cost Breakdown for RASCAL 1st Stage.

This analysis results in a DDT&E cost of just under $3 B USD (FY'04) with a TFU of $446 M USD (FY'04). The engine DDT&E number includes a cost for developing the MIPCC technology, but no cost for developing the F-100 turbofans. When these numbers are compared to the F-22 program the TFU of the baseline comes out slightly high. (The TFU of the F-22 was backed out from the initial production order of six aircraft at a cost of 1.6 billion dollars with a learning curve of 85% [[14]]). The surface control and actuation seems high for this vehicle, but the complexity factor was left at 0.9 since the aerodynamic maneuvers performed by the first stage are occurring at high dynamic pressures and therefore require expensive actuators.

The second and third stages were again modeled together as two engines. The CERs are similar to the ones used in the first stage analysis except they have been adjusted to use expendable data from Transcost [12]. For the upper stages the fuel casings for the solid propellant (for both the hybrid and the solid) are considered propulsion weight along with the nozzles.

In Millions of $	DDTE	TFU
Body Group	$ 36.09	$ 3.59
Power	$ 4.52	$ 1.13
Avionics	$ 28.64	$ 1.56
Engine 2nd Stage	$ 174.88	$ 2.98
Engine 3rd Stage	$ 14.81	$ 0.80
Total	**$ 417**	**$ 17**

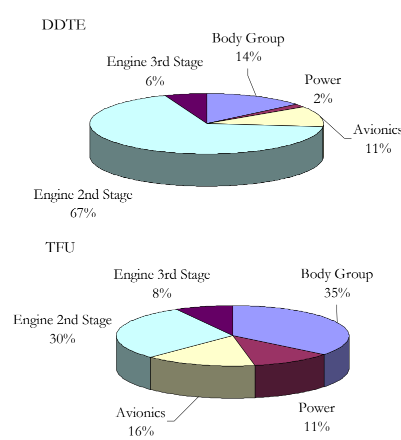

Figure 17: Cost Breakdown for Upper Stages.

As this figure depicts the DDT&E for the upper stages are $417 M USD (FY '04) with a TFU of $17 M USD (FY '04). This TFU number has been used taking aggressive cost cutting numbers for the structure, power, and avionics (0.4, 0.3, and 0.3 respectively). These numbers were justified since the structure and power systems are much simpler than systems NAFCOM was created to design. The engines seem to dominate both the DDT&E as well as the TFU. That is because the engines comprise of the majority of the vehicle weight (Only the power systems, connecting structure, and LOX tanks are not considered propulsion).

Once these numbers were calculated they were combined into a cost calculating spreadsheet. This spreadsheet calculated the required number of flights and resulting number of first stages (one first stage required for every 50 flights per year). This was then combined with the operations model into one spreadsheet to calculate the overall costs of the program. Learning curves for both the first and upper stages was set to 85%. The upper stage takes advantage of this learning curve tremendously due to the number of flights necessary to reach the cost goals of $750,000 per flight or $5,000 per pound.

To calculate the overall cost of the RASCAL program some assumptions were made. First it was decided that the program will begin in 2005 and last for 20 years (spreads the DDT&E over the life of the program). It was also decided that different scenarios would be looked at and the number of flights necessary to reach the RASCAL cost goals would be calculated. The cases are described below:

Case 1: Only the upper stages are purchased every flight. The production of the first stage as well as the DDT&E are paid for by some other agency (NASA, USAF, etc.).
Case 2: The upper stages are purchased for every flight as well as the necessary first stages. The DDT&R is paid for by some other agency (NASA, USAF, etc.).
Case 3: The total cost for the program is paid for by RASCAL. This includes the entire vehicle as well as all DDT&E and facilities

Each of the cases was then calculated for the baseline and evaluated against each of the RASCAL cost goals

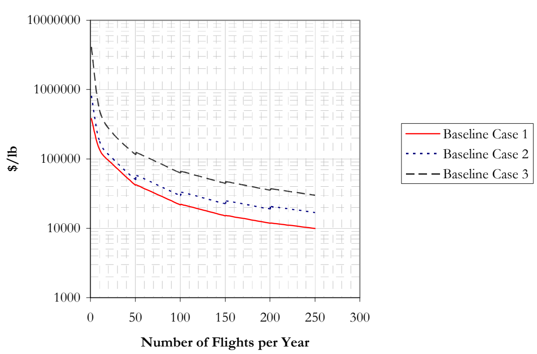

Figure 18: Baseline Dollars per Pound of Payload.

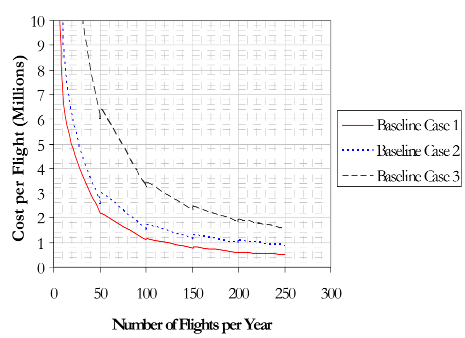

Figure 19: Baseline Cost per Flight.

As the above figure shows only Case 1 reaches the desired goals in less than 250 flights per year (155 flights). Case 2 takes 336 flights per year where Case 3 takes 660 flights per year to reach the RASCAL goal of $750,000 per flight. The somewhat saw-tooth profile of the cost charts is due to the fact that a new first stage must be purchased every fifty flights per year (A first stage can only fly 50 flights per year for 20 years). Every time a new first stage must be purchased there is a cost jump for all three cases (Case 1 the small jump is only due to the pilot, where as in the other cases it is due to the pilot and aircraft. When the second cost goal of RASCAL is analyzed the answer is not as promising. Due to the actual payload of 52 lbs calculated in the performance section no case reaches the $5,000 per pound goal in 250 flights. Case 1 takes 594 flights per year, Case 2 takes 1511 flights per year, and Case 3 takes 2490 flights per year to meet the RASCAL cost goal. This is of course unreasonable numbers since the demand will never be close to two flights a day, which is the best case scenario for $5,000 per pound.

IV. Baseline Conclusions:

After the RASCAL design has been evaluated it has been found to be underperforming and well above the cost numbers quoted as goals. The payload is far below the stated performance goal and the costs are far higher than the cost goals. The 52 lbs of payload is only 20% of the stated goal of 250 lbs. This is well below the goal and the weight of most small satellites. The flight rates necessary to reach the two cost goals are summarized in Table 8.

Table 8: Cost Goal Summary for Baseline.

	\$750,000 per Flight	\$5,000 per lb
Case 1	155	594
Case 2	336	1511
Case 3	660	2490

Only the case 1 scenario approaches reasonable flight number of about once every 2.5 days for \$750,000 a flight. Even this number may be unachievable if the demand is not there.

V. Design Alternatives:

Because the shortcomings in the baseline RASCAL design the same DSM was used to evaluate different alternatives to this baseline. These alternatives were first evaluated on a performance basis and then the most promising alternative will be chosen to be completely evaluated as the GT RASCAL design. The alternatives looked at in the RASCAL designs are as follows in Table 9.

Table 9: Design Alternatives for RASCAL Baseline.

Alternatives Number	Description:
1	Changing the hybrid oxidizer from H_2O_2 to LOX
2	Changing the first stage main structure (wing, tail, and fuselage) to a Metal Matrix Composite (MMC).
3	Add a fourth kick stage so the second and third stages only have to propel the rocket to an intermediate orbit (80X270)
4	Combine Alternative 1 and 3
5	Combine Alternative 1, 2, and 3

Each of these alternatives will be closed in the performance aspects of the DSM (all except cost, operations, and economics). Unless noted all alternatives represent one change off of the baseline at a time.

1. Alternative 1:

The first alternative will be to change the second stage oxidizer from hydrogen peroxide to liquid oxygen. The advantage of making this change is that the I_{sp} of the second stage will increase to 340 seconds vs. the 310 seconds for the hydrogen peroxide. A disadvantage to this change is that the second stage will become larger. This is due to the fact that the HTPB/LOX combination operates at a lower O/F ratio (1.9 vs. 6.5). This smaller mixture ratio results in a larger rocket. This alternative configuration was closed in the internal iteration loop of the DSM and this alternative resulted in the overall payload capacity of the baseline RASCAL design increasing from 52 lbs to 75 lbs.

2. Alternative 2:

The second alternative involves changing the baseline first stage structural material from standard aluminum to a metal matrix composite. This MMC will not only make the design much lighter, it will also eliminate the need for a thermal protection system (up to 1500 F). This is due to the fact that the reuse temperature of the MMC is higher than the temperatures encountered by the first stage. The disadvantage to the MMC is that it is an immature technology, and must be developed and tested far more that the typical aluminum structure. This will cause the DDT&E of the design to increase dramatically over the baseline. This alternative configuration was closed in the internal iteration loop of the DSM and this alternative resulted in the overall payload capacity of the baseline RASCAL design increasing from 52 lbs to 110 lbs.

3. *Alternative 3:*

The third alternative involves changing the trajectory of the upper stages so that the second and third stages put the payload into a transfer orbit (80 nmi by 270 nmi orbit). A fourth stage solid motor is then added to the upper stage to circularize the payload at the 270 nmi circular orbit. An advantage to this alternative is that the upper stages will become smaller each performing a smaller ΔV. A disadvantage of this approach is that the fourth stage will add dry mass to the system as well as involve another component that could fail. This could result in a lower total system reliability. This alternative could result in an increased cost due to the addition of the extra stage. (The GT RASCAL cost analysis will show this cost is actually made up for in the weight reduction). This alternative configuration was closed in the internal iteration loop of the DSM and this alternative resulted in the overall payload capacity of the baseline RASCAL design increasing from 52 lbs to 211 lbs.

4. *Alternative 4:*

The fourth alternative involves combining the LOX hybrid with the fourth stage kick motor. This will carry the advantages of improving the upper stages without altering the first stage design. It will result in a slightly more expensive rocket than the baseline, but will start to approach the payload numbers set out in the RASCAL program. This alternative configuration was closed in the internal iteration loop of the DSM and this alternative resulted in the overall payload capacity of the baseline RASCAL design increasing from 52 lbs to 224 lbs.

5. *Alternative 5:*

The fifth and final alternative involves combining the LOX hybrid with the fourth stage kick motor and the MMC upper stage. This design should result in the highest performance since it uses all of the enhancing alternatives presented. This alternative should also result in the most expensive vehicle since the LOX hybrid adds length to the first stage, the fourth stage adds the cost of a new stage, and the new MMC technology needs to be matured. When this configuration was closed in the internal iteration loop the DSM the payload increased from 52 lbs to 303 lbs. This is above the desired 250 lbs of payload set out in the RASCAL goal. This higher payload means that this alternative can be further scaled down to achieve the 250 lb goal.

A summary of the payload capacity of the alternatives is presented as Figure 20. The LOX Hybrid second stage shows promise for improving the performance of the second stage by adding efficiency (higher I_{sp}), but comes with the cost of having a longer vehicle. The MMC alternative also adds performance, but at the cost of using an unproven technology on the first stage. This technology may not come to maturity by the time the RASCAL is ready to fly. Adding the fourth stage to the design adds the most performance without changing the first stage. This fourth stage adds performance by allowing the second and third stages to propel the satellite into a transfer orbit. This transfer orbit does not incur the same steering losses that are obtained by flying directly into a circular orbit. A combination of the alternatives needs to be used to create the optimal design. Only the fourth and fifth alternatives are capable of achieving the required performance so those designs will be further optimized to the RASCAL performance goals and then compared.

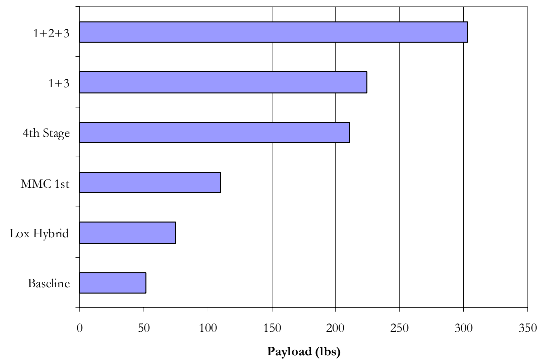

Figure 20: Payload Capacity for Alternative Designs

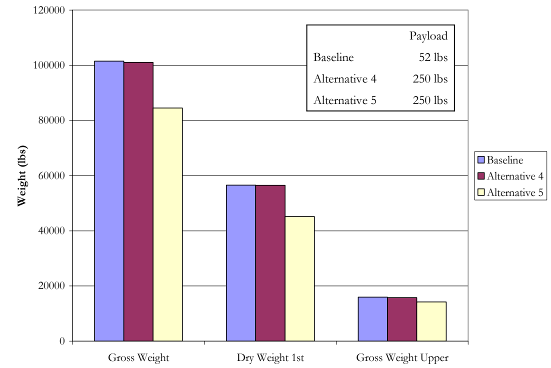

Figure 21: Optimized Alternatives Performance Comparison

To fully evaluate the designs the fourth and fifth alternatives (1+3, 1+2+3) were reoptimized to the RASCAL goals (250 lbs to sun-synchronous orbit). This was possible by scaling up alternative four and scaling down alternative five. This reoptimization was done to compare the two vehicles with the same performance. This was be completed by combining both the first and second stages into one trajectory deck and letting the optimizer find the smallest vehicle (weight consumed) that can get 250 lbs to the 98 degree sun-synchronous orbit. These vehicles will be closed in much the same way as the baseline with both vehicles being evaluated in the DSM and the resulting performance and costs compared.

Each of the alternatives was able to meet the performance goals set out by the RASCAL program. A summary of the total weights of the resulting designs is included as Figure 21. As this figure shows, at each stage the fifth alternative results in a smaller vehicle. The MMC technology offered a lighter, better performing first stage, which allowed the second stage to achieve a smaller ΔV. The baseline is also included in this figure even though the payload of the baseline is only 20% that of the alternative designs.

The design alternatives were costed in much the same way as the baseline the alternative designs were costed using a combination of the NAFCOM tool as well as Transcost CERs. The cost analysis results for cost per flight follows as Figure 22.

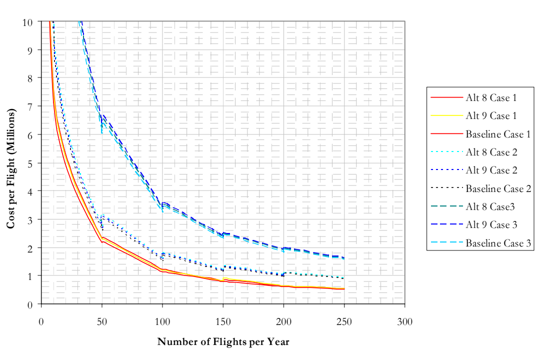

Figure 22: Cost per Flight Comparisons of Optimized Alternatives.

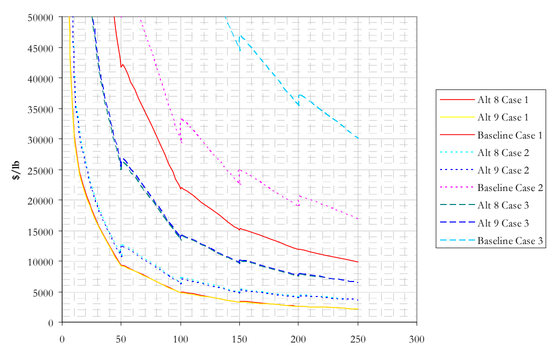

Figure 23: Dollars per Pound of Payload of Optimized Trajectories.

As this figure shows the baseline has the cheapest cost per flight of every design. This is misleading because cost per flight does not take into account the fact that the baseline only carries 52 lbs to orbit. If only the alternatives which reach the performance requirement are evaluated alternative 5 results in the lowest cost per flight for Case 1 and 2, with alternative 4 being the cheapest for Case 3. The reason alternative 4 is cheaper in Case 3 is due to the higher DDT&E costs of alternative 5 over alternative 4 ($3.675 B vs. $3.48 B respectively). This higher DDT&E is due to the increased cost of having a MMC first stage, which is not a fully mature technology. The increased cost of the MMC is taken into account through the use of complexity factors on the CERs. The complexity factors take into account the fact that some designs may cost more per pound than conventional designs due to the complexity of the materials or processes used in the manufacture of the design.

Even though the complexity factors were set higher for the MMC design alternative 4 has a higher TFU than alternative 5 ($464 M vs. $446 M). This is due to alternative 5 having a much lower weight per stage than alternative 4. This weight savings results in a cheaper vehicle than alternative four even with complexity factors that are almost twice as high. This is due to the weight-based CERs favoring the lighter vehicle with the high complexity factors over the heavier vehicle.

The dollars per pound of payload was also calculated for both alternatives as well as the baseline (Figure 23). This dollar per pound of payload comparison penalizes the baseline for only taking 52 lbs versus the 250lbs of the alternatives. As this figure shows the two alternatives are very similar again, with alternative 5 being slightly better in case 1 and 2 and slightly worse in case 3.

VI. GT RASCAL:

From this alternative analysis one design was chosen to be the Georgia Tech RASCAL design. From the alternative analysis either alternative 4 or alternative 5 were both feasible and viable solutions. Alternative 5 was chosen as the Georgia Tech design because of its lower costs on economic Cases 1 and 2. Case 3 was determined to be a non-factor since the flight rates necessary to reach this case were about one flight a day for the $5,000 per pound of payload and two flights a day for the $750,000 per flight goal. These flight rates will probably never be attainable, and therefore case three should not be a deciding factor on which alternative to chose.

Table 10: Comparison of GT RASCAL with Baseline.

	Baseline	GT RASCAL
Length	89 ft	95 ft
Payload Bay Length	42 ft	48 ft
GTOW	101,502 lbs	84,549 lbs
Gross Weight Upper Stages	16,000 lbs	14,273 lbs
Payload	52 lbs	250 lbs
Technologies	MIPCC, Hybrid 2nd Stage	MIPCC, MMC, Hybrid 2nd Stage

As this table shows the GT RASCAL is lighter in both the GTOW as well as the upper stage gross weight. The GT RASCAL design carries five times the payload to orbit only using one more technology than the baseline, as well as a fourth stage.

A weight breakdown was conducted for the GT RASCAL design. This dry weight breakdown for the RASCAL first stage follows (Figure 34). As this figure shows the TPS and hydraulic systems were eliminated in the GT RASCAL design. The TPS was eliminated due the high temperature resistance of the MMC and the hydraulic fluid was eliminated by using electromechanical actuators for all flight control systems. Since the propulsion segments remained static when compared with the baseline design, the percentage of propulsion weight actually increases due to the decreased weight of the system.

Gross Weight	84,550 lb
Dry Weight	5,183 lb
Wing Group	6,200 lb
Tail Group	660 lb
Body Group	7,570 lb
TPS	0 lb
Landing Gear	2,600 lb
Main Propulsion	16,980 lb
Electrical Conversion and Distribution	1,140 lb
Hydraulic Systems	0 lb
Surface Control and Actuators	1,820 lb
Avionics	1,910 lb
Environmental Control	200 lb
Personnel Equipment	220 lb
Dry Weight Margin	5,890 lb

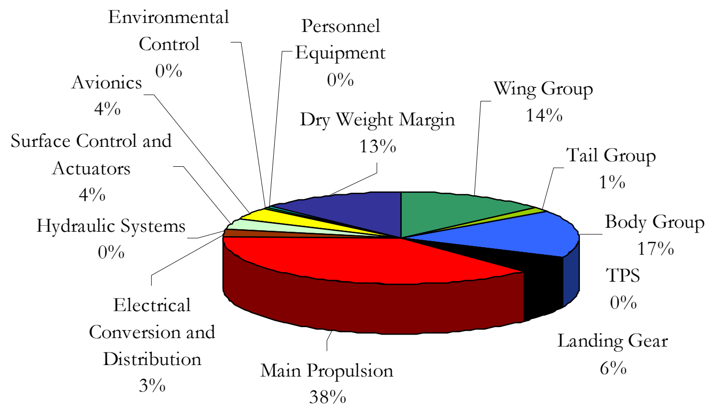

Figure 24: Weight Breakdown of 1st Stage of GT RASCAL.

A weight breakdown was also completed for the upper stages of the GT RASCAL design. The following figure shows how the gross weight of the upper stages has decreased due to the fact that the higher Thrust to Weight Ratio (T/W) first stage can propel the upper stage higher and faster. (T/W increases since the 4 MIPCC F-100 thrust is constant and the MMC first stage is lighter). Also the transfer orbit, although requiring a faster insertion speed, doesn't have as high of gravity and thrust vectoring losses as the higher orbit. Therefore the upper stage doesn't have to provide as much energy to the system and can therefore be smaller.

The third stage of the GT RASCAL also benefits from the weight savings in the first stage. The third stage is 500 lbs lighter than the baseline. This can be attributed to the same energy savings as the second stage as well as the fact that a fourth stage has been added.

Gross Weight	14,270 lb
Dry Weight	2,180 lb
Structure	1,160 lb
Propulsion	370 lb
Power	130 lb
Dry Weight Margin	280 lb

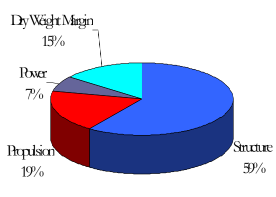

Figure 25: Weight Breakdown of GT RASCAL 2nd Stage.

Gross Weight	2,750 lb
Dry Weight	370 lb
Structure	110 lb
Propulsion	100 lb
Power	30 lb
Avionics	180 lb
Dry Weight Margin	50 lb

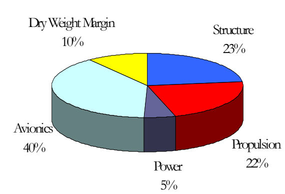

Figure 26: Weight Breakdown of GT RASCAL 3rd Stage.

The Fourth and final stage of the GT RASCAL design utilizes a very small apogee kick motor. This motor is just over 32 lbs with a dry weight of 9lbs. This engine was sized using a simple two body orbital mechanics as well as the rocket equation

When looking at the zoom maneuver almost all of the propellant is fuel. That is because the first stage accelerates to Mach 3 before it starts to climb out of the atmosphere and loses the majority of its oxidizer intake. The zoom maneuver uses more water than lox because the water is used from Mach 1 onward where are the LOX is not used until later in the trajectory. The water is the major factor in the MIPCC system since it increases the density of the flow and cools the turbo machinery, where as the LOX mainly stabilizes the combustion at high altitudes.

A final mission profile for the GT RASCAL design is also included (Figure 27). As this figure shows the release of the second stage is at a higher altitude and a higher velocity than the baseline. This allows the upper stage rockets to be much smaller and in turn reduces the first stage weight further. This mission profile also shows that the GT RASCAL design releases the payload from the bottom of the first stage. This was decided since when the first stage drops the payload there is still over 1200 lbs of lift on the first stage. When they payload is released it will fall faster than the first stage and therefore will be farther away that the same payload dropped from the top of the first stage.

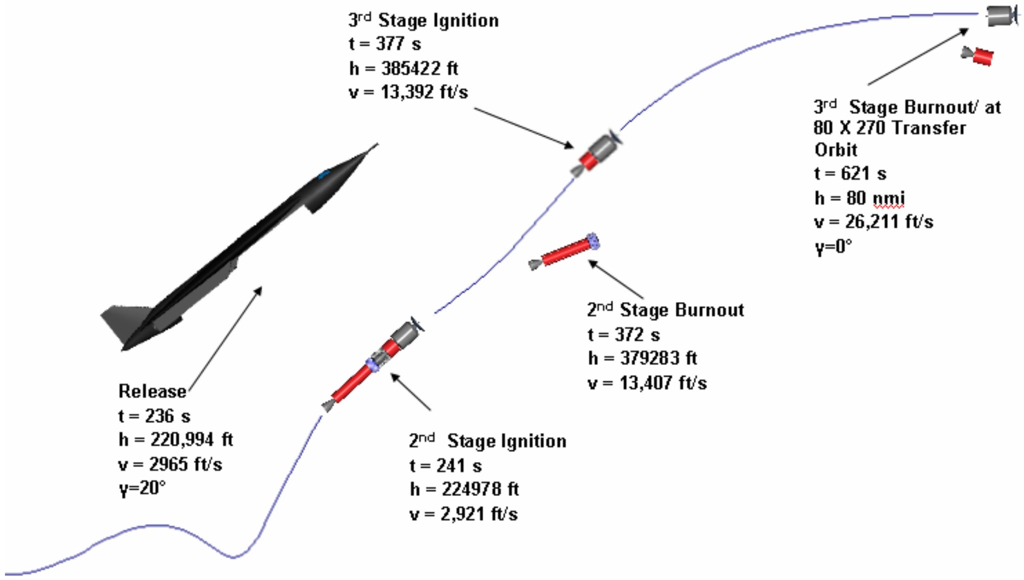

Figure 27: GT RASCAL Mission Profile.

When the final cost numbers of the GT RASCAL design were calculated, they were compared with the initial RASCAL goals. A summary of the number of flights necessary to obtain the two RASCAL cost goals follow as Table 11.

Table 11: GT RASCAL Flights per Year to Attain $5,000 per Pound of Payload.

Case 1: Purchase Only Expendables	91 flights per year
Case 2: Purchase Only Flight Vehicles	125 flights per year
Case 3: Purchase Vehicles and DDT&E	363 flights per year

As Table 11 shows, the GT RASCAL design can meet the case 1 economic scenario with only 91 flights and two first stage aircraft, while the case 2 economic scenario can be meet with just 125 flight and three first stage aircraft. The third case, as previously noted, if very tough with even the GT RASCAL design. For this case the vehicle must fly about once a day with eight first stage vehicles. The alternative four design actually falls slightly below this number at 361 flights per year. The second cost goal of $750,000 dollars per flight is slightly harder to obtain. For case 1 it takes 176 flights per year with four first stage aircraft. Case 2 requires 350 flights with seven first stage aircraft, while Case 3 requires 742 flights per year with a fleet of 15 first stage aircraft.

VII. Conclusions:

For this project both the baseline vehicle and five alternative vehicles were examined. When the baseline vehicle was evaluated it was found to fall short of the performance as well as the cost goals set by DARPA for the RASCAL program. The baseline vehicle was found to only carry 52 lbs to the 270 nmi sun synchronous orbit. When the baseline vehicle was costed it was found to meet the first cost goal of $750,000 per flight in just 155 flights per year for economic Case 1, 366 flights per year for Case 2, and 660 flights per year for Case 3. Unfortunately the $5,000 dollar per pound goal was hindered by the smaller than expected payload capacity and the number of flights per year was greater than 500 for all cases.

Several alternatives of the baseline design were analyzed to try to produce a feasible and viable design. From this alternative analysis two were continued through the entire design process. These optimized alternatives were both able to reach the performance goals set out by DARPA as well as being very similar in the number of flights necessary to meet the cost goals. The alternative 5 design was chosen as the GT RASCAL design because of the lower number of flights necessary to meet both cost goals of $750,000 per flight and $5,000 per pound for two of the three economic cases. The GT RASCAL only cost slightly more than alternative 4 in the Case 3 economic scenario. This was the most unlikely scenario since it required an enormous amount of flights to meet the RASCAL cost goals.

The GT RASCAL design can meet DARPA's performance goals and reach the cost goals of $5,000 per pound of payload with eight first stage vehicles flying 46 times per year for a total of 369 flights per year. Different economic cases were also evaluated to try and meet the cost goals in a less ambitious number of flights per year. It was found that if the DDT&E was paid for by another party (NASA, DOD, etc.) the cost goals can be met with just three vehicles flying 42 times per year for a total of 125 flights per year.

Acknowledgments

The author would like to thank the members of the Space Systems Design Lab at the Georgia Institute of Technology.

References

[1]. Space Space Launch Corporation, URL: http://www.spacelaunch.com/RASCAL.asp, Last visited 5/2/04.

[2]. Alford, William, "NASA Assessment of United States Navy F-14 Airplane Designs," Langley Working Paper LWP-794, Hampton Virginia, 1969.

[3]. Pratt and Whitney Products: F-100, URL: http://www.pratt-whitney.com/prod_mil_f100.asp, Last visited 5/2/04.

[4]. Bechtel, Ryan, "Turbine Based Engine Analysis Tool". URL: http://www.ssdl.gatech.edu/~ssdl/T-BEAT/T-BEAT.html, Last visited 4/15/04.

[5]. Carter, Preston, "Mass Injection and Precompressor Cooling Engines Analyses," AIAA-2002-4127, 38th AIAA Joint Propulsion Conference and Exhibit, Indianapolis, Indiana, July 7-10 2002.

[6]. Humble, Ronald, Larson, Wiley, and Wertz, James, Space Mission Analysis and Design, 3rd Ed. Microcosm, New York: 1999.

[7]. Roskam, Jan, Airplane Design Part I: Preliminary Sizing of Airplanes, DAR Corporation, Lawrence, Kansas: 1997.

[8]. Powell, R.W., et. Al., "Program to Optimize Simulated Trajectories (POST) Utilization Manual, Volume II, Version 5.2," NASA Langley Research Center, Hampton, VA; Martin Marietta Corporation, Denver, CO., October 1997.

[9]. Rohrschneider, R., "Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design," AE8900 Special Project, Georgia Institute of Technology, April 26, 2002.

[10]. Lockheed Martin Space Systems Company: Thermal Protection, URL: http://www.lockheedmartin.com/michoud/products/tpp/tpp_ma25.htm, Last visited 5/2/04.

[11]. Morris, W. Douglas, "X-15/HL-20 Operations Support Comparison," NASA Technical Memorandum 4453, June 1993.

[12]. Koelle, Dietrich, "Handbook of Cost Engineering for Space Transportation Systems with Transcost ," TransCostSystems, Liebigweg, Germany, 2000.

[13]. Crippen, Dan L, "The Effects of Aging on the Costs of Operating and Maintaining Military Equipment," Congressional Budget Office, August 2001. http://www.cbo.gov/showdoc.cfm?index=2982&sequence=0, Last visited 5/2/04.

[14]. Rodrigues, Louis J. "Progress in Meeting F-22 Cost and Schedule Goals,"

General Accounting Office,1999.

General References:

1. Benson, T., "Earth Atmosphere Model," June 4, 2002, http://www.grc.nasa.gov/WWW/K- 12/airplane/atmos.html, Last visited 5/2/04.
2. Raymer, Daniel P., Aircraft Design a Conceptual Approach, AIAA Education Series, Reston, VA, 1999.
3. Wall, Robert, "Hot Rod to Space," Aviation Week. McGraw Hill Corp: September 22, 2003.