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B. St. Germain, J. R. Olds, J. McIntire, D. Nelson,
J. Weglian
Space Systems Design Laboratory
Georgia Institute of Technology
Atlanta, GA

L. Ledsinger
The Aerospace Corporation
El Segundo, CA

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Starsaber: A Small Payload-Class TSTO Vehicle Concept Utilizing Rocket-Based Combined Cycle Propulsion

Brad St. Germain*, John R. Olds†, Laura Ledsinger‡, James McIntire*, Douglas Nelson§, John Weglian*
Space Systems Design Lab
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA, 30332-0150

ABSTRACT

This paper introduces *Starsaber*, a new conceptual launch vehicle design. *Starsaber* is a two-stage-to-orbit (TSTO) vehicle capable of putting a 300 lb. class payload into low Earth orbit (LEO). The vehicle is composed of a reusable winged booster, powered by two hydrocarbon fueled ejector ramjet (ERJ) engines, and a LOX/RP-1 expendable upper stage. The vehicle utilizes advanced structural and thermal protection system (TPS) materials, as well as advanced subsystems.

Details of the conceptual design process used for *Starsaber* are given in this paper. Disciplines including mass properties, internal and external configuration, aerodynamics, propulsion, trajectory simulation, aeroheating, and cost estimation are used in this study. A baseline design was generated and a 2-level 15 variable Taguchi L16 array was used to determine key system variables' influence on vehicle weight and cost. Based on these preliminary results the *Starsaber* vehicle was optimized for both minimum weight (gross and dry weight) and recurring cost. The lowest recurring cost vehicle was estimated to have a recurring cost per flight of \$2.01M, a gross liftoff weight of 168,000 lb. and a booster length of 77 ft.

* - *Graduate Research Assistant, School of Aerospace Engineering, Student member AIAA.*

† - *Associate Professor, School of Aerospace Engineering, Senior member AIAA.*

‡ - *Former Graduate Student Currently Working for The Aerospace Corporation, Member AIAA.*

§ - *Graduate Research Assistant, School of Aerospace Engineering, Member AIAA.*

NOMENCLATURE

ANOM	analysis of the mean
CER	cost estimating relationship
DoE	design of experiments
DSM	design structure matrix
ERJ	ejector ramjet
GLOW	gross liftoff weight (lb.)
Gr/Ep	graphite epoxy
H_2O_2	hydrogen peroxide
I_{sp}	specific impulse (sec)
LEO	low Earth orbit
LOX	liquid oxygen
MER	mass estimating relationship
q	dynamic pressure (psf)
RBCC	rocket-based combined cycle
RP	rocket propellant
TRF	technology reduction factor
Ti-Al	titanium aluminide
TPS	thermal protection system
TSTO	two-stage-to-orbit
T/W	thrust-to-weight ratio
WBS	weight breakdown statement

INTRODUCTION

One of NASA's goals is to identify key vehicle technologies that will enable significantly lower cost launch services for the ultra-lite and small payload community. This 300 lb. – 500 lb. payload class is often associated with University Explorer scientific missions. Budgets for these flights are typically limited (less than \$1M - \$1.5M for a dedicated flight), but scientific and educational value can be significant. Aggressive new concepts and technologies are needed to address this potential user base.

This paper summarizes part of a conceptual study conducted by the Space Systems Design Laboratory (SSDL) at Georgia Tech with the support and collaboration of NASA's Marshall Space Flight Center (MSFC). The goal of this study is to investigate a promising concept for small payload-class missions using rocket-based combined cycle (RBCC) propulsion technology. NASA's MSFC currently has a development effort in RBCC engines. Previous work completed under this study included the analysis of a small payload-class two-stage-to-orbit system utilizing a Hankey wedge-shaped LOX/LH2 ejector scramjet powered booster[1].

CONCEPT OVERVIEW

Starsaber, as seen in Figure 1, is a two-stage vehicle that uses a conical winged-body booster along with a low cost expendable upper stage.

Figure 1: *Starsaber* Concept.

The booster is powered by two ejector ramjet engines and is fully reusable. The fuel/oxidizer combination used for the ejector ramjet engines is one of the factors considered in the design of experiments trade study performed. The fuel choices are propane and JP, while the oxidizer choices are LOX and hydrogen peroxide (H_2O_2). The low cost upper stage is expendable. It uses a LOX/RP-1 rocket to place the 300 lb. payload into low Earth orbit.

Mission Profile

The unpiloted *Starsaber* vehicle takes off and lands horizontally from a notional airfield at the Kennedy Space Center. The initial acceleration occurs in ejector mode. The vehicle then transitions to ramjet mode and flies along a constant dynamic pressure (q) boundary until the booster and enclosed upper stage have accelerated to the rocket mode transition point. After rocket transition the vehicle accelerates off the dynamic pressure boundary to a high altitude staging point. The upper stage is then jettisoned as the dynamic pressure falls below 1 psf. The booster then descends and turns back toward KSC and begins its powered cruise-climb flyback maneuver, while the upper stage and payload continue to accelerate to a circular 200 nmi. low Earth orbit. This mission profile is shown in Figure 2.

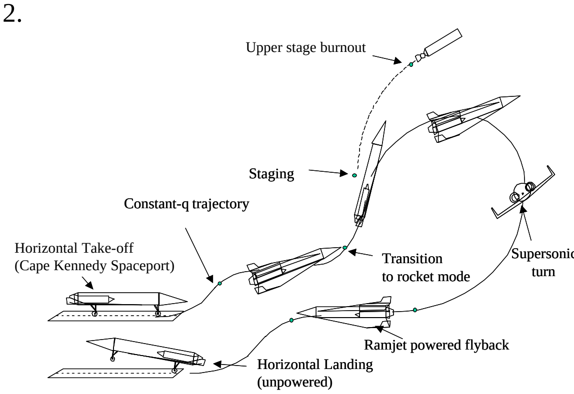


Figure 2: *Starsaber* Mission Profile.

DESIGN PROCESS

Starsaber is designed using a collaborative, multidisciplinary design process. An integrated design team is used with each team member responsible for a specific discipline. Team members each execute an individual disciplinary analysis tool and these disciplines are coupled in an iterative conceptual design process in which information about each candidate design is exchanged between the disciplines until the propellant mass fractions of each segment of the mission converge. The design process is most conveniently represented by the design structure matrix (DSM) shown in Figure 3. The main iteration loop identified in the DSM is expanded in Figure 4. These figures outline the design process for the *Starsaber* booster. The DSM for the upper stage consists of a simple iteration loop between the weights and sizing and trajectory disciplines.

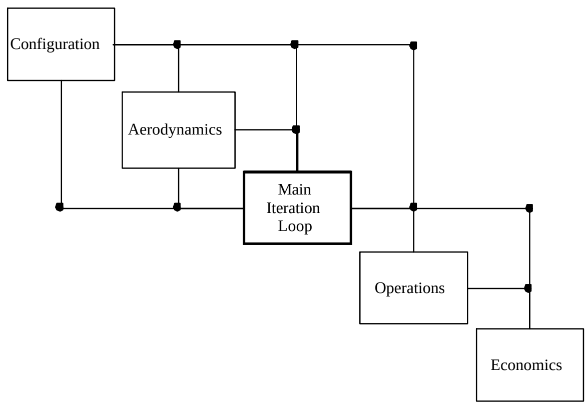


Figure 3: Design Structure Matrix.

Design structure matrices are useful for showing the interactions between the various disciplines used for the analysis of *Starsaber*. The lines on the top of the diagram represent the feed-forward loops. These show what information must be passed from the current discipline to a subsequent discipline. The lines below the diagonal represent feedback loops and show what information must be passed upstream. Disciplines that are connected by feedback loops require iteration between the disciplines to achieve a converged design. The main iteration loop shows the strong coupling between the propulsion, performance (trajectory), and the weights and sizing disciplines. The aeroheating discipline is only weakly coupled. This strong coupling inside the main iteration loop is responsible for the many iterations required to achieve a converged vehicle design.

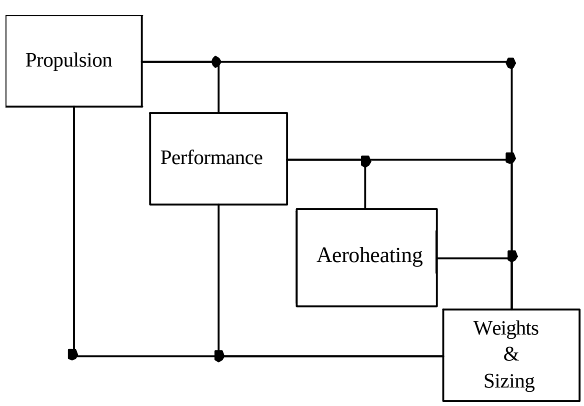


Figure 4: Main Iteration Loop.

At the beginning of the design process a brainstorming session is used to get an initial configuration for the design. All disciplinary experts have equal input in this session. After the initial design is determined the configuration and aerodynamic disciplines work together to determine a feasible packaging and aerodynamic configuration. The vehicle is then converged and properly sized for the 300 lb. payload using the main iteration loop. The vehicle is considered converged when the change between the gross and dry weight of the vehicle does not exceed 0.1% between iterations. The operations and economic analysis is conducted after a converged vehicle design is achieved. The following sections give a more detailed description of the individual disciplines.

DISCIPLINARY ANALYSIS

Configuration

The initial vehicle layout is set after consultations between the aerodynamics engineer and the configuration engineer. This is done to arrive at a vehicle shape that has the desired aerodynamic performance as well as one that has a high packaging efficiency. After the outer mold line is set, SDRC I-DEAS solid modeling software is used to determine the internal configuration of the vehicle. For a given reference vehicle length and vehicle mixture ratio, the configuration engineer draws the locations of the main propellant tanks, upper stage and RCS tanks. From this drawing, reference surface areas and other key geometric features are determined. These values are incorporated into the Microsoft Excel© weights and sizing spreadsheet. Since the layout of the *Starsaber* vehicle is composed of relatively simple geometric shapes a mostly analytical model is used to determine the fuselage and propellant volumes. This model is incorporated into the weights and sizing spreadsheet and along with the key geometric features supplied by the configuration model, allows the weights and sizing spreadsheet to determine all tank and vehicle lengths, surface areas and volumes.

The oxidizer tank, fuel tank, RCS tanks, and payload bay containing the upper stage occupy the internal volume of the booster. The integral fuel tank is located in the forward section of the booster, with the payload bay in the middle and an integral oxidizer tank in the rear. A 3-view of the *Starsaber* vehicle is shown in Figure 5.

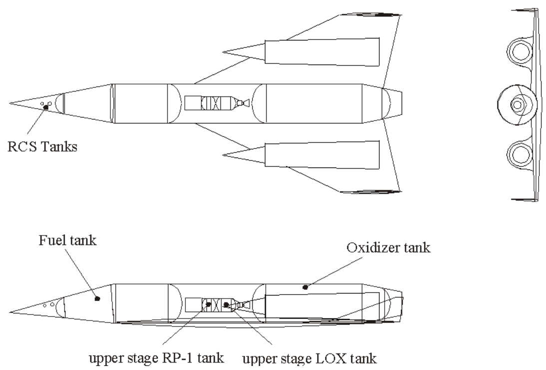


Figure 5: *Starsaber* 3-View.

Aerodynamics

The aerodynamic properties of *Starsaber* are evaluated using a conceptual design tool entitled APAS[2] (Aerodynamic Preliminary Analysis System). APAS, which is written in Fortran, was developed by Rockwell International as an aid in the design of the Space Shuttle. APAS couples two subprograms that separately perform the low speed and high speed aerodynamic analysis. UDP (Unified Distributed Panel) is used for Mach numbers up to Mach 3.5. This program uses the geometry created within APAS to perform a vortex lattice method on the body panels. HABP (Hypersonic Arbitrary Body Program) is used to analyze the hypersonic flight regime.

APAS requires several data inputs in order to perform the aerodynamic analysis. These inputs include the vehicle's external geometry and parameters such as the reference wing planform area, leading edge sweep angle, and an estimate of the position of the center of gravity of the vehicle. Figure 6 shows the *Starsaber* geometry file used for the aerodynamic analysis.

APAS is able to provide the trajectory discipline tables of lift and drag coefficients over a wide range of altitudes, Mach numbers, and angles of attack. Pitching moments are also generated in APAS, but because the trajectory is flown untrimmed they were not used in the analysis. During the design process, the *Starsaber* vehicle is photographically scaled to achieve the proper propellant volume and vehicle mixture ratio. This scaling does not affect the relative external geometry of the vehicle. The aerodynamic coefficients generated by APAS will remain constant during the analysis, but the actual lift and drag values scale with the vehicle's reference area. Therefore, the aerodynamic analysis is only required at the beginning of the design process.

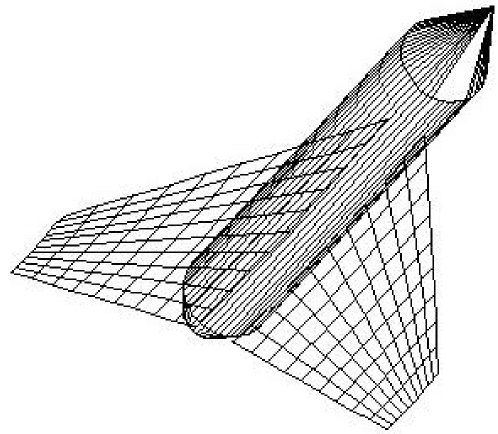


Figure 6: *Starsaber* APAS Geometry.

Weights & Sizing

The weights and sizing analysis for *Starsaber* uses a photographic scaling set of parametric mass estimating relationships (MERs) that have a NASA Langley heritage. These mass estimating relationships are combined with the analytical configuration model mentioned earlier, to form the weights and sizing spreadsheet. This spreadsheet receives required mass and mixture ratios from the trajectory analysis for both the booster and upper stage and then photographically scales the vehicle to meet these requirements. Since changing the vehicle scale changes the gross weight, capture area, sea level static thrust requirements, and other vehicle parameters, the disciplines in the main iteration loop shown in Figure 4 must be iterated until the vehicle size converges. This process usually takes 4 to 5 iterations, depending on the initial guesses of the various vehicle parameters.

The MERs used for the *Starsaber* analysis are based on near-term materials and construction techniques. Therefore these relations are adjusted by a technology reduction factor (TRF) to allow their use for the advanced materials and technologies utilized on *Starsaber*. The material used for the prime structure of the booster is either graphite epoxy or an advanced metal matrix composite (e.g. titanium-aluminide) depending on the particular vehicle design being analyzed. Graphite epoxy is also used for the propellant tanks. Several other advanced subsystems are assumed for *Starsaber*. They include an autonomous flight control system, lightweight avionics, and a vehicle health monitoring system. The upper stage is made of low-cost conventional materials and uses a relatively simple LOX/RP-1 rocket engine.

For the lightest gross weight design, the converged vehicle has a gross weight of 71,950 lb. and a dry weight of 17,400 lb. The upper stage weighs 3,400 lb. including the 300 lb. payload. Graphical breakdowns of the vehicle's gross and dry weight are shown in Figures 7 and 8.

The weights and sizing discipline supplies a great deal of information to the other analyses. The trajectory analyst uses the vehicle gross weight, wing reference area, upper stage weight, and maximum wing normal force. The cost analyst uses the complete 28-point weight breakdown statement and the required sea level thrust is used by the propulsion discipline.

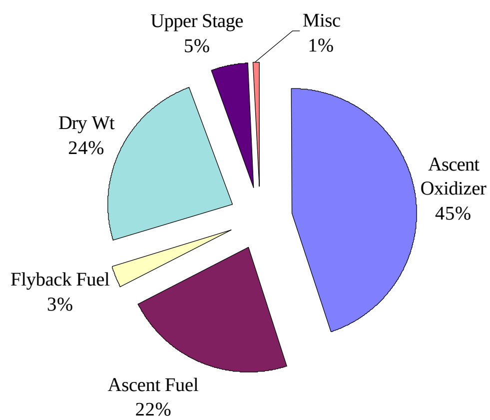


Figure 7: Gross Weight Breakdown.

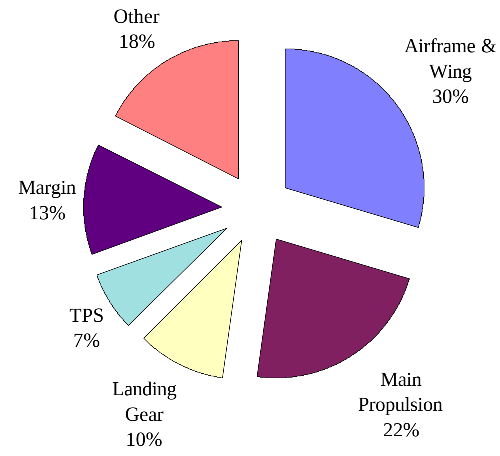


Figure 8: Dry Weight Breakdown.

Propulsion

The propulsion system for the *Starsaber* booster is analyzed using SCCREAM[3]. SCCREAM, the Simulated Combined Cycle Rocket Engine Analysis Module, is a one-dimensional code developed at Georgia Tech that models many different types of Rocket-Based Combined Cycle (RBCC) propulsion systems. This code is used to determine the performance characteristics of the two ejector ramjet engines used to power the booster. The output obtained from SCCREAM is an engine performance deck that is pre-formatted to be used in the trajectory program. This deck contains engine thrust or thrust coefficient, and specific impulse (I_{sp}) for a range of altitudes and Mach numbers for each operating mode of the ejector ramjets.

The ejector ramjet engines accelerate the vehicle to the staging Mach number. They also provide the ramjet propulsion used for the powered flyback of the booster to the launch site after separation from the upper stage. The two hydrocarbon-fueled ejector ramjets are axisymetric with a varying inlet throat. Figure 9 shows the notional engine shape analyzed in SCCREAM.

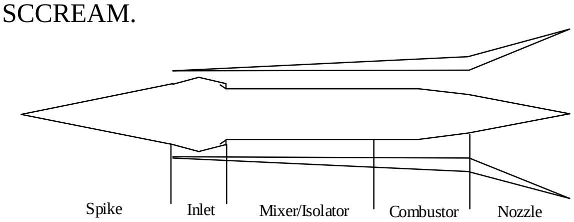


Figure 9: Notional Ejector Ramjet.

A rocket primary located in the engine flow path provides the initial power for the acceleration to ramjet speeds. The engines are sized to provide the required sea level static thrust-to-weight requirement specified for the particular vehicle design being analyzed. The required inlet area is also specified for each design.

During the trajectory analysis, the transition from ejector to ramjet mode is modeled by linearly throttling the ramjet up while the ejector is ramped down over the half Mach number preceding the transition point. The same procedure is used for the transition from ramjet to all-rocket mode. The rocket performance calculations for the all-rocket mode use the same rocket primary subsystem from the ejector mode with a larger expansion area.

Performance

Starsaber follows a branching trajectory; therefore the upper stage and booster must be modeled separately after the staging point. The booster ascent and upper stage ascent are modeled using POST[4], while the booster flyback is modeled in Microsoft Excel©. POST, the Program to Optimize Simulated Trajectories, is a three degree-of-freedom code that was written by Lockheed Martin and NASA. It is a generalized event-oriented trajectory optimization code that numerically integrates the equations of motion given the aerodynamic and propulsive characteristics of the vehicle. The program minimizes the given objective function, usually propellant consumed, while meeting the given trajectory constraints.

As mentioned above the booster ascent trajectory was modeled in POST. This involves the portion of the trajectory from horizontal takeoff until the staging point. The trajectory is constrained by a maximum dynamic pressure boundary, a 3g maximum acceleration in all-rocket mode, and a maximum wing normal force load during the pull-up maneuver at the beginning of the all-rocket mode. The value of the maximum dynamic pressure allowed during the trajectory was 1600 psf. This constraint limits the internal engine pressure and vehicle heat loads. The wing normal force limit represents a compromise between wing structural concerns and the more fuel-efficient, sharp pull-up maneuver at the beginning of all-rocket mode.

Starsaber first operates in ejector mode, then transitions to ramjet mode and flies along a constant dynamic pressure boundary until the all-rocket mode is reached. It then continues in rocket mode until the staging point is reached. The specific Mach numbers where these transitions occur are determined by the design of experiments array for the particular case being analyzed. For the lowest recurring cost case, the ejector/ramjet transition occurs at Mach 3.5, the ramjet/rocket transition occurs at Mach 5.5, and staging occurs at Mach 14. The initial dynamic pressure boundary followed during ramjet mode is also determined by the design of experiments. For the lowest recurring cost vehicle a q of 1500 psf is the initial value. After reaching this q-boundary, the optimizer in POST is allowed to vary the dynamic pressure boundary followed during the remainder of the airbreathing mode, as long as the maximum dynamic pressure constraint is met. An additional constraint imposed in the DoE on maximum engine static pressure limited, in some cases, the q-boundary that could be flown. Once the booster ascent is optimized, the ending conditions (altitude, speed, latitude, longitude, etc.) are used as inputs into remaining the two branches of the trajectory.

The upper stage branch is also analyzed using POST and is very straightforward. It involves only one engine operation mode because the upper stage is an all-rocket vehicle. After reaching the required orbit using a single burn trajectory, the mass ratio of the upper stage can be calculated and used in the weights and sizing analysis.

The flyback segment of the trajectory is more difficult to analyze in POST. Therefore, it was decided that a different analysis method was needed for the flyback segment. The solution was to create a Microsoft Excel© workbook to numerically approximate the values of interest from the flyback. The idea was to create simple approximations that could be used together to generate a basic picture of what is happening during the flyback segment.

The flow of information in the workbook is simple. The initial location, speed, and altitude of the vehicle are input into the spreadsheet based on the output of the POST ascent analysis. Then using some basic aerodynamic approximations, the vehicle glides into a turn around until it is pointed back toward the launch site. The distance back to the launch site is then calculated and passed to the section of the workbook that calculates how much fuel will be burned during the flight back to the launch site. Once this distance is known, the program uses a form of the Breguet range equation to calculate the mass ratio of the vehicle during flyback. This section of the workbook is the most involved and merits further explanation.

In order to apply the range equation, data is needed from other disciplines. First, the workbook must know what the engine performance of the vehicle is so that it can properly keep track of fuel consumption. This data comes from SCCREAM and is represented as a function of both altitude and Mach number. The workbook performs a linear interpolation to determine which values to use. Figures 10 and 11 show plots of ramjet thrust coefficient and Isp versus Mach number for a constant altitude of 60,000 ft.

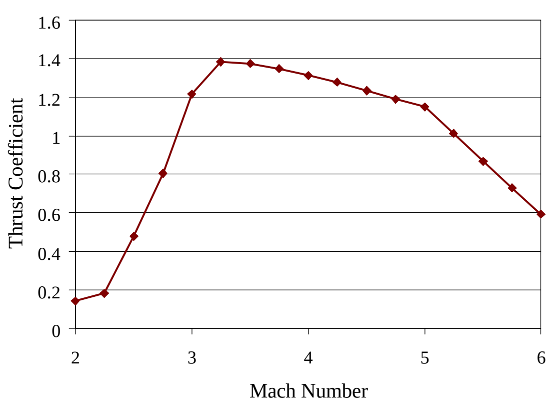


Figure 10: Ramjet Thrust Coefficient versus Mach Number (Constant 60,000 ft. Altitude).

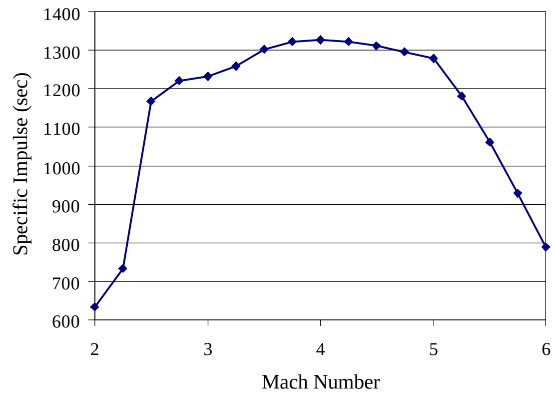


Figure 11: Ramjet Specific Impulse versus Mach Number (Constant 60,000 ft. Altitude).

The second discipline that must provide data is aerodynamics. The workbook needs to know lift and drag coefficients to accurately model the fuel usage. Like the engine data, the aero data is also presented as a function of Mach number and must be interpolated. With these inputs, the workbook discretizes the total flight distance that was previously calculated and applies the range equation at each discreet step. As the vehicle burns fuel and becomes lighter, the vehicle is allowed to ascend until weight equals lift in a “cruise-climb” scenario. Adding up the total fuel burned in each step gives the total amount of fuel burned during the flyback and thus the mass ratio.

Because the optimum settings of Mach number and angle of attack are not know a priori, a grid search is performed on both of these parameters to find the best combination. It is assumed that during the flyback, the Mach number and angle of attack are held constant.

After the trajectory analysis in complete, the booster ascent and flyback mass ratios, booster mixture ratio, and upper stage mass ratio are used in the weights and sizing discipline to determine the vehicle’s weight and overall size. The booster ascent and flyback trajectories are also sent to the aeroheating analysis to determine the TPS requirements for the vehicle.

Aeroheating

The aeroheating analysis for *Starsaber* is performed using two separate tools. The first tool, MINIVER[5], is a thermal analysis code that was written by NASA and performs a 2-D flow analysis over the vehicle. Trajectory information, including angle of attack, altitude, velocity, and sideslip angle as a function of time are input into MINIVER along with the vehicle geometry. MINIVER then models the vehicle using simple geometric shapes and calculates the centerline temperature distributions, convective heat rates, and total heat loads for the simplified vehicle.

Once the MINIVER analysis is completed a Georgia Tech developed tool, TCAT[6] (Thermal Calculation Analysis Tool) is used to determine the type and thickness of thermal protection needed for each section of the vehicle. TCAT allows the analysis of TPS materials from the NASA Ames' TPS-X[7] database and has an internal optimization routine that allows for the calculation of the minimum TPS material thickness required to protect the vehicle substructure.

Several different TPS material types are used to protect the *Starsaber* booster. On the leeward side of the booster, flexible AFRSI blankets are used. Ceramic TUFI tiles protect the majority of the windward side and SHARP material is used on the stagnation point and leading edges of the vehicle. SHARP materials are ultra-high temperature ceramics, such as hafnium diboride, which are under development at NASA Ames as an alternative technology to actively cooled leading edges[6]. Reinforced carbon-carbon (RCC) tiles are used in the nose between the SHARP and TUFI tiles.

Operations

The operations analysis is completed using the enhanced Architectural Assessment Tool (AATe)[8]. This spreadsheet-based, parametric, ground processing operations model was created by NASA KSC. The inputs to AATe are qualitative and quantitative answers to questions regarding the vehicle's attributes. These questions cover the number and type of propellant tanks, TPS material, vehicle size, engine type, etc. The vehicle is then judged using the Space Shuttle as the baseline concept. The results are then compiled into a final quantitative measure of the vehicle operability.

Using the results from the operations analysis, AATe is able to predict the ground operations cost associated with the reusable parts of the vehicle. For the operational cost analysis, it is assumed that the company operating the *Starsaber* vehicle is using a large fictitious spaceport at KSC and is therefore able to share common facilities with other companies.

Economic Analysis

The tool used for the economic analysis of the *Starsaber* vehicle is CABAM[9]. CABAM (Cost and Business Analysis Module) is a spreadsheet tool developed at Georgia Tech that uses parametric cost estimating relationships (CERs) to determine the cost of the launch system. The inputs to CABAM include a weight breakdown of the booster and upper stage, technology and complexity factors, and operations cost numbers.

The economic analysis assumes that *Starsaber* is developed and built as a government asset, but is operated by a fictitious commercial company named Small RLV, Inc. This company operates the *Starsaber* fleet out of a notional spaceport at KSC. This spaceport is used by multiple vehicle operations with a shared staff and facilities. Other assumptions made in the economic analysis are as follows:

- the government pays for all of the design, development, testing and evaluation (DDT&E), fleet acquisition, and facilities expenses.
- the government subcontracts to Small RLV Inc. to operate the vehicle 24 times per year.
- primary labor and other ground operations costs are provided by Small RLV Inc.
- Small RLV Inc. makes a 10% "fee" above the recurring cost of the flight.

Using these assumptions the following are representative results for the *Starsaber* economic analysis. Typical recurring costs per flight ranged from between \$2M – \$2.7M (1999 dollars) depending on the particular configuration being analyzed. Figure 12 shows a sample recurring cost breakdown for *Starsaber*. The liability insurance cost was assumed to be \$100K per launch. The LRU (line replacement unit) hardware cost is the maintenance hardware cost for the booster. Shown in Table 1 is a sample non-recurring cost breakdown for *Starsaber*. The non-recurring costs include both the DDT&E costs and the theoretical first unit costs (TFU). Typically the non-recurring cost ranged from \$1,500M to \$3,000M depending on the specific vehicle design. The values shown in Table 1 are for the lowest non-recurring cost design analyzed.

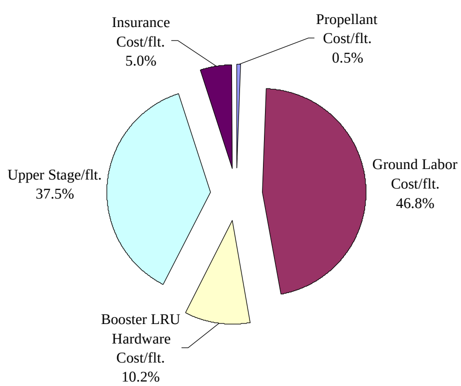


Figure 12: Typical Recurring Cost Breakdown.

Table 1: Typical Non-Recurring Cost (1999 Dollars).

Items	Non-Recurring Cost
DDT&E	\$1,316 M
Booster Airframe	\$1,203 M
Booster Engines	\$85 M
Upper Stage	\$28 M
TFU	\$273 M
Booster Airframe	\$210 M
Booster Engines	\$60 M
Upper Stage	\$3 M
Total Non-Recurring	\$1,589 M

EXPENDABLE UPPER STAGE

The upper stage system chosen for *Starsaber* is an expendable pump-feed LOX/RP-1 system (Figure 13). An expendable option is used because of the low flight rates envisioned for a small payload-class vehicle, and therefore the associated difficultly in recovering the development costs of a reusable upper stage. The cost trends for an expendable upper stage show that the reduction in development and operations/support cost outweigh the cost of the expended hardware per flight.

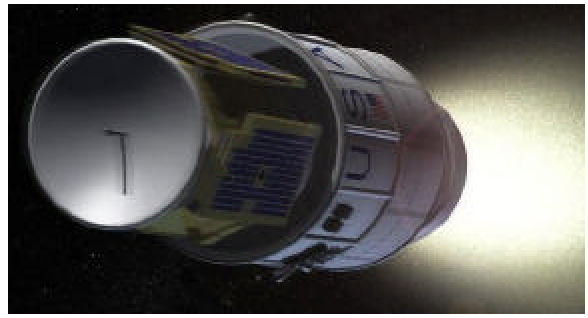


Figure 13: Expendable Upper Stage.

Since the upper stage is expendable the production costs of each stage should be minimized to reduce the expense of each mission. In this regard, a pump-feed LOX/RP-1 gas generator cycle engine was chosen as the propulsion system. This engine operates with a chamber pressure of 650 psia, an area ratio of 50, and a fuel mixture ratio of 2.17. The engine is sized to give the upper stage a vacuum thrust to gross weight at staging ratio of 1.0 and has a vacuum I_{sp} of 328 seconds. A pressure-fed engine was considered as a low cost alternative, but the propellant volume required exceeds the practical limit for a pressure-fed engine system. Also, the large tank weight associated with the pressure-fed system made it impractical for this application. The other components of the upper stage include graphite epoxy tanks and structure and a low cost avionics package.

DESIGN OF EXPERIMENTS

A Taguchi L16 design of experiments array was used to determine the effect of fifteen different design variables on *Starsaber's* overall cost and weight. The fifteen variables chosen for the DoE were selected after consultation between the Space Systems Design Laboratory at Georgia Tech and engineers at NASA's Marshall Space Flight Center. The variables included in this study are:

- the booster's prime structure material
- the fineness ratio of the booster fuselage
- the ratio of the wingspan to body diameter of the booster
- the ejector/ramjet transition Mach number
- the ramjet/rocket transition Mach number
- the staging Mach number
- the initial dynamic pressure boundary for ramjet mode
- the maximum allowable engine static pressure
- the overall vehicle thrust-to-weight ratio at takeoff
- the booster's oxidizer
- the booster's fuel
- the ratio of the booster's engine capture area to fuselage cross sectional area
- the primary rocket maximum chamber pressure
- the maximum expansion ratio for the booster's engines
- and the maximum wing loading

Table 2 is the DoE array used in this analysis and shows the settings used for each variable. As can be seen the variables include mass property/configuration variables, performance (trajectory) variables, aerodynamic variables, as well as variables affecting propulsion. They were chosen because each was seen as a possible important factor in the vehicle's overall weight and cost. The DoE is used to determine the actual importance of each variable and which variable settings give the more desirable solutions.

For each case, the final output metrics evaluated were the vehicle's dry weight, gross weight, non-recurring cost, recurring cost per flight, and recurring price per flight. The dry weight includes only the first stage (booster), while the gross weight includes the upper stage. The recurring price per flight is the price charged by the fictitious company, Small RLV Inc., which is operating the *Starsaber* fleet and includes a 10% "fee" above the recurring cost of the vehicle.

Table 2: *Starsaber* Design of Experiments Array.

Case #	Prime Structure	Fuselage Fineness Ratio (L/D)	Wing span/body diameter (b/D)	Finish Ejector Ramjet Transition (Mach)	Finish Ramjet Rocket Transition (Mach)	Staging Mach Number (Mach)	q boundary at M_i (psf)	Max Engine Static Pressure (psi)	Overall Vehicle T/W at takeoff	Oxidizer	Fuel	Twe Airfs/engine capture area	Primary Rocket max Pc (psi)	Max expansion ratio (Ae/At)	Max Wing Normal Load/Wing Loading	Dry Weight (lb)	Gross Weight (lb)	Non Recurring Cost ($M)	Recurring Cost/Flight ($M)	Recurring Price/Flight ($M)
1	Gr/Ep	6	4.5	3	7	9.5	1200	300	0.6	H2O2	Propane	0.7	2000	1.5	3	22,050	106,100	1,610	2.38	2.62
2	Gr/Ep	8	4.5	3.5	5.5	14	1500	150	0.8	LOX	JP	0.7	2000	1.5	3	34,850	212,200	2,185	2.12	2.33
3	Gr/Ep	8	5.5	3	5.5	14	1500	150	0.6	H2O2	Propane	0.9	1000	2	3	57,600	394,550	2,735	2.17	2.39
4	Gr/Ep	6	5.5	3.5	7	9.5	1200	300	0.8	LOX	JP	0.9	1200	2	3	34,600	168,050	2,160	2.52	2.77
5	Gr/Ep	10	4.5	3	5.5	14	1000	300	0.6	LOX	Propane	0.9	1000	1.5	1.5	30,350	181,850	2,045	2.10	2.31
6	Gr/Ep	10	4.5	3.5	7	9.5	1500	150	0.6	H2O2	JP	0.9	1000	1.5	1.5	22,150	134,000	1,705	2.45	2.70
7	Gr/Ep	10	5.5	3	7	9.5	1500	150	0.8	LOX	Propane	0.7	2000	2	1.5	24,[illegible]	110,[illegible]	1,8[illegible]	2.43	2.74
8	Gr/Ep	10	5.5	3.5	5.5	14	1500	300	0.5	H2O2	JP	0.7	2000	2	1.5	41,500	291,450	2,290	2.11	2.52
9	Ti-Al	6	4.5	3	5.5	9.5	1500	300	0.8	H2O2	JP	0.9	2000	2	1.5	10,500	67,200	1,700	2.41	2.65
10	Ti-Al	8	4.5	3.5	7	14	1000	150	0.8	LOX	Propane	0.9	2000	2	1.5	54,700	412,150	2,790	2.28	2.51
11	Ti-Al	8	5.5	3	7	14	1000	150	0.5	H2O2	JP	0.7	1000	1.5	1.5	62,900	474,000	2,950	2.21	2.44
12	Ti-Al	6	5.5	3.5	5.5	9.5	1200	300	0.8	LOX	Propane	0.7	1200	1.5	1.5	20,250	67,900	1,760	2.42	2.66
13	Ti-Al	10	4.5	3	7	14	1500	300	0.5	LOX	JP	0.7	1000	2	3	41,200	332,300	2,415	2.20	2.42
14	Ti-Al	10	4.5	3.5	5.5	9.5	1000	150	0.6	H2O2	Propane	0.7	1000	2	3	16,050	77,050	1,545	2.40	2.64
15	Ti-Al	10	5.5	3	5.5	9.5	1000	150	0.8	LOX	JP	0.9	2000	1.5	3	17,[illegible]	76,[illegible]	1,7[illegible]	2.51	2.77
16	Ti-Al	10	5.5	3.5	7	14	1500	300	0.8	H2O2	Propane	0.9	2000	1.5	3	40,[illegible]	346,[illegible]	2,540	2.13	2.34

The design of experiments array presented above is composed of sixteen different *Starsaber* designs. For each of these designs, the iterative design process described earlier was used to determine the vehicle properties. For every case, a converged design was reached using the design structure matrix shown in Figures 3 & 4. The average design time for each vehicle ranged from 3 to 5 days, depending on the difficultly associated with the different variable combinations.

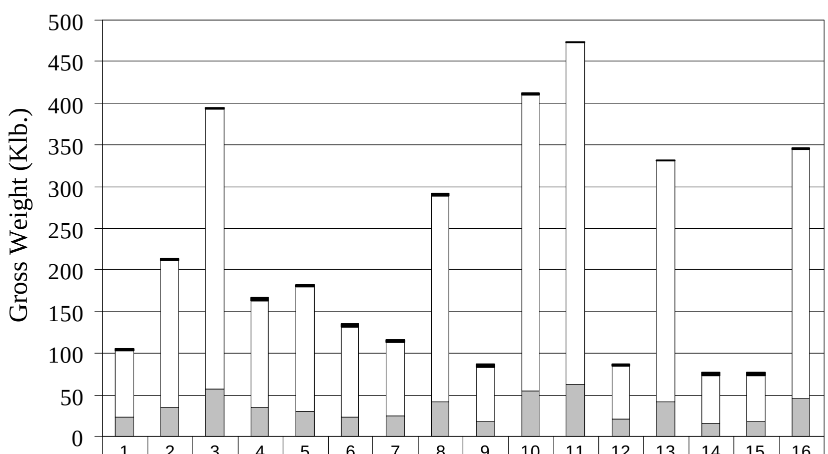
From the DoE array, the analysis of the mean technique (ANOM) was used to determine the effect of each variable on the five performance metrics. Also, the variable settings producing the lowest weight and recurring cost vehicles were predicted. It should be noted that the two-level DoE used for this experiment cannot capture any quadratic effects of the design variables on the metrics. A two-level array gives either the high or low settings of each variable as the "near optimum" answer. For a more detailed determination of the optimal variable settings, the results from this 2-level analysis could be used as a screening test and the most important and influential variables could be included in a 3-level DoE.

RESULTS

Detailed weight breakdowns and economic analysis are available for each vehicle, however, for this paper only the overall vehicle gross and dry weights, recurring cost per flight and non-recurring costs will be presented. The gross weight comparison for each case of the DoE is shown in Figure 15.

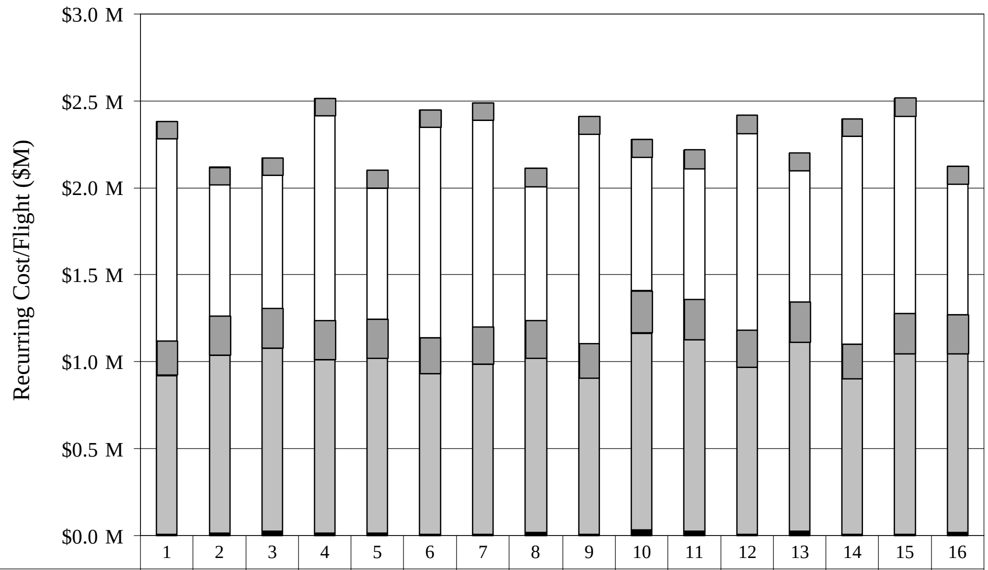
As can be seen from Figure 15 the gross weight varies greatly depending on the vehicle design. The cases with the higher staging Mach numbers are the highest weight vehicles, while the lower staging Mach number vehicles tend to be the lightest. The lower recurring cost cases have the opposite trend (Figure 16). The higher staging Mach number vehicles have the smallest upper stages and therefore the recurring cost is less, because a large factor in the recurring cost is the cost of a new expendable upper stage.

After the analysis of all sixteen cases in the DoE was complete, it was possible to determine the effect of each variable on the performance metrics and to determine which variable settings yield the lowest cost and lightest gross and dry weight versions of *Starsaber*.



	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Upper Stage (Klb.)	3.4	1.7	1.8	3.5	1.7	3.6	3.5	1.8	3.6	1.8	1.7	3.3	1.7	3.6	3.3	1.7
Propellant (Klb.)	79	176	335	128	150	108	88	248	65	356	409	64	289	57	55	299
Dry Weight (Klb.)	24	35	58	35	30	23	25	42	18	55	63	20	41	16	18	46

Figure 15: Gross Weight Breakdown.



	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Insurance Cost/flt. ($M)	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10	0.10
Upper Stage/flt. ($M)	1.17	0.76	0.77	1.19	0.76	1.22	1.19	0.77	1.20	0.77	0.76	1.14	0.76	1.20	1.14	0.76
LRU Hardware Cost/flt. ($M)	0.20	0.22	0.23	0.22	0.22	0.20	0.21	0.22	0.20	0.24	0.23	0.21	0.23	0.20	0.23	0.22
Labor Cost/flt. ($M)	0.91	1.02	1.05	1.00	1.01	0.93	0.98	1.00	0.90	1.13	1.09	0.96	1.09	0.90	1.04	1.03
Propellant Cost/flt. ($M)	0.01	0.02	0.02	0.01	0.01	0.01	0.01	0.02	0.01	0.03	0.03	0.01	0.03	0.00	0.01	0.02

DOE Case #

Figure 16: Recurring Cost Breakdown.

Pareto diagrams are used to graphically show what variables influence the metrics the most. According to the Pareto principle, 20% of the variables will account for 80% of the effect on the given metric[10]. This principle allows a Pareto chart to be used as a screening test to determine which variables should be included in a more detailed analysis. The Pareto charts for gross liftoff weight and recurring cost per flight are shown below in Figures 17 and 18.

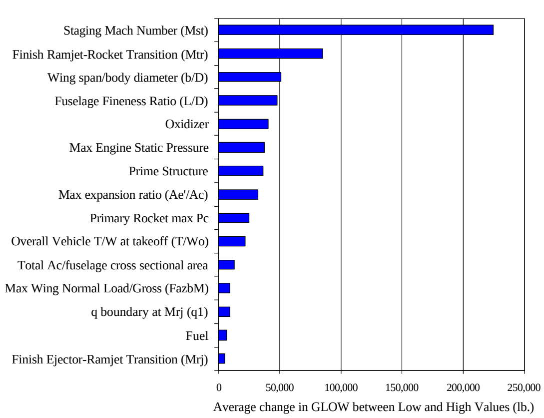


Figure 17: Pareto Chart for Gross Liftoff Weight.

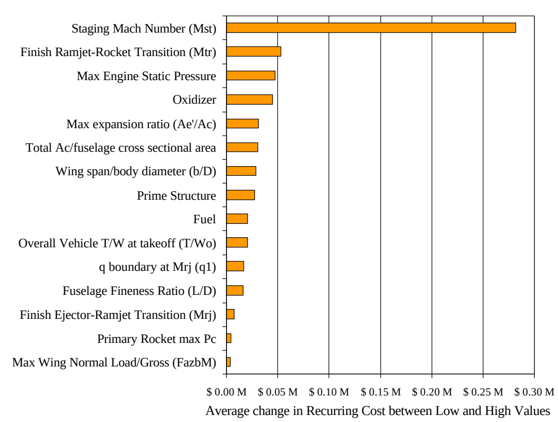


Figure 18: Pareto Chart for Recurring Cost per Flight.

The dominance of a variable was determined by finding the difference between the averaged value of the metric for all cases in the DoE where the variable was at its lowest setting and for all the cases where the variable was at its highest setting. The charts above show that the most dominant variable is the staging Mach number, followed by the ramjet/rocket transition Mach number.

The ANOM tables used to generate the Pareto charts were also used to determine the settings for the lowest recurring cost, lowest gross weight, and lowest dry weight vehicle designs. The specific settings determined by the ANOM tables were not one of the sixteen previously run cases, so additional designs where converged using the same iterative design process discussed earlier. The ANOM analysis predicted the same variable settings when the gross and dry weights were used as the optimization variable. Therefore only two new vehicle designs where required. The variable settings and results are shown in Figure 19.

	Lowest Recurring Cost Solution	Lightest Weight Solution
Prime Structure	Gr/Ep	Gr/Ep
Fuselage Fineness Ratio (L/D)	10	10
Wing Span/Body Diameter (b/D)	4.5	4.5
Finish Ejector-Ramjet Transition (Mrj)	3.5	3.5
Finish Ramjet-Rocket Transition (Mtr)	5.5	5.5
Staging Mach Number (Mst)	14*	8.5*
q boundary at Mrj (psf)	1500	1500
Max Engine Static Pressure (psi)	300	300
Overall Vehicle T/W at Takeoff (T/W_o)	0.9	0.9
Oxidizer	H2O2*	LOX*
Fuel	Propane	Propane
Total Ac/Fuselage Cross Sectional Area	0.7	0.7
Primary Rocket Max Pc (psi)	2000	2000
Max Expansion Ratio (Ae'/Ac)	1.5	1.5
Max Wing Normal Load/Gross (FazbM)	3	3
Dry Weight (lb)	26,250	17,375
Gross Weight (lb)	168,250	71,950
Non-Recurring Cost ($M)	1,890	1,580
Recurring Cost/Flight ($M)	2.01	2.43
Recurring Price/Flight ($M)	2.21	2.67

*setting changed

Figure 19: “Optimized” Vehicle Designs.

The lightest gross/dry weight vehicle has the lower staging Mach number and uses LOX because of the increased engine performance. The lower staging Mach number gives a lighter vehicle because the booster can be much smaller. The lowest recurring cost vehicle has the higher staging Mach number, which gives a smaller and cheaper expendable upper stage. It also uses hydrogen peroxide (H_2O_2) for an oxidizer because of the operational cost benefits of using a non-cryogenic oxidizer.

The lowest recurring cost vehicle did in fact have a lower recurring cost than all sixteen of the original cases. The vehicle optimized for weight has a lower GLOW than all the original cases, but it has the second lowest dry weight. The reason for this is the selection of graphite epoxy (Gr/Ep) instead of titanium aluminide (Ti-Al) as the prime structure. If the prime structure for the lowest weight solution is switched to Ti-Al and the vehicle is re-converged, the dry weight decreases to below 16,000 lbs. making it lighter than the sixteen original cases. The statistical reason for the selection of Gr/Ep over Ti-Al by the ANOM analysis can be explained by the effect of confounding. Confounding is when one variable in the DoE statistically affects another. In this case, the prime structure column appears to be confounded by the interaction between the staging Mach number and rocket transition Mach number columns. For the Ti-Al cases, instances of the high staging Mach number occur with the high ramjet-rocket transition Mach number while the high staging Mach number for the Gr/Ep cases occurred with the low ramjet-rocket transition Mach number. The combination of high staging and ramjet-rocket transition Mach numbers leads to a large booster stage and therefore a heavier vehicle. Since the majority of the heavy vehicles occurred when the material selection was Ti-Al, the ANOM analysis chose Gr-Ep. However, even with this confoundment issue, the results from the DoE analysis are very encouraging and give vehicles with significantly lower recurring cost and gross liftoff weight when compared to the initial reference designs considered by the team. The use of a DoE and ANOM analysis allowed the team to analyze only a small fraction of the 2^{15} possible design combinations, but still obtain near-optimal results for both vehicle cost and weight.

TRADE STUDIES

The variable with the largest influence on the vehicle cost and weight is the staging Mach number. Since this variable is so influential to the results, an additional trade study was conducted to further analyze its influence. Figure 20 shows the effects of staging Mach number on the non-recurring and recurring costs of the lowest recurring cost vehicle. To generate this graph, the fourteen DoE variables besides the staging Mach number where left at their lowest recurring cost settings. Then a sweep of staging Mach numbers was performed.

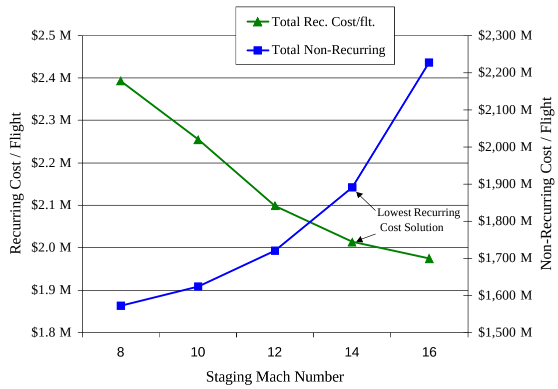


Figure 20: Effect of Staging Mach Number on Cost.

Figure 20 shows that increasing the staging Mach number further reduces the recurring cost of the system. This is because a higher staging Mach number results in an even smaller expendable upper stage. This trend will continue until the flyback fuel costs and booster maintenance costs become dominant and start driving the recurring cost up again. The reduction in recurring cost is at the expense of the non-recurring cost. Higher staging Mach numbers give a larger booster and therefore the development and production costs of the vehicle increase dramatically.

SUMMARY

Starsaber is a TSTO horizontal takeoff vehicle sized to place a 300 lb. payload into LEO. A systems engineering method utilizing a Taguchi L16 design of experiments array was used to examine system variable interactions and the sensitivities of performance metrics to these variables. Two "optimal" *Starsaber* designs where determined using ANOM techniques. The first was designed for lowest recurring cost and the second was designed for lowest gross/dry weight. Within the ranges examined, each of these vehicles showed a preference for higher dynamic pressure trajectories, high chamber pressure rocket primaries, high fineness ratio fuselages, high allowable internal engine pressures, and high takeoff thrust-to-weight ratios. There were several key differences between the two vehicles. The lowest recurring cost vehicle preferred an H_2O_2/Propane fuel-oxidizer combination, while the lowest weight vehicle preferred LOX/Propane. A JP fueled vehicle was not preferred by either solution for the evaluation criteria considered. The lower recurring cost vehicle has a higher staging Mach number (Mach 14), while the lightest weight vehicle has a lower staging Mach number (Mach 8.5). This result should be expected because the upper stage of *Starsaber* is expendable, and the high staging Mach number vehicle throws away a smaller, cheaper upper stage. A trade study was performed on staging Mach number because of its dominant influence on the performance metrics. It was found that a further increase in staging Mach number resulted in a lower recurring cost vehicle, but this is at the expense of vehicle weight and non-recurring cost.

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