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Hyperion: An SSTO Vision Vehicle Concept Utilizing Rocket-Based Combined Cycle Propulsion

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ABSTRACT

This paper reports the findings of a conceptual launch vehicle design study performed by members of the Space Systems Design Laboratory at Georgia Tech. *Hyperion* is a conceptual design for an advanced reusable launch vehicle in the Vision Vehicle class. It is a horizontal takeoff, horizontal landing single-stage-to-orbit (SSTO) vehicle utilizing LOX/LH2 ejector scramjet rocket-based combined cycle (RBCC) propulsion. *Hyperion* is designed to deliver 20,000 lb. to low earth orbit from Kennedy Space Center. Gross weight is estimated to be 800,700 lb. and dry weight is estimated to be 123,250 lb. for this mission. Preliminary analysis suggests that, with sufficient launch traffic, *Hyperion* recurring launch costs will be under $200 per lb. of payload delivered to low earth orbit. However, non-recurring costs including development cost and acquisition of three airframes, is expected to be nearly $10.7B. The internal rate of return is only expected to be 8.24%.

Details of the concept design including external and internal configuration, mass properties, engine performance, trajectory analysis, aeroheating results, and concept cost assessment are given. Highlights of the distributed, collaborative design approach and a summary of trade study results are also provided.

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NOMENCLATURE

C_t	thrust coefficient
I_{sp}	specific impulse (sec.)
I*	equivalent trajectory averaged I_{sp} (sec.)
MR	mass ratio (gross weight/burnout weight)
q	dynamic pressure (psf)
T/W_e	installed engine thrust-to-weight

INTRODUCTION

NASA Marshall Space Flight Center is currently conducting a ground test program to evaluate rocket-based combined cycle (RBCC) engines. These multi-mode engines combine the best aspects of rocket propulsion (high thrust-to-weight) and airbreathing propulsion (high I_{sp}). Previous research has shown that vehicles utilizing RBCC propulsion are attractive candidates for future space launch missions.

As part of its Advanced Reusable Technologies[1] program, NASA solicited advanced RBCC vehicle designs from several aerospace contractors. These ‘Vision Vehicle’ designs were groundruled to be single-stage, LOX/LH2, ejector scramjet vehicles. Preference was given to horizontal takeoff designs. Georgia Tech’s *Hyperion* design was created to serve as an independent assessment of this class of vehicle, albeit at a lower payload (20 klb to LEO vs. 25 klb to Space Station). As with the larger Vision Vehicles, the primary objective of the *Hyperion* concept design project is to determine whether RBCC propulsion and other advanced technologies can be used to produce a vehicle that could significantly reduce the cost of space access. A preliminary version of *Hyperion* was entered as a candidate in NASA’s recent Highly Reusable Space Transportation (HRST) study[2] and was shown to produce attractive recurring cost benefits.

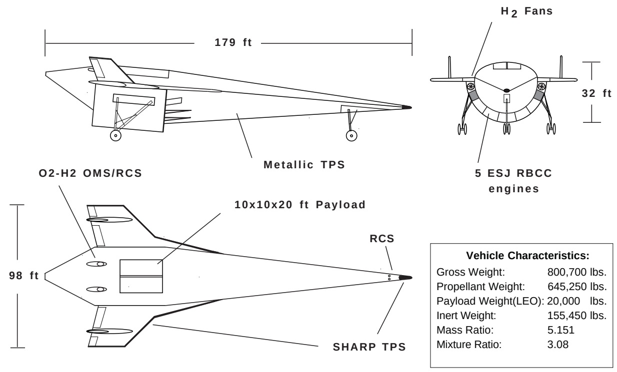

Figure 1. *Hyperion* Concept Configuration.

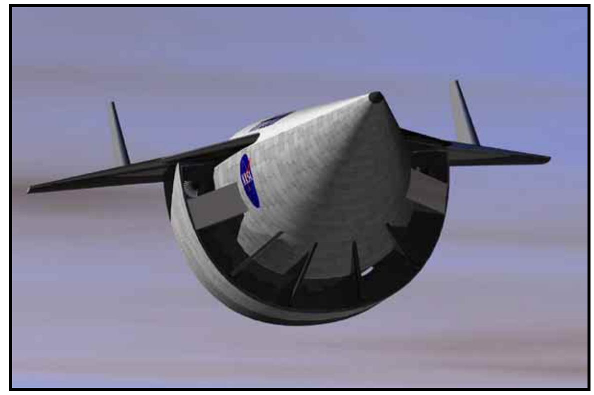

Figure 3. *Hyperion* Loiter with Ducted Fans.

THE HYPERION CONCEPT

Concept Overview

Hyperion (Figure 1 and Figure 2) is a single-stage vehicle with a conical forebody, highly swept wings, and twin vertical winglets. It is powered by five LOX/LH2 ejector scramjet RBCC engines mounted on its undersurface. These engines provide the primary motive power to accelerate the vehicle into orbit. The baseline concept is designed to deliver 20,000 lbs. of payload into a 100 nmi. x 28.5° circular orbit from Kennedy Space Center (KSC). Entry is unpowered, but 5 minutes of loiter and go-around capability is provided by two H2 low-thrust ducted fans, one under each wing (Figure 3).

Hyperion uses a number of advanced technologies in addition to the ejector scramjet engines. Lightweight metal matrix composites such as titanium-aluminides are used for the primary structure and wings. Graphite composites are used to construct the main propellant tanks. To avoid active cooling, the high temperature nosecap, wing leading edges, and cowl leading edges are constructed of Ultra-High Temperature Ceramic (UHTC) TPS. Metallic TPS tiles and high temperature TABI blankets are used to protect the acreage areas of the vehicle. In addition, lightweight power, avionics, and electromechanical surface control actuators are used. The vehicle is capable of autonomous operation and thus has no pilots. Initial operational capability is expected to be in the year 2010 – 2015. Each airframe is assumed to have been designed for long life operation (estimated to by 1000 flights per airframe and 500 flights per engine).

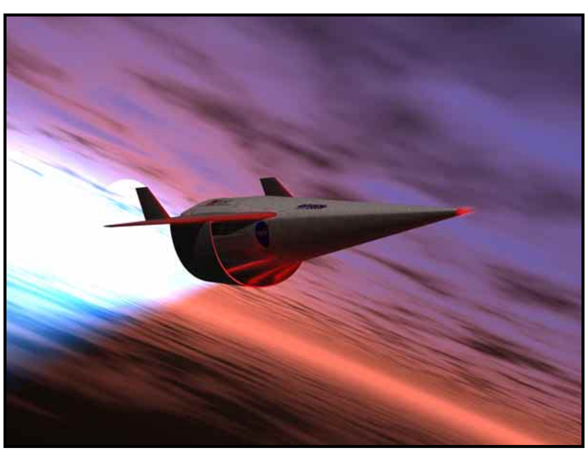

Figure 2. *Hyperion* RBCC Ascent.

Mission Profile

Hyperion operates from a notional airfield at KSC. Initial take-off thrust is provided by the ejector mode of the RBCC engines. The vehicle is designed for a thrust-to-weight ratio of 0.6 at horizontal takeoff. Ejector mode is used to accelerate the vehicle onto a 2000 psf dynamic pressure boundary at Mach 3 where the RBCC operating mode is shifted to ramjet operation (increase in I_{sp}, decrease in thrust). The vehicle smoothly shifts to scramjet mode around Mach 5.5, and continues to accelerate to Mach 10. At this point, the vehicle uses its internal rocket-mode to finish its climb to LEO. The payload is released (Figure 4), and the vehicle is de-orbited for the return to KSC. Once in the vicinity of KSC, the ducted fans are uncovered to provide up to 5 minutes of high

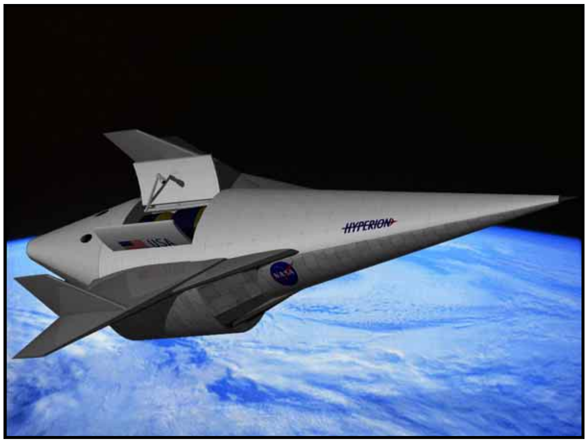

Figure 4. *Hyperion* On-Orbit.

efficiency loiter. *Hyperion* is designed to carry a crew module of 6 astronauts or a cargo module. The baseline design is capable of delivering approximately 11,100 lb. to Space Station orbit or 20,000 lb. to low earth orbit (LEO).

COLLABORATIVE DESIGN PROCESS

Hyperion was designed using a collaborative, team-oriented approach at the Space Systems Design Laboratory at Georgia Tech. An integrated design team of disciplinary experts was assembled. Each team member used a conceptual design tool to conduct his or her engineering analysis in a highly coupled and iterative concept convergence process similar to that described in reference 3. Table 1 lists the represented engineering disciplines and the conceptual design tools used by analysts in each.

Table 1. Disciplinary Representation.

Discipline	Analysis Tool
Aerodynamics	APAS (UDP, HABP)
CAD and Layout	SDRC I-DEAS
RBCC Propulsion	SCCREAM
Trajectory Optimization	POST 3-D
Aeroheating/TPS	MINIVER/TPS-X
Weights & Sizing	in-house spreadsheet
Ground Operations	AATe
Cost and Economics	CABAM

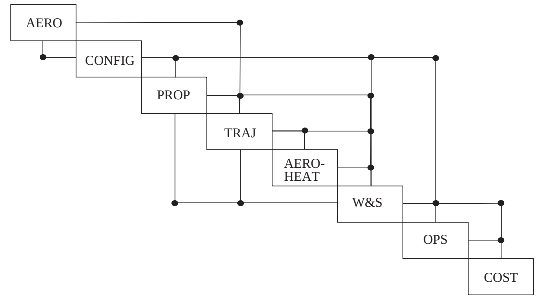

Figure 5. *Hyperion* Design Structure Matrix.

Data was exchanged between the team members according to the coupling links in the *Hyperion* Design Structure Matrix (DSM, Figure 5). In the DSM, the data links above the diagonal represent feed forward data from one analyst to a subsequent analyst. Feedback links below the diagonal represent iteration loops for which an initial guess must be made and then iteration performed to converge the results of the two disciplines. For example, a strong iteration loop is present between propulsion, performance (trajectory optimization), and mass properties (weights & sizing). As the vehicle size and capture area changes, the engine performance must be updated and the trajectory re-optimized. During the conceptual design process, the convergence tolerance was taken to be a change of less than 0.1% in gross weight between iterations.

BASELINE DESIGN RESULTS

Aerodynamics

The external fuselage configuration of *Hyperion* consists of a 9.0° half-angle cone on the lower surface and an elliptical cross section upper surface. The tail was shaped to provide a large expansion surface for scramjet and all-rocket modes of operation. Using APAS[4], the wings were positioned and sized to provide static stability at hypersonic and landing conditions (with flaps). In addition, the wing area was sized to limit the landing speed to under 200 knots. Actual landing speed was estimated to be 145 knots.

For the baseline configuration, a theoretical wing planform area of 5,900 ft^2 was required (extending into the fuselage, but discounting strake area). The leading edge sweep of the outboard wing section is 65°. The theoretical aspect ratio of the wing is 1.2 and the taper ratio is 0.25. The wing airfoil is a 5% thick biconvex airfoil with a small leading edge radius to reduce wave drag. The vertical tipfin controllers were sized to have a planform area of 2.5% of the wing theoretical area. These tipfins are used for active lateral control and were not sized to provide static stability in yaw.

An aerodynamic database consisting of tables of lift and drag coefficients were generated across the ascent trajectory speed regime using APAS. At each Mach number and altitude pair of interest, analysis was performed for a range of angles-of-attack. These data tables were provided to the trajectory analyst. Subsequent vehicle scaling was done photographically and the aerodynamic coefficients were assumed to remain nearly constant during scaling. Aerodynamic analysis was therefore only required at the start of the design process. Note that in the force accounting system used, all forebody pressures were included as aerodynamic drag and the propulsive force was taken to be from the front of the cowl to the tail of the vehicle (cowl-to-tail system).

Internal Configuration & Layout (CAD)

Propellant tanks were packaged in the fuselage of the vehicle using SDRC-IDEAS, a solid modeling CAD program. At the final LOX/LH2 mixture ratio of 3.08 (by weight), *Hyperion* is dominated by internal LH2 tanks containing normal boiling point liquid hydrogen. As shown in Figure 6, a transparent view of the fuselage, aftbody volumes are occupied by LH2 tanks. These tanks are partially integral, that is, they share a common wall with the airframe where possible. Two additional cylindrical LH2 tanks flank the centerbody. A 10 ft. x 10 ft. x 20 ft. cargo bay was reserved for the 20,000 payload. A single "belly" LOX tank holds the required liquid oxygen. Two separate LH2 tanks along sides of the center body hold dedicated fuel for loiter operations using the ducted fans.

One of the key outputs of the packaging discipline is the fraction of total internal fuselage volume that is occupied by ascent propellants (propellant packaging efficiency, PEF). Since the tank configuration changes slightly with vehicle scale (payload volume is fixed), three different internal layouts were created — one each at three different vehicle scales. A 1-D curve was created to allow interpolation between the points on the curve. For the final, converged baseline *Hyperion* design, the vehicle length was 179 ft. tip-to-tail and the PEF was 72.5%.

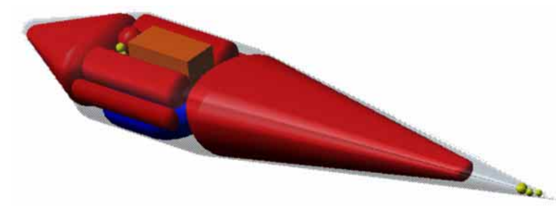

Figure 6. *Hyperion* Internal Layout CAD Model.

Propulsion

The propulsion system analysis was performed using the 'Simulated Combined Cycle Rocket Engine Analysis Module' (SCCREAM).[5] SCCREAM is a one-dimensional analysis code that is capable of analyzing all modes of RBCC engine operation. The final output from SCCREAM is an engine deck preformatted for use in a trajectory simulation program. This engine deck includes engine thrust, thrust coefficient, and I_{sp} for a range of altitudes and Mach numbers for each operating mode.

Hyperion uses five liquid oxygen and hydrogen ejector scramjet (ESJ) engines to inject the vehicle into a 50 x 100 nmi. interim transfer orbit. Figure 7 shows the engine layout and station identifications used by SCCREAM. The engines were mounted on the lower half of the vehicle, which provided 9° of forebody compression. An engine cowl height of 4.6 feet was determined based on a Mach 10 shock-on-lip condition for a conical bow shock. Each engine has an average width of 10.0 feet. A variable inlet geometry and exit nozzle were assumed.

A LOX/H2 rocket primary with a chamber pressure of 2,000 psi and an ejector mode mixture ratio of 8.0 was selected. The engines were sized at sea-level-static (SLS) conditions to meet the vehicles' overall takeoff thrust-to-weight ratio of 0.6. Each engine is thus capable of producing 96,100 lbs. of thrust at SLS, with an I_{sp} of 389 seconds. For *Hyperion*, the secondary-to-primary flow ratio at SLS was 2.3.

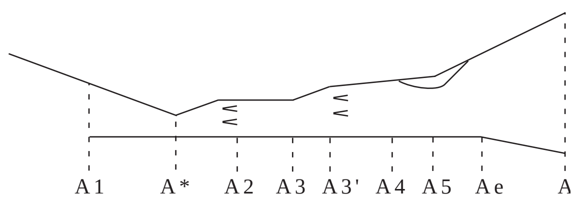

Figure 7. *Hyperion* ESJ Engine Configuration.

Table 2 provides the internal engine geometry values and fuel injection properties for a single *Hyperion* engine. With a minimum internal contraction ratio of 2.1, the lowest possible Mach number at which the inlet could start for ramjet operation was predicted to be Mach 3.0.

Figure 8 shows the net specific impulse versus Mach number during ejector mode operation. Between Mach 2.5 and 3.0, transition to ramjet mode is modeled by linearly throttling the ejector mode down while the ramjet mode is ramped up.

Figure 9 shows the net thrust coefficient (C_t) versus Mach number for ramjet and scramjet mode operation for a single engine. To obtain the thrust coefficient, the thrust was normalized by the dynamic pressure (q) and inlet area of 46.0 ft^2. Note that the propulsion force accounting system in SCCREAM is cowl-to-tail. All forebody pressures are included in aerodynamic drag calculated by APAS. Forebody calculations are performed in SCCREAM to determine mass capture at various flight conditions, but the pre-compression effects are not used to reduce the cowl-to-tail thrust coefficients and I_{sp}'s.

Table 2. *Hyperion* ESJ Engine Data.

inlet area, A_1	46.0 ft^2
primary throat, A_t	3.97 ft^2
mixer area, A_3	25.56 ft^2
combustor break, $A_{3'}$	38.3 ft^2
combustor exit, A_4	38.3 ft^2
maximum exit area, $A_{e'}$	184.0 ft^2
combustor efficiency, η_c	95.0%
mixer efficiency, η_m	90.0%
nozzle efficiency, η_{nozz}	98.0%

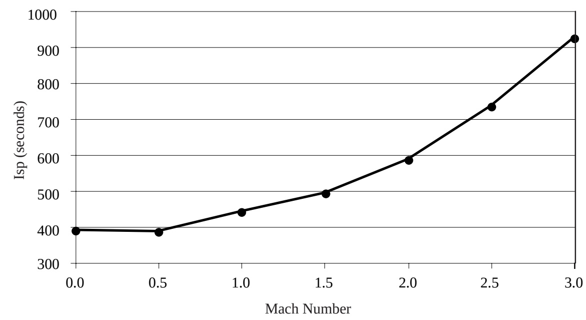

Figure 8. Ejector Mode Net Specific Impulse.

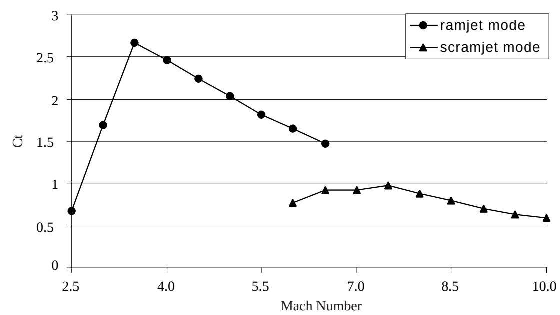

Figure 9. Thrust Coefficient vs. Mach Number.

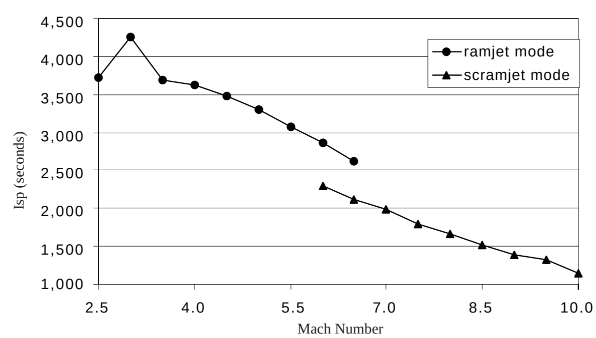

Figure 10. Net I_{sp} vs. Mach Number.

Evident in Figure 9 is the significant increase in performance due to the inlet starting at Mach 3. Figure 10 shows the net specific impulse in ramjet and scramjet modes. When operating in all-rocket mode after Mach 10, each of Hyperion's engines can generate 116,600 lbs. of thrust, at a vacuum Isp of 455 seconds. The rocket performance calculations use the same rocket primary subsystem from the ejector mode, operating with an assumed expansion ratio of 180 and a more optimal rocket-mode mixture ratio of 7.0.

Performance (Trajectory Optimization)

The trajectory analysis was performed by the three degree-of-freedom version of the Program to Optimize Simulated Trajectories—POST[6]. POST is a Lockheed Martin and NASA code that is widely used for trajectory optimization problems in advanced vehicle design. It is a generalized event-oriented code that numerically integrates the equations of motion of a flight vehicle given definitions of aerodynamic coefficients, propulsion system characteristics, weight models, etc. Numerical optimization is used to satisfy trajectory constraints and minimize a user-defined objective function.

The trajectory for *Hyperion* is constrained by a dynamic pressure boundary that provides optimal RBCC performance, by changes in pitch rates that provide smooth ejector and rocket pull-ups, and by orbital termination criteria. The dynamic pressure boundary that is flown is 2000 psf during the ramjet and scramjet modes between Mach 3 and Mach 9 at which point the vehicle begins to pull up. Transition to all-rocket mode is complete by Mach 10. The q boundary is constrained through implementation of a linear feedback control guidance scheme in which the dynamic pressure is held by controlling angle-of-attack.[7] *Hyperion* flies to a 50 nmi. x 100 nmi. x 28.5˚ parking orbit. A separate OMS propulsion system is used to circularize the orbit at 100 nmi. and then later deorbit the vehicle. The baseline LOX/LH2 OMS is designed to deliver 350 fps of on-orbit ΔV.

Figure 11 shows a graph of Mach number and altitude vs. time. A plot of the dynamic pressure as a function of Mach number is given in Figure 12. The 2000 psf boundary can clearly be seen in the figure. *Hyperion's* dynamic pressure is not exactly on the boundary at Mach 3, but it is within an acceptable tolerance. The linear feedback control algorithm quickly guides *Hyperion* to the boundary. In this portion of the trajectory, the angle-of-attack is allowed to vary within a range of 0˚ and 10˚. The angle-of-attack profile for the entire trajectory can be seen in Figure 13; the dynamic pressure is held between ~195 seconds and ~500 seconds. The Mach number transitions between the four engine modes (ejector, Mach 0 – Mach 2.5; ramjet, Mach 3 – Mach 5.5; scramjet, Mach 6 – Mach 9; and rocket, Mach 10 – orbit insertion) are modeled as a linear ramp down of the preceding mode and a linear ramp up of the following mode.

The objective of the trajectory is to maximize the final weight, or burnout weight. For the converged

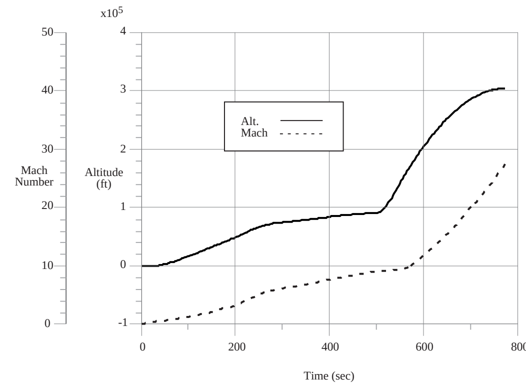

Figure 11. Mach & Altitude vs. Time.

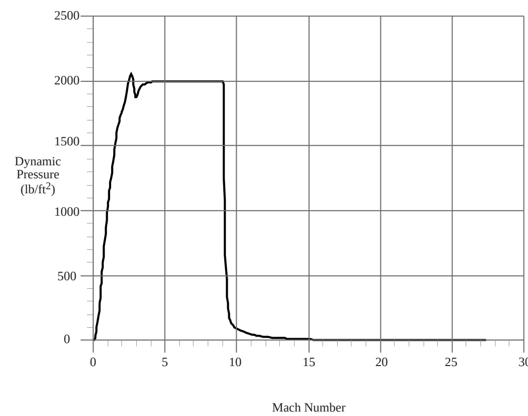

Figure 12. Dynamic Pressure vs. Mach.

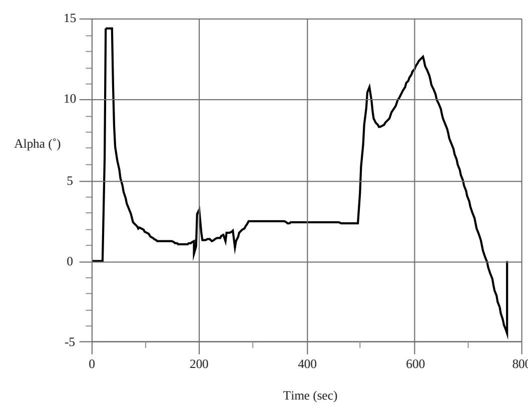

Figure 13. Angle-of-Attack Profile.

baseline, the Mass Ratio (MR) of the ascent was determined to be 5.151. The ideal ascent ΔV provided by the propulsion system is 33,105 fps, including 7,110 fps of drag losses (measured inertially). Therefore the I* for the ascent is estimated to be 464 sec. accounting for all losses. For the baseline 20,000 lb. payload mission, the required LOX/LH2 mixture ratio was determined to be 3.08.

Aerothermal Analysis

The thermal protection materials and unit weights for *Hyperion* were scaled from a previous RBCC SSTO design flying a similar trajectory and using similar technologies[8]. On that earlier vehicle, MINIVER[9] was used to estimate surface heating rates, heat loads, and radiation equilibrium temperatures along a 2000 psf q boundary trajectory.

For *Hyperion,* a metallic (Inconel surface) standoff tile system is baselined for the high temperature windward side of the fuselage. A lightweight blanket system, TABI, is used for the leeward fuselage surface. Since the exposed wing is constructed of a high-temperature titanium-aluminide, large sections of the wing are designed to be hot structure. To avoid the complexities of active cooling, an ultra-high temperature ceramic (UHTC) is employed on the small radius nosetip and wing leading edges. This material is being developed by NASA – Ames and is capable of withstanding temperatures as high as 4,500° F. Additional information about the various types of advanced thermal protection system (TPS) materials selected can be found at reference 10.

Mass Properties

A spreadsheet model consisting of approximately 75 parametric mass estimating relationships (MERs) was created to estimate the weight and size of the converged *Hyperion* vehicle. For example, MERs were included that estimate the wing weight based on surface area and wing loading. The propellant tank MER was based on design pressure, materials, and internal volume. These MERs have a NASA Langley heritage, but were adjusted to account for advanced materials technologies, construction techniques, and lightweight subsystems. The Georgia Tech WATES[11] tool was used estimate the installed T/W_e of the ejector scramjet engine given its geometry and operating conditions. For the converged baseline, the installed T/W_e was estimated to be 28.8.

Given a MR (or propellant mass fraction) and a mixture ratio requirement from the trajectory optimization discipline, the spreadsheet was used to scale the vehicle up or down until the available MR matched that required. Changing PEF and engine T/W_e were also accounted for during this process. Once the vehicle was “closed” within the Weights & Sizing discipline, the results were sent back to the Propulsion discipline to iterate several times around the Propulsion – Trajectory – Aeroheating - Weights loop shown in the DSM in Fig. 5. This entire process was repeated until the gross weight was converged to within 0.1%.

The baseline *Hyperion* design has a gross weight of 800,700 lb. and a dry weight of 123,250 lb. Fuselage length is 179 ft. from tip to tail. Figure 14 shows a graphical breakout of the largest contributors to dry weight. Table 3 lists selected summary items from the weight breakdown structure (WBS). The full WBS is not included in this paper for brevity, but includes 28 major headings with several subcategories under each. A 15% overall dry weight growth margin

Table 3. *Hyperion* Top-Level Weight Statement.

WBS Item	Weight
Wing & Tail Group	19,200 lb.
Body Group	28,150 lb.
Thermal Protection System	7,600 lb.
Main Propulsion (includes ESJ)	20,750 lb.
OMS/RCS Propulsion	2,500 lb.
Subsystems & Other Dry Weights	28,950 lb.
Dry Weight Margin (15%)	16,100 lb.
Dry Weight	123,250 lb.
Payload to LEO	20,000 lb.
Other Inert Weights	12,200 lb.
Insertion Weight	155,450 lb.
LH2 Ascent Propellant	142,350 lb.
LOX Ascent Propellant	502,900 lb.
Gross Weight	800,700 lb.

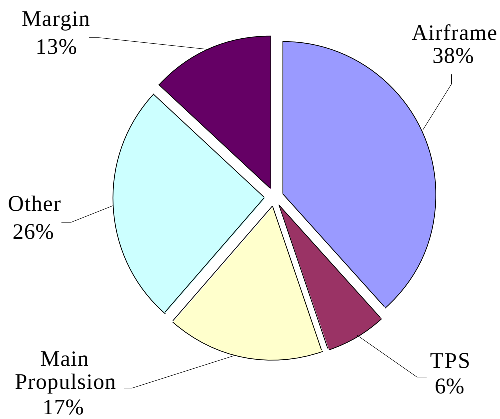

Figure 14. *Hyperion* Dry Weight Breakdown.

was included to account for the likelihood of weight increases.

Operations

Hyperion is designed to be a highly operable and highly reusable space transportation system. "Design for Operability" guidance was provided by the operability team of NASA's HRST[2] study and the AATe spreadsheet tool developed at KSC[12]. Technologies such as vehicle health monitoring and built-in test equipment are included in the design to make checkout and inspection easier, and therefore reduce turnaround time and labor costs. Long life and very reliable airframe components (1000 flights before replacement) and engine components (500 flights before replacement) reduce scheduled maintenance actions and lower inventory costs. The use of toxic fluids such as hypergols has been avoided. The LOX and LH2 propellants are both normal boiling point liquids (no slush LH2). Electro-mechanical actuators are used in place of hydraulics to reduce maintenance costs. The two ducted fans make it possible to taxi and even self-ferry *Hyperion* (with no payload or LOX). The complexities of requiring a separate transport aircraft are avoided.

Hyperion is assumed to be operated by a commercial company using a future spaceport and runway at Kennedy Space Center. The spaceport infrastructure is assumed to be a shared asset provided by the federal or local government similar to today's airports. Spaceplane operators pay a user's fee per flight, but are not required to build the spaceport or perform runway maintenance, etc. An estimated streamlined operations crew of only 450 personnel are required to operate a fleet of three *Hyperion* vehicles. The fleet was assumed to be capable of flying up to 150 – 175 flight per year (turnaround times of less than 1 week per airframe). The spaceport user's fee was estimated to be $50,000 per launch.

ECONOMIC AND COST ANALYSIS

After the *Hyperion* vehicle configuration was determined, a conceptual assessment of its development cost, production costs, fleet size, operational costs, and even its potential revenue stream was determined. This assessment was made using Georgia Tech's CABAM cost and business modeling spreadsheet.[13]

CABAM (Cost and Business Analysis Module) was developed at Georgia Tech in response to the need to have a tool that provides a financial assessment of conceptual launch vehicle design. This tool incorporates not only the cost attributes associated with a project, but also identifies the potential revenue streams and projects a number of evaluation metrics such as net present value, internal rate of return, return on investment, etc.

CABAM uses data from the NASA Commercial Space Transportation Study[14] (CSTS) and user entered competition models to approximate the price elastic behavior of potential markets. The 'medium' market growth models from the CSTS study was used for the baseline, but the nuclear waste disposal market was not included. For conservatism, all cargo traffic from the CSTS model was assumed to be destined for the International Space Station (ISS) orbit. In addition, a 15% penalty for incompatibilities between multiple manifested payloads was assumed.

The goal of the present research was to identify the optimum pricing strategy that results in maximum internal rate of return (IRR). IRR is defined as the discount rate for a certain project that results in a $0 net present value. Neglecting risk, higher IRR's are better.

Hyperion Business Model

Hyperion will be operated by a private business, RLV Inc., with the government assisting in the initial development of the launch service. The U.S. government is a very heavy user of launch services and launch cost reductions will ultimately benefit the taxpayers. Therefore, the government was assumed to pay for 100% of the RBCC engine non-recurring development cost (DDT&E) and 20% of the Hyperion airframe DDT&E. In addition, the government also guaranteed commercial debt loans made to RLV Inc. so that financing could be obtained at a reduced interest rate (10%). All airframe and engine production costs as well as all operations and financing costs are borne by RLV Inc.

The economic environment used in this analysis consisted of an inflation rate of 3%, tax rate of 30%, and discount rate of 20%. A relatively high hurdle rate of 20% is chosen to account for the risky nature of this project. The program starts in 1999 with a projected initial operating capability (IOC) in 2011 with termination in 2025. A 20% cost margin was added to both DDT&E and theoretical first unit (TFU) costs.

Economic Results Optimized for IRR

The optimized business scenario resulted an IRR of 8.24% with a fleet size of three *Hyperion* vehicles, 450 personnel in the company, and a total steady state flight rate of 146 flights per year (106 commercial cargo, 27 government cargo, 8 commercial passenger flights/year, and 7 government astronaut flights to each market). RLV Inc. operates for 15 steady state years after a two year ramp up and flies a total of about 2,471 flights. The venture is predicted to break even two years after initial operations begin with a total Life Cycle Cost (LCC) for the program of $19.69B (99$) with an initial debt-to-equity ratio of 3. Non-recurring costs (DDT&E, engine and airframe production, but not financing costs) of the entire venture is estimated to be $10.68B (99$) of which the U.S. government is expected to contribute $1.45B (99$). Figure 15 shows the non-recurring cost distribution for *Hyperion*.

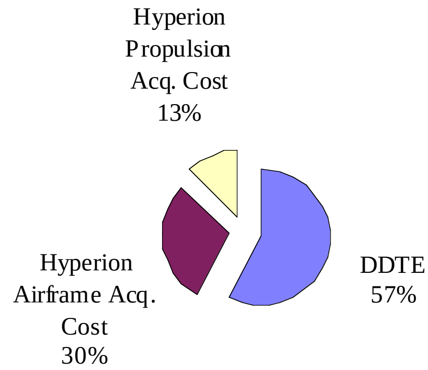

Figure 15. Non-Recurring Cost Breakdown.

Specific market price results are given in Table 4 in 1999 constant year dollars. Recall that the prices per pound of cargo reflect an ISS destination. A commercial cargo price of $800/lb. would therefore generate about $7.55M in revenue per flight (accounting for the reduced average *Hyperion* payload capacity to ISS of 11,100 lb.). Note that the less price elastic government traffic models result in a higher optimized market price for government missions compared to commercial missions. That is, the size of the government launch market is relatively constant over a wide range of prices, so the IRR optimization tends toward a higher price.

Table 4 - Optimized CSTS Market Prices for *Hyperion*

Market	Price	Traffic
Comm. Cargo	$800/lb.	1,000 klb/yr.
Comm. Pass.	$0.62M/pass.	46 pass./yr.
Gov't Cargo	$1,845/lb.	250 klb/yr.
Gov't Pass.	$8.27M/pass.	39 pass./yr.

Relative to current expendable launch vehicle prices in this class, the optimized market prices represent only about a factor of five decrease in price for commercial payloads and a factor of two decrease in price for government payloads. The reductions are more significant with respect to the Space Shuttle, but dramatic multiple orders of magnitude decreases in access to space costs do not appear likely given the current models and assumptions if the proposed company is to achieve an attractive rate of return for its investors. It should also be noted that an IRR of 8.24% appears unattractive for such a new launch vehicle venture with its myriad of risks. Investors and company decision-makers might demand a return as high as 35% or more for a program of this type with significant risk and uncertainty.

Hyperion Recurring Costs Per Flight

For *Hyperion*, aggressive assumptions were made to determine recurring costs. For this study, recurring costs were assumed to be the sum of the following four items (in 1999 dollars): 1) labor costs at a \$100,000/yr. encumbered rate per employee, 2) Line replaceable unit spares at 0.10% of airframe weight replaced per flight at an average cost of \$10,000/lb. of hardware, 3) propellant costs at \$0.10/lb of LOX and \$0.25/lb of LH2 based on the assumption of an on-site propellant production facility, and 4) insurance costs were \$50,000/launch. This insurance is for limited liability coverage for the vehicle.

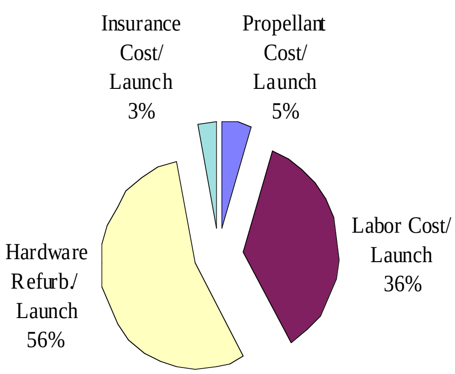

Figure 16. Recurring Cost Breakdown.

Based on these assumptions and the 146 flight annual rate, each flight of *Hyperion* is estimated to cost \$1.64M (1999\$). A recurring cost breakdown is given in Figure 16. For a typical Space Station cargo delivery mission with an average actual payload delivered of 11,100 lb., the recurring cost per pound of payload is \$148/lb. The cost per pound for delivering payloads (20,000 lb) to a 100 nmi. due east orbit is potentially lower. It is important to note that this is a somewhat artificial value. *Hyperion* customers pay the optimized launch price in Table 4, not the recurring cost. The price includes recurring costs, amortized hardware and design costs, financing costs, and company profit and thus is several times higher.

TRADE STUDY

The airbreathing-to-rocket transition Mach number was selected for investigation. The baseline vehicle was designed for a maximum A/B Mach number of 10, with scramjet-to-rocket transition beginning at Mach 9 (recall that a one Mach number transition was used between modes). Two alternate transition cases were examined. They were beginning transition at Mach 8, obtaining all-rocket mode operation at Mach 9, and transitioning at Mach 10, with all-rocket operation by Mach 11. For each case considered, the cowl height was adjusted so that the shock-on-lip condition was obtained at the maximum airbreathing Mach number.

Figure 17 shows the trade study results. It is apparent that the gross weight was not affected significantly by the transition Mach number within the ranges tested. The lower transition case (Mach 9) GLOW weight was slightly higher due to the accompanying higher mixture ratio and increased LOX load. The high Mach number case (Mach 11) had a lower mixture ratio, but this GLOW benefit was partially lost because the overall vehicle became larger.

Hyperion's dry weight is shown to decrease significantly at the lower transition Mach number. The vehicle benefits from the lower transition number from both lower drag losses and higher mixture ratio (denser vehicle). Additionally, the inlet area can be larger due

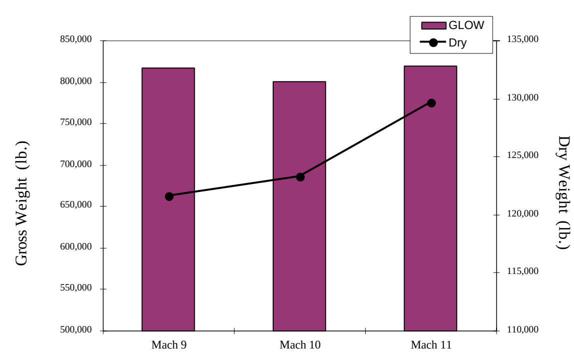

Figure 17. Trade Study Results

to the lower shock-on-lip constraint. This improves ramjet and scramjet thrust at the lower Mach numbers and results in a significantly lower dry weight. The results of this trade study are still being assessed to determine if the baseline flight profile for *Hyperion* should be changed. Specifically, the impacts on the economic viability of the concept need to be examined.

SUMMARY

A new airbreathing SSTO concept based on RBCC propulsion has been presented. *Hyperion* (Figure 18) is second generation RLV designed to deliver 20,000 lb. to low earth orbit. Advanced propulsion, materials, and systems technologies are used throughout the vehicle. A collaborative, team-oriented design process was used to perform the conceptual design. For the baseline mission, gross weight was determined to 800,700 lb. and dry weight was 123,250 lb.

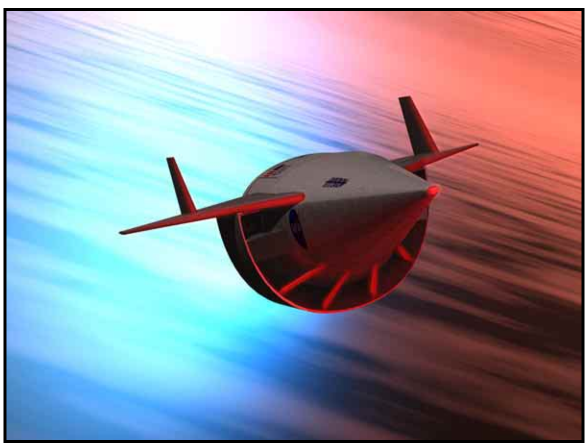

Figure 18. *Hyperion* Concept.

Economic results were somewhat disappointing. Even with optimistic assumptions regarding government investment and using price elastic CSTS markets, the maximum IRR of 8.24% of *Hyperion* still cannot compete with the average Standard and Poor's (S&P) annual compounded return of over 12%. While recurring cost per pound of payload is shown to be less than $200/lb., the optimum price that must be charged to potential customers to maximize IRR represents only a factor of 4 – 5 improvement over current launch prices. Without more government assistance to lower investment hurdles (i.e. offset non-recurring costs) and a higher demand for overall launch services (i.e. higher flight rates), the probability of achieving a commercial, economically viable *Hyperion* vehicle is low.

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