

StarRunner: A Single-Stage-to-Orbit, Airbreathing, Hypersonic Propulsion System

Patrick Biltgen^{*}, Jarret Lafleur^{*}, Josh Loughman^{*}, Robert Martin^{*},
Kevin Flaherty^{*}, Min Cho^{*}, Keith Becker^{*}, Chester Ong^{*}, John R. Olds[†]

*School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA 30332-0150*

ABSTRACT

In response to the request for proposal (RFP) for the 2003 AIAA Undergraduate Team Engine Design Competition, the FAS Propulsion Design team from the Georgia Institute of Technology presents *StarRunner: A Single-Stage-to-Orbit (SSTO), Airbreathing, Hypersonic Propulsion System*. Low-cost, highly reliable access to low-Earth orbit (LEO) and the International Space Station (ISS) is an area of continuing research and debate. *StarRunner* is proposed to supplement a notional Crew Transfer Vehicle through the ability to deliver a 25,000 lb payload to the ISS. The horizontal takeoff/horizontal landing (HTHL) vehicle makes use of a turbine-based combined cycle (TBCC) propulsion system consisting of 14 low-bypass-ratio turbofan engines and a dual-mode ramjet/scramjet propulsion system for high-speed flight. The vehicle also takes advantage of ultra-high-temperature ceramic thermal protection materials and uses hydrogen fuel for regenerative cooling of engine components. *StarRunner* is compatible with standard runways, with a gross takeoff weight of approximately 1,000,000 lbs, and has a cost per pound to orbit of approximately \$825/lb. This advanced, fully reusable space transport vehicle and integrated propulsion system design demonstrates student efforts to understand issues facing the space launch community. Future enabling and enhancing technologies for TBCC SSTO launch vehicles are explored and analyzed. The final *StarRunner* design addresses and proposes several innovative solutions to traditional problems.

INTRODUCTION

Safe, inexpensive, and reliable access to space are the primary goals of every space launch system. With the impending retirement of Space Shuttle in 2010, a need has been identified for a highly-reusable vehicle capable of transporting 25,000 lbs of cargo to and from the International Space Station (ISS). To this end, the 2002-2003 AIAA Undergraduate Team Engine Design Competition sought a vehicle capable of performing this mission with an entry-into-service (EIS) date of 2020 and utilizing the latest technology advances in hypersonic airbreathing propulsion. This design competition was offered by the airbreathing propulsion technical committee; however, the integrated nature of the airframe and propulsion system required that both the engines and vehicle be designed and sized to operate together. Although the design required extensive analysis in aerodynamics, structures, thermal protection systems, and rocket engine design, this paper primarily addresses the systems and technologies required to develop a feasible airbreathing propulsion system for access-to-space launch vehicles.

The *StarRunner* design was carried out by a team of eight undergraduate students from the Georgia Institute of Technology under the direction of Dr. John Olds and Mr. William J. Escher. The team, composed of five freshmen, a junior, and two seniors, combined classroom knowledge with Georgia Tech

^{*} Undergraduate Student, School of Aerospace Engineering, Student Member AIAA.

[†] Associate Professor, School of Aerospace Engineering, Associate Fellow AIAA.

vehicle and propulsion systems design methodologies to develop a feasible system that satisfied the requirements set forth by the AIAA Airbreathing Propulsion Technical Committee. In August 2003, *StarRunner* was selected as the first-place design in the 2002/2003 AIAA Undergraduate Team Engine Design Competition¹.

NOMENCLATURE

ACES	Air Collection and Enrichment System	NEPP	NASA Engine Performance Program
AFRSI	Advanced Flexible Reusable Surface Insulation	RBCC	Rocket Based Combined Cycle
APAS	Aerodynamic Preliminary Analysis System	RFDU	Rotational Fractional Distillation Unit
ASDL	Aerospace Systems Design Laboratory	RFP	Request for Proposal
CABAM	Cost and Business Analysis Module	SCCREAM	Simulated Combined-Cycle Rocket Engine Analysis Module
DDT&E	Design, Development, Test and Engineering	SSDL	Space Systems Design Laboratory
EIS	Entry-into-Service	SSTO	Single-Stage-to-Orbit
GLOW	Gross Liftoff Weight	TBCC	Turbine-Based Combined Cycle
HTHL	Horizontal Takeoff/Horizontal Landing	TFU	Theoretical First Unit
IHPTET	Integrated High Performance Turbine Engine Technology	TPS	Thermal Protection Systems
IRR	Internal Rate of Return	TRL	Technology Readiness Level
ISS	International Space Station	TUFI	Toughened Uni-Piece Fibrous Insulation
LEO	Low-Earth Orbit	UEET	Ultra-Efficient Engine Technology
LH2	Liquid Hydrogen	UHTC	Ultra-High Temperature Ceramics
LOX	Liquid Oxygen	VAATE	Versatile Affordable Advanced Turbine Engines
NASP	National Aerospace Plane	WATE	Weight Analysis of Turbine Engines

CONCEPT OVERVIEW

StarRunner is a single-stage-to-orbit (SSTO) vehicle powered primarily by airbreathing propulsion systems. Its 14 low-bypass ratio turbofan engines and seven dual-mode ramjet/scramjet engines are powered by hydrogen fuel. A LOX/LH2 linear aerospike tail rocket provides the necessary ΔV for the transition from hypersonic flight at Mach 10.5 to orbital velocity. The structure of the vehicle includes non-integral graphite epoxy propellant tanks and utilizes Titanium Aluminide (Ti-Al) for the wings, tails, and a majority of the internal structure. The two-dimensional forebody uses four fixed ramps to provide inlet compression while an actively cooled movable inlet cowl translates vertically to provide maximum capture area and shock-on-lip conditions at the design Mach number of 10. To reduce gross liftoff weight (GLOW) and reduce the number of turbofan engines required for takeoff and transonic operation, an innovative Air Collection and Enrichment System (ACES) is used to liquefy atmospheric oxygen to fill the rocket propellant tanks while cruising under ramjet power at low equivalence ratios for just under one hour. Also, a takeoff sled is utilized to support the GLOW of the vehicle and reduce the weight of the main landing gear to only accommodate a normal landing or emergency return-to-launch-site landing. These two systems allow *StarRunner* to operate using existing HTHL architectures. The converged vehicle is 257 ft long and weighs roughly 1,000,000 lbs at liftoff. The launch price for commercial and government customers is approximately \$825/lb.

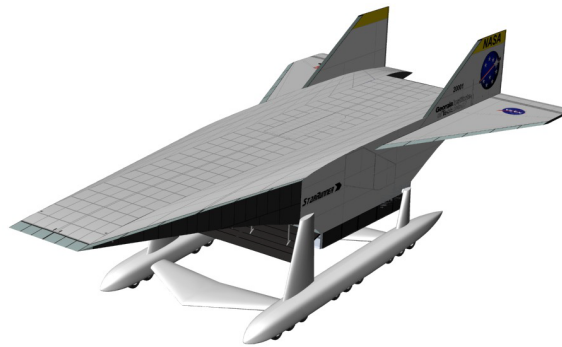


Figure 1. StarRunner Mounted on the Takeoff Assist Sled.

VEHICLE CONFIGURATION

StarRunner's overall shape is dictated by the need to fly and accelerate in the hypersonic regime. The entire lower surface can be considered part of the engine, making up a virtual inlet and exhaust nozzle. As with all spacecraft, internal space is at a premium in this design: Hundreds of thousands of pounds of fuel must be carried, but an airframe that is too large will have excessive weight and drag. A three-view inboard profile of *StarRunner* is shown in Figure 2.

Three non-integral hydrogen tanks with a tank pressure of approximately 25 psia, shaped to the vehicle contours, supply fuel to the turbofan, ramjet, scramjet, and rocket propulsion systems. Combined, the tanks hold 669,000 lbs of normal boiling point liquid hydrogen. The two oxygen tanks, with a pressure of approximately 38 psia, are shaped in a more desirable "pill capsule" configuration, and are located on either side of the 40' x 15' x 15' payload bay. The tanks are empty at liftoff and are filled by the ACES with 859,000 lbs of 98% pure liquid oxygen during the 60 minute Mach 3 cruise under ramjet power.

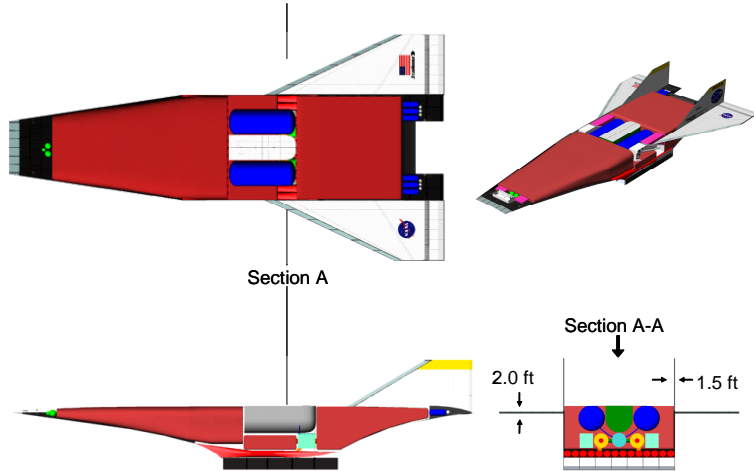


Figure 2. *StarRunner* Inboard Profile.

StarRunner uses three airbreathing propulsion systems in addition to a linear aerospike tail rocket. The initial part of the flight from Mach 0 to 3 is powered by 14 hydrogen-burning low-bypass-ratio turbofan engines located beneath the center hydrogen tank. A dual-mode ramjet/scramjet engine is located beneath the turbofan flowpath. The ramjet operates from Mach 3 to 6, where the engine internal static pressure begins to approach material limits and the scramjet is engaged. The four-ramp inlet features a curved section that folds open to increase the capture area, and hence mass flow rate, for the low-speed portion of the flight. The turbofan engines are exhausted along the aftbody of the vehicle, which also functions as an expansion surface for the ramjet/scramjet. The space between the turbojet engines and the payload bay is filled with the oxygen liquefaction equipment for the ACES. The primary rocket engine, ignited at Mach 10.5 after scramjet cutoff, is based on the RS-2200 Linear Aerospike developed for the *VentureStar* SSTO vehicle². Each engine produces 495,000 lbs of thrust with an I_{sp} of 455 seconds (vacuum) and a mixture ratio of 6:1. Three of these engines are used on *StarRunner*, and they are mounted in a cutout on the aft fuselage. The propulsion systems comprise 38% of the 376,000 lb empty weight of the vehicle. Although the inclusion of turbofan engines and the ACES dramatically increase the empty weight of *StarRunner*, the turbofan engines provide an efficient means to accelerate the vehicle to Mach 3. The ACES removes the requirement to carry 859,000 lbs of liquid oxygen from the ground, reducing GLOW. Increases in GLOW dramatically increase the number of turbofan engines required to accelerate through Mach 1. The hex-airfoil wing of the *StarRunner* is made primarily of Ti-Al metal matrix composite and the wing carry-through structure is reinforced with Silicon Carbide (SiC) fibers. The dual horizontal tails are similarly constructed and all control surfaces use electromechanical actuators to eliminate the complexity and weight of hydraulic systems. Electrical power for the vehicle and ACES is provided by the turbofan engines and auxiliary power units. The sizes of the wing and tail are based on historical guidelines from similar vehicles³. Aerodynamic coefficients for this configuration are determined across the flight regime using the Aerodynamic Preliminary Analysis Software (APAS).

As indicated in Figure 1, a wheeled sled is used in place of a takeoff gear to reduce GLOW by nearly 100,000 lbs. The main landing gear, located on the outer sides of the oxygen tanks, swings aft into position.

This configuration reduces the weight of the main gear since it is designed to accommodate a landing with the 25,000 lb payload and reserve propellants only. The drawback to this configuration is that extensive propellant venting must take place in the event of a return-to-launch-site abort. The nose gear folds horizontally into the forebody and is shorter than the overall height of the vehicle, causing the nose gear touchdown to put *StarRunner* into a negative angle of attack as used on the Space Shuttle. The landing gear weight is further reduced by the elimination of retraction mechanisms. A summary weight statement is shown below in Table I.

Table I: Summary Weight Statement.

Element	Element Weight (lb)	Total Weight (lb)
Wing and Tail	11,479	
Body and Tankage	139,725	
Thermal Protection	23,727	
Landing Gear	10,551	
Rocket Engines	25,667	
ACES weight	32,056	
Ramjet/Scramjet	24,000	
Turbofan Propulsion	36,400	
Misc. Systems	23,983	
Dry Weight Margin	49,138	
Dry Weight		376,726
Cargo	25,000	
RCS/OMS/Landing Prop (incl. reserves)	43,198	
Ascent LH2	556,353	
Takeoff with ACES		1,001,275
Ascent LOX (Filled In-Flight)	848,250	

PROPULSION SYSTEM DESIGN

The design of the turbofan engines utilized in the low speed portion of the flight is conducted using the NASA Engine Performance Program (NEPP). This FORTRAN computer program performs one dimensional, steady-state thermodynamic performance of aircraft gas turbine or jet engine configurations⁴.

The Weight Analysis of Turbine Engines (WATE) computer code was developed by Boeing Military Airplane Development with support from the Garrett Corporation as a subprogram to NASA's thermodynamic cycle analysis code, NEPP. The goal of the WATE code is to provide "representative" propulsion system weight and dimension estimates for conceptual design analyses. A conceptual flowpath diagram of the engine is also provided. For the analysis performed in this study, NEPP and WATE are linked to perform simultaneous calculations, allowing the designer to change performance parameters and predict not only the engine thrust and fuel flow, but also the engine diameter and total engine weight, which are critical to the vehicle sizing.

The engine selected for *StarRunner* is a loosely based on the F110-GE-132 low-bypass-ratio turbofan. The baseline engine configuration produces approximately 32,000 lbs of thrust at sea level with afterburner. Applying modifications to account for technology advances expected by the year 2020, the maximum thrust is increased to 40,000 lbs, fuel burn is decreased, and engine weight is reduced. Although the RFP for the competition allowed a dual-fuel system, hydrogen was selected due to its high I_{sp} and potential for engine and airframe cooling. It was determined that a Kerosene/JP-based vehicle would weigh about the same, but fuel cooling would be difficult beyond Mach 6. Since hydrogen fuel was required for the scramjet and rocket, all engine systems use the same fuel from cross-fed tanks.

StarRunner's engine, the KT40, is designed with electric engine concepts in mind. Vehicle systems are driven by electrical power generated by the KT40, and the engine takes advantage of new technologies such as magnetic bearings to reduce the need for exotic and complex lubricants. The important design parameters of the KT40 engine are shown in Table II.

Table II: Design Values for the NASA Engine Performance Program (NEPP).

Fan Pressure Ratio	3.6	Rotor Inlet Temp (T41)	3000° F
Fan Efficiency	90%	Reheat Temp (Afterburner)	3400° F
Design Bypass Ratio	0.40	Extraction Ratio	0.85
Compressor Pressure Ratio	8	Turbine Blade Max Metal Temp.	1800 °F
Compressor Efficiency	91%	Turbine Vane Max Metal Temp.	1900 °F
Burner Efficiency	98%	Total Uninstalled Engine Weight	2600 lbs
HPT Efficiency	90.5%	Overall Diameter	4 ft
LPT Efficiency	91.5%	Overall Length	11.83 ft

The design parameters represent those consistent with upper-bound predictions for a 2020 EIS advanced gas turbine engine. The rotor inlet temperature and afterburner temperature rely on the application of advanced turbine and combustion chamber alloys in development under the UEET, IHPTET, and VAATE initiatives. The WATE conceptual flowpath of the KT40 engine is shown in Figure 3. The weight breakdown of the KT40 is shown below in Table III. These weights are calculated using the WATE model, and technology k-factors are applied to represent advanced technologies such as compressor blisks which cannot be modeled using the WATE software.

Table III: Component Weight Summary of the KT40 Engine.

Component	Weight (lbs)	Component	Weight (lbs)
Fan	904	Afterburner	236
Compressor	357	Nozzle	296
Combustor	57	Accessories	100
High Pressure Turbine	218	Misc. Cases/Ducts	200
Low Pressure Turbine	232	Total Uninstalled Weight	2,600

The above table illustrates the uninstalled weight of the KT40 engine. The thrust/weight ratio of this engine is approximately 15, and is considered extremely aggressive. Although an engine with this thrust/weight is optimistic by today's standards, a fully airbreathing system that does not rely on rocket power for a thrust assist through the transonic drag rise will require small and extremely powerful engines. The VAATE program, sponsored by the Air Force Office of Special Research, seeks to increase the performance to cost ratio by a factor of 10 by 2020. The success of an airbreathing propulsion access-to-space vehicle such as *StarRunner* relies on government funded advances in propulsion technologies to advance the state of the art, contributing to a very high thrust/weight ratio and reduced fuel consumption.

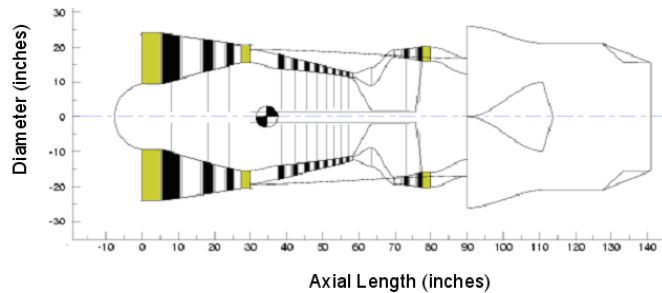


Figure 3. *StarRunner* KT40 Turbofan Engine.

The high-speed flight regime is characterized by four phases. Initially, *StarRunner* flies at an altitude of 45,000 feet at Mach 3 for a ramjet-powered cruise while the ACES equipment liquefies and stores the 859,000 lbs of liquid oxygen required. The vehicle then ascends along the RFP-recommended 2,000 psf dynamic pressure trajectory to Mach 6 at approximately 75,000 feet where the ramjet transitions to scramjet mode. This engine operates along this trajectory to approximately Mach 10.5, where the vehicle pulls off the 2,000 psf dynamic pressure trajectory and ascends at constant Mach number to 115,000 feet where the rocket is ignited. Ramjet/scramjet performance is determined using SCCREAM⁵, which requires engine geometry based inputs to compute thermodynamic performance for the calculation of thrust and I_{sp} . To ensure that the engine design is optimized over the flight path, the iSIGHT⁶ integration and optimization software package is utilized. A Windows command-line version of SCCREAM is integrated into the

iSIGHT graphical environment and the input variables for SCCREAM such as the area ratios at each engine station and the position and angle of fuel injectors are identified as parameters for optimization. The objective function for the optimization algorithm is either Thrust or I_{sp} , depending on the flight regime. The trajectory code was executed with thrust-optimized and I_{sp} -optimized engine configurations, and the design team selected the option that minimizes GLOW while also minimizing the amount of movable geometry required for the ramjet/scramjet system. The optimization results are shown in Figure 4. The lower surface is represented by the dotted red line (Mach 3-6) and moves as Mach number increases to the dashed green position (Mach 10). The upper surface of the engine and bottom of the vehicle are shown in the solid red line. While the ramjet is in operation from Mach 3-6, the engine is optimized to be in the full-open position. The ramjet throat, labeled as station “*” moves up and down to minimize spillage drag and maintain a normal shock at the inlet to avoid an “unstarted inlet” condition.

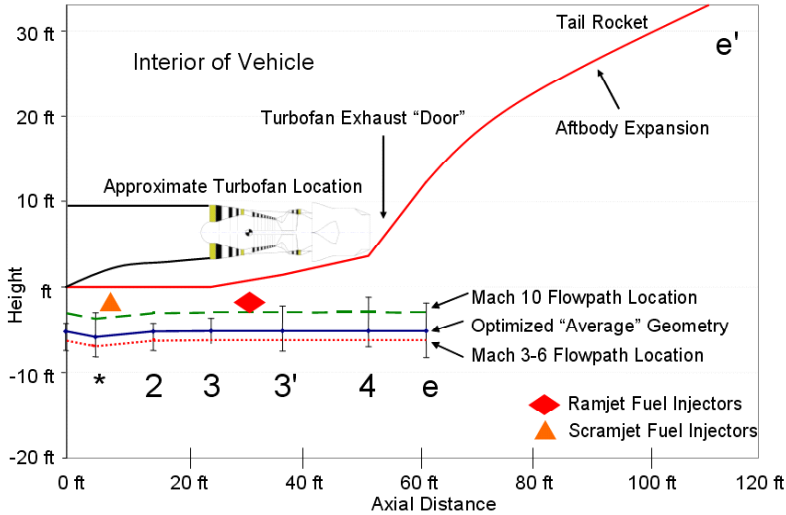


Figure 4. Optimized Ramjet/Scramjet Geometry.

The ramjet fuel injectors are located just prior to station 3' and consist of an array of burners similar to the afterburner on the F-15 or F-16. As the Mach number increases from 6 to 10 and the scramjet is activated, the lower surface of the engine is raised, decreasing the capture area and moving toward a shock-on-lip condition at Mach 10. At this time, the ramjet injector array is retracted into the upper surface of the engine and a series of swept-ramp fuel injectors near station “*” are activated to promote mixing due to the short length of the engine and the supersonic speed of the incoming air in scramjet mode. To reduce the amount of variable geometry in the high-temperature aft section of the ramjet engine, a thermal choke technique, maintained by a digital engine control system is used in place of a fixed converging/diverging nozzle. The extra complexity of removing this nozzle in scramjet mode heavily penalized the airbreathing system.

The linear aerospike rocket engine, which ignites after the vehicle reaches Mach 10.5, uses liquid hydrogen and liquid oxygen as the propellant combination. The vacuum I_{sp} of the engine is 455 seconds, and the combined thrust of the three engines is approximately 1.5 million lbs. The engine is located at the end of the expansion ramp and is protected from the ramjet/scramjet exhaust with retractable TPS-coated doors. The actuators for the doors allow them to close again in the vacuum of space to protect the rocket engines during reentry.

TRAJECTORY OPTIMIZATION

Due to the extremely integrated nature of the engine and vehicle in the *StarRunner* design, a rapid and efficient trajectory code that easily integrates with an engine performance deck was required. Several commercial legacy codes such as POST and OTIS were examined for their feasibility; however, due to the complexity of the ACES flight mode, a separate 600-line trajectory code was written in MATLAB for this design study. The details of the code operation during the ACES cruise are discussed in a subsequent section.

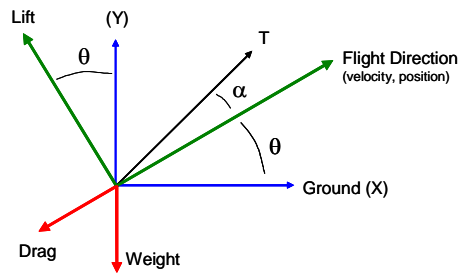


Figure 5. Force Balance in Trajectory Code.

The simple code is based upon the force balance in each Cartesian direction as shown in Figure 5:

$$\sum F_x = T \cos(\theta + \alpha) - D \cos(\theta) - L \sin(\theta) \quad (1)$$

$$\sum F_y = T \sin(\theta + \alpha) - D \sin(\theta) + L \cos(\theta) - W \quad (2)$$

Where T is the thrust of the vehicle, W is the current vehicle weight, and L and D are the lift and drag calculated by APAS. The flight path angle, θ , and angle of attack, α , are optimized along the flight path to converge the trajectory code with a minimum GLOW as the engine performance parameters are perturbed. The focus of the trajectory code is the high-fidelity modeling of the physics of the propulsion system. To simplify the vehicle analysis and reduce code complexity, the code assumes a point mass in two dimensions with a flat, non-rotating Earth. The acceleration is calculated from Newton's second law in each direction, and integrated over one second intervals for the entire trajectory. The optimization routine of the code determines the optimum trajectory to minimize fuel burn with a given number of engines and a provided engine deck. Using an iterative process, the ideal number of turbofan engines was determined, which defines the width of the vehicle to accommodate 14 engines with a four foot diameter. Another optimization loop was performed to determine the optimum scramjet/rocket transition Mach number. A feedback loop was added to the trajectory code to transition when the thrust/drag ratio of the scramjet was below a certain tolerance. Airbreathing ascent beyond this point provided diminishing energy increases for a relatively constant fuel input. The trajectory plot of *StarRunner* is shown in Figure 6. Below Mach 3, flight path is optimized for minimum fuel burn using the turbofan performance deck. At Mach 3 and 45,000 feet, the ramjet engines are engaged for the ACES cruise. After filling the oxygen tanks, *StarRunner* climbs along the 2000 psf dynamic pressure curve using ramjet/scramjet power until reaching Mach 10.5, where a pull-up maneuver is executed using the scramjet to ascend to 115,000 feet and rocket ignition.

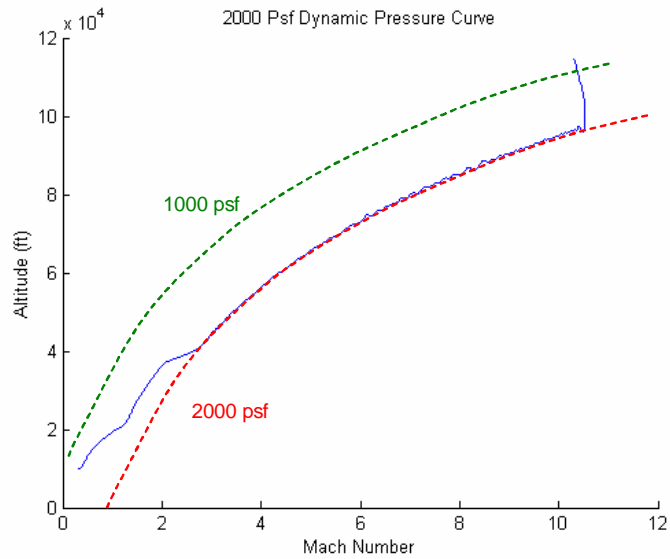


Figure 6. *StarRunner* Trajectory Plot.

is optimized for minimum fuel burn using the turbofan performance deck. At Mach 3 and 45,000 feet, the ramjet engines are engaged for the ACES cruise. After filling the oxygen tanks, *StarRunner* climbs along the 2000 psf dynamic pressure curve using ramjet/scramjet power until reaching Mach 10.5, where a pull-up maneuver is executed using the scramjet to ascend to 115,000 feet and rocket ignition.

AIR COLLECTION AND ENRICHMENT SYSTEM (ACES)

The requirement to maximize the use of airbreathing propulsion systems drove the design to seek ways to greatly reduce GLOW. The primary means identified for GLOW reduction is an ACES. Research on this concept began in the 1960s and has been carried through to ground testing. ACES were proposed for the NASP vehicle in the 1980s and more recently in the “Alchemist” ACES⁷. *StarRunner*'s ACES utilizes a Rotational Fractional Distillation Unit (RFDU), powered by an electric motor using electricity generated by two of the turbofan engines. This device physically separates low temperature oxygen from nitrogen. Future concepts may utilize a vortex tube technology⁸, performing the separation using a much lighter system with few moving parts; however, the TRL of this system was determined to be too low for implementation in *StarRunner*. The chilling system to liquefy incoming atmospheric oxygen and nitrogen utilizes one stage for high temperature and one for low temperature heat exchange. As shown in Figure 7, a portion of the ramjet inlet air is directed into the ACES Inlet. After trace elements such as Argon and CO₂ are removed, the incoming air is directed through the high temperature heat exchanger or pre-cooler. This heat exchanger primarily uses chilled nitrogen flowing back through the system to decrease the temperature of the air by several hundred degrees. At the same time, the cold nitrogen is raised back near ambient

temperature before being dumped into the exhaust of the ramjet engines, providing additional combustor cooling. The ACES air then enters the low temperature heat exchanger or condenser. This device uses liquid hydrogen to decrease the temperature of the air to levels at which both nitrogen and oxygen are liquefied. The liquid mixture then enters the RFDU which spins to physically separate the oxygen and nitrogen. An RFDU diameter of 8 feet is capable of providing 98% purity for the LOX. After separation, the oxygen is pumped into the two storage tanks near the payload bay of the vehicle, while the nitrogen is pumped back through the system to provide cooling in both heat exchangers. This “waste” nitrogen accounts for 73% of the cooling required by the system.

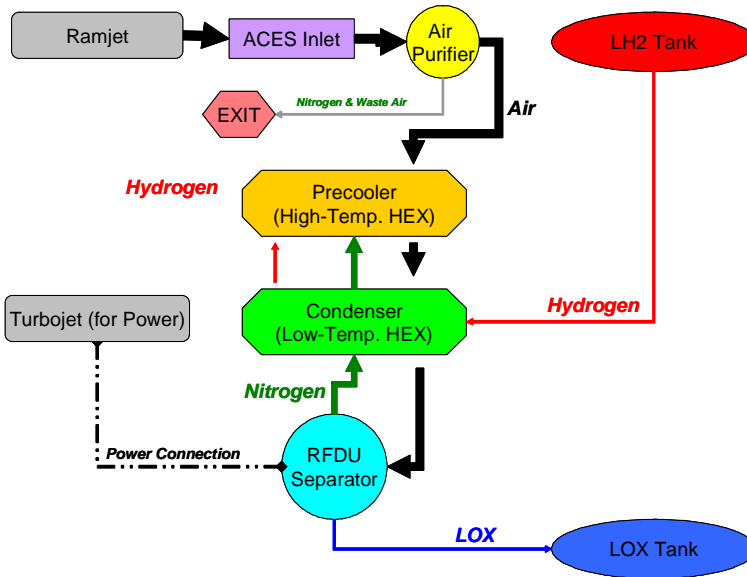


Figure 7. Schematic of the *StarRunner* ACES.

The collection rate of oxygen depends primarily on two factors: the inlet flow rate of air and the amount of liquid hydrogen used for cooling. Although it is possible to design a system which collects liquid oxygen much faster by using more hydrogen fuel, the ramjet engines can only accommodate a specific amount of hydrogen before the equivalence ratio of the propulsion system exceeds 1.0 or the vehicle needs to dump LH2 overboard. To prevent this, the inlet flow rate and collection rate of the system were matched to the fuel flow requirements of the ramjet engine. During the ACES cruise, the cooling requirements for the ramjet components are not a major design constraint, so the thermal heat sink of the hydrogen fuel can be used primarily for oxygen liquefaction. At maximum performance, 320 lb/s of 98% pure LOX can be liquefied and stored.

During the ACES collection phase, the trajectory code maintains cruise at Mach 3.0 at 45,000 feet, increasing the angle of attack slightly as oxygen is collected and the mass of the vehicle increases. Initially, the ramjet engines are operated at part-power; maximum power at the beginning of ACES cruise would cause the vehicle to accelerate or climb, and the ACES is designed to operate at a given Mach number and altitude. As oxygen is liquefied, the ramjets are throttled up to maintain the cruise profile. Approximately thirty minutes into the cruise, the ramjet engines reach 100% throttle. Since the ramjet thrust is a function of mass flow and fuel/air ratio, air must be diverted from the ACES to the ramjets to increase the mass flow to maintain the ACES cruise. During the second half of the ACES cruise, the oxygen collection rate decreases by approximately 15% over the remaining thirty minutes. According to the trajectory code, the cruise ends successfully and the mission continues when 859,000 lbs of LOX have been collected. In the model, the mission fails if the collection rate decreases to zero or if the vehicle burns too much hydrogen during the cruise. The optimized design requires approximately 100,000 lbs of LH2 to power the ramjet engines over the time required to collect the necessary LOX. Without the ACES, *StarRunner's* empty weight decreases, but the GLOW increases to the point at which the turbofan engines cannot provide sufficient thrust to overcome the transonic drag rise without the use of the rocket. Trade studies revealed that the turbofan-powered system required ACES, whereas the exclusion of ACES drove the design from a TBCC to an RBCC configuration inconsistent with the RFP requirement to maximize the use of airbreathing propulsion. The weight of the system, approximately 32,000 lbs, was estimated from Reference 9. This weight is based on the use of the RFDU for separation, while a vortex tube-based system may reduce the weight by as much as 30%. The system is estimated to be approximately 8 feet in diameter and 10 feet long, based on the ground-test RFDU models and estimates for the “Alchemist” system.

THERMAL PROTECTION SYSTEMS (TPS)

Unlike the Space Shuttle and all historical launch vehicles, the maximum heating loads experienced by *StarRunner* occur during ascent rather than reentry, due to the relatively long time the vehicle spends at supersonic and hypersonic speeds within the Earth’s atmosphere. The primary TPS for the vehicle utilizes advanced passive protection systems similar to the tiles and RCC panels on Space Shuttle. TPS materials for each vehicle area are shown below in Table IV. The nose and wing leading edge will encounter a maximum heating load of 3530°F for approximately 15 seconds at the end of the scramjet ascent prior to rocket ignition; however, during the Mach 3 ACES cruise, the vehicle experiences a constant temperature on these surfaces of around 1550°F. Ultra High Temperature Ceramics (UHTC’s) such as Halfnium Diboride provide the potential to handle this temperature. High temperature areas behind the vehicle nose and wing leading edge utilize Advanced Carbon-Carbon panels. The lower surface of the vehicle requires thermal protection that can be provided by NASA-developed TUFU tiles; however, the viability of these TPS materials under high dynamic pressures must be confirmed. Due to the lower temperatures experienced, the upper surfaces of the vehicles utilize AFRSI blankets where appropriate.

Table IV: *StarRunner* Thermal Protection Systems.

TPS Material	Location	Max Temp.	Density (lb/ft ³)	Emissivity	% of Total TPS Weight
Halfnium Diboride (UHTC)	Nose and Wing L.E. (6’’)	4500 °F	594.3	0.90	9.7%
Adv. Carbon/Carbon (ACC)	Nose and Wing L.E. (12’’)	2900 °F	99.9	0.90	4.1%
Titanium Aluminide	Wing/Tail Structure	1900 °F	28.5	0.80	27.5%
AFRSI Blankets	Vehicle Upper Surface	1600 °F	6.0	0.87	29.9%
TUFU Tiles w/ AETB-8	Vehicle Lower Surface	2400 °F	81.8	0.87	28.9%

The inlet lip, due to shock/boundary layer interactions and shock/shock interactions, requires active cooling using a series of copper NARloy-Z pipes and liquid hydrogen fuel similar to the water-cooled solution used for the 10 second test flight of the X-43A in March 2004¹⁰. Hydrogen is also used to provide cooling for the ramjet/scramjet components in the hypersonic regime. Using the first-order estimation techniques from Reference 11, the fuel flow of hydrogen to the scramjet is sufficient to provide the required cooling up to a Mach number of approximately 11. This reinforces the selection of Mach 10.5 as the scramjet/rocket transition Mach number.

COST AND ECONOMICS

After converging the design process on a configuration that satisfies the performance requirements of the competition RFP, it was also necessary to build a valid business case for the SSTO vehicle and estimate the cost of the system. *StarRunner*’s cost estimate uses the Cost and Business Analysis Module (CABAM), developed by the Georgia Tech SSDL¹². This Excel-based tool calculates the DDT&E cost, operations cost, and first unit production cost (TFU). Several programmatic assumptions were made for *StarRunner*. The DDT&E phase of the program is assumed to start in 2011. The first vehicle will be purchased and completed in 2018, with several test flights during 2018 and 2019. The first full-scale mission will be a government ISS ferry mission, conducted in late 2019. Following successful completion of the test phase, the spacecraft will begin a low-rate operations phase where between 25% and 75% of the maximum flight rate for commercial and government markets will be conducted. Meanwhile, two additional vehicles will be purchased and constructed. The program will begin full-scale operations by 2023, and will continue until 2040. Extension of the program will be possible, and upgrades to the vehicle can occur by using money obtained during the 20 year operational period. This program closely follows that of the Space Shuttle¹³.

To establish a valid business case, government contribution of 50% toward the airframe development and 100% towards the propulsion system development will be required. Using public-domain estimates from a variety of sources for estimated current and future markets for government and commercial satellite launches, a correlation between the launch price and number of launches can be established. Using these predictions, CABAM will determine the Internal Rate of Return (IRR) for a given flight rate. By selecting the desired IRR, the cost per pound (\$/lb) to orbit is established. The results of the CABAM parametric

study are shown in Figure 8. Based on the market assumptions, as the \$/lb to orbit increases, the number of customers willing to accept this price decreases, reflecting the decrease in flights per year with an increase in \$/lb. The IRR can be maximized in two ways: with a very high price and a very low flight rate, or by selecting the highest flight rate with a high IRR. As illustrated in Figure 8, a local maximum in the IRR function exists at a flight rate of 54 flights/year and a cost per pound to orbit of \$825/lb. This corresponds with an IRR of 8.95%, and puts the \$/lb value below the often-quoted \$1000/lb desired value for next generation reusable launch vehicles.

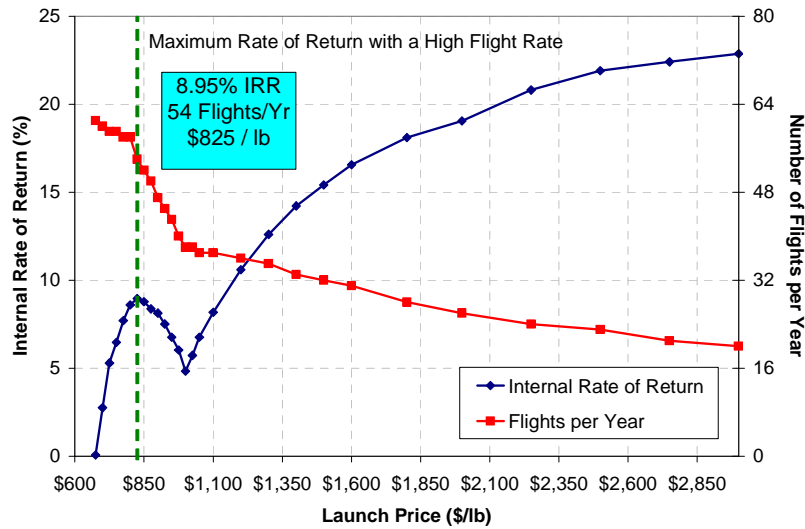


Figure 8. Determination of IRR and Flight Rate.

The weight-based CABAM module is calibrated for the technology factors estimated for the 2020 EIS of *StarRunner*. With the market and IRR assumptions defined above, the estimated DDT&E cost of *StarRunner* is \$10,909 M (2003 USD), and a TFU cost of approximately \$2,933 M (2003 USD). The KT40 turbofan engine is estimated using PRICE H cost estimation software¹⁴ and a modified baseline engine model from the Georgia Tech ASDL. Sixty engines will be produced to provide the three *StarRunner* airframes with sufficient spares. The engine cost model requires weight inputs from the WATE flowpath model and technology k-factors that are consistent with the predicted level of engine technology for the 2020 EIS. Finally, the overall engine complexity factor is set at the upper bound to represent the difficulty of developing a new centerline turbofan engine using hydrogen fuel. Using this model, the estimated cost per engine of the KT40 is \$5.3 M (2003 USD) for a production run of 60 engines. If the modification of an existing turbofan engine such as the F110 or F119 were a program requirement, the DDT&E cost for the engine modifications is approximately \$684 M (2003 USD), while the more likely result of a new centerline engine and technology demonstration program may cost several billion dollars. The development of a new core will likely require government involvement and commonality with future hypersonic global strike military systems. The high cost of the KT40 in comparison to other engines in this thrust class is a result of the low production run of the engine program. If the same technology assumptions are used over a JSF-like production run of around 3,000 engines, the cost per engine drops to under \$3M per engine, which is consistent with predicted values for a 35-40,000 lb thrust class afterburning turbofan engine program. The overall breakdown of the cost categories for *StarRunner* is shown in Figure 9. The actual mission operations, including operations, maintenance, insurance, and reusable systems, accounts for 35% of the total program cost, while the development of launch facilities and vehicle DDT&E is nearly 10% of the total cost. As shown in the figure, the greatest expenditure category is the financing cost of the program,

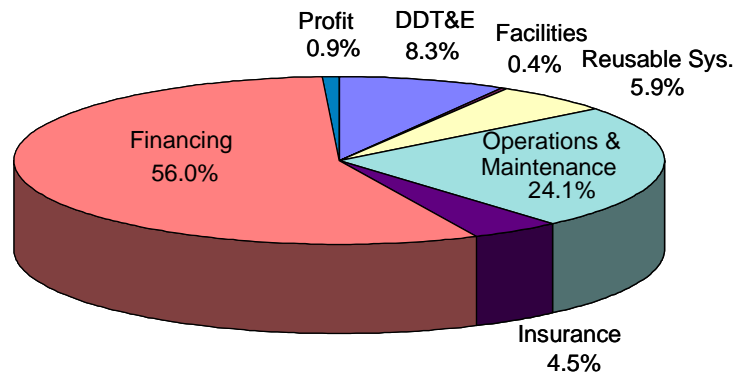


Figure 9. Summary of *StarRunner* Cost and Economics.

due to the large initial outlays required to develop, construct, and test *StarRunner*. Due to the relatively low IRR, the total estimated profit for the program is only 1%. With the market assumptions predicted and technologies required to field a viable reusable launch vehicle, the basis of a commercial program on the economics of *StarRunner* would be difficult. The *StarRunner* concept would require extensive government involvement and support for the development of the vehicle, maturation of advanced technologies from low TRL to flight readiness, and operation of *StarRunner* over a twenty year time frame. Although the *StarRunner* economics were based on the assumptions of the design competition, in addition to supporting the ISS, *StarRunner*'s highly reusable nature would make this architecture ideal for the newly announced Moon/Mars initiative which may require multiple launches to assemble interplanetary vehicles in LEO.

CONCLUSION

The *StarRunner* concept represents a solution to the demanding requirements of a next-generation reusable launch vehicle that utilizes airbreathing propulsion systems. Throughout the design process, aggressive but realistic assumptions of future technologies were utilized to demonstrate the viability of an airbreathing, hypersonic SSTO access-to-space vehicle. Technologies required to realize this advanced system include advanced compressor and turbine alloys, blisk construction techniques, combustor ceramic matrix composites, and technologies that drastically increase the thrust-to-weight ratio over current turbofan engines. Following the successful test of the X-43A vehicle in 2004, further testing of scramjet combustion systems over a wide range of Mach numbers must be conducted, and reusable hydrogen-powered scramjet demonstrators should be flight tested to validate the *StarRunner* technologies on a smaller scale. Additionally, the development of highly efficient linear aerospike rocket engines should be reexamined following the cancellation of the X-33 program. Also, due to the high aerodynamic heating loads experienced on ascent, new thermal protection systems such as UHTC's must be developed and flight-proven to allow *StarRunner* to use passive cooling for the airframe to retain the cooling capacity of its hydrogen fuel for the engine components. Finally, a reliable and robust ACES must be developed with low-weight components that rapidly liquefy atmospheric oxygen and store it for later use.

The integrated nature of the propulsion system and airframe for the SSTO vehicle required that the student design team members not only possess knowledge of each discipline, but also understand the complex interactions of the multidisciplinary design. Many iterations between team members were required to understand the complex interactions of the physics of the problem, and compromises were reached through trade studies and iteration between the various disciplines. Ironically, the inexperience of the mostly-freshmen team contributed to the unorthodox approach to this complex, real-world problem. The enthusiasm of the team's younger members and desire to succeed in defiance of this lack of experience were two of the driving factors in keeping the team members engaged in the project from week to week. Team members learned that while many concepts may be technically feasible, the reality of cost and time constraints on the project may limit the final design. Additionally, the team quickly learned that off-the-shelf solutions for vehicles requiring such advanced technologies are rarely available, confirming the initial prediction that, "if this were easy, someone would have done it already." Nevertheless, the competition provided students with a thorough overview of the exciting challenges facing them as they begin their careers in aerospace engineering.

This educational program not only provides students with a real-world application for classroom methodologies, but also encourages student involvement in extracurricular activities related to aerospace engineering, space vehicle design, and propulsion system analysis. Two team members have entered the graduate program at GT, while four former freshmen are undergraduate research assistants at the University.

ACKNOWLEDGEMENTS

The Georgia Tech Student Design Team, FAS Propulsion Systems, would like to thank our competition advisors, Dr. Olds and Mr. Escher, in addition to the staff of the ASDL, SSDL, and representatives from the Space Propulsion Synergy Team (SPST) and SpaceWorks Engineering, Inc who served as mentors and project reviewers. Finally, we would like to thank the AIAA and the Airbreathing Propulsion Technical Committee for their continued sponsorship of the yearly design competitions that promote student creativity through research and design.

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