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Thomas M. Krivanek, Joseph M. Roche, and John P. Riehl
Glenn Research Center, Cleveland, Ohio

Daniel N. Kosareo
ZIN Technologies, Inc., Cleveland, Ohio

April 2003

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AFFORDABLE FLIGHT DEMONSTRATION OF THE GTX AIR-BREATHING SSTO VEHICLE CONCEPT

Thomas M. Krivanek, Joseph M. Roche, and John P. Riehl
National Aeronautics and Space Administration
Glenn Research Center
Cleveland, Ohio 44135

Daniel N. Kosareo
ZIN Technologies, Inc.
Cleveland, Ohio 44135

ABSTRACT

The rocket based combined cycle (RBCC) powered single-stage-to-orbit (SSTO) reusable launch vehicle has the potential to significantly reduce the total cost per pound for orbital payload missions. To validate overall system performance, a flight demonstration must be performed. This paper presents an overview of the first phase of a flight demonstration program for the GTX SSTO vehicle concept. Phase 1 will validate the propulsion performance of the vehicle configuration over the supersonic and hypersonic air-breathing portions of the trajectory. The focus and goal of Phase 1 is to demonstrate the integration and performance of the propulsion system flowpath with the vehicle aerodynamics over the air-breathing trajectory. This demonstrator vehicle will have dual mode ramjet/scramjets, which include the inlet, combustor, and nozzle with geometrically scaled aerodynamic surface outer mold lines (OML) defining the forebody, boundary layer diverter, wings, and tail. The primary objective of this study is to demonstrate propulsion system performance and operability including the ram to scram transition, as well as to validate vehicle aerodynamics and propulsion airframe integration. To minimize overall risk and development cost the effort will incorporate proven materials, use existing turbomachinery in the propellant delivery systems, launch from an existing unmanned remote launch facility, and use basic vehicle recovery techniques to minimize control and landing requirements. A second phase would demonstrate propulsion performance across all critical portions of a space launch trajectory (lift off through transition to all-rocket) integrated with flight-like vehicle systems.

INTRODUCTION

The achievement of SSTO will result in a smaller, less expensive vehicle for access to space. The GTX program[1] is a liquid oxygen/liquid hydrogen fueled, vertically launched, horizontal landing, SSTO concept shown in figure 1. This concept utilizes a RBCC engine system that is designed to combine the high thrust–to-weight performance of a rocket with the fuel efficiency of the ram/scramjet airbreathing engine into a single highly integrated propulsion system. This system operates in four modes. Mode 1 is a low speed rocket/air augmented system for launch to Mach 2.5; mode 2 is a ramjet system for Mach 2.5 to Mach 5.5; mode 3 uses scramjet operation for Mach 5.5 to Mach 11; and mode 4 is the rocket only operation for Mach 11 to orbit insertion.

A key factor for enabling SSTO launch system technology is the enhanced specific impulse of the air-breathing modes. Development of the propulsion system through component level performance validation is ongoing. However, test facilities are limited in scale, test medium composition, Reynolds number, and test time. Therefore, flight-testing is critical and required to validate the overall system. A well-designed program utilizes a mixture of ground testing and an evolutionary flight demonstration program to systematically characterize performance while minimizing the cost and technical risk. Such a flight demonstration program incorporates existing technology to the largest extent possible. This design utilizes existing materials, instrumentation, robust systems and existing boosters, and will maximize technical results while minimizing cost and risk.

In the case of the GTX concept, a critical technical challenge is the operation of the propulsion system in the pure air-breathing portions of the trajectory from Mach 2.5 to Mach 11, (modes 2 and 3). Validation of propulsion/airframe integration and ram/scram performance of the reference vehicle is the primary objective of the Phase 1 flight demonstrator shown in figure 2. The focus of the current study is to develop the requirements and a realistic cost estimate for the Phase 1 vehicle by accurately defining the mission trajectory, operational parameters, and the test objectives. A second phase would demonstrate the rocket integration with flight-like vehicle systems over the complete trajectory from lift-off to mode 4 transition.

This study provides a candidate demonstrator to validate ramjet and scramjet performance and operation over the Mach 2.5 to 7.5 range. Restricting operation to this range eliminates the oxidizer required for rocket operation and reduces the aero-thermal heating that drives up the vehicle size and cost. Limiting the flight to below Mach 8.0 permits the use of existing metallic materials, (and structures), Thermal Protection Systems (TPS), and eliminates the need for active cooling of external surfaces. The current study serves to layout and optimize the demonstrator architecture and mission trajectory to achieve the required performance validation.

VEHICLE ARCHITECTURE AND SIZING

The demonstration vehicle, shown in figure 3, has three propulsion pods, is 37.3 feet long with a 14.5 ft wingspan, resulting in a 26% scale of the GTX SSTO reference vehicle. Three Black Brant SRM's[2] are used to propel the vehicle from lift-off to the ramjet takeover speed of approximately Mach 2.4. The vehicle was scaled using the GTXSizer[3] model to reach at least Mach 7.5. Lift off weight is also limited by the thrust capability of the SRM's, which must provide sufficient thrust to accelerate the demonstration vehicle to ramjet take over. The resultant propulsion system scale also allows for full scale component testing in existing NASA and contractor facilities prior to final design.

The structure is capable of accommodating the solid boosters nestled between the RBCC nacelles. The SRM's are mounted on the vehicle in a cluster that can be vectored. The trimming of the vehicle before booster separation requires vectoring of the boosters by gimbaling the forward booster connection and moving the aft end of the boosters through an internal thrust ring with two perpendicular hydraulic actuators. Separation is accomplished by simultaneous firing of explosive bolts at each of the booster hard points. Actuation of the aero-control surfaces and independent fueling of the engines to create differential thrust about the vehicle centerline, simulating thrust vectoring, trim the vehicle after SRM separation.

A vehicle weight summary is presented in table 1. Sub-systems include the fuel delivery system, the power distribution, guidance and control, data acquisition and telemetry, and range safety as shown in figure 4. The liquid hydrogen fuel (LH_2) is plumbed with a regenerative cycle turbopump of the RL10a variety.[4] The fuel tank is a filament wound graphite epoxy structure. The tank is covered with Airex insulation. The vehicle structure is protected with a metallic aero shell backed with Saffil insulation. The vehicle purge system incorporates gaseous nitrogen to control internal icing, subsystem temperatures and to provide a diluent gas to manage hydrogen concentration issues. Actuators are a mix of electro/hydraulic and electro-mechanical systems based on the load and response requirements of the translating centerbodies, control surfaces, and booster thrust vectoring rings. Most of these systems and components are existing units employed on recent X-Vehicle programs.[5–7] These components have been flight-tested, thus minimizing technical risk and development cost. Passive recovery methods are also used to eliminate concerns over low speed aero trim and handling performance of an un-powered vehicle and the need for an autonomous landing system (minimizing the cost and complexity of landing gear and structure). This passive method provides for contingency recovery locations for aborted missions as well. It also reduces the number of preliminary checkout test flights required and the risk of vehicle loss during those checkouts. The recovery system is based upon existing tested concepts.[8]

VEHICLE OPERATION

The flight vehicle prelaunch assembly and checkout could be conducted at the X-33 launch facility at HayStack-Butte, Edwards Air Force Base, California, a picture of which is shown in figure 5. This facility was built in 1999 and has the capability for cryogenic hydrogen handling, incorporates the latest tracking and telemetry equipment, and data acquisition ground network system.[9] The facility is designed for a horizontal buildup with a rotation to vertical orientation for launch.

The launch sequence commences with SRM ignition followed by vehicle release upon thrust equalization. The vehicle accelerates for 28 seconds under booster thrust reaching Mach 2.3, where booster thrust tailoff begins. When the vehicle reaches Mach 2.5, at 32.4 seconds, the SRM's are separated. The vehicle coasts for 6 seconds to allow the dynamic pressure to decrease to 1500 psf. Tank head pressure is used to speed up the RL-10a turbopump to 25,800 rpm in two seconds. The fuel is delivered at 980 psi, by the regeneratively powered pump, which first pumps the fuel thru heat exchangers that actively cool the walls of the engine combustor and nozzle. The ramjet engine flowpath is then fueled and ignited. The heat exchangers are metallic with 0.25 inch tube on plate construction. The air mass flow through each engine is 63 lb/sec at Mach 2.4 and drops to 39 lb/sec at Mach 7.8. The fuel flow rate drops accordingly from 1.74 lb/sec to 1.5 lb/sec per engine for an equivalence ratio of 1.0 to 1.3. The vehicle is fueled for 112 seconds as it accelerates to a final Mach number of 7.8.

At takeoff the engines are configured to the full open centerbody position to reduce drag during the boost stage and for initial inlet starting. The centerbody then translates aft during flight until it reaches the Mach 6 shock-on-lip position, where it remains throughout the SCRAM portion of the trajectory. At the completion of the powered trajectory the centerbodies are retracted to completely close off the flowpath to increase engine drag and to minimize internal heating.

After completion of the powered trajectory the vehicle decelerates through a series of banks and turns, and descends to reach it's landing site. The vehicle reaches the latitude and longitude of Dugway Proving Grounds, at an altitude of 30,000 ft and Mach number of 0.5. For the remaining descent one drogue parachute is deployed to slow the vehicle and at approximately 12,000 ft the main stage of two parachutes are deployed from the leeward side of the vehicle at the top two booster attachment points. This slows the vehicle down to a vertical fall rate of 16 ft/sec. At 6000 ft, (1500 ft above ground level) air bags are deployed from the windward surface just forward of the bottom booster front mount and from the two wings near the nacelle attachment. After touch down the vehicle is retrieved via helicopter using an attachment at the parachute rings and returned to the recovery site. The use of parachutes and air bags for recovery minimizes complexity and cost.

DEMONSTRATOR TRAJECTORY

The trajectory was optimized using the computer program OTIS[10] and designed to achieve maximum Mach number with the available fuel. Vehicle propulsion and aerodynamic coefficients were obtained from the GTX reference vehicle. The maximum dynamic pressure is limited to 1500 psf in the RAM/SCRAM portions of flight. Transition from RAM to SCRAM operation was constrained to occur at a Mach number between 5.0 and 5.5.

The launch coordinates used were –117.64° W longitude, 34.89° N latitude and 2700 feet altitude above sea level, (HayStack Butte launch site). The trajectory simulation included constraints to force a northeastern trajectory as shown in figure 6. Trajectory parameters are presented in table 2. Booster separation occurs 3.2 miles downrange within the Edwards AFB test range. The vehicle consumes all its propellant 82 nautical miles downrange. The vehicle coasts, descends, and lands at the Dugway Proving Grounds in Utah. The descent trajectory was optimized for minimum time to descend and still produced a range contingency on distance as seen in figure 7 where the plot indicates a climb up to the end point of Mach 0.5 at 30,000 ft.

TRACEABILITY TO REFERENCE VEHICLE

The objective of the Phase 1 flight demonstration test is to validate the *system* performance over the RAM and SCRAM operating ranges of the SSTO vehicle. Two key performance parameters are the effective specific impulse (I_{eff}), which is defined as:

$$I_{eff} = \frac{T_{net} - D_{veh} - W \sin\gamma}{\dot{m}_p} \tag{1}$$

which becomes:

$$I_{eff} = I_{sp}\left(1 - \frac{D_{veh}}{T_{net}} - \frac{W \sin\gamma}{T_{net}}\right) \tag{2}$$

where:

$$I_{sp} = \frac{T_{net}}{\dot{m}_p} \tag{3}$$

The equivalent, effective specific impulse (I^*) over a given velocity interval is defined by:

$$I^* = \frac{\Delta V_{1-2}}{\int_{\Delta V_{1-2}} \frac{dv}{I_{eff}}} = \frac{\Delta V_{1-2}}{g_c \ln \frac{m_1}{m_2}} \quad (4)$$

From equation 4, the propulsion performance is confirmed if the vehicle completes the trajectory. The weights and available fuel are known (m_1 and m_2) leaving only the performance (I^*), as the single variable if the velocity change is achieved.

The flight trajectories of the demonstrator and the reference vehicle are presented in figure 8. Major differences in the low speed trajectory are due to the use of the SRM's to achieve the initial ramjet velocity and are not relevant to the present study. The variation in altitude and dynamic pressure at Mach 2.5 is small. At Mach 7.8 the conditions are the same for both vehicles. A comparison of the I_{sp} and I_{eff} performance to the reference vehicle is shown in figure 9. The I_{sp} of the demonstrator matches the reference vehicle (i.e. the same propulsion database was used). The higher I_{eff} of the demonstrator is due primarily to the difference in the thrust to weight between the demonstrator and the reference vehicle, as seen in figure 10. The demonstrator accelerates approximately 46% faster than the reference vehicle resulting in a steeper flight path angle to maintain constant dynamic pressure, as seen in figure 11. The weight fraction of the demo vehicle is much larger than the reference vehicle since it is designed to only carry the fuel required for the air-breathing portion of the reference mission.

The angle of attack is roughly half that of the reference vehicle due to the lower weight of the demonstrator, as seen in figure 12. This lower angle of attack results in a lower lift to drag ratio as seen in figure 13. The drag and thrust are proportional to the square of the scale, while the weight is proportional to the cube of the scale. This produces a relatively smaller weight term with respect to the thrust and drag. From equation 2, this results in the slightly higher I^* being achieved for the demonstration vehicle over the same velocity range, since the terms subtracted are smaller in magnitude. The flight demo I^* is determined to be 2180 seconds. The reference vehicle calculated I^* over the same velocity is 2101 seconds.

TECHNICAL CHALLENGES

Maintaining engine operation throughout the ramjet/scramjet modes of the flight is the primary objective. The fuel delivery system must sense the engine performance; adjust the fuel flow rate and schedule, to maintain the required thrust and trajectory. Balancing the thrust produced by the three engines also requires a fast response rate system for controlling the centerbodies, fuel, and flight surfaces.

The highest temperatures seen by the vehicle are at the nose and wing leading edges. These surfaces are made of CMC materials to withstand these elevated temperatures over the short duration they are experienced as shown in figure 14. Thermal limitations of the metallic vehicle, however, require skin temperatures below 1600 °F to avoid accelerated structural failure. These thermal limits necessitate the flight dynamic pressure constraint. Although the simulation would be more representative at a higher dynamic pressure, a compromise must be reached to allow the vehicle to survive. Active cooling of the engine flow paths, which will experience pressures up to 140 psi, requires a robust heat exchanger design.

Vehicle sealing issues are found at the translating inlet centerbodies, SRM front attachments, the vectoring tie-plates for the boosters, vehicle elevons and rudder surfaces, parachute and airbag cavities, and umbilical connection ports. Work in seal design and development from existing programs will be used to address these issues. Attachment of the solid boosters remains an area of future work to establish structural details. The simultaneous separation of all three SRM's must occur cleanly.

COST FACTORS

Efforts to minimize the cost of the demonstrator vehicle include the use of flight-tested avionics components and existing materials and manufacturing methods. Cost allowances have been included for potential modifications to existing hardware to provide a higher degree of confidence in the estimate. It is assumed that engine development and testing will be completed through component ground tests and ultimately through a flight-like engine flowpath effort, the

GTX Rig. 5 Project.[11] Only those provisions required for engine integration are estimated.

The three Black Brant SRM's required per flight are assumed to be available from existing military excess stock. The largest cost factor for each booster will be demonstrator modifications, such as removing the fins, guidance and control systems, and the payload bay and adding the structural reinforcements and attachment structure. The cost of the rockets and associated telemetry/range safety services were added.

Additional assumptions include the use of an existing unmanned remote launch facility with only minimal refurbishment, building of two vehicles (one prototype and one flight), a test matrix to cover six flights, no preliminary captive carry or drop tests required, and modification and use of the X-33 Environmental Impact Assessment. Operational costs for the six test flights are included in the project estimate. Ground support is primarily the onsite technical personnel associated with range services. These costs are included with the rocket costs and based on recent GRC experience with Black Brant rockets used for microgravity experiments. Refurbishment costs for additional test flights were included for the vehicle and engines based on factors provided in the TRANSCOST Model, version 6.2 (dated October 1998). Based on unit cost, refurbishment factors of 2.8% for the vehicle less engines and 13% for the engines were used. Vehicle subsystem costs were derived using an analogy to the X-34 project data provided in the NASA/Air Force Cost Model (NAFCOM) database.

The government cost additions includes a fee of 10%, contingency/reserve at 30%, and program support at 15%. Program support includes NASA institutional costs, nominal direct project support, and contract oversight. The cost for the Phase 1 project is estimated to be about $325 M in fiscal year 2002 dollars as shown in table 3, with the above changes incorporated.

The vehicle requires integration of existing operational control systems and thoroughly tested end components. Significant ground tests are required and the availability of a prototype unit results in significant risk reduction providing potential spares for every vehicle subsystem. Other possible cost and risk reducing measures include the use of existing data from telemetry studies, recovery tests, and systems analysis.

CONCLUSIONS

This study has provided a conceptional design of a scaled flight demonstration vehicle, which serves to validate the air-breathing ram/scram performance of the GTX reference vehicle configuration. Existing SRM's can provide the thrust to propel the vehicle from launch to ram takeover. The vehicle accelerates to Mach 7.8 on the available tanked propellant. Maximum use of existing technology and flight-tested components has been implemented in the design in order to achieve an affordable and low risk demonstrator vehicle. The cost for the first phase has been estimated at $325M using the stated assumptions. This flight demonstrator will provide a benchmark in high speed air-breathing propulsion and can provide a test bed for future hypersonic experiments.

SYMBOLS

AVTCS	Air Vehicle Thermal Control System
CMC	Ceramic Matrix Composite
DGPS	Differential Global Positioning System
ESP	Electronically Scanned Pressures
ECLSS	Environmental Control and Launch Subsystem
EPD&C	Electrical Power Distribution and Control
FTS	Flight Termination System
INS	Inertial Navigation System
I_{eff}	Effective Specific Impulse
I_{sp}	Specific Impulse
I^*	Equivalent, Effective Specific Impulse over a Velocity Range
OML	Outer Mold Lines (define vehicle external skin)
OTIS	Optimal Trajectories by Implicit Simulation Computer Program
PDU	Power Distribution Unit
RAM	Ramjet Engine
SCRAM	Supersonic Combustion Ramjet Engine
TRANSCOM	Transportation Cost Model
VPP&D	Vent, Pressurization, Purge and Dump System
$\dot{m}_p$	Propellant mass flow
m_1	Vehicle initial mass
m_2	Vehicle final mass
T_{net}	Net Thrust
W	Vehicle Weight
γ	Flight Path Angle w.r.t. horizontal

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Table 1. GTX Phase 1 Flight Demonstrator – Weight Summary.

Assembly	Components	Weight
Aero Shells	Nose Fairing, Midbody, Nozzle, Thrust Ring Forward & Aft	958
TPS	Nose Fairing, Cowl Leading Edges, Midbody, Aftbody, LH2 Tank	123
Propellants	LH2	589
Tankage	LH2, Nitrogen	315
Wings	Tail & Leading Edge, Left Wing & Right Wing, Left Wing & Right Wing Leading Edges	271
Engine 1 Top Dead Center	Centerbody Aero-exposed, Centerbody Combustion Side, Cowl Aero-surface & Ribs, Cowl Leading Edge, Cowl Flow Path Area, Rocket, Closeout Skirt, Diverter Pad	494
Engine 2 Port Side	Centerbody Aero-exposed, Centerbody Combustion Side, Cowl Aero-surface & Ribs, Cowl Leading Edge, Cowl Flow Path Area, Rocket, Closeout Skirt, Diverter Pad	509
Engine 3 Starboard Side	Centerbody Aero-exposed, Centerbody Combustion Side, Cowl Aero-surface & Ribs, Cowl Leading Edge, Cowl Flow Path Area, Rocket, Closeout Skirt, Diverter Pad	509
Recovery System	Drogue Chute, Main Parachutes, Airbags, Structural Attachments	409
Equipment	Solid Booster Supports, Actuators, Batteries, Hydraulics, RL10a Turbopump, AVTCS, Data Acquisition, DGPS, ESP, ECLSS, EPD&C, Instrumentation, FTS, Range Safety, Telemetry, VPP&D	1,820
TOTALS	Vehicle Dry Weight (w/30% weight contingency) Vehicle Wet Weight Propellant Fraction	7,033 7,622 7.73%

Table 2. GTX Phase 1 Flight Demonstrator – Trajectory Summary.

Event	Elapsed time, sec	Altitude, ft	Free-stream Mach #	Flight path angle, deg	Dynamic pressure, lb/ft^2	Vehicle weight, lbm	Down range distance, nmi
1. Launch	0	2,700	0	90	0	16,031	0
2. Booster B/O and separation	32.4	35,802	2.5	25	2271	9192	3.2
3. Ramjet initiation	38.8	41,862	2.4	21	1500	7,622	5.1
4. Scramjet transition	92.7	75,956	5.4	4	1453	7,336	31
5. Begin coast	151	91,139	7.8	0	1500	7,052	82
6. Deploy drogue chute	564	30,000	0.5	90	114	7,052	324
7. Vehicle touchdown	~	4,349	0	0	0	6,820	345

Table 3. GTX Phase 1 Flight Demonstrator—Cost Summary.

GTX Project Estimate (FY02M$)	
WBS Item	**TOTAL**
Structural/Mechanical Subsystem	27.3
Thermal Protection Subsystem	9.7
Avionics/Telemetry/Power Subsystem	14.8
Propulsion Subsystem (less engines)	37.3
Recovery Subsystem	1.6
Engine Integration Provisions	15.9
Hardware Subtotal	106.6
Test HW/Test/GSE/SE&I/PM	74.7
Total Vehicle Cost	181.3
Fee	18.1
Contingency	54.4
Government Program Support	44.8
Total Vehicle Estimate	298.6
Sounding Rocket & Range Services	27.0
Total Project Estimate	**$325.6**

Figure 1.—GTX Reference vehicle–Configuration 10c.

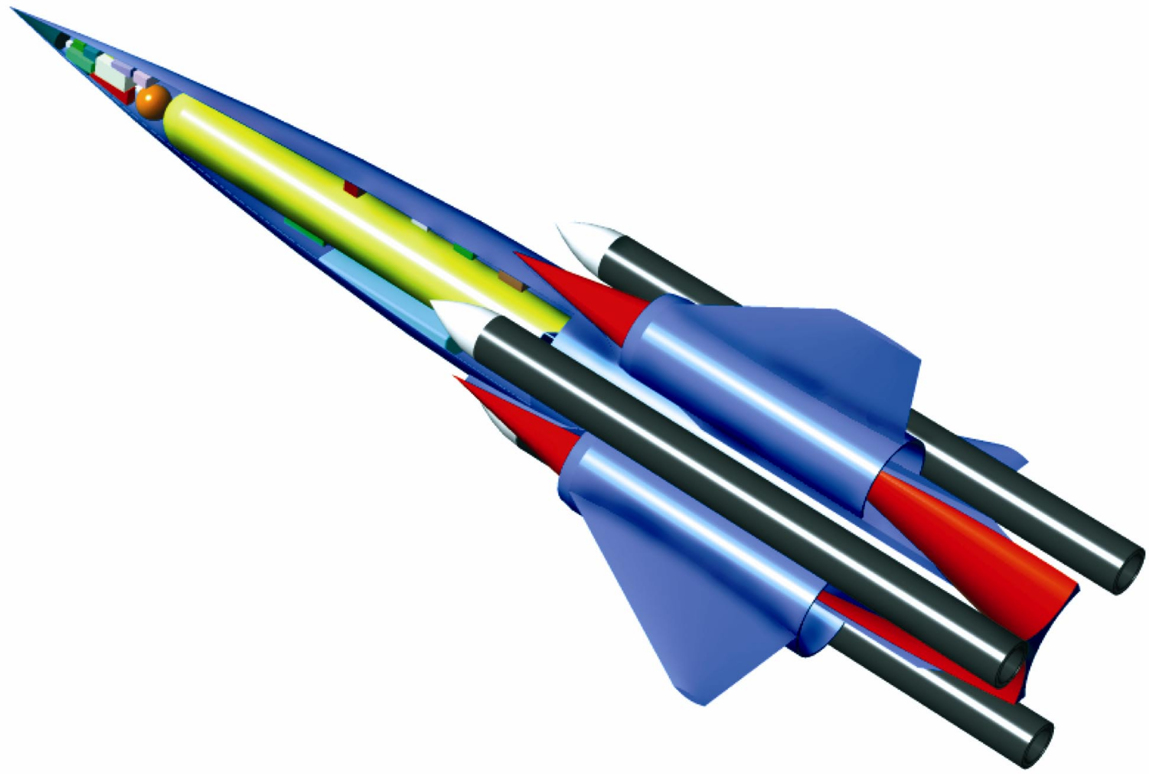

Figure 2.—GTX Phase 1 Flight Demonstrator Vehicle, 7,622 lbs.

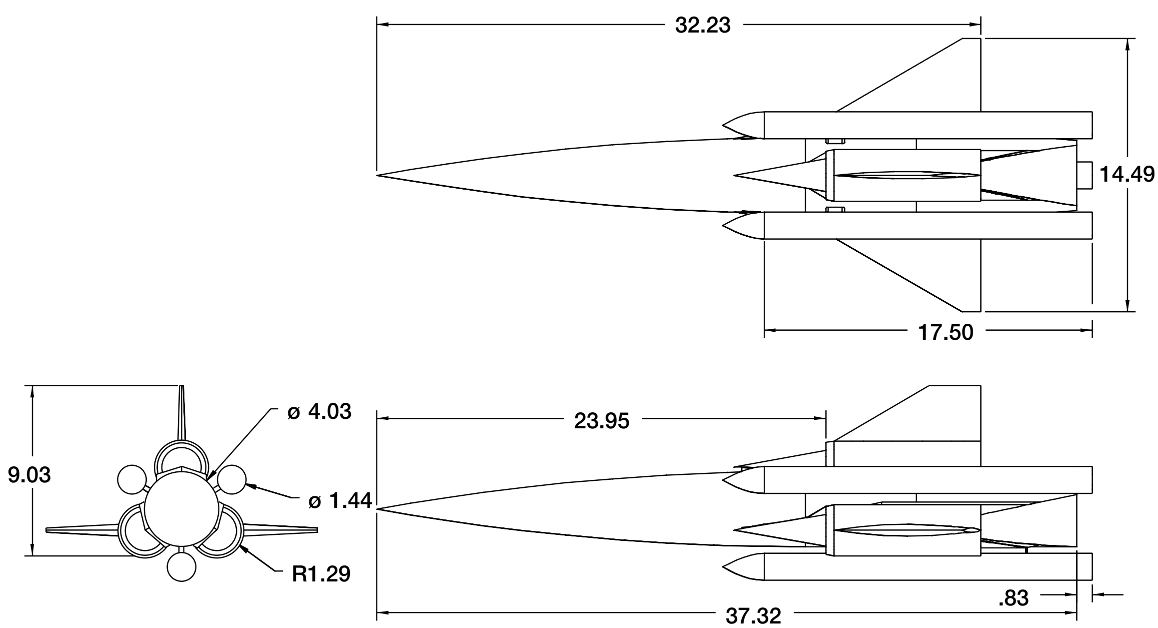

Figure 3.—GTX Flight Demonstrator Dimensions (ft).

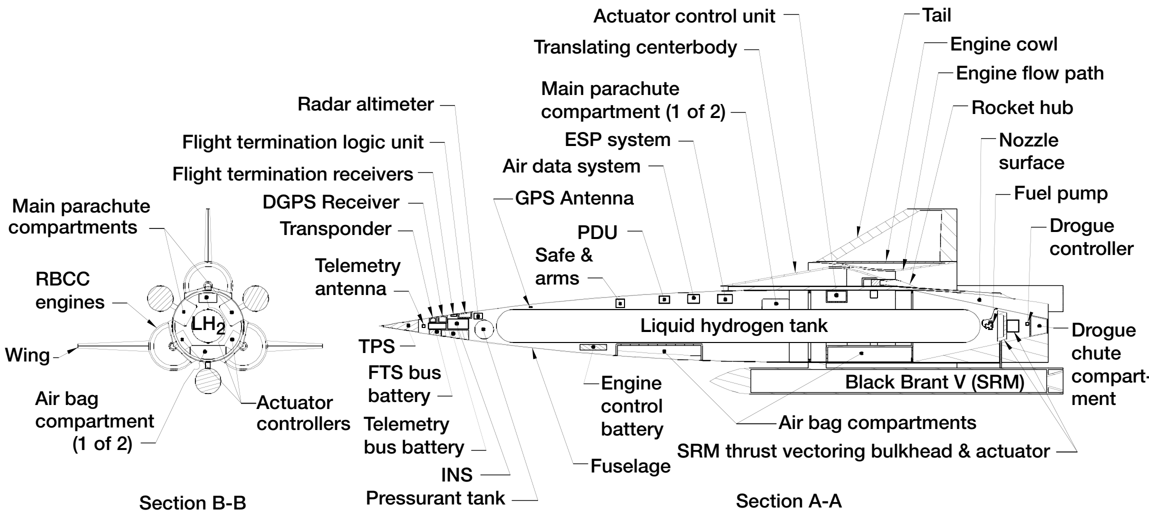

Figure 4.—GTX Phase 1 Flight Demonstrator Components–Sectional.

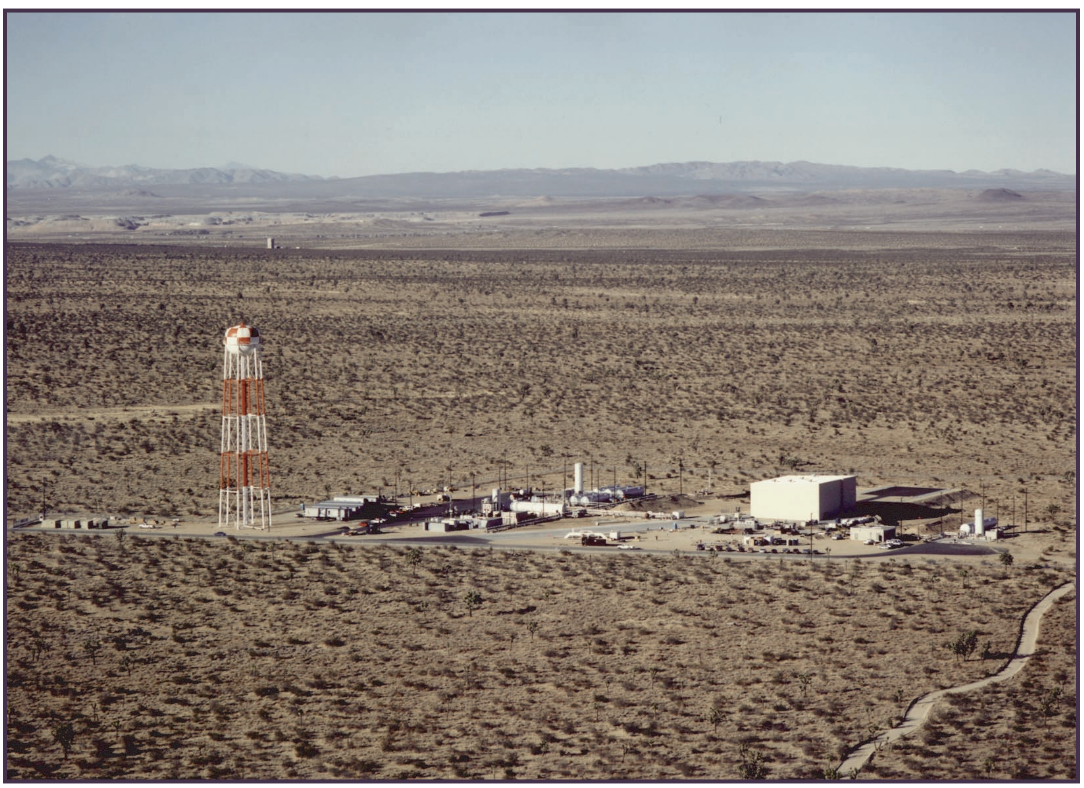

Figure 5.—X-33 Launch Facility–Haystack Butte, Edwards AFB.

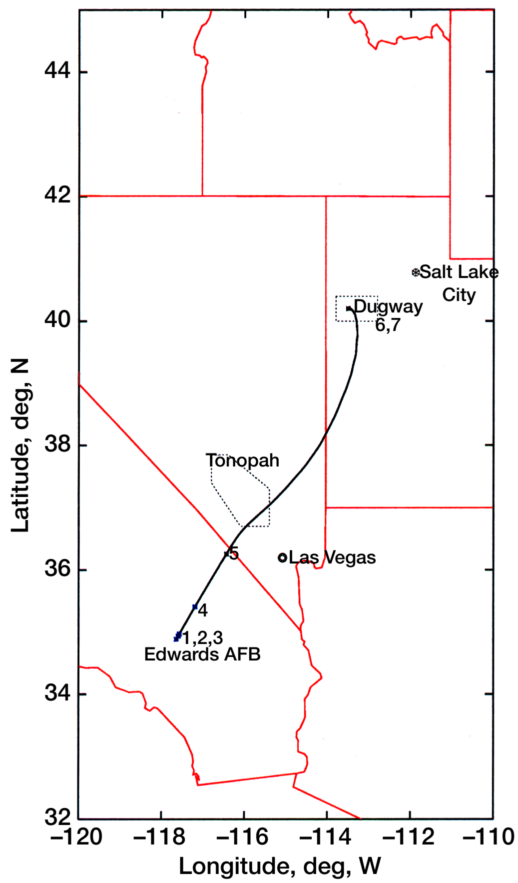

Figure 6.—GTX Phase 1 Flight Demonstrator Trajectory Map.

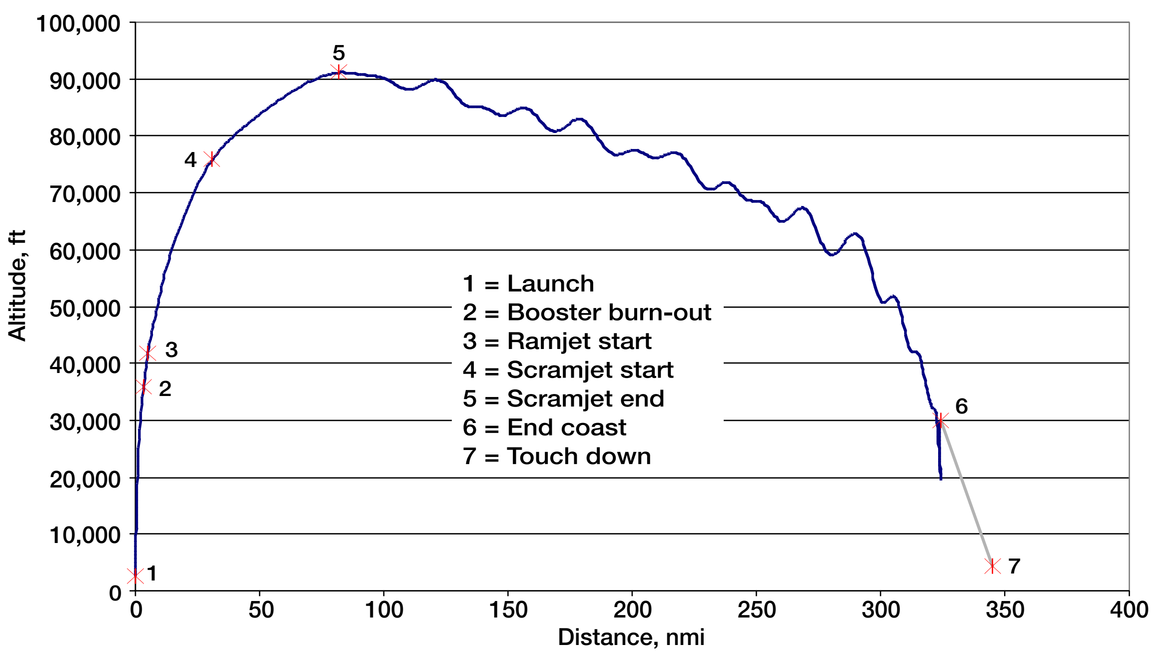

Figure 7.—GTX Phase 1–Flight Demonstrator Distance Plot.

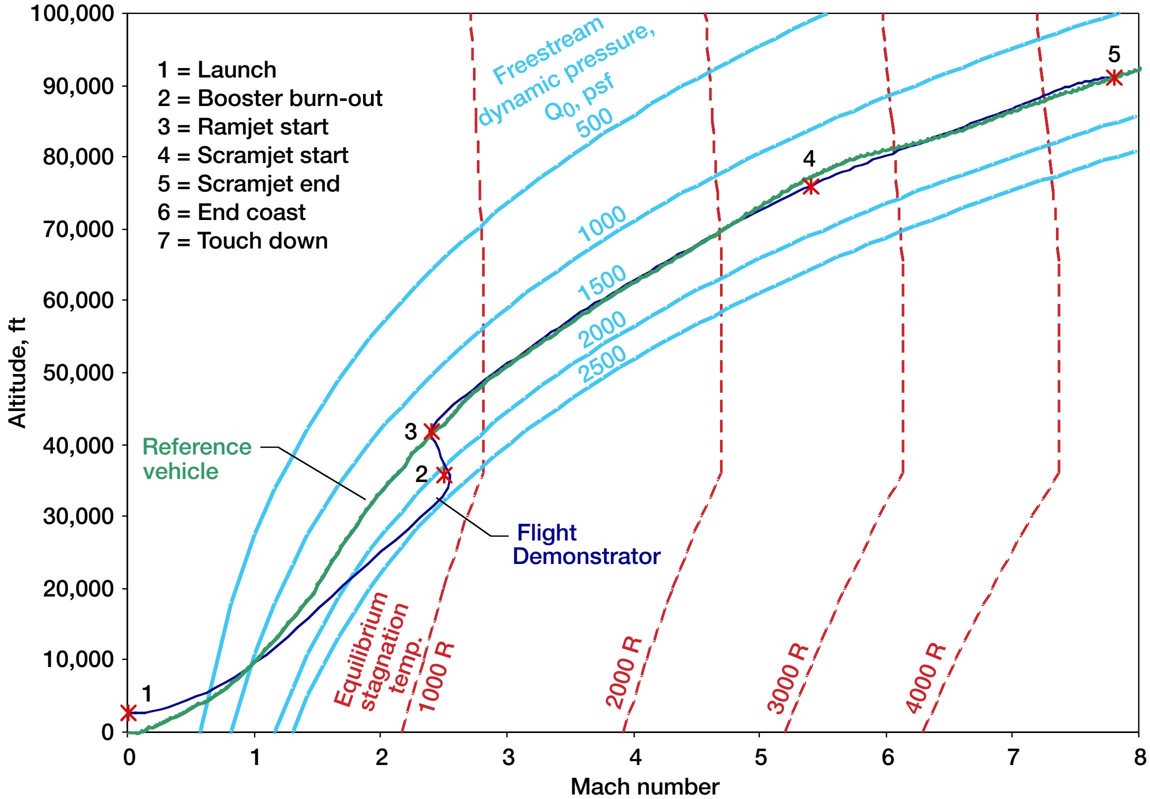

Figure 8.—GTX Phase 1–Flight Demonstrator Trajectory Comparison.

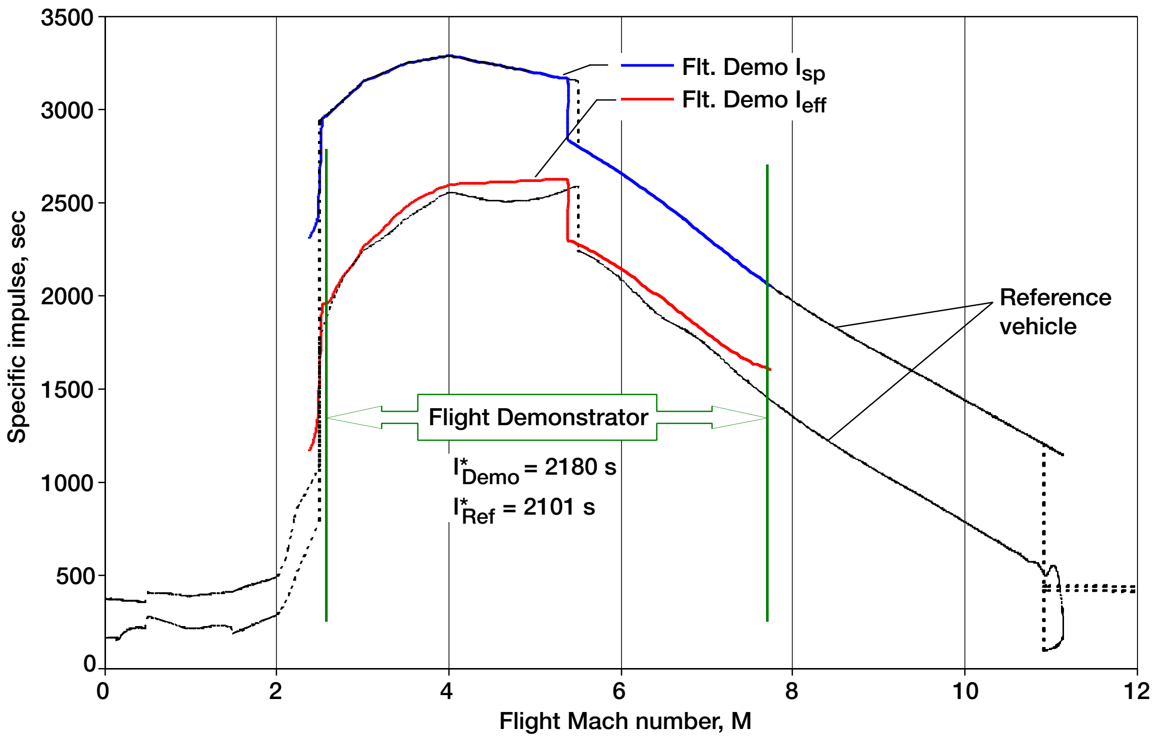

Figure 9.—GTX Phase 1–Flight Demonstrator Performance.

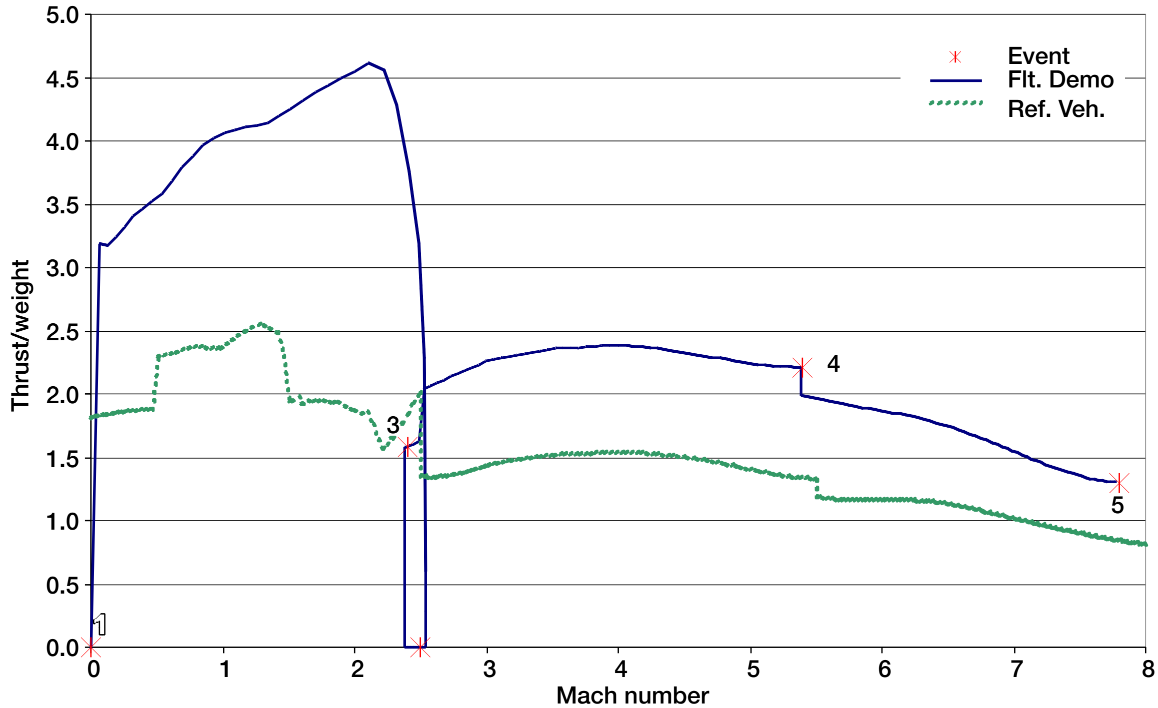

Figure 10.—GTX Phase 1-Thrust to Weight Comparison.

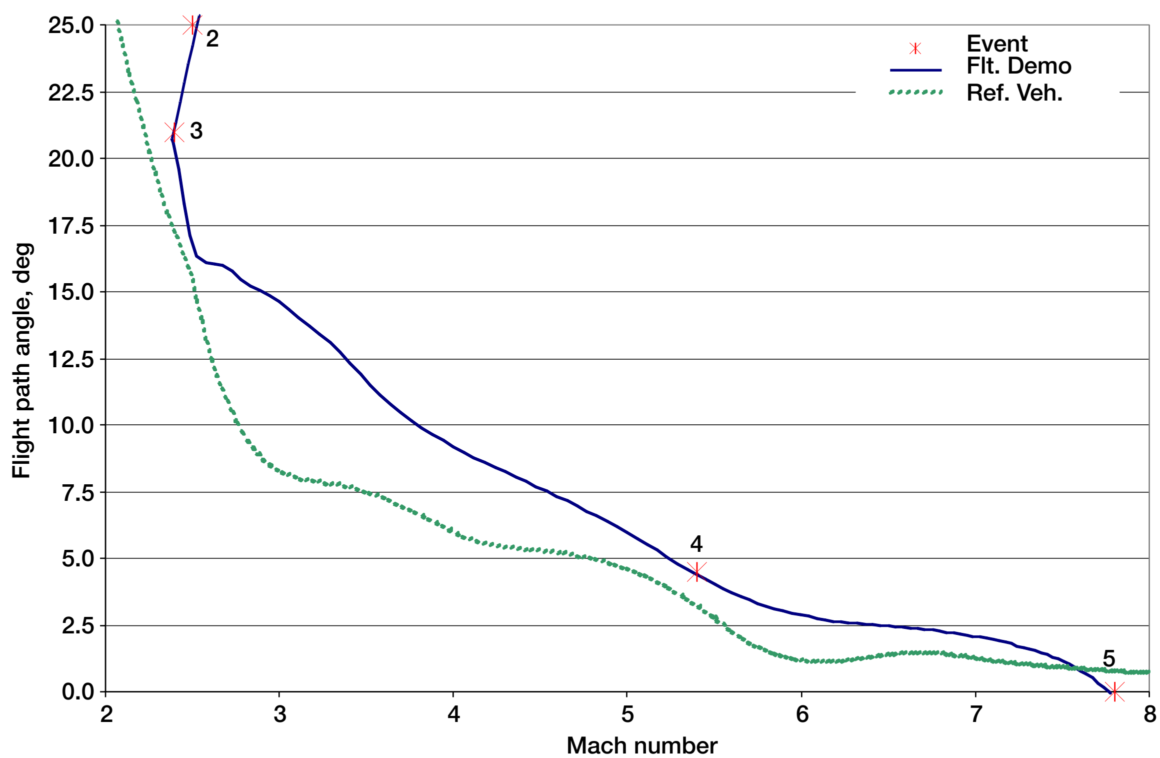

Figure 11.—GTX Phase 1–Flight Path Angle Comparison.

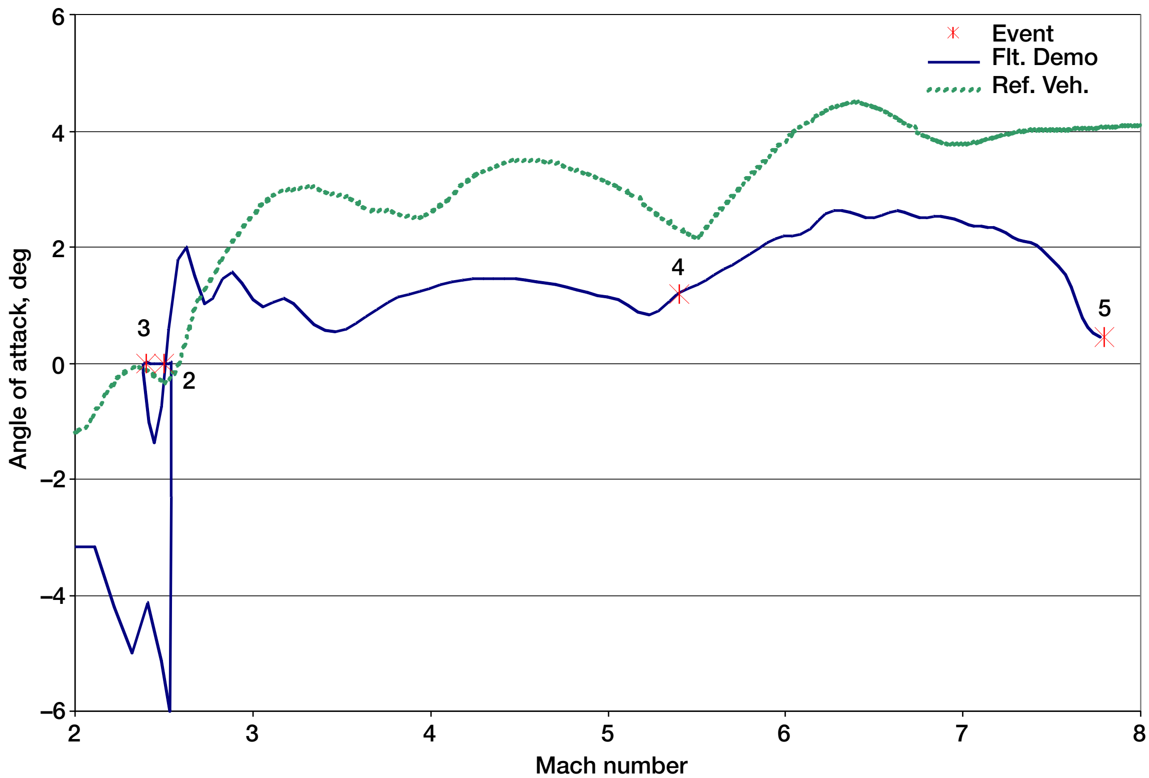

Figure 12.—GTX Phase 1–Angle of Attack Comparison.

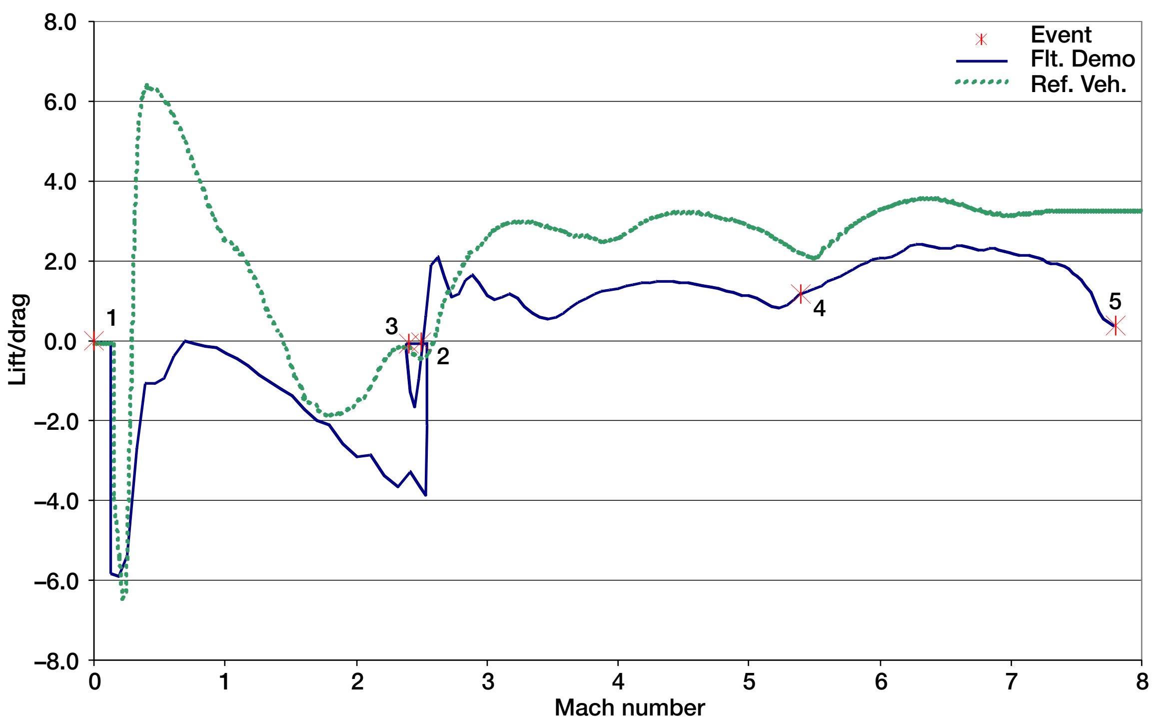

Figure 13.—GTX Phase 1–Lift to Drag Comparison.

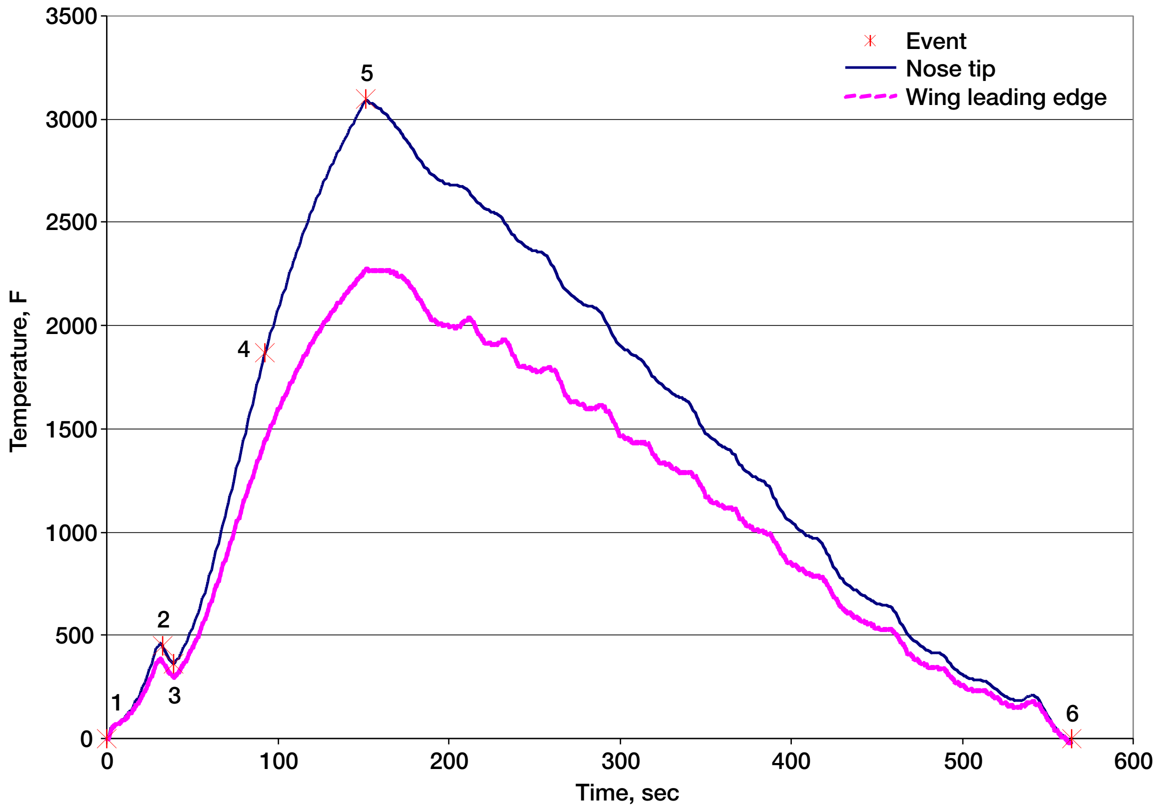

Figure 14.—GTX Phase 1–Nose and Leading Edge Temperatures.

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Thomas M. Krivanek, Joseph M. Roche, John P. Riehl, and Daniel N. Kosareo

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National Aeronautics and Space Administration
John H. Glenn Research Center at Lewis Field
Cleveland, Ohio 44135–3191

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13. ABSTRACT ***(Maximum 200 words)***

The rocket based combined cycle (RBCC) powered single-stage-to-orbit (SSTO) reusable launch vehicle has the potential to significantly reduce the total cost per pound for orbital payload missions. To validate overall system performance, a flight demonstration must be performed. This paper presents an overview of the first phase of a flight demonstration program for the GTX SSTO vehicle concept. Phase 1 will validate the propulsion performance of the vehicle configuration over the supersonic and hypersonic air-breathing portions of the trajectory. The focus and goal of Phase 1 is to demonstrate the integration and performance of the propulsion system flowpath with the vehicle aerodynamics over the air-breathing trajectory. This demonstrator vehicle will have dual mode ramjet/scramjets, which include the inlet, combustor, and nozzle with geometrically scaled aerodynamic surface outer mold lines (OML) defining the forebody, boundary layer diverter, wings, and tail. The primary objective of this study is to demonstrate propulsion system performance and operability including the ram to scram transition, as well as to validate vehicle aerodynamics and propulsion airframe integration. To minimize overall risk and development cost the effort will incorporate proven materials, use existing turbomachinery in the propellant delivery systems, launch from an existing unmanned remote launch facility, and use basic vehicle recovery techniques to minimize control and landing requirements. A second phase would demonstrate propulsion performance across all critical portions of a space launch trajectory (lift off through transition to all-rocket) integrated with flight-like vehicle systems.

14. SUBJECT TERMS

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