Research in Hypersonic Airbreathing Propulsion at the NASA Langley Research Center

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Abstract

The NASA Langley Research Center has been conducting research for over 4 decades to develop technology for an airbreathing-propelled vehicle. Several other organizations within the United States have also been involved in this endeavor. Even though significant progress has been made over this period, a hypersonic airbreathing vehicle has not yet been realized due to low technology maturity. One of the major reasons for the slow progress in technology development has been the low level and cyclic nature of funding. The paper provides a brief historical overview of research in hypersonic airbreathing technology and then discusses current efforts at NASA Langley to develop various analytical, computational, and experimental design tools and their application in the development of future hypersonic airbreathing vehicles. The main focus of this paper is on the hypersonic airbreathing propulsion technology.

Introduction

The NASA Langley Research Center (LaRC) has an ongoing research program for over 4 decades to develop hypersonic airbreathing propulsion technology for possible applications in hypersonic airplanes, launch vehicles, and missile systems. A number of other organizations within the United States and other countries have also been involved in this endeavor since the 1950's. However, an airbreathing-propelled hypersonic vehicle has not yet been realized. The fastest aircraft propelled by an airbreathing engine is the SR-71 Blackbird that reaches speeds slightly over Mach 3 using a turbojet engine. Missiles have used ramjet propulsion to fly up to about Mach 5. What is limiting our ability to "fly" faster, in fact, all the way to orbit? It has to do with a number of technological challenges that still need to be overcome (Ref. 1). For example, an efficient and reliable propulsion system is necessary that makes use of the oxygen within the earth's atmosphere for combustion of the fuel. This engine should be able to

operate over an extraordinary large range of flight conditions, including speeds from zero to orbital and altitudes from sea level to the upper level of the atmosphere. It is necessary to integrate several engine cycles from low to high speed, each working efficiently through its operating range, while at the same time, being effective aerodynamically. The airbreathing engine for hypersonic speeds should be able to accomplish stable, efficient mixing and combustion within a reasonable size combustor. Similarly, the vehicle should possess necessary structural integrity for a reusable system despite the extremely hostile environmental conditions. This means that the conventional practice of using higher temperature materials for vehicle structure as speed increases is not adequate. Rather, new materials and active cooling of airframe and engine structure is required. Another challenge is in developing reliable design tools that can be used for aerodynamic and propulsion system design and performance prediction. These tools need to model a very broad spectrum of physical and chemical phenomena such as laminar/transitional/turbulent flow, real gas effects in external air flow, mixing and combustion, turbulence and chemical kinetics interactions, etc. Finally, it is necessary to demonstrate that the hypersonic vehicle along with its engine system is capable of routine operations. This requires detailed analysis of complete vehicle, extensive ground testing, and flight testing of experimental vehicles. One of the major reasons for the slow progress in meeting these technological challenges in the development of a hypersonic airbreathing vehicle has been with the low level and cyclic nature of funding.

NASA began its research in supersonic combustion ramjet (scramjet) engines in the 1960's with a program to advance the technology for manned vehicles (Ref. 2). The Hypersonic Research Engine (HRE) project provided the focus for this effort; the original goal was to test a regeneratively cooled, flight-weight engine on the X-15 research airplane. The HRE was an axisymmetric engine with a translating spike to position the shocks and to control

the inlet airflow over a Mach number range of 4 to 8. Combustor operation was controlled by streamwise staging of fuel injection over the Mach number range. Although the original goal of flight tests was never achieved, two engines were built and ground tested for structural design and internal performance. The full-scale Structural Assembly Model (SAM), a flight-weight model constructed of Hasteloy-X (shown in Fig. 1), with hydrogen cooling, was tested in the NASA Langley 8-Ft High Temperature Tunnel (8'-HTT). The full-scale Aerothermodynamic Integration Model (AIM), a boiler-plate model constructed of Nickel-200 with water cooling and hydrogen fuel burning, was tested in the NASA Glenn Plumbrook Hypersonic Test Facility (HTF). These tests verified the structural and cooling design and demonstrated the feasibility of good internal thrust performance over a range of flight speeds.

Figure 1.- HRE SAM in NASA Langley 8 ft. HTT

Although good internal thrust performance remains a primary goal for any propulsion system, the engine must also achieve high installed performance (i.e., internal thrust minus external drag), requiring proper aerodynamic integration of the propulsion system with the remainder of the vehicle. This led to research in scramjet propulsion concepts that can be intimately integrated with the airframe of the vehicle as shown in Fig. 2. During the 1970's and early 1980's, NASA Langley was engaged in an in-house program to develop an airframe-integrated scramjet concept and ground demonstrate its performance potential. This program included research on engine components (inlets, combustors, and nozzles), computational fluid dynamics for internal reacting and non-reacting flows, component integration (sub-scale engines), high-temperature materials and structures, and flow diagnostics. In addition, the Department of Defense (DoD) and industry were also involved in this

technology development, again only at modest level of effort.

These research efforts were substantially augmented during the National Aero-Space Plane (NASP) Program that spent over $3B between 1984 and 1995. The goal of this program was to develop an airbreathing-propelled, single stage to orbit vehicle.

Figure 2.- Airframe-Integrated, Airbreathing-Propelled Hypersonic Vehicle

The NASP program brought together individual efforts at NASA, DoD, and industry under one umbrella. Facilities were revived, advanced computational tools were developed, and a large number of engineers were trained in hypersonics over a short period of time. Broad-based university research programs in hypersonic technologies were funded. At its peak, over 5,000 engineers and scientists were involved in the program. As part of the hypersonic airbreathing propulsion technology development, a number of small- and large-scale, dual-mode scramjet engines were designed and ground tested. Based on these tests, a large-scale (10 x 16 x 142 inches) Concept Demonstration Engine (CDE), shown in Fig. 3, was fabricated towards the end of the NASP program and tested in the 8'-HTT at Mach 6.8 simulated flight conditions. The objectives of the CDE test were to demonstrate performance and operability limits of the large-scale integrated scramjet engine and to verify flowpath design methods for application to flight. Even though significant progress was made during this program in all aspects of hypersonic technology, it was still not adequate to produce an airbreathing-propelled, single stage to orbit vehicle. The NASP program did not produce the X-30 research vehicle because of constraints imposed on size and cost.

Recently, once again, the United States has initiated a number of hypersonic technology programs within NASA and DoD. One such program is NASA's Hyper-X (X-43) program, with a goal to fight demonstrate and validate as many technologies

as possible for a scramjet-powered vehicle at Mach 7 and 10. NASA Langley and Dryden Flight Research Center are jointly leading the Hyper-X program. The first flight for Mach 7 in June 2001 was unsuccessful due to the booster failure. The second flight for Mach 7 is scheduled for December 2001, with a Mach 10 flight to follow. In another effort, started in 1997 under NASA's Advanced Space Transportation Program (ASTP), wind tunnel testing of a Rocket-Based Combined Cycle (RBCC) engine was
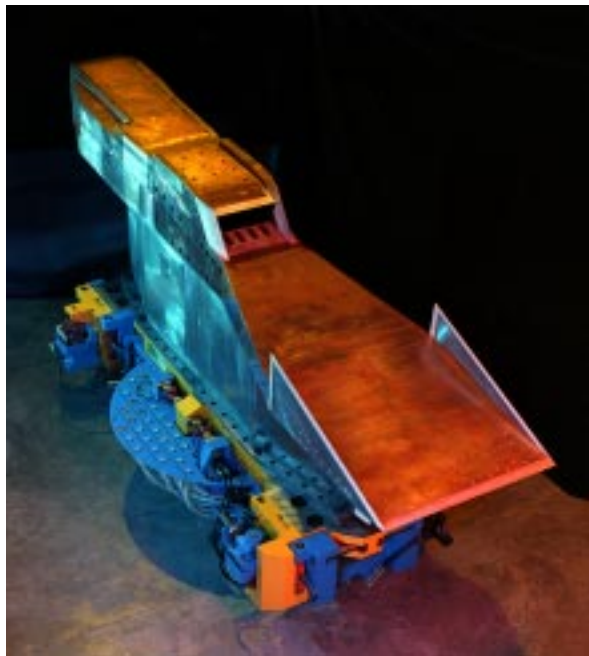

Figure 3.- CDE in 8-'HTT

conducted for airframe-integrated scramjet flowpath concepts containing integrated rockets for low and high-speed thrust. A new follow-on technology program that focuses on NASA 3rd Generation Reusable Launch Vehicle (RLV) goals, is currently being planned and implemented under the NASA Marshall Space Flight Center led Advanced Space Transportation Program. The goal of this program is to reduce cost and increase reliability and safety. Assuming 1,000-2,000 flights per year, the 3rd generation goals are $100/lb of payload to Low Earth Orbit (LEO) and 10^{-6} failure rate. The program includes design tools development and ground and flight technology demonstration. Systems studies are being used to evaluate numerous vehicle architectures that include single-stage to orbit (SSTO) or two-stage to orbit (TSTO) with vertical or horizontal takeoff. A number of low to high Mach number propulsion cycles are being developed with hydrogen, hydrocarbon, or dual fuel.

Within DoD, several hypersonic programs have been undertaken in the past 5 years. These include the Air Force Research Laboratory's Hypersonic Technology (HyTech) Program, the Defense Advanced Research Programs Agency (DARPA) Affordable Rapid Response Missile Demonstrator (ARRMD) Program, and the Army Scramjet Technology Development Program. In addition, United States Air Force (USAF) is also studying hypersonic aircraft under future strike force. There is an ongoing effort in the United States to bring all of these NASA and DoD programs under a joint National Hypersonic Program.

The paper briefly discusses the outcome of several systems studies comparing rocket versus airbreathing-propelled aerospace vehicles, including various low- to high-speed propulsion cycles being developed for hypersonic airbreathing-propelled vehicles to orbital speeds. This is followed by a discussion of computational and experimental tools for hypersonic propulsion flow path design and performance evaluation. Although NASA Langley has strong capabilities in aerothermodynamics, structures and materials, and controls, all critical to the successful development of hypersonic vehicles, they are not discussed in the paper. Finally, the paper provides a brief overview of future technological challenges.

Engine Cycles for Hypersonic Airbreathing Vehicles

Engine selection for a hypersonic airbreathing vehicle is dependent on the mission (Ref. 3). Efficient hypersonic flight requires scramjets. For this paper, scramjet refers to a dual-mode scramjet that can operate in supersonic, subsonic, and mixed supersonic/subsonic modes without the use of a second throat. Dual-mode scramjets can operate over a large speed range from about Mach 3 to 15. For space launch vehicles, a rocket is required for orbit insertion, in-space maneuvering, and deorbit. Below Mach 3-4, numerous options are available such as turbojet, rocket, air-augmented rocket, ejector ramjet, conventional ramjet, pulse detonation, or liquid air cycle engine (Ref. 4). These engine cycles can be put together into numerous combinations, all of which are captured in Refs. 4 and 5. They are historically divided into combined-cycle engines (Ref. 4) and combination engines (Ref. 5). The two approaches are illustrated by the RBCC engine and turbojet-scramjet-rocket combination engine (TBCC). The RBCC engine, illustrated in Fig. 4, operates in 3 modes: (1) air augmented rocket from Mach 0 to 3; (2) dual-mode scramjet to Mach 10 – 15, and (3) rocket to orbit. The

turbojet-scramjet-rocket combination engine is illustrated in Fig. 5. This over/under turbojet/dual-mode scramjet pure airbreathing system operates to Mach 15, at which point the rocket, located in the scramjet external nozzle, takes over. The rocket mounted within the scramjet nozzle allows thrust vector through the vehicle center of gravity.
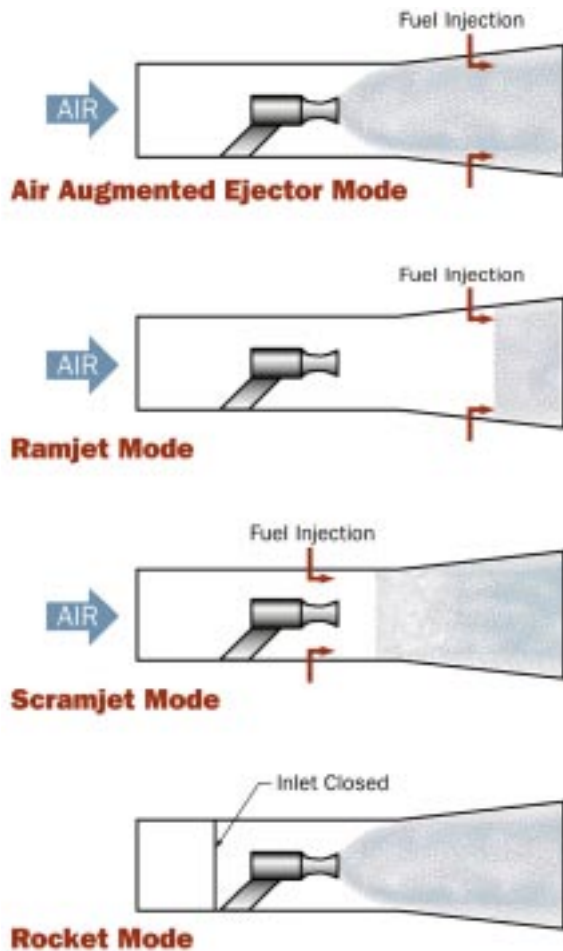

Figure 4.- RBCC Operating Modes and Performance Benefits

Due to the apparent simplicity, the RBCC approach has gained significant attention within the United States. Rocket-based engines (Ref. 4) have potential for "synergistic" integration of rockets into the airbreathing (ramjet or scramjet) flowpath due to the large area ratio nozzle afforded by the airbreathing flowpath. The rockets can operate as ejectors or air-augmented rockets, and may include a liquid air compression engine (LACE) system to off-load part of the rocket oxidizer requirements. The rocket systems used in the RBCC flowpath are assumed to be capable of providing orbital and in-space propulsion needs.
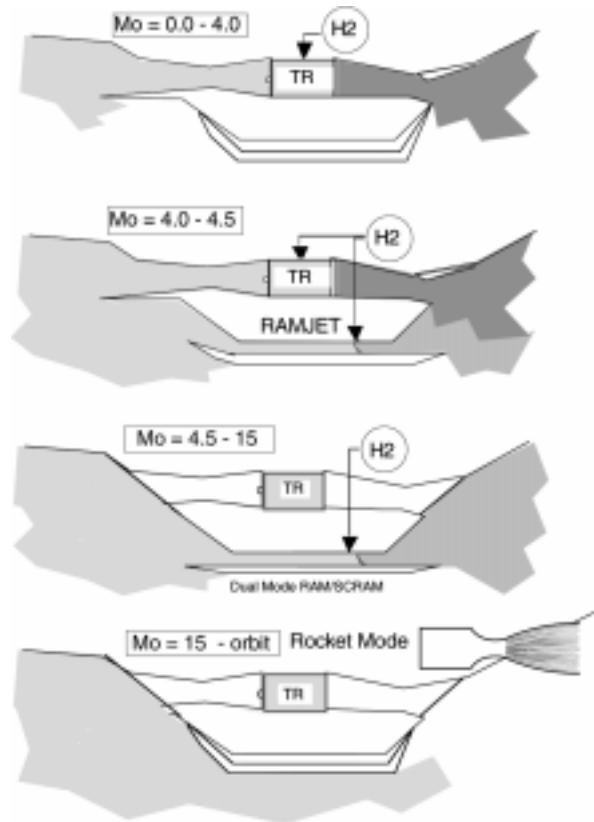

Figure 5.- Turbojet-Scramjet-Rocket Combination Engine

System Analysis

Systems studies are required to identify potential vehicle configurations, components, and approaches, and focused technology development. For rocket vehicles, such studies are straightforward and well understood. The rocket performance is essentially independent of the vehicle aerodynamics, and the vehicle aerothermal (aerodynamics and thermal) environment is simply tied to trajectory. For most rocket systems, the vehicle structure is cold, protected by a passive thermal protection system. The dominant design driver is vehicle structural efficiency. In addition, the rocket weight and thrust efficiency have been studied and optimized for years.

For an airbreathing vehicle, systems studies are more complex due to the highly integrated nature of the airframe and engine (Ref. 6). Air flow requirements of hypersonic engines require that they be aerodynamically integrated with the airframe to capture as much of the vehicle pre-compressed flow as possible (Ref. 5). Therefore, the vehicle and engine must be developed together. A formal design process

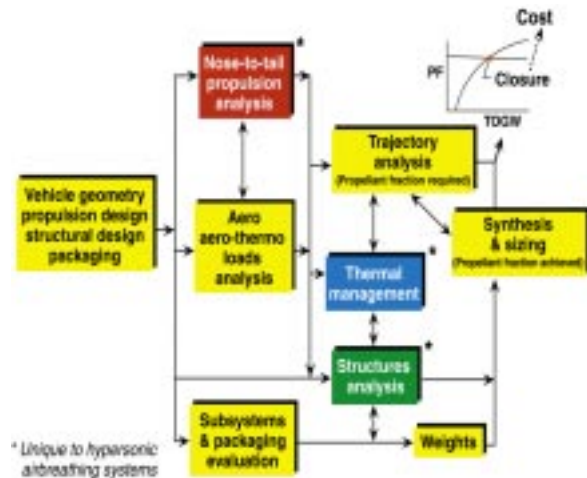is illustrated in Fig. 6. Engine and aerodynamic performance, structure, weight, systems and

Figure 6.- Design Method for Airbreathing Hypersonic Systems

Table 1.- Vehicle Fidelity Assessment

packaging, and thermal management are iterated as the vehicle is flown to determine the 'required' propellant. Finally, the vehicle is resized to achieve the propellant required to meet the mission and define a "closed" configuration. Because of a relatively low level of maturity of this technology, vis-à-vis rocket-powered configurations, substantial improvements, refinements, and optimization are not only possible but a significant part of the process.

Systems analysis methods for airbreathing vehicles have evolved dramatically over the past 20 years. These methods can be executed at several levels (Ref. 3), as noted in Table 1. The lowest level, designated "0" in Table 1, does not require a physical vehicle or engine geometry. Level zero analysis utilizes ideal engine cycle performance, historical L/D and C_d values for aerodynamic performance, design tables (or weight fractions) for structure and components weight, the "rocket equation" for flight trajectory, and estimates for packaging. Uncertainty in the design of airbreathing vehicles using this approach has been shown to exceed hundreds of percent.

Level 1 analysis utilizes uncertified cycle performance and/or CFD, impact theory, unit or uncertified finite element model (FEM) weights, single equation packaging relations, and energy state vehicle performance. This level of analysis does not capture operability limits, and thus has large uncertainties that can exceed 100%.

Level 2 analysis uses the same methods as used in Level 1 (but certified) for propulsion, aerodynamics, structure and weights, trimmed 3-DOF (degree of freedom) vehicle performance analysis, and multiple equation, linear or non-linear packaging relations. Certification is only achieved by demonstration that the methods work on the class of problems being simulated. For example, Level 2 analytical models utilize corrections for known errors, such as inlet mass spillage, relevant empirical fuel mixing models (Ref. 7), shear and heat flux models (Ref. 8), etc. This empirical approach is based on experimental data from wind tunnel tests, structural component tests, etc. Higher level methods (CFD, FEM) are used to refine the vehicle closure. Uncertainty in the projected closure weight for an airbreathing system using this approach is estimated at 10-20% for CFD-rich to 15-30% for certified cycle-rich analysis.

The highest design level (Level 3) is achieved only by having a significantly large fraction of the actual vehicle manufactured and tested. Wind tunnel and other ground testing provide less verification than flight tests. Although numerous components have been built and ground tested, flight data is required for the highest level of design. This has not yet been done for a hypersonic airbreathing vehicle; however, the projected uncertainty with this approach is expected to be 5-10%.

Whatever the level of system analysis, closure is achieved by sizing the vehicle so that the propellant fraction required for the mission is equal to the propellant fraction available (packaged within the sized vehicle). However, the reported closure weight is only as good as the lowest level of analysis used in the "closure."

Summary of Airbreathing vs. Rocket Vehicle Study

Vehicle takeoff gross weight and/or dry weight have long been used as figures of merit to show benefits of rocket and/or airbreathing systems. Mission flexibility, design robustness, safety, reliability, and cost are other factors that are being considered to quantify the benefits of airbreathing versus rocket systems. This section will discuss both sets of figures of merit.

Airbreathing and rocket vehicle designs for space access have been examined since the early 1960's. In 1990's, two major studies were performed by NASA – The 1993 Access to Space (ATS) Study (Refs. 9), and the 1997 Advanced Launch Vehicle Study (led by Talay and Hunt of NASA Langley, unpublished). The ATS Study established a reference, horizontal takeoff/horizontal landing (HTHL) airbreathing/rocket SSTO vehicle design for 25Klb payload delivery to the Space Station in a lifting-body configuration (provided by NASA Langley). The ATS team also established a HTHL/TSTO design and a vertical takeoff/horizontal landing (VTHL) SSTO rocket vehicle design.

The 1997 Advanced Launch Vehicle Study was a follow-up evaluation to the ATS Study with a focus on enhancing the design fidelity of the VTHL/ SSTO rocket and the HTHL/SSTO airbreather, on understanding some of the design pros and cons, and on quantifying the mission flexibility of the two SSTOs. The airbreather won all mission flexibility contests.

The 1994 Commercial Space Transportation Study (CSTS) examined mission/market elasticity and provided space access goals for the Agency -- reduce the cost and increase the reliability, safety and flexibility of going to space. The vision was a vehicle that operated more like a commercial airliner than a research vehicle. NASA has adopted these goals – the 2025 "Third Generation" goals are $100/lb cost of payload delivery and 10^{-6} failure rate. These goals may sound unobtainable based on current values of $10,000 per pound to orbit and l/200 crew losses, but they assume a significant growth in flight operations, from 4-8 per year to 1,000 to 2,000 per year.

Within the United States, system studies are continuing to search for the best airbreathing vehicle architectures. The NASA MSFC led ASTP is sponsoring screening studies to conduct a broad review of all prospective configurations, approaches, and architectures. These architectures include HTHL and/or VTHL SSTO and TSTO systems; hydrogen, hydrocarbon, and/or dual-fuels; and many propulsion options including RBCC and TBCC which incorporate dual-mode scramjet capability, as discussed previously.

A preliminary vehicle/propulsion matrix examination, designated as the Airbreathing Launch Vehicle (ABLV) Study, was sponsored by the Advanced Reusable Technology (ART) project of ASTP at NASA Marshall in 1998. The study ended in 2000 and was led by NASA Langley. It was supported by Boeing under a contract. The mission was to deliver 25Klb payload to the Space Station in a single stage. Fourteen unassisted HTHL and four VTHL vehicle variants were examined. The propulsion emphasis for the HTHL vehicles was on under-slung, single-duct RBCCs; and under-slung, two-duct turboramjet (over)/dual mode ramjet (under). The reference vehicle design to which all other vehicles were compared was a lifting body with an under-slung one-half duct (over)/single duct (under) – LACE ejector ramjet/dual-mode scramjet propulsion system. Some results from the ABLV study are presented in Fig. 7.

Four airbreathing engine approaches were considered in the vehicles represented in Fig. 7, with primary difference being characterized by the vehicle oxygen fraction. Two approaches resulted in low oxygen fraction: the turbojet-scramjet-rocket 'TRJ+SJ+R' combination engine and ScramLACE combined cycle engine. The other airbreathing approaches used more LOx; first for low speed rocket operation to Mach 3, and second, for lower speed rocket takeover from the scramjet mode. The heavier RBCCs in Fig. 7 ('min' and 'max') had higher LOx fraction because of the limited (to about Mach 10-12) scramjet operating range. The SSTO rocket, represented in Fig. 7, was a VTHL design from the Advanced Launch Vehicle Study. It had a full-flow, staged combustion cycle rocket main propulsion system with an uninstalled thrust-to-weight of 83.

RBCC-powered VTHL SSTO's for the same mission sized-out at about 1.6 to 1.7 Mlbs. In the ABLV study, these vehicles appeared more favorable than the unassisted horizontal takeoff counterpart because of the lack of the fixed wings sizing required for takeoff at guideline speeds; however, they still required fixed wings for flying the ascent, lifting trajectory and for horizontal landing.

Two Stage to orbit HTHL launch configurations are currently being examined – a' la the
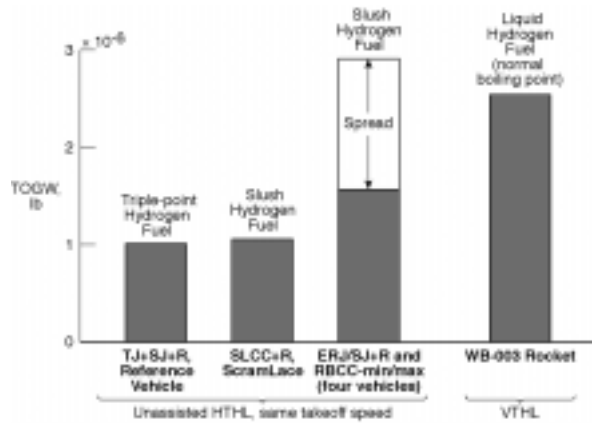
Figure 7.- Airbreathing Benefit to Vehicle Size
(25Klbs to Space Station)

Figure 8.- Launch Window ΔV Penalty

TSTO system established in the ATS study in 1993. The TSTO ATS system employed a lifting body for both first and second stages, with the upper stage powered by a rocket. It was an all-liquid hydrogen-fueled system that staged at Mach 5. Hydrocarbon fueled, hydrogen fueled, and/or dual fueled HTHL TSTO systems are being examined with staging Mach numbers to 8 for the hydrocarbon-fueled first stage and to Mach 16 for the hydrogen-fueled first stage.

Mission Flexibility

Extensive evaluation of mission flexibility for the VTHL/SSTO rocket and the HTHL/SSTO airbreather was performed in the 1997 Advanced Launch Vehicle Study at Langley. The study showed that with the airbreather, during flight within the atmosphere, aerodynamic lift and a substantially enhanced engine efficiency could be used on ascent to change inclination and achieve alternate orbits or intercept alternate orbits—to effectively chase the ascending node! This allows the airbreather a greatly expanded launch window vis-a-vis the SSTO rocket as indicated in Fig. 8.

Offset launch for rapid rendezvous up to 15° have been calculated for the airbreather, compared to only 2° for the rocket system. Other launch benefits of the airbreather include: lower orbital inclination access; changing orbit on demand during launch; synergistic plane changes; and mission recall. Basing flexibility also favors the airbreather because of runway requirements versus launch pad requirements. Likewise, on reentry, the airbreather's large cross range (over 2.5 times the rocket vehicle) allows rapid/immediate de-orbit to a safe landing over a huge
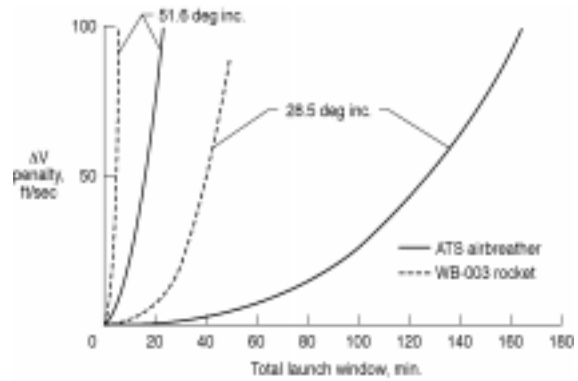
landing footprint. The airbreather also has self-ferry capability, which greatly opens the number of potential affordable recovery sites. This feature also eliminates the need for a carrier vehicle to bring the launch vehicle back to home base when landing elsewhere as with the Space Shuttle.

Design Robustness

Quantification of relative design robustness was a fall-out of the 1997 Advanced Launch Vehicle Study. The airbreathing SSTO also had a more robust design than the SSTO rocket due to larger vehicle dry weight fractions (26% versus 9%), structural mass fractions (14% versus 4.5%), and reduced weight growth sensitivity – one pound increase in dry weight for the rocket systems adds over 10 pounds to its TOGW while one pound of dry weight increase for the airbreather adds only 3.7 pounds to its TOGW.

Safety

Safety issues include abort, powered landing and go around, propellant flow rates, and failure rates. HTHL airbreathing vehicles allow abort during the takeoff role, or soon after takeoff because the system is designed to land partially loaded. Vertical takeoff (VTO) eliminates the potential for abort over the initial part of the trajectory. Rocket systems, for the foreseeable future, will be limited to VTHL, with landing only possible after burning or dumping propellants. Once again, the cross range capability of airbreathing vehicles enhances safety by dramatically increasing the number of potential recovery sites.

For rockets, VTO requirements and the huge propellant loads result in thrust requirements more than 6 times that of an airbreathing system ($3 + X10^6$ vs. $5 + X10^5$). The associated maximum propellant flow rate differences are even more troublesome:

7,500 lb./sec. vs. 120 lb./sec. Because of these large flow rates, large rocket engines fail catastrophically. Fixing this critical limitation has been estimated to double the engine weight, with a 10:1 impact on the takeoff gross weight (TOGW). Recently completed failure rate studies show a significant benefit for the airbreathing vehicle. These studies were performed on a pure rocket-powered and a horizontal takeoff RBCC-powered vehicle. The RBCC vehicle used 40 'low' pressure rocket chambers, producing 234 Klb thrust, compared to 7 'high' pressure rocket chambers producing 3 Mlb thrust for the all rocket vehicle. Considering rocket durability and thrust levels/duration for the two systems, the all rocket system was estimated to have an order of magnitude higher failure rate than the RBCC vehicle. The turbine-based system has not been assessed yet but is expected to be even better. In addition, the scramjet failure rate is expected to be miniscule in comparison because it utilizes rocket cooling technology, but has $1/4^{th}$ the peak or $1/10^{th}$ the average thermal load of a rocket (Fig. 9).
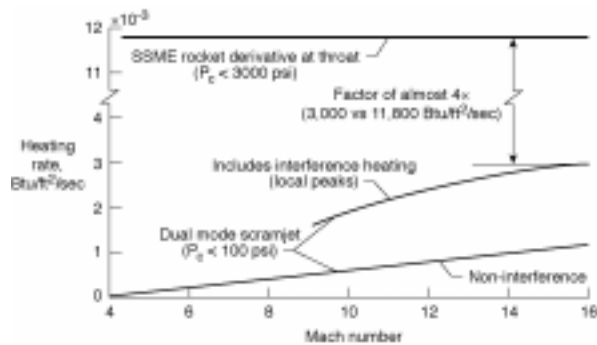

Figure 9.- Maximum Heating Rate in Combustor/Combusion Chamber for SSTO Vehicles

Another safety feature of the airbreathing systems, particularly turbine-based systems, is the potential for powered go around. These and other issues are being considered in the development of safety models.

Reliability

Airbreathing and rocket launch vehicle systems and subsystems are assumed to utilize the same technologies so they should not serve as a discriminator. The reliability of airbreathing systems is expected to be high due to lower thermal loads and lower pressure requirements for the fuel turbo pump. For a turbine-based system, the turbojet reliability is well known, and will not be compromised by short duration (5 min.) operation at full power, or near idle

cruise of a lightly loaded vehicle. Small rockets (order of 100 Klb thrust) for orbital insertion can be over designed without serious impact on vehicle performance or size (0.1% increase in TOGW for the HTHL airbreathing vehicle vs. 20% for the VTHL rocket).

Airframe thermal loads are clearly greater for the airbreathing vehicles than for the rocket vehicles. Structural designs for the airbreathing vehicles have been demonstrated in ground tests, but some, like the vehicle leading edges, require moderate-pressure fuel cooling. Durability of such systems must be validated.

Cost

Cost models for airbreathing systems are being developed under the ASTP to address DDT&E, production, operations, and life-cycle costs. Technology development costs are certainly greater for the airbreathing vehicle. These development costs have been estimated at about $12B. DDT&E and production cost differences between the two systems are generally considered to be small. Operational cost models for airbreathing vehicles do not exist yet. However, a HTHL single stage vehicle is inherently simpler than a vertical takeoff (single or two stages) system. Therefore, the turnaround time and workforce requirements should be less for the airbreathing vehicle.

Other operational cost considerations include propellant usage, vehicle losses, and transportation. Propellant requirements for the rocket increase operational cost through additional cost of fuel and increased logistics. Propellant cost is considered trivial by the current launch vehicle community, which is accustomed to huge budgets and infrequent flights. This attitude probably will change with increase in flight frequency, as in the airline business.

Cost of vehicle replacements and other costs associated with catastrophic loses clearly favor the airbreathing system. Transportation costs, such as returning the Shuttle from DFRC to Kennedy Space Flight Center, again favor the airbreathing vehicle, particularly for vehicles with turbojet systems. Cost reduction will also be achieved by alternate use such as endoatmospheric operations, hypersonic cruise operations, and other military applications. These activities eventually will share development and operations costs.

Flight vehicle utilization probably has the greatest impact on operating cost and is often over

looked. Even the Shuttle cost can be reduced by more than an order of magnitude by increasing the number of flights to 1,000-2,000 per year. Do the uncertainties associated with the existing systems risk limit vehicle utilization?

Computational Tools for Scramjet Engine Analysis and Design

Computational fluid dynamics (CFD) has emerged as an extremely valuable and cost-effective engineering tool in aerodynamic design and analysis. The usefulness of CFD can be attributed to many factors. With the rapidly increasing cost of wind tunnel testing, the development cost for a vehicle can be greatly reduced by the judicious application of relatively simple and inexpensive computational tools. Parametric investigations can be performed with short turn around time to discard designs with obviously poor performance. When the stage in a development cycle is reached where wind tunnel testing is required, CFD can be very helpful in model design and placement of instrumentation, and as a diagnostic tool to help explain certain unexpected flow phenomena observed in measured data. An area where computational tools have made significant progress is hypersonic airbreathing propulsion. These tools are being used effectively in the analysis and design of various engine components as well as the complete hypersonic vehicle. A combination of low to high fidelity computational tools is used in these applications depending upon the stage of the design and the level of confidence desired.

Computational analyses of inlets typically employ codes that solve the Euler equations, iterated with the boundary-layer equations for viscous effects, for initial design. More detailed calculations utilize either the parabolized Navier-Stokes (PNS) equations, or the Reynolds-Averaged Navier-Stokes (RANS) equations if significant flow separation must be considered. All of the calculations typically solve the steady-state equations so that simulations can be completed in reasonable times. Turbulence is modeled using either algebraic or two-equation turbulence models with empirical compressibility corrections and wall functions. Transition modeling is not typically employed. Thermodynamic properties are generally determined by assuming that the inlet flow behaves as a perfect gas or equilibrium air. Calculations are conducted on fixed grids of 100,000 to 2,500,000 points in multi-zone domains. A limited degree of dynamic grid adaptation is employed when necessary. The most limiting area of accurate simulation of complex inlet flow is the modeling of transition and turbulence. Some promising work is now underway to develop new algebraic Reynolds stress turbulence models (Ref. 10 and 11). Advances in large eddy simulations (LES), with sub-grid scale models appropriate for high-speed compressible flow, may allow this technique to be applied to inlet flows in the future (Ref. 12 and 13). However, transition modeling for high-speed inlets still remains mostly nonexistent.

Combustor analysis is typically done using either the PNS or RANS based codes, depending upon the region of the combustor being modeled and the degree of flow separation and adverse pressure gradient being encountered. Steady-state methods are normally used for complete combustor analysis and performance prediction. However, limited unsteady analysis may be used for understanding the physics of fuel air mixing or for combustion instabilities. Turbulence is again modeled using algebraic or two-equation models with empirical compressibility corrections and wall functions. A few codes model turbulence-chemistry interactions based on probability density functions. Fuel-air combustion is modeled with reduced reaction set. For example, hydrogen-air chemistry in a hydrogen-fueled scramjet is normally modeled using a nine species, eighteen reactions system (Ref. 14), although other mechanisms are available for use as the case may dictate. Hydrocarbon fuel combustion requires much more complex reaction mechanisms that must be further reduced to allow practical computations. Calculations in each case are typically conducted on fixed structured grids of 200,000 to 2,500,000 points in multi-zone domains. Typical run times on a Cray C-90 computer range from 10 to 300 hours. It is obvious from the computer resource requirements that these times need to be improved by a factor of 10 to 100 if advanced computational tools are to be routinely used in the design process. Recent work by Thomas et al. (Ref. 15) on multi-grid methods offer promise for substantial increase in convergence rates, but the application of these methods to high-speed reacting flows offer additional challenges. Dynamic grid adaptation will become even more important for accurately capturing the complex flow structure in the combustor, in particular, the shock interactions and vortical structure in the flow.

Turbulence modeling for high-speed reacting flows continues to be a challenge. Promising work is going on in this area using several approaches. Techniques using velocity-chemical composition probability density functions have been successfully applied to incompressible reacting flows, and are now being extended to model compressible reacting flows (Ref. 16). Although not ready for practical use in

combustor flow analysis, work is also underway to apply LES techniques to such flows. Sub-grid scale models utilizing a filtered mass density function for the LES of turbulent reacting flows appear particularly promising for the future (Ref. 17).

Nozzle analysis is usually less demanding than combustor and, in general, combustor analysis codes can be applied to nozzles. In many cases, Euler codes iterated with boundary layer codes can be used for initial engineering design studies. Due to expanding flow with favorable pressure gradient, nozzle flow is more readily amenable to PNS codes. However, finite rate analysis is required throughout the nozzle to assess the continuing degree of reaction, and to determine the extent of recombination reactions that add to the available thrust. Transition and turbulence modeling also continues to remain an issue for nozzles. In addition, the nozzle wall boundary layer may relaminarize locally in the region of large favorable pressure gradient and transition back to a turbulent boundary layer further downstream, thus posing an additional modeling challenge. The reduced kinetics models for nozzle flows appear to be reasonably accurate, although some further work may be warranted to improve the recombination process.

The ultimate use of the computational tools is in the nose-to-tail analysis and design of the complete vehicle. This requires the development of a system of computational codes that appropriately selects the required type of code for most efficient and accurate analysis. The codes in this system are made compatible by creating proper interfaces with each other. It is also possible to use a single RANS-based code with appropriate modeling for different regions. Reference 18 describes a number of codes that have been developed and applied in hypersonic vehicle analysis over the years. This paper briefly describes below one such recent code, VULCAN (Ref. 19), that has shown a lot of promise in the analysis of hypersonic airbreathing propulsion and is based on the state of the art CFD technology. An application of this code is also presented as an illustration.

VULCAN (Viscous Upwind aLgorithm for Complex flow ANalysis) is a cell-centered finite-volume, structured grid, multi-block code which solves the equations governing inviscid and viscous flow of a calorically perfect gas or of an arbitrary mixture of thermally perfect gases undergoing non-equilibrium chemical reactions. VULCAN allows the flow domain to be decomposed into regions in which the most suitable algorithm (elliptic or marching) can be utilized. The inviscid fluxes are computed using the MUSCL scheme of Vanleer with the approximate

Riemann solver of Roe or the low dissipation flux splitting scheme of Edwards. VULCAN solves spatially elliptic flow by marching the unsteady form of governing equations in time. Spatially hyperbolic flow (e.g. the Euler equations in supersonic flow) and parabolic flow (e.g., the parabolized Navier-Stokes equations) are solved by space marching while iterating the unsteady equations in pseudo-time to a steady state solution at each cross-stream plane.

A number of two-equation, k-omega type turbulence models have been incorporated in VULCAN. The code also has available an explicit algebraic Reynolds stress model. All the models have been modified to account for compressibility effects.

VULCAN has been used extensively by NASA as well as by the United States Air Force to study and simulate scramjet flow fields. One such simulation (Ref. 20) was recently conducted to study an ethylene-fueled scramjet combustor employing an aerodynamic ramp fuel injector and a cavity flame-holder. The aerodynamic ramp fuel injector consisted of a three-by-three array of flush-wall injectors. The injectors in the first, second, and third row were angled at 15, 30, and 45 degrees, respectively, relative to the wall. The nominal separation between the injectors was 7.0 jet diameters in the streamwise direction and 2.5 jet diameters in the spanwise direction. The injector design is intended to produce fuel vortex interactions to enhance fuel-air mixing but without the complications of a physical intrusion in the supersonic stream.

Under the modeled conditions of P_t = 574.5 kPa, T_t = 1223.0 K, M = 1.8 and Phi = 0.70, the combustor operated as a dual-mode scramjet in which the heat release is sufficient to choke the incoming supersonic air stream. The pre-combustion shock train aids flame stabilization by increasing the static temperature and pressure of the air stream (reducing the ignition delay) and decelerating the flow (increasing the residence time). The position and strength of the three-dimensional pre-combustion shock train and the combustor heat release distribution are strongly coupled. To accurately model a dual-mode scramjet flow field, the code must adequately resolve several complex physical processes including three-dimensional shock-boundary layer interaction, turbulent mixing in high-speed subsonic and supersonic streams, and kinetics of hydrocarbon fuels.

A schematic of the combustor flowpath is shown in Fig. 10. In this study, the computational domain extended streamwise from the throat of the facility nozzle to the exit plane of the combustor, and

from the centerline of the duct to the sidewall in the spanwise direction. The streamwise, spanwise, and normal directions were represented by the x, y and z directions, respectively, and x = 0 corresponds to the exit of the facility nozzle. Distance is non-dimensionalized by the height of the duct at the exit of the facility nozzle, H. The isolator and combustor extended from 0.0 < x/H < 12.3 and 12.3 < x/H < 33.3, respectively. The three rows of injectors were located at x/H = 13.5, 13.7 and 13.9, and the cavity extended from x/H = 14.7 to x/H = 16.5. The computational mesh consisted of 22 blocks and 1.06 million grid points.

Chemical kinetics was modeled with a three-step global ethylene-air kinetics model with six species: C_2H_4, O_2, CO, H_2, CO_2, and H_2O. Adjustments were made to the rates of the kinetics model to account for the effect of the small amount of OH in the vitiated heater flow (approximately one part per million) on ignition delay.
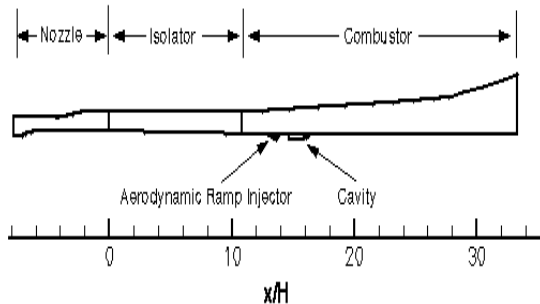

Figure 10.- Schematic of the combustor flow path

The calculated heat release region within the combustor is illustrated in Fig. 11, showing the static temperature contours within the combustor and in the cavity flame-holder region. The stoichiometric surface is indicated with a black contour line. The injected fuel did not react until it interacted with the hot fluid within the cavity. The flame spreads rapidly downstream of the cavity primarily along the stoichiometric surface. The cavity was instrumental in flameholding. Also necessary for flameholding was the role of the pre-combustion shock train in reducing the flow velocity and raising the static temperature. The shock train spanned the region 6.2 < x/H < 12.5 and lowered the mass-weighted one-dimensional Mach number from 1.78 and 0.87 and the u-velocity from 950 m/s to 520 m/s. More importantly, the mass-weighted one-dimensional static temperature increased from 740K to 965K.
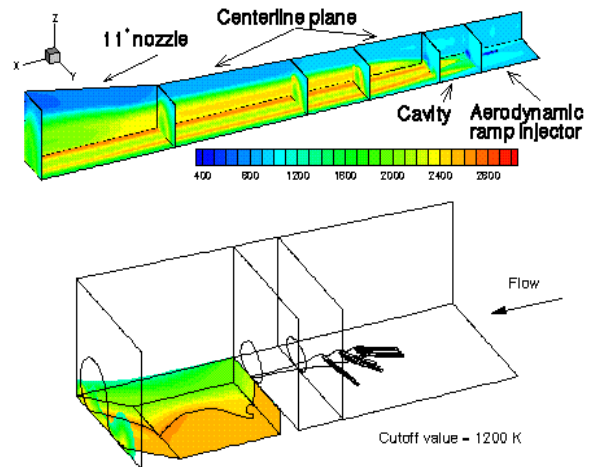

Figure 11.- Contours of static temperature [K] within the combustor and in the vicinity of the cavity.

A comparison of the experimental data with the computed wall pressure data (normalized by the pressure at the exit of the facility nozzle) is presented in Fig. 12. Sidewall pressure data is also included in the figure. The experimental data were obtained at similar, but not identical, conditions to those used in the calculations. The location of the pre-combustion shock train and the peak pressure are in good agreement, although the computed wall pressures are slightly lower than the corresponding experimental values.
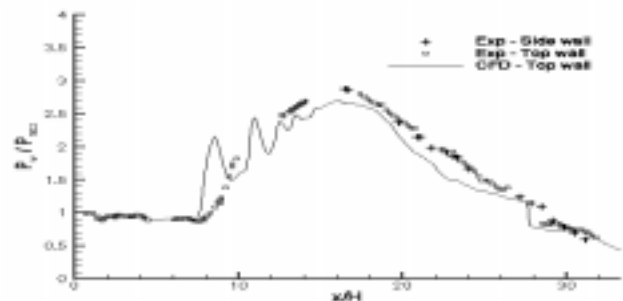

Figure 12.- Experimental and CFD pressures along the top wall of the combustor

NASA Langley Scramjet Test Complex/Experimental Design Tools

Hypersonic Airbreathing propulsion technology development so far has depended primarily on the ground-based facilities. These facilities are distinguished from hypersonic aerothermodynamic facilities by the requirement that the propulsion test facilities duplicate the stagnation

enthalpy of flight, the Mach number entering the engine or engine component under test, and the oxygen mole fraction of atmospheric air. NASA Langley has a facility complex (Fig. 13) for scramjet engine testing that can simulate flight enthalpies and dynamic pressure from Mach 3.5 to near orbital speed. A brief description of the facility complex is presented here along with an example of its use in an integrated experimental program to understand effects of geometric scale, dynamic pressure, and test gas differences on engine operability and performance at a given Mach number or flight enthalpy. More detailed discussion on facilities is given in Ref. 21.

The scramjet test complex at NASA Langley consists of five facilities, the Direct-Connect Supersonic Combustion Test Facility (DCSCTF), the Combustion-Heated Scramjet Test Facility (CHSTF), the Arc-Heated Scramjet Test Facility (AHSTF), the 8'-HTT, and the Hypersonic Pulse Facility (HYPULSE) shock-expansion tube/tunnel. The purpose of the DCSCTF is to test scramjet combustors in flows with stagnation enthalpies duplicating that of flight at Mach numbers from 4 to 7.5 in direct-connect, or connected-pipe, fashion so that the entire facility test gas mass flow passes through the combustor. The flow at the exit of the facility nozzle simulates the flow exiting an inlet and entering the combustor of a scramjet in flight. Scramjet nozzle geometric simulations can also be added to the scramjet combustor exit. The required stagnation enthalpy in the facility is achieved through hydrogen-air combustion with oxygen replenishment to obtain a test gas with the same oxygen mole fraction as atmospheric air (0.2095). Two two-dimensional (rectangular) contoured nozzles are currently available to attach to the facility combustion heater to simulate scramjet combustor entrance conditions. The first is a Mach 2 nozzle with exit dimension of 1.52 x 3.46 inches and the second is a Mach 2.7 nozzle with exit dimension of 1.50 x 6.69 inches. The facility is typically used to assess the mixing, ignition, flameholding, and combustion characteristics of the combustor models.

The CHSTF is used to investigate the operability and performance of complete (inlet, combustor, and partial nozzle) sub-scale scramjet component integration models in flows with stagnation enthalpies duplicating that of flight at Mach numbers from 3.5 to 6. Similar to the DCSCTF, the stagnation enthalpy is achieved through hydrogen-air combustion with oxygen replenishment. The flow at the exit of the facility nozzle simulates the flow entering a scramjet engine module in flight. The facility may be operated with either a Mach 3.5 or 4.7,

contoured nozzle with square cross sections, measuring 13.26 x 13.26 inches at the nozzle exit. The nozzle exhausts as a free jet into the test section that is 96 inches long and has a cross section of 42 x 30 inches. Either hydrogen or ethylene (both in gaseous forms at ambient temperature) may be used as fuel in the test engine. The facility allows the use of silane as an ignitor/pilot gas to aid in the combustion of the primary fuel.

The purpose of the AHSTF is to investigate operability and performance of sub-scale, component integration models of airframe-integrated scramjet engines at simulated enthalpies of flight from Mach 4.7 to 8. The stagnation enthalpy necessary to simulate flight Mach number for the engine tests is achieved by passing air through a rotating electric arc. Two square cross-section contoured nozzles for Mach 4.7 and 6 are available for use in the facility. These nozzles have an approximate cross section of 11 x 11 inches. Test times normally range from 30 sec at Mach 8 to 60 sec at Mach 4.7 conditions. This facility also allows the use of silane as a pilot gas.

The Langley 8'-HTT is a combustion-heated hypersonic blowdown-to-atmosphere wind tunnel that provides simulation of flight enthalpy for Mach 4, 5, and 7. The exit of the facility nozzle is 8 ft. in diameter that exhausts as free jet in a 26 ft/-diameter test chamber. The length of the test section test space is 12 ft. but scramjet engine models longer that 12 ft. can be mounted in this space if the model is not required to be fully retracted for tunnel start up. Stable wind tunnel test conditions can be maintained for up to about 60 sec. In addition to hypersonic propulsion testing, the facility can also be used for testing of structural and thermal protection system components with simulation of ascent or entry heating profiles by a radiant heater system. The desired stagnation enthalpy is generated by the combustion of air and methane in a pressurized chamber. Oxygen is replenished for airbreathing propulsion tests. This facility is the largest scramjet test facility in the country and has been used to conduct large-scale, airframe-integrated scramjet engine tests under the NASP and Hyper-X programs.

The Langley HYPULSE facility is located at and operated by GASL, Inc., in Ronkonkoma, New York, under a contract from NASA Langley. It can operate in both a reflected shock tunnel mode for conditions from Mach 4 to 12 and a shock-expansion mode for conditions from Mach 12 to near-orbital speeds. Most experience with HYPULSE is in the expansion tube mode at simulated Mach numbers of about 14 and 15 to study fuel injectors and
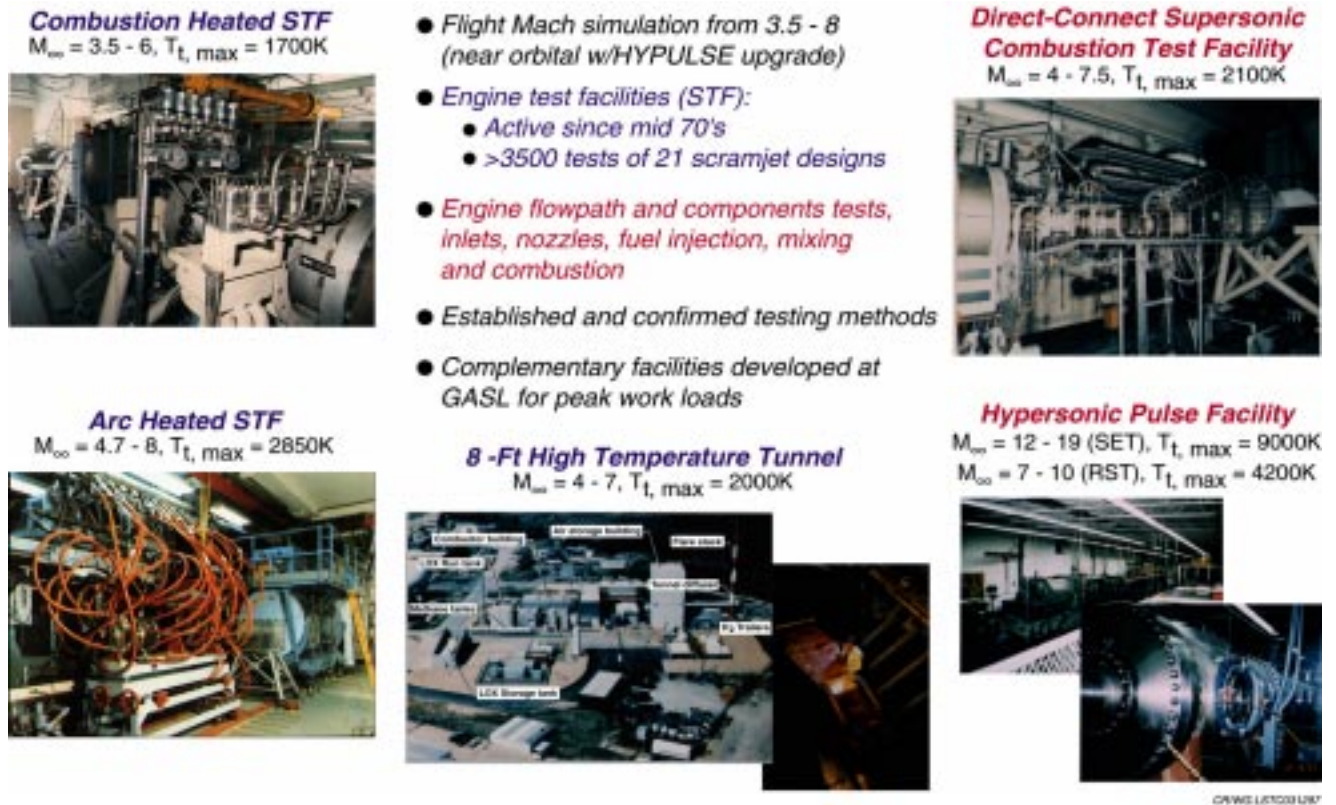
Figure 13.- NASA Langley Scramjet Test Complex

combustors. The tube exit (or combustor entrance) Mach number varies from 4.8 to 5 depending on the facility operations. When operated in the tunnel mode, HYPULSE expands the test gas to Mach 6.5 using a 26-inch diameter axisymmetric nozzle. Since the test gas in the shock-expansion tube/tunnel is never fully stagnated, high levels of dissociation (typical of reflected shock tunnels at high flight Mach numbers) do not occur. However, some dissociation does occur at the secondary diaphragm as a result of shock reflection during the rupture process. Optical access is provided for Schlieren images and laser diagnostics and instrumentation is available for collecting pressure, heat transfer, and temperature data.

The Langley Scramjet Test Complex was recently used in a systematic manner in the Hyper-X program to develop the Mach 7 hydrogen-fueled, airframe-integrated, dual-mode scramjet propulsion system. A detailed description of the engine development and its integration with an airframe is given in References 22 through 25. However, the use of test facilities in the Hyper-X program as a design tool is described here as an example. Figure 14 shows

a roadmap of the Mach 7 flowpath development and verification test program that involved three engine models in three facilities from the NASA Langley Scramjet Test Complex. The facilities used were the AHSTF, 8'-HTT, and HYPULSE. The engines tested were the Hyper-X Engine Model (HXEM), the Hyper-X Flight Engine (HXFE), and the HYPULSE Scramjet Model (HSM). These facilities and engines allowed an integrated test program to isolate and measure the effects on engine operability and performance caused by geometric scale, dynamic pressure, and test gas differences between tests. These differences, encircled in Fig. 14, exist due to test technique and facility differences. The effects of these differences must be properly accounted for in the design and analysis methodologies when using wind tunnel test results as an integral part of the vehicle/engine design.

The HXEM was tested in the AHSTF and 8'-HTT and provided the bulk of the data in this integrated test program. It was a full height, partial width, and truncated length flowpath model that could be tested in smaller facilities, including the AHSTF, but the model incorporated the same structural design and active cooling as the flight engine to allow testing
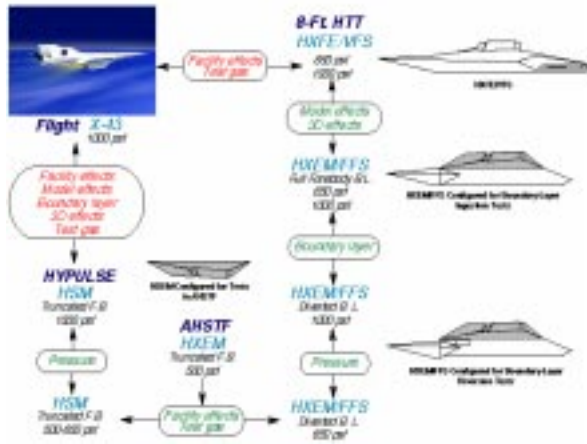
Figure 14.- Hyper-X Mach 7 Flowpath Verification Test Roadmap

at full flight dynamic pressure in the 8'-HTT. Tests of the HXEM in the 8'-HTT provided engine data at flight dynamic pressure, as well as data in a CH$_4$-Air-O2 combustion-heated facility at low dynamic pressure for comparison with the AHSTF results. For the 8'-HTT tests, the HXEM was mounted on the Hyper-X Full Flowpath Simulator (FFS) which provided a partial simulation of the Hyper-X airframe, as shown in Fig. 15. The mounting of the HXEM on the FFS allowed for ingestion or diversion of the forebody boundary layer to quantify the effects of boundary layer ingestion on engine performance.
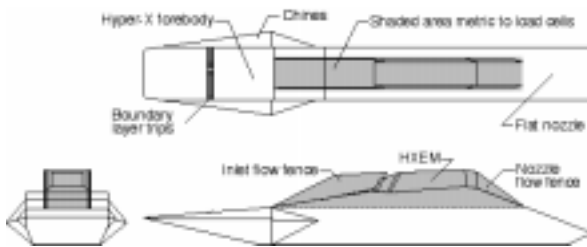

Figure 15.- HXEM Mounted on FFS

The HXFE was a spare Hyper-X engine dedicated to ground testing. This engine was mounted on a Vehicle Flowpath Simulator (VFS) that accurately represented the forebody and aftbody of the Hyper-X. The 12-ft. long HXFE/VFS simulated exactly the entire full-scale engine flowpath of the Hyper-X. Figure 16a shows the details of the HXFE/VFS and Fig. 16b shows it mounted in the 8'-HTT on the same support pedestal and force measurement system as the HXEM/FFS. The HXFE incorporated a removable panel encompassing the second and third forebody ramps so that either a flight

thermal protection system (TPS) panel or a more heavily instrumented metal panel could be installed and tested to quantify the effects of the surface temperature and roughness of the flight TPS on the engine operability and performance.
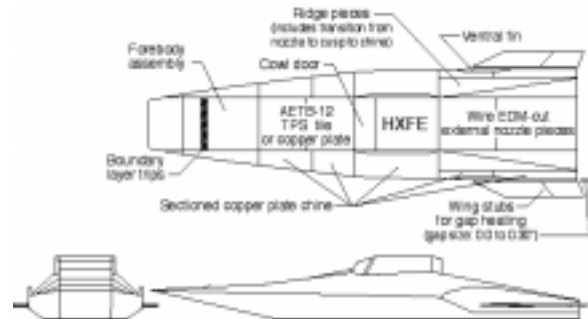

Figure 16a.- HXFE/VFS Details

The HSM flowpath was identical to the HXEM flowpath and was used to provide several important pieces of information on the Mach 7 verification roadmap. Tests of HSM in the HYPULSE facility provided Mach 7 tests at full flight dynamic pressure and enthalpy simulation, comparisons of performance for high-to-low pressure simulations, and a direct low-pressure comparison of clean air pulse facility results with the AHSTF. In addition, the short test duration alleviated thermal design issues, which allowed optical access to the scramjet flowpath for schlieren and planar images of fuel mixing and combustion.

The integrated test program discussed above allowed the design and performance assessment of an airbreathing engine flowpath and associated systems, including an understanding of the impact of various factors associated with the ground-based facilities on the performance and operability of the engine. The Hyper-X flight will allow for the first time a direct comparison of a full scale, airframe-integrated, scramjet engine flowpath performance in flight to the performance predicted in ground facilities under this integrated test program.
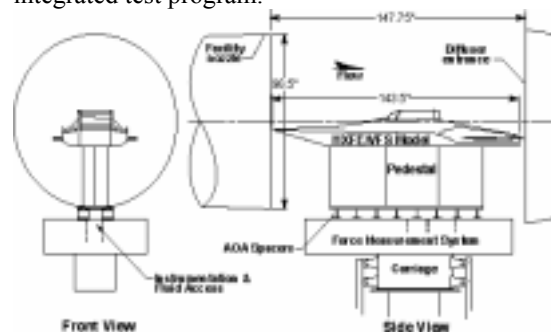

Figure 16b.- HXFE/VFS in the 8'-HTT

Future Technology Development Needs

Future needs of technologies for hypersonic airbreathing vehicles for access to space are discussed in Ref. 26 and are briefly detailed here. Some of the key future development requirements include:

- Hypervelocity combustion and engine performance prediction and optimization
- Efficient low-speed engine/cycle integration
  - Rocket-based combined cycle
  - Turbojet-scramjet combination engines
- Integrated engine performance and vehicle controls over the flight envelope
- Efficient cooled structures and durability testing
- Three–dimensional design methods with multi-disciplinary considerations
- Flight experience to quantify cost and reliability

Hypervelocity scramjet testing and databases are required to reduce performance uncertainty in the Mach 12 – 18 speed. Research in this speed range started in the late 1980's, and remains a pacing technology for efficient launch vehicles. Thrust augmentation approaches being studied for this speed range require experimental demonstration. These approaches include pre-mixed, shock-induced combustion, LOx addition, film and transpiration cooling, and novel mixing enhancement strategies. This database will validate the higher maximum operating Mach number of the scramjet, which is critical for effective airbreathing launch vehicles.

Basic system studies of combined cycle engines are required. Integration of rockets into scramjet flowpath must be accomplished with great care to avoid degrading scramjet performance. Current RBCC designs do not operate effectively above Mach 10-12, due to rocket drag. Alternate rocket-scramjet integration approaches have been proposed, which need to be investigated. Effectiveness for in-space operation should also be considered.

Turbojet-scramjet combination engine integration has not been seriously studied since the 1970's. This combination engine remains the only useful approach to efficient hypersonic airplane applications (vehicles which remain within the atmosphere and do not require a rocket for space application). These combination engines have also been shown as leading contenders for launch vehicles. Integration issues include durable variable geometry structures, inlet interactions during transition from one engine cycle to the other, and low inlet pressure recovery for the variable-geometry low-speed (turbojet or LACE) inlet. Mach 4 capable turbine-

based engines are also required, as discussed in Refs. 3 and 26.

Cooled structural concepts, material, coatings, and small test elements, developed in the NASP program, have not been tested. Restarting this activity is essential for validation of structural approaches assumed and used for efficient engine operation above Mach 10. In addition, work need to be focused on hot hydrogen fuel lines and valves.

Scramjet development is limited by methods to tailor the flowpath shape and optimize performance and weight, without violating operability limits. Even though two-dimensional approaches require less computer resources, three-dimensional effects in flowpath design are important and must be considered. This can be achieved by making cleaver use of existing, validated CFD codes.

Flight remains the only way to fully integrate and test the complete scramjet system. The ASTP Program, discussed in Ref. 26, provides a logical approach to flight testing and final development of this hypersonic propulsion technology.

Acknowledgement

References

1. Heiser, W.H. and Pratt, D.T.: Hypersonic Airbreathing Propulsion. AIAA Education Series. AIAA, Inc. Washington, D.C., USA. 1994.
2. Andrews, E.H. and Mackley, E.A.: NASA's Hypersonic Research Engine Project – A Review. AIAA Paper No. 93-2323, June 1993.
3. McClinton, C.R., Hunt, J.L., Ricketts, R.H., Reukauf, P., and Peddie, C.L.: Airbreathing Hypersonic Technology Vision Vehicles and Development Dreams. AIAA Paper No. 99-4978, November 1999.
4. Escher, W.J.D.: Synerjet for Earth/Orbit Propulsion: Revisiting the 1966 NASA/Marquardt Composite (Airbreathing/Rocket) Propulsion Study. SAE Progress in Technology Series, PT-54, Chapter 3: The 'Synerjet Engine'.

5 .   Douglas Aircraft Co.: Two-Stage Aerospace Plane Study. DAC-47551, 1964.

6 .   Hunt, J.L. and McClinton,C. R.: Scramjet Engine/Airframe Integration Methodology. AGARD-CP-600, Vol. 3: Sustained Hypersonic Flight, Paper C-35, 1997.

7.   Anderson, G.Y. and Rogers, R.C.: A Comparison of Experimental Supersonic Combustor Performance with an Empirical Correlation of Non Reactive Mixing Results. NASA TM X-2429, Oct. 1971.

8 .   Pinckney, S.Z.: Turbulent Heat-Transfer Prediction Methods for Application to Scramjet Engines. NASA TN D-7810, 1974.

9 .   Hunt, J.L.: Airbreathing/Rocket Single-Stage-to-Orbit Design Matrix. AIAA Paper No. 95-6011, April 1995.

10. Gatski, T.B. and Rumsey, C.L.: Linear and Nonlinear Eddy Viscosity Models. Closure Strategies for Modeling Turbulent and Transitional Flows. Edited by B.E. Launder and N.D. Sandham, Cambridge University Press, 2000.

11. Girimaji, Sharath S.: A Galilean Invariant Explicit Algebraic Reynolds Stress Model for Turbulent Curved Flows. Physics of Fluids, Vol. 9, No. 4, April 1997.

12. Gicquel, Laurent Y. M.: Velocity-Filtered Density Function for Large Eddy Simulation of Turbulent Flows. PhD Dissertation, State University of New York at Buffalo, June 2001.

13. Okong'o, N., Knight, D., and Zhou, G.: Large Eddy Simulations Using an Unstructured Grid Compressible Navier-Stokes Algorithm. International Journal of CFD, Vol. 13, pp. 303-326, 2000.

14. Jachimowski, C.J.: An Analytical Study of the Hydrogen-Air Reaction Mechanism with Application to Scramjet Combustion. NASA TP-2791. February 1988.

15. Thomas, J.L., Diskin, B., and Brandt, A.: Distributed Relaxation Multigrid and Defect Correction Applied to the Compressible Navier-Stokes Equations. AIAA Paper No. 99-3334, 1999.

16. Delarue, B.J. and Pope, S.B.: Calculation of Subsonic and Supersonic Turbulent Reacting Mixing Layers using Probability Density Function Methods. Physics of Fluids, Vol. 10, 487-498, 1998.

17. Jaberi, F.A., Colucci, P.J., James, S., Givi, P., and Pope, S.B.: Filtered Mass Density Function for Large Eddy Simulation of Turbulent Reacting Flows. J. Fluid Mechanics, Vol. 401, 85-121, 1999.

18. Kumar, A., Gnoffo, P.A., Moss, J.N., and Drummond, J.P.: Advances in Computational Capabilities for Hypersonic Flows. AGARD-CP-600, Vol. 3: Sustained Hypersonic Flight, Paper C-13, 1997.

19. White, J. A. and Morrison, J. H.: A Pseudo-Temporal Multi-Grid Relaxation Scheme for Solving the Parabolized Navier-Stokes Equations. AIAA Paper No. 99-3360, June 1999.

20. Eklund, D.R., Baurle, R.A., and Gruber, M.R.: Numerical Study of a Scramjet Combustor Fueled by an Aerodynamic Ramp Injector in Dual-Mode Combustion. AIAA Paper No. 2001-0379, January 2001.

21. Guy, R.W., Rogers, R.L., Rock, K.E., and Diskin, G.S.: The NASA Langley Scramjet Test Complex. AIAA Paper No. 96-3243, July 1996.

22. Voland, R.T., Rock, K.E., McClinton, C.R., and Pinckney, S.Z.: Hyper-X Engine Flowpath Design (U). 1997 JANNAF CS/PSHS/APS Joint Meeting, West Palm Beach, FL, October 1997.

23. Rock, K.E., Voland, R.T., Rogers, R.C., and Huebner, L.D.: NASA's Hyper-X Scramjet Engine Ground Test Program. ISABE 99-7214, September 1999.

24. Huebner, L.D., Rock, K.E., Witte, D.W., Ruf, E.G., and Andrews, E.H.: Hyper-X Engine Testing in the NASA Langley 8-Foot High Temperature Tunnel. AIAA Paper No. 2000-3605, July 2000.

25. Huebner, L.D., Rock, K.E., Ruf, E.G., Witte, D.W., and Andrews, E.H.: Hyper-X Flight Engine Ground Testing for X-43 Flight Risk Reduction. AIAA Paper No. 2001-1809, April 2001.

26. McClinton, C.R., Andrews, E.H., and Hunt, J.L.: Engine Development for Space Access: Past, Present, and Future. ISABE 2001-1074, September 2001.