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Abstract

This paper provides an overview of NASA's focused hypersonic technology program, called the Hyper-X Program. The Hyper-X Program, a joint NASA Langley and Dryden program, is designed to move hypersonic, air breathing vehicle technology from the laboratory environment to the flight environment, the last stage preceding prototype development. The Hyper-X research vehicle will provide the first ever opportunity to obtain data on an airframe integrated scramjet (supersonic combustion ramjet) propulsion system at true flight conditions and the first opportunity for flight validation of experimental wind tunnel, numerical and analytical methods used for design of these vehicles. A substantial portion of the program is experimentally based, both for database development and performance validation. The program is now concentrating on Mach 7 vehicle development, verification and validation and flight test risk reduction. This paper concentrates on the aerodynamic and propulsion experimental programs. Wind tunnel testing of the flight engine and complete airframe integrated scramjet configuration flowpath is expected in 1998 and 1999, respectively, and flight test is planned for 2000.

Introduction/Program Overview

Airbreathing propulsion offers substantial advantages for hypersonic flight applications, as illustrated in figure 1. It is an essential ingredient for sustained endoatmospheric hypersonic cruise applications, such as "global reach" vehicles, and can significantly improve the performance of space launch vehicles (ref. 1). Airbreathing scramjet (supersonic combustion ramjet) engines promise to improve mission effectiveness by reducing on-board propellant load in favor of payload and by increasing operational flexibility.

The pacing airbreathing hypersonic technology is certainly the scramjet engine. Hypersonic airbreathing propulsion has been studied by NACA/NASA for nearly 60 years (ref. 2). Numerous scramjet tests have been performed in a host of ground facilities (ref. 3 - 7). Significant improvements in design methods, experimental databases, experimental facilities and testing methods, as well as demonstrated engine performance have been made over the years for both propulsion and hypersonic aerodynamics. However, many of these tests have limitations, as discussed in Ref. 8.

NASA, as well as the hypersonic community at large, has long recognized the requirement to fully integrate hypersonic airbreathing propulsion engines with the vehicle airframe (ref. 9). This integration is difficult to demonstrate in ground based, experimental facilities. To mature hypersonic airbreathing propulsion technology for application in the future, NASA has initiated the Hyper-X Program (ref. 8, 10). The goal of the Hyper-X Program is to demonstrate and validate the technology, the experimental techniques, and computational methods and tools for design and performance predictions of hypersonic aircraft with airframe-integrated hydrogen fueled, dual-mode combustion scramjet propulsion systems. Accomplishing this goal requires flight demonstration of a hydrogen-fueled scramjet powered hypersonic aircraft.

Previous studies and other work leading up to this program were discussed in ref. 8, and are briefly addressed by figure 2. To meet cost constraints, the design of the flight test portion of the program utilized existing, largely National Aero-Space Plane (NASP) program developed technology, including vehicle design, design tools, databases, flight test approaches and "off-the-shelf" flight test equipment which was identified for the NASP proposed HYFLITE and HySTP experiments. Conceptual design for the flight experiment was accomplished in 1995 under a contract to McDonnell Douglas Aerospace (MDA), and preliminary design was accomplished by MDA between Feb. and Oct. 1996. The Hyper-X Program was approved by NASA Headquarters in July 1996, and officially started in Sept. 1996. The current Hyper-X launch vehicle (HXLV) and

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research vehicle (HXRV) contractor activity commenced in Oct. 1996 and March 1997 respectively. The launch vehicle and launch support contractor is Orbital Sciences Corporation, and the research vehicle contractor is a Micro Craft lead team, which also includes Boeing North American, GASL and Accurate Automation (see ref. 8 for additional details).

The technology portion of the program concentrates on three main objectives:

1) risk reduction - i.e., preflight analytical and experimental verification of the predicted aerodynamic, propulsive, structural and integrated air-vehicle system performance and operability of the Hyper-X research vehicle (HXRV),
2) flight validation of design predictions, and
3) methods enhancements - i.e., continued development of the advanced tools required to improve scramjet-powered vehicle designs.

These objectives include experimental, analytical and numerical activities applied to design the research vehicle and scramjet engine; wind tunnel verification of the propulsion-airframe integration , including performance and operability; vehicle aerodynamic and thermal database development; thermal-structural design; boundary layer transition analysis and control; flight control law development; and flight simulation model development. Much of the current technology effort is expended in objective 1, "risk reduction."

The primary objective of the flight test portion of the Hyper-X Program is to provide data required to advance key hypersonic Technology Readiness Levels (TRL's) from the laboratory level to the flight environment level, a NASA requirement before proceeding with a larger, crewed X-plane or prototype program. In addition, the flight test portion must address and advance many flight test techniques, such as experimental derivation of aerodynamic performance, and robust sensors and controls to accurately determine the flight condition.

As a part of this program, three autonomously-controlled research flights at speeds up to Mach 10 will demonstrate, validate, and extend scramjet technology. Each of the 12-foot-long, 5-foot-wingspan hypersonic aircraft, illustrated in figure 3, will have a single airframe-integrated scramjet. These vehicles will be boosted to flight test conditions using a modified Pegasus™ booster, air launched from the NASA Dryden Flight Research Center (DFRC) B-52 (see figure 4). The desired test condition for the Hyper-X in free flight is a dynamic pressure of 1000 pounds per square foot. The research vehicle will be boosted to approximately 95,000 feet for Mach 7, and 110,000 feet for Mach 10, corresponding to Reynolds number ~ 12 million, based on vehicle length.

Nominal flight sequence for the Mach 7 test is illustrated in figure 5. Following drop from the B-52 and boost to the predetermined stage separation point, the HXRV will be ejected from the booster-stack and start the programmed flight test. Stage separation is accomplished by pistons pushing the research vehicle and rotating the adapter "jaw" as illustrated by the CFD simulation presented in figure 6. Once separated from the booster, the HXRV will commence unpowered controlled flight to prepare for the powered test sequence. The powered test sequence, illustrated in figure 5, includes 5 to 10 seconds of scramjet operation. Following the powered engine test and 15 seconds of aerodynamic parameter identification maneuvers (ref. 11 and 12) the cowl door will be closed. The vehicle will then fly a controlled deceleration trajectory. In the process, short-duration programmed test inputs will be superimposed on the control surface motions to aid in the identification of aerodynamic parameters. These fully autonomously controlled vehicles will fly preprogrammed 700 to 1000 nautical-mile due-west routes in the Western Test Range (WTR) off the California coast, telemetering approximately 500 channels of test data.

The airframe design and flight test techniques are expected to prove readily adaptable for other advanced-propulsion-technology studies. This flight test of a scramjet-powered vehicle will focus technology on key propulsion-airframe integration issues, provide data to validate hypersonic vehicle design tools, and extend ground based wind tunnel testing methods.

This paper presents an overview of the experimental portion of the Hyper-X Program, including aerodynamic, propulsion and complete vehicle testing in wind tunnels. These wind tunnel tests are an integral part of the overall program and should not be considered as stand-alone activities.

Flight Test Vehicle Design and Experimental Databases

Hyper-X related propulsion and aerodynamic wind tunnel testing started in 1995 and 1996 respectively. The Hyper-X Program includes extensive utilization of wind tunnels to support the research vehicle design. Although conceptual vehicle design was based on previously demonstrated analytical and numerical design method predictions, the final aero database is experimentally based. Likewise, experimental data are a key part of the propulsion flowpath design and provide critical input to the predicted engine flight performance.

Successful demonstration of the predicted vehicle performance will be considered as validation of the use of these wind tunnels in the design process. Experimental wind tunnel tests include aerothermodynamic, propulsion flowpath, and integrated aero-propulsion tests. These will be discussed in the following sections.

Aerodynamic Facilities and Databases

Aerodynamic wind tunnels and wind tunnel models for various test programs, summarized in Table 1, are being utilized to develop the aerodynamic database, provide limited aero-heating data, develop the boundary-layer trip design, and calibrate the flush air-data system (FADS). In order to develop the flight aerodynamic database, extensive wind tunnel testing is planned and has been conducted on the Hyper-X research vehicle (HXRV), the Hyper-X launch vehicle booster stack (HXLV), and stage separation sequence. The primary focus of these wind tunnel tests has been to provide complete 6 degree-of-freedom (6 DOF) aerodynamic force and moment models, including control surface effectiveness, for use in the development of flight control laws as well as to provide performance data to aid in the propulsion system development process. Because of the program schedule and budget, these tests have been both prioritized and incrementally developed. Table 2 illustrates the aero database requirements, and is used to explain program priorities, development approach, and status. Data are required over a large range of flight Mach numbers for both the booster and research vehicle. Stage separation and boundary layer transition analyses are only required for the scramjet test point. (Boundary layer control is only an issue for inlet operation.) The highest priority data are that required to get the research vehicle to the scramjet-powered test condition. This includes the following phases: boost (Mach 0.8 to Mach 10); stage separation (Mach 6 and 10 wind tunnel conditions, which bracket the flight test conditions); and research vehicles at test conditions (Mach 6 and 10). The second priority is the research vehicle flight back to high subsonic speeds (Mach 4.6 to ~0.8). The lowest priority is subsonic operation of the research vehicle. The approach for generation of the database is illustrated by table 2 and figure 7. This database development approach features a 3-step process which is required for this fast paced, parallel design/development program. Each of the flight conditions is first "scoped," i.e., quick inexpensive tests are performed to obtain coarse data for preliminary validation of the basic configuration. These tests are performed with inexpensive scaled models, reducing the risks associated with configuration changes. The principal intent of these initial tests was to bracket the anticipated flight envelope in order to ensure the vehicle was capable of trimmed, controlled flight at the design test point and during the decent to terminal conditions. Next, the configuration is tested in more "detail" to define control effectiveness over the entire range of expected flight operation. Finally the configuration is "benchmarked," where the test matrix becomes fine enough to accurately identify the non-linear characteristics. Typical test matrices of angle of attack (α or AOA), side slip angle (β), and control surface deflection (rudder, δ_r, aileron, δ_a, and elevon, δ_e) are illustrated in figure 7 for each of these three phases of testing. Note that the primary difference between "detailed" and "benchmark" is the fineness of the incremental control surface deflections.

The primary wind tunnels that have been utilized in testing the Hyper-X research vehicle and launch vehicle stack to date include 3 hypersonic tunnels: NASA Langley's 20-Inch Mach 6 and 31-Inch Mach 10 facilities (ref. 13) and the AEDC VKF Tunnel B (ref. 14). Several transonic-supersonic tunnels have been utilized: the then McDonnell Douglas - St. Louis (now Boeing) Polysonic Wind Tunnel (PSWT) facility (ref. 15), the Lockheed/Martin Vought "High Speed Wind Tunnel" (ref. 14), and the NTS operated former Lockheed 4' Trisonic Wind Tunnel (ref. 14). In addition, 2 subsonic tunnels have been used: the Vigyan subsonic tunnel and the Boeing North American subsonic tunnel. Discussion of the low speed wind tunnel tests will not be included herein, because of the preliminary nature of these lower priority tests. Table 1 summarizes the test matrix planned and performed to date. Table 3 presents highlights of the test capabilities for these wind tunnels. A brief description of these wind tunnels and the Hyper-X related aerothermodynamic wind tunnel models is presented in the next section.

Hypersonic Aerodynamic Facilities:

NASA Langley 20-Inch Mach 6 Tunnel. The NASA Langley 20-Inch Mach 6 Tunnel is a blowdown facility with a 20-inch by 20-inch square test section that operates at a fixed Mach number of 6, using air as the test medium. Typical test conditions in this facility include nominal freestream unit Reynolds numbers of 0.5, 2.1, 3.9, and 7.1 million/ft. Model angle-of-attack can be varied in a pitch-pause mode from -20° to +20°, depending upon the length and position of the model in the test section. The design of the injection system permits the pitch-pause sequence at fixed angles of sideslip of up to 5°.

NASA Langley 31-inch Mach 10 Tunnel. The NASA Langley 31-Inch Mach 10 Tunnel is also a blowdown facility, with a 31-inch square test section that operates at a fixed Mach number of 10, using air as the test medium. Typical test conditions in this facility include nominal freestream unit Reynolds numbers ranging from 0.55 to 2.15 million/ft. Model angle-of-attack can be varied in a pitch-pause mode from -20° to +20°, depending upon the length and position of the model in the test section. The design of the injection system permits the pitch-pause sequence at fixed angles of sideslip of up to 5°.

AEDC VKF Tunnel B. This facility has a 50" diameter test section and operates in a closed circuit, recycling, variable density continuous mode at a Mach number of 6 or 8. Typical test conditions in this facility include nominal freestream unit Reynolds numbers ranging from 0.3 to 4.7 million/ft, with total temperature ranging from 800 to 1500°R. A two-sting mounting rig, referred to as the Captive Trajectory System (CTS) rig, allows models to be place in close proximity to quantify interaction aerodynamic forces and moments. This rig has the capability for modeling stage separation motion (ref. 15), accounting for measured forces. However, that capability was not included in the current study. For these tests, the facility was operated in a pitch-pause mode, across ranges of relative linear (x, y, z) and angular (pitch, yaw, roll) displacements.

Hypersonic Aerodynamic Experimental Models:

A large range of models have been or will be tested in these hypersonic wind tunnels (see table 1). Hyper-X research vehicle models include 12-inch (8.33% of full scale) and 18-inch (12.5%) steel model of the research vehicle and a 33%-scaled model of the research vehicle forebody. The HXRV models include variable control surfaces, including the all moving horizontal tails and twin vertical rudder surfaces. The 12" model was tested over a range of angles-of-attack and sideslip at both Mach 6 and 10. A photograph of the model in the 20-Inch Mach 6 Tunnel is shown in Figure 8. Generally, these tests were performed with rudder, δ_r, aileron, δ_a, and elevon, δ_e, control surface deflections from -20° to +20°, -20° to +20° and -10° to +10° respectively. The rudders and horizontal tails could be deflected in 10o increments. The 18" HXRV model is currently being built and will include both blade and sting mounts to quantify blade interference effects on lift and pitching moment. In addition, this model will have higher fidelity increments (2.5°) on the control surface deflections.

The 12-inch (8.33% of full scale) stainless steel model of the Hyper-X research vehicle with variable control surfaces was also tested in the 20" Mach 6 and 31-Inch Mach 10 Tunnels in close proximity to the research vehicle adapter to simulate stage separation, as illustrated in figure 9. To date, only the fixed-jaw stage separation scenario has been tested in these facilities. Future tests will include various "jaw" rotation angles for comparison with the AEDC stage separation data to determine feasibility of this single-sting approach for stage separation aerodynamic database development for the Mach 10 flight.

A 3% cast steel HXRV with fixed control surfaces was mounted on a high fidelity 15-inch (3% of full scale) steel model of the Hyper-X launch vehicle with variable HXLV control surfaces and tested over a range of angles-of-attack and sideslip. A photograph of the model in the 20-Inch Mach 6 tunnel is shown in Figure 10. This model has control surface deflections of +/- 20° in 5° increments. This model was tested both with and without (post stage separation simulation) the research vehicle.

The 33%-scale forebody model was used to investigate boundary layer trip effectiveness and impact on the engine airflow and forebody aerothermal loads. One of the boundary layer trips included in the study is illustrated in figure 11. These tests were performed using both a macor surface, as shown for global thermography and discrete thin film gages, and a steel surface with pressure measurements. This model is designed to quantify the heat transfer patterns downstream of various trip designs. These tests confirmed trends noted in CFD design studies, and refined the boundary layer trip designs.

A separate 8.33% scale model of the research vehicle (12-inch), booster-to-research vehicle adapter, and booster were fabricated for the AEDC stage separation test. This model could be combined into the full launch vehicle (~40") stack, or tested in various combinations. The primary objective of the AEDC tests was to quantify the aerodynamic loads during the HXRV-HXLV stage separation. These loads are required for the complicated simulation modeling of the stage separation sequence. The AEDC von Karman Facility Tunnel B was selected for this test because of the existing dual support/force balance CTS rig. This test included variations in interference loads due to the effects of adapter "jaw" position (including 4 "jaw" angles), effects of relative position (translation and rotation) of the research vehicle and booster-adapter combination ,and variations in initial flight conditions (AOA and sideslip) which might result from booster dispersion from the nominal trajectory. The tests have provided data to compare with CFD predicted loads. The stage separation tests included a series of HXRV control surface parametrics including elevator and aileron combinations, over a range of angles-of-attack and sideslip. This test also included Mach 6 tests on the 12-inch (8.33% scale) research vehicle, the 40" long complete launch vehicle stack and the launch vehicle minus the research vehicle. These tests provide a direct comparison with HXRV, HXLV, and limited stage separation results from the LaRC hypersonic tunnels. A schematic of the vehicle mounting for the AEDC stage separation tests is presented in figure 12.

Transonic - Supersonic Aerodynamic Facilities:

Three wind tunnels, with essentially equal capabilities, were used for the transonic - supersonic tests. These tunnels were used for force and moment development, and flutter testing of the HXRV and HXLV. Models were tested using incremental AOA sweeps, with sideslip up to 5°. These facilities are briefly discussed below:

Boeing (formerly McDonnell Douglas) Polysonic Wind Tunnel (PSWT). This facility has a 4 x 4 x 9 foot test section and operates in a blowdown to atmosphere mode with a Mach range of 0.5 to 5.8. Typical test conditions in this facility include nominal freestream unit Reynolds numbers ranging from 2 to 50 million/ft.

Lockheed/Martin Vought Systems High Speed Wind Tunnel. This facility has a 4 x 4 x 5 foot test section and operates in a blowdown to atmosphere mode with a Mach range of 0.2 to 5.0. Typical test conditions in this facility include nominal freestream unit Reynolds numbers ranging from 2 to 38 million/ft.

NTS - Lockheed 4-Foot Trisonic Wind Tunnel. This facility has a 4 x 4 foot test section and operates in a blowdown to atmosphere mode with a Mach range of 0.1 to 5.0 (to 2.0 for these tests). Typical test conditions in this facility include nominal freestream unit Reynolds numbers ranging from 1 to 43 million/ft.

Transonic - Supersonic Aerodynamic Models

A 24-inch (16.67% of full scale) aluminum model of the HXRV was tested at Mach numbers ranging from 0.8 to 4.6 in the PSWT with a series of control surface parametrics including horizontal tail, differential tail, and rudder deflection combinations, over a range of angles-of-attack and sideslip. These tests provide the basis for the HXRV's supersonic (second priority) aerodynamic database.

A 30-inch (6% of full scale) steel model of the HXLV (figure 13) was tested by Orbital at Mach numbers ranging from 0.8 to 4.6 in the L/M-Vought and NTS wind tunnels. These tests included a series of control surface parametrics including horizontal tail, differential tail, and rudder deflection combinations, over a range of angles-of-attack and sideslip. Two entries in Vought (12/96 and 9/97) and one in NTS (6/97), in conjunction with the LaRC Mach 6 and 10 tests, have provided a complete database covering the Mach 7 boost trajectory flight conditions and vehicle configurations.

Aerodynamic Tests Results

The aerodynamic wind tunnel results are used primarily in formation of the aerodynamic database for the research vehicles. This is accomplished working in conjunction with analytical (S/HABP, ref. 16 and 17) and CFD (GASP, ref. 18; USM3D, ref. 19; and SAMcfd, ref. 20) methods. A preliminary aerodynamic database was developed from results of fifteen experimental programs on 11 separate wind tunnel models utilizing over 1,000 wind tunnel runs. The aerodynamic database includes boost, stage separation, research vehicle unpowered and scramjet powered flight at the test point, and unpowered flight back to subsonic speeds. The test point aerodynamic database must include both inlet cowl open and closed unpowered configurations as well as powered effects over a large range of power settings. The aerodynamic and propulsion database is being filled in with additional wind tunnel tests and CFD solutions, and used in flight simulation of all stages of the flight.

Flight simulations, utilizing this aero database, demonstrate that the vehicle is controllable over the anticipated flight envelop, and meets the flight test requirements. For example, figure 14 presents elevator position and angle-of-attack as a function of time, from stage separation through the scramjet powered portion of the flight back to inlet cowl closure. Initially the elevator controls are locked, and the vehicle is assumed to be at the launch vehicle stage separation condition of zero degrees AOA. Aerodynamic and separation forces drive the vehicle nose down initially, toward negative AOA. When active control is established the control system commands the all-moving wing up to maintain zero degrees AOA. At 0.5 seconds from separation the flight controls switch to the powered flight attitude, which is 2 degrees AOA. For this simulation the cowl door opens between 2-2.5 seconds. Tare, no fuel operation, is maintained for 5 seconds. At 8 seconds after separation, engine fuel ignitor (silane-H_2 mixture) and fuel flow are initiated. The ignitor is turned off at about 9.5 seconds, as the fuel is ramped up to full power. Full power is maintained from about 11 to 14 seconds in this simulation. The fuel ramp down is complete at 14.5

seconds. Five seconds of engine tare data and 15 seconds of parameter identification (PID) maneuvers are performed before the cowl flap is closed 35 seconds after stage separation. During this entire process the elevator excursions are within reasonable limits, and vehicle response is adequate for the flight test. Preliminary analyses indicate that stability and controllability of the research vehicle is within acceptable margins during all segments of the flight.

Propulsion Facilities and Databases

Wind tunnel testing commenced in 1995 to verify the engine design (performance and operability), evaluate the engine operating characteristics required to develop autonomous engine controls for flight test, and support generation of the propulsion contribution to the aerodynamic database. Tests planned and completed are listed in Table 4. Although a substantial database existed for this class of engine, the proposed flight test engine required specific tailoring for the small scale to provide the desired thrust efficiency to assure vehicle acceleration. Tests were started for the Mach 5, 7 and 10 design points, but the main emphasis to date has been on the Mach 7 engine flowpath.

Facilities utilized for the engine flowpath development are summarized in Table 5. Highlights of these facilities are presented in the next section, followed by a description of existing or planned engine flowpath models designed or used to support the Hyper-X research vehicle development. All of these facilities have vacuum capability for simulation of engine exhaust to flight static pressure, so that will not be discussed for each facility. The first two facilities discussed are used for component development only; the next four facilities are designed for Mach 3-8 scramjet engine module testing. One of these is also large enough to test the complete Hyper-X research vehicle. The last facility is a pulse tunnel which has been recently upgraded for complete scramjet engine module testing at Mach 7 to 10 flight conditions.

Propulsion Facilities:

The NASA LaRC Mach 4 Blowdown Facility (M4BDF) (ref. 21), is a 9" x 9" Mach number 4 wind tunnel, that utilizes ambient temperature, high pressure air in a blow down mode. This facility is utilized for scramjet inlet and isolator studies. Operation at stagnation pressures to 200 psi provides aerodynamic simulation to Reynolds number of about 21 million per foot. The data acquisition system (DAS) includes 128 channels of analog and digital data from an Electronic Scanning Pressure (ESP) system. The 9" by 9" test section includes windows for Schlieren and other non-intrusive measurements, and through-the-wall probe actuators.

GASL Leg II (ref. 22), currently configured as the former NASP parametric Direct Connect combustor Module (DCM), utilizes a H_2-Air-O_2 combustion heater to provide flight simulation to about Mach 9 enthalpy, with total pressure capability adequate for full flight dynamic pressure simulation in excess of 4000 psf. For Mach 5 flight simulation, the Mach 2.2 facility nozzle was used to provide test gas to the parametric combustor hardware. Test gas contaminants for this facility are similar to those noted below for the CHSTF. The Leg II facility includes about 400 channels of instrumentation in the DAS, automatic fuel control, thrust-drag load cell, bulk calorimetric measurements and pitot gas sampling/mass spectrometer analysis capability.

The NASA Langley Research Center (LaRC) Arc-Heated Scramjet Test Facility (AHSTF) provides high test gas enthalpy duplicating flight conditions at Mach numbers from 4.7 to 8, utilizing a rotating electric arc, Linde "N3" heater. Arc-heated air (at 3000 Btu/lbm) is mixed with ambient air to provide the desired test gas enthalpy. Pressure is limited in this facility to simulation of about 600 psf flight dynamic pressure at Mach 7 and about 800 psf at Mach 5. Two nominal 11" square exit nozzles are available as listed in Table 5. Facility test gas contaminants are predominantly NO, at about 0.02 mole fraction for the Mach 7 test conditions. This facility DAS includes about 650 channels of data, a 6-component force balance, and pressure probe and gas sampling survey capability. The AHSTF is the primary Mach 7 work-horse scramjet facility at Langley, and is more fully characterized in ref. 3. Figure 15 shows a Hyper-X scramjet model installed in the AHSTF test cabin.

The NASA LaRC Combustion-Heated Scramjet Test Facility (CHSTF) provides high test gas enthalpy duplicating flight at Mach numbers from 3.5 to 6. This facility utilizes a H_2-Air-O_2 combustion heater for test gas production. The facility test gas has an oxygen mole fraction identical to atmospheric air. Facility test gas contaminants are predominantly water vapor, at about 0.08 and 0.18 mole fraction for the Mach 4 and 5.5 test conditions, respectively. The effect of this contaminant on test results is summarized in ref. 3. Two 13.25" square exit nozzles are available; the Mach 4.7 being the same as the lower Mach number nozzle in the AHSTF, to promote facility comparisons. The maximum Mach 5 flight dynamic pressure simulation from this facility is about 1500 psf. The facility DAS includes about 600 channels of data, with the engines mounted on a thrust-drag load cell. This is the primary lower Mach number (<5) scramjet development facility at Langley (ref. 3).

The GASL Leg IV or V (ref. 22) combustion-heated freejet test facility provides flight enthalpy simulation to Mach numbers in excess of 8, and can provide full flight dynamic pressure simulation at Mach 7 enthalpy. These facilities share a set of Mach 2, 3.4, 4.7, 5.5 or 6 13.25" square exit nozzles. For leg V the test gas is heated using a H_2-Air-O_2 combustion heater, and for leg IV the test gas is heated using a two stage approach. First, air can be preheated using a pebble-bed storage heated. Then the hot air is utilized in an otherwise standard combustion heater. High total pressure provides flight simulated dynamic pressure up to 1,700 psf at Mach 7 enthalpy, and in excess of 2000 psf at Mach 5. Facility test gas contaminants are predominantly water, at 0.08 and 0.20 mole fraction or less for the Mach 5 and 7 test conditions, respectively. These facilities shares instrumentation with the DCM, Leg II discussed above with the addition that engines are mounted on a single component thrust-drag balance. Final selecting between these two facilities will be made based on facility availability/schedule.

The NASA LaRC 8-ft High Temperature Tunnel (8' HTT) provides a unique capability to test the complete hyper-X research vehicle or full length engine flowpath at the Mach 7 flight conditions. This facility was placed in service in the 1960's to conduct aerothermal loads, aerothermostructures and high-enthalpy aerodynamic research (ref. 23). The high enthalpy test gas is produced by burning methane and air at high pressure, then expanding it through an eight-foot exit-diameter hypersonic nozzle to the 12.5 foot long test cabin. During the late 1980's and early 1990's the tunnel was modified with an oxygen replenishment system to allow scramjet engine performance testing over a range of flight Mach numbers between 4 and 7 (Table 5). The primary test gas contaminants created are water vapor and carbon dioxide, at about 0.18 and 0.09 mole fraction respectively for the Mach 7 test conditions (see ref. 3). A schematic representation of the 8' HTT configured for airbreathing propulsion tests in shown in figure 16. To facilitate starting the tunnel and to protect the models from startup and shutdown dynamic loads, models are typically stored beneath the hypersonic test stream and inserted into the stream after steady-state hypersonic flow is established. A hydraulic elevator system inserts the model mounted on a three component force balance into the test stream in approximately 1.5 seconds. The data acquisition system can accommodate about 1000 channels of ESP measured pressure and 500 channels of general stain gage measurements. Of these 500 channels, up to 31 high frequency (106 Hz. or 1 MHz.) measurements and a three component force balance are available.

The NASA Hypersonic Pulse Facility (HYPULSE) at GASL. operates over a flight Mach enthalpy from about 5 to orbital velocity, with full pressure simulation to Mach 17 for scramjet combustor testing. This facility operates in the expansion tube or expansion tunnel mode for simulated flight Mach numbers greater than 12. For Hyper-X testing, the facility operates in the reflected shock mode, utilizing a shock-induced detonation (SID) high pressure driver. At Mach 7 conditions, full flight pressure (dynamic pressure greater than 1000 psf) is provided, with test times on the order of 2 - 3 ms. The only significant facility test gas contaminant is NO, at less then 0.01 mole fraction for the Mach 10 test conditions. At Mach 7, the facility provides a "clean" air test gas. The facility instrumentation is currently limited to about 200 channels at 1 MHz sampling rate, used primarily for wall pressure and heat transfer. As pointed out in ref. 24, the short test times preclude hot wall effects, but permit easy optical access to the combustor, and facilitate acquisition of accurate heat flux measurements compared to continuous test facilities. Non-intrusive measurements systems are available to infer the level of fuel mixing (ref. 25) and combustion efficiency (ref. 26), and thrust measurements using stress-wave type balances and metric surfaces are being developed.

Propulsion Flowpath Models

Three types of experiments support the Hyper-X Program flowpath development: 1) inlet and combustor components; 2) full scale, partial-width engine module, which includes part of the external forebody-inlet and afterbody-nozzle surfaces; and 3) full length flowpath. Most of the component level testing required for the Hyper-X flowpath design and design methods validation was accomplished as part of the NASP Program. Therefore, the majority of the Hyper-X flowpath tests are of the integrated flowpath type. This section presents a brief description of the ground-based propulsion wind tunnel models.

The Generic Inlet Model (GIM) used in the M4BDF was modified for the Hyper-X Program to study inlet starting phenomena. Two approaches were investigated, and the variable geometry illustrated in figure 17 was selected to facilitate both closing the inlet for unpowered flight and inlet starting at the flight test point. This model was also modified to include the correctly scaled inlet cowl lip radius, internal contraction ratio, and closely modeled cowl lip station normalized boundary layer thickness.

The Hyper-X Combustor Development (HXCD) model is a specific use of the parametric direct connect module hardware in GASL Leg II to develop a parametric dual mode ramjet combustor design database and associated regression models. The major change to the existing GASL hardware was addition of new fuel injectors. The HXCD model incorporated various combustor divergence angles, injector locations, fueling strategies, etc. The primary factor of goodness for test results was measured incremental thrust. However, extensive wall pressure and limited gas sample and pressure probe measurements were also collected. Results from this test, performed at Mach 5 enthalpy, provided the starting point for the Mach 5 and 7 Hyper-X combustor detailed designs. Figure 18 illustrates this parametric hardware.

The dual-fuel experimental (DFX - named for the preliminary design contract) engine (figure 15) was developed in 1996 by modification of NASP engine hardware, to provide a rapid performance/operability evaluation of the Mach 7 Hyper-X design using the LaRC AHSTF. The DFX engine simulates the full scale, partial width (44%) engine flowpath, as illustrated in figure 19. This simulation includes the correct cowl leading edge radius and all flowpath lines from the mid-forebody to mid-afterbody as denoted by the shaded region in figure 19. The DFX engine, fabricated predominantly from copper, is heat sink cooled, and features limited parametric capability required for preliminary feasibility studies. Two design features limit the utility of this engine. First, this engine utilized an existing engine starting mechanism. To reduce the internal contraction ratio for inlet starting, the cowl is rotated about the cowl leading edge, which is different than the Hyper-X approach (illustrated in figure 17). In addition, this uncooled engine is limited to about 30 second tests at the reduced dynamic pressure (500 psf) of the AHSTF. It can not be used, as is, in the full pressure (1000 psf) environment in the 8' HTT or the Leg IV engine test facility.

The Hyper-X engine module (HXEM) was designed to overcome the limitations mentioned for the DFX. The HXEM shares the DFX partial width simulation. It also incorporates additional parametric capability for Mach 5 and 7 testing, includes limited active cooling to allow testing at full flight dynamic pressure, and includes the articulated, two-position inlet cowl leading edge used to close off the inlet. Current plans are to test this engine at Mach 7 in the AHSTF, GASL Leg IV, and in the 8' HTT. Tests in the AHSTF provide a direct comparison with DFX results. Tests in GASL Leg IV provide results for full pressure and enthalpy simulation, comparisons of performance for high-to-low pressure tests, and a direct, low-pressure comparison of H_2-Air-O_2 combustion heated facility results with arc-heated facility results. Tests in the 8' HTT also provide full pressure simulation, as well as a comparison of CH_4-Air-O_2 combustion-heated data for comparison with the Leg IV and AHSTF results. More importantly, this test provides a benchmark of the 8' HTT facility effects before testing the Hyper-X research vehicle.

The Hyper-X Full Flowpath Simulator (HXFFS) is a non-flight partial simulation of the Hyper-X vehicle, designed to be mated with either the Hyper-X research vehicle flight engine (HXFE) or with the partial width HXEM. The lower half of the Hyper-X Research Vehicle, the external propulsion flowpath, is accurately modeled by the HXFFS, as illustrated in figure 20. This "boiler-plate" model will also include the correct forebody leading edge radius, flight boundary layer trips, and the ability to incorporate either steel, copper, or Advanced External Thermal Barrier (AETB-12) tile as used in flight. Because of size, this model can only be tested in the 8' HTT tunnel. Tests with the HXFE and HXEM in the 8' HTT will provide quantification of the effects of partial width testing, and provide a connection between the flight engine and the remainder of the engine flowpath wind tunnel tests. Test with the HXFE on the HXFFS will allow full flowpath testing without interference with the first HXRV development.

The HYPULSE Combustor Model (HCM), illustrated in figure 21, is a full scale, partial-width (15%) combustor and internal nozzle segment of the Mach 7 and 10 Hyper-X engine flowpath. This model is a slightly modified version of the NASP HYPULSE Mach 13-17 combustor model, and is used for parametric combustor development tests and facility shakedown leading to tests of a complete engine module. Instrumentation for the Mach 7 and 10 HYPULSE-RST inauguration were limited to wall pressure and heat flux measurements. Future tests will also include non-intrusive measurements and force measurements. Combustor data from these tests at Mach 7 have been utilized for direct comparison to the AHSTF-deduced combustor performance, illustrating the effects of facility contaminants on fuel auto-ignition.

The HYPULSE Scramjet Module (HSM) is illustrated in figure 22. Like the HXEM, this model is a full scale, partial width (44%) flowpath. Because of the short test times, the HSM is made from aluminum, reducing model cost. The engine will also be parametric, for testing at both Mach 7 and 10 in the 84" test cabin at HYPULSE. Model alignment in the 26" diameter nozzle exit flow will be adjusted to duplicate the flight cowl lip Mach number. Experimental measurements will include surface pressure and heat flux, force, and non-intrusive water vapor and fuel plume images. This model will be tested at Mach 7 to provide full pressure, clean test gas for comparison with results from the other facilities, and a direct comparison of pulse and continuous tunnel scramjet engine performance. The latter comparison is to provide confidence in the pulse tunnel scramjet testing, which is the only means available for flight simulation at a Mach number of 10.

Typical Results from Propulsion Tests

Mach 7 engine performance and operability were verified in reduced dynamic pressure tests of the "DFX" (dual-fuel experimental) engine in the NASA Langley Arc Heated Scramjet Test Facility. These test results verified predicted engine performance (including force and moments as well as inlet and combustor component performances) and operability (ignition requirements, flameholding limits and inlet interaction limitations). The tests demonstrated excellent engine performance and operability, and provided data for validation of the Hyper-X design methods. Preliminary experimental results for the Mach 5 and 10 scramjet combustor design have been obtained using the direct connect combustor module rig (DCM) and the HCM in the HYPULSE facility. These tests have also demonstrated design performance. Additional tests will be performed at Mach 7, 5, and 10 using the DFX, HXEM, and HSM engines. These wind tunnel tests of the HXRV flowpath will include wind tunnel test gas variants for comparison with flight: both partial and full dynamic pressure data at full flight enthalpy; long (20-30 seconds) and short duration (2 ms) tests; "clean," "vitiated" and "arc" heated test gas; and hydrogen and hydrocarbon-fueled vitiated heaters. In addition, the wind tunnel test will include engine controls. The current schedule for these tests is presented in figure 23.

Wind Tunnel Tests of the Hyper-X Research Vehicle (HXRV)

In addition to the standard aerodynamic force and moment tests and engine flowpath tests described above, two full length flowpath tests are planned in the NASA Langley 8-foot High Temperature Tunnel (8' HTT). First, a "boiler plate" vehicle, the HXFFS model with the HXFE will be tested, followed by the second Mach 7 Hyper-X research vehicle.

The primary objective of the 8' HTT test of the Hyper-X research vehicle is to demonstrate the readiness of the integrated engine/vehicle system for flight. This objective is composed of three components: a) verification of the propulsion system; b) limited structural integrity verification; and c) verification of selected subsystems in the flight environment. The propulsion system verification will include demonstrating 1) that the inlet will start on the full-length/width vehicle with boundary layer trips, wall roughness/temperature, and cowl actuation at near flight test conditions, 2) that the fuel control system is operational, and 3) that the engine will operate over the fuel sequence without unstarting the inlet. In addition, this test will provide the aerodynamic force and moment incremental data that results from the cowl inlet opening and closing and the fuel-on (powered) portion of the flight. This test will not provide overall vehicle force and moment data because the full scale research vehicle span extends beyond the tunnel core flow (ref. 23), as well as the aerodynamic interference effects associated with the large mounting strut that is required to support the model in the tunnel and house the fueling purge and instrumentation interfaces (Figure 24). However, because the propulsion flowpath is well within the high quality core flow of the tunnel, the test should provide good quality flowpath force increments. This data will aid in the benchmarking assessments of the computational predictions included in the aero-propulsion database. The preliminary test of the flight engine with a boiler-plate vehicle allows rapid closure on the most critical issues without risking the first vehicle (schedule or hardware).

Only limited structural verification will be performed since the test article will not encounter the entire heating history that the research vehicle experiences on the boost/flight trajectory to the test condition. The vehicle's carbon-carbon leading edge; actively-cooled cowl and sidewall leading edges; thermal protection system, coatings and seals; and the engine surfaces will see a portion of the high-enthalpy flow and will be checked for structural integrity following each run and following the entire test series.

Subsystem verification includes demonstration that the HXRV subsystems employed in the 8' HTT tests are working in concert to minimize risk of subsystem interface problems for flight. Other system validation and verification testing will be performed at DFRC without hydrogen fuel or the harsh hypersonic environment. Presently, subsystems that are planned for the 8' HTT test include: HXRV flight computer, cowl flap actuation system, water-cooling system, fuel and purge systems, and most vehicle and engine instrumentation.

Additional objectives for the test are to characterize the engine performance and operation at Mach 7 in order to provide data for ground-to-flight correlation and to develop expanded ground-test capabilities for fully-integrated hypersonic vehicles with airbreathing propulsion systems. This test signifies the first time that a full-scale, scramjet-powered flight vehicle will be tested in a wind tunnel prior to flight.

The Mach 7 engine will be delivered in April 1998 for testing in the Langley 8' HTT. The second vehicle will be delivered in Sept. 1999, for testing before the first flight. The first flight is currently scheduled for Jan. 2000. These tests provide flight test risk reduction and will allow direct comparison of the wind tunnel methods and results with flight performance. Flight data will provide validation of the hypersonic facilities for slender-body aerodynamics and scramjet engine testing.

Although it is beyond the scope of this paper, it is important to be mindful of the importance of analysis in the comparison of ground and flight data. Facility contaminants, flow distortion, and turbulence levels make a direct comparison of ground and flight results very difficult and perhaps misleading. The experimental wind tunnel data will be compared with that predicted for the wind tunnel conditions. Then the flight results will be compared with that predicted for the flight environments thereby preventing misinterpretation of the test results.

Summary

This paper discussed highlights of NASA's hypersonic technology program Hyper-X, with a focus on the wind tunnel testing portion of the program. The Hyper-X Program is designed to elevate scramjet powered hypersonic technology readiness (TRL's) from the wind tunnel to the real flight environment, a mandatory step before proceeding to prototype or other vehicle development. The wind tunnel part of the program includes research vehicle design support, flight test risk reduction, and flight vehicle verification. The flight focus of the program provides a significant challenge to some wind tunnel test methodologies, as no airframe integrated scramjet-powered vehicle has gone to flight. As a direct result, ground based experimental wind tunnel methods, as well as design methods, are continuing to be improved to meet program objectives. The flight test will provide critical data required to validate design methods, including analytical, computational and experimental methods. Flight test plans call for the first Hyper-X research vehicle to fly at Mach 7 in early CY 2000.

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Facility	HXRV	HXLV	HX-Stage Sep.	Boundary Layer
LaRC 31" M-10	**8.3% S&C** **33.% AH** 12.5% HF B/S	3.% QL 3.% S&C*	**8.33% Single Sting (Fixed Jaw)** (Rotating Jaw)	**33.% AH, FADS**
LaRC 20" M 6	**8.3% S&C** **33.% AH** 12.5% HF B/S	3.% QL 3.% S&C*	**8.33% Single Sting (Fixed Jaw)** (Rotating Jaw)	**33.% AH, FADS**
AEDC VKF - B Mach 6	**8.3% S&C**	**8.3% S&C***	**8.3% Dual Sting (Rotating Jaw)**	
PSWT M 0.4-4.6	**16.7% S&C**			
L/M Vought M 0.4-4.6		**6.% S&C**		
NTS M 0.4-2.0		**6.% S&C**		

Notes:
QL Preliminary data, "quick look"
S&C Stability and control
B/S Blade/sting mounting interference study
( )* Data obtained for the launch vehicle without HXRV attached/interference
HF High fidelity
AH Aerodynamic Heating
Bold Test completed

Table 1. Aerodynamic Test Matrix.

Flight Mach	Booster Stack	Research Vehicle	Stage Separation	Boundary Layer Transition
0.2				
0.4				
0.6				
0.8	S	S		
0.95	S	S		
1.1	S	S		
1.4	S	S		
2.0	S	S		
2.8	S	S		
3.5	S	S		
4.6	S	S		
6.0	D	S	D	D
10.0	D	S	D	D

1st Priority	
2nd Priority	Not Applicable

S=Scoped quick; outline edges of envelope; evolving keel line/configuration (addresses flyability; preliminary sim/trajectory development)
D=Detailed define control effectiveness on current configuration (addresses controllability; used in sim)
B=Benchmarked fully defined control characteristics; uncertainty analysis (current configuration; current flight trajectory including dispersions; used to build flight control laws)

Table 2. Requirements and Status.

Facility	Mach Number	Test Section	Type	Unit Rey. No.
M-10 HRNT	10	31"x31"	Blowdown	0.5 - 2.2 $\times 10^6$/ft
M 6, 20"	6	20"x20"	Blowdown	0.5 - 7.1 $\times 10^6$/ft
PSWT	M 0.4 - 4.6	48" x 48"x9'	Blowdown	2 - 50 $\times 10^6$/ft
L/M TF	M 0.4 - 4.6	48"x48"x5'	Blowdown	2 - 38 $\times 10^6$/ft
NTS M 0.4 - 4.6	M 0.4 - 4.6	48" by 48"	Blowdown	1 - 43 $\times 10^6$/ft
VKF - B	6, 8	50" Diam	Continuous	0.3 - 4.7 $\times 10^6$/ft

Table 3. Aerodynamic Test Facility Statistics.

Facility	Fligh Mach Number Simulation		
	Mach 7	Mach 5	Mach 10
DCM (GASL)		HXCD	
CHSTF (LaRC)		DFX HXEM	
AHSTF (LaRC)	DFX HXEM	DFX HXEM	
LEG V (GASL)	HXEM		
8' HTT (LaRC)	HXEM, HXFE Hyper-X,	HXEM	
HYPULSE (GASL)	HCM HSM		HCM HSM

Table 4. Propulsion Test Matrix.

Facility	Primary use	Flow energizing method (Max $T_{t,\infty}$ (°R))	Simulated flight Mach No.*	Nozzle exit Mach No.	Nozzle exit size (in.)	Test section dimensions (ft)	Dynamic Pressure PSF
Direct-Connect Module GASL	Combustor tests	H_2/O_2/Air combustion (3800)	4.0 to 7.5	2.2	4.71 x 6.69	----------	>4000 PSF @ M5 & 7
Combustion-Heated Scramjet Test Facility (CHSTF)	Engine tests	H_2/O_2/Air combustion (3000)	3.5 to 5.0 4.7 to 6.0	3.5 4.7	13.26 x 13.26	2.5 W x 3.5 H x 8 L	1500 PSF @ M5
Arc-Heated Scramjet Test Facility (AHSTF)	Engine tests	Linde (N=3) Arc Heater (5200)	4.7 to 5.5 6.0 to 8.0	4.7 6.0	11.17 x 11.17 10.89 x 10.89	4 dia. x 11 L	800 600
8-ft High Temperature Tunnel (8' HTT)	Engine tests	CH_4/O_2/Air combustion (3560)	4.0 5.0 6.8	4.0 5.0 6.8	96 diameter	8 dia. x 12 L (26 dia. chamber)	1500 PSF @ M7
Hypersonic Pulse facility (HYPULSE)	Engine and Combustor tests	RST (15 550)	7 10	6.5	24 dia.	7 dia.	>2000 PSF @ M7 ~800 @ M10
GASL Leg IV	Engine tests	Peebble-Bed + H_2/O_2/Air combustion (5200)	5 7	4.7 6	13.26 x 13.26	2.5 W x 3.5 H x 8 L	1200 1700

Table 5. Propulsion Test Facility Statistics.

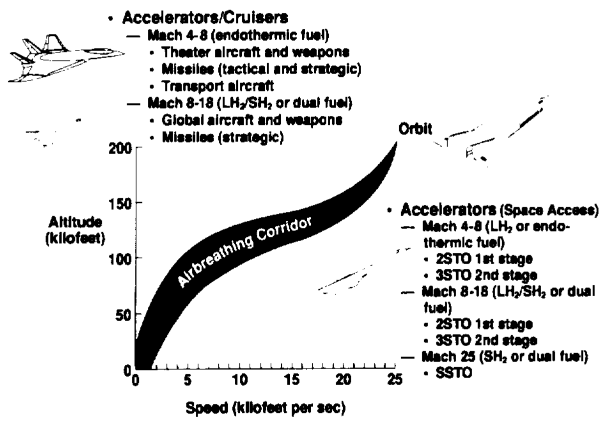

Figure 1. Potential Airbreathing Hypersonic Vehicle Applications.

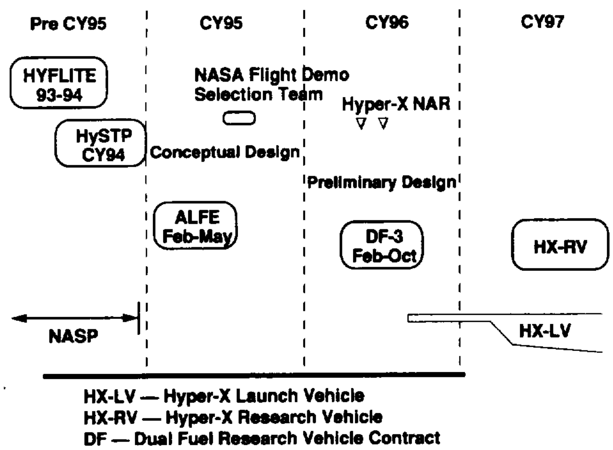

Figure 2. Recent Hypersonic Flight Test Design Studies.

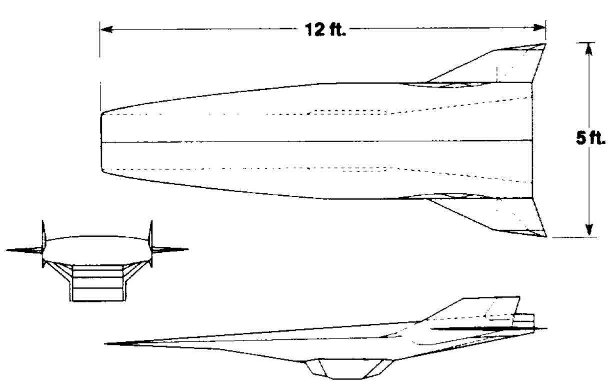

Figure 3. Hyper-X Research Vehicle Configuration.

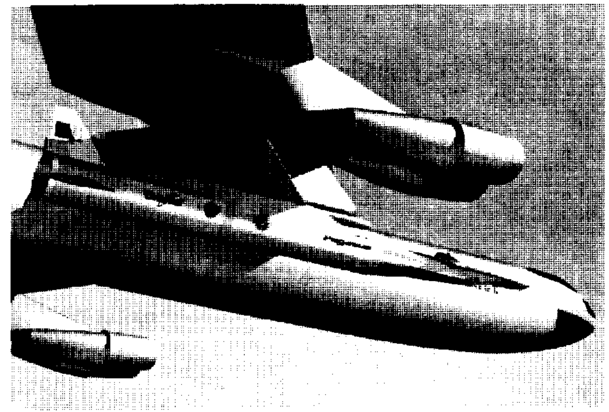

Figure 4. HXLV on B-52.

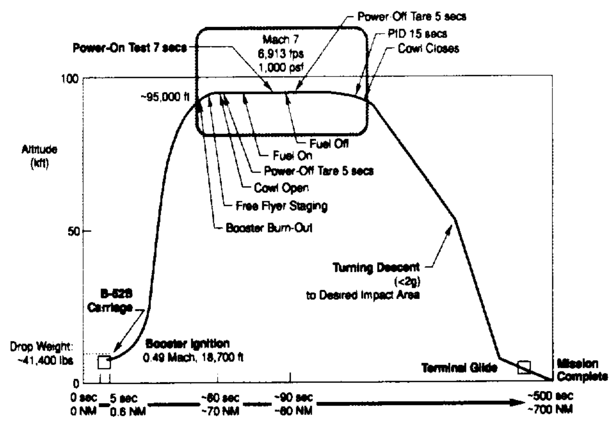

Figure 5. Nominal Hyper-X Flight Trajectory.

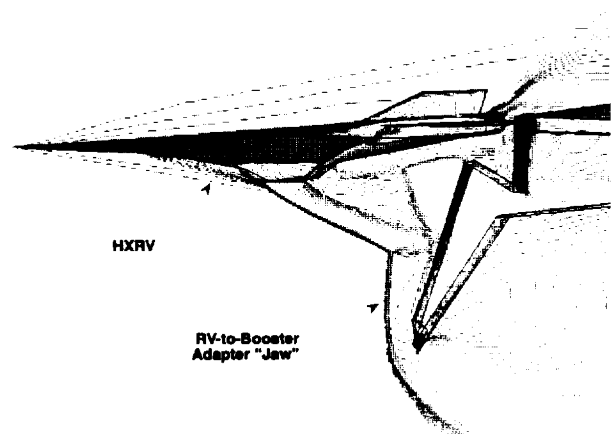

Figure 6. Typical CFD Solution for RV Stage Separation.

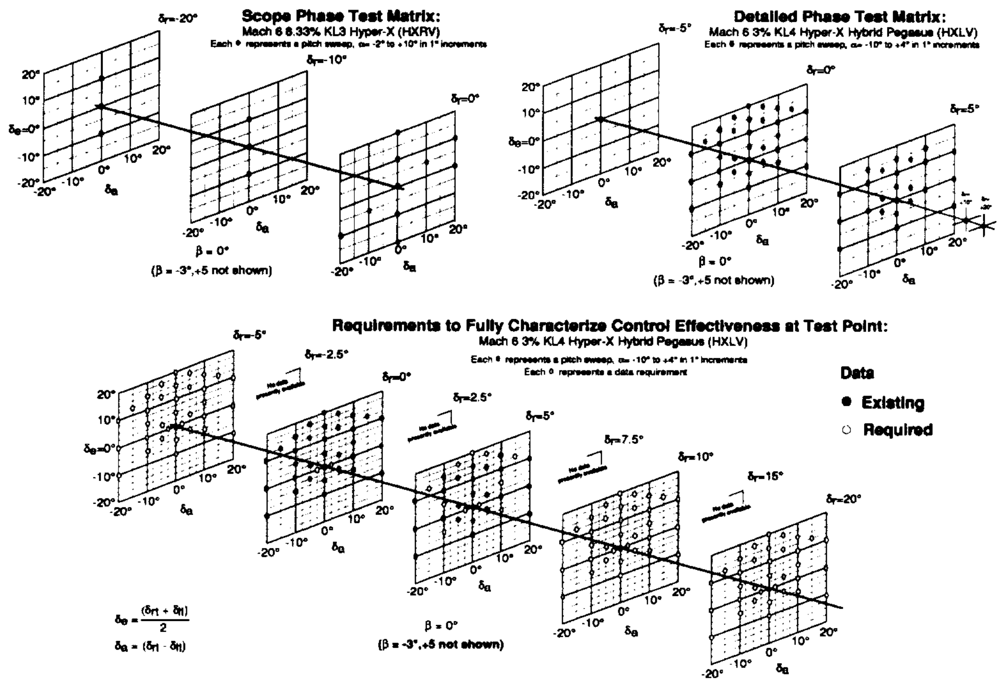

Figure 7. Aero Database Phases.

Figure 8. Photo of the 8.33% Scale Wind Tunnel Model in the NASA Langley 20-Inch Mach 6 Tunnel.

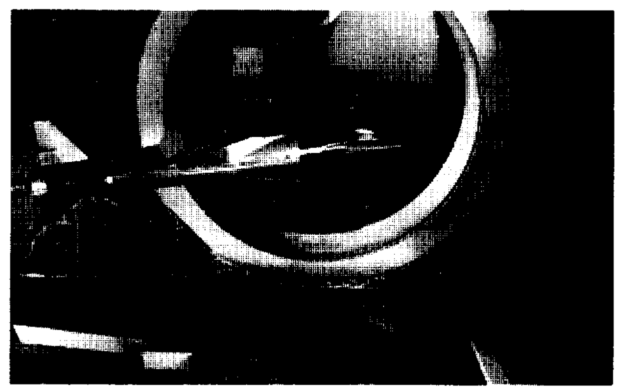

Figure 10. 3% Booster Stack Model in 20-Inch Mach 6 Tunnel.

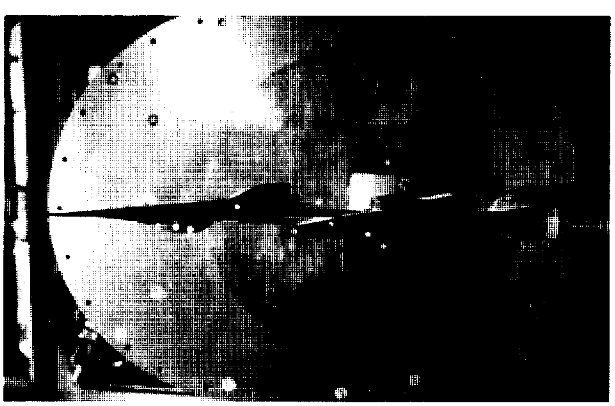

Figure 9. 8.33% Stage Separation in 20-Inch Mach 6 Tunnel.

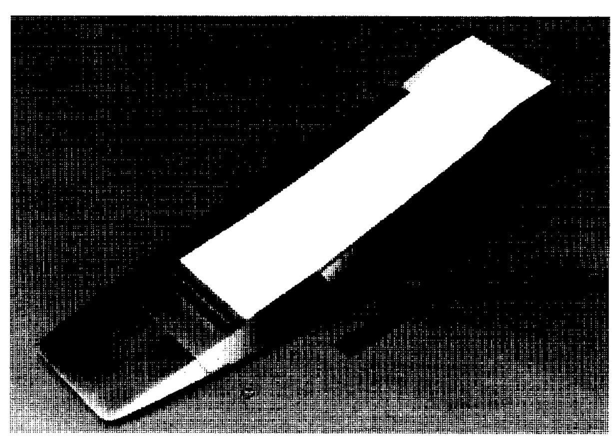

Figure 11. Boundary Layer Transition/Tripping Model.

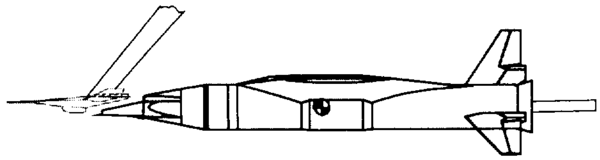

Figure 12. Mach 6 Stage Separation Test.

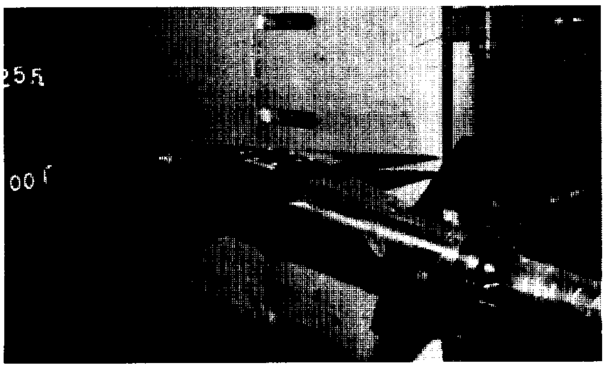

Figure 13. 6% HXLV in Lockheed Martin Tunnel.

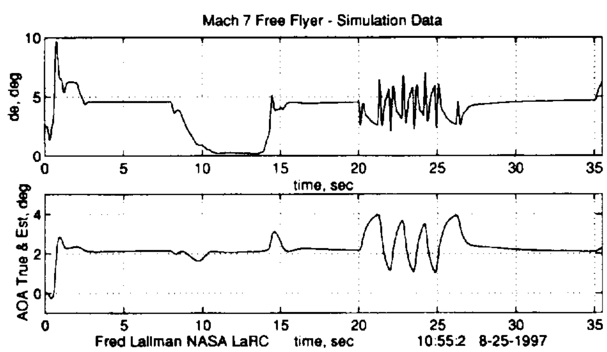

Figure 14. Flight Control Evaluation (elevator and angle-of-attack history).

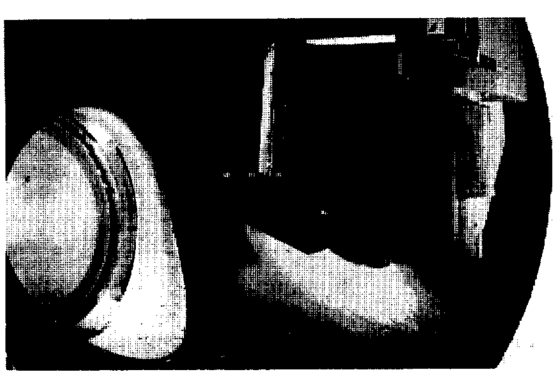

Figure 15. DFX Engine in AHSTF.

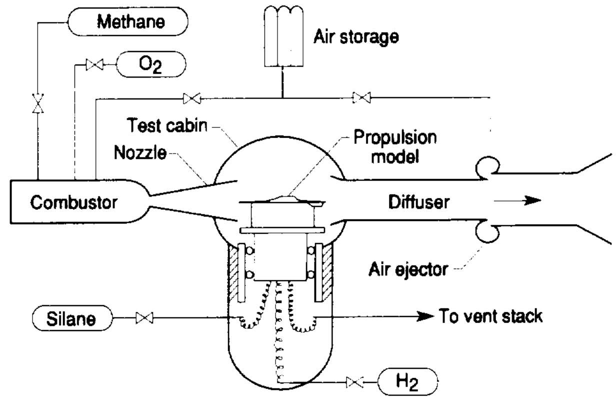

Figure 16. Schematic Drawing of 8' HTT for Air-breathing Propulsion Testing.

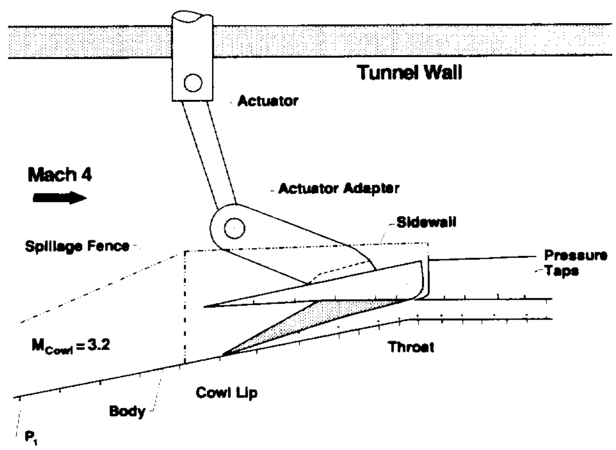

Figure 17. Generic Experimental 2-D Inlet Model.

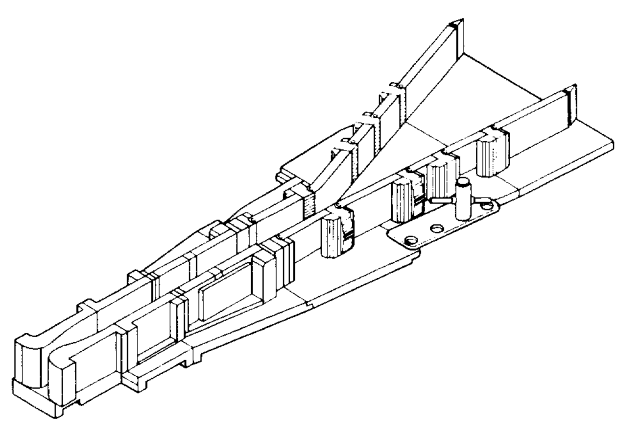

Figure 18. Schematic of Leg II Direct-Connect Combustor Module.

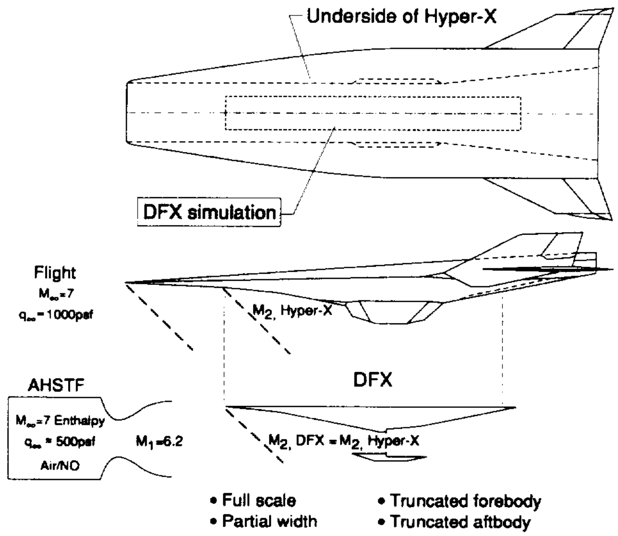

Figure 19. DFX Simulation Compared to Hyper-X.

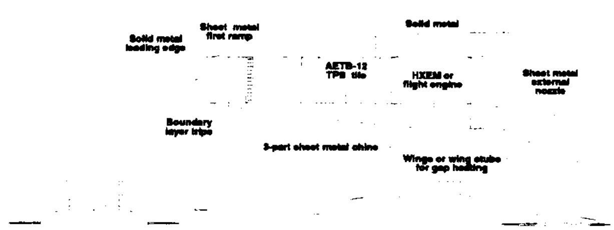

Figure 20. HXFFS model airframe characteristics.

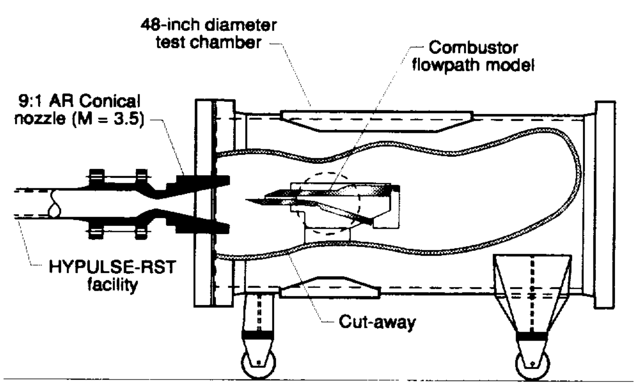

Figure 21. Sketch of HYPULSE RST and HCM Scramjet Model.

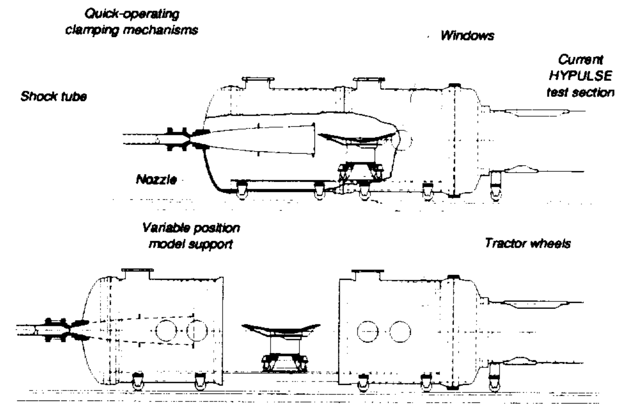

Figure 22. Installation of HSM in the 84-Inch HyPULSE Test Cabin.

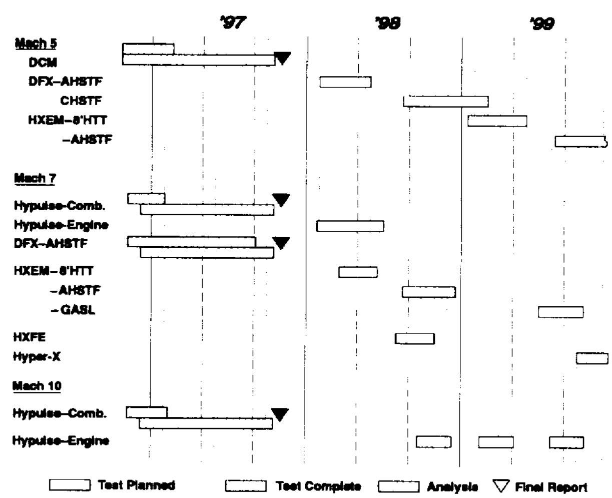

Figure 23. Engine Test Schedule.

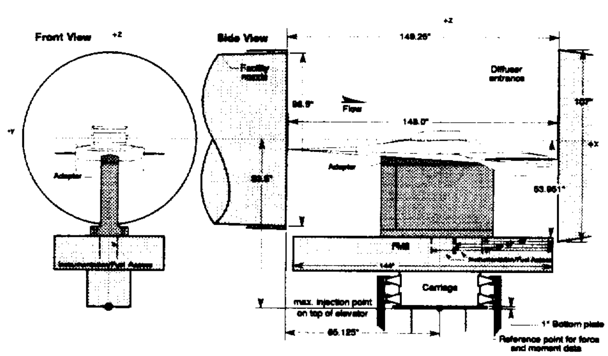

Figure 24. Installation Drawing of the Hyper-X Research Vehicle in the 8' HTT.