The NASA Langley Scramjet Test Complex

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Abstract

The NASA Langley Scramjet Test Complex consists of five propulsion facilities which cover a wide spectrum of supersonic combustion ramjet (scramjet) test capabilities. These facilities permit observation of the effects on scramjet performance of speed and dynamic pressure from Mach 3.5 to near-orbital speeds, engine size from Mach 4 to 7, and test gas composition from Mach 4 to 7. In the Mach 3.5 to 8 speed range, the complex includes a direct-connect combustor test facility, two small-scale complete engine test facilities, and a large-scale complete engine test facility. In the hypervelocity speed range, a shock-expansion tube is used for combustor tests from Mach 12 to Mach 17+. This facility has recently been operated in a tunnel mode, to explore the possibility of semi-free-jet testing of complete engine modules at hypervelocity conditions. This paper presents a description of the current configurations and capabilities of the facilities of the NASA Langley Scramjet Test Complex, reviews the most recent scramjet tests in the facilities, and discusses comparative engine tests designed to gain information about ground facility effects on scramjet performance.

Nomenclature

Ar	argon
CH_4	methane
CO_2	carbon dioxide
FPI	fuel plume imaging
h	enthalpy
H_2	hydrogen
H_2O	water vapor
G	combustor gap size
LOS	line-of-sight
$\dot{m}$	facility total mass flow rate
M	Mach number
N_2	nitrogen
NO	nitric oxide
NO_2	nitrogen dioxide
O	atomic oxygen
O_2	oxygen
OH	hydroxyl radical
p	pressure
q	dynamic pressure
Re	Reynolds number
SiH_4	silane
T	temperature
v	velocity
X	axial distance
α_w	wedge angle simulating aircraft angle-of-attack
ϕ	fuel equivalence ratio
η	efficiency
ΔF	fuel-on minus fuel-off scramjet axial force

Subscripts

c	combustion
inj	fuel injection condition
m	mixing
tg*	test gas
t,plen	stagnation condition in the facility plenum chamber
t,1	stagnation condition downstream of aircraft bow shock
t,∞	stagnation condition upstream of aircraft bow shock
1	static condition downstream of aircraft bow shock
2	static condition at scramjet combustor entrance
∞	static condition upstream of aircraft bow shock

*The flow exiting the nozzles of the facilities of the NASA Langley Scramjet Test Complex can be representative of flow conditions upstream of a flight vehicle, downstream of a vehicle bow shock, or at the entrance to a scramjet combustor. Therefore, in this paper, the flow exiting the nozzles and entering the test model will be referred to as "test gas" with the subscript "tg."

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Introduction

Scramjet research in the United States historically has been cyclic in nature with periods of very intense activity surrounded by comparative lulls (ref. 1). The first period of scramjet activity began in the early 1960's and ended with the conclusion of the Hypersonic Research Engine (HRE) project (ref. 2) about 1975. After a national lull in scramjet research activity from about 1975 to 1985, research efforts again became intense with the start of the National Aero-Space Plane (NASP) Program. Significant accomplishments were made during the NASP Program, but research activity declined dramatically with the termination of the program in 1994.

Coincident with these active scramjet research periods was the emergence of new and reactivated hypersonic propulsion test facilities for airbreathing engines (ref. 3). These facilities are distinguished from hypersonic aerodynamic 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. The propulsion community realized at the start of the NASP Program that the country was severely lacking in airbreathing propulsion test capability. Most of the facilities used during the HRE era were either on standby, mothballed, or disassembled. The only government facilities actively operating as scramjet test facilities were NASA Langley's Direct-Connect Supersonic Combustion Test Facility (refs. 4-5), Combustion-Heated Scramjet Test Facility (refs. 6-8), and Arc-Heated Scramjet Test Facility (refs. 9-11). In addition, prior to the start of the NASP Program, Langley's 8-Foot High Temperature Tunnel (8-Foot HTT) was undergoing renovation and upgrade with the addition of an oxygen replenishment system and two new nozzles (Mach 4 and 5) to supplement the existing Mach 7 nozzle. This extension of capabilities enabled the 8-Foot HTT to be used as a scramjet test facility (refs. 12-14).

As the NASP Program gained momentum after 1985, new and reactivated propulsion facilities were a part of the technology development plan. NASP-funded Engine Test Facilities were constructed at Aerojet (ref. 15) and modified at Marquardt (ref. 16) with the goal of testing complete subscale scramjet engine modules at Mach 5 and 8. The NASA Lewis Hypersonic Tunnel Facility (HTF), with Mach 5, 6, and 7 capabilities, was also reactivated just after the NASP Program (ref. 17). Direct-connect tests were conducted from Mach 2.5 to 8 in GASL's hydrogen-combustion-heated facility, a direct-connect combustor test facility was assembled at the Johns Hopkins University/Applied Physics Laboratory (JHU/APL) for tests from Mach 5 to 8 (ref. 18), and the Direct-Connect Arc Facility (refs. 19 and 20) was assembled at NASA Ames Research Center for combustor tests from Mach 9 to 12. At hypervelocity speeds, scramjet combustor research was conducted up to Mach 18 in Cal Tech's new T5 reflected shock tunnel (ref. 21) and semi-free jet scramjet combustor tests were conducted at Mach 9.3 in the Calspan shock tunnel (ref. 22) and in the NASA Ames 16-Inch Shock Tunnel from Mach 12 to 16 (ref. 23). In addition, the NASA Langley expansion tube, which had been dismantled in 1982, was reassembled at GASL and renamed the HYPULSE shock-expansion tube (refs. 24-26). This facility, which was reactivated in 1989, provided combustor test capability at Mach 13.5 and 17. During the NASP Program, the HYPULSE test capability was extended to include Mach 14 and 15 conditions at higher pressures.

Currently, in 1996, scramjet research is once again at a low point. The only activities are continuing NASA Langley research in hydrogen-fueled scramjets and an Air Force effort in hydrocarbon-fueled scramjet research. True to history, some of the NASP-era propulsion facilities are no longer active, and some of the active facilities are devoted to research areas other than propulsion. One exception is the NASA Lewis HTF where a rocket-based combined cycle engine is being tested. The other exception, as in the period from 1975 to 1985, is the propulsion facility complex at NASA Langley Research Center. This paper describes the current configurations and capabilities of the facilities of the NASA Langley Scramjet Test Complex, reviews results from some of the most recent scramjet tests in the facilities, and discusses comparative engine tests designed to gain information about ground facility effects on scramjet performance.

NASA Langley Scramjet Test Complex Facilities

The five propulsion facilities comprising the NASA Langley Scramjet Test Complex are listed in Table 1. The complex consists of the Direct-Connect Supersonic Combustion Test Facility (DCSCTF), the Combustion-Heated Scramjet Test Facility (CHSTF), the Arc-Heated Scramjet Test Facility (AHSTF), the 8-Foot High Temperature Tunnel (8-Foot HTT), and the Hypersonic Pulse Facility (HYPULSE) shock-expansion tube/tunnel. A flight Mach number/altitude map showing current operational ranges of the facilities of the Scramjet Test Complex with constant stagnation pressure, stagnation enthalpy, and dynamic pressure lines noted is presented in figure 1. This complex is unequaled in the range of scramjet performance parametric studies which are possible. The facilities permit the observation of the effects on scramjet performance of speed and dynamic pressure (Mach 3.5 to near-orbital speeds), engine size (Mach 4 to 7 in nozzle exits of approximately 1-foot square and 8-foot diameter), and test gas composition (Mach 4 to 7 with arc-heated, hydrogen-combustion-heated, and methane-combustion-heated flows).

Direct-Connect Supersonic Combustion Test Facility

The purpose of the Direct-Connect Supersonic Combustion Test Facility (refs. 4-5) 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 stagnation enthalpy necessary to simulate flight Mach number for the combustor tests 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).

Facility Configuration: As shown in figure 2, the DCSCTF is located in a 16- x 16- x 52-ft. test cell with forced air ventilation through the entire cell. Test air is supplied from a high pressure bottle field and is regulated to 550 psia (nominal) prior to entering the test cell. Gaseous hydrogen is supplied from tube trailers at a maximum pressure of 2400 psia and is regulated to 720 psia. Similarly, oxygen is supplied from trailers at a maximum pressure of 2400 psia and is regulated to 720 psia prior to entering the test cell. Purge nitrogen is also supplied from a tube trailer at a maximum pressure of 2400 psia with the pressure regulated to 230 psia.

During facility heater (Figure 3) operation, oxygen is injected into the airstream from instream injectors and premixed before hydrogen is injected. The hydrogen is injected into the air/oxygen mixture from instream injectors centered in holes located in a baffle plate upstream of the water-cooled combustor section. Ignition of the gas mixture is achieved using an electric-spark-activated hydrogen/oxygen torch ignitor.

Various facility nozzles can be attached to the facility combustion heater to simulate scramjet combustor entrance conditions. Two nozzles currently are available for use in the DCSCTF; both are two-dimensional (rectangular) contoured nozzles. The first is a Mach 2 nozzle with throat dimensions of 0.846 x 3.46 inches and exit dimensions of 1.52 x 3.46 inches; the second is a Mach 2.7 nozzle with throat dimensions of 0.356 x 6.69 inches and exit dimensions of 1.50 x 6.69 inches. Vacuum for altitude simulation is provided by a 70-foot diameter vacuum sphere/steam ejector system (requiring up to 25,000 lbm/hr of steam).

Gaseous hydrogen (at ambient temperature) is the primary fuel used in the scramjet combustors tested in the DCSCTF, although tests using other types of fuel are conducted occasionally. The hydrogen fuel for the scramjets comes from the same trailers as the hydrogen for the facility heater and is regulated to 720 psia before entering the scramjet fuel manifolds. A 20/80-percent mixture of silane/hydrogen (by volume) is supplied from K-size cylinders (maximum storage pressure of 2400 psia) for use as an ignitor/pilot of the primary fuel in the scramjet combustor.

The data acquisition system for the DCSCTF, which has recently been updated, consists of a commercially available software package (AutoNet) running on a Pentium processor. The new system incorporates a NEFF 300 signal conditioner and a NEFF 600 amplifier/multiplexer capable of supporting 128 channels. In addition to the A/D capabilities of the NEFF, up to 512 static pressure measurements can be recorded using a Pressure Systems Incorporated (PSI) 8400 electronic sensing pressure (ESP) system and sixteen 32 port modules. Nonintrusive laser-based diagnostics are commonly used in the DCSCTF and the combustor test section can be mounted on a thrust-measuring system.

Test Capabilities: The DCSCTF normally operates at heater stagnation pressures between 115 and 500 psia and at heater stagnation temperatures between 1600 and 3800 R. Test gas mass flow rates range from 1 to 30 lbm/s. The Mach number/altitude map for the DCSCTF is shown in figure 4. The left boundary is the lower temperature limit of stable operation of the heater (~1600 R) and the right boundary represents the maximum operational stagnation temperature (~3800 R). The lower (diagonal) boundary reflects the maximum allowable heater pressure (~500 psia) and the upper boundary reflects the lowest pressure for stable heater operation (~115 psia); however, these pressures translate into higher simulated stagnation pressures on the flight envelope when typical scramjet inlet and aircraft bow shock losses are included. (An inlet kinetic energy efficiency of 0.985 was assumed.) The standard operating conditions of the DCSCTF are shown by the symbols on Figure 4 and are tabulated in Table 2. The normal test schedule is 2 or 3 test days per week. Tests average 20 to 30 seconds duration with multiple tests (5 to 10) per day.

Test Gas Contaminants: Calculated test gas compositions for the standard operating conditions of the DCSCTF are tabulated in Table 2. The data are listed only if the species mole fractions are 0.0001 or greater. These calculations were made with finite-rate chemistry during expansion through the facility nozzle. The primary contaminant in the test gas is water vapor which varies from 0.083 mole fraction at Mach 4 conditions to 0.358 at Mach 7.5 conditions. A small amount of nitric oxide (0.004 mole fraction) is also present in the test stream at the Mach 7.5 condition.

Recent Scramjet Tests: The DCSCTF has been in operation since 1969. Extensive tests of various combustor and fuel injector configurations have been conducted (ref. 27, for example) with the primary interest being the distributions of fuel-test gas mixing and combustion. One of the most recent tests (ref. 28) investigated the effects of combustion on mixing using both swept compression and expansion ramp fuel injectors (Figure 5). As shown in Figure 6, combustion efficiency was higher for the expansion ramp configuration. These results, coupled with computational fluid dynamic (CFD) analyses of the two ramp configurations, demonstrated the detrimental effects of combustion on fuel-test gas mixing (Figure 7). The heat release associated with combustion was shown to reduce the longitudinal vorticity. Although delayed in the expansion ramp model, once combustion began, it occurred rapidly because of the premixing.

The current test program in the DCSCTF is the investigation of a rocket-based combined cycle (RBCC) engine in cooperation with industry partners Aerojet and GASL. The fuel for this engine is gaseous hydrogen. Both static and flight simulation tests will be conducted.

Combustion-Heated Scramjet Test Facility

The purpose of the Combustion-Heated Scramjet Test Facility (refs. 6-8) is to test complete (i.e., includes inlet, combustor, and partial nozzle), subscale, scramjet component integration models in flows with stagnation enthalpies duplicating that of flight at Mach numbers from 3.5 to 6. The flow at the exit of the facility nozzle simulates the flow entering a scramjet engine module in flight. As in the DCSCTF, the stagnation enthalpy necessary to simulate flight Mach number for the engine model flowpath tests 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).

Facility Configuration: As shown in figure 8, the CHSTF is located in a 16- x 16- x 52-ft. test cell with forced air ventilation through the entire cell. This test cell is adjacent to that of the DCSCTF and the two facilities share the same gas, vacuum, and data acquisition systems. As noted for the DCSCTF, test air for the CHSTF is supplied from a high pressure bottle field and is regulated to 550 psia (nominal) prior to entering the test cell. Gaseous hydrogen is supplied from tube trailers at a maximum pressure of 2400 psia and is regulated to 720 psia. Similarly, oxygen is supplied from trailers at a maximum pressure of 2400 psia and is regulated to 720 psia prior to entering the test cell. Purge nitrogen is also supplied from a tube trailer at a maximum pressure of 2400 psia with the pressure regulated to 230 psia.

During facility heater operation, oxygen is injected into the airstream from instream injectors centered in holes in a baffle plate and premixed upstream of the plane where hydrogen is injected (Figure 9). The hydrogen is injected into the air/oxygen mixture from instream injectors centered in holes located in a baffle plate upstream of the air-cooled combustor section. Ignition of the gas mixture is achieved using an electric-spark-activated hydrogen/oxygen torch ignitor. The facility may be operated with either a Mach 3.5 or a 4.7 nozzle. Both nozzles have square cross sections and are contoured to exit dimensions of 13.26 x 13.26 inches. The nozzle flows exhaust as free jets into the test section which is 42 inches high x 30 inches wide x 96 inches long. A diffuser catch cone extends to within about 22.5 inches of the nozzle exit. Vacuum for altitude simulation is provided by a 70-foot diameter vacuum sphere/steam ejector system (requiring up to 25,000 lbm/hr of steam).

Either gaseous hydrogen or gaseous ethylene (both at ambient temperature) may be used as fuel in the scramjet engines tested in the CHSTF. The hydrogen fuel for the scramjets is supplied from the same trailers as the hydrogen for the facility heater and is regulated to 720 psia before entering the scramjet fuel manifolds. The ethylene fuel is supplied from an industrial tube trailer at a maximum pressure of 1200 psia. A 20/80-percent mixture of silane/hydrogen (by volume) is supplied from K-size cylinders at 2400 psia for use in the scramjet model as an ignitor/pilot gas to aid in the combustion of the primary fuel.

The data acquisition system for the CHSTF is common with the DCSCTF and has recently been up-dated as previously discussed. The system consists of a commercially available software package (AutoNet) running on a Pentium processor. The system incorporates a NEFF 300 signal conditioner and a NEFF 600 amplifier/multiplexer capable of supporting 128 channels. In addition to the A/D capabilities of the NEFF, up to 512 static pressure measurements can be recorded using a PSI 8400 ESP system and sixteen 32 port modules. Scramjet thrust/drag is measured with a load cell.

Test Capabilities: The hydrogen-air-oxygen heater is rated for a maximum pressure of 500 psia and a maximum temperature of 3000 R. The facility normally operates at heater stagnation pressures between 50 and 500 psia and at stagnation temperatures between 1300 and 3000 R. Test gas mass flow rates range from 15 to 60 lbm/s. The Mach number/altitude map for the CHSTF is shown in figure 10. For each facility nozzle, the simulated flight Mach number range may be increased through appropriate increases in stagnation enthalpy by assuming that the facility nozzle supplies flow at conditions behind the vehicle bow shock. Thus, scramjet tests in the facility at stagnation enthalpies greater than that corresponding to the nozzle exit Mach number represent various degrees of aircraft forebody precompression.

The left vertical boundary of the flight simulation envelope is the nozzle exit Mach number of 3.5, and the right vertical boundary reflects the maximum heater operating temperature of 3000 R. The upper inclined boundary represents the minimum operating pressure of 50 psia up to an altitude where a simulated flight dynamic pressure of 250 lbf/ft^2 is imposed as a limit. The lower inclined boundary reflects the maximum mass flow rate to the heater at the Mach 3.5 limit and the maximum heater operating pressure at the Mach 6 limit. The standard operating conditions of the CHSTF are shown by the symbols on Figure 10 and are tabulated in Table 2.

The normal test schedule of the CHSTF is 2 to 3 days per week. Tests average 20 to 30 seconds duration with multiple tests (5+) per day.

<u>Test Gas Contaminants:</u> Calculated test gas compositions for the standard operating conditions of the CHSTF are tabulated in Table 2. These calculations were made with finite-rate chemistry through the facility nozzle. The primary contaminant in the test gas is water vapor, which varies from 0.085 mole fraction at Mach 4 conditions to 0.179 at Mach 5.5 conditions.

<u>Recent Scramjet Tests</u>: The CHSTF has been in operation since 1978. Extensive tests have been conducted to investigate the operability and performance of various component integration engines. Scramjet engines tested in this facility (see Table 3) include the NASA 3-Strut, NASA Parametric, NASA Step-Strut, NASP Government Baseline, Rocketdyne A2 , Pratt and Whitney C, JHU/APL B1, and, most recently, the Rocketdyne Hydrocarbon-Fueled Scramjet. Since 1978, a total of 1874 scramjet tests have been conducted in the CHSTF.

The purpose of the Rocketdyne Hydrocarbon-Fueled Scramjet test series was to establish a database for a gaseous ethylene-fueled, fixed-geometry, complete scramjet engine module (Figure 11). The tests were conducted at simulated Mach 4 flight conditions with the ethylene acting as a surrogate for cracked JP fuel. Extensive hydrogen-fueled tests also were conducted to compare with the ethylene-fueled tests. The results indicate that flameholding with ethylene fuel was much more difficult to achieve than with hydrogen; hence, a pilot flame (provided by silane-hydrogen gas) was required to maintain ethylene combustion at the simulated Mach 4 conditions. However, as shown in figure 12, by piloting the ethylene and by systematically varying the fuel and pilot fuel injection locations (within available locations), thrust performance and engine operability were achieved comparable to the best performance with hydrogen fuel (ref. 29).

Arc-Heated Scramjet Test Facility

The purpose of the Arc-Heated Scramjet Test Facility (refs. 9-11) is to test complete, subscale, scramjet component integration models in flows with stagnation enthalpies duplicating that of flight at Mach numbers from 4.7 to 8 (see Figure 13). The flow at the exit of the facility nozzle simulates the flow entering a scramjet engine module in flight. The stagnation enthalpy necessary to simulate flight Mach number for the engine tests is achieved by passing air through a rotating electric arc.

<u>Facility Configuration:</u> The main air flow enters the arc heater from the NASA Langley 5000 psig air system. This air, with flow rates ranging from 0.50 to 2.20 lbm/s, passes through the rotating electric arc in the Linde (N = 3) heater (Fig. 14), where it is heated to a stagnation enthalpy of approximately 3000 Btu/lbm and a stagnation pressure that does not exceed 660 psia. The power to the arc is from two 10-megawatt direct-current power supplies connected in series with stabilizing ballast resistors. Power can be varied in 33 increments. Maximum power delivered to the arc is approximately 13 megawatts and the maximum power into the air is approximately 6.5 megawatts. As the arc-heated air exits the heater, ambient temperature (bypass) air from a second leg of the 5000 psig air system is injected radially from two axial stations of a conically diverging section. The quantity of bypass air is varied from approximately 1.0 to 10.0 lbm/s to dilute the heated gas to achieve the required test stagnation enthalpy in the facility plenum chamber (usually between 500 and 1600 Btu/lbm). The arc heater and nozzle throat sections are cooled with deionized water which can be supplied at pressures up to 1400 psig.

Mach 4.7 and 6 nozzles are available for use in the AHSTF. Both are square cross-section contoured nozzles. Exit dimensions of the Mach 4.7 nozzle are 11.17 x 11.17 inches and the exit dimensions of the Mach 6 nozzle are 10.89 x 10.89 inches. Mach 4.7 nozzles in both the CHSTF and the AHSTF and the capability of overlapping test conditions allow the testing of a scramjet engine in the two facilities with only a difference in test gas composition. The AHSTF nozzle flows exhaust as free jets into a 4-foot diameter test section which is 11 feet in length. The test gas and scramjet exhaust gases are diffused to subsonic velocities in a 33.5 foot long, 4-foot diameter, straight-pipe diffuser prior to entering a subsonic diffuser (Fig. 13). This 24.5-foot long diffuser diverges to a 10-foot diameter, 13-foot long duct which houses an air-to-water heat exchanger to cool the hot gases prior to entry into a 100-foot diameter vacuum sphere. The sphere can be evacuated to 0.020 psia by a 3-stage steam ejector which uses 26,000 lbm/hr of steam.

Hydrogen fuel (at ambient temperature) enters the test section from a 31.46-ft^3 storage bottle with a usual fill pressure of 1200 psig; the pressure is regulated to a maximum pressure of 625 psig prior to entering the scramjet fuel manifolds. A 20/80-percent mixture of silane/hydrogen (by volume) is supplied from K-size cylinders at 2400 psia (regulated to a maximum of 650 psig) for use as an ignitor/pilot gas to aid in the combustion of the hydrogen fuel.

The data acquisition system for the AHSTF consists of a Modcomp 9230 computer with 32-bit CPU, 192 analog channels, and 16 digital channels. Model static pressures are measured with a 448 channel PSI 8400 ESP system. Forces and moments on the scramjet models are measured with a six-component force balance.

Test Capabilities: The Mach number/altitude operating map for the AHSTF is shown in figure 15. This map shows envelopes for the Mach 4.7 and the Mach 6 nozzles. For each facility nozzle, the simulated flight Mach number range may be increased through appropriate increases in stagnation enthalpy by assuming that the facility nozzle supplies flow at conditions behind the vehicle bow shock. The wedge-angle scales at the bottom of the figure indicate the aircraft attitude required to shock the air (through a single oblique shock) from the simulated flight Mach number to the Mach number at the exit of the facility nozzle. Thus, scramjet tests in the facility at stagnation enthalpies greater than that corresponding to the nozzle exit Mach number represent various degrees of aircraft forebody precompression.

For the Mach 4.7 nozzle envelope, the left vertical boundary is the nozzle exit Mach number. The lower inclined boundary reflects the design stagnation pressure of the nozzle (210 psia) and the right vertical boundary reflects the design stagnation temperature of the nozzle (2700 R). The upper inclined boundary reflects the minimum power setting of the arc heater power supplies.

For the Mach 6 nozzle envelope, the left vertical boundary is the nozzle exit Mach number. The lower inclined boundary reflects the maximum heater design pressure (660 psia) at Mach 6; however, the limiting factor becomes maximum power available at Mach 7 and Mach 8. The right vertical boundary reflects the maximum operational stagnation temperature of the nozzle, ~ 5200 R. The upper inclined boundary reflects a minimum reasonable nozzle exhaust back pressure of 0.10 psia. The standard operating conditions of the AHSTF are shown by the symbols on Figure 15 and are tabulated in Table 2. Nozzle exit Mach number and static temperature in Table 2 were calculated from measured nozzle exit stagnation temperature, pitot pressure, and static pressure assuming vibrationally frozen flow and finite-rate chemistry in the nozzles. The vibrationally frozen flow at the nozzle exit was assumed to remain frozen during the probe flow processes.

The normal test schedule of the AHSTF is two test days per week with four tests per test day. Test times normally range from 30 seconds at flight Mach 8 simulated conditions to 60 seconds at flight Mach 4.7 simulated conditions.

Test Gas Contaminants: The primary contaminants in air flows heated by electric arcs after passing through a plenum chamber and a converging/diverging nozzle are nitrogen oxides (primarily nitric oxide, NO) and copper contamination (from copper electrode erosion). The copper contamination level in the AHSTF has not been measured. A discussion of copper contamination in arc heaters is presented in reference 30. Calculations of test gas compositions in the AHSTF (ref. 31) at the standard operating conditions are tabulated in Table 2. These calculations employed finite-rate chemistry throughout the arc heater, plenum chamber, and the facility nozzles. The levels of NO in the test flow, verified by measurement (refs. 10 and 31), range from 0.011 mole fraction at $M_\infty = 4.7$ to 0.031 mole fraction at $M_\infty = 8$. Oxygen deficits due to the formation of NO are not replenished, and the facility operates at $M_\infty = 4.7$ with 0.203 oxygen mole fraction and at $M_\infty = 8$ with 0.194 oxygen mole fraction.

Recent Scramjet Tests: The AHSTF has been in operation for scramjet testing since 1976. Extensive tests have been conducted to examine the operability and performance of component integration engines. Scramjet engines tested in this facility (see Table 3) include the NASA 3-Strut; NASA Parametric; Rocketdyne A, A1, A2, and A2+; Pratt and Whitney C; NASP SX-20; and, currently, the NASP SXPE. Since December 1976, a total of 1264 scramjet tests have been conducted in the AHSTF.

This SXPE test series was continued by NASA Langley's Hypersonic Airbreathing Propulsion Branch after the termination of the NASP Program (as were the NASP Concept Demonstration Engine (CDE) tests in the 8-Foot HTT). The purposes of the SXPE tests were to establish a database for comparison with the larger scale CDE and to test geometric changes designed to improve the performance of the CDE. The SXPE (Figure 16) is a 12.5-percent scale and the CDE is a 30-percent scale of the NASP X-30 engine. Comparisons of scale and flow conditions between the three engines are shown in figure 17 (ref. 32). The SXPE tests have been and continue to be conducted at Mach numbers from 5 to 8. A database has been established at Mach 7 for comparison with the larger scale CDE and significant progress has been made in identifying procedures and geometric changes that improve scramjet performance.

8-Foot High Temperature Tunnel

The forerunner of the 8-Foot High Temperature Tunnel was the 8-Foot High Temperature Structures Tunnel. This facility was designed in the 1950's and was placed in service during the mid-1960's. For the following 20-plus years, the methane/air-heated facility was used to conduct research in the areas of aerothermal loads and aerothermostructures at simulated Mach 7 flight conditions. During the late 1980's and early 1990's, the tunnel was modified with the addition of an oxygen replenishment system and new Mach 4 and 5 nozzles (refs. 12-13). These modifications allowed the use of the facility as a Mach 4, 5, and 7 propulsion test facility and the facility became operational in that capacity in 1993. The facility capability for aerothermal loads and aerothermostructural research remains intact and is, in fact, enhanced by the modifications. However, in this paper, the 8-Foot HTT will only be discussed in its function as a propulsion test facility.

The purpose of the 8-Foot HTT is to test complete, larger scale and multiple-module, scramjet component integration models in flows with stagnation enthalpies duplicating that in flight at Mach 4, 5, and 7. The flow at the exit of the facility nozzle simulates the flow upstream of the aircraft bow shock in flight. The stagnation enthalpy necessary to simulate flight Mach number for the engine tests is achieved through methane-air combustion with oxygen replenishment to obtain a test gas with the same oxygen mole fraction as atmospheric air (0.2095).

Facility Configuration: A schematic of the 8-Foot High Temperature Tunnel is shown in Figure 18. Air is supplied from a 6000 psia bottle field, methane from bottles at 6000 psia, and liquid oxygen (LOX) from an 8000 gallon run tank at 2290 psia. The 8-Foot HTT combustion heater (Figure 19) consists of a laminated high-strength carbon steel pressure vessel, a stainless steel outer liner, and a Nickel 201 inner liner. High pressure air from the bottle field enters the pressure vessel through a torus at the upstream end, flows downstream between the pressure vessel and the outer liner, turns 180°, and flows upstream in the annular space between the inner and outer liners, thereby cooling the inner liner which is exposed to the hot combustion products and thermally protecting the carbon steel pressure vessel. The inner liner ends at approximately the mid-point of the combustor and the outer surface of the LOX injector ring forms a new annular channel with the outer liner which accepts the air flow from the annular space between the inner and outer liners. The LOX is injected at the beginning of the 20-inch-long annular space between the LOX ring and the outer liner where it mixes with the air. At the exit of the LOX injector ring, the oxygen-enriched air flows into the area bounded by the outer liner, turns 180°, and flows downstream to the methane fuel injection region. The methane is injected from 700 fuel injection orifices located on the downstream faces of 15 concentric rings of manifold tubing. The methane is mixed with and burns in the oxygen-enriched air in the half-length of the combustor upstream of the facility nozzle to provide stagnation pressures to 2000 psia and stagnation temperatures to 3560 R.

The combustion products (with an oxygen mole fraction of 0.2095) exiting the combustor are expanded through an air-transpiration-cooled nozzle throat section (Fig. 19), which has a throat diameter of 5.62 inches. The nozzle geometry downstream of this throat section depends upon the desired test Mach number. If Mach 4 or 5 tests are required, a mixer section and the appropriate throat and nozzle section downstream of the mixer are substituted for a portion of the original Mach 7 nozzle (Fig. 20). In the mixer section, ambient-temperature air is added to the combustion-heated test gas to achieve the stagnation enthalpy for either Mach 4 or Mach 5 flight-simulation propulsion tests.

The exit of the facility nozzle is 8 feet in diameter and the free-jet flow exhausts into a 26-foot diameter test chamber. A flow survey apparatus (which contains 13 pitot pressure probes, 13 iridium/iridium 40-percent rhodium stagnation temperature probes, and 11 static pressure probes) is positioned just downstream of the nozzle exit. This apparatus is standard test section hardware and can be rotated to various positions during a test to measure nozzle exit conditions. The length of the test section test space is 12 feet. The scramjet models which are mounted in this space can be longer than 12 feet if they do not have to be fully retracted (on the hydraulic injection system) in order to start the tunnel flow. The scramjet models are mounted on a three-component force balance system. Hydrogen fuel for the scramjet models is stored in trailers at 2200 psia and a 20/80-percent mixture of silane/hydrogen (by volume), for primary fuel ignition and/or piloting, is stored in a single trailer at 2300 psia. Maximum hydrogen flow rate to the scramjets is 5.1 lbm/s and the hydrogen is supplied at ambient temperature. Downstream of the test section, the flow is captured by the diffuser collector ring, processed by a straight-pipe diffuser, and pumped by an annular air ejector (requiring up to 1500 lbm/s) and mixing tube. The flow then passes through a second minimum and exhausts through a subsonic diffuser to the atmosphere.

The 8-Foot HTT data system consists of a low-speed data acquisition subsystem, a high-speed data acquisition subsystem, and a post-test data processing subsystem. The low-speed data acquisition subsystem is comprised of a Pentium personal computer (PC) running AutoNet data acquisition software; the NEFF 500/600/300 data acquisition system which can acquire 512 channels of thermocouple, wire strain-gage-based transducers, and preconditioned signals at 50 samples per second per channel; and the PSI 8400 data acquisition system which acquires pressure data for a maximum of 1000 channels at 10 samples per second per channel. This subsystem performs data acquisition and recording, primary data reduction and near real-time displays. The high-speed data acquisition subsystem is comprised of a 486 PC running NEFF software, which acquires and records data from the NEFF 490 system for a maximum of 31 channels at 1 million samples per second per channel. The post-test data processing subsystem consists of a workstation and associated peripherals running customized software which performs customer-specific secondary data processing, distribution, and archival.

Test Capabilities: For scramjet testing, the maximum heater stagnation conditions are 2000 psia and 3560 R. The facility has been operated at heater stagnation pressures between 1000 and 2000 psia and at stagnation temperatures between 3000 and 3560 R. However, in the standard operating mode, the combustion heater is always operated at 2000 psia and 3560 R with a gas mass flow rate exiting the combustor of 416 lbm/s. The mass flow rate for the nozzle section transpiration cooling air is 60 lbm/s (nominal). For Mach 5 operation, an additional 350 lbm/s of air is added in the mixer section, and, for Mach 4 operation, an additional 917 lbm/s of air is added in the mixer section. The Mach number/altitude map for the 8-Foot HTT is shown in figure 21. The vertical shaded regions at Mach 4 and 5 represent the new Mach 4 and 5 nozzles of the 8-Foot HTT. Recall that this facility is normally operated at stagnation enthalpies corresponding to the nozzle exit Mach numbers, thereby simulating flight conditions ahead of an aircraft bow shock. The spread of the shaded region for the Mach 6.8 nozzle depicts operation where the flow stagnation enthalpy is dropped below a level corresponding to nominal Mach 7 flight. However, the lower flight Mach numbers are achieved at the expense of increased water vapor condensation in the flow and, hence, are not considered standard operating conditions for scramjet testing. The upper boundary on the operating envelope is a 750 psia stagnation pressure limit in order to stay above the critical pressure of oxygen and the lower boundary is the 2000 psia stagnation pressure limitation imposed by the oxygen run tank pressure limit. The standard operating conditions of the 8-Foot HTT are shown by the symbols on Figure 21 and are tabulated in Table 2. A nozzle exit calibration for the Mach 6.8 case is presented in reference 14. For one shift operation, test frequency is normally one test per day and three per week. Test times are approximately 30 seconds on point.

Test Gas Contaminants: Calculated or measured (at Mach 6.8) test gas compositions for each of the standard operating conditions of the 8-Foot HTT are listed in Table 2. The calculated compositions were obtained with finite-rate chemistry in the facility nozzle expansion. The primary contaminants are water vapor and carbon dioxide. At Mach 6.8, the mole fraction of the H_2O contaminant is 0.182 and the mole fraction of the CO_2 contaminant is 0.091. No carbon monoxide or unburned hydrocarbons were detected and NO was not measured. Finite-rate calculations (shown above the measured values in Table 2) indicate that the NO mole fraction at Mach 6.8 is about 0.003.

Recent Scramjet Tests: Since becoming operational as a scramjet test facility in 1993, the 8-Foot HTT has been entirely devoted to tests (see Table 3) or calibrations associated with the NASP Concept Demonstration Engine (CDE). The CDE (Figure 22) is a 30-percent photographically scaled model of the NASP X-30 flight engine. Data from these tests are being used in conjunction with data from tests of the NASP SXPE (a 12.5-percent photographically scaled model of the NASP engine, Figure 16) to assist in evaluating scale, stagnation pressure, and test gas composition effects upon scramjet performance. Illustrations of the relative sizes of and test conditions for the SXPE in the AHSTF, the CDE in the 8-Foot HTT, and the X-30 engine in flight were previously presented in Figure 17. As noted in reference 32, tests of the SXPE and the CDE (at approximately Mach 7 conditions) provide data over a significant pressure/scale range; for instance, the ratio of the Damkohler first number (residence time/reaction time) for the flight case to that of the SXPE and CDE tests is 84 and 7.8, respectively.

HYPULSE Shock-Expansion Tube/Tunnel

The primary purpose of the HYPULSE shock-expansion tube/tunnel (refs. 24-26) is to provide airbreathing propulsion and aerothermodynamic test capability in the hypervelocity flight regime, from Mach 12 to near-orbital speeds. The NASA HYPULSE facility is currently located at and operated by GASL, Inc., in Ronkonkoma, New York, under a contract through NASA Langley Research Center. After years of service at NASA Langley in the study of hypervelocity aeroheating, the expansion tube was moved to GASL in 1988 and configured to deliver a test gas suitable for the study of hydrogen-fueled scramjet combustors at hypersonic flight conditions in support of the NASP Program. Most test experience to date has been accomplished using HYPULSE in the expansion tube mode at simulated flight Mach numbers (actual flight stagnation enthalpy) of about 14 and 15 (although some early tests were run at Mach 17) to study scramjet fuel injectors and combustors. The tube exit (combustor entrance) Mach numbers varied from 4.8 to 5, depending on the facility operation. Recently, a nozzle has been constructed and, configured in an expansion tunnel mode, the facility has been operated at Mach 14 conditions (ref. 33).

Facility Configuration: A schematic of the facility is presented in Figure 23, with the four major components—driver, intermediate (or shock) tube, acceleration tube, and test section/dump tank—identified. Dimensions of the components are given in Table 4 along with maximum pressure ratings. The lengths of the shock and acceleration tubes are variable depending on the location of the secondary diaphragm. The values listed are for the configuration depicted in Figure 23 which was used during most of the scramjet combustor flowpath testing at conditions simulating nominal Mach 14 to 15 flight. Length values in parentheses are for the facility configured for hypervelocity test conditions with stagnation enthalpy simulating flight Mach 15 and above. Additional details of the HYPULSE facility are given in references 24 and 25.

The principal of operation of the shock-expansion tube is illustrated by the distance-time (wave) diagram in Figure 24. At time zero, the tube sections are charged with their respective gases, which are separated by diaphragms as noted. The driver gas is typically 380-400 atm of cold helium (or 15-percent hydrogen in helium). The intermediate tube is filled with the test gas and the acceleration tube and the test chamber contain nitrogen at a very low pressure, typically about three orders of magnitude below the intermediate tube pressure. Flow is initiated by the main diaphragm rupture, which initiates a shock wave. This primary shock passes through the test gas (as illustrated on the x-t diagram in Figure 24), changing it to condition state 2. The secondary diaphragm is ruptured by the shock, which then accelerates into the lower pressure acceleration gas. To equilibrate the pressure between states 20 and 5, unsteady upstream expansion waves are generated. This unsteady expansion processes the test gas from condition state 2 to state 5, adding kinetic energy to the moving test gas, thus avoiding the high dissociative temperatures at stagnation conditions. The test gas exits the acceleration tube at conditions suitable for testing scramjet combustors in semi-direct connect mode. In the ideal x-t diagram of Figure 24, the test time is between the passage of the acceleration/test gas (secondary) interface and the arrival of either the tail of the secondary expansion or the expansion wave reflected from the primary (test gas/driver gas) interface. However, partial reflection of the shock, which occurs during the rupture of the secondary diaphragm, and viscous effects cause the shock to slow down, shortening the useful test time. Typically, test times with cold driver gases are between 300-400 µs. Test frequency is normally two per day and about seven per week.

Test Capabilities: A review of some of the HYPULSE test conditions is presented in reference 26. Generally, the test conditions are identified as the flight Mach number simulated by the stagnation enthalpy of the test gas. The current test envelope for HYPULSE is defined by the solid line in figure 25. This region is bounded on the bottom (low altitude) by driver fill-pressure limits. The left boundary is essentially the stagnation enthalpy for Mach 12 flight, below which test time is too short. The right boundary is drawn along an essentially constant stagnation pressure line where discrete test points have been demonstrated. The upper boundary line as drawn is arbitrary, but for scramjet testing, the limiting upper boundary is governed by static pressure requirements for scramjet combustor operation. Because HYPULSE currently is used to simulate combustor entrance conditions, the stagnation pressure limits of the flight simulation envelope were set by adjusting the equivalent test gas stagnation pressure for inlet total pressure losses assuming an inlet kinetic energy efficiency of 0.985.

Specific test conditions defined for HYPULSE scramjet combustor semi-direct connect testing are given in Table 5. Of the test conditions, the M13.5 condition was used in the early phase of the NASP Program for unit injector tests. Because of the high static temperature and low pressure, it has been replaced by the M14HP condition. In the search for a new Mach 14 test point, the M14LP condition was found with a lower (more realistic) temperature. Further modification of the facility operation through the mixing of about 10-15 percent H_2 in the He driver gas, produced the M14HP point, with nearly one-half atmosphere static pressure and a consistent temperature. The differences in air and N_2 conditions are believed to be due mainly to the acceleration of the shock wave caused by some chemical reaction of the H_2-air at the test gas/driver gas interface. The conditions with air test gas are repeatable and this point has been used extensively in fuel injector combustor flow tests during the NASP Program. The conditions identified with suffix 'SD' are obtained by using a secondary driver in which a downstream running detonation has been initiated by the primary shock. The conditions labeled with suffix 'NZ' are obtained at the exit of a 4:1 area ratio (12-inch exit diameter) nozzle attached directly to the acceleration tube. This mode of operation provides a semi free-jet test capability of a scramjet engine segment in a flow with about a 6-inch diameter core. Details of the nozzle design and test conditions are discussed in reference 33. A lower enthalpy condition corresponding to about Mach 12 flight has been identified (using the shock-induced detonation drive operation) and used to conduct some exploratory tests of a combustor-nozzle segment. Current standard test points for HYPULSE are shown by the symbols on Figure 25 and are tabulated in Table 2.

Test Gas Contaminants: Since the test gas in the HYPULSE 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. Attempts to quantify the dissociation through experimental measurements have been partially successful, and indicate mole fractions of NO between 0.030 and 0.050 at Mach 14 and 17 conditions (ref. 34). The level of atomic oxygen, O, has yet to be measured, but chemistry models indicate levels of about 0.010 mole fraction in the test gas at conditions of interest in scramjet testing. Calculated test gas compositions at the standard HYPULSE operating conditions are tabulated in Table 2. These compositions were calculated assuming that the test gas is in chemical equilibrium behind the incident shock in the shock tube and that no large amount of recombination occurs during the unsteady expansion process. The oxygen contained in the NO and O reduces the molecular oxygen mole fraction to approximately 0.199 at Mach 12 and 0.180 at Mach 17.

Recent Scramjet Tests: The HYPULSE facility has been used to study scramjet mixing and combustion for a wide range of fuel injector designs and arrangements. The test apparatus for most of the tests was a rectangular combustor model (RCM), shown schematically in Figure 26 positioned in the HYPULSE test chamber. The combustor duct is constant area with a 1- by 2-inch cross section and is 28 inches in length. Fuel injector blocks were generally located about 8 inches from the leading edge, leaving some 20 inches for mixing and combustion. Optical access was provided by sidewall windows, which were 12 inches long and extend to about 8 inches downstream of the injector location. Heated hydrogen fuel (typically about 1440 R) for the combustor models can be provided by a small gaseous detonation-driven shock tunnel (ref. 35) that fits entirely inside the dump tank and test section, replacing the usual fuel supply (Ludweig) tube. Standard data sources for the combustor models are wall static pressures and thin-film thermocouples for heat flux data. Additionally, the model sidewalls have ports which are used for mounting skin friction gauges (refs. 36 and 37) and a water vapor measurement system (ref. 38). To date, however, wall shear inferred from the processed skin friction data is inconclusive when compared with local heat-flux measurements. The three channel line-of-sight (LOS) system, shown installed in the combustor model in Figure 26, uses tunable diode lasers, but requires knowledge of the local static temperature to obtain accurate values of water vapor content. Images of the injected fuel plume are obtained by scattering a sheet of laser light off silica particles injected with the fuel (ref. 39). This technique has proven to be very successful and the images obtained have been used to infer the level of mixing of the fuel for a variety of fuel injection configurations (ref. 40). The fuel plume imaging (FPI) system is intended to provide a method to estimate the integral fuel mixing through the mapping of the fuel plume. The water vapor diagnostic is intended to provide an estimate of the combustion efficiency though the fraction of water vapor produced. The HYPULSE data acquisition/storage system consists of 152 channels with a one microsecond response and 512 kilobytes of data storage per channel.

The model has been used extensively in testing both single- and multiple-fuel injector configurations (refs. 41-43). A typical example of the scramjet fuel injectors tested in the HYPULSE is shown in the RCM as an inset in figure 27. The injector is a 10° ramp with four sonic orifices arranged in the base as shown in the cross section view. This injector has been tested in the model at different test conditions and with different measurement systems in place. Generally, the RCM was instrumented with pressure and heat flux gages along the top and bottom walls and, in some tests, the FPI system was operative allowing images of the fuel plume to be acquired. The most recent tests (ref. 43) involved the injection of heated hydrogen fuel at conditions that matched previous tests at the M14HP condition (see Table 5) with cold fuel. Data for both the heated and cold fuel were analyzed with an established method to obtain mixing and combustion performance. The results are summarized by the comparisons presented in Figure 27, which shows the ratio of normalized pressure distributions for combustion and mixing for runs with cold fuel (solid symbol) and hot fuel (open symbol) at a nominal fuel equivalence ratio of 2. These data indicate a small rise (in the range of 10 percent) in the pressure due to combustion over that of the mixing, for both hot and cold fuel injection. This small change in pressure due to heat release is characteristic of the combustion processes at hypervelocity conditions.

Facility Upgrade: Since the installation of HYPULSE at GASL, plans have been proposed to upgrade operation to higher pressures, thus making semi-free jet testing of scramjet engine segments possible. Currently, the staff at GASL is working the design of modifications to HYPULSE to enable detonation drive and to convert to a full tunnel mode of operation by the addition of a large (nominally 26-inch exit diameter) nozzle. A sketch of a proposed modified tunnel facility is illustrated in Figure 28. Projected operation of the detonation-driven expansion tunnel would be at conditions bounded by the dashed line above Mach 12 in Figure 25. Detonation drive also will permit operation as a reflected shock tunnel at conditions bounded by the dashed line from Mach 5 to 12 in Figure 25. Facility nozzle design is underway, with the 4:1 area ratio nozzle (ref. 33) providing a prototypical example. Other concepts are being investigated, including methods of using skimmer nozzles to capture only the core flow from the expansion tube.

Ground Facility Effects on Scramjet Performance

In the simulation of scramjet flight in ground test facilities, flight stagnation enthalpy, engine (or component) Mach number, and oxygen mole fraction are normally duplicated. However, the projection of scramjet ground test data to atmospheric flight situations is still a difficult task because of many inherent differences between these propulsion environments (ref. 3). Some of the usual differences are gas composition, turbulence level, and nonequilibrium vibrational and chemistry effects (ref. 44). Test gas stagnation pressure and engine size also add to the issue of flight projection of scramjet data obtained in the various facilities of the NASA Langley Scramjet Test Complex.

Experimental and analytical studies of these ground facility effects on scramjet performance are relatively rare in the literature. More studies deal with the effects of flow contaminants than with the other effects noted above. However, data on the effects of these flow contaminants on scramjet performance in the literature (such as refs. 45-50) are primarily from analytical and numerical studies looking at ignition delay and reaction rates and singular experiments such as ignition experiments in shock tubes. Some of the numerical studies have simulated flows with contaminants in portions of scramjet engines (refs. 51-52) and one study (ref. 31) has calculated the effects of nitric oxide (in a one-dimensional sense) on scramjet performance. However, it is not possible to draw absolute conclusions about the effects of flow contaminants on the performance of a propulsion system unless all aspects of the problem are studied simultaneously in a flowing system with the inclusion of ignitors and flameholders and with a knowledge of whether the combustion is mixing-controlled.

Recently, two sets of comparative performance data on complete, hydrogen-fueled scramjet modules have been obtained in three facilities of the NASA Langley Scramjet Test Complex. These data, which are still in analysis, are expected to provide valuable information about ground facility effects on scramjet performance. The first set of data was obtained from tests of the NASP CDE in the 8-Foot HTT and the NASP SXPE in the AHSTF. Recall that the SXPE is a 12.5-percent scale and the CDE a 30-percent scale of the NASP X-30 engine (see Fig. 17). One of the objectives of these tests was to determine the effect of engine size on scramjet performance. However, the determination of scale effects in these comparative tests is complicated by differences in test gas contaminants (NO in the AHSTF; CO_2 and H_2O in the 8-Foot HTT) and by differences in test gas stagnation pressure.

The second set of data was obtained from tests of the NASA Langley Parametric Scramjet in both the CHSTF and the AHSTF. In these tests, the geometric configuration of the engine was identical and the facility stagnation temperature and pressure were the same in the two facilities. Therefore, the data should indicate the differences in scramjet performance with the only test difference being the nitric oxide contaminant in the AHSTF flow and the water vapor contaminant in the CHSTF flow. Generic results of scramjet thrust performance (figure 29) show that scramjet thrust performance in the AHSTF is significantly higher than in the CHSTF. This trend is in agreement with ignition experiments and calculations (at similar local conditions) where NO enhances ignition and H_2O delays ignition; the result raises the issue of adequate flameholding and whether the combustion was mixing controlled.

A third experiment (not currently scheduled) would provide direct information on the effects of engine scale on scramjet performance. In this experiment, the SXPE would be tested in the 8-Foot HTT for comparison with the CDE at the same test conditions. In turn, these tests would provide comparative data from the SXPE in the 8-Foot HTT (water vapor and carbon dioxide contaminants) and the AHSTF (nitric oxide contaminant). Of course, the stagnation pressure and the turbulence level also would be different in tests in the two facilities.

For final analysis of ground facility effects on scramjet performance, computational fluid dynamics (CFD) must be used together with the experimental data. CFD can provide the bridge between various ground facility tests and between ground facility tests and flight tests of scramjets.

Concluding Remarks

The facilities comprising the NASA Langley Scramjet Test Complex, which cover a wide spectrum of supersonic combustion ramjet (scramjet) test capabilities, have been described in this paper. These facilities permit the observation of the effects on scramjet performance of speed and dynamic pressure from Mach 3.5 to near-orbital speeds, engine size from Mach 4 to 7, and test gas composition from Mach 4 to 7. The current configurations and capabilities of the facilities have been discussed in enough detail for potential users to determine if the facility test area sizes are sufficient for their purposes, to understand the standard and potential facility operating conditions, and to learn the data acquisition capabilities of the facilities. In addition, the paper has reviewed the most recent scramjet tests in the facilities so that the reader has knowledge of the types of tests that are normally conducted in the facilities. Finally, comparative engine tests which provide information about ground facility effects on scramjet performance were discussed to indicate factors that must be considered in using the ground facility scramjet data for projection to flight situations.

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Table 1.- Facilities of the NASA-Langley Scramjet Test Complex.

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)
Direct-Connect Supersonic Combustion Test Facility (DCSCTF)	Combustor tests	$H_2/O_2/Air$ combustion (3800)	4.0 to 7.5	2.0 2.7	1.52 x 3.46 1.50 x 6.69	———
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	2.5W x 3.5 H x 8 L
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
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 (25 dia chamber)
Hypersonic Pulse facility (HYPULSE)	Combustor tests	Shock-Expansion Tube/Tunnel (15 550)	12.0 14.0 15.0 17.0	4.7 4.8 5.0 7.2	6.12 diameter	4 dia x 34.5L

*Based on stagnation enthalpy and altitude simulation

Table 2.- Standard Test Conditions of the NASA-Langley Scramjet Test Complex.

Facility	M_0	$p_{t,plen}$ (atm)	$h_{t,plen}$ (Btu/lbm)	$T_{t,plen}$ (°R)	$\dot{m}_{tg}$ (lbm/s)	M_{tg}	p_{tg} (atm)	T_{tg} (°R)	Test gas mole fractions*								
									N_2	O_2	Ar	H_2O	CO_2	NO	NO_2	O	OH
DCSCTF	4.0	7.8	430	1640	4.34	2.0	.990	959	.6987	.2095	.0083	.0832	.0003	———	———	———	———
	7.5	26.5	1290	3780	7.21	2.7	1.021	1905	.4248	.2074	.0051	.3594	.0002	.0039	———	———	.0002
CHSTF	4.0	6.3	430	1640	29.1	3.4	.090	523	.6970	.2095	.0083	.0849	.0003	———	———	———	———
	5.5	19.7	750	2563	19.1	4.5	.055	574	.6036	.2094	.0072	.1793	.0002	.0003	———	———	———
AHSTF	4.7	13.6	514	2021	11.0	4.8	.033	381	.7756	.2034	.0093	———	———	.0111	.0006	———	———
	5.0	13.1	572	2251	10.0	4.8	.031	424	.7749	.2028	.0093	———	———	.0125	.0005	———	———
	5.5	11.2	692	2642	7.9	4.8	.027	513	.7735	.2014	.0093	———	———	.0154	.0004	———	———
	6.0	35.1	792	2979	6.41	6.1	.018	391	.7699	.1978	.0093	———	———	.0224	.0006	———	———
	7.0	28.6	1108	3989	4.51	6.0	.014	550	.7704	.1985	.0093	———	———	.0216	.0002	———	———
	8.0	23.0	1531	5183	3.16	6.0	.012	768	.7659	.1940	.0093	———	———	.0306	.0002	———	———
8'HTT	4.0	10.9	418	1640	1333	4.0	.067	412	.7081	.2095	.0085	.0491	.0248	———	———	———	———
	5.0	19.4	680	2350	765	5.0	.030	444	.6520	.2094	.0078	.0870	.0438	———	———	———	———
	6.8	136	1153	3560	416	6.7	.019	449 †	.5098 .5127	.2080 .2079	.0061 .0064	.1820 .1820	.0912 .0910	.0029 ———	——— ———	——— ———	——— ———
HYPULSE	12.0	6807	3105	9135	26.4	4.7	.722	2575	.7705	.1988	.0094	———	———	.0200	———	.0015	———
	14.0	551.7	3767	10 270	15.0	4.8	.471	3040	.7608	.1875	.0093	———	———	.0336	———	.0088	———
	15.0	2000	4225	11 630	38.8	5.0	1.174	3130	.7564	.1829	.0093	———	———	.0416	———	.0098	———
	17.0	13700	5805	15 550	24.9	7.2	.446	2275	.7540	.1804	.0093	———	———	.0461	———	.0102	———

* Calculated unless otherwise noted † Measured values of composition in this row

Table 3.- Airframe-Integrated Scramjet Tests in NASA-Langley Scramjet Engine Test Facilities.

Program	Engine	CHSTF	AHSTF	8' HTT	Tests	Total tests	Time period
NASA	Three-strut	178	90	0	268	963	Pre-NASP (1976 - 1987)
	Parametric	238	212	0	450		
	Step-strut	245	0	0	245		
NASP	Government baseline	114	0	0	114	1217	NASP Pre-Team (1987 - 1990)
	Rocketdyne A	0	69	0	69		
	Rocketdyne A1	0	55	0	55		
	Rocketdyne A2	177	138	0	315		
	Rocketdyne A2r	0	72	0	72		
	Pratt & Whitney C	31	202	0	233		
	JHU/APL B1	359	0	0	359		
	NASP SX-20	0	160	0	160	326	NASP-Team (1990 - 1994)
	NASP SXPE	0	142	0	142		
	NASP CDE	0	0	24	24		
NASA	Parametric	400	0	0	400	683	Post-NASP (1994 - 1998)
	Rocketdyne HC	132	0	0	132		
	SXPE	0	124	0	124		
	CDE	0	0	27	27		
		1874	1264	51	3189	3189	

Table 4.- HYPULSE Dimensions and Pressure Limits.

Component	Length (ft)	Diameter (ft)	Pressure limit (atm)
Driver	8.01	0.54	1360
Shock tube	39.76 (24.67)	0.50	340
Acceleration tube	32.87 (47.97)	0.50	340
Testsection/Dump tank	34.46	4.00	10

Table 5.- HYPULSE Scramjet Combustor Test Conditions.

Test conditions	Test gas	$h_{t,\infty}$ (BTU/lbm)	P_{tg} (atm)	V_{tg} (ft/s)	T_{tg} (°R)	M_{tg}
M 13.5	Air	4250	0.177	12630	4230	4.1
M 12SD*	Air	3235	0.772	11330	2575	4.7
M 14LP	Air	4000	0.147	13060	2475	5.5
M 14HP	Air N2	3915 3355	0.471 0.372	12435 11580	3040 2620	4.8 4.7
M 14SD	Air	4300	1.172	13270	3130	5.0
M 14NZ	Air	3745	0.046	12360	1495	7.0
M 15NZ*	Air	4645	0.221	14450	1820	7.1
M 17NZ*	Air	5935	0.446	16355	2275	7.2

* Test condition demonstrated/calibrated, but no combustion test data

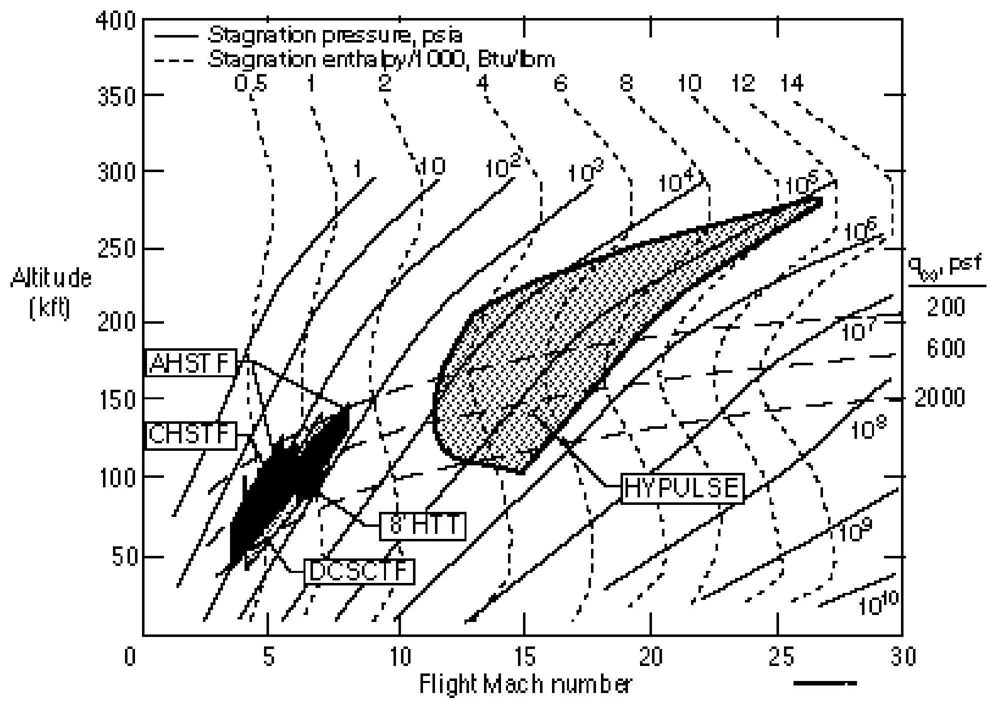

Figure 1.- Flight simulation map of NASA-Langley scramjet test facilities.

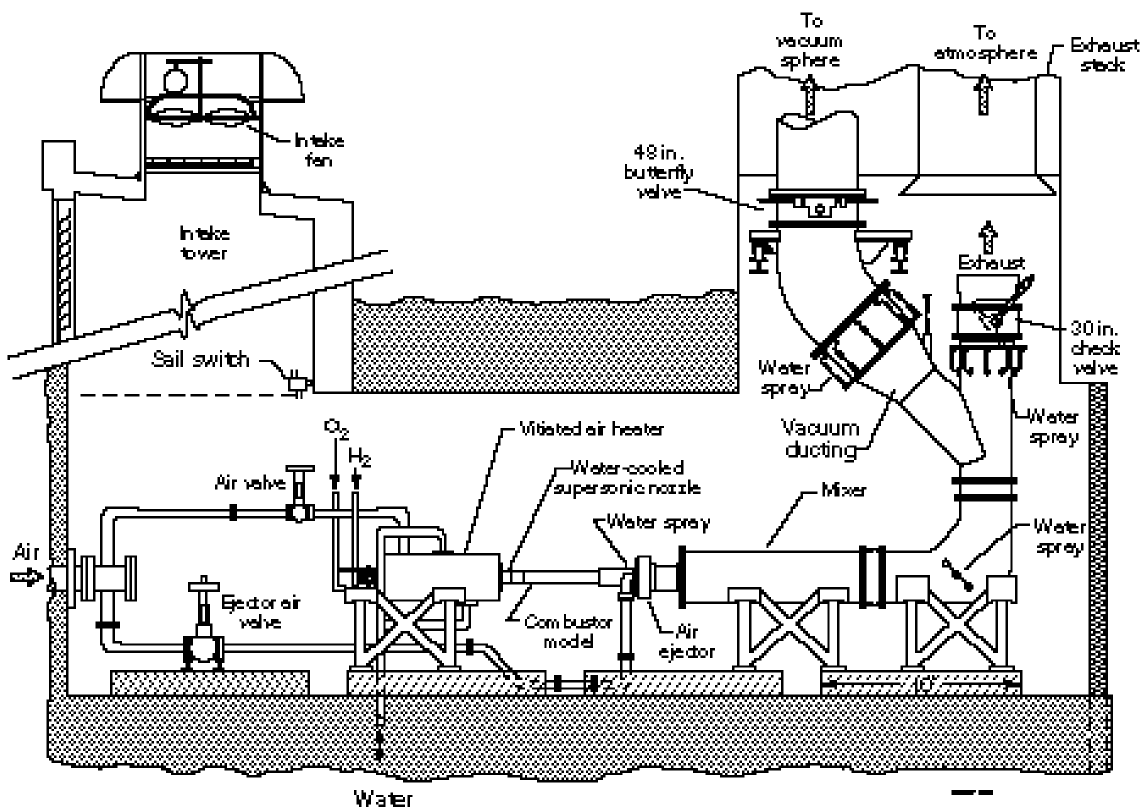

Figure 2.- Direct-Connect Supersonic Combustion Test Facility.

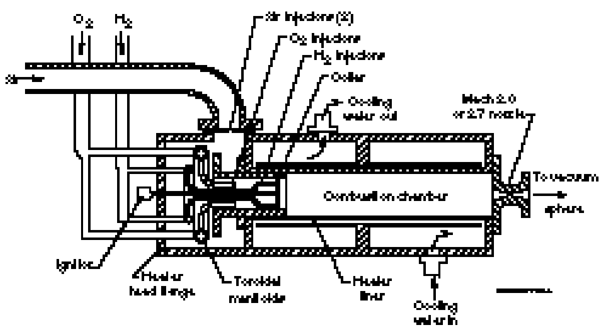

Figure 3.- DCSCTF H_2-O_2- Air combustion heater.

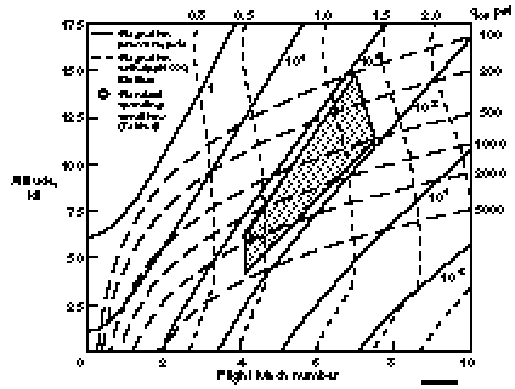

Figure 4.- DCSCTF flight simulation envelope.

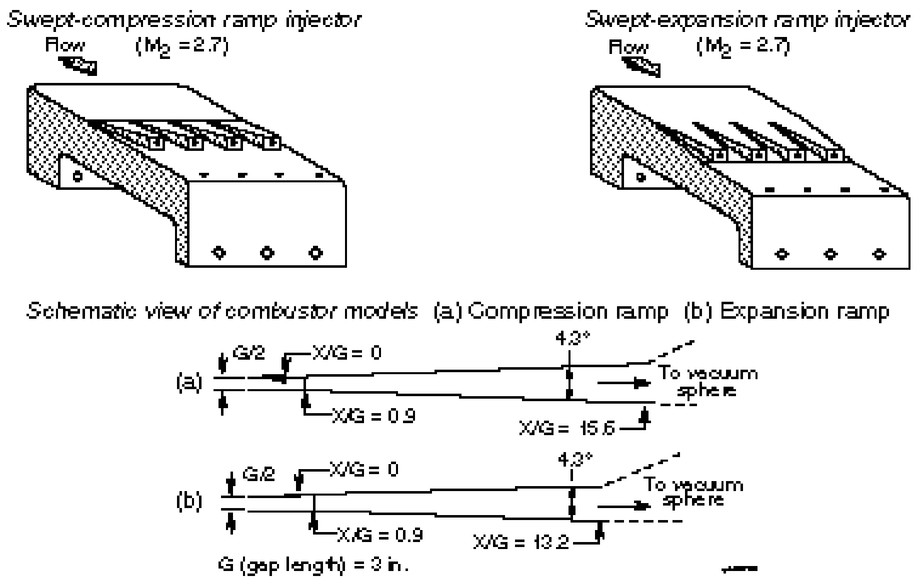

Figure 5.- Swept-ramp injector geometries tested in the DCSCTF.

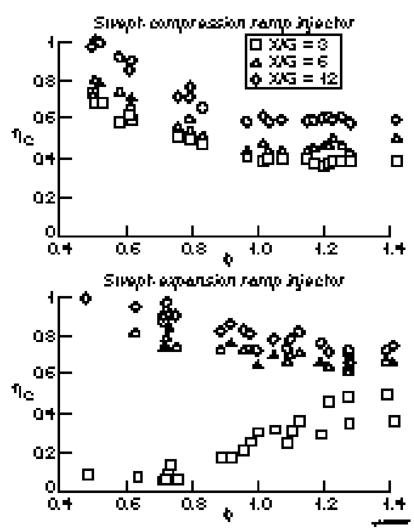

Figure 6.- Combustion efficiency of swept-ramp injectors.

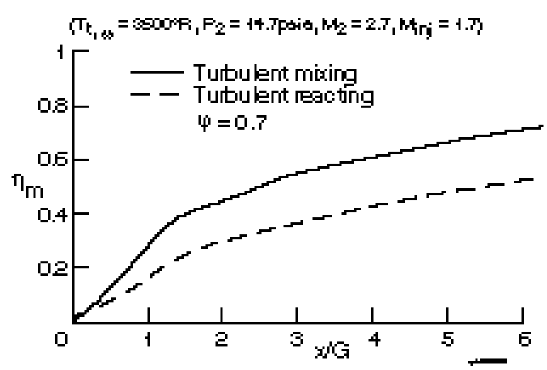

Figure 7.- Effect of reaction on mixing for swept-compression ramp injector.

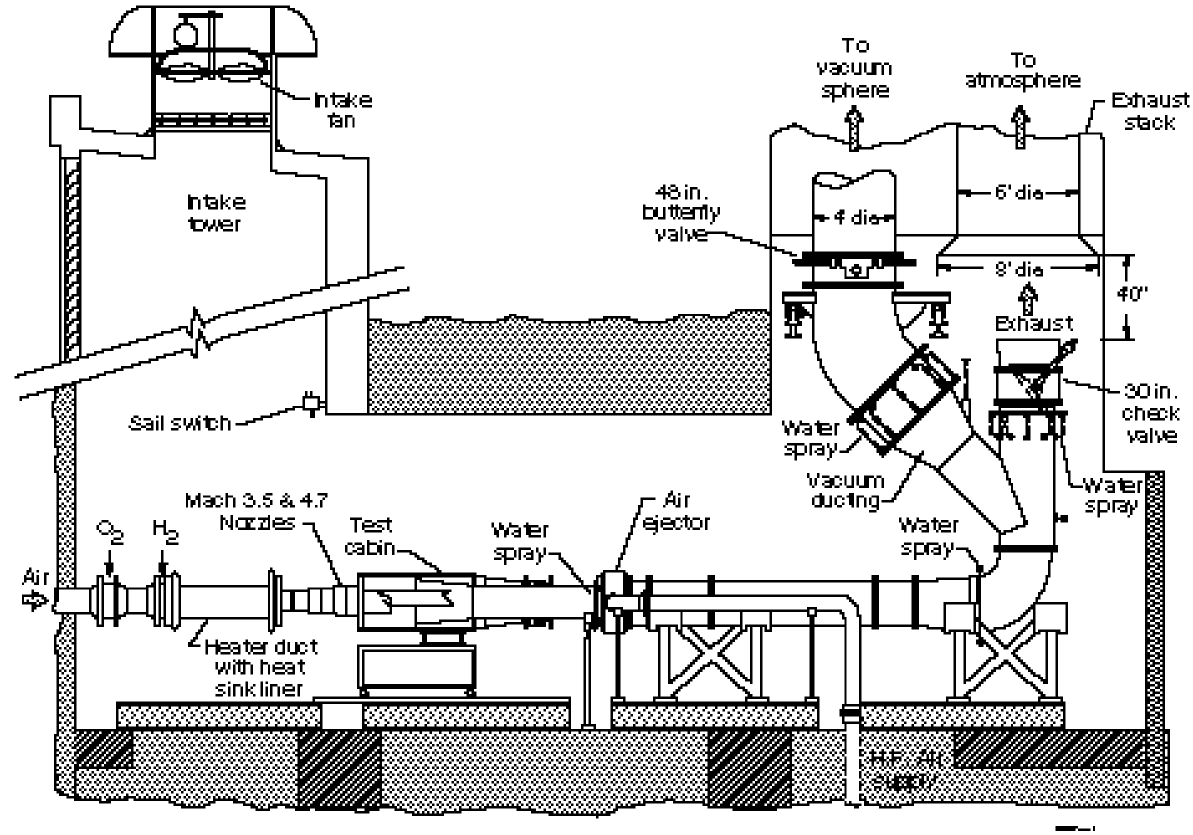

Figure 8.- Combustion-Heated Scramjet Test Facility.

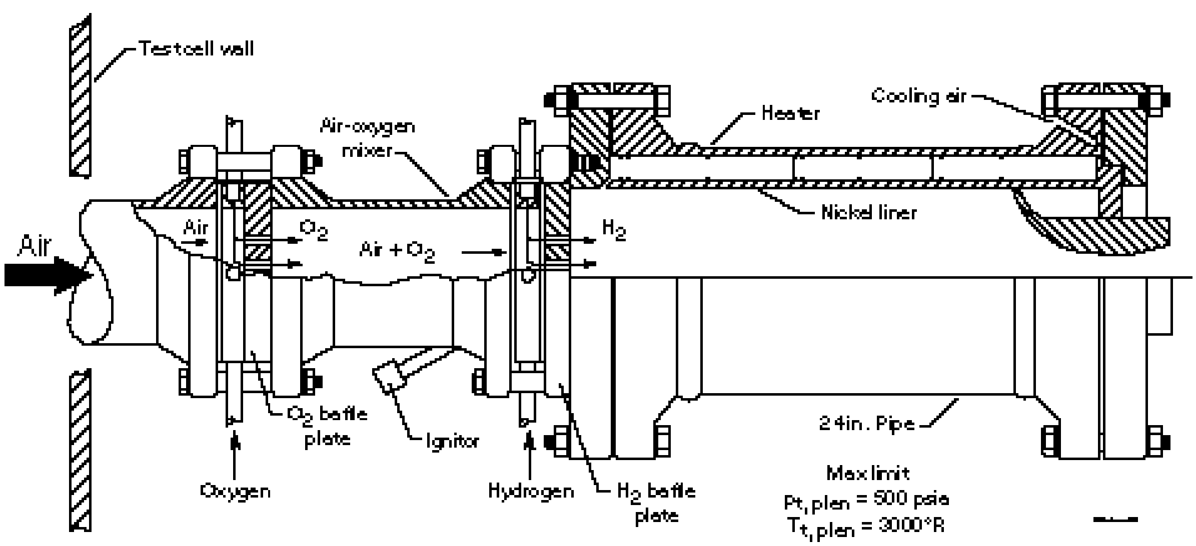

Figure 9.- CHSTF H_2-O_2-Air combustion heater.

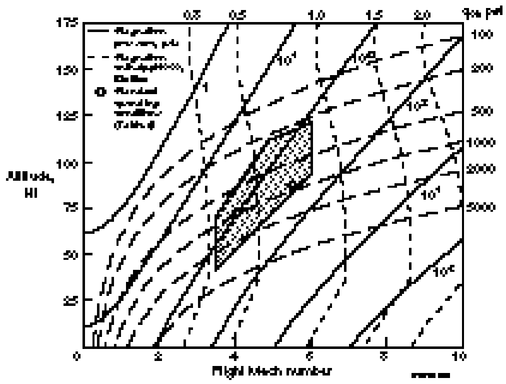

Figure 10.- CHSTF flight simulation envelope.

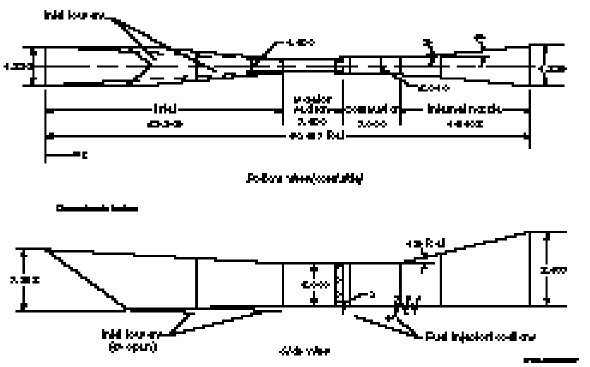

Figure 11.- Hydrocarbon-fueled scramjet model tested in the CHSTF.

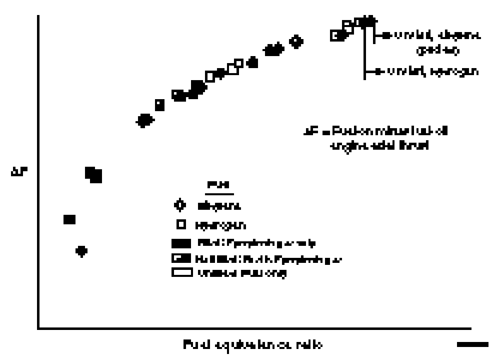

Figure 12.- Maximum performance with hydrogen and ethylene fuels in the CHSTF tests.

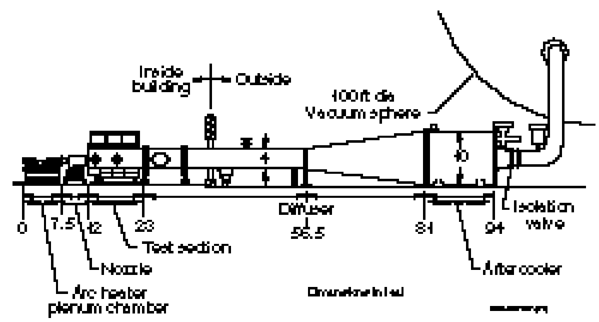

Figure 13.- Arc-Heated Scramjet Test Facility.

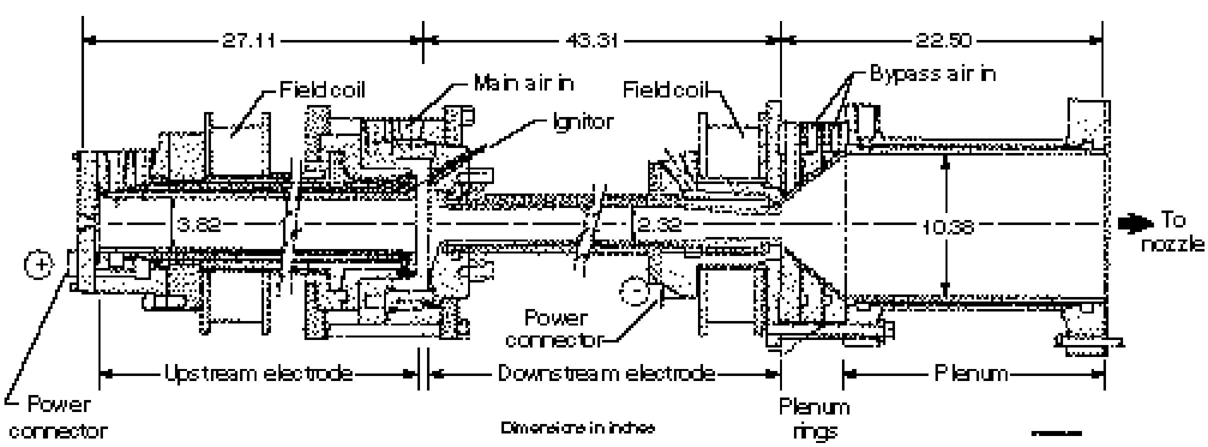

Figure 14.- AHSTF heater and plenum chamber.

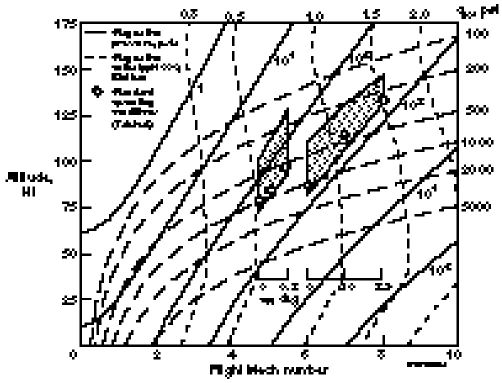

Figure 15.- AHSTF flight simulation envelope.

Figure 16.- SXPE model in the AHSTF.

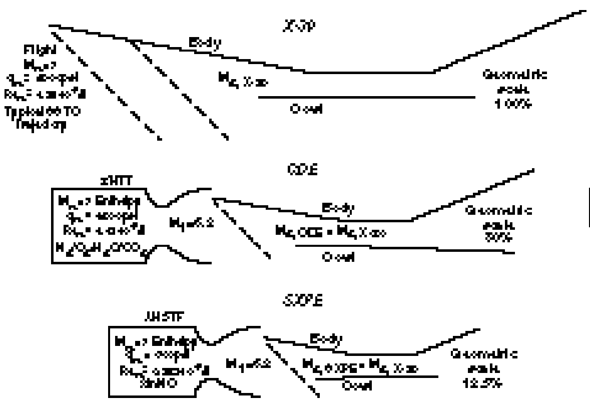

Figure 17.- Comparison of CDE and SXPE simulations with flight.

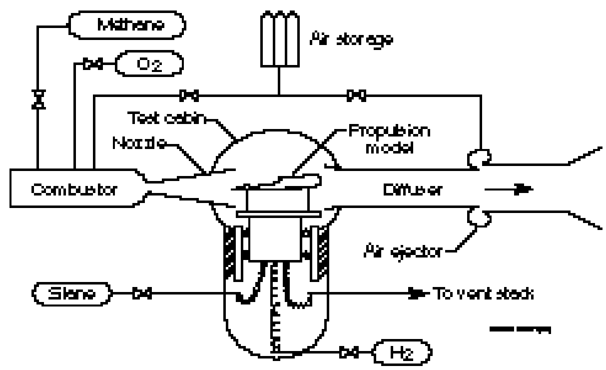

Figure 18.- Eight-foot High Temperature Tunnel.

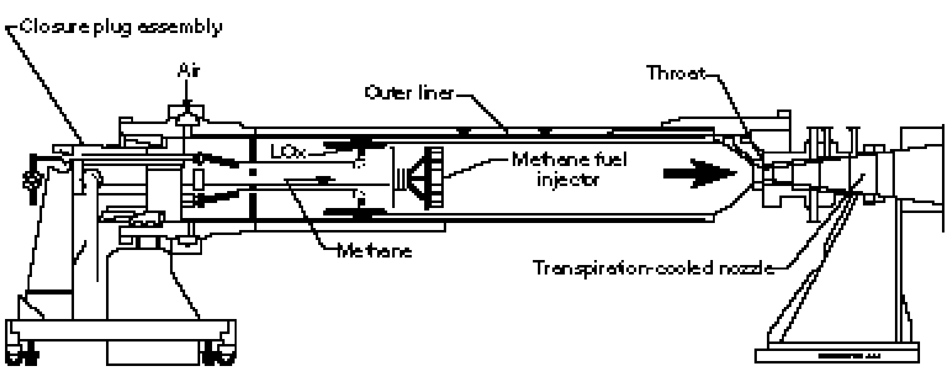

Figure 19.- 8-Foot HTT CH_4-Air-O_2 combustion heater.

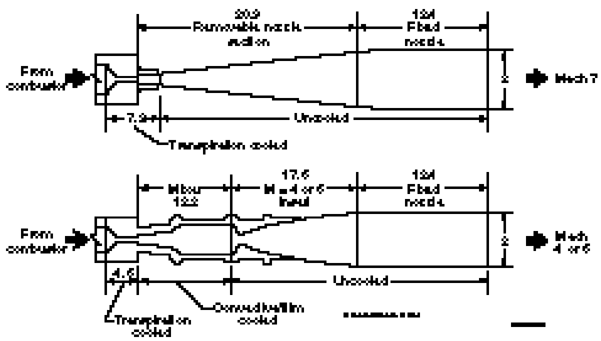

Figure 20.- Modified 8-Foot HTT nozzles for alternate Mach numbers capability.

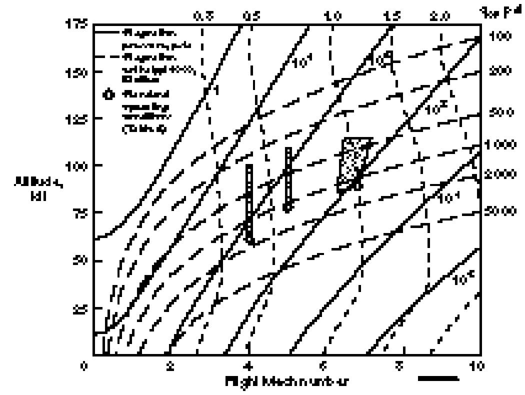

Figure 21.- 8-Foot HTT flight simulation envelope.

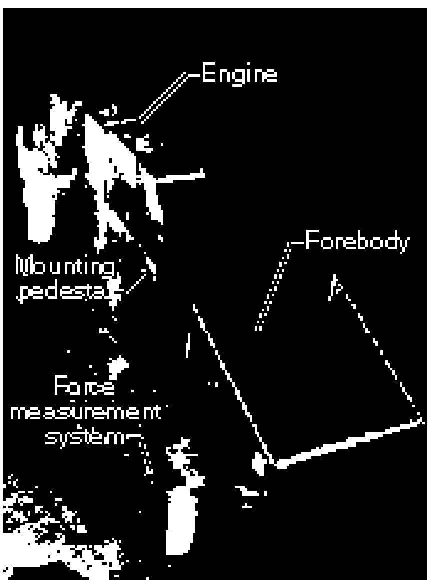

Figure 22.- Concept Demonstration Engine (CDE).

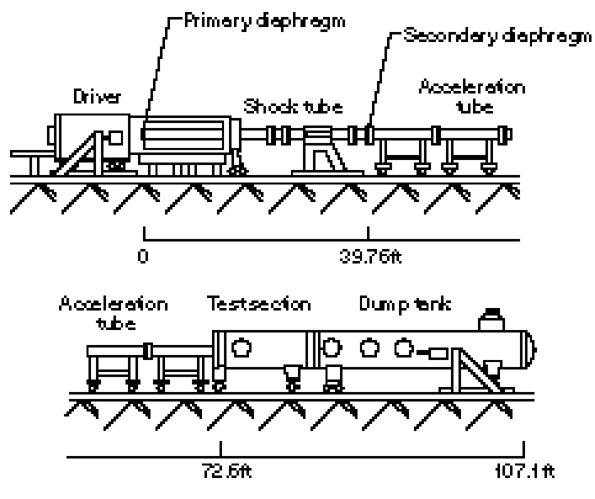

Figure 23.- Arrangement of NASA HYPULSE shock-expansion tube.

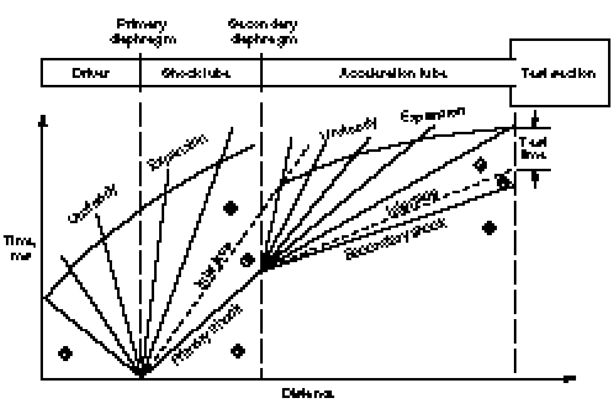

Figure 24.- Wave diagram illustrating HYPULSE operation.

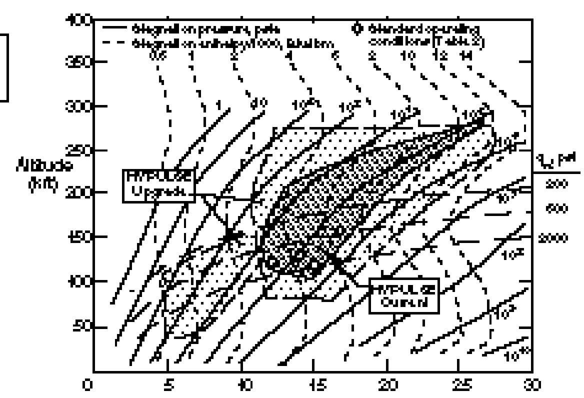

Figure 25.- HYPULSE flight simulation envelope.

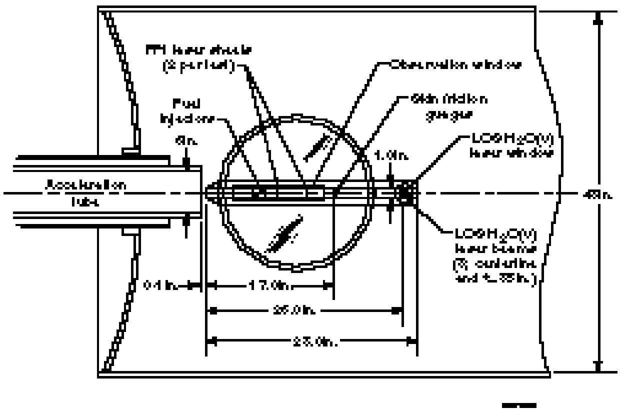

Figure 26.- Rectangular combustor model in HYPULSE test chamber.

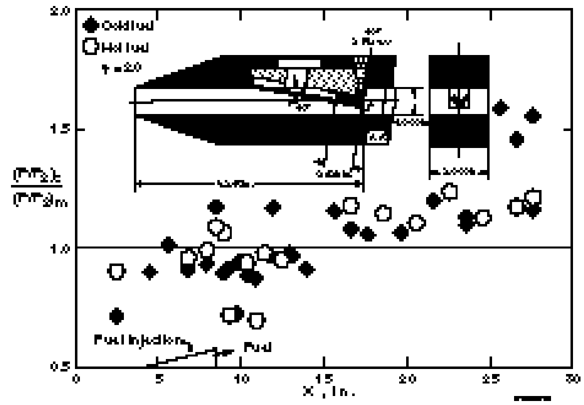

Figure 27.- Comparison of normalized combustor pressures for hot and cold fuel; HYPULSE tests at Mach 14.

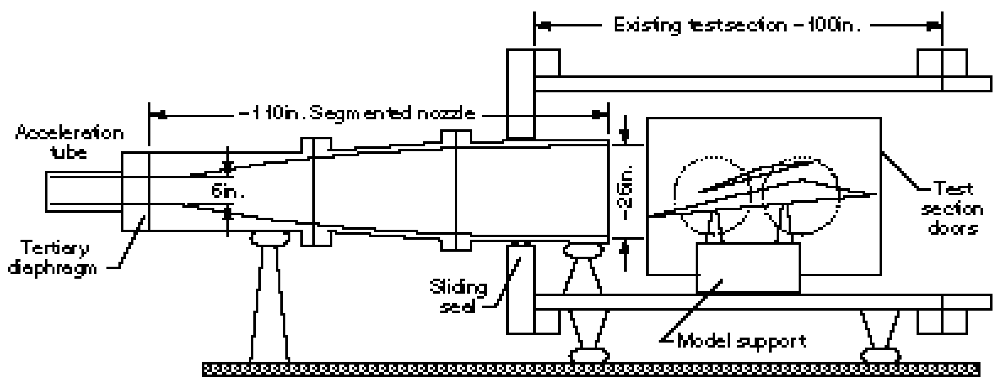

Figure 28.- Conceptual arrangement of HYPULSE as shock-expansion tunnel.

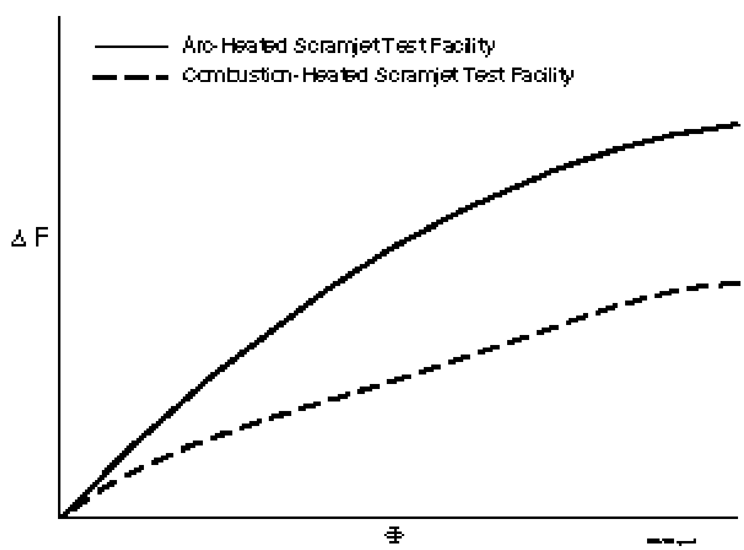

Figure 29.- Thrust performance of parametric scramjet in the CHSTF and the AHSTF.