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Abstract

Tests of the Hyper-X scramjet engine flowpath have been conducted in the HYPULSE shock tunnel at conditions duplicating the stagnation enthalpy at flight Mach 7, 10, and 15. For the tests at Mach 7 and 10 HYPULSE was operated as a reflected-shock tunnel; at the Mach 15 condition, HYPULSE was operated as a shock-expansion tunnel. The test conditions matched the stagnation enthalpy of a scramjet engine on an aerospace vehicle accelerating through the atmosphere along a 1000 psf dynamic pressure trajectory. Test parameter variation included fuel equivalence ratios from lean (0.8) to rich (1.5+); fuel composition from pure hydrogen to mixtures of 2% and 5% silane in hydrogen by volume; and inflow pressure and Mach number made by changing the scramjet model mounting angle in the HYPULSE test chamber. Data sources were wall pressures and heat flux distributions and schlieren and fuel plume imaging in the combustor/nozzle sections. Data are presented for calibration of the facility nozzles and the scramjet engine model. Comparisons of pressure distributions and flowpath streamtube performance estimates are made for the three Mach numbers tested.

Nomenclature

A	Area
D	Diameter
H	Enthalpy
M	Mach number
P	Pressure
q	Dynamic pressure
R	Radial coordinate
S_{norm}	Coordinate normal to forebody
T	Temperature
X	Axial coordinate
ϕ	Equivalence ratio

Subscripts:

1	Facility test gas
5	Shock, acceleration tube end
In	Inflow condition
max	Maximum value
N	Nozzle exit
pit	pitot pressure
pln	plenum pressure
t	Stagnation condition
inf, ∞	Flight condition

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Acronyms and Abbreviations:

4FSS	Four-frame sequential schlieren
AR	Nozzle area ratio
CFD	Computational Fluid Dynamics
DAS	Data Acquisition System
FB	Forebody
FPI	Fuel Plume Imaging
HSM	HYPULSE Scramjet Model
HYPULSE	HYpersonic PULSE facility
LaRC	Langley Research Center
LHI	Laser Holographic Interferometry
NASP	National Aero-Space Plane
PIW	Path-Integrated Water vapor device
RST	Reflected-Shock Tunnel
SET	Shock-Expansion Tunnel
SFM	Silane-hydrogen Fuel Mixture

Introduction

The development of a scramjet engine flowpath concept for operation across the Mach 6 to 15+ flight regime has been pursued in numerous programs for more than 25 years.[1-3] A large database of scramjet components and engine models has been established in research programs by NASA Langley,[4,5] Johns Hopkins University, Applied Physics Laboratory,[6,7] GASL, Inc.[8] and others, at test conditions duplicating flight enthalpies below Mach 8. During the National Aero Space Plane (NASP) program, a rejuvenation of hypersonic test facilities capable of duplicating flight speeds from Mach 7 to 18 enabled some testing of scramjet components and engine flow paths. These tests, although limited in number and scope, did provide the impetus to move forward with enhancing the airbreathing propulsion research capabilities within this Mach number range. One test facility that came out of the NASP effort for achieving hypervelocity test conditions is the NASA HYPULSE shock expansion tube.

The NASA HYPULSE shock tunnel,[9-11] located at and operated by GASL, Inc., has been upgraded from the expansion tube configuration used in support of NASP to include a detonation driven reflected-shock tunnel (RST) operating mode that is capable of delivering a test flow at conditions duplicating Mach 7 to 10 flight speeds. The upgrade also includes a test chamber that allows free-jet testing of scramjet engine at the same scale as used in other blow-down test facilities at NASA Langley. In addition, the facility can be converted into a shock-expansion tunnel (SET) that can deliver test conditions suitable for scramjet tests up to Mach 16 flight speed duplication.[11]

The results from a series of ground tests conducted in the HYPULSE shock tunnel in support of the Hyper-X Mach 10 flowpath development have begun to bridge the gap in scramjet flowpath data from the Mach 7 limit of combustion- or electric arc-heated test facilities, to the hypervelocity conditions achievable in shock-heated pulse tunnels. Hyper-X is a program to flight test an airframe-integrated scramjet engine at Mach 7 and 10.[12] These Hyper-X tests included operation at conditions that duplicated the stagnation enthalpy at flight Mach 7, 10, and 15 conditions. The focus of the Mach 7 tests was to extend and connect the existing database for scramjets to pulse facilities and thereby anchor future hypervelocity test data in HYPULSE to the well-established Mach 7 database from conventional (combustion- or arc-heated) blow-down scramjet test facilities. Testing at conditions duplicating Mach 10 flight enthalpy used essentially the same engine hardware as the Mach 7 tests, with some internal flowpath changes dictated by scramjet design methodology. The Mach 10 test series provided an initial data set at a test condition that has twice the stagnation enthalpy as Mach 7 and the extension in continuity of scramjet design methods into the hypervelocity regime. Tests at a condition duplicating nominal Mach 15 flight energy, which is an additional factor of two increase above Mach 10, were conducted with HYPULSE configured as a shock-expansion tunnel (SET). The scramjet engine model used in these tests was the same configuration as in the Mach 10 series and provided a preliminary set of data for a scramjet flowpath that was essentially the same across the major portion of the expected operational flight regime. In addition, the Mach 15 tests provided the first definition of HYPULSE SET test technique for scramjet testing. This paper reviews the test technique in the dual-mode HYPULSE shock tunnel, and provides comparative results of the scramjet flowpath operation over the flight Mach number range.

Test Facility

HYPULSE Shock Tunnel

Description: The NASA HYPULSE shock tunnel facility, shown in Figure 1, is a dual-mode pulse facility that can be configured as a reflected-shock tunnel (RST) or a shock-expansion tube/tunnel (SET), depending on the desired simulation. The shock tunnel is located at and operated by GASL, Inc. In the RST mode stagnation enthalpies in the test gas corresponding to flight speeds from Mach 5 to 10+ can be duplicated. In the SET mode stagnation enthalpies in the test gas corresponding to flight speeds from Mach 12 to 25 can be duplicated. The major components of the facility are: (1) the driver, which can be either cold helium at high-pressure (up to 83 MPa or 12,000 psi), or (1a), a stoichiometric hydrogen-oxygen mixture diluted with a noble gas (helium or argon) detonation wave; (2) a shock tube section that is about 22m (72 ft) long and 15 cm (6-in.) diameter; (3) the test chamber 6.1m (20-ft) long by 2.13m (7-ft) diameter; and (4) a dump tank 9.14 m (30-ft) long and 1.27m (50-in) diameter. An upstream portion of the shock tube section is used as the detonation driver (1a), for test conditions above Mach 7. A downstream section of the shock tube (2a) is used as the acceleration tube in the SET mode.

Operation: The shock wave is generated by the sudden rupture of a double diaphragm separating the high-pressure cold helium driver from the low-pressure gas in the shock tube that will become the test gas. In detonation drive mode, this driven shock passes into the detonable mixture that initiates a stronger shock to heat and compress the test gas in the shock tube. The test chamber contains the facility nozzle and the scramjet test hardware. In RST mode, the reflection of the incident shock at the shock tube-nozzle interface produces the nozzle plenum conditions at near stagnation. In the SET mode, the test gas flow is processed by the passage of the incident shock and then accelerated by an unsteady expansion fan to a steady flow at the tube exit, achieving very high enthalpy through the addition of kinetic energy, without being stagnated. The operational envelopes of the HYPULSE shock tunnel simulation for both RST and SET modes are presented in Figure 2. The lines of constant dynamic pressure cover the generally accepted airbreathing flight corridor to near orbit.

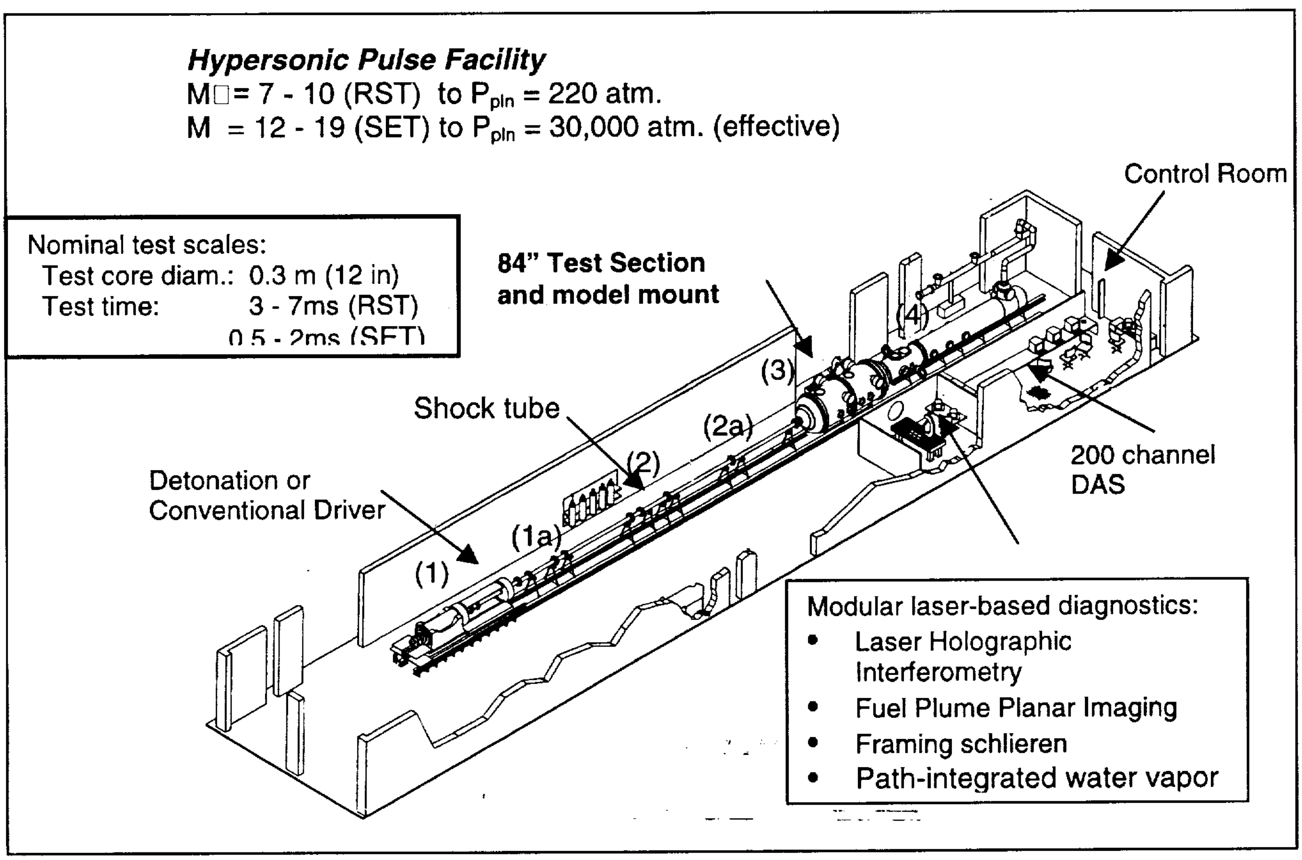


Figure 1.- Schematic of the NASA HYPULSE Shock Tunnel at GASL, Inc.

To date, test points for scramjet tests have been established that duplicate the stagnation enthalpy at flight Mach 7 and 10 in RST mode and at Mach 15 in SET mode. A test point for studying aero-heating in planetary entry at Mach 21 also has been defined in SET mode. Additional details and descriptions of HYPULSE operation are in the literature.[9-11]

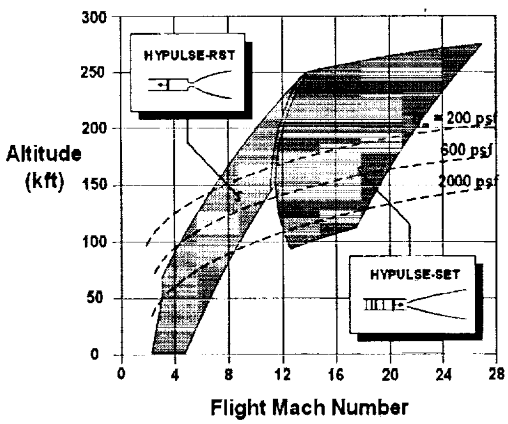


Figure 2.- HYPULSE operational envelope.

Test Conditions: The primary nozzle for RST operation is an axisymmetric contour with an area ratio of 175 (AR-175 nozzle designation), with a throat diameter of 5.1 cm (2-in) and an exit diameter of 66.7 cm (26.25-in). This nozzle was designed to deliver an exit core flow with a local Mach 6.5, (typical of the forebody flow on an aerospace vehicle at Mach 10) at the nominal flight Mach 10 condition. When operated at the flight Mach 7 condition, the nozzle delivers an exit flow at Mach 7.3 in the slightly over-expanded core flow. Based on calibration test results from earlier Mach 10 tests, a smaller throat piece was made with a diameter of 4.45 cm, (1.75-in) for the Mach 10 tests to obtain a nozzle with a geometric area ratio of 225 (AR-225). This nozzle produced a core flow with an exit Mach of 6.9 at the Mach 10 condition. The tests duplicating flight Mach 15 conditions in SET mode were made with the nozzle designed by GASL for the flight Mach 21 aero-heating tests. This SET nozzle captured the entire flow at the exit of the 15-cm (6-in.) diameter acceleration tube and expanded the flow through an area ratio of 16 (AR-16), delivering a nominal Mach 12 flow at the nozzle exit. This nozzle has a hyperbolic contour that approaches a 10-deg half-angle cone at the exit. Nominal test conditions of the HYPULSE facility to simulate scramjet tests at flight Mach 7, 10, and 15 are given in Table 1.

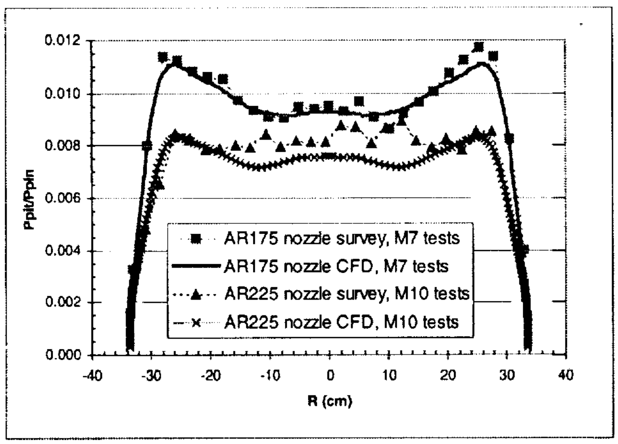
Table 1. Nominal Mach 7, 10, and 15 test conditions in HYPULSE.

	Plenum (stagnation)			HYPULSE Nozzle exit core					
Flight Mach (nominal)	H_{pln} (Btu/lb) [MJ/kg]	P_{pln} (psia) [MPa]	T_{pln} (R) [K]	Test mode	Nozzle AR (geom.)	M	P (psia) [kPa]	T (R) [K]	V (fps) [m/s]
7	1056 2.456	1440 9.932	3852 2140	RST	175	7.32	0.101 0.696	374 208	6940 2116
10	2093 4.87	3371 23.26	6932 3851	RST	225	6.91	0.416 2.872	846 470	9836 2998
15	4958 11.5	253,900 1751*	13625 7570*	SET	16	13.5	0.095 0.655	567 315	15745 4800

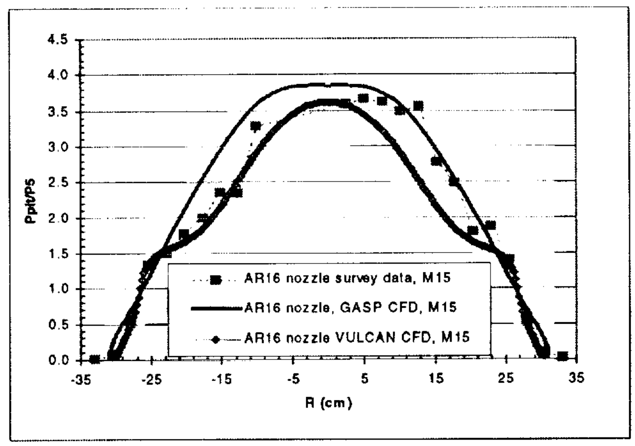
* Effective stagnation state for isentropic process in chemical equilibrium.

Calibrations

The nozzle exit values in Table 1 were determined from pitot pressure surveys and CFD analyses of the nozzles at selected benchmark test conditions. The pitot pressure profiles for a vertical survey and CFD results at 2.54 cm from the axisymmetric nozzle exit are presented in Figure 3. The data were obtained with a 30-probe pitot rake with probes spaced 2.54 cm (1-in) apart. For the Mach 7 and 10 test conditions shown in Figure 3(a), the pitot pressure data are normalized by the measured nozzle plenum pressure P_{pln}. The CFD results were computed with the GASP™ code,[13] starting at the nozzle plenum. The plenum state was determined by computing the reflected shock process knowing the initial pressures, the measured shock speed, and assuming the test gas to be in chemical equilibrium. The expansion from the plenum through the nozzle used finite rate air chemistry. Two CFD solutions are given for the Mach 15 test condition in Figure 3(b). One was made with the GASP code and the other with the VULCAN code.[14] Both codes started at the end of the 15-cm diameter shock-acceleration tube with profiles of the primitive variables derived from measured pitot-pressure profiles, the wall static pressure, and mean gas velocity based on the measured shock speed. Both of the CFD results compare well with the single pitot survey data that have been normalized by the static pressure measured at the shock tube – nozzle interface.



(a) Mach 7 and 10; RST mode.



(b) Mach 15; SET mode

Figure 3.- HYPULSE nozzle exit surveys

Data Systems

The data acquisition system (DAS) at HYPULSE includes a cluster of digital oscilloscopes to acquire data with piezoelectric quartz-crystal pressure transducers and thin-film thermocouple heat-flux gages. The pressure sensors have 500 kHz frequency response and sensitivities ranging from 1 to 100 mV/psi, with a measurement uncertainty of less than 5% of full scale. Heat flux gages are platinum thin-film resistance thermometers painted on machinable ceramic substrates and are manufactured and calibrated at GASL. The run times of HYPULSE are short, on the order of a few milliseconds of established flow, which enables optical access through uncooled windows. The integrated optical system[11,15,16] at HYPULSE includes a high-speed four-frame sequential schlieren system (4FSS), a laser holographic interferometer (LHI), a laser-based fuel plume planar imaging (FPI) system, and a path-integrated water vapor (PIW) measurement.

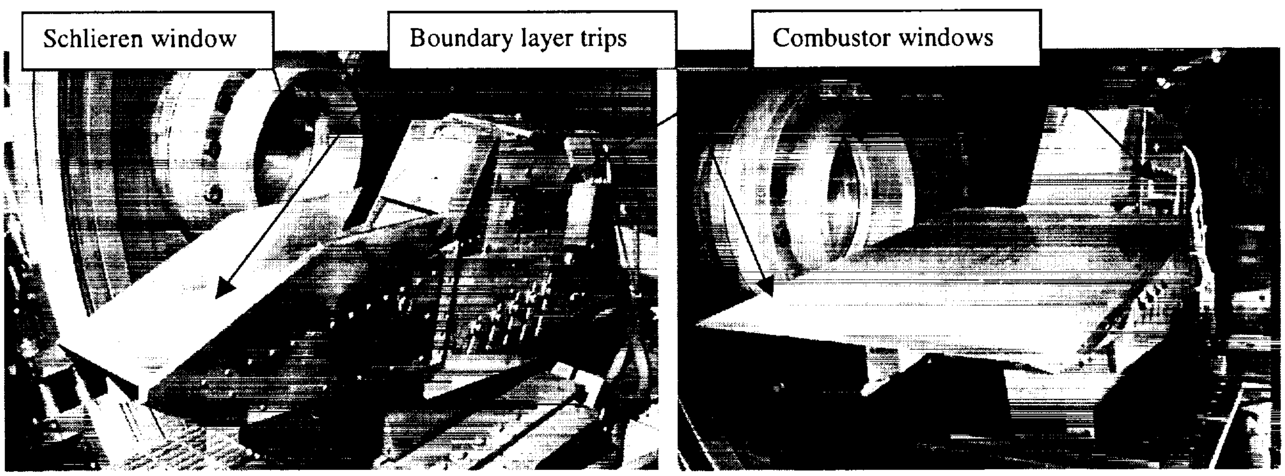
Scramjet Tests

Hardware

Figure 4 shows photographs of the HSM hardware mounted cowl-side up in the HYPULSE test chamber. The internal width is about 16.8 cm (6.6-in.), with an overall length of about 200 cm (80-in). For the Mach 7 tests, the HSM (shown in the left photo) had a ramped forebody with side fences to enable capture of a 2-D streamtube, and to be similar to concepts tested in blow-down test facilities in the LaRC Scramjet Test Complex.[5] For the Mach 10 and 15 tests, the forebody was changed to a flat unfenced ramp (right photo) to give engine inlet conditions to enable better simulation of forebody conditions on a generic hypersonic vehicle. Boundary layer trips were placed on the forebody, as indicated in the photos, to prompt boundary layer transition on the forebody. The trips were designed and positioned on the forebody based on the particular test flow properties at the facility nozzle exit.[17] The engine isolator/combustor section had windows to permit optical access for schlieren and fuel-plume planar images that were acquired in some tests. The same HSM hardware was used in all tests with minor changes to the internal geometry between Mach 7 and 10. Mach 15 tests were made with the Mach 10 hardware. Instrumentation on the model consisted of piezoelectric transducers and thin-film thermocouples for heat flux data on the body side and cowl side surfaces.

Flight Simulation

As illustrated in the sketches of Figure 5, the HSM represents a portion of the airbreathing propulsion flowpath of a conceptual space access vehicle. The test model is shown mounted cowl-side up and positioned relative to the nozzle exit. The forebody and aftbody are truncated to achieve a model size compatible with the test facility limitations. The truncation is indicated by the dashed lines from the conceptual vehicle to the model sketches. Generally, the ground test simulation of flight condition is achieved by replicating the stagnation enthalpy, Mach number, and static pressure at the cowl plane. The HYPULSE facility is operated at the flight stagnation enthalpy; however, flight simulation is not exactly matched in Mach and pressure because of facility limitations. One of these parameters can be matched by mounting the HSM at some angle to the facility flow, as indicated in the figure. Values of freestream and engine inflow conditions for a typical aerospace vehicle along a 1000 psf dynamic pressure trajectory are given in Table 2 for Mach 7, 10, and 15.



HSEM for Mach 7 tests; two-ramp, fenced FB.

HSEM for Mach 10 & 15 tests; flat, unfenced FB.

Figure 4. - Photos of HSE model for Mach 7, 10, and 15 in HYPULSE test chamber.

Table 2. Typical flight vehicle conditions at Mach 7, 10, and 15

Flight at Q=1000psf				Typical vehicle engine inflow*		
Flight Mach	H_∞ [MJ/kg]	P_∞ [kPa]	T_∞ [K]	M_{in}	P_{in} [kPa]	T_{in} [K]
7	2.48	1.40	225	5.05	9.32	400
10	4.92	0.68	233	6.55	8.59	515
15	11.52	0.30	249	8.46	9.02	770

*Assumed generic hypersonic vehicle flying Q = 1000 psf trajectory.

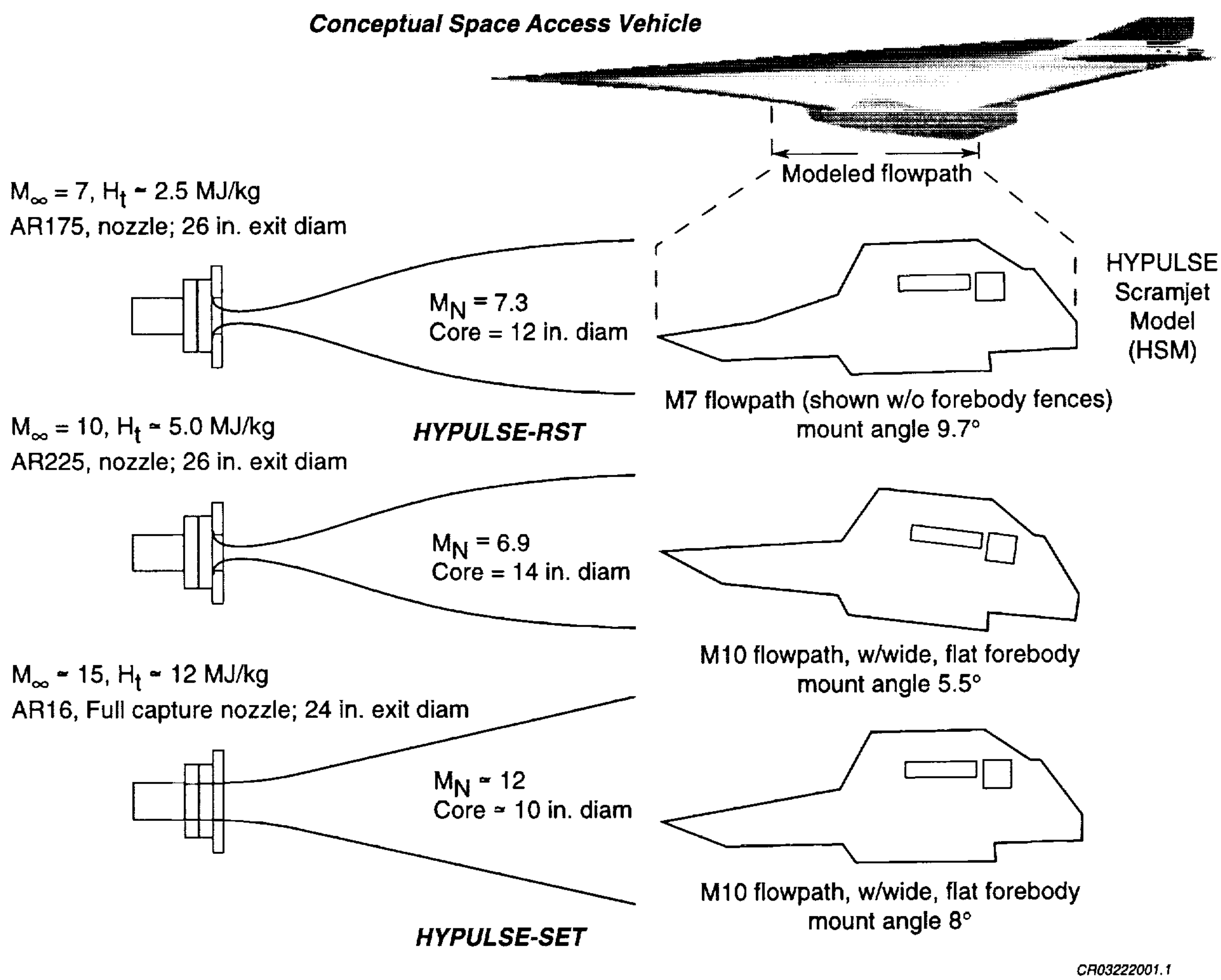


Figure 5. Test conduct of the HYPULSE Scramjet Engine Model at Mach 7, 10, and 15

Test Conduct

HSM Calibration: The most essential quantity for testing airbreathing propulsion flowpaths is the engine mass capture. For these HSM tests a hybrid procedure was used to obtain engine mass capture values, in which CFD of the HSM forebody flow was used with forebody experimental pressure distribution data and pitot pressure surveys normal to the forebody surface near the cowl. The forebody CFD calculations were started from the nozzle exit CFD solution using either a 2-D profile or mass averaged core values. The pitot pressure survey rake is indicated in the forebody sketch and a schlieren image in Figure 6 for the Mach 7 tests. The pitot rake has 23 probes spaced 0.50 cm (0.20 in.) apart. Each probe is instrumented with a piezoelectric transducer mounted in the rake body to minimize response time. The shocks from the model leading-edge and forebody ramp are observed in the schlieren. Pitot data at multiple lateral locations were obtained, depending on the test series. The pitot rake also was used in the Mach 10 and 15 test entries to map the engine inlet flow.

Comparisons of the HSM forebody pitot pressure data and CFD predictions are presented in Figure 7. The pitot data profiles for the Mach 7 and 10 test series, with HYPULSE in RST mode, in Figure 7(a), are an average of all the lateral surveys with each set normalized by the facility nozzle plenum pressure. Standard deviations of the data are indicated for the run-to-run variation with the rake at different lateral positions. The data indicate large variation in the vicinity of the forebody shocks, particularly for the Mach 7 series. In addition, there is large variation at the outer edge as a result of interference caused by the forebody fences used in the Mach 7 test series. The Mach 10 series exhibit a much more uniform profile out to where the forebody leading edge shock is crossed. The CFD profiles were obtained with the GASP code as indicated and compare well for the Mach 10 case. The Mach 7 CFD results are significantly different from the data; yet further examination of the computations and data processing revealed no clear cause. However, the estimated mass capture from the CFD and a value obtained by integration of the pitot data profiles agreed to within a few percent. CFD results of the HSM forebody flow at the Mach 15 test condition are not yet complete for comparison with the calibration pitot data. The shape of the pitot data profile is attributed to the conical flow from the facility nozzle.

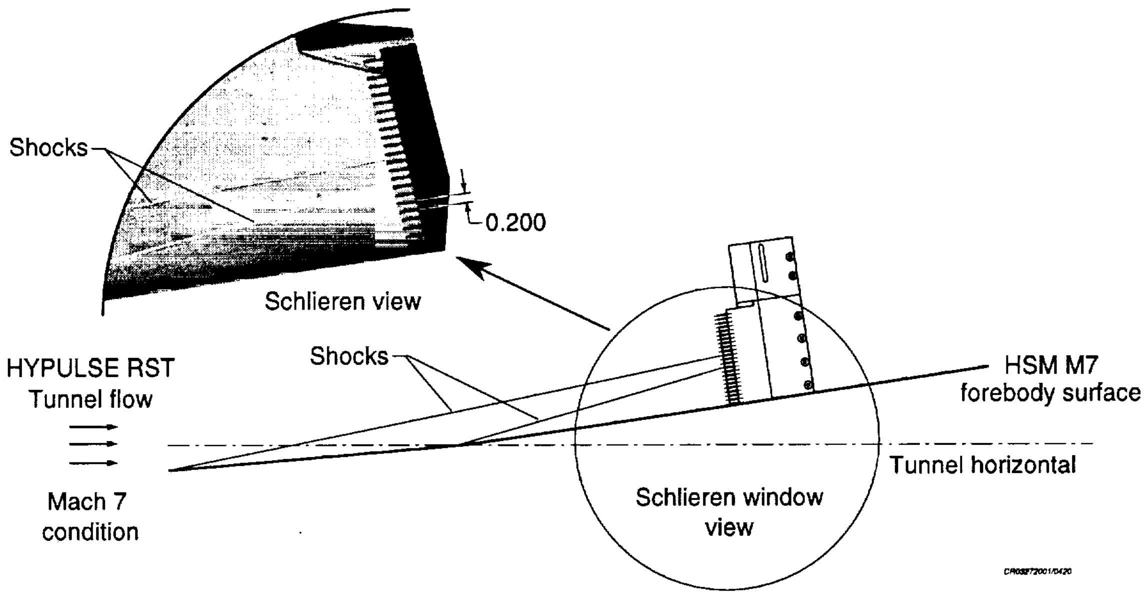
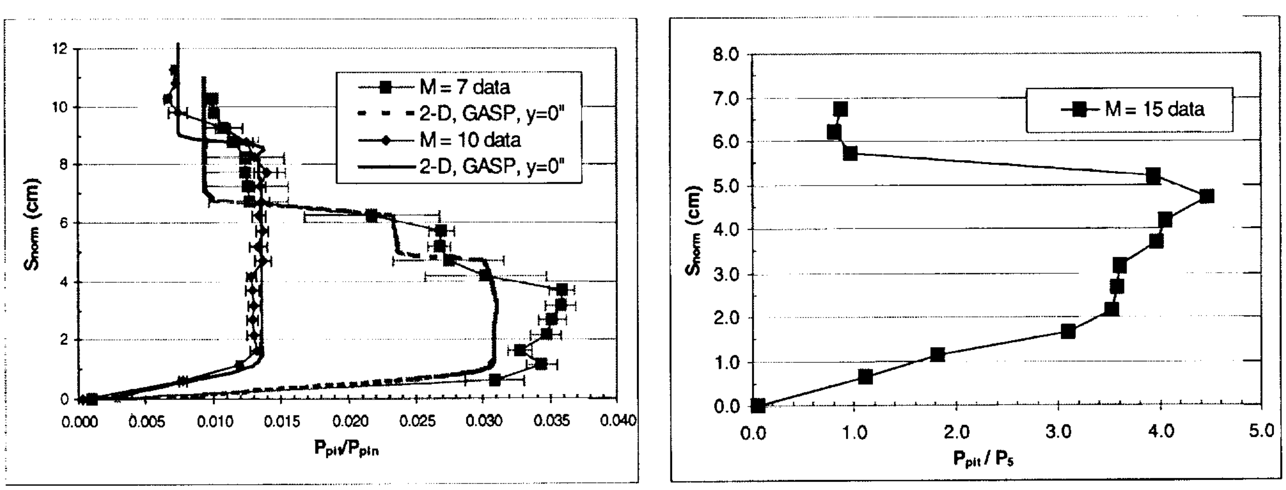


Figure 6.- Forebody pitot survey and schlieren for Mach 7 tests



(a) Mach 7 & 10 tests.

(b) Mach 15 tests

Figure 7.- HSEM Forebody pitot data with CFD

Test Procedure: The test runs in each series included variation of the injected fuel equivalence ratio, the fuel mixture, and engine inlet conditions. Fuel equivalence ratio variation included fuel-off runs (referred to as "tare"), and nominal ϕ = 0.8, 1.2, and 1.5, depending on the mass capture and the fuel mixture. The fuels were pure H_2 and H_2 in mixtures with small amounts of silane (SiH_4), a pyrophoric gas that is used as a scramjet combustor ignition aid, particularly at the Mach 7 condition. Silane-fuel mixtures (SFM), containing 2% and 5% SiH_4 by volume in H_2, were used in both Mach 7 and 10 tests to provide a qualitative assessment of fuel ignition. Silane hydrogen mixtures have been used for many years in the blow-down scramjet test facilities at LaRC.[5] The Mach 15 tests were conducted with pure H_2 fuel, but some runs included 2% SFM. These Mach 15 tests are preliminary as the first scramjet engine tests in the SET mode of HYPULSE, which has not yet been fully calibrated and developed. However, the Mach 15 tests do provide useful information in the development of test techniques and facility operational mode at this hypervelocity condition.

Engine inlet parameters were varied to examine some effects of flight simulation. These parameters included the mounting angle of the HSM hardware relative to the tunnel flow (to change the inlet pressure levels to be typical of flight dynamic pressure), and test gas composition (to replenish the O_2 depleted by NO formation in the facility nozzle plenum in the RST mode at Mach 10). Estimated composition of the Mach 10 test gas at the nozzle exit was about 6% NO by mass, that reduced the O_2 to about 19.5% by mass. At the Mach 7 condition, dissociation was slight, with about 0.5% by mass of NO. In the SET mode, very little dissociation occurs because the test gas is not stagnated; however, estimates have not yet been made. A summary of the test runs made in the three test series is given in Table 3.

Table 3. HYPULSE Scramjet Engine Model (HSEM) Tests

Test Condition	Mode	Nozzle AR	Stagnation Pressure. [Mpa]	Mounting Angle [deg]	Calibration Runs	Engine Test Runs
Mach 7	RST	175	9.93	9.7	16	27
Mach 10	RST	225	23.26	5.5	4	29
Mach 15	SET	16	~3030 (effective)	8	1	9

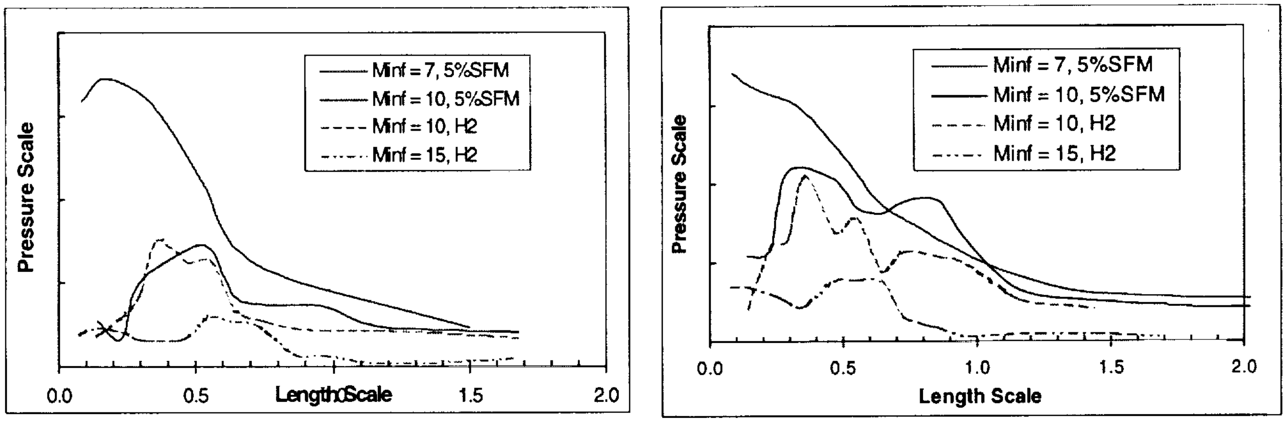
Results

Results of the HSM tests are presented in the form of flowpath pressure distributions and a stream thrust performance parameter for each test series.

HSM Pressures

Comparisons of the HSM normalized pressure distributions from tests at Mach 7, 10, and 15 conditions are given in Figure 8. The individual pressure distributions have been smoothed and scaled from the test values to conditions expected on a typical aero-space vehicle in flight at q = 1000 psf across the tested flight Mach number range. The smoothing better illustrates the general effect of flight Mach number (stagnation enthalpy) on pressure rise by filtering the changes in internal shock angles as a result of combustion.

Comparisons for fuel lean and fuel rich operation are shown in Figures 8(a) and 8(b), respectively. The Mach 7 series used a silane fuel mixture (nominally 5%) in all tests; therefore, comparisons between pure H_2 fuel and SFM fuel are for the Mach 10 and 15 tests only. The pressure distributions generally exhibit lower values as the flight Mach number increases, as is expected because of the higher energy in the mainstream airflow. At the Mach 7 condition, the combustor is operating in dual-mode, with a separated flow entering the combustor section, as evidenced by the higher pressure at the combustor entrance. The Mach 10 data show a small difference between the pure H_2 and 5%SFM fuels, except in the near field where the onset of pressure rise occurs closer with the SFM fuel, particularly for the fuel rich tests results in Figure 8(b).



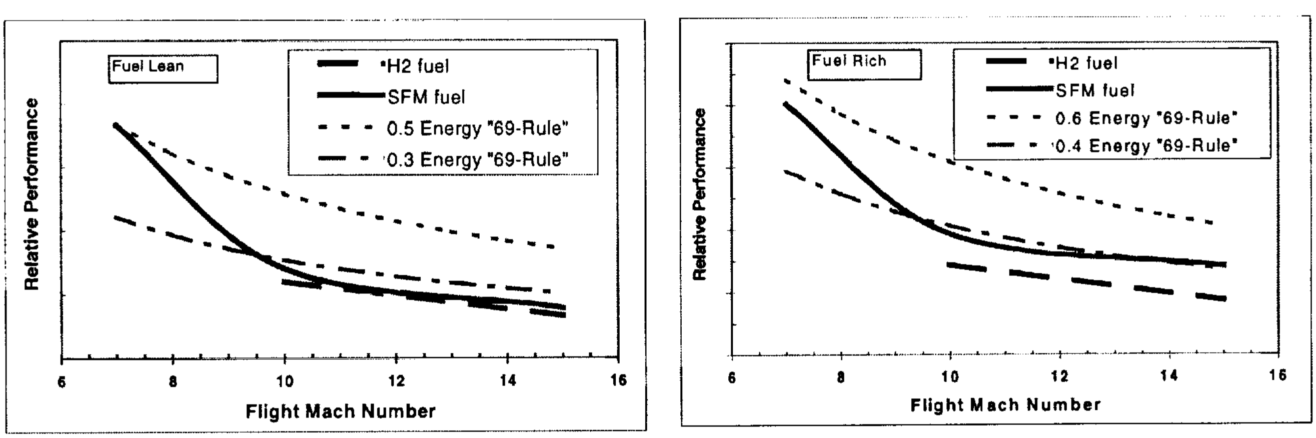
(a) Fuel lean cases (b) Fuel rich cases

Figure 8.- Comparison of engine pressure with flight Mach number.

Performance Estimates

Comparisons of the HSM flowpath estimated performance are presented in Figure 9. The parameter was derived from the streamtube thrust and scaled from the test conditions to the generic flight vehicle at flight q = 1000 psf. For the fuel lean runs in Figure 9(a) the performance follows the expected trend of decreasing with increasing flight Mach number (stagnation enthalpy). In general, if the flight enthalpy (~ velocity squared) doubles, streamtube thrust performance is reduced by about half. Note that the effect of the SFM fuel on this performance parameter is slightly better than pure H_2 fuel. Similar results are shown for the fuel rich cases in Figure 9(b).

A performance scaling parameter was derived from the energy available in the fuel. This value, which has been referred to as the "Rule of 69",[18] is that the ratio of the energy available from burning all the fuel to the kinetic energy in the captured air stream tube is equal to about $69/M^2_\infty$. For the fuel rich cases, the fuel energy is limited to stoichiometric fuel-air ratio. Because some of the fuel heat release goes into thermal, some to combustor boundaries, and some is not released due to incomplete mixing and combustion, equivalent performance values were computed for fuel rich operation assuming 60% and 50% conversion to kinetic energy for the Mach 7 and 10 conditions, respectively. These results are plotted in Figure 9(a) and (b). For the fuel lean cases, with a nominal $\phi = 0.8$, the amount of fuel energy going into kinetic energy was reduced by an additional factor of 0.8 from the fuel rich values. The fuel energy parameter exhibits about the same trend, although not quite as much decrease with increasing Mach as the data.



(a) Fuel lean cases (b) Fuel rich cases

Figure 9.- Comparison of HSM performance with flight Mach number

Concluding Remarks

A series of scramjet engine flowpath tests have been conducted in the HYPULSE shock tunnel at conditions duplicating the stagnation enthalpy at flight Mach 7, 10, and 15. For the tests at Mach 7 and 10 HYPULSE was configured as a reflected-shock tunnel. At the Mach 15 condition, HYPULSE was operated as a shock-expansion tunnel. The test conditions matched the stagnation enthalpy of a scramjet engine on a typical aerospace vehicle accelerating through the atmosphere along a 1000 psf dynamic pressure trajectory. Test parameter variation included fuel equivalence ratios from lean (nominally 0.8) to rich (1.5+) and variation in fuel composition from pure hydrogen to mixtures of 2% and 5% silane in hydrogen by volume. The small amount of silane was used as an ignition aid. Other test parameters were inflow pressure and Mach variations made by changing the scramjet engine mounting angle in the HYPULSE test chamber. Data sources were wall pressures and heat flux, and schlieren and fuel plume imaging in the combustor/nozzle sections for some runs.

Comparison of the wall pressure distributions in the scramjet flowpath, when normalized and scaled to the assumed flight conditions for a typical space access vehicle, indicated a trend of decreasing pressure rise with increasing flight Mach number. This effect seems to be more noticeable for fuel lean operation. A performance estimate based on the streamtube thrust indicated the general trend that the performance decreased by about one-half when the stagnation enthalpy (flight Mach squared) was doubled. An estimate of performance based on the increase in engine flow kinetic energy due to the heat release from burning the fuel was presented for comparison.

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