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A SUBSONIC TO MACH 5.5 SUBSCALE ENGINE TEST FACILITY

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Abstract

The Hypersonic Propulsion Branch at NASA-Langley has focused its research on the supersonic-combustion ramjet (scramjet) airframe-integrated propulsion system concept since about 1969. Engine component (inlet and combustor) research was performed and documented in the early 1970's. During this same period, two existing ground facilities were modified to allow tests of subscale, component-integration engine models at simulated Mach 4 and 7 flight conditions. The Mach 4 facility, in which air-hydrogen combustion is used to heat the test gas, has been recently modified to make it suitable for tests of ramjet type models at subsonic and transonic Mach numbers as well as the present supersonic Mach number capability. These modifications are described, and the extended test capabilities of the facility are documented in this paper.

Nomenclature

d	nozzle exit equivalent diameter (14.97), in.
H	height of square nozzle, in.
$H_{t,\infty}$	free-stream total enthalpy, BTU/lb.
M_1; M_{ex}	Mach number behind vehicle bow shock or at facility nozzle exit
M_{EJ}	facility air ejector exit Mach number
M_∞	free-stream flight Mach number
p	static pressure, psia
p_{cab}	facility test cabin pressure, psia
p_{ex}	facility nozzle exit wall pressure, psia
p_{pitot}; $p_{t,2}$	total pressure behind a normal shock (total pressure at $M_{ex} < 1.0$), psia
$p_{t,1}$; $p_{t,ex}$	total pressure behind vehicle bow shock; facility nozzle exit, psia
$p_{t,brn}$	facility burner total pressure, psia
$p_{t,\infty}$	free-stream flight total press., psia
q_∞	free-stream flight dynamic pressure, psf
R	facility nozzle exit radius, in.
$T_{t,brn}$; $T_{t,ex}$	facility burner or nozzle exit total temperature $(= T_{t,\infty})$, °R
$T_{t,\infty}$	free-stream flight total temperature, °R
X_D	distance in facility exhaust duct (see Fig. 13), in.
y	vertical distance from facility nozzle centerline, in.
γ	ratio of specific heat
θ	vehicle precompression angle, deg.
$\dot{w}$	flow rate, lbs/sec.

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Introduction

NASA-Langley's involvement in airbreathing hypersonic engine research and development began in the early 1960's. This research was focused around the Hypersonic Research Engine (HRE), an axisymmetric ramjet/scramjet that was designed to be pod-mounted to an X-15 for flight tests up to Mach 8. The X-15 program was terminated before the flight tests could be conducted, but two engine models were fabricated and tested in ground test facilities as shown in Figure 1.[1-3] These models were used to successfully demonstrate the ramjet/scramjet cycles with variable engine geometry (a translating inlet spike) and to initiate development of regeneratively-cooled flight hardware. After completion of the HRE program, the Hypersonic Propulsion Branch (HPB) at NASA-Langley initiated a program to study ramjet/scramjet propulsion systems that were highly integrated with the vehicle to obtain a more efficient propulsion system for hypersonic speeds which was designed for minimum drag and weight and to process as much air as possible. This airframe-integrated propulsion system encompasses the entire underside of the vehicle (Fig. 2). The aircraft forebody is used for part of the inlet compression process and the aftbody for part of the nozzle expansion process. The propulsion system includes a cluster of identical individual rectangular engine modules of a size and shape suitable for ground testing. An internal view of one module of the Langley first generation airframe-integrated scramjet is shown in Figure 2. A considerable amount of inlet and combustor component research related to this airframe-integrated engine concept was performed within the HPB in the 1960-80's.[4] In the mid 1970's two existing facilities at Langley were modified to allow tests of subscale, component-integration engine models at simulated Mach 4 and 7 flight conditions. Four different engine concepts have since been tested, and results from three of these engine tests are summarized in Reference 5.

The Mach 4 engine test facility consisted mostly of surplus hardware assembled in an existing test cell. The test gas was heated by air/hydrogen combustion with oxygen replenishment to allow propulsion testing. The present facility heater has a pressure limit of 190 psia and a temperature limit of 2250°R because of safety considerations. A new heater will increase these limits to 600 psia and 3000°R and a new facility nozzle and other optional ducting hardware have greatly extended the operating range of the facility. Plans are also underway to incorporate a 70-foot vacuum sphere into the facility system. In this paper, the original and the modified features of the facility are described and the extended operating range of the facility is documented. In addition, special facility problems associated with scramjet engine testing are discussed and the relationship of this facility to the entire Langley Scramjet Engine Test Complex is described.

Langley Scramjet Research

Basic Approach

Extensive inlet and combustor research programs, both analytic and experimental, have been and are being conducted at LaRC in support of the airframe-integrated scramjet concept.[4] Inlet and combustor designs generated in this component research are then assembled to form component-integration engine models. These subscale engine models are tested in ground facilities to define and resolve interactions between the various engine components and to determine the overall engine performance. The models are of the heat-sink type and are generally representative of one module of a cluster shown on the aircraft in Figure 2.

The ground facility tests must simulate the flight conditions of an airframe-integrated scramjet as closely as possible. The simulation logic is depicted in Figure 3. A vehicle flying at supersonic or faster speeds compresses the flow across the forebody bow shock. At a particular altitude, increased aircraft speed results in increased stagnation conditions such as total pressure and total enthalpy (or temperature). The flow conditions downstream of the bow shock are different in that the velocity (Mach number) and total pressure are decreased, the static pressure is increased, and the total enthalpy or total temperature remains the same. It is this flow condition just ahead of the propulsion system, depicted in Figure 3(a) by M_1, that has to be duplicated in an engine test facility.

The arrangement shown in Figure 3(b) schematically represents an engine test facility that produces true-velocity, true-temperature, and true-pressure flow for flight simulation. Flight freestream total enthalpy or total temperature is duplicated by a heater (with oxygen replenishment, as required). A facility contoured nozzle exhausts the heated, simulated-air flow to the engine module inlet at a Mach number (M_1) that simulates the vehicle forebody precompressed flow. The altitude, or dynamic pressure, simulated by the nozzle exit flow is dependent upon the heater total pressure. Also, the scramjet engine modules are mounted in the facility so that all or a portion of the facility-nozzle top-surface boundary layer is ingested by the model in partial simulation of the ingestion of a vehicle forebody boundary layer by a flight engine.

In an arrangement such as presented in Figure 3(b), flight conditions are duplicated as closely as possible, i.e., true total enthalpy, static pressure, static temperature, model scale, test gas composition, turbulence level, etc. Although total enthalpy can be duplicated in subscale ground engine test facilities; the other factors must be examined more closely. Heating the test gas to the correct total enthalpy to simulate the flight Mach number usually is accompanied by flow contamination, i.e., H_2O, NO_x, etc, that may effect engine fuel combustion. Expansion of the heated air in the facility nozzle leaves the vibrational mode in a nonequilibrium state and, above $T_{t,1}$ = 4000°R, chemical nonequilibrium (oxygen dissociation first) can occur. These affect nozzle exit flow properties and parameters; i.e., P, T, ρ, M. Facility turbulence level which is almost certainly an unknown in engine tests, varies with each facility and could have an effect on fuel-air mixing. Engine size (scaling effect) is very important if combustion is not mixing-controlled since chemical kinetics do not scale (mixing does scale). Therefore, poor thrust performance in subscale tests due to kinetics problems does not necessarily mean that a larger engine would have the same poor performance. Facility model interactions can occur which will negate or make data interpretation difficult; marginal facility diffusers can cause this problem. Although the factors outlined above warrents caution in interpretation of subscale scramjet ground facility data, these tests are important and informative. Subscale scramjet ground facility tests can be performed relatively inexpensively, can yield large quantities of data, and this data is valuable in the study of combustor-inlet interaction; fuel injector size, spacing, and staging; flameholding; and thrust performance.

Engine and Component Test Facilities

The Hypersonic Propulsion Branch has assembled a group of facilities that enables such scramjet research to be conducted at the NASA-Langley Research Center. Photographs of the facilities are shown in Figure 4.

Small-scale inlet experimental tests for "screening" potential inlet designs are performed in the Mach 4 Blowdown Tunnel that uses unheated air and has a 9 x 9-inch test section. Much can be learned from such tests such as the inlet starting capability, restart capability, inlet air mass capture, and surface pressures that can be compared to computational fluid dynamic (CFD) results. The models are generally relatively inexpensive with considerable model geometric versatility. Larger-scale inlet tests are also performed in larger aerodynamic wind tunnels at Langley.

Small-scale direct-connect combustor tests that simulate a portion of the engine combustor are conducted in Test Cell #2 Direct-Connect Combustor Facility (Fig. 4) to provide basic research data on supersonic fuel-air mixing, ignition, and combustion processes. The hot test gas is supplied to the combustor models by a hydrogen-air-oxygen combustion heater, to simulate air with enthalpy levels ranging up to Mach 7 flight speeds.[6] Various facility nozzles produce the desired combustor entrance flow conditions.

Much can be learned from individual component experimental basic research, but when these components are integrated into an engine configuration a new set of problems may be encountered. Flow conditions supplied to direct-connect combustor tests are generally shock free uniform flow conditions. The flow delivered by an inlet attached to a combustor entrance in an engine configuration includes reflected shocks and non-uniformity. It was therefore desirable to provide at Langley the capability to conduct subscale engine tests to perform inlet and combustor basic research in an engine environment. To fulfull this requirement, two engine test faclities have been assembled by the Hypersonic Propulsion Branch (Fig. 4). These

facilities permit inexpensive, highly productive, combustion and engine research tests to be conducted on small-scale, gaseous-fuel-burning, scramjet models. One is an electric-arc-heated facility with 11-inch square exit contoured nozzles that exhausts through a diffuser into a 100-ft vacuum sphere.[7,8] Another facility utilizes a hydrogen-burning heater to provide a hot test gas with the proper oxygen content that is expanded through a 13.26-inch square exit contoured nozzle; an air ejector is employed to aid exhausting to the atmosphere.[9] Both facilities yield free-jet tunnel flow for subscale scramjet tests at simulated Mach 7 and Mach 4 flight conditions in the arc-heated and combustion-heated facilities, respectively.[5] The same size models (frontal view about 6 by 8 inches and 72 inches long) are tested in both of these facilities. The combustion-heated facility is the topic of the remainder of this paper.

Facility Description and Discussion

Basic Configuration with Air Ejector

Description. - The Combustion-Heated Scramjet Test Facility (CHSTF) basic configuration is represented in Figure 5.[9] This figure has a schematic sketch of the basic configuration along with a photograph and some salient features of the facility. All of the facility hardware is contained in a 16 x 16 x 52 foot test cell as shown in the sketch with the flow direction from left to right. Air is supplied by a centralized 600 psi distribution system. (The air supply into the Test Cell Complex and its controls are shared along with the gaseous propellant supplies and controls with the Test Cell #2 Direct-Connect Combustor Facility.) Oxygen is injected to mix with the air to yield an oxygen-rich air mixture. Hydrogen is then injected into this oxygen-rich air mixture in the facility heater and ignited with a hydrogen/oxygen torch igniter. The facility hydrogen and oxygen flow rates are controlled so that the resulting combustion product mixture contains approximately 21 percent free oxygen by volume to simulate the oxygen content of air. The remaining test gas is a mixture of nitrogen and water vapor; the higher the stagnation temperature for higher simulated flight speed, the greater the hydrogen flow rate required for combustion and thus the larger the water vapor content in the test gas. The mass fraction of the test gas constituents at simulated flight Mach number total temperatures are presented in Figure 6. Mach 4 flight simulation (T_t = 1640°R) results in a nominal test gas composition of 6 percent water, 70 percent nitrogen, and 24 percent oxygen by mass; 9, 70, and 21 percent, respectively by volume. Expansion of the test gas is accomplished through a Mach 3.5 contoured nozzle that has square cross sections and is 13.264 inches square at the exit. The Mach 3.5 free-jet exhaust flow simulates the forebody precompressed flow of a vehicle at Mach 4.0 flight conditions as discussed for Figure 3. A test cabin contains the free-jet exhaust flow that is received by an exhaust catch cone to yield a free jet length (from nozzle exit to cone entrance) of 1.5 nozzle exit equivalent diameters. The catch cone is connected to a 19-inch diameter straight duct supersonic exhaust diffuser (about 8-diameters long). The tunnel flow (30 to 60 pounds per second) at the exit of the diffuser is pumped upon by an annular air ejector (about 200 pounds per second) with an exit Mach number of 4.16 (can be manually changed to M = 3.72). These flows mix in a 25-inch diameter mixer duct that is about 5.5-diameters long and is then turned vertically by the turning elbow. The flow exhausts from the elbow as a free-jet and enters a 6-foot diameter duct in the test cell ceiling exhaust tower. This relationship results in a low-pressure, high-volume flow ejector that entrains test cell air into the vertical exhaust duct. This ejector action, in turn, induces flow to enter through the test cell ceiling entrance tower and "wash" over the facility hardware and piping to ensure purging from the test cell any gaseous propellant leakage that may occur (four hydrogen detectors are strategically located in the test cell, test cabin, and exhaust tower).

A subscale scramjet (about 6 x 8 x 72 inches) is installed in the facility free-jet flow at the exit of the facility nozzle as discussed for Figure 3(b). Engine model gaseous hydrogen fuel is supplied, up to 700 psia, through six individually controlled systems. These six systems can be routed to different engine fuel injection stations that can be used as desired during a test. One of these systems generally controls the supply of a phyorphoric gas used for engine fuel ignition. This same system can also be easily converted to supply and control a higher pressure (up to 1000 psia) gaseous hydrogen fuel. A 1000 psi air supply (1 inch pipe) is also available in the test cell that may be connected to one of the six control systems. Also for fuel ignition, two high-voltage electric transformers and their controls are available as standard test options in the test cell.

Facility instrumentation. - The facility was heavily instrumented during calibration tests and during initial engine model tests. The main computer of the data acquisition system (DAS) is a multiprogrammable, 32-bit, general purpose digital computer with parallel 500,000 word paged memory. The DAS is designed for high-speed data acquisition and includes a console terminal, two disk drives (a 24 and a 48 megabyte size), a card reader, two 800 bits/in. tape drives, and a line printer. A graphics terminal attached to the main computer over an RS-232 interface permits desired data to be plotted at a rate of 2400 bits/sec. A series of electronic amplifiers allows up to 192 analog signals to be digitized. An electronically scanned pressure (ESP) measurement system is attached and controlled over a standard IEEE-488 interface. The ESP is used for some of the facility duct pressure measurements, but its main purpose is to provide model pressure measurements up to 75 psia. Model thrust and drag is measured with a one-component strain-gauge force balance.

Data are recorded at a scan rate of 10 frames per second and are generally printed at a rate of 2 per second for a nominal 20-second run. The printed data are in engineering units, ratios, and/or calculations using raw data; printing starts within 1 second after the end of a test. Data are immediately plotted on the graphics terminal from which a standard set of plots is obtained on a hard copier for on-site preliminary run analysis. Once selected data are stored, the

data can be recovered for additional analysis either on site or by remote terminals interfaced with the computer.

Operational conditions. - A portion of the operational corridor typical of scramjets is shown in Figure 7(a). This corridor represents the operation of an engine concept that would operate as a ramjet or with mixed subsonic-supersonic combustion over the flight Mach number range of 3 to 6 and as a scramjet for flight Mach numbers above 6. A range of flight conditions that can be simulated (without precompression consideration) by the basic facility configuration is represented by the hatched area of Figure 8(a). The vertical bar represents the present facility nozzle with a nominal exit Mach number of 3.5. The flight simulation region (hatched area) is dictated by the Mach 3.5 nozzle, the present maximum operational heater pressure limit of 190 psia, and the pumping capability of the air-ejector which exhausts to the atmosphere (nozzle Mach number, model/diffuser arrangement/size dependent).

Proper control of the heater oxygen and hydrogen allows the heater test gas temperature to be varied to simulate flight stagnation temperatures over a range of Mach numbers represented in Figure 7(b); the basic configuration has a heater maximum total temperature limit of 2250°R or about Mach 5 simulation. Since the facility nozzle exit Mach number remains essentially constant (slight change because of temperature change), the corresponding range of angles (assumed wedge) that the vehicle undersurface, relative to the horizontal, has to attain to maintain a constant Mach number (3.5 or M_1 in Fig. 3(a)) ahead of the inlet is also represented in Figure 7(b). Associated curves for flight free-stream stagnation pressure and dynamic pressure assuming a constant total pressure ahead of the engine inlet of 190 psia, which is the present heater maximum pressure limit, are presented in Figure 7(b). (Similar curves could of course be represented at lower heater total pressure values.) As the total temperature simulation value is increased the simulated precompression angle may not necessarily simulate a realistic flight cruise or acceleration condition; however, the variable total temperature capability allows engine environment combustion studies to be performed very easily at various total temperatures that simulate those for various flight Mach numbers.

Modifications

The flight simulation capability of the basic facility configuration is being expanded to allow a larger range of true simulation in subscale engine tests in the CHSTF. Obvious limitations for scramjet testing are the test gas heater total pressure and temperature limits of 190 psia and 2250°R, respectively. New design maximums are: (1) a heater total pressure of 600 psig (the air supply limit), and (2) a heater total temperature of about 3000°R. In order to take advantage of such expanded test gas heater capabilities, new facility nozzles with different exit Mach numbers are required to properly simulate the flight precompressed flow conditions entering the engine inlet. Heater and nozzle extended capabilities are relatively inexpensive revisions to expand the facility flight simulation capability. Another more expensive revision is a replacement of the air ejector exhaust system with a vacuum sphere exhaust system sufficiently sized to allow 20-30 second hot test times. Plans for such revisions are underway; required facility hardware and the effect on the facility flight simulation capability will be discussed in the following sections.

Heater - The relative position of the mixer and heater assembly in the basic facility configuration is shown in the schematic of Figure 5. The schematic in Figure 8(a) shows some of the major details of this assembly. As previously mentioned, the air and oxygen are premixed in the mixer section between the oxygen and hydrogen baffle plates; thus the hydrogen is injected into oxygen-rich air. Both baffle plates have 2 rings of orifices, 10 orifices in the inner ring and 20 in the outer ring, through which injector tubes pass. Ignition of the heater propellants is provided by a hydrogen-oxygen torch ignitor that is installed as shown in Figure 8(a). Premixing the air and oxygen results in good mixing and thus allows the length-to-diameter ratio of 3.3 to be sufficient for good combustion in the heater. The design of the passages through the oxygen and hydrogen baffle plates was such that the air Mach number is about 0.9 and the air-oxygen mixture Mach number is about 0.7 through the respective baffle plates. Oxygen and hydrogen are injected through orifices at about M = 0.7.

The present test gas heater shown in Figure 8(a) has a longitudinal-slot cooling water jacket (450 gal./min. water flow). This heater duct is an item of surplus equipment for which the design documents were not available. After safety tests were performed, the duct was sanctioned for use in the secured test cell at maximum operating limits of 190 psia and 2250°R which contribute to the present restricted operational range of the facility.

The replacement test gas heater to be installed in 1987 will extend the facility operational range. The design of this heater duct incorporates a 0.75-inch thick, heat-sink 201-nickel liner with an 18.0-inch internal diameter contained within a 24.0-inch diameter schedule-40 carbon steel pipe. Provisions are made to allow the option of the liner to be back-side cooled with air flowing from the downstream end toward the upstream end and into the heater chamber as shown in Figure 8(b). An adaptor flange will be used to attach the new heater to the existing mixer section/hydrogen baffle plate shown in Figure 8(a). A downstream end flange is a replaceable adaptor flange to accept various facility nozzles.

Test setup configuration. - Two facility test configurations are shown in the schematics of Figure 9. The hardware arrangement for a supersonic test setup is shown in Figure 9(a). This setup uses either the present Mach 3.5 nozzle or the new Mach 4.7 nozzle which are connected directly to the test gas heater downstream flange. The engine model is installed at the facility nozzle exit, as discussed for Figure 3(b), suspended from a mounting beam by four flex beams that allow longitudinal movement of the engine. A preloaded one-component force balance senses the longitudinal thrust/drag force of the engine. The nozzle extension, the exhaust flow catch cone, and

the initial portion of the supersonic diffuser are shown in Figure 9(a) (latter two shown as dashed lines).

The test setup for subsonic/transonic experimental investigations is represented by the schematic of Figure 9(b). In this setup the test cabin, model installation, and exhaust system remain the same as shown in Figure 9(a). The subsonic/transonic configuration is obtained by replacing the facility air/oxygen mixer section, the hydrogen-air-burning heater duct, and the contoured supersonic nozzle with ducting that delivers unheated air to the engine model. Two nozzle blocks can be easily exchanged to yield Mach 1.2 flow or subsonic flow with the convergent Mach 1.0 nozzle. Various subsonic Mach number flows may be obtained by adjusting the chamber total pressure. The nozzle exit static pressures for both nozzle blocks is slightly below atmospheric pressure without the use of the facility air ejector. When the facility air ejector is operated at low flow rates, the nozzle exit pressures are about one-half atmosphere which simulates an altitude of about 17,500 feet. During subsonic tests without the facility air ejector operating, the test cabin pressure changes slightly as combustion occurs in the engine. This in turn affects the tunnel operating pressure, which causes the facility nozzle exit Mach number to change slightly.

Nozzles. - The present facility supersonic nozzle, shown in the sketch of Figure 10(a), is an uncooled contoured square nozzle designed on the basis of streamline-tracing the flow of an axisymmetric nozzle. The throat is 4.976 in. square (throat area of 24.81 in.^{2}), and the flow exit is nominally 13 in. square (actual geometric nozzle exit dimensions are 13.264 in. to account for boundary layer displacement thickness). At a total temperature of 1640°R, the nozzle-exit Mach number is 3.50. The nozzle entrance, which protrudes into the heater duct, makes a transition from a circular to a square cross section. The throat section was constructed with a large mass of stainless steel for heat sink, and the downstream expansion section of the nozzle was constructed of 0.183-in-thick carbon steel with external stiffening webs. The nozzle sidewalls and bottom wall were extended at the exit to ensure that shocks, generated when the ratio of test cabin to nozzle-exit static pressure increased from 1.0 to 2.0, would not enter the internal flow region of an engine model.

An additional nozzle, available in the latter part of 1987, is shown in Figure 10(b). This nozzle is similar to the Mach 3.5 nozzle but has an exit Mach number of 4.7 to simulate airframe-integrated precompression at a flight Mach number of 5.5. A copper water-cooled throat section is incorporated in the design to allow operation at Mach 6.0 flight stagnation temperature (3000°R). The downstream contoured section is uncooled and is fabricated by the same method as the present Mach 3.5 nozzle. The throat is 2.588 inches square (throat area of 6.70 in.^{2}) and the exit is the same as the Mach 3.5 nozzle, 13.264 inches square.

Nozzle exhaust flows. - Exhaust flow profiles of the existing facility nozzles are presented in Figure 11. These results were obtained with multiprobe rakes positioned at the nozzle exit plane along the vertical centerline.

Profiles for the Mach 3.5 nozzle at 95 psia and 1600°R heater test gas stagnation conditions are presented in Figure 11(a). The exit instream pitot pressures and total temperatures were obtained by use of a 13-probe (one-inch spacing) stationary rake with alternate pitot pressure and total temperature probes (seven and six probes, respectively). Boundary layer pitot pressure surveys were obtained with a 7-probe stationary rake. Static pressure instream surveys were made but the results were the least reliable of the survey measurements. The nozzle wall exit static pressure was assumed as an instream constant static pressure and used with the measured instream pitot pressures. This was not a valid assumption because at some locations in the stream near the boundary layer edge, total pressures greater than the test gas heater total pressure were computed from the resultant Mach numbers. Therefore, a computed total pressure trend associated with the wall static and pitot pressure measurements in the boundary layer was faired into a constant value equal to the heater test gas total pressure. This total pressure trend was then used with the measured pitot pressures to yield the ratio of pitot to total pressure profile shown in Figure 11(a). The total temperature profile is nearly constant and is faired through the boundary layer toward an adiabatic wall recovery temperature. The Mach number profile shown was derived from the pitot-to-total pressure ratio profile. Since the square nozzle contours are obtained from the streamline tracing technique using axisymmetric nozzle flows, some total pressure loss is likely to occur in the axis region of the nozzle exhaust flow. Such reduced total pressures would tend to flatten the Mach number profile in the axis region. The Mach number of 3.55 indicated on the Mach number figure was calculated using the nozzle wall exit static pressure-to-heater test gas total pressure ratio.

Exit flow profiles for the transonic axisymmetric nozzle are presented in Figure 11(b). These profiles were obtained using the same 13-probe rake which was used for the Mach 3.5 nozzle except all probes were converted to measure pitot pressure. Boundary layer surveys were not performed. The measurements of the two end probes were in line with the nozzle exit walls (12.0 inch diameter) and may be in a disturbed flow. The pressure profiles are a ratio of the nozzle wall exit static pressure to measured pitot pressures. The Mach number profiles shown are a function of these pressure ratios. Noted at the top of the figures are the tunnel chamber total pressure, nozzle wall exit static pressure, and Mach number as a function of the ratio of nozzle wall static-to-tunnel total pressure. The profiles shown for the transonic nozzle are representative of test conditions with and without the operation of the facility air ejector.

The various profiles shown in Figure 11(c) for the subsonic nozzle are the result of different tunnel total pressures as indicated at the top of the figure (the pitot measurements are of course instream total pressure measurements). Most of these tests were performed with the nozzles exhausting into the closed test cabin without the

facility air ejector in operation. Slight aspiration of the test cabin resulted with the nozzle exhaust flow entering into the exhaust duct catch cone; the nozzle exit static pressures indicated at the top of the figure are very nearly equal to the test cabin pressure. In order to obtain near sonic conditions, the facility air ejector had to be used; the elongated-diamond-symbol profiles in Figure 11(c) are results obtained with the air ejector in operation.

Expanded operational conditions. - The addition of the new test gas heater and Mach 4.7 nozzle results in expanded facility operational conditions that simulate true flight conditions (without precompression) as represented by the hatched area of Figure 12. The higher pressure and temperature limits of the new heater allow both lower altitude and higher Mach number simulation. A larger range of vehicle precompression simulation is available with the additional Mach 4.7 nozzle. Higher altitude/lower dynamic pressure simulations remain limited by the operational capability of the air ejector as shown in Figure 12. Also shown in Figure 12 are the subsonic and transonic nozzle flight simulation capabilities.

Test Experience

Exhaust duct flow. - Flow conditions throughout the facility ducting can be determined from duct wall pressure and pitot-pressure rake measurements; the relative locations of the wall and rake pressure measurements are indicated in Figure 13(a). Two special problems are associated with testing scramjets in the facility. First, installation of a scramjet model with the inlet top surface aligned with or near the facility nozzle top wall creates asymmetric flow profiles in the exhaust duct (greater pressure losses on the top side of the ducting). Second, the facility exhaust duct mixing length had to be made shorter than optimum because of the containment of the facility within an existing test cell; this results in the asymmetric flow still being in existence at the entrance of the turning elbow. These flow conditions are evident in Figure 13(b). The undisturbed trends (circle symbols) are evident at the start of a test and at moderate engine fuel flow rates. During tests of poor engine fuel combustion or at high fuel flow rates, some of the fuel is not burned entirely within the engine. This unburned fuel apparently ignites in the facility exhaust duct and generates increased pressures in the duct that eventually affect the tunnel flow in the region of the engine. This condition is referred to as facility-engine interaction and duct pressure measurements associated with the disturbed duct flow are represented by the square symbol trends in Figure 13(b). The test cabin static pressure and the exhaust straight duct supersonic diffuser exit static pressure and flow Mach number are good indicators of facility flow breakdown (facility-engine interaction) and are generally observed during a test and/or during post test on-site analyses of the test data. Large engine models cause more facility sensitivity to these facility-engine interactions.

A few techniques have been successful in minimizing these interactions. Water sprayed into the diffuser catch cone and at the elbow turning vane leading edges has been effective. Also, an increased length of the facility nozzle exit extension decreased the facility free-jet length (about 0.75 equivalent diameters) and made the nozzle flow less sensitive to increases in the exhaust-duct back pressure. With these techniques incorporated, the exhaust duct flow trends remained undisturbed (circle symbol curve, Fig. 13(b)) as engine fuel flow rate was increased to much higher levels of flow rates before the exhaust duct flows became disturbed (similar to square symbol curves) than was possible prior to the resolutions. Prior to the resolution of the interaction problem, the increased pressures in the exhaust system affected the engine measurements and distinction was difficult between engine combustor-inlet interactions and facility engine interactions. When the engine inlet flow "unstarted," the facility nozzle flow was drastically affected. After the problem resolutions, engine combustor-inlet interactions, caused by model combustor fuel-burning pressure rise, could be obtained before the occurrance of or without a facility-engine interaction. Also, after the changes were made, even with the inlet unstarted and a facility exhaust duct flow breakdown, the ejector could maintain the cabin pressure sufficiently low so that the facility nozzle flow was seldom affected.

Models tested. - Four different subscale scramjet engine models have been tested in the present configuration of the CHSTF at nominal stagnation conditions of 92.4 psia and 1640°R which simulated Mach 4 flight at a dynamic pressure of 1000 psf. The models were large relative to the facility nozzle exit of 169 sq. inches. Some of the model configurations could be varied resulting in different inlet cowl frontal areas which, for the four models, ranged from 12 to 30 percent of the facility nozzle exit area; maximum external cross sectional areas ranged from 31 to 49 percent. When these models were installed with a catch cone, nozzle extension, and water sprays as discussed previously, supersonic flow could be established through the exhaust diffuser (pitot and wall static pressures at the diffuser exit indicated the flow to be at Mach number of about 1.4 at zero engine fuel flow). During such conditions, the cabin pressure was very nearly equal to the facility nozzle exit pressure. As the engine fuel flow was increased, the diffuser flow Mach number decreased and for some high fuel flow rate tests, the diffuser exit flow Mach number became or was about to become subsonic. The cabin pressure for these cases increased to a point that a facility-engine interaction occurred. As mentioned previously, larger model blockage caused more sensitivity to facility-engine interactions. It is therefore recommended that a conservative approach should be taken and an attempt made to design for a model cowl area blockage of 25 percent or less and/or 35 percent model maximum area blockage. Also, a facility should be designed with diffuser and mixer duct lengths of optimum length where possible. This approach should result in more stable exhaust duct flow conditions.

Facility with Vacuum Sphere

As mentioned previously, the simulated flight envelope for engine tests in the CHSTF is currently limited in altitude simulation by the air ejector exhaust system. This ejector system

consumes large quantities of the facility air supply while providing only marginal altitude simulation. In addition, facility-model interactions have, during some tests, interfered with obtaining data at model full throttle setting. To solve these problems, the installation of a vacuum sphere system is planned which will be shared by the CHSTF and the Test Cell #2 Direct-Connect Combustor Facility. This project is in the final design phase and is earmarked for 1988-89 construction funds.

The new vacuum sphere will provide the needed reduced back pressure capability to allow flight condition simulation for engine testing in the CHSTF with altitude simulation from 30,000 to 120,000 feet and run times of 20 seconds. In addition, the vacuum sphere will eliminate the high air flow required by the present air ejector and effectively uncouple the test cell from the other facilities with which it competes for air.

Description. - The facility will be connected to the 70-foot vacuum sphere by approximately 200 feet of 48-inch vacuum pipe line. A remotely operated butterfly valve will be installed in the vacuum pipe near the sphere entrance. A remotely operated butterfly valve will also be installed at the facility/sphere suction connection in the test cell. The vacuum sphere and connecting piping will be evacuated by a four stage condensing steam ejector discharging to atmosphere through an exhaust silencer. An existing 6-inch steam line operating at 350 psig and 26,000 pounds per hour will supply the steam ejector. Condenser water will be supplied by an existing cooling water system. Since the system will be shared with Test Cell #2, each facility will be provided with a remotely operated key-interlocked isolation butterfly valve and remotely operated atmospheric purge valve. Piping from the test cells to the sphere will be run from the exhaust end of the test cells up, over, and across the roof to the sphere.

The schematic sketch shown in Figure 14 depicts the facility hardware/ducting arrangement in the test cell. Exhaust ducts associated with the facility air ejector will be removed (see Fig. 5) and new ducting will be installed as shown in Figure 14. For safety considerations, a small amount (about 5 lbs. per sec.) of continuously flowing purge air is required through the tunnel. During this purge flow, the sphere will be isolated from the tunnel by the 48-inch butterfly valve near the test cell ceiling. The 30-inch duct, with its butterfly valve open, directs this purge flow to the vertical exhaust duct to the atmosphere. The purge flow sets up an ejector effect, as in the present facility air ejector configurations, to create air flow in the test cell to "bathe" the entire hardware of the facility. This action is assisted by a fan installed in the test cell ceiling intake tower; the main function of the fan, however, is to provide test cell bathing air flow during tunnel flow to the vacuum sphere.

The vacuum sphere system is to be installed so that, if required, the present air ejector system could be reinstalled. That is, the 48-inch isolation butterfly valve at the test cell ceiling would isolate the vacuum sphere and the new ducting shown in Figure 14 would be replaced with the present configuration that was shown in Figure 5.

Operational Conditions. - The proposed vacuum sphere system will allow Mach 3.5 to 6 flight conditions (without precompression) at altitudes from 30,000 to 120,000 feet to be simulated as represented by the hatched area of Figure 15. The low altitude and low Mach number simulation (high dynamic pressure simulation) may require the optional return to the facility air-ejector configuration. This addition of the vacuum system will greatly enhance the Combustion-Heated Scramjet Test Facility capability to better serve the NASA and national ramjet/scramjet community.

Relationship to NASA-Langley Scramjet Test Complex. - The enhanced capabilities of the Combustion-Heated Scramjet Test Facility represents both an improvement to the facility itself and also contributes to the enhancement of the overall capability of the NASA-Langley Scramjet Test Complex.[10] This complex includes the Langley 8-Foot High Temperature Tunnel (8-FT HTT) which is presently part of the Aerothermal Loads Complex. By 1989 the facility will have an oxygen replenishment system and new facility nozzles.[11] These enhancements give this tunnel the capability of engine tests at Mach 4, 5, and 7 flight conditions (8-foot diameter nozzle exits) of large-scale engine models (about 20 x 28 inches), a cluster of multiple engine modules, or engine models that have full nozzle expansion surfaces. The Arc-Heated Scramjet Test Facility (AHSTF) now has a much expanded test capability[8] with two different nozzles (M = 4.7 and 6.0) and is part of the NASA-Langley Scramjet Test Complex. Upon completion of the modifications to the 8-FT HTT and those discussed herein for the CHSTF along with the extended test capability of the AHSTF, the Scramjet Test Complex at Langley will be unique in the western world. The flight simulation envelope of the test gas heaters (without precompression) for the 8-FT HTT and the AHSTF are shown in Figure 16 superimposed on the operational envelope of the CHSTF that was presented in Figure 15. The potential operational envelope of this complex would extend over a flight Mach number range from 3.5 to 7. The lower Mach number limit shown for each of these facilities represents the lowest Mach number nozzle available for each facility.

Concluding Remarks

A test facility has been assembled at NASA Langley Research Center to provide the capability for subscale air-breathing ramjet/scramjet engine tests. Hydrogen burned in air with oxygen replenishment yields simulated air with the proper oxygen content for combustion tests. The present configuration is a free-jet wind tunnel with a three-dimensional Mach 3.5 nozzle (nominal 13-in-square exit) and an exhaust air ejector for simulation of Mach 4 flight conditions at the inlet of an airframe-integrated scramjet. The facility is designated the Combustion-Heated Scramjet Test Facility (CHSTF).

Details of the CHSTF were described herein and modifications to enhance the test capability range from subsonic/transonic and Mach 3.5 to 5.5 were discussed. This facility allows relatively inexpensive subscale test of ramjet-type engines to be conducted.

References

1. Hypersonic Research Engine Project - Phase II Structures and Cooling Development Final Technical Data Report. NASA CR-112087, May 1972.

2. Hypersonic Research Engine Project - Phase II Aerothermodynamic Integration Model Development Final Technical Data Report. NASA CR-132654, May 1975.

3. Hypersonic Research Engine Project - Phase II Aerothermodynamic Integration Model (AIM) Test Report. NASA CR-132655, May 1975.

4. Northam, G. Burton and Anderson, G. Y.: Supersonic Combustion Ramjet Research at Langley. AIAA-86-0159, January 1986.

5. Guy, Robert W.; Pinckney, S. Zane; and Andrews, Earl H., Jr.: Summary of NASA Langley's Scramjet Engine Tests. 1985 JANNAF Propulsion Meeting, San Diego, California, April 9-12, 1985. (Conf.)

6. Russin, William Roger: Performance of a Hydrogen Burner to Simulate Air Entering Scramjet Combustors. NASA TN D-7576, 1974.

7. Guy, Robert W.; Torrence, Marvin, G.; Sabol, Alexander P.; and Mueller, James N.: Operating Characteristics of the Langley Mach 7 Scramjet Test Facility. NASA TM-81929, 1981.

8. Thomas, Scott R. and Guy, Robert W.: Expanded Operational Capabilities of the Langley Mach 7 Scramjet Test Facility. NASA TP-2186, 1983.

9. Andrews, Earl H., Jr.; Torrence, Marvin G.; Anderson, Griffin Y.; Northam, G. Burton; and Mackley, Ernest A.: Langley Mach 4 Scramjet Test Facility. NASA TM-86277, 1985.

10. Andrews, Earl H.: Scramjet Test Complex. Langley Aerospace Test Highlights, NASA TM-87703, p. 34, 1985.

11. Kelly, H. N. and Wieting, A. R.: Modification of NASA Langley 8-Foot High Temperature Tunnel to Provide a Unique National Research Facility for Hypersonic Air-Breathing Propulsion Systems. NASA TM-85783, May 1984.

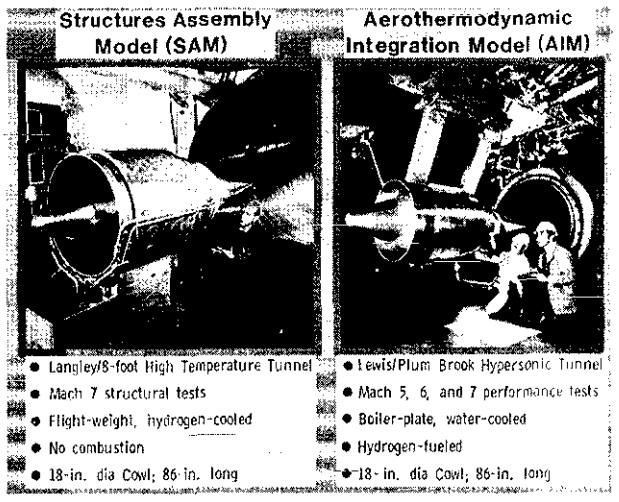


Fig. 1. NASA-Langley Hypersonic Propulsion; Hypersonic Research Engine Project, 1965 to 1974.

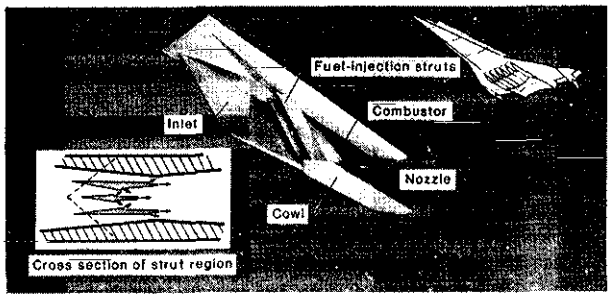
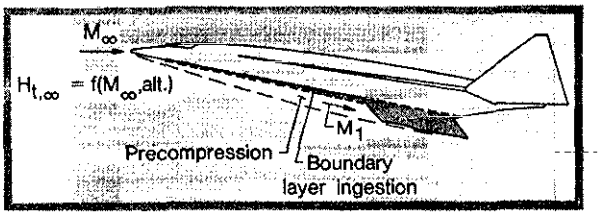
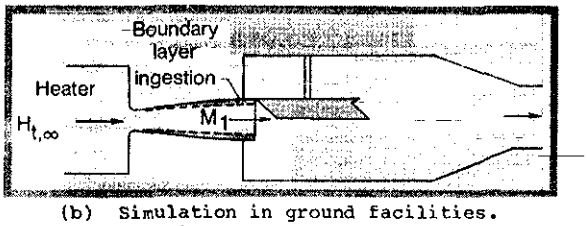


Fig. 2. NASA-Langley Hypersonic Propulsion; Airframe Integrated Concept, 1970 to present.



(a) Flight conditions.



(b) Simulation in ground facilities.

Fig. 3. Flight simulation for engine tests.

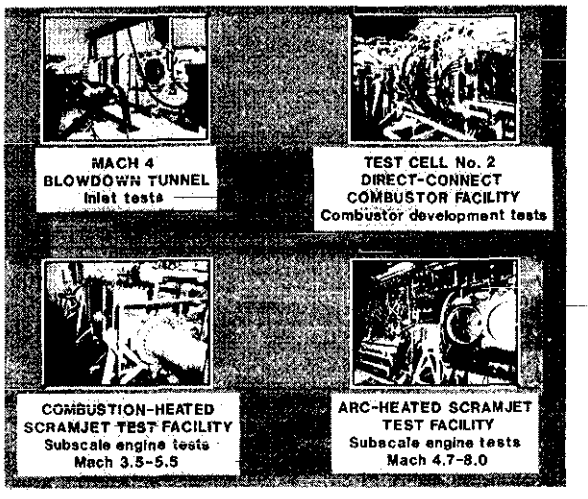


Fig. 4. Hypersonic Propulsion Branch's Engine and Component Test Facilities.

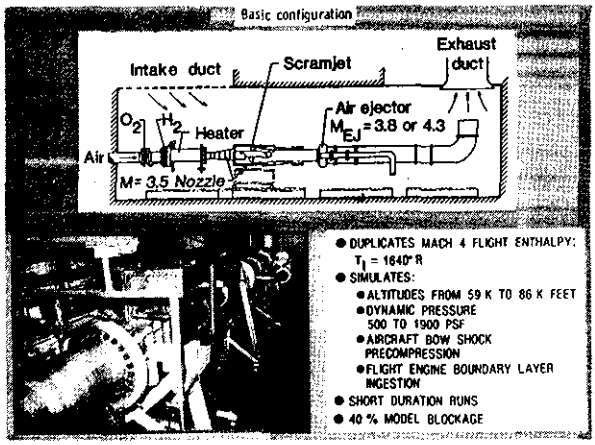


Fig. 5. Combustion-Heated Scramjet Test Facility basic configuration.

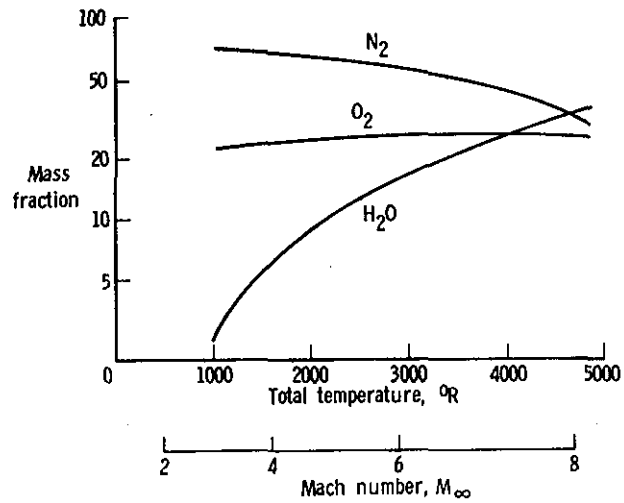
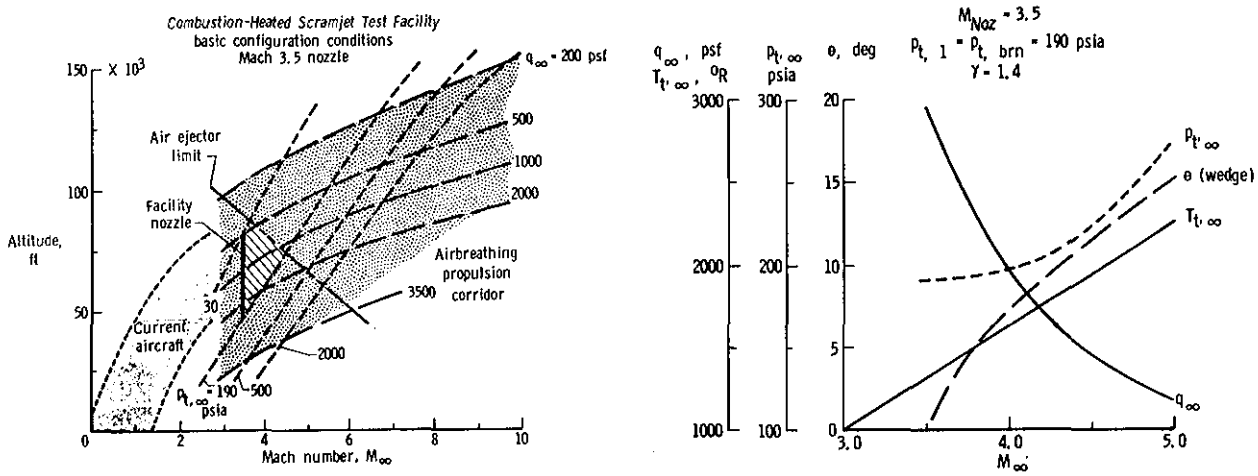


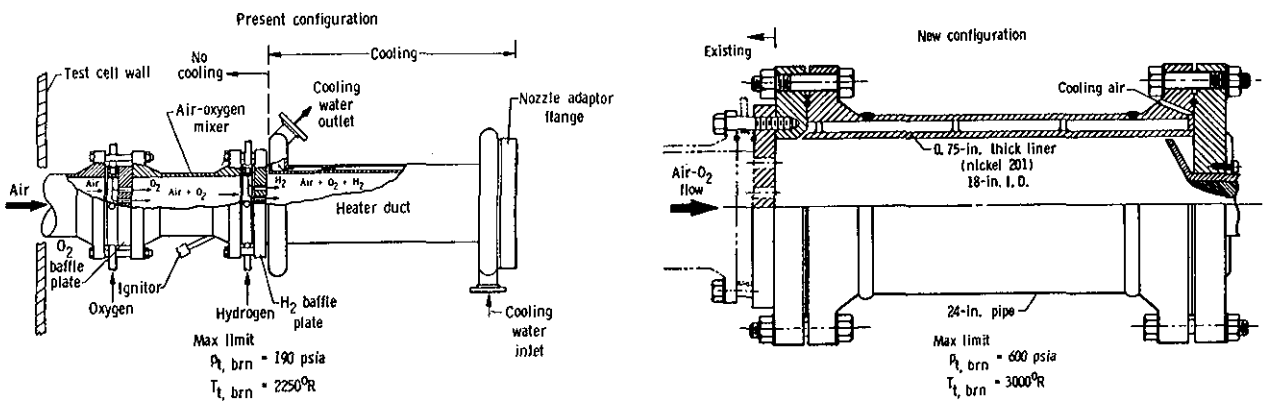
Fig. 6. Test gas flow constituents; air-hydrogen combustion with oxygen replenishment.



(a) Flight/test envelope; no precompression.

(b) Variable Mach number simulation.

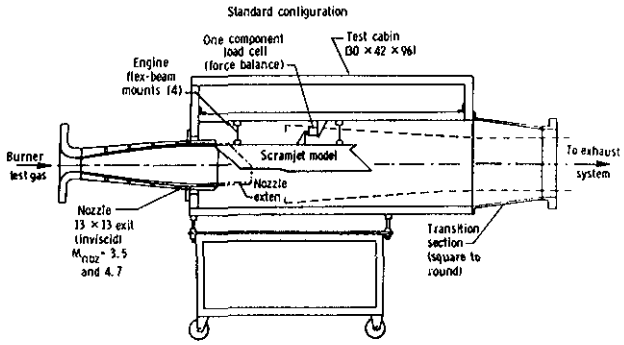
Fig. 7. Facility conditions of basic configuration.



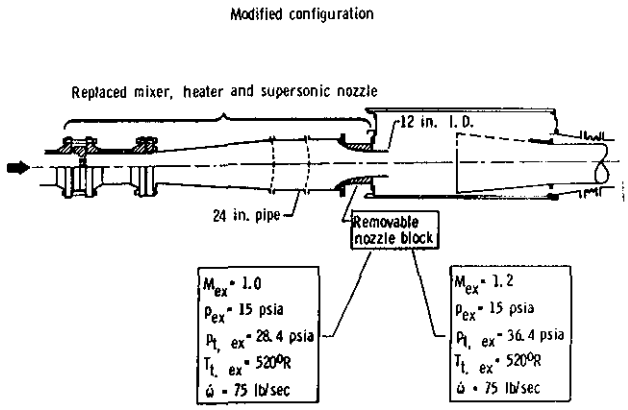
(a) Water cooled.

(b) Heatsink liner.

Fig. 8. Test gas heater concepts.

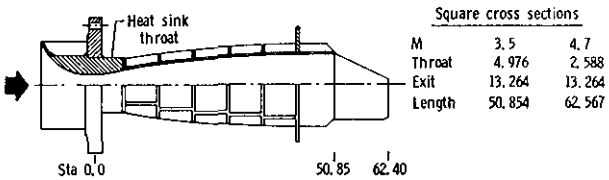


(a) Supersonic test setup.



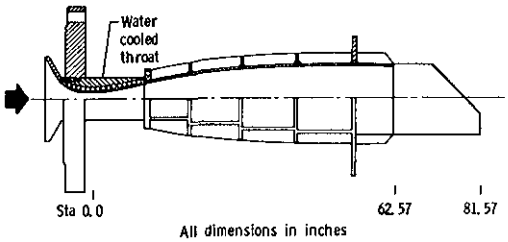
(b) Sub/transonic test setup.

Fig. 9. Facility configurations.



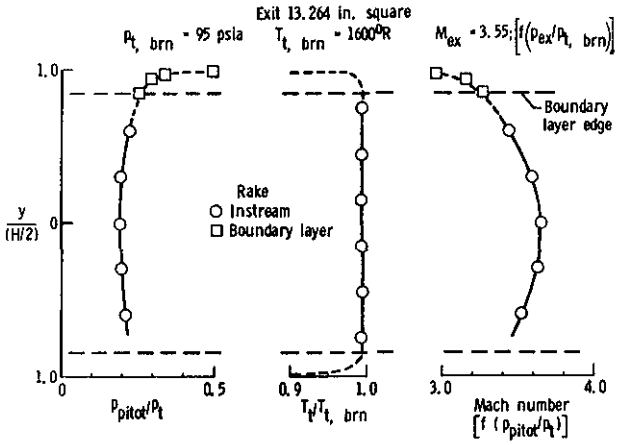
Square cross sections		
M	3.5	4.7
Throat	4.976	2.588
Exit	13.264	13.264
Length	50.854	62.567

(a) Mach 3.5.

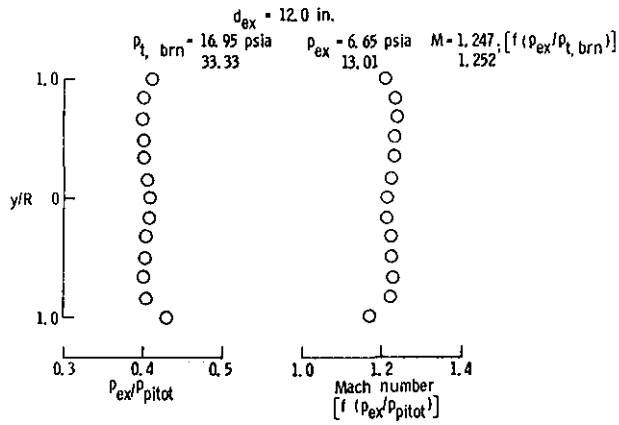


(b) Mach 4.7.

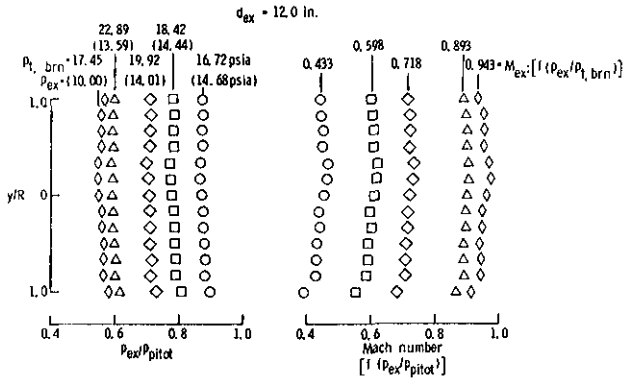
Fig. 10. Facility supersonic nozzles.



(a) Supersonic; M = 3.5.



(b) Transonic; M = 1.2.



(c) Subsonic; M = 0.5, 0.8, 1.0.

Fig. 11. Nozzle exit flow profiles.

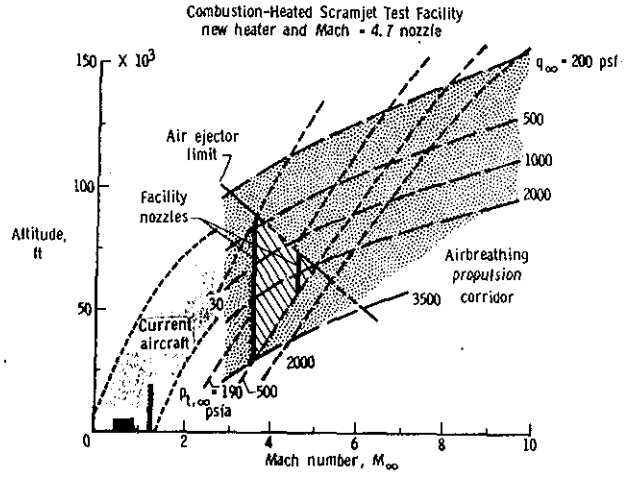


Fig. 12. Expanded flight/test envelope; no precompression.

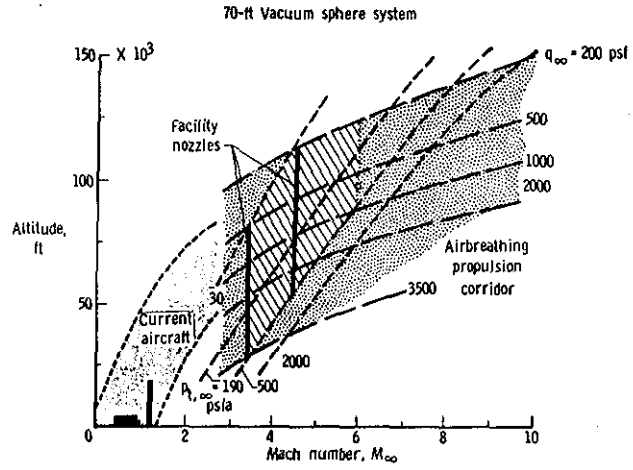
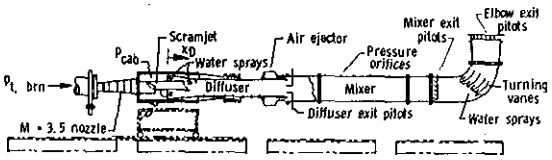
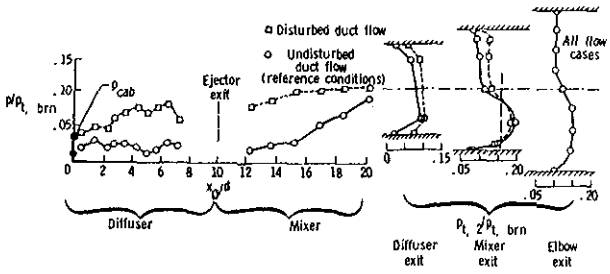


Fig. 15. Facility full potential flight/test envelope; no precompression.



(a) Facility duct and instrumentation layout.



(b) Pressure distributions and profiles.

Fig. 13. Exhaust duct flow; supersonic setup.

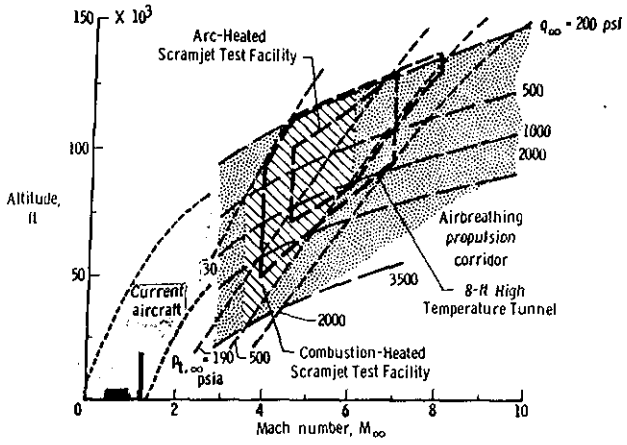


Fig. 16. Scramjet Test Complex Envelope; no precompression.

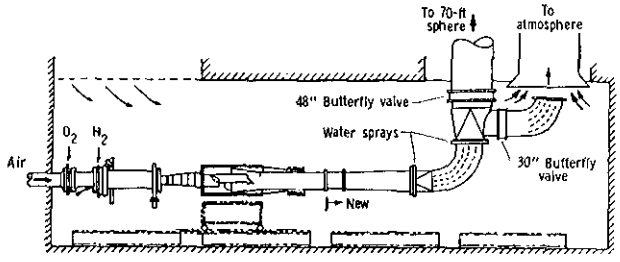


Fig. 14. Planned 70-foot vacuum sphere system.