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HYPERSONIC PROPULSION RESEARCH

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INTRODUCTION

The NASA Langley Research Center has been conducting hypersonic propulsion research since the 1960's. In 1965, the Hypersonic Research Engine (HRE) project was undertaken to demonstrate internal performance of a scramjet with the X-15 as a test platform. The X-15 program was terminated in 1968 before the scramjet was ready for flight testing. However, two regeneratively cooled models were constructed and the internal performance of the engine was demonstrated in ground tests at the Lewis Research Center's Plum Brook test facility. Following the HRE program, the Hypersonic Propulsion Branch developed a modular airframe-integrated scramjet concept. This engine concept yields higher installed performance by reducing drag associated with mounting the engine. The airframe-integrated scramjet uses the vehicle forebody as a part of the compression surface, attaches engine modules directly to the underside of the vehicle, and uses the aft end of the airframe as a nozzle expansion surface. The development of technology for the modular airframe-integrated scramjet has been the focus of this research for the past several years.

As part of this research, a variety of inlet concepts have been explored and characterized. The emphasis of the inlet program has been the development of the short (light weight), fixed geometry, side-wall-compression inlets that operate efficiently over a wide Mach number range. As hypersonic combustion tunnels were developed, programs to study the parameters controlling fuel mixing and combustion with single and multiple strut models were conducted using direct connect test techniques. These various tests supported the design of subscale engine test hardware that integrated inlet and combustor technology and allowed the study of the effect of heat release on thrust and combustor/inlet interaction. A number of subscale (8-in.-high by 6-in.-wide) engine tests have demonstrated predicted performance levels at Mach 4 and Mach 7 simulated flight conditions.

This paper summarizes a few of the highlights from this research program.

ARTIST CONCEPT OF A SINGLE-STAGE-TO-ORBIT VEHICLE

An artist's concept of the current focus of much of the nation's hypersonic research is shown in figure 1. Since February 1985 when President Reagan announced that the country would build a National Aero-Space Plane (NASP) there has been an accelerated program at government and industrial laboratories to access and define the technology advances required to make single stage to orbit possible. NASA has teamed with the Defense Department, industry, and the universities to develop the technologies necessary to design and construct an experimental aircraft needed to demonstrate hypersonic flight by using airbreathing propulsion. The proposed research airplane, designated the X-30, is not an operational vehicle but a test bed to demonstrate the various technologies required for the design of three classes of vehicles - hypersonic transports that would cruise at Mach 5 or higher, space transports, and transatmospheric military vehicles.

Figure 1. - Artist concept of a single-stage-to-orbit vehicle.

AIRFRAME-INTEGRATED SUPERSONIC COMBUSTION RAMJET

Figure 2 shows the Langley concept for an airframe-integrated supersonic combustion ramjet (scramjet). The engine modules are mounted on the underside of the vehicle and use the vehicle forebody as part of the nozzle compression. The compression is completed by the inlet sidewalls and the fuel injection struts. The use of fuel injection struts to complete the compression shortens the inlet and provides locations for distributing the fuel in the incoming compressed air. The relative amount of parallel and transverse injected fuel is used to control the mixing and thus the heat release distribution required for the desired performance as a function of flight Mach number. The modular design allows testing of a single engine in a smaller facility than that required if the total propulsion system had to be tested.

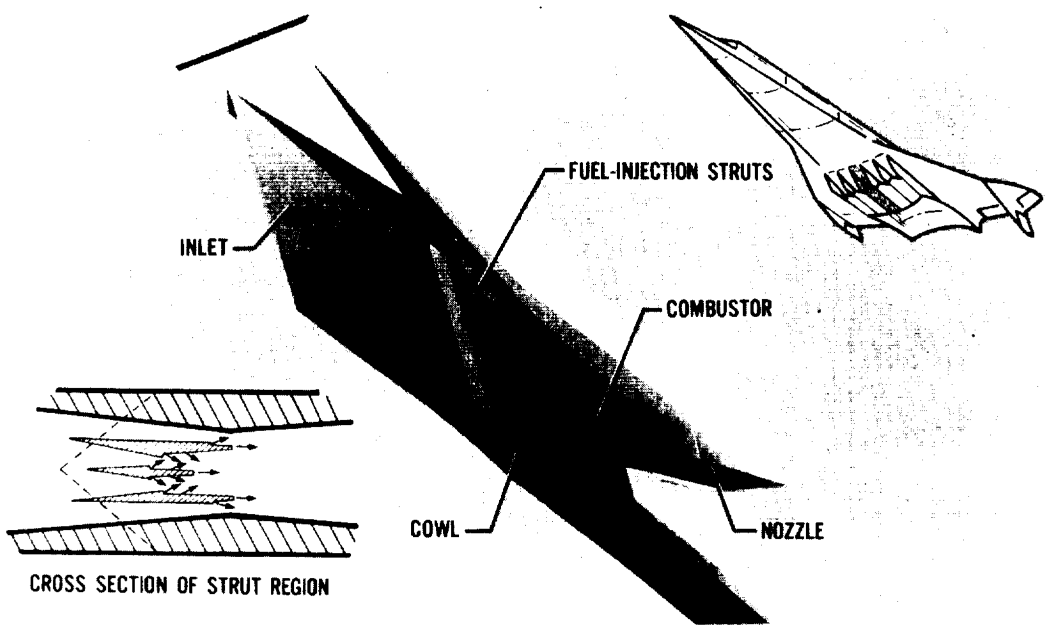


Figure 2. - Airframe-integrated supersonic combustion ramjet.

HYPERSONIC INLET DESIGN

Several inlet designs (fig. 3) have been constructed and evaluated at Mach 3 to 6. Currently the testing range is being extended to Mach 18 to support the NASP program. The three strut inlet with a swept combustor was the first inlet of this class completely characterized - mass capture, pressure recovery, contraction ratio, and throat Mach number as a function of free stream Mach number. Computational fluid dynamics is becoming more mature and is being used to design and evaluate alternate inlet designs like the rectangular to circular inlet and the two strut inlets shown. The three-module inlet model is being used to determine the effect of an inlet unstart on the performance of the remaining started inlets.

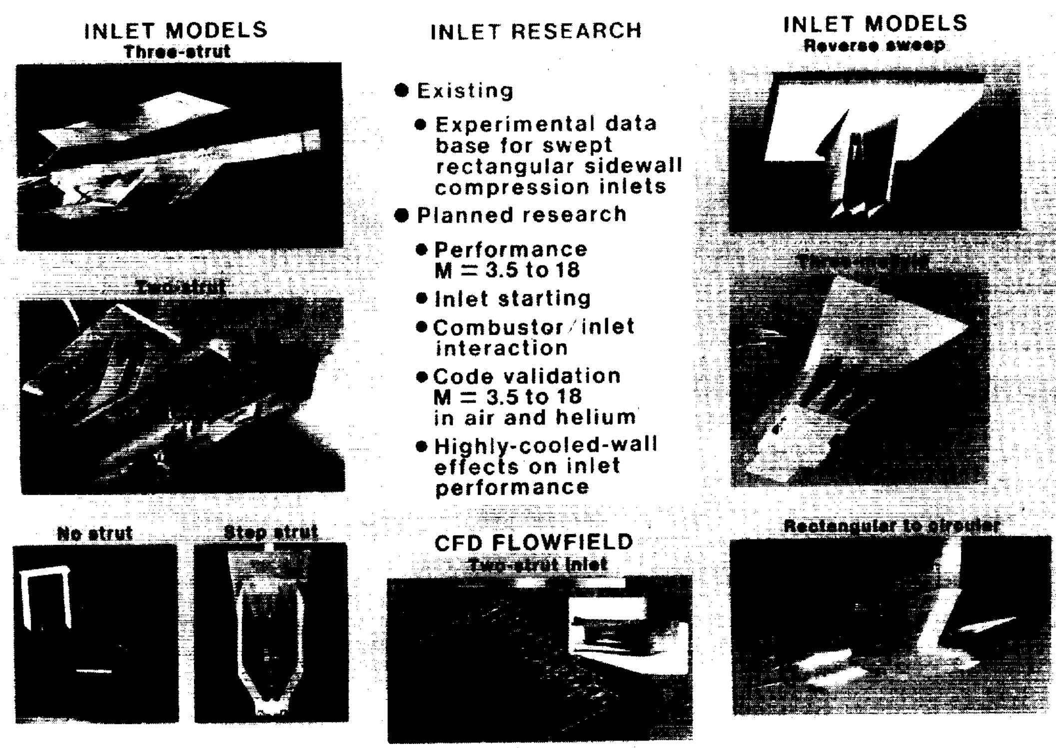


Figure 3. - Hypersonic inlet design.

SCHEMATIC OF DIRECT CONNECT TEST APPARATUS

In order to reduce the size of the facility required to test a scramjet engine, direct connect tests of the combustor are conducted by simulating the inlet flow into the combustor region. Typical direct connect test hardware are shown in figure 4. This hardware is being used to determine combustor geometry effects on scramjet performance. The figure shows some of the geometry variables being investigated.

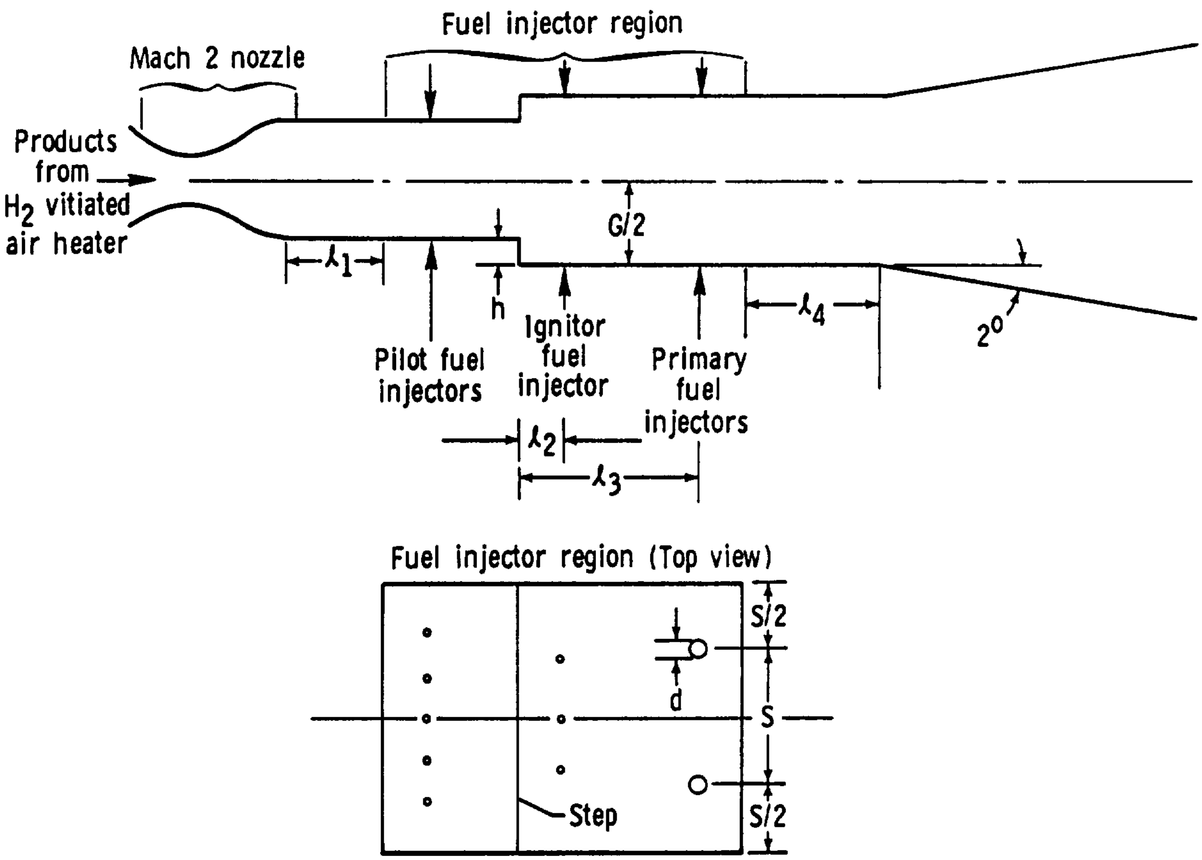


Figure 4. - Schematic of direct connect test apparatus.

DUAL-MODE COMBUSTOR OPERATION

Using direct connect test hardware, the two different pressure distributions shown in figure 5 were obtained by varying the total temperature and the fuel injection location with the Mach number at the combustor entrance held constant at 1.7. The Mach 4 distribution with a major portion of the duct operating subsonic had most of the fuel injected parallel to the flow. For the Mach 7 total temperature data, most of the fuel was injected perpendicular to the air flow and much of the combustion occurred at supersonic conditions.

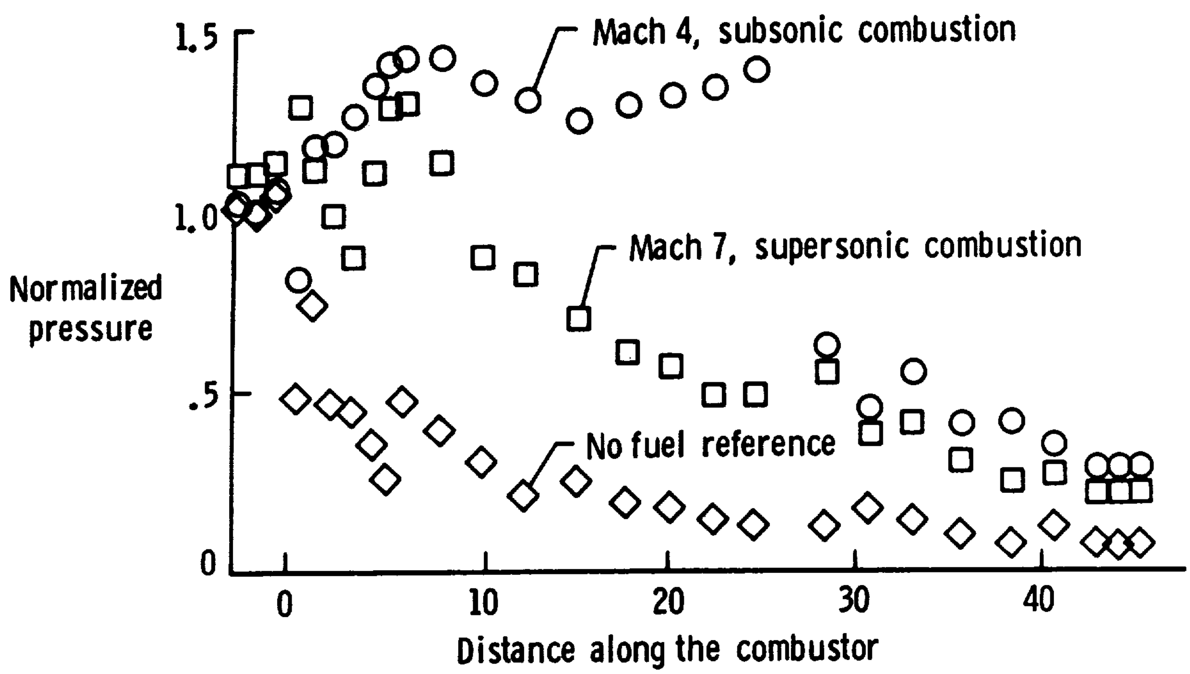


Figure 5. - Dual-mode combustor operation.

UPSTREAM INTERACTION LIMIT

Figure 6 shows the maximum equivalence ratio that can be injected in the direct connect hardware as a function of total temperature. The Mach number entering the combustor from the facility nozzle was 2.0. Adding a constant area section 2.2 duct heights downstream of the fuel injector region reduced the amount of fuel that could be added without causing a pressure rise at the combustor entrance station. The duct divergence downstream of the fuel injectors of the constant area section was 2° on the top and bottom walls.

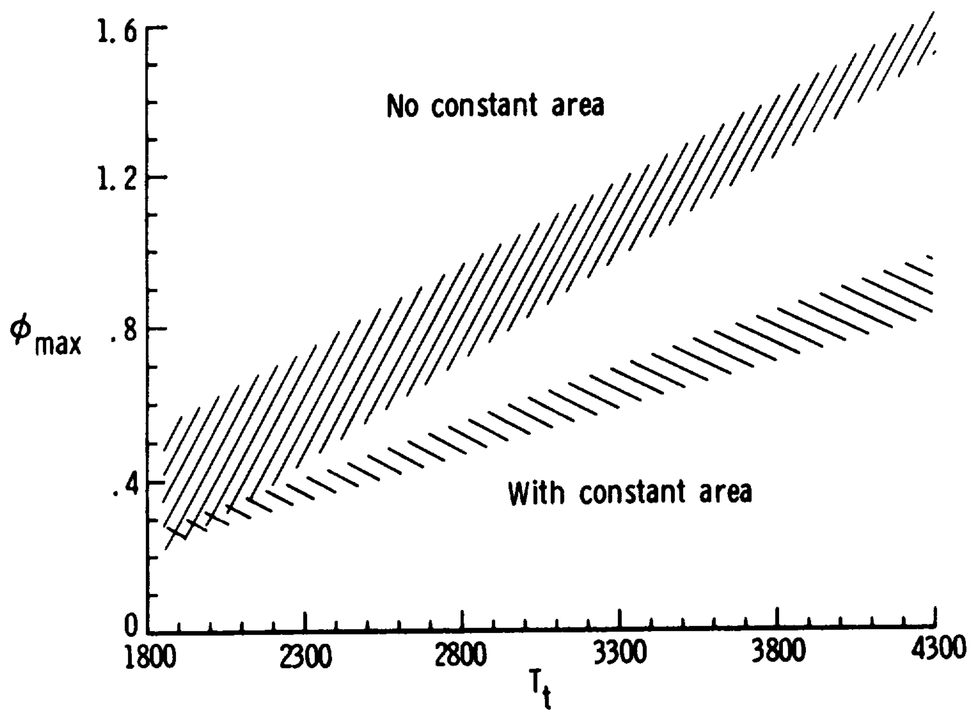


Figure 6. - Upstream interaction limit (effect of constant area combustion).

PRESSURE DISTRIBUTIONS FOR PILOTED TESTS

A number of tests have been conducted to explore ignition and flameholding techniques for hypersonic combustion. At low flight speeds, the total temperature is not high enough for autoignition; at high speeds the reduced residence time may make flameholding difficult. One ignition technique explored was the use of an electrically driven plasma jet operating on a hydrogen/argon mixture. Figure 7 shows the pressure distributions for a Mach 2 combustor operating below the autoignition temperature (total temperature 1400 °R). When 50 percent hydrogen/argon mixture was used in the plasma jet operating at less than 1 kW, the pressure distribution indicated considerable pressure rise. However, when the torch was operated on all argon the pressure distribution was similar to the no fuel distribution. These results indicate that the hydrogen atoms generated in the plasma, not the thermal heating of the gases were the ignition source.

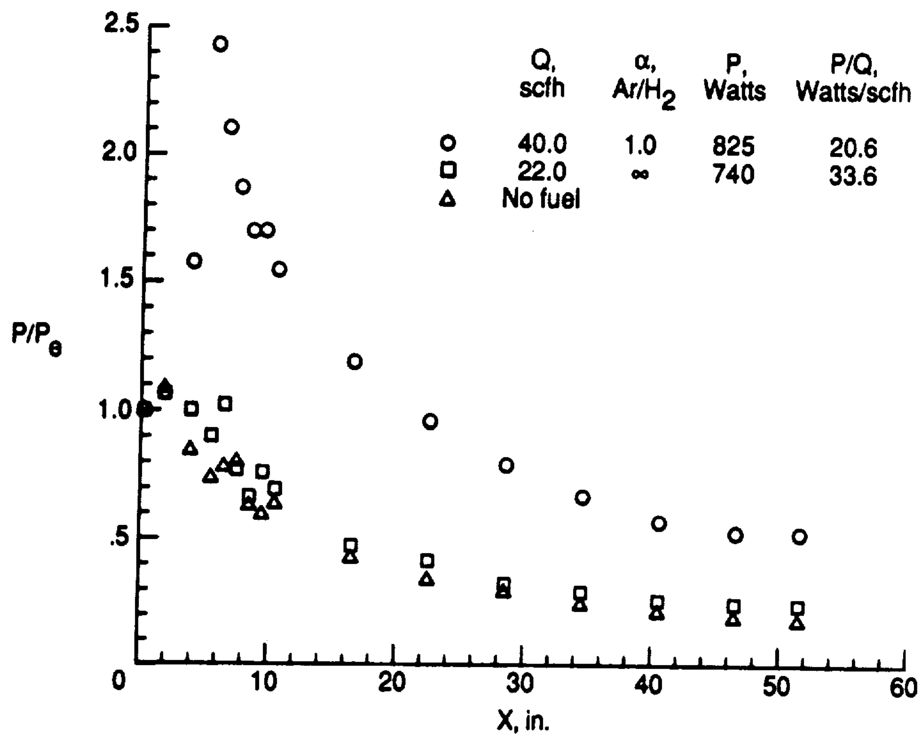


Figure 7. - Pressure distributions for piloted tests (AR/H_2 and argon plasmas, T_t = 1400 R, ϕ = 0.28).

COMPARISON OF CFD AND EXPERIMENTAL MIXING RESULTS

The use of large computers for the calculations of complex flows is a new tool available to the scramjet designer. In order for these codes to be used with confidence, validation tests with well defined boundary conditions must be conducted and compared with the corresponding computed results. Figure 8 shows that the result of one such experiment where a sonic transverse jet was injected into supersonic Mach 2.06 flow. The lines are the calculated mass contours and the dots are the edge of the jet as defined by laser induced fluorescence of iodine. This favorable comparison indicates that the CFD code could be extended to other similar geometries with confidence. In fact, the code has been modified to include chemical reactions and similar reacting flow experiments are planned.

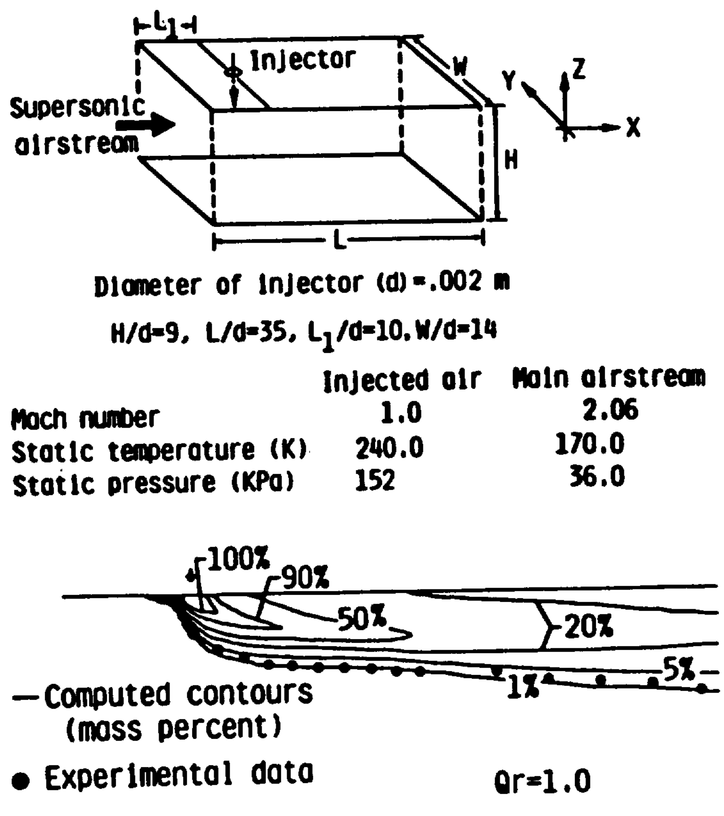


Figure 8. - Comparison of CFD and experimental mixing results.

SHOCK TUNNEL SUPERSONIC COMBUSTION

At high hypersonic speeds, there are no steady state facilities capable of simulating the total enthalpy and pressure encountered in flight. The best experimental simulation of hypersonic combustion is conducted in pulse facilities that operate on the order of 1/1000 of a second. Results from such tests are shown in figure 9 for Mach 9 and 16 simulated flight conditions. Again CFD calculations were used to predict the flow. The Mach 16 results were not predicted as well as the lower speed test results. At these higher speeds where ground tests facilities are not available, CFD will have to be used to predict performance for NASP.

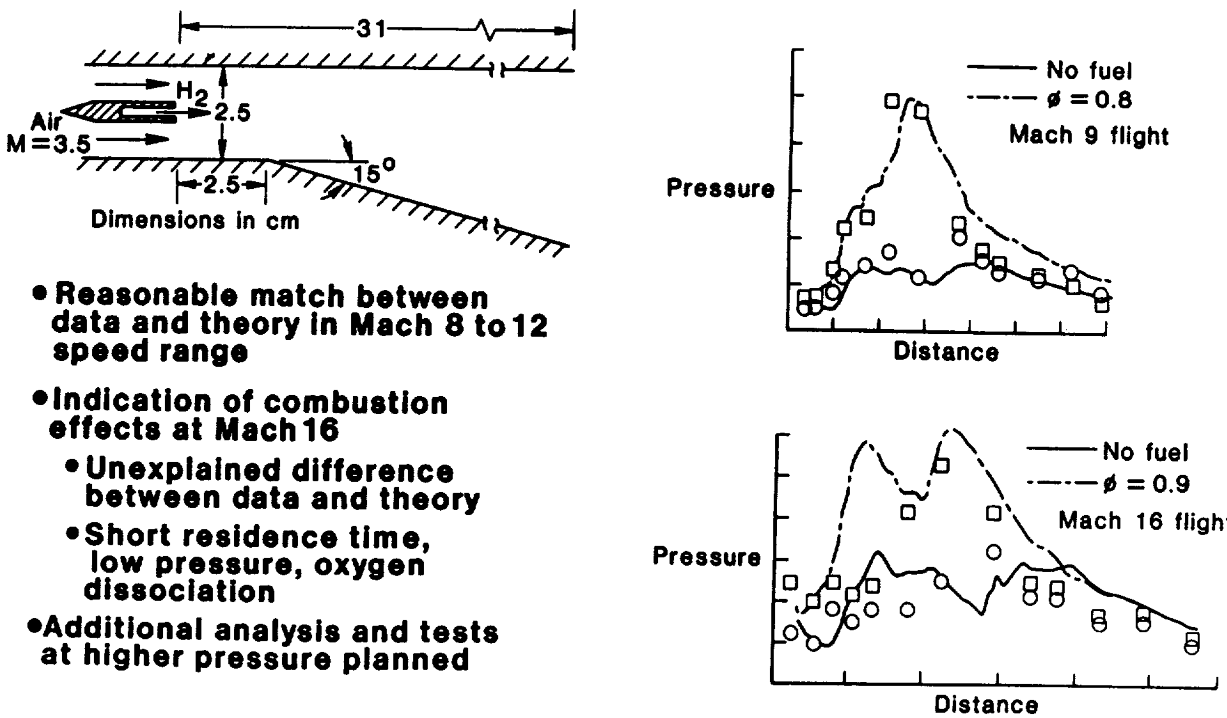


Figure 9. - Shock tunnel supersonic combustion (University of Queensland).

MULTISPECIES CARS SCHEMATIC

In order to validate CFD codes, instream measurements of temperature and species profiles are required. In reacting supersonic streams it is difficult to use sampling probes for the measurements due to shock effects. Nonintrusive optical techniques are being developed to replace probe measurements. One technique being used is Coherent AntiStokes Raman Scattering, CARS. A schematic of the CARS system is shown in figure 10. This system can measure temperature, oxygen, and nitrogen during a 10-nanosecond laser pulse at 10-Hz rate. The short pulse time allows measurements to be made in high speed turbulent flow.

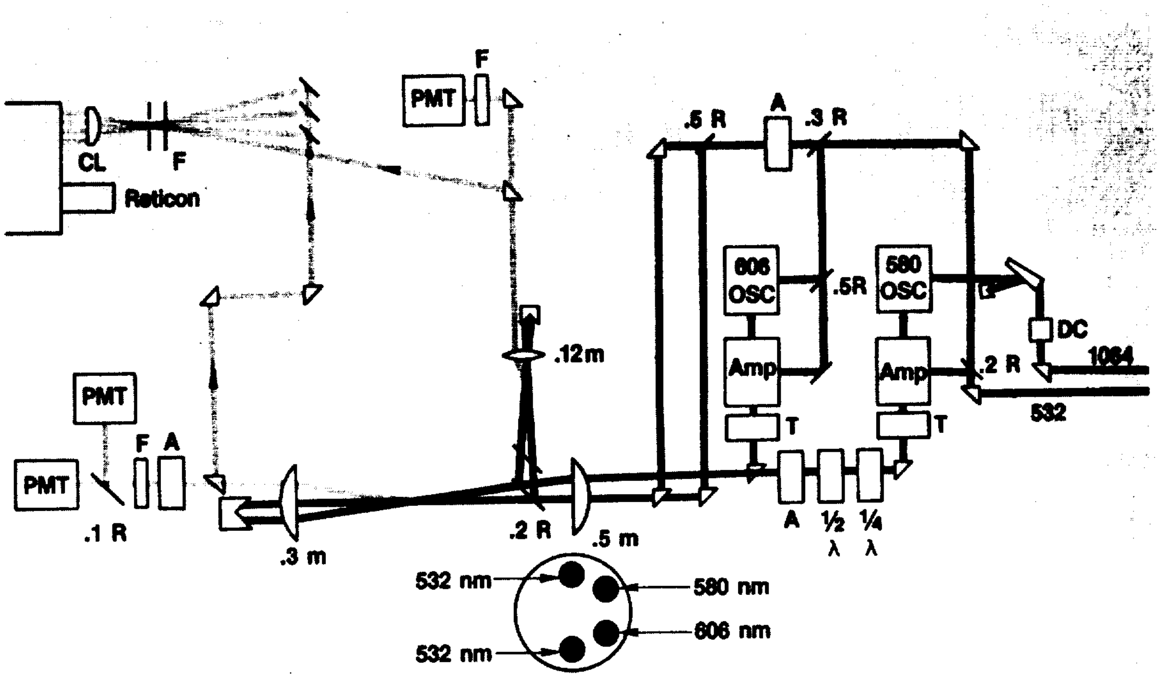


Figure 10. - Multispecies cars schematic.

SUPERSONIC COMBUSTION EXPERIMENTS

A simple coaxial-jet, supersonic combustion experiment has been developed for CFD validation. The sonic jet is injected into a Mach 2 high enthalpy simulated air stream generated by the combustion and expansion of hydrogen and air with oxygen make up. A cross section of the apparatus is shown in figure 11 along with data taken in the exit plane. On the data plots the dashed lines are the calculated means and the solid lines are the CARS measurements. Tests are underway to make similar measurements to map the radial and axial profiles with and without fuel injection. This experiment is also being simulated by using a reacting flow CFD code.

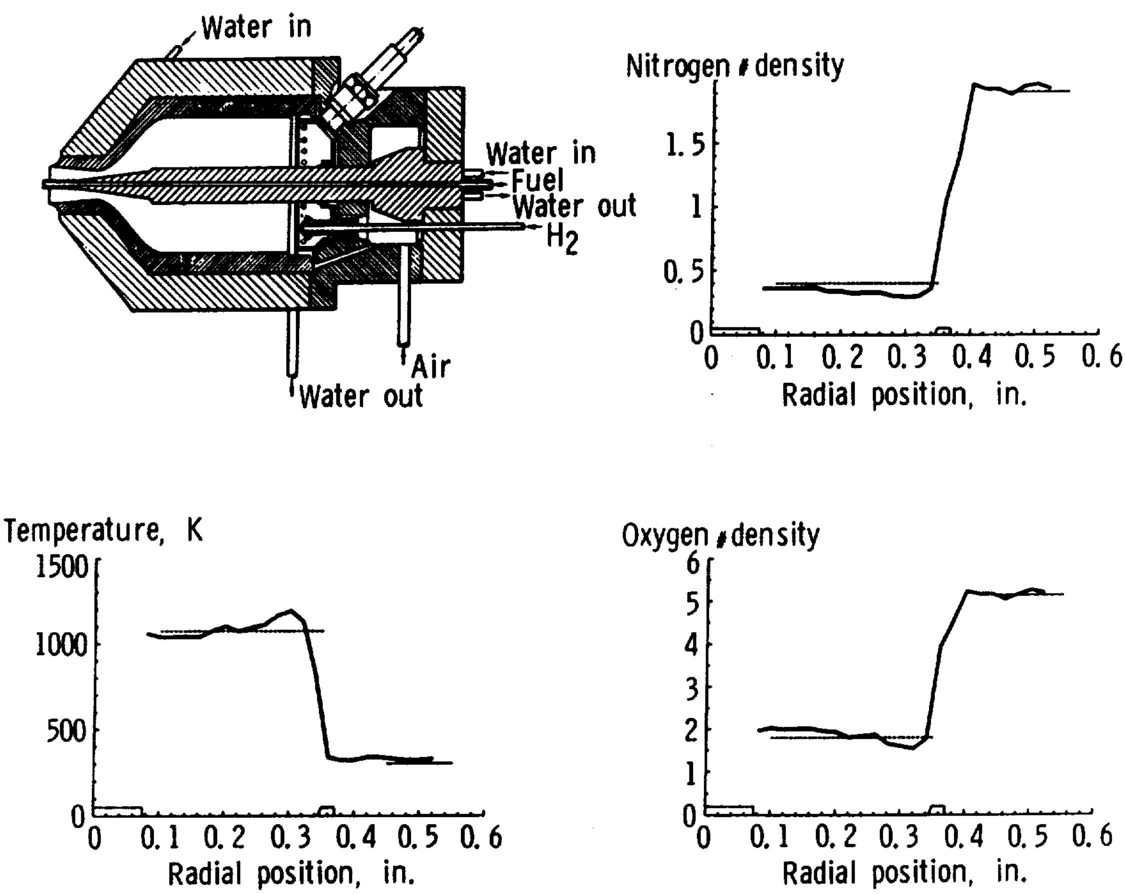


Figure 11. - Supersonic combustion experiments.

SIMULATION REQUIREMENTS FOR HYPERSONIC SCRAMJET COMBUSTOR

In order to demonstrate scramjet performance in ground test, large high enthalpy facilities are required. Figure 12 shows the total enthalpy required as a function of flight Mach number up to Mach 25 or near orbital speed. Also shown on the figure are the static temperatures, stagnation temperatures and the stagnation pressures required for proper simulation. The right most curve shows the flight path energy and the band shows the combustor flow conditions after the air is processed by the inlet. Even the combustor conditions rise rapidly with flight Mach number. In fact, hydrogen or air heaters can only be used to simulate flight speeds up to about Mach 8. Beyond this speed pulse facilities are required.

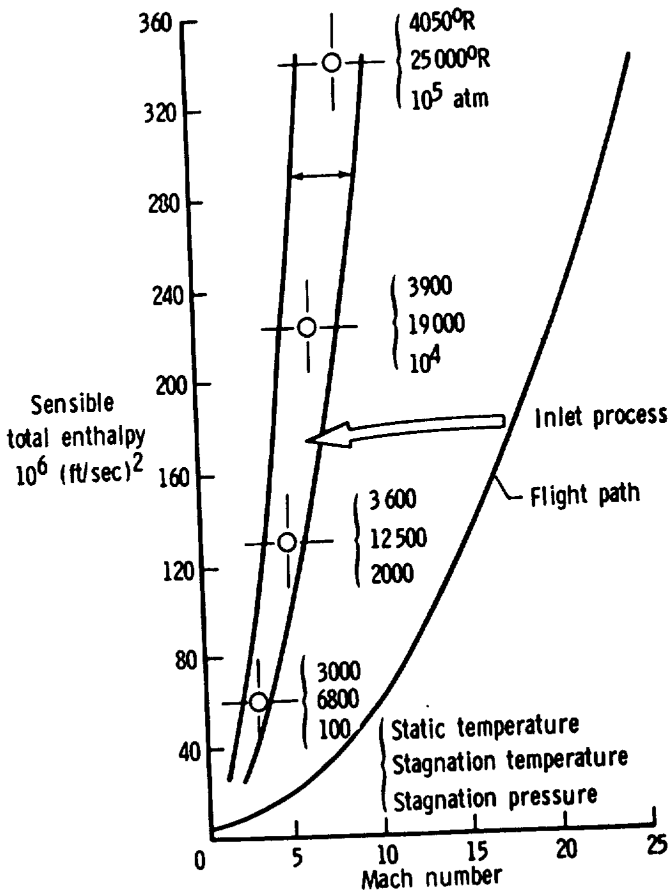


Figure 12. - Simulation requirements for hypersonic scramjet combustor tests.

SIMULATION CAPABILITIES OF OPERATIONAL AND NEAR-TERM FACILITIES

Figure 13 shows, on similar coordinates, the operational range of several combustion tunnels. Note that the T3 tunnel at the University of Queensland in Australia is the only facility that covers the higher Mach numbers. This facility has been used to produce flow simulating Mach 12.

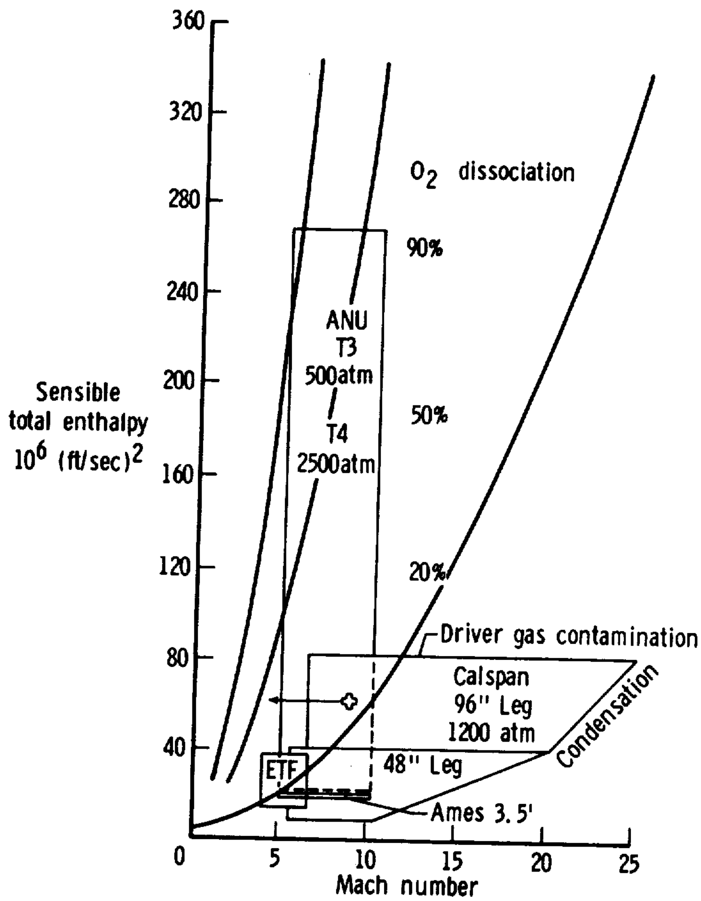


Figure 13. - Simulation capabilities of operational and near-term facilities.

EFFECTS OF VITIATION ON IGNITION TIME FOR STOICHIOMETRIC H_2 IN EQUILIBRIUM AIR

One of the problems associated with testing in high enthalpy ground test facilities is the fact that the energy is usually added to the flow by some mechanical, electrical, or combustion process that contaminates or dissociates the air. Engine test facilities that use combustion of hydrogen or hydrocarbons can produce simulated Mach numbers of about 7 with oxygen content maintained at atmospheric levels. When these facilities are used, the tests are said to be conducted in vitiated air. Figure 14 shows the effects of vitiation on the ignition delay of hydrogen as a function of static pressure and temperature. At low temperatures and pressures the effects become significant.

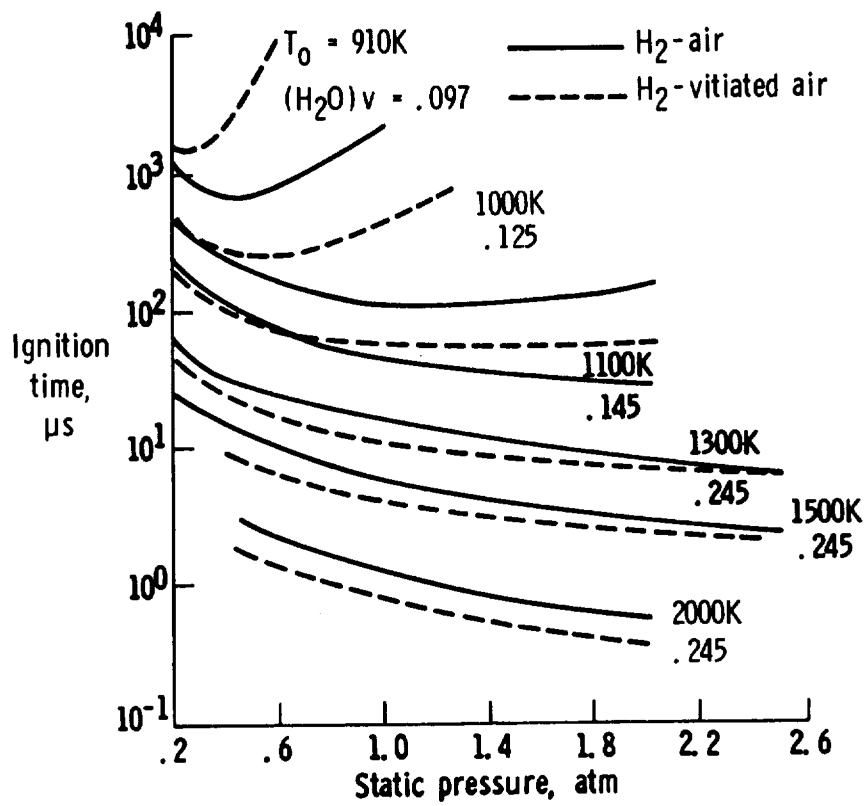


Figure 14. - Effects of vitiation on ignition time for stoichiometric H_2 in equilibrium air.

STRUTLESS PARAMETRIC ENGINE MODEL

In order to integrate the inlet and combustor technology, subscale engine tests are conducted in arc or vitiated scramjet test facilities at Langley from Mach 3.4 to 7 flight conditions. Figure 15 shows the strutless parametric engine that is being used to study the sensitivity of performance to engine geometric variables, fuel injection location, fuel level, contraction ratio and inlet sweep.

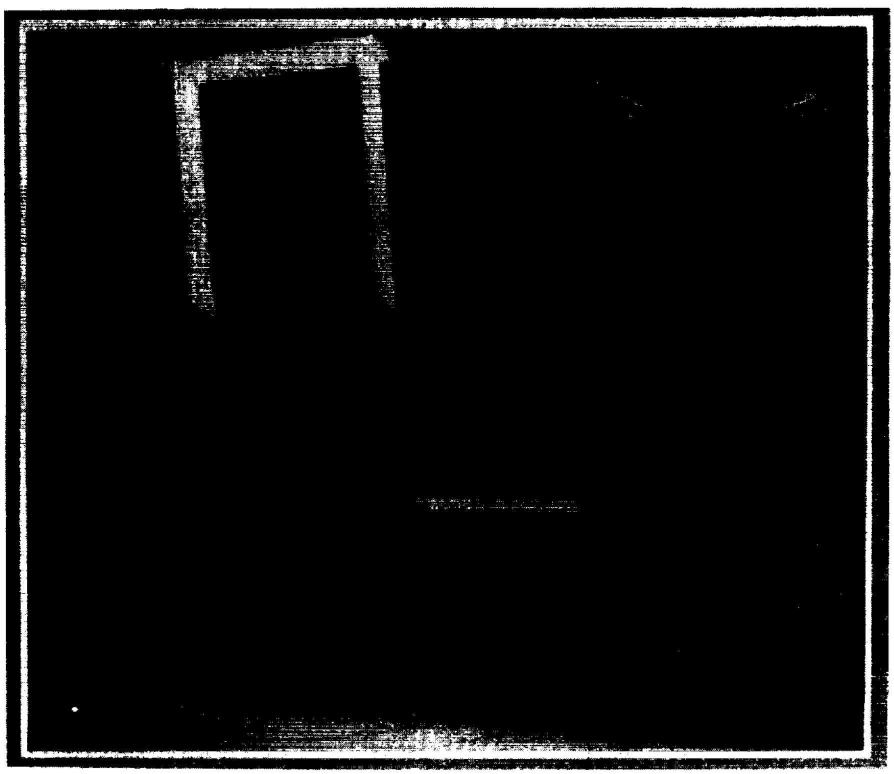
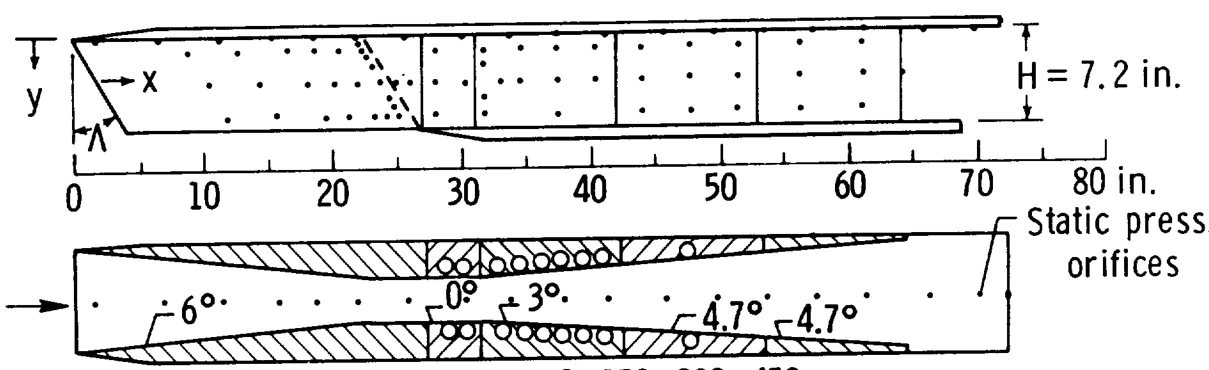


Figure 15. - Strutless parametric engine model.

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

STRUTLESS PARAMETRIC ENGINE MODEL DETAILS

The sketch in figure 16 shows the details of the strutless model. The sidewalls are adjustable and the segmented side wall locations can be independently moved to tailor the internal geometry. Fuel can be injected from a number of side wall locations. The engine is mounted on a thrust balance in the tunnel and has numerous pressure ports to measure wall pressures.



- Swept inlet; Λ = 0°, 15°, 30°, 45°
- Unswept combustor
- Variable position sidewalls
  - Geometric contraction ratio 4 to 7
  - Sidewall steps
- Variable position cowl
- Multiple fuel injection stations
  - Hydrogen fuel
  - Silane ignition
- In-stream fuel injection struts optional

Figure 16. - Strutless parametric engine model details.

MACH 4 ENGINE PERFORMANCE

Figure 17 shows typical pressure and force data obtained in Mach 4 tests of the strutless scramjet shown in the schematic. The peak pressure in the combustor increases as more fuel is injected into the engine and the pressure rise moves upstream closer to the inlet exit. However, no combustor-inlet interaction occurred since the combustion-induced pressure rise did not move onto the forward-facing sidewalls of the inlet.

The thrust curve shows a comparison of data from the strutless engine with that from an earlier version of a scramjet and with theory assuming mixing-controlled combustion. Solid symbols indicate that combustion of the primary hydrogen fuel was assisted by a pilot gas (silane/hydrogen) while the open symbols indicate that the fuel was all hydrogen. Agreement of the two sets of data (piloted and unpiloted) indicate that the wedges shown in the schematic were good flameholders. Agreement of the data with theory indicates that the combustion was mixing-controlled. And, finally, comparison of the data with that from the other scramjet shows the improvement in performance obtained by redesign.

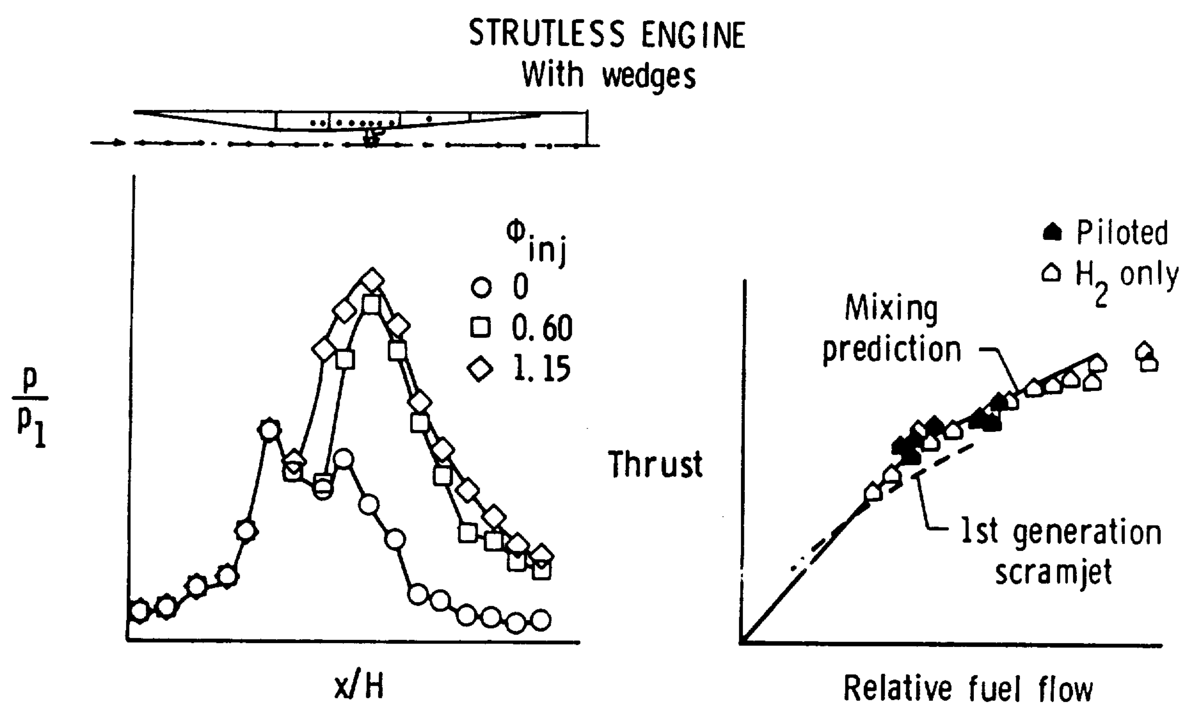
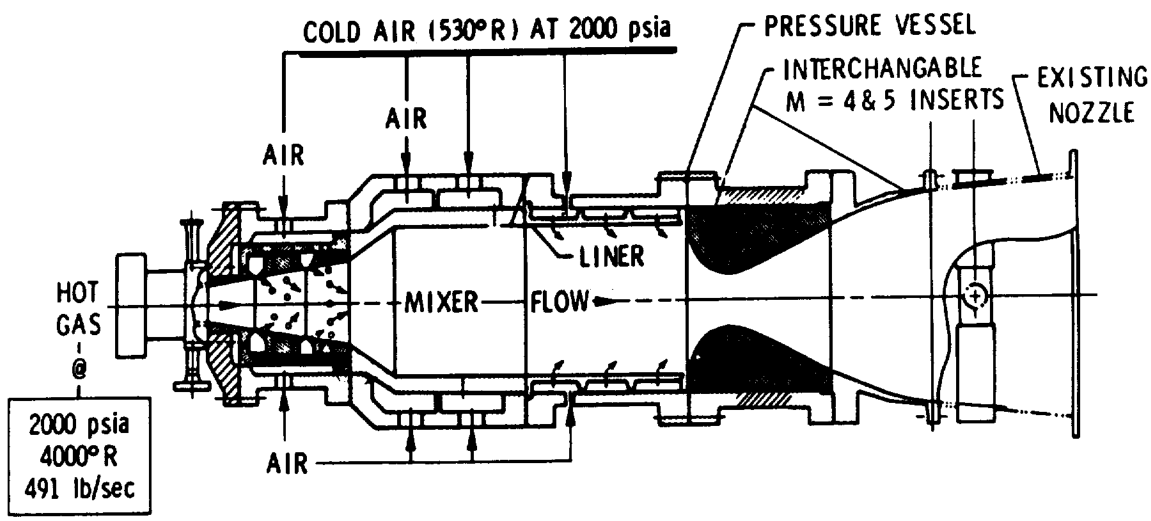


Figure 17. - Mach 4 engine performance.

MIXER/ALTERNATE MACH NUMBER NOZZLES FOR 8-FOOT HTT

The Langley 8-foot test diameter high temperature tunnel is being modified for propulsion testing. When completed this facility will allow testing of larger scale models and complete missile size scramjets. Figure 18 indicates the operating points at maximum total pressure of the 8-foot HTT with the Mach 4, 5, and 7 nozzles. A schematic is also shown of the mixer/throat section which is used with the Mach 4 and Mach 5 nozzles. When the Mach 7 nozzle is in service, the entire mixer/throat section (shown in section view) is replaced by the first portion of the Mach 7 nozzle. The hardware upstream and downstream of the cross-hatched section is used with all three nozzles.

During Mach 7 tests, no unheated air is added to the 4000 °R vitiated air exiting combustion heater. However, to obtain the lower Mach 4 and Mach 5 total enthalpies, unheated bypass air is added as shown in the schematic in a fashion similar to that employed in the Langley Arc-Heated Scramjet Test Facility.



MACH NUMBER	THROAT DIAMETER, in.	MIXER TOTAL PRESSURE, psia	MIXER TOTAL TEMPERATURE, °R	COLD AIR FLOW RATE, lb/sec	TOTAL FLOW RATE, lb/sec
4.0	28.5	234	1640	1436	1927
5.0	17.1	444	2350	597	1088
7.0	5.6	2000	4000	0	491

Figure 18. - Mixer/alternate Mach number nozzles for 8-foot HTT.

SCRAMJET DEVELOPMENT

As high speed propulsion matures and new larger facilities become available, testing will move from single module subscale testing to testing of larger scale and multi-engine modules in the 8 ft tunnel and then to flight testing (fig. 19). The larger or near full scale tests will build confidence in the mixing and flameholding models and will allow realistic engine cooling evaluations in flight type hardware. The multi mode test will be used to study the influence of one module on the inlet flow of an adjacent module and allow the study of an inlet unstart on the total engine performance.

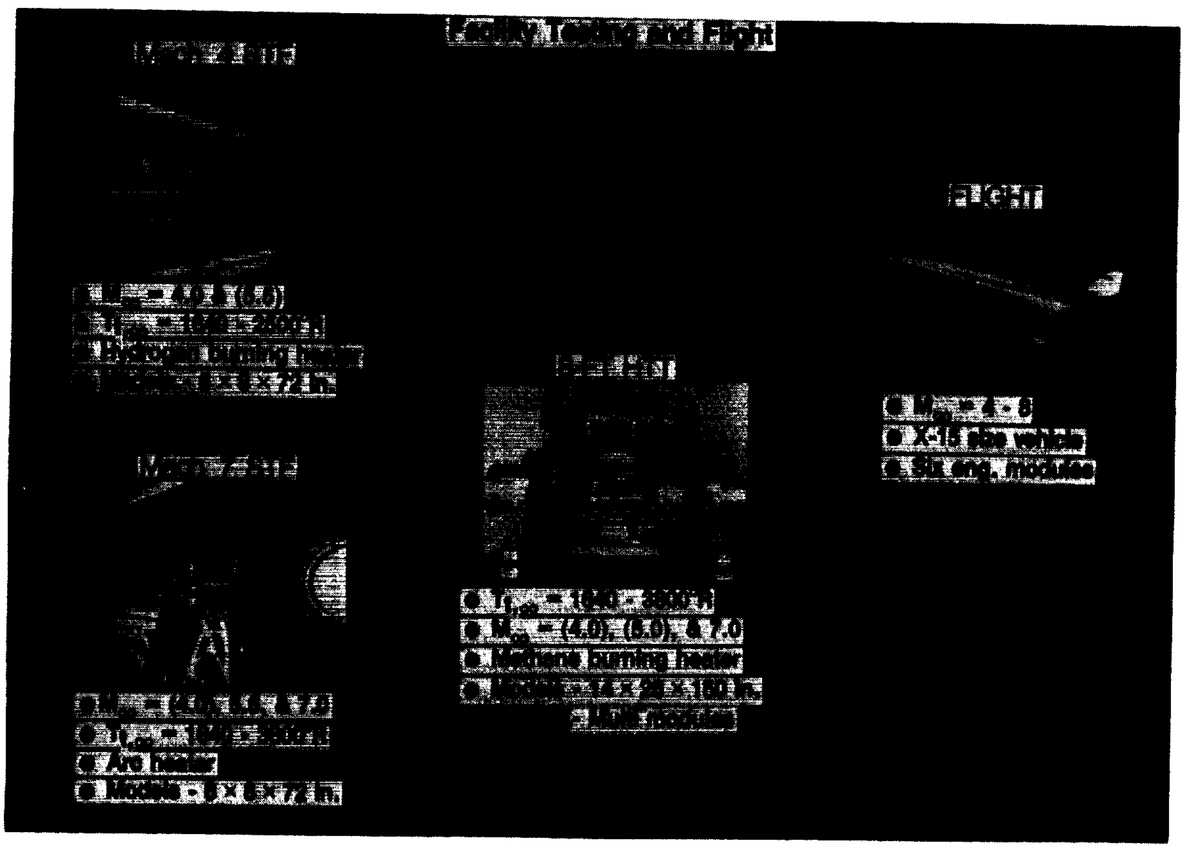


Figure 19. - Scramjet development.