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MODIFICATION OF NASA LANGLEY 8-FOOT HIGH
TEMPERATURE TUNNEL TO PROVIDE A UNIQUE
NATIONAL RESEARCH FACILITY FOR HYPERSONIC
AIR-BREATHING PROPULSION SYSTEMS

H. N. KELLY AND A. R. WIETING

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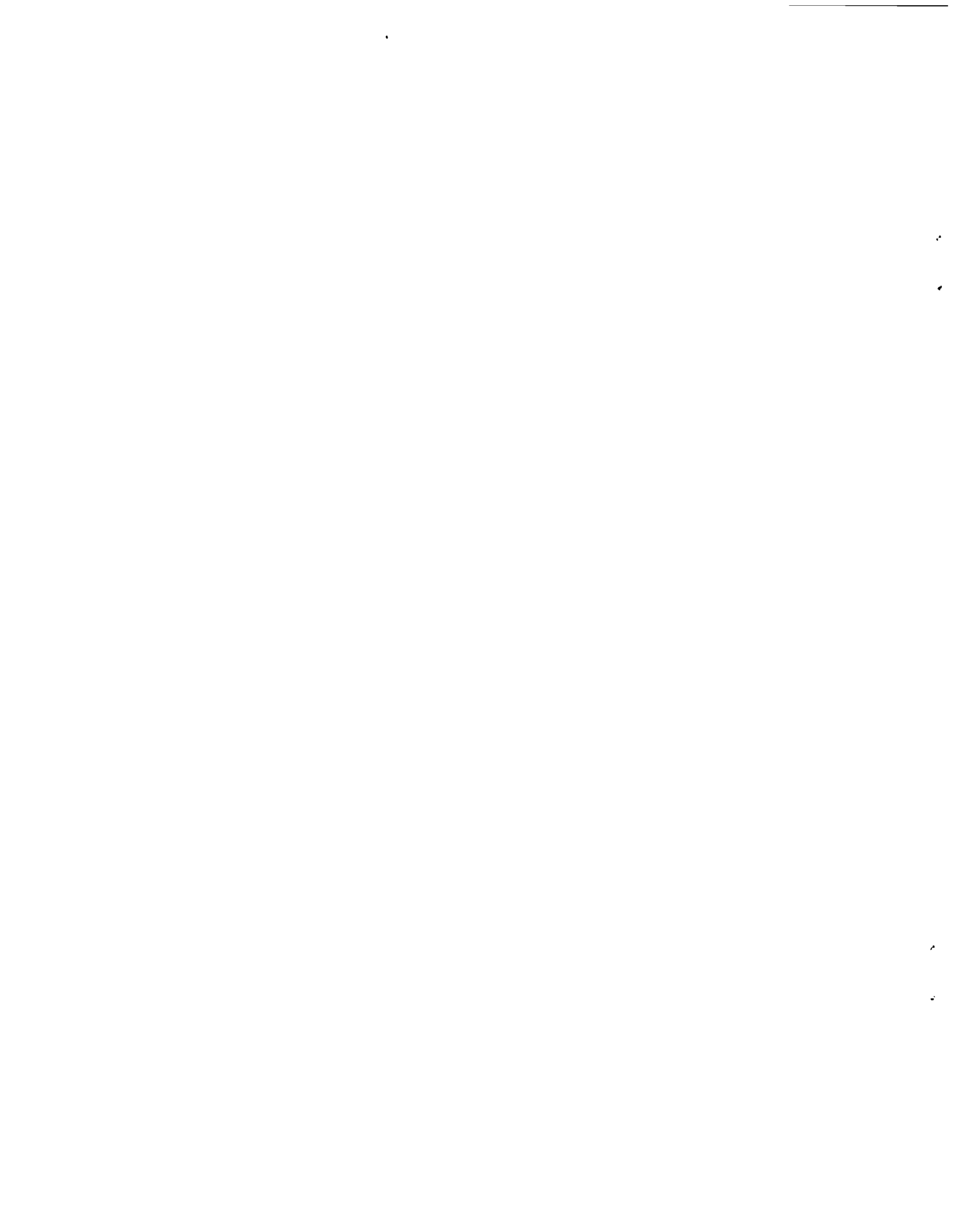
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Abstract

This paper describes a planned modification of the NASA Langley 8-Foot High Temperature Tunnel (8' HTT) to make it a unique national research facility for hypersonic air-breathing propulsion systems and discusses some of the ongoing supporting research for that modification. The modification involves: (1) the addition of an oxygen-enrichment system which will allow the methane-air combustion-heated test stream to simulate air for propulsion testing; and (2) supplemental nozzles to expand the test simulation capability from the current nominal Mach number to 7.0 to include Mach numbers 4.0, 4.5, and 5.0. Detailed design of the modifications is currently underway and the modified facility is scheduled to be available for tests of large scale propulsion systems by mid 1988.

Introduction

Current facility capabilities for testing propulsion or missile systems at high Mach numbers are severely limited. The Marquardt Company, the Chemical Systems Division of United Technology Corporation, and Air Force facilities at Arnold Engineering Development Center (Aero Propulsion Test Unit and Aero Propulsion System Test Facility) can accommodate full-scale ramjets and missiles up to about Mach 4. Other test facilities (for example, those at NASA Langley and General Applied Science Laboratories) can simulate flight Mach numbers up to 7, but they are small and can only accommodate incomplete subscale engines or engine components. No existing facility in the U.S. can provide both true-temperature, high-Mach-number flow and large scale.

The NASA Langley 8-Foot High Temperature Tunnel (8' HTT) has many of the attributes desirable for a propulsion test facility, in particular, size and true temperature simulation for Mach 7 flight; however, the high energy level required to simulate Mach 7 flight is obtained by burning high pressure methane and air and the resulting products of combustion are used as the test medium. Thus the test stream is oxygen depleted and will not support combustion. In addition it would be highly desirable to be able to test at lower Mach numbers nearer the range where transition from turbojet or rocket to ramjet or scramjet mode of operation is anticipated.

The present paper describes a planned modification of the 8' HTT to make it a unique national research facility for hypersonic

air-breathing propulsions systems and discusses some of the ongoing supporting research for that modification. The modification involves: (1) the addition of an oxygen-enrichment system which will allow the methane-air combustion-heated test stream to simulate air for propulsion testing; and (2) supplemental nozzles to expand the test simulation capability from the current nominal Mach number of 7.0 to include Mach numbers 4.0, 4.5, and 5.0. The modified facility will retain the present capability for aerothermostructural and aerothermal loads research for high speed vehicles and aerospace re-entry vehicles.

Research Opportunities

The modified facility will complement the two large scale Air Force facilities at the Arnold Engineering Development Center. As illustrated by figure 1, in which the operating envelopes of the three facilities are superimposed on projected operating range of air-breathing propulsion systems¹ the Aero Propulsion Test Unit and Aero Propulsion System Test Facility appear completely adequate for propulsion research for Mach numbers below approximately 4.5. The modified 8' HTT will extend that range to Mach 7 and provide coverage in the ramjet to scramjet transition region.

The large, approximately 8' diameter by 12-1/2' long, open jet test section will enable the modified facility to support both NASA sponsored research on scramjet engines and development tests of DOD missiles. As shown in figure 2 the facility can accommodate multiple modules of an airframe-integrated concept that includes a vehicle forebody/inlet, combustor, and vehicle/afterbody nozzle. Previously LaRC efforts to develop a technology base for hydrogen and hydrocarbon scramjets have been successful^{2,3} but have been limited by facility size to the testing of single modules including only the inlet and combustor. The larger facility will permit the investigation of (1) potentially critical interactions between multiple engine modules at angles-of-attack and yaw, (2) the effects of engine scale on performance, and (3) the effects on performance of the integrated afterbody/nozzle, configurations for which the airframe provides up to 50-percent of the total thrust. The large scale will also permit the thermal/structural evaluation of flight-weight, fuel-cooled engine structures. Such tests must be made with essentially full-scale components since scaling of structure components is generally not applicable because of the adverse effects on fabrication and manufacturing techniques.

The facility was used for flight weight engine structure studies in the late 60's and early 70's. The hydrogen cooled structural assembly model (SAM) of the hypersonic research engine,⁴ which is shown in the test section of the 8' HTT in figure 3, was

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a full-scale, flight-weight, but non-combusting, replica of the dual mode scramjet. The model, which featured an intricate hydrogen cooling system, was exposed to a total of approximately 30 minutes of hypersonic flight environment and 55 thermal cycles but could not be used for studies involving combustion because of the oxygen-deficient products-of-combustion test medium. A water-cooled boiler-plate model was used for engine performance tests in a NASA Lewis Research Center facility which was closed in 1974.

The 8' HTT can also accommodate all-up missiles up to 12 feet in length. For configurations with forward inlets and internally ducted engine facility size may not be critical, but for configurations such as that shown in figure 4 that rely on the forebody to perform some of the inlet functions⁵ size is important. The facility must be large enough to accommodate significant model angle-of-attack excursions without breaking down the flow or introducing spurious interactions with the engine exhaust flow.

8 Foot High Temperature Tunnel

The 8-Foot High Temperature Tunnel (8' HTT) which was formerly designated the 8-Foot High Temperature Structures Tunnel (8' HTST) is shown schematically in figure 5. The facility, which was designed in the late 1950's was, as the former designation implied, intended primarily for high temperature structures evaluations. The unmodified facility, as indicated by the information included in the figure is currently capable of simulating flight conditions at altitudes from approximately 80,000 to 130,000 feet, at Mach numbers from approximately 5.8 to 7.3. The wide range of Mach numbers is uncharacteristic of fixed nozzle supersonic wind tunnels and, as will be discussed more fully in subsequent sections, is believed to be associated with water vapor condensation in the product of methane-air combustion test medium. Because of the resulting losses in the nozzle the Mach number changes as pressure and temperature in the combustor are changed with the lower temperatures, and less significantly higher pressures, producing lower Mach numbers. Furthermore, the total pressure in the test section may be considerably lower than that in the combustor.

The facility is equipped with a hydraulic elevator system which permits the model being tested to be stored in a pod beneath the test section during the tunnel start-up and shut-down transients. Once steady-state conditions are established the model is moved into position as shown by the triple exposure of a model insertion in figure 6. The carriage on which the model rides weighs approximately 15 tons and can be moved into position in as little as one second. Two interchangeable carriages are available: (1) a pitch carriage (shown) on which the models are sting supported from a curved strut and can be pitched through an angle-of-attack range of ± 20 degrees, and (2) a yaw carriage on which the models are mounted on a turntable flush with the floor and can be rotated through an angle of yaw range of ± 20 degrees.

Also visible in the upper, right-hand corner of figure 6 is the flow survey rake which resembles

a king-size windshield wiper. The rake, which is electromechanically operated, can accommodate up to 37 interchangeable probes (i.e., temperature, pressure, gas sampling probes, heat flux gages) and can provide a flow field survey at any preselected longitudinal position in the test section immediately prior to the insertion of a model or after its retraction from the flow stream.

The combustor in which the hot test gas is generated is illustrated by the cut away model shown in figure 7. High pressure air from a 6000 psi storage source is introduced through a torus at the upstream end of the combustor. To protect the combustor pressure vessel from the hot combustion gases the air flows to the downstream end of the combustor in the annular space between the pressure vessel and a liner where it turns and flows back through the annular space formed by a second liner to approximately the mid point of the combustor. At that point the inner liner in the 8' HTT terminates (although the model shows the inner liner extending almost to the upstream end of the combustor) and the air is dumped into the 3-foot diameter central portion of the combustor. Air in the central portion flows downstream at a relatively low velocity (≈ 30 ft/sec) through the fuel spray bar where the methane fuel is introduced. The fuel spray bar is a series of tubes formed into concentric circles with a pattern of hole derived experimentally to produce a uniform temperature distribution in the test section. The methane is supplied through two large pipes (not present in the model) which extend from the upstream end of the combustor to the spray bar which is located slightly downstream of the truncated inner liner. Combustion occurs in the downstream half of the combustor and the hot gases from the combustor flow through a convergent-divergent, conical/contoured nozzle into the test section. The convergent section of the nozzle is water cooled and the nozzle throat, which experiences the most severe thermal environment, is film cooled using an annular film of air introduced just upstream of the throat. The water and film cooled components have been a source of continuing trouble⁶ and will be modified as part of the forthcoming facility modification.

Combustion Products Test Medium

Although the 8' HTT has been in operation over twenty years and combustion-heated facilities using products of combustion enriched with oxygen (commonly called vitiated air) have been used for engine and engine component tests at the Langley Research Center and elsewhere, the question of the suitability of combustion products as a test medium is a recurring one. The question was addressed in considerable detail in reference 7. That study concluded that the most important consideration in using products of combustion as a test medium was the condensation of water vapor which can have a significant effect on the test flow parameters if the temperature is low enough and therefore test parameters should be determined from conditions measured in the test section. The report indicates that for an equivalence ratio (ratio of the actual fuel to air ratio to the ratio of fuel to air for stoichiometric reaction) of 0.8 with a corresponding theoretical total temperature of 3600°R the flow at the exit of the nozzle would be subcooled by approximately 100°R below saturation

but that there would be little or no effect on the flow parameters. However at an equivalence ratio of 0.5, theoretical total temperature of 2655°R, condensation significantly effects Mach number and static pressure but has little effect on dynamic pressure. Pressure ratios obtained from calibrations in the 8' HTT and presented in figure 8 indicate that results based on conditions measured in the test section and those based on conditions measured in the combustor tend to converge as the total temperature approaches 4000°R, the temperature required for true temperature simulation for the Mach 7 design condition. These results suggest that the effects of condensation vary continuously with temperature and are insignificant if the combustor is operated near stoichiometric conditions.

Experimental aerodynamic data (heating and loading distributions, and aerodynamic forces and moments on various aerodynamic bodies) also presented in reference 7 indicate that if stream conditions are known and proper values of thermodynamic and transport properties of the combustion are used in the reduction of the data good correlation with theory and with experimental results measured in air can be obtained.

Results of theoretical studies of ignition and reaction times of hydrogen in vitiated air (hydrogen-air-oxygen products of combustion)⁸ indicate that, in the temperature range of primary interest for hypersonic propulsion studies, the effects of vitiation tend to reduce the ignition times relative to those in air but that the total reaction time is little effected. Results of more recent calculations, with the vitiated air consisting of the products of combustion of methane, air, and oxygen, presented in figure 9, indicate a similar reduction in ignition time. Thus ignition rates in propulsion systems tested in products of combustion may lead to overly optimistic conclusions and may require adjustment of the tests data or the addition of suppressants to retard ignition. However additional studies are required to fully assess the significance of the variation of ignition time.

Facility Modifications

O₂ Enrichment. - After considering several alternatives, an oxygen enrichment system that injects LOX directly into the tunnel combustor as indicated by the schematic diagram in figure 10 was selected. The system uses an existing low pressure tank for long term LOX storage, and a large high pressure LOX run tank that is pressurized from an existing high pressure nitrogen system. The system has the advantages that (1) it is separate from the air supply system and can be kept "oxygen clean," (2) only the combustion air is oxygen enriched, (3) an oxygen vaporizer is not required, and (4) the system is pressurized only during run time. As indicated by the figure, the 8000 gallon LOX run tank and most of the valves are excess equipment from the Air Force Rocket Propulsion Lab. Although the surplus equipment results in a considerable cost savings, the 2300 psi pressure rating of the run tank is not as high as desired. Because of this pressure limitation, the facility will be limited to a maximum dynamic pressure of approximately 2000 psf or minimum simulated altitudes from approximately 50,000 feet at Mach 4

to 90,000 feet at Mach 7. To provide for future growth, the piping and the valves will be rated for 5000 psi operation.

As shown in figure 11 LOX will enter the combustor through the air supply torus located at the upstream end of the combustor (see fig. 7). The LOX piping will penetrate the torus at, and extend through, 8 of the 20 existing 3-inch diameter holes used to distribute the incoming air into the combustor. The selected holes will be enlarged to account for the additional blockage incurred by the LOX piping. Split segment manifolds will be used to further distribute the LOX through 64 tubes which will conduct the LOX to the downstream end of the combustor where it will be sprayed into the counterflowing air in the annular space between the two liners. The torus was selected for the point of penetration of the combustor pressure vessel because (1) stresses in the torus are low, (2) the 6A350 LF3 steel of the torus is more suitable for low temperatures than the carbon steel of the main pressure vessel, and (3) modification of the existing combustor would be minimized. The annular space between the two liners at the downstream end of the combustor was selected as the LOX injection site because (1) it provides sufficient distance for mixing of the air and oxygen, and (2) the nickel 201 and stainless steel of the inner and outer liners, respectively, are compatible with LOX whereas the carbon steel of the pressure vessel is not.

Selection of materials and overall safety are overriding concerns in the design of the oxygen enrichment equipment, since, in contrast to most high energy facilities which are ensconced behind bunkers, the 8' HTT is immediately adjacent to occupied offices. A recent NASA publication⁹ (and consultation with the authors of the report) has proven to be an invaluable aid in the selection of materials (both metallic and non-metallic), the practical design of LOX system components, and the selection of operational parameters. The basic philosophy adopted for the O₂ control system is to keep it as simple as possible and to avoid rapid response both for safety reasons and to avoid possible interactions with other control systems in the facility. During operation combustion will be established first, then oxygen flow will be ramped to a preestablished level with only minor modulations possible during a test. Rapid response instrumentation is being developed to permit monitoring of the oxygen concentration¹⁰ and the equipment could be incorporated in a computer controlled control system as indicated in the reference report; however, that approach is not presently contemplated.

To provide first-hand experience in handling oxygen and to develop operational procedures a mock-up of the proposed LOX system has been installed and is being evaluated in the Langley 7" HTT which is a pilot facility for the 8' HTT. So far the mock-up has provided valuable experience in handling and transferring cryogenic fluids. In the initial test in which oxygen was actually injected it was found that the addition of oxygen did not introduce any large transients but that it did significantly increase the efficiency of the burning process as evidenced by a 200-300°F increase in the total temperature. There also appears to be a slight decrease in the turbulence level. However the higher temperature coupled with

an apparent upstream movement of the flame front overheated the fuel spray bar. This problem has been overcome in the pilot facility by slightly increasing the fuel pressure differential. It is anticipated that ultimately some fine tuning of the fuel pressure will be required in the full scale facility to prevent similar overheating and at the same time not affect the stability of the combustion process. Subsequently the uniformity of the mixing will be explored in the pilot facility through surveys of conditions in the test section (temperature, pressure, and oxygen concentration).

Alternate Mach Numbers. - Test conditions in the unmodified 8' HTT are at the lower end of the scramjet range (fig. 1). However for Mach numbers in the range for ramjet testing and, in particular, the critical turbojet to ramjet transition range, the nozzle must be reconfigured. To obtain the desired Mach numbers and provide high dynamic pressures, a dual throat/mixer concept similar to that used to convert Tunnel C at the AEDC Von Karman Gas Dynamics Facility from a Mach 10 to a Mach 4 true temperature tunnel was selected.¹¹ This concept has also been used to provide increased capabilities for the Langley Scramjet Test Facility.¹² For the 8' HTT application, as illustrated by figure 12, a section of the existing nozzle will be removed and a mixer with interchangeable nozzle sections for the desired Mach numbers will be installed. The interchangeable nozzles will consist of two parts--a nozzle throat insert, which may or may not be cooled depending upon the Mach number, and an uncooled downstream section that provides a transition from the insert to the existing nozzle. The purpose of the mixer is to increase the total mass flow and to reduce the temperature by adding ambient temperature air to the hot gas flowing from the existing combustor so as to provide true temperature simulation and maintain high dynamic pressures at the lower Mach numbers. The mixer approach was selected in lieu of enlarging the existing throat because initiating and maintaining combustion with higher velocities in the combustor is difficult and because the mixer approach facilitates changing Mach numbers. For operation at Mach numbers of 4.0, 4.5, and 5.0 only the nozzle sections would be changed and the mixer flow rates adjusted to obtain the desired conditions; for operation at Mach 7 the entire mixer section would be removed and the original nozzle section reinstalled. To facilitate configuration changes each of the components will be "floated" on air bearings. The mixer configuration differs somewhat from that used at AEDC in that the 8' HTT mixer will use aerodynamic means to promote mixing in lieu of the "core breaker" (perforated metal plate) and screens used at AEDC.¹³ The aerodynamic concept which has been demonstrated in the Langley Mach 7 Scramjet Test Facility¹² was selected because conditions in the 8' HTT mixer are too severe for uncooled metallic plates and screens. Details of the mixing system will be established in the 7" HTT pilot facility.

Air transpiration cooling using a platelet concept pioneered by the Aerojet Techsystems Company (formerly the Aerojet Liquid Rocket Company) will be used for the cooled portion of the nozzles. In this concept, which has been successfully used to provide thermal protection for several different applications involving very high heat flux environments (see for example ref. 14),

hydraulic passages are photoetched on thin sheets or platelets. (A photograph of typical platelets for a nose cone application is shown in figure 13.) The platelets are then stacked to form a complete coolant distribution system. By varying the design of the individual platelets and order of stacking very precise distribution of the coolant can be obtained and the design can be tailored for any particular application. In addition to the mixer nozzle inserts transpiration cooled components will be developed to replace the water cooled convergent section and the film cooled throat section of the present Mach 7 nozzle (see fig. 7). The new nozzle components will eliminate persistent problems with water leakage and will significantly reduce the nozzle cooling air requirements. Because of the higher efficiency of transpiration cooling relative to slot film cooling the coolant flow will be reduced from the present rate which ranges up to approximately 30-percent of the total nozzle flow to less than 7 percent of the total. This reduced flow rate should be reflected in an increase in the hot gas flowing through the nozzle, a reduction in condensation losses, and a larger core (region of uniform temperature) in the test section. Although the nozzles will be larger than any components previously fabricated there appear to be no insurmountable technical obstacles to their manufacture. For the main nozzle the transpiration cooling also provides the potential for accommodating the higher enthalpy flow that would be required if, at a later date, it was decided to take advantage of the potential of burning methane in pure oxygen and reconfiguring the nozzle to provide a true temperature, Mach 10-11 facility.

As with the LOX system, a subscale development model of the mixer concept is being investigated in the 7" HTT. The model, shown in figure 14, provides considerable flexibility for investigating the effects of mixer configuration and air injection mode on the mixing process. In theory the upstream parallel flow injectors will tend to stabilize, energize, and compress the incoming hot gas stream in order to minimize the losses incurred as the flow becomes subsonic. The interdigitated upstream normal injectors and the six fingers that protrude into the stream will provide the bulk of the cold air for mixing. (If sufficient mixing can be obtained with the normal injector, the fingers will be eliminated.) The remaining cold air that is introduced through the downstream injectors is primarily for wall cooling. Instrumentation for the model will include a survey rake that can be installed at different axial locations in addition to wall static pressure and temperature sensors.

Schedule

As indicated by the schedule presented in figure 15 the preliminary engineering design study has been completed and conceptual designs have been established for all components. These concepts are being verified in the previously described, ongoing tests in the 7" HTT which should be completed before the end of 1984. Meanwhile, detail design of the full scale modifications are underway and should be completed by mid 1985. The construction phase is scheduled to begin in mid 1985 and extend for approximately two years with the facility shutdown for actual modification during the last six months. Following construction a period of six months to a year will be required for shake-down

and calibration; thus, the facility should be ready for propulsion testing by early to mid 1988.

Concluding Remarks

Conceptual designs have been established for modifications to the Langley 8-Foot High Temperature Tunnel to make it suitable for the testing of hypersonic, air-breathing propulsion systems. The modifications involve the addition of oxygen to the test stream to make it suitable for combustion studies and supplemental nozzles to expand the test range of the facility. Direct LOX injection into the combustor from a nitrogen pressurized run tank has been selected for the oxygen enrichment system and a dual throat/mixer concept with interchangeable nozzle inserts for Mach 4.0, 4.5 and 5.0 has been selected to expand the test range. The latter, which will be interchangeable with the present Mach 7 nozzle, will mix ambient temperature air with the hot gas from the combustor so as to provide true temperature simulation and to maintain high dynamic pressure at the lower Mach numbers. Subscale developmental tests to verify conceptual design selected for the facility modification are currently underway in the Langley 7-Inch High Temperature Tunnel. In addition to these modifications, the upstream portion of the present Mach 7 nozzle will be replaced with transpiration cooled components to increase the efficiency and reliability of the nozzle. When the modifications are completed in 1988 the 8-Foot High Temperature Tunnel will provide a unique national facility for large-scale, hypersonic, air-breathing propulsion systems research.

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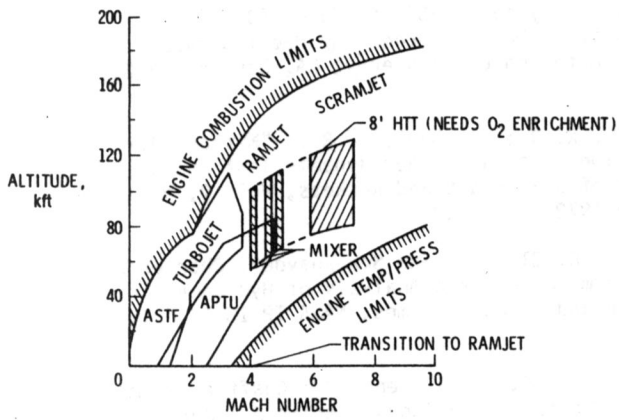


Fig. 1 Operational envelopes for LaRC 8' HTT and AEDC Propulsion Facilities (ASTF & APTU).

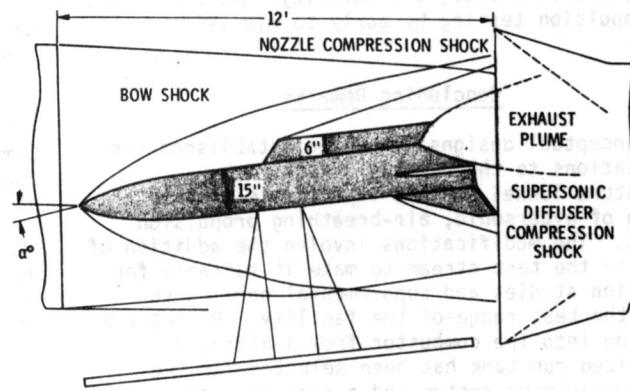


Fig. 4 Typical air-breathing missile.

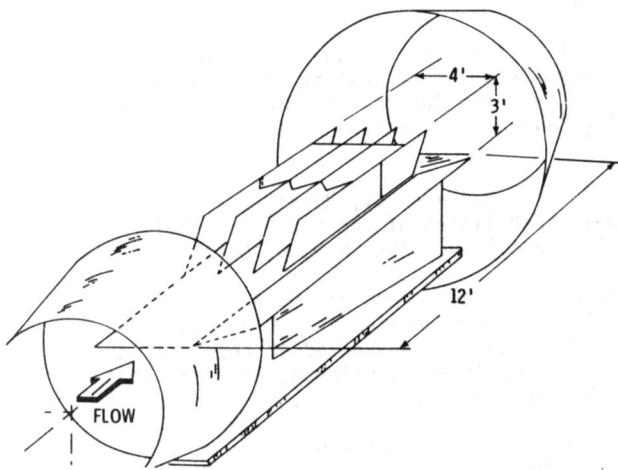


Fig. 2 Multiple module scramjet test model.

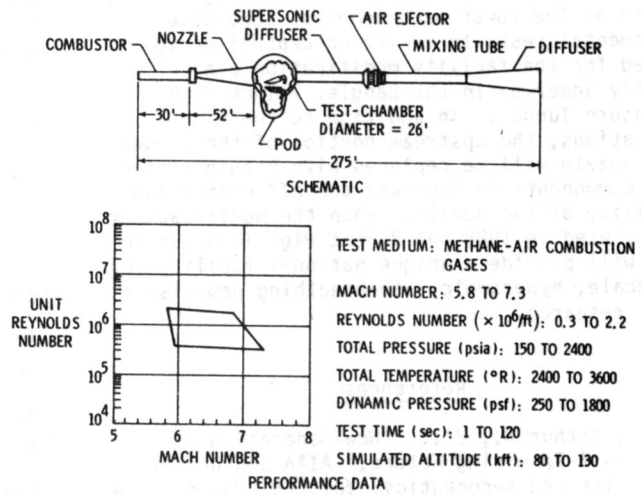


Fig. 5 Langley Research Center 8-Foot High Temperature Tunnel.

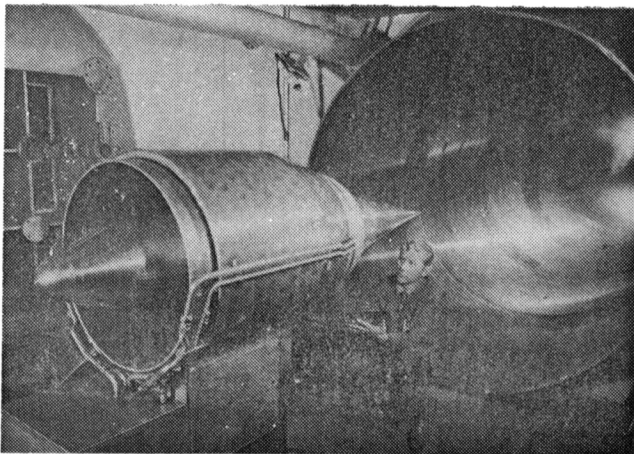


Fig. 3 Structural assembly model of hypersonic research engine in the LaRC 8' HTT.

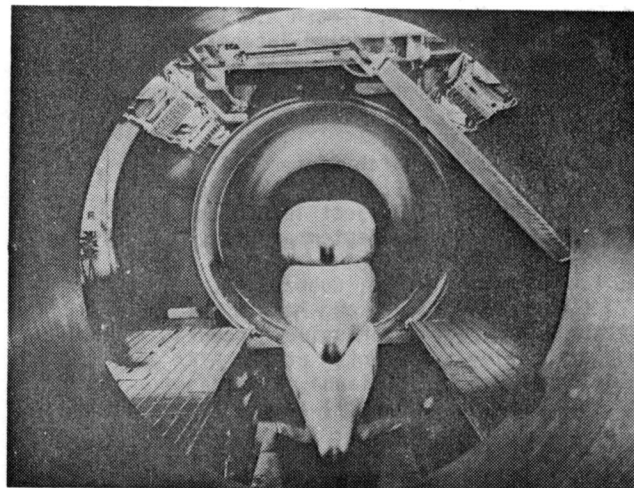


Fig. 6 Triple exposure of model entering 8' HTT test section.

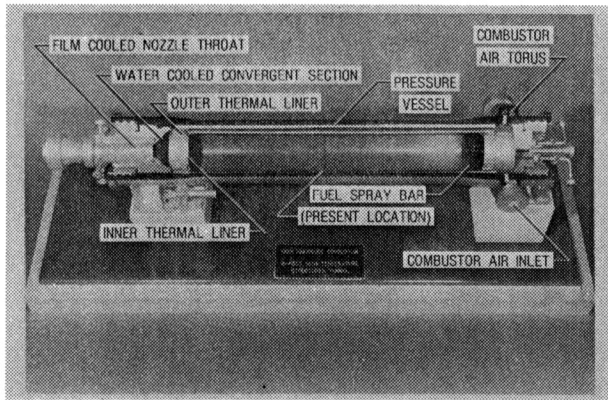


Fig. 7 Cutaway model of 8' HTT combustor.

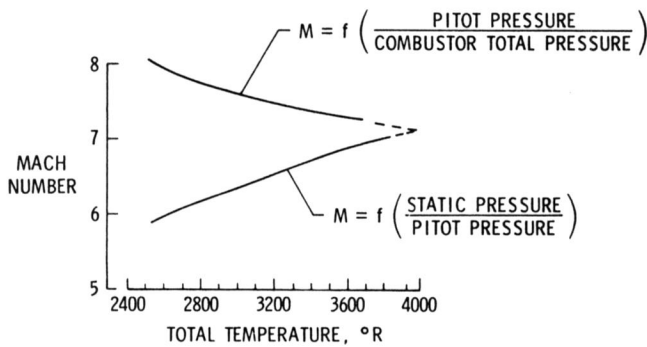


Fig. 8 Mach number based on conditions measured in 8' HTT test section and in combustor.

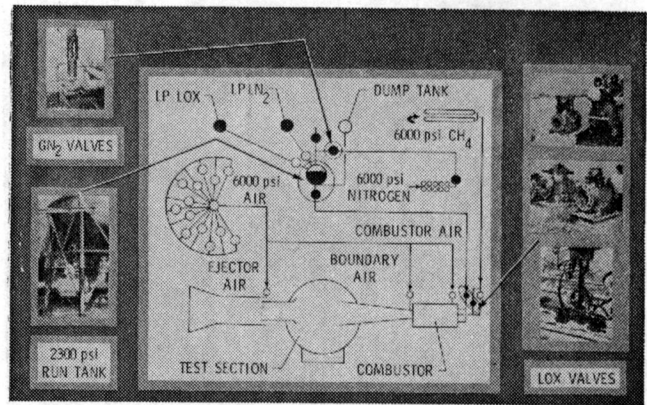


Fig. 10 Proposed LOX enrichment system. Photographs show available AFRPL surplus equipment.

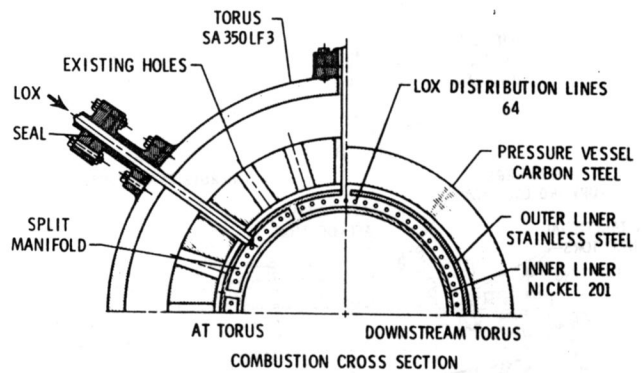


Fig. 11 Details of LOX penetration and distribution.

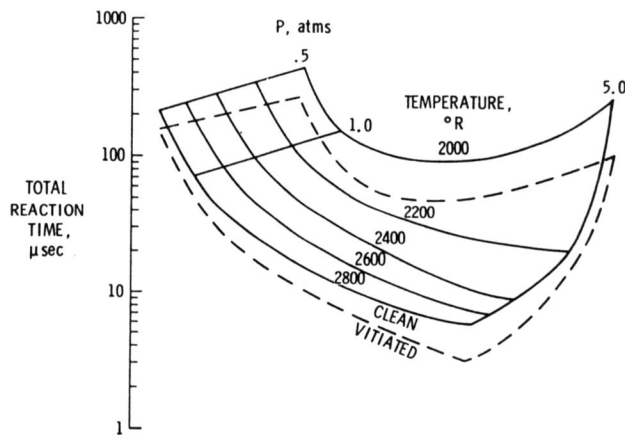
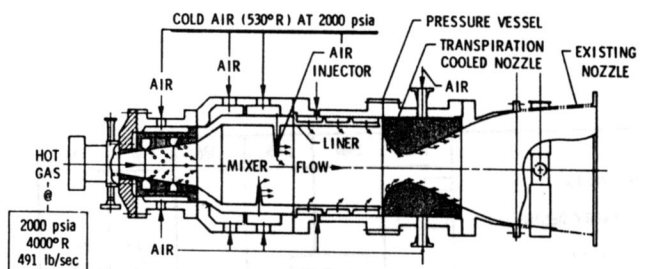


Fig. 9 Reaction times for H₂-air and H₂-vitiated air (CH₄-Air-O₂ products of combustion).



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
4.5	22.2	316	1970	933	1424
5.0	17.1	444	2350	597	1088

Fig. 12 8' HTT mixer with Mach 5 nozzle.

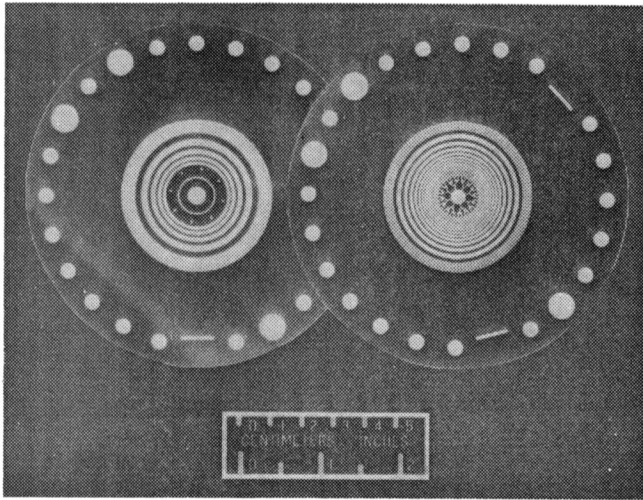


Fig. 13 Typical platelets for transpiration cooled nose cone.

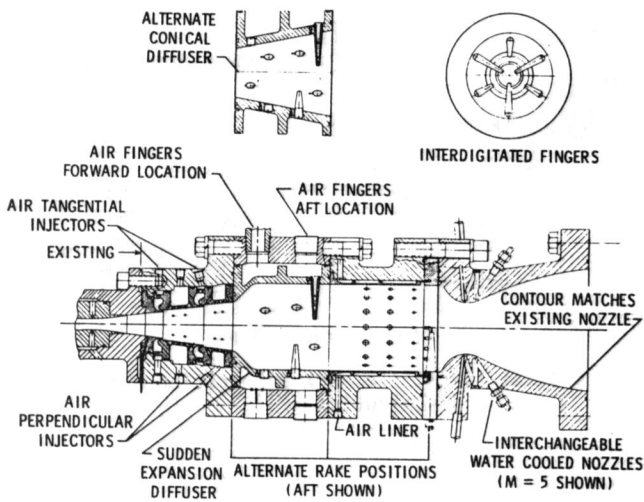


Fig. 14 Development model of mixer and Mach 5 nozzle for 7" HTT.

	1983	1984	1985	1986	1987	1988
REQUIREMENTS	█					
PRELIMINARY DESIGN	█					
FINAL DESIGN		█				
CONSTRUCTION			█			
CHECKOUT AND CALIBRATION					█	
O ₂ ENRICHMENT DEV. (7' HTT)	█					
MIXER DEV. (7' HTT)		█				

Fig. 15 Schedule for 8' HTT modification.

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16. Abstract This paper describes a planned modification of the NASA Langley 8-Foot High Temperature Tunnel (8' HTT) to make it a unique national research facility for hypersonic air-breathing propulsion systems and discusses some of the ongoing supporting research for that modification. The modification involves: (1) the addition of an oxygen-enrichment system which will allow the methane-air combustion-heated test stream to simulate air for propulsion testing; and (2) supplemental nozzles to expand the test simulation capability from the current nominal Mach number to 7.0 to include Mach numbers 4.0, 4.5, and 5.0. Detailed design of the modifications is currently underway and the modified facility is scheduled to be available for tests of large scale propulsion systems by mid 1988.					
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