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Abstract

The goals of the NASA Hypersonic Research Engine (HRE) Project, which began in 1964, were to design, develop, and construct a hypersonic research ramjet/scramjet engine for high performance and to flight-test the developed concept over the speed range from Mach 3 to 8.[1] The project was planned to be accomplished in three phases: project definition, research engine development, and flight test using the X-15A-2 research airplane, which was modified to carry hydrogen fuel for the research engine. The project goal of an engine flight test was eliminated when the X-15 program was canceled in 1968. Ground tests of engine models then became the focus of the project. Two axisymmetric full-scale engine models, having 18-inch-diameter cowls, were fabricated and tested: a structural model and a combustion/propulsion model. A brief historical review of the project, with salient features, typical data results, and lessons learned will be presented.

Introduction

For several years prior to 1964, considerable research had been conducted in experimental investigations of airbreathing engine components (inlets, combustors, and nozzles). Direct-connect combustor tests were conducted to demonstrate the validity of supersonic combustion. The status of this component technology in the early 1960's indicated a high potential for significant advances in hypersonic airbreathing propulsion using a supersonic combustion ramjet (scramjet) engine with hydrogen as both a coolant and the fuel (a regenerative system). The research results, however, had not been integrated into a complete engine having high performance and operational flexibility over any significant range of speed beyond that obtainable with turbojet engines. NASA's Hypersonic Research Engine Project (HREP) was formulated in 1964 to meet the need for a program to effect this integration and to accelerate advancement of the technology of airbreathing propulsion for hypersonic atmospheric flight. Langley Research Center was the lead center with the Ames, Flight, and Lewis Research Centers participating.

The HRE Project's main research objective was to demonstrate high internal thrust performance for a scramjet engine over a Mach number range of 4 to 8; the engine was meant for research and was not in any sense meant to be a small-scale prototype of a propulsion system for any particular flight mission. This task was to be accomplished by means of broad objectives, such as: (a) provide focus for application and integration of fundamental and engine component research; (b) generate comparable engine ground and flight test data as a basis for future decisions; (c) guide and stimulate hypersonic airbreathing propulsion research; and (d) establish the validity of existing hypersonic engine research, developmental methods, and future requirements. To meet these broad objectives the HRE Project was planned to be accomplished in three phases: project definition in Phase I, research engine development in Phase II, and flight tests using the X-15A-2 research airplane as a test vehicle in Phase III (see figure 1). In January 1968,

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during Phase II development, the goal of the project to flight test an engine came to an end when the X-15 program was canceled. Therefore, ground tests of engine models became the driving focus of the project. To fulfill the project's redirected goals, two axisymmetric full-scale models with 18-inch-diameter cowls were constructed. One was a full-scale, water-cooled, gaseous hydrogen-burning Aerothermodynamic Integration Model (AIM) that was tested in the NASA Lewis Plum Brook Hypersonic Test Facility at Mach 5, 6, and 7. A second full-scale model was of flight-weight structure and hydrogen cooled (gaseous H_2 at LN_2 temperatures). This model, the Structures Assembly Model (SAM), was tested in the NASA Langley 8-Foot High-Temperature Structures Tunnel at Mach 7. No combustion in this engine was possible because of an oxygen-deficient tunnel stream.

This present paper will present an historical review, salient features, typical data results, and lessons learned from the HRE Project. Many of the contractors' documents were made available as NASA contractor reports and several NASA formal reports were generated during the HRE Project. A list of 149 of these publications is contained in an appendix of reference 2.

Research/Development Program

The HRE axisymmetric configuration, figure 2, had a controlled translating spike that could be moved fore and aft from inlet closeoff to full open and to intermediate positions. Inlet closeoff was required to minimize the use of hydrogen coolant before and after the engine test portion of the X-15A-2 flight and to minimize foreign-object damage to the engine during takeoff and landing. At the onset of the engine test during the flight, the spike was translated aft from inlet closeoff to allow the inlet to start. The inlet spike was in a fixed position for Mach 4 to 6 operation with the spike-tip shock falling outside of the cowl lip, as shown on the underside of the spike in figure 2. From Mach 6 to 8, the spike-tip shock impinged on the cowl lip, as shown on the top side of the spike in figure 2. This shock-on-lip condition was maintained by translating the spike in a forward direction as the Mach number increased. Subsonic combustion was planned over the Mach range of 4 to 6, with transition from subsonic to supersonic combustion from Mach 5 to 6, and all supersonic combustion up to Mach 8. The staged fuel injectors, which were used to accomplish these mode changes, are shown in figure 2. The flight-weight structure fabrication techniques are also illustrated in the inserts of the figure.

Flight Associated

Development items associated with the flight experiment included fuel systems, a fuel/engine control system, X-15 integration studies, flight instrumentation, and ground support systems for flight.[2] A close-coupled, boot-strap, liquid-hydrogen turbopump was completely developed. Hot (1500° R) and cold (50° R) hydrogen control valves were also developed along with a breadboard of the control system. The valves and turbopump had been developed to the point where they were considered to be prototype flight configuration hardware, but were never used since the X-15 program was canceled prior to any flight tests.

Fabrication/Structures

Fabrication techniques for the cooled structures were developed and partial sections were fabricated. Many tests were performed on sections of the cowl leading edge, the spike tip, the manifold crossovers, and the internal strut.[2] Fabrication of defect-free parts was a learning process in that components were made, and remade, until usable or repairable parts were available; the success ratio was approximately one out of three.

Inlet Program

In the inlet development test program, two different models were tested—a one-third scale model and a two-thirds scale model. The one-third scale model was tested at Mach 4 in the Unitary Plan Wind Tunnel at Langley Research Center where inlet starting problems were encountered. During hot flow tests at Mach 4 in a facility at the Ordnance Aerophysics Laboratory (OAL), Dangerfield, Texas, the model was precooled and the inlet started upon model injection into the tunnel flow but unstarted as the model surface temperature increased. At NASA Langley, the starting problem was studied using actively cooled surfaces to allow the inlet to start and remain started during steady state test conditions. A two-thirds scale inlet model

(12-inch-diameter inlet cowl) was then fabricated with active nitrogen (vaporized liquid) cooling and tested at the Arnold Engineering Development Center (AEDC) in tunnels A and B over the Mach number range of 3 to 8. The pressure recoveries and mass flow ratios were found to meet the required inlet performance criteria.[2]

Combustor Program

A combustor research and development program was conducted with a two-dimensional combustor model that permitted the study of staged fuel injection, angled fuel injection, and geometry scaling. A study of the combustion kinetics was performed for the diverging combustor in order to determine the optimum station to inject the hydrogen fuel. Staged fuel injection tests were also performed at the United Technologies Lab using a modified version of a two-dimensional model. Test results were compared to theoretical analysis and the results of this program fed into the engine design.[2]

Nozzle Program

Another subprogram that was conducted during this project was an engine exhaust nozzle research and development program. This program had two major categories—determination of experimental performance and design analysis optimization. Two one-third scale nozzle models were fabricated and tested; the configuration of one was optimized for Mach 6 flight conditions and the other for Mach 8 conditions. One nozzle had surface cooling (liquid nitrogen) to allow determination of the overcooling effect upon performance. Direct-connect tests were conducted at the Fluidyne facility in Minneapolis, Minnesota, with an unheated air supply at several incoming Mach numbers. The tests permitted assessment of internal engine centerbody mounting-strut losses, entrance Mach number, plug truncation, initial boundary layer thickness, and wall cooling effects. The presence of internal struts resulted in performance degradation, ΔC_T= 0.008-0.009, and cooling produced a decrement of ΔC_T=0.006.[2] (C_T is internal thrust coefficient based on maximum capture area.)

Flight Program

The X-15 airplane that was to be used for the HRE flight test was designated the X-15A-2. This aircraft was modified with the addition of a section to include the hydrogen fuel tank for the HRE. For high-speed flights (Mach 6-8), external drop fuel tanks were attached and used by the X-15 up to about M = 3.5 and an ablative thermal protective cover was applied over the entire aircraft.

Simulated Engine Tests

Preliminary flights, prior to the flight test program, were performed to test the X-15A-2 for controllability with a simulated HRE attached to the underside of the aircraft. The model did not have internal flow passages but did have the external shape of the HRE. Two flight tests were performed with this simulated engine attached, as shown in figure 3. The first flight was conducted at a maximum flight speed of approximately Mach 3.5 without the drop tanks or ablative coating (figure 3(a)). The second flight was performed at a maximum flight speed of Mach 6.7 on October 3, 1967, with the drop tanks attached and the ablative coating applied; this was the last successful X-15 flight. During this latter flight, structural damage to the aircraft/engine pylon occurred, as a result of shock impingement, to the point that the simulated HRE model fell from the underside of the vehicle on the aircraft's final landing approach. The X-15 was a very austere program at that time, and a decision was made the following year, January 1968, to terminate the X-15 program. Since the HRE would not be flight tested on an X-15, the HRE Project focus was redirected.

HREP Redirection

With the cancellation of the X-15 program, ground tests of engine models became the driving focus of the HRE Project. The objectives for the project then became: 1) completion of the development of the engine aerothermodynamic design and testing of a full-scale, water-cooled, gaseous hydrogen-burning aerothermodynamic integration model (AIM) of the research engine in order to verify engine performance; and 2) completion of the development of the structural design and validation of the full-scale engine structure by testing of a full-scale hydrogen-cooled structures assembly model (SAM) of the research engine.

Structures Assembly Model (SAM)

At the time of the X-15 program termination, the HREP was conducting a flight-weight structures program where engine components (inlet spike, outer shell, etc.) were fabricated and the fabrication processes were evaluated by various destructive and nondestructive testing. A decision was made to assemble the fabricated parts from the flight-weight engine structures program, the engine vibration model, and additional required parts into a structural test engine; the Structures Assembly Model (SAM). This flight-weight engine development program had the means to determine system feasibility, establish aerodynamic design methods, design and fabricate a light-weight cooled structure, and to test the structure, the SAM, in a wind tunnel at conditions simulating Mach 7 flight conditions.[3-5]

Facility/Model

The SAM was tested in the NASA Langley 8-Foot High-Temperature Structures Tunnel from 1971 to 1972. The facility is a hypersonic blowdown tunnel in which the energy level for simulating hypersonic flight is obtained by burning methane and air in a high pressure combustor.[6] The resulting combustion gases are expanded through a contoured nozzle with an 8-foot exit diameter to obtain a nominal Mach 7 flow in an enclosed 14-foot long open-jet test section. The combustion-heated tunnel flow, not replenished with oxygen (which left about 4 percent oxygen by volume), was suitable for structural tests but unsuitable for combustion tests.

The SAM is shown installed in the wind tunnel in figure 4. The base plate of the mounting strut was flush with the floor of the tunnel that allowed the proper alignment of the model in both pitch and yaw. The SAM heat-exchange skin was fabricated of Hasteloy-X.[5] Maximum hot surface temperature of 2000° R was chosen to satisfy a hot-surface creep-rupture life criterion. The cold structural surface temperature was limited to avoid creep deformations. The SAM engine consisted of flight-weight structure with all aerodynamic surfaces of brazed, plate-fin sandwich construction with hydrogen cooling as shown in figure 2. Hasteloy-X was used in all shells with the hot skin being 0.015-inch thick. Fin density ranged from 16 to 28 per square inch, and fin height varied from 0.020 to 0.153 of an inch as a function of operating temperature, heat fluxes, and geometry requirements. Thermal fatigue life of the structures used in SAM was a function of the temperature difference across the structure. The design fatigue life was approximately 100 cycles. With several flight engines, this life was considered sufficient to meet the flight-test objectives.

SAM Tests and Results

SAM tests.- Tests of the SAM were conducted at various tunnel flow stream total pressures and total temperatures. A run was defined as a blowdown of the wind tunnel in which the model was inserted into the gas stream on the tunnel centerline; during cooling performance tests, a run consisted of a single cycle and during the thermal cycling tests each tunnel run generally consisted of two cycles.[5] The longest time in the tunnel for any one run occurred at 2200 psia and 3000° R during which the model was in the stream for 116 seconds. At the maximum tunnel conditions, 3300 psia and 3400° R, the run times were 35 to 40 seconds. The average run time at the lower tunnel conditions was between 50 and 60 seconds. Because of the differences in tunnel total temperatures, the tunnel nozzle exit Mach number varied from 6.3 to 6.8 (due to water vapor condensation).

SAM test results.-Surface temperatures of the SAM are presented in figure 5 for steady-state conditions: 3320 psia and 3400° R. The test data are represented by the dashed-line curves and the solid-line curves represent results of a Mach 8 analysis. The steady-state test surface temperatures were generally lower than predicted for the Mach 8 temperatures. Model surface discolorations, which are an indicator of surface temperature, were observed at several locations during inspection after tests with reduced hydrogen coolant flow rates; results are represented by the symbols in figure 5. Such discolorations did not occur for runs with the design hydrogen coolant flow rate.

SAM test summary.- A summary of the thermal fatigue data for the SAM tests is shown in the table of figure 6. Fifty-five cycles were performed during these tests for a total instream time of 29.7 minutes. During these tests, the measurements of ΔT, surface temperature, and internal cooling passage pressures were used to calculate the combined thermal and mechanical stresses. These calculations were thereby used to estimate the amount the engine material elastic limit was exceeded for each thermal cycle. From these results, the total damage fraction was estimated. The 55 test thermal cycles were estimated to amount to a damage fraction of 0.46 (out of a 100-cycle life). The structural program accomplishments included the development of excellent flight-weight hydrogen-cooled structure hardware for the SAM and the partial validation of that structure during ground tests. The results indicated a need for higher design surface temperatures, lower ΔT's, and a different cooling jacket concept to assure longer engine life with coolant flow rates less than or equal to that required for stoichiometric fuel burning. Some foreign object (debris in the tunnel stream) damage to the cowl leading edge occurred early in the test program. The deformations were deep enough to close some of the 0.020-inch-high fin passages and one dent resulted in a small leak. Numerous other damage areas occurred during subsequent tests, however, none of the areas showed serious signs of distress. Such results indicate that the leading edge as designed had considerable tolerance toward foreign object damage.

Aerothermodynamic Integration Model (AIM)

The AIM was fabricated and delivered to the NASA Lewis Plum Brook Station in August 1971 and prepared for installation in the Hypersonic Tunnel Facility (HTF).

Facility/Model

The HTF is a blowdown enclosed free-jet tunnel designed for propulsion testing with true oxygen composition, temperature, and altitude simulation for the Mach number range of 5 to 7.[7] The facility used an induction-heated, drilled-core graphite storage heater to heat nitrogen. Ambient temperature oxygen was then mixed with the heated nitrogen downstream of the heater to produce synthetic air. Diluent nitrogen was also added with the oxygen in the mixer at tunnel operating Mach numbers below 7 to supply the correct temperature and weight flow to the free-jet nozzles. Altitude simulation was provided by a tunnel diffuser/single-stage steam ejector exhaust system. Three interchangeable axisymmetric contoured nozzles (42 inches exit diameter) provided nominal test Mach numbers of 5, 6, and 7.

The AIM was fabricated from nickel 200 with boiler-plate construction and water cooling. It had an 18-inch-diameter at the cowl lip and was about 87 inches in length (varied with spike translation). Heated (1500° R) gaseous hydrogen was the fuel.

A schematic, presented in figure 7, describes the installation of the AIM in the HTF; components of the AIM are also indicated. The spike, inner shell, and nozzle plug formed the centerbody and the outerbody consisted of the cowl leading edge, outer shell, and nozzle shroud. The outerbody was connected to the centerbody by six internal struts, which also served as passages for the centerbody fuel and instrumentation. The outer shell was attached to two main mounting struts that were connected to the thrust bed. The thrust bed was hung from flex plates to allow free movement. The thrust/drag load cell was mounted to the thrust bed and a "hard-point" beam. The engine outer cowl and main mount strut aerodynamic covers were not attached to the engine but to the hard point. The locations of the fuel injectors are also depicted in figure 7. A frontal view of the AIM installed in the HTF, looking downstream into the facility diffuser, is shown in figure 8.

AIM Tests and Results

AIM tests.- The main goal of the AIM tests was to determine internal thrust performance for a complete engine over the Mach number range of 5 to 7. This performance over a Mach number range was achieved using staged fuel injection for distributed heat release. The effects of the following parameters on thrust performance were investigated: fuel/air ratio, angle of attack, various simulated altitudes, and various inlet contraction ratios. Determination of inlet-combustor interaction limits for this configuration was another goal. Combustion mixing length in this configuration was studied by injecting the fuel at the different axial locations and taking exit-flow gas samples.[8] Fuel autoignition and ignitor (H_2-O_2 torch) performance were also studied during these tests. An important engine operation goal was to demonstrate a controlled combustion-mode transition from supersonic to subsonic and back to supersonic combustion. Heat transfer and engine cooling requirements with combustion were determined from the results of these tests.

AIM inlet performance.- Theoretical preformance predictions for the AIM inlet were documented and the AIM inlet test data were analyzed and compared to the predictions.[9,10] The AIM supersonic total pressure recovery was found to correlate with free-stream and throat Mach numbers, as shown in figure 9.[10] The figure shows the expected result: higher total pressure losses with increased flow compression. The correlation includes spike position changes, different Reynolds numbers, and 0° and 3° angle-of-attack data. The lower values of recovery correspond to the 3° angle-of-attack tests.

AIM combustor performance.- To understand the flow phenomena inside the combustor, the test data were analyzed one-dimensionally using the equations of momentum, energy, continuity, and state with reactants and products of combustion in chemical equilibrium. In the combustor, the arithmetic average of the inner and outer wall static pressure distributions were used with the one-dimensional analysis to determine the flow condition and performance. The results of the analysis are shown in figure 10.

Supersonic combustion efficiency varied with the injector configuration as expected. Combustion efficiency for six different fuel injector combinations are presented in figure 10. Data scatter was observed with the 1a, 1b, 2a, 2c injectors and this injector configuration produced a lower combustion efficiency than the 1b, 2a, 2c configuration.

The sizes and locations of fuel injectors were selected to obtain desired mixing schedules by optimizing the fuel penetration and jet spreading. To increase the mixing efficiency, the injector design in each stage was interdigitated to cover the maximum mixing area. In the final configuration, however, the injectors in the first stage (1a, 1b) were inline and opposed to each other (fabrication error).

Examination of the static pressure distributions for different injector combinations revealed that the interaction between the first and second injector stages had significant effects on the overall combustor performance. The shorter distance between stages using injectors 1a, 1b, 4 and 2c (see fig. 7), or a larger disturbance generated by a single-sided injection from the second stage using injectors 1a, 1b, and 2c appeared to have enhanced the combustion process. The same interaction between the first and second stages, with most of the fuel in the second stage, gave the best performance. Injector 1b only was used for the first stage and injectors 2a, 2c for the second stage. Fuel injectors 1b, 3a, and 3b were used during the subsonic combustion mode.

AIM internal performance.- Internal performance vs. test Mach number is presented in figure 11 at an equivalence ratio of unity (fuel-air ratio = 0.0293). The test data (open-circle symbols) were obtained from the mid range of the thrust values at the test Mach numbers. The Mach 5 data were for subsonic combustion, and the Mach 6 and 7 data were for supersonic combustion. (Mach 7 data were corrected for test total temperature lower than flight simulation.)[11]

The HRE internal performance goals are shown as the cross-hatched bands in figure 11 at an equivalence ratio of unity. The lower lines represent minimum specified values, which were essentially met with the AIM engine. Points on the upper line of the band were considered to be obtainable only for engines optimized for a particular Mach number.

The test data were obtained for a water-cooled engine, whereas performance goals were based on a regeneratively cooled system. For the majority of tests, the AIM was over-cooled with more heat removed than was replaced by the heated hydrogen fuel. To obtain a realistic comparison of test data (open-circle symbols) with test goals, the thrust coefficient and specific impulse were corrected for a regeneratively cooled system (filled-circle symbols). Essentially, the correction involved calculating combustor exit conditions at the same enthalpy as a regenerative system and at test total pressure and combustor efficiency, then expanding the flow to the nozzle exit to determine gross thrust for a regeneratively cooled system. The correction to Mach 5 data was relatively small (about 2 percent), while at Mach 7 it was relatively large (about 12 percent). Close agreement between corrected performances and predictions is evident in figure 11.

Analyses of the test data indicated that the AIM nozzle performance was about 3 to 4 percent lower than expected relative to the 1/3-scale nozzle model tests. Estimates for internal thrust and impulse that would have been attained had the nozzle performance been the same as measured in the 1/3-scale nozzle model tests are shown as the triangle symbols which are above the minimum performance goals in figure 11. These differences in performance were postulated to be related to the turbulence energy generated by the combustion processes and not recovered in the AIM nozzle.

AIM test summary.- A summary of the AIM tests is presented in figure 12. Facility/engine checkout tests were performed from September 1972 through May 1973. During this time, the tunnel/model shrouding was modified to yield good tunnel operations.[11] Some binding of the engine metric and non-metric hardware was detected; this was corrected prior to resumption of tests in the fall of 1973. The first complete fuel-burning test was conducted at Mach 6 conditions on October 5, 1973. A majority of the tests were performed at the Mach 6 conditions (see figure 12). Some tests were conducted at different total pressures than the nominal value to determine the effect of altitude (dynamic pressure) on engine performance. One test was performed at each of the three Mach numbers at an angle of attack of 3°. A total of 52 complete tests were conducted for a total test time (steady-state conditions) of almost 112 minutes. The last test was performed on April 22, 1974, at Mach 5 conditions.

The AIM program was a major testing accomplishment of a complete (inlet, combustor, and nozzle), large-scale engine to demonstrate high internal thrust performance for a scramjet/ramjet engine over a Mach number range. Maximum thrust performance of the AIM was close to predictions. An unexpected result was observed for staged fuel injection. A strong stage interaction occurred and the second stage combustion efficiency was reduced by oxygen depletion near the wall. The fuel-air ratio effect was very similar to that expected. Performance degradation of about 15 percent was noted at 3° angle of attack. The effects of variations in the inlet contraction ratio were also determined. Stable inlet operation was observed during all tests; inlet unstarts were determined for various fuel injection locations. Single-stage mixing lengths were the same as predicted.[12] Fuel autoignition and use of fuel ignitors were successfully demonstrated. An important accomplishment was the demonstration of a smooth transition from a supersonic to subsonic mode of combustion. Measured heat loads to the various components of the AIM indicated that the engine overall heat transfer was very close to predictions. Details of the AIM tests and analyses are contained in references 11 through 13.

Concluding Remarks

The original Hypersonic Research Engine Project (HREP) objectives included the necessary research, the engine design, the fabrication, and the ground and flight testing of a ramjet/scramjet engine. The flight test performance objective and correlation of ground test and flight test data were eliminated when the X-15 Program was canceled. However, a new level of ramjet/scramjet technology was established by the HRE ground test program. The ramjet/scramjet ground test thrust performance was measured using an 18-inch diameter (at cowl lip), water-cooled, boilerplate hydrogen-burning Aerothermodynamic Integration Model (AIM). These tests were conducted in the NASA Lewis Plum Brook Station Hypersonic Test Facility at Mach numbers of 5, 6, and 7. The M = 5 and 6 test conditions were full simulation (total pressure and total temperature) of flight. The Mach 7 condition was limited to a total temperature of about 3200° R because of a facility limitation. Good ramjet/scramjet engine performance was obtained over the Mach number range tested. Engine wall temperatures were much colder in the combustor and nozzle for the water-cooled engine than would be expected for flight; therefore, an energy balance process was used to correct the measured engine performance values to flight-type engine performance levels.

Measured scramjet engine structural performance very close to predicted structural performance was obtained using a hydrogen-cooled, flight-weight Structures Assembly Model (SAM) engine in tests in the NASA Langley 8-Ft. High-Temperature Structures Tunnel at a Mach number of 7. Surface pressures, temperatures, and heat fluxes were measured throughout the engine. Predicted flight-like surface temperatures were obtained by undercooling the surfaces; predicted flight-like structural temperature differences, or hot surface temperature minus coolant-side surface temperature, were obtained by overcooling the engine structure. In this way, the flight-type thermal stresses were duplicated during the tunnel testing. The flight-type heat fluxes in the combustor and nozzle could not be duplicated because of an oxygen-deficient test stream (4 percent by volume) in the 8-Ft. HTST. Since there was no combustion within the engine, the internal engine maximum heat fluxes were only 40 percent of expected flight values. Approximately 46 percent of the predicted thermal fatigue life of the SAM (100 cycles) was used during the repeated testing. No problems with the cooling or cooling control systems were found. Tunnel debris caused perforations in the hydrogen-cooled leading edge of the SAM cowl. This damage caused no distress to the cooling system or problems to the leading edge or internal cooling jackets.

HREP Lessons Learned

Briefly stated and hopefully of future use, the following represent some of the lessons learned during the HRE program.

1. Free-jet engine tests with high blockage engine models should be preceded by small-scale model tests to explore tunnel starting and engine/facility interaction.

2. Purging of engine internal cavities of an engine being used for thrust measurements should be done with care to avoid unwanted tare forces.

3. Thrust measuring models should be assembled in a manner similar to the tunnel installation configuration to avoid binding problems between metric and non-metric parts. (That is, if the model is to be suspended, then assembly should be performed using a "hanging" assembly rig.)

4. Tunnel starting loads analyses are usually performed considering the pressure loads only. Thermal loads should also be considered to avoid bolt/rivet shearing that may occur because of peak thermal loads.

5. Correcting measured engine thrust values to flight requires sufficient ground measurements and flight analyses to make the correct energy balance.

6. Inlet boundary layer transition is difficult to achieve artificially and causes a large total-pressure loss at hypersonic speeds. Transition occurs naturally in high adverse pressure gradient regions and corresponds to the highest total-pressure recovery.

7. Combustion mode transition, i.e., subsonic to supersonic or the reverse, was relatively easy to achieve by switching the fuel injection locations, and thus heat distribution, in the HRE diverging combustor with a 5-percent local area reduction present at the aft end of the combustor (a result of the presence of internal struts).

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Fig. 1. The HRE on an X-15A-2 aircraft prior to launch from a B-52.

Fig. 4. SAM installed in the NASA Langley 8-Foot High-Temperature Structures Tunnel

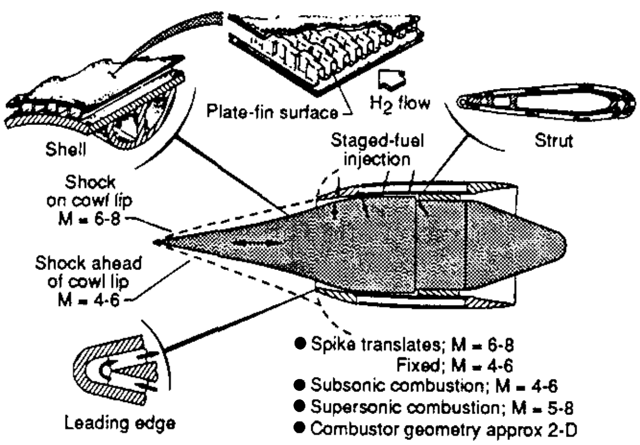


Fig. 2. Hypersonic Research Engine concept and flight engine design features.

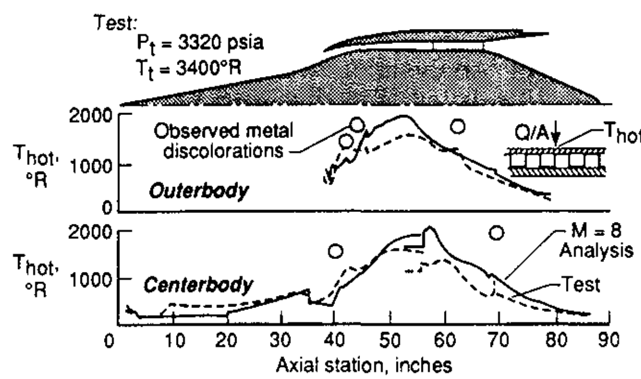
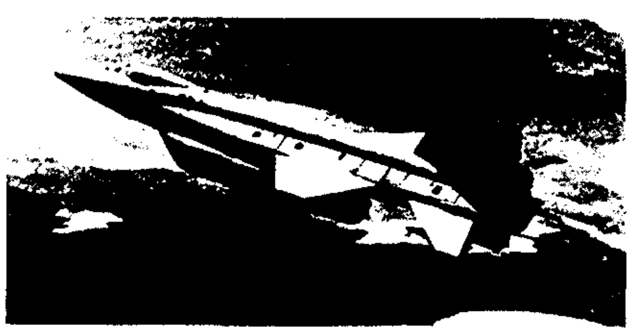


Fig. 5. SAM surface temperatures.

a) Mach 3.5 flight



b) Mach 6.7 flight

Fig. 3. X-15A-2 aircraft flights with simulated HRE attached.

Tunnel total conditions		Number of	Time in Stream,	Avg Cycle Temperatures		Damage fraction,
psia	°R	Cycles	(sec)	T_{max}, °R	Δ T, °R	%
950	2600	5	172	1360	733	1.30
1300	2700	3	135	1445	950	2.12
1380	2700	33	851	1446	906	20.50
1500	2700	3	138	1571	1152	3.77
2200	3000	5	266	1591	1287	8.36
2800	3300	1	58	1435	1224	1.19
3300	3400	5	163	1522	1350	8.46
Totals		55	1783	———		45.70
			29.7 min			

Fig. 6. SAM tunnel tests thermal fatigue summary.

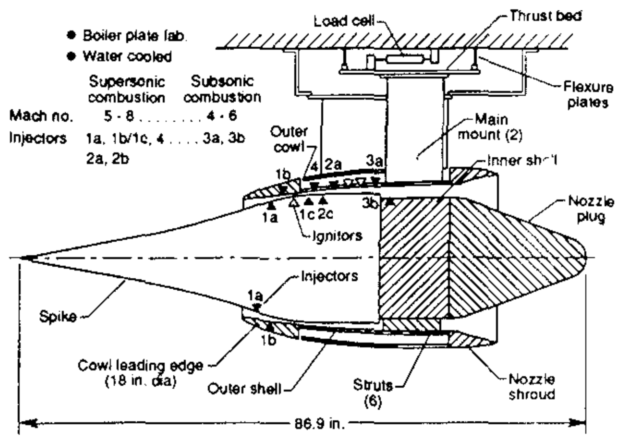


Fig. 7. AIM design features and installation schematic.

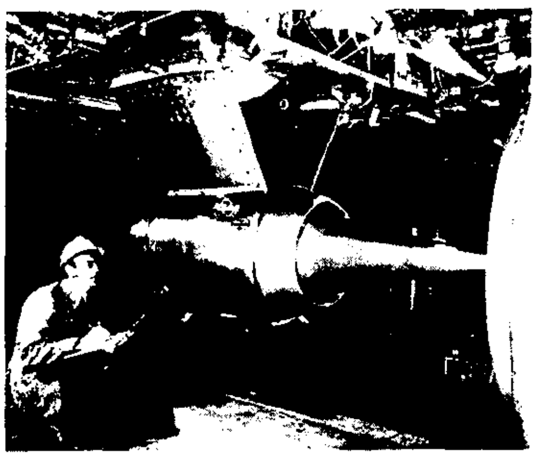


Fig. 8. AIM installed in the NASA Lewis Plum Brook Hypersonic Test Facility.

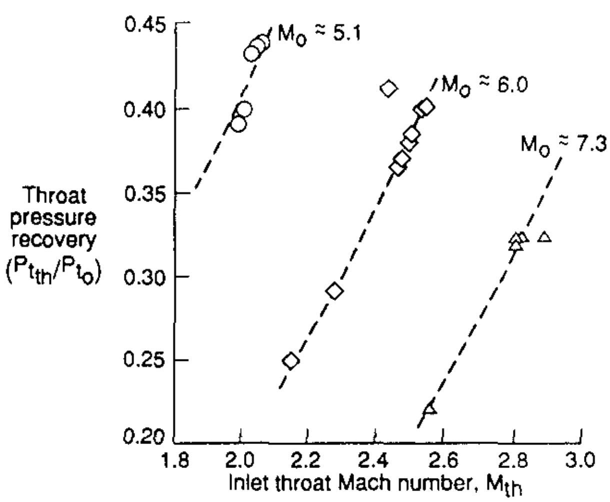


Fig. 9. *AIM inlet performance; supersonic total-pressure recovery correlation.*

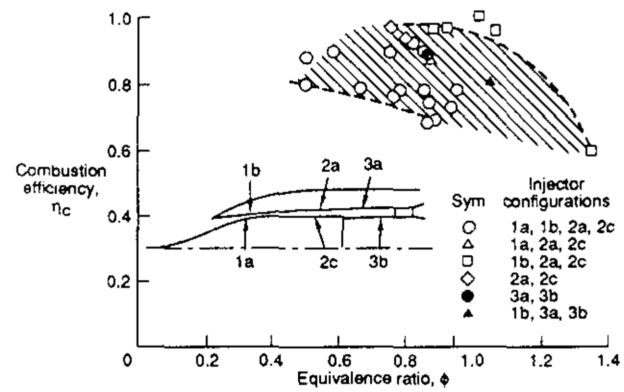
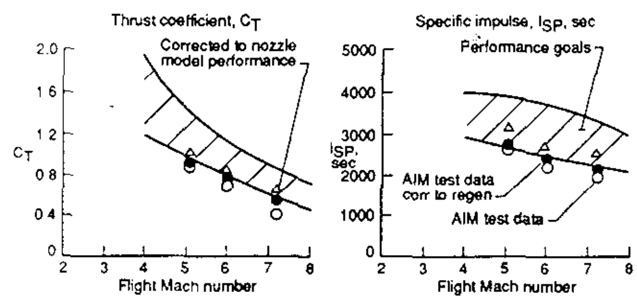


Fig. 10. AIM combustor performance; Mach 6 tests.



a) Thrust coefficient, C_T b) Specific impulse, I_{SP}

Fig. 11. AIM internal performance; ϕ = 1.0.

Mach No.	No. of Tests	Time at test cond (min/sec)	P_{TO} (psia)	T_{TO} (°R)
5	5	19' 30"	210/415	2210/3000
6	36	63' 17"	466/750/930	1500/3000
7	11	28' 57"	1000	3000/3500
	52	111' 44"		

- Test period
  - Mach 6 from Oct 5, 1973 to Dec 19, 1973
  - Mach 7 from Jan 22, 1974 to Mar 18, 1974
  - Mach 5 from Mar 20, 1974 to Apr 22, 1974

Fig. 12. *AIM test summary.*