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THE NASA HYPERSONIC RESEARCH ENGINE PROGRAM

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INTRODUCTION

This paper provides an overview of the NASA Hypersonic Research Engine Program, describes the engine concept which was evolved, and summarizes the accomplishments of the program.

The National Aeronautics and Space Administration undertook the Hypersonic Research Engine Program (HREP) as an in-depth program of hypersonic airbreathing propulsion research to provide essential inputs to future prototype engine development and decision making. An airbreathing liquid-hydrogen-fueled research-oriented scramjet was to be developed to the performance goals shown in Figure 1. The work was many faceted, required aerodynamic design evaluation, structures development, and development of flight systems such as the fuel and control system, but the prime objective was investigation of the internal aerothermodynamics of the propulsion system. At flight speeds below Mach 6, the combustion mode was to be at the contractor's option; above Mach 6, supersonic combustion was specified.

RESEARCH ENGINE CONCEPT

To meet these requirements, an axisymmetric dual-combustion mode design illustrated in Figure 2 was selected. The capture diameter was 0.457 meter (18 in.), the area of the exit nozzle was twice the capture area, and the overall length with the translating spike in the full-forward closed position was 2.13 meters (84 in.). An external-internal compression inlet having a significant degree of external compression minimized inlet wetted surface and associated cooling load. Translation of the inlet spike provided for adjustment of the internal area contraction at higher flight speeds and minimization of inlet spillage at lower flight speeds.

PROGRAM EVOLUTION

At its inception, the hypersonic research engine program plan provided for aerothermodynamic development, first at the subscale component level, followed by component integration and engine performance at full scale for a concurrent development of structures and subsystems, and then for airborne experiments which would be the culmination of the program. This program was subsequently restructured to accommodate retirement of the X-15 flight test vehicle and deactivation of an intended ground-based facility. These program changes redirected structural evaluation toward Mach 7 true-temperature testing in the Langley 8-foot high-temperature structures tunnel of an assembly of the structural components (the structures assembly model, SAM, Figure 3) as the final act in structural development. The restructured program retained aerothermodynamic development essentially unchanged except for the deletion of the final step, building and flight testing of the unified product. Flight system development, having already reached a point where feasibility was insured, was discontinued.

FUEL SYSTEM

The hydrogen system, Figure 4, consisted of a number of circuits supplied by a turbine-driven pump and regulated by special-purpose valves all under command of a digital computer which provided overall control of the system. Four high-pressure cryogenic valves distribute the hydrogen among the engine cooling passages and three high-temperature valves of 922°K (1200°F design) redistribute the collected hot jacket effluent to the fuel injectors. In addition, a turbine control valve regulated the flow of hot hydrogen to the pump drive, and a waste (dump) valve permits operating the system when desired at engine fuel-consumptions values below coolant requirements. The computer provides all logic and control signals necessary for (1) operating the translating inlet spike, (2) operating the combustor fuel feed and distribution as required by speed and altitude for programmed equivalence ratios, (3) regulating the coolant flows in the several circuits to maintain desired skin temperatures, and (4) performing numerous safety and self-checking functions.

STRUCTURES

The SAM configuration was the culmination of the structures research and development effort and reflects the design concepts evolved for the flight engine. The configuration is a Hastelloy X plate-fin monocoque structure with local stiffening as required to resist buckling. The stiffening rings double fuel-injection mani-folds or fuel collector manifolds. The SAM is hydrogen cooled except for a water-cooled cowling outer surface which is part of the wind-tunnel installation. A hydraulic actuator was incorporated in the design to provide for positioning of the variable-geometry inlet.

In as much as the vitiation-heated test facility lacked the oxygen replenishment required for testing with combustion, the SAM was fitted with only a single row of fuel injectors. This model was successfully tested at a nominal Mach 7 true temperature and altitude. In the SAM, as in a complete engine, the aerodynamic interferences are reproduced which cause uneven heating and the thermal expansions that give rise to structural interactions. The SAM investigation demonstrated the capability, by appropriate design, to cope with nonlinearities and other peculiarities inherent in a total engine structure.

THERMODYNAMIC COMPONENT DEVELOPMENT

Aerodynamic development at the component level was done at reduced scale with a view to arriving at preferred component characteristics, and experimental verification thereof, at minimum time and cost. Combustion studies were made by using a quasi-two-dimensional variable-geometry combustion rig provided with separately heated test stream (vitiated and oxygen replenished) and gaseous hydrogen fuel. Subsonic and supersonic combustion modes were investigated in this rig. Combustion efficiencies in excess of 95 percent were shown to be quite attainable, and an initial investigation of the complex inter-related problems of staged injection in diverging supersonic combustion was made. These studies indicated poor efficiencies for supersonic combustion in a diverging duct. This investigation showed a need for further research at full scale and with better simulation.

FULL-SCALE PERFORMANCE ENGINE

The aerothermodynamic integration model (AIM, Figure 5) was the "proof of the pudding" for the aerothermodynamic design of the engine. The engine configuration reflects the aerodynamic contours established in the subscale component program. The engine is constructed from nickel and is water-cooled. Heavy duty, nonflight, laboratory models such as the AIM are commonly referred to as "boilerplate" models, a somewhat misleading term. The thick-plate construction at high heat fluxes necessitated a very sophisticated structural design and placed unusually severe demands on the fabrication technology. For example, zirconium copper was required to form the tip of the cowl leading edge, where the thermal conductivity and high-temperature strength requirements exceeded the capability of nickel 200. Because of stress and dimensional stability requirements, explosive bonding was used for attaching the copper tip to the nickel. Water cooling was elected as a matter of convenience in testing and controlled such that proper simulation of the temperature of the hydrogen-cooled flightweight wall could be obtained at points of importance. Heated hydrogen was used to properly simulate the flight engine combustion and ignition characteristic in the burner. The design had provisions for 266 pressure measurements, 138 temperature measurements, and 5 gas sampling probes. The engine was tested at the NASA Lewis Plum Brook Facility at Mach 5, 6, and 7. The facility was capable of providing nonvitiated, true temperature simulation over this Mach range up to a total pressure of 81.5 atm (1200 psia).

RESULTS

The combustion efficiency levels measured in the AIM are presented in Figure 6. Figure 7 presents the experimental data compared to the predicted performance. The overall conclusions from the AIM model are presented in Figure 8. The total test time is shown in Figure 9.

The SAM model was thermally cycled in the Langley 8-ft. high temperature structures tunnel. The measured heat fluxes and surface temperatures are shown in Figures 10 and 11. Total test time in the tunnel is shown in Figure 12. These tests indicated at the time that the design of regenerative cooled flightweight structure capable of taking variable highly non-uniform heat loads was feasible.

The HRE program was the only program in its day that totally addressed all the issues facing the design of a high Mach number hydrogen cooled supersonic combustion ramjet.

HRE PERFORMANCE GOALS

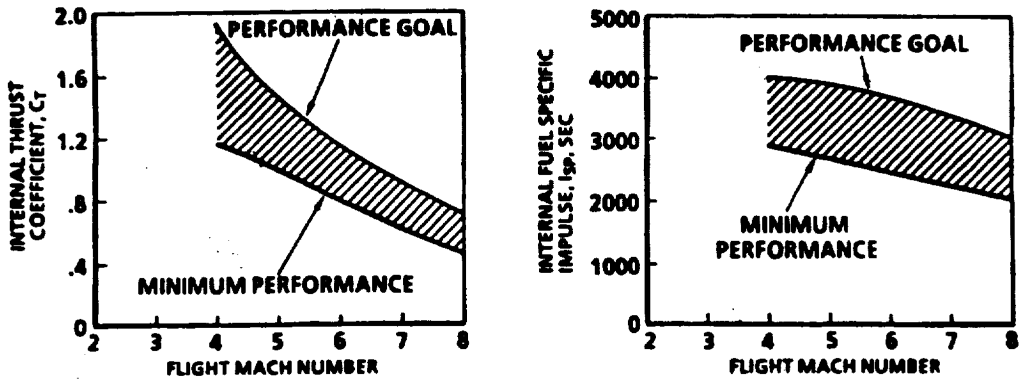
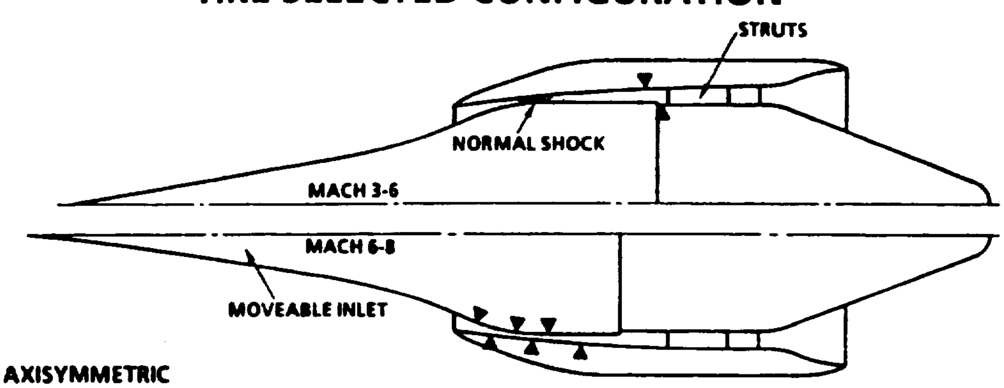


FIGURE 1

HRE SELECTED CONFIGURATION



- **AXISYMMETRIC**
  - INLET DIAMETER = 18.0 IN.
  - EXIT DIAMETER = 25.1 IN.
  - ENGINE LENGTH = 86-91 IN.
- **HIGH INLET CONTRACTION RATIO**
  - VARIABLE GEOMETRY
- **DISTRIBUTED COMBUSTION**
  - SUBSONIC COMBUSTION, M_0 = 3 TO 6
  - SUPERSONIC COMBUSTION, M_0 = 6 TO 8

▲ FUEL INJECTORS

FIGURE 2

HRE SAM ENGINE ASSEMBLY

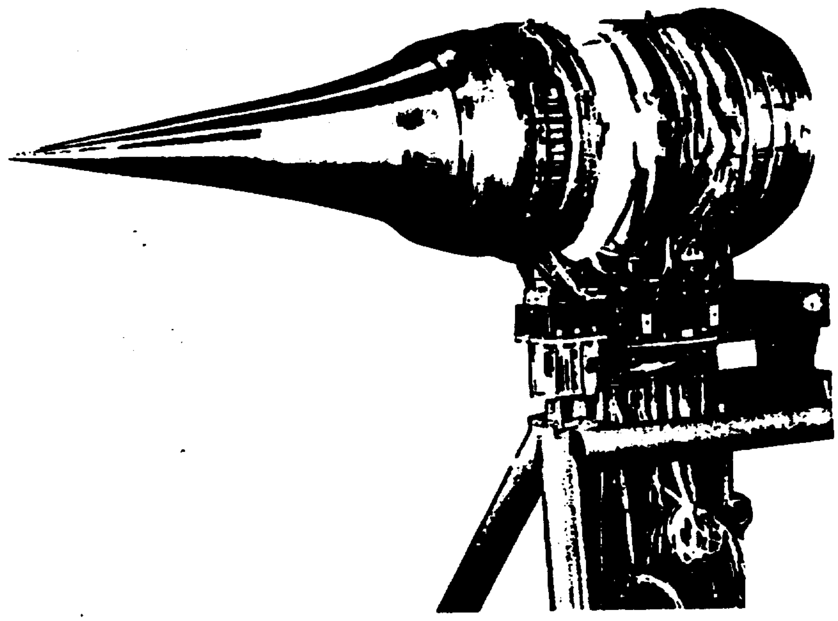


FIGURE 3

FUEL AND CONTROL SYSTEMS

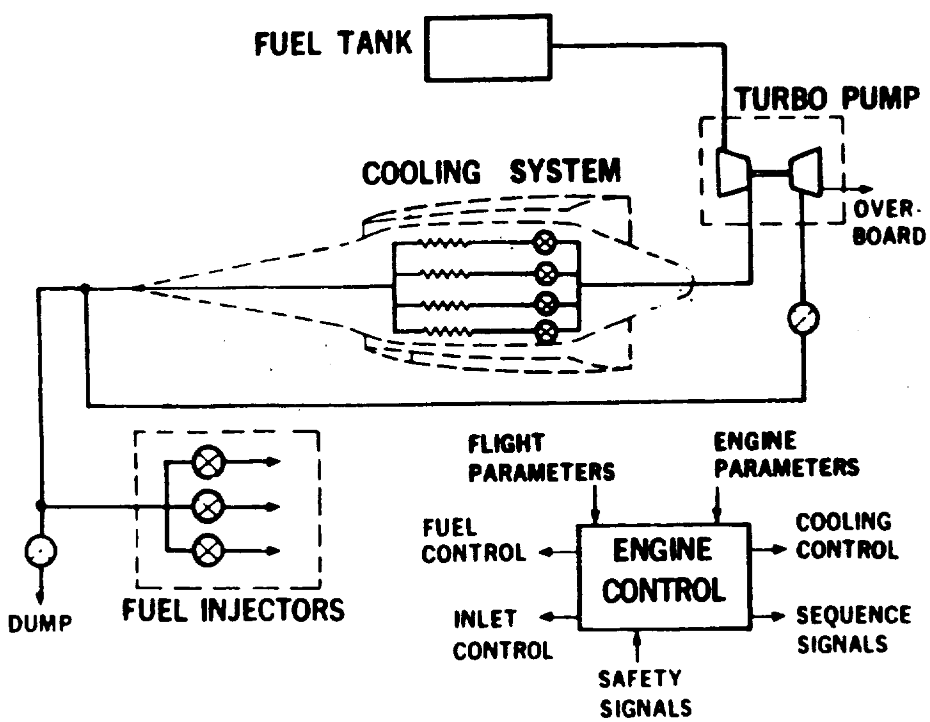


FIGURE 4

AIM FINAL ASSEMBLY

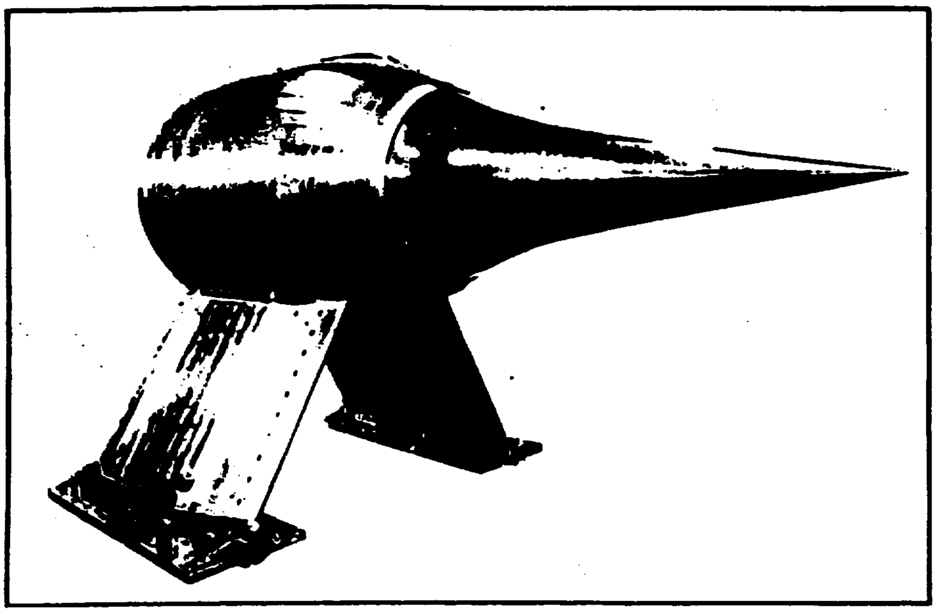


FIGURE 5

MACH 6 SUPERSONIC COMBUSTOR EFFICIENCY

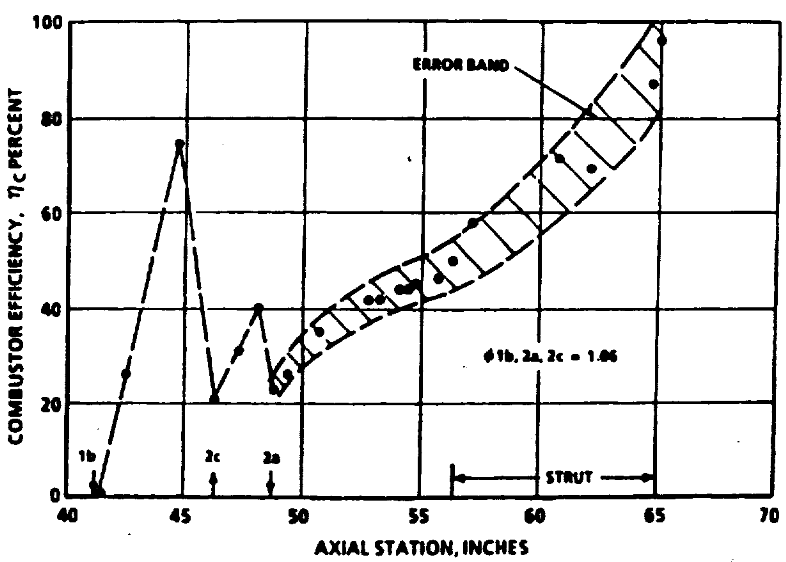


FIGURE 6

CORRECTED MACH 7.25 INTERNAL PERFORMANCE
SUPERSONIC COMBUSTION

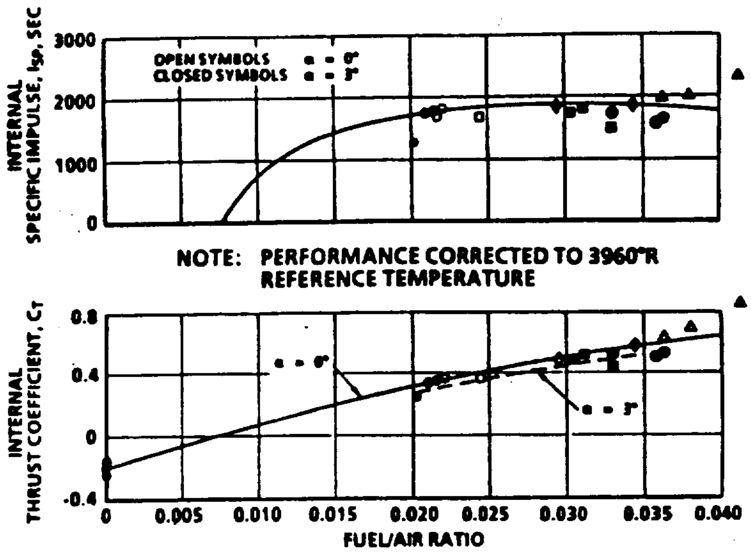


FIGURE 7

AIM INTERNAL PERFORMANCE COMPARISON

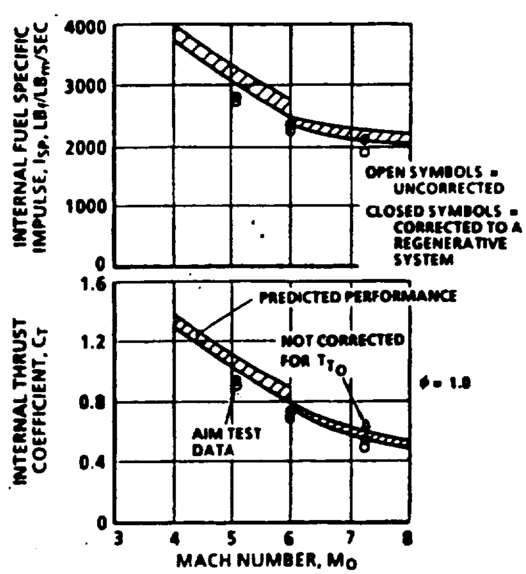


FIGURE 8

TEST SUMMARY

MACH NO.	NO. OF TESTS	TIME AT TEST COND (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

FIGURE 9

HEAT FLUXES

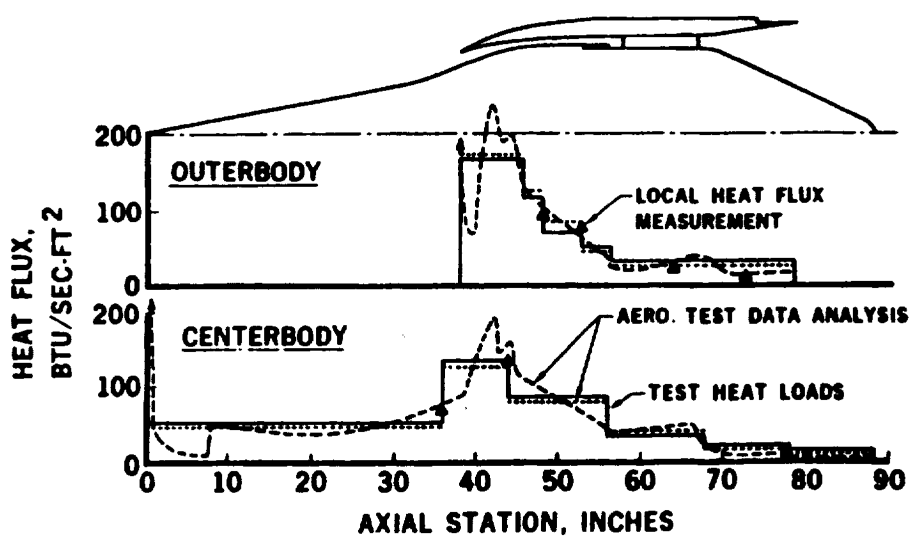


FIGURE 10

SURFACE TEMPERATURES

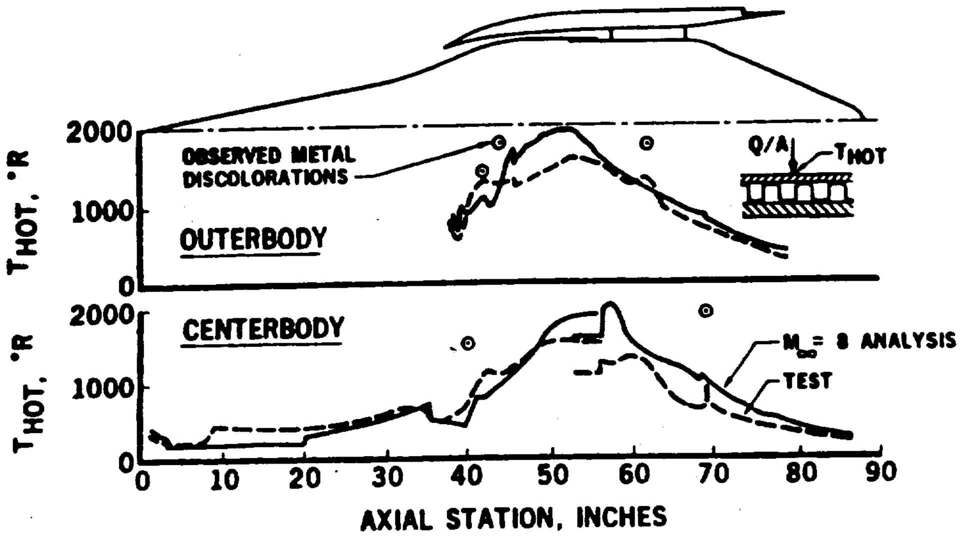


FIGURE 11

WIND TUNNEL TESTS
THERMAL FATIGUE SUMMARY

TUNNEL TOTAL CONDITIONS		NO. OF CYCLES	TIME IN STREAM, SEC	AVG. CYCLE TEMPERATURES		DAMAGE FRACTION, PERCENT
PSIA	°R			T_{MAX}, °R	ΔT, °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.			

FIGURE 12