Preprint Series

MAR. 1976

SUPERSONIC COMBUSTION RAMJET (SCRAMJET) ENGINE DEVELOPMENT IN THE UNITED STATES

PAUL J. WALTRUP
Johns Hopkins University
Applied Physics Laboratory

GRIFFIN Y. ANDERSON
National Aeronautics and Space Administration
Langley Research Center

FRANK D. STULL
Aero Propulsion Laboratory
Wright-Patterson Air Force Base

The 3rd International Symposium
on Air Breathing Engines
Munich, Germany, March 1976

THE JOHNS HOPKINS UNIVERSITY ■ APPLIED PHYSICS LABORATORY
Johns Hopkins Road ∘ Laurel, Maryland ∘ 20810
Operating under Contract N00017-72-C-4401 with the Department of the Navy

SUPERSONIC COMBUSTION RAMJET (SCRAMJET) ENGINE DEVELOPMENT IN THE UNITED STATES

Paul J. Waltrup[†]

The Johns Hopkins University
Applied Physics Laboratory [*]
Laurel, Maryland

Griffin Y. Anderson [††]

National Aeronautics and Space Administration
Langley Research Center
Hampton, Virginia

Frank D. Stull [†††]

Aero Propulsion Laboratory
Wright-Patterson Air Force Base, Ohio

ABSTRACT

This survey of supersonic combustion ramjet (scramjet) engine development in the United States covers development of this unique engine cycle from its inception in the early 1960's through the various programs currently being pursued and, in some instances, describing the future direction of the programs. These include developmental efforts supported by the U. S. Navy, National Aeronautics and Space Administration, and U. S. Air Force. Results of inlet, combustor, and nozzle component tests, free-jet engine tests, analytical techniques developed to analyze and predict component and engine performance, and flight-weight hardware development are presented. These results show that efficient scramjet propulsion is attainable in a variety of flight configurations with a variety of fuels. Since the scramjet is the most efficient engine cycle for hypersonic flight within the atmosphere, it should be given serious consideration in future propulsion schemes.

ZUSAMMENFASSUNG

Es wird ein Uberblick über die in USA erzielte Entwicklung des Staustrahltriebwerks mit Uberschallverbrennung gegeben, der den Entwicklungsverlauf dieser einzigartigen Triebwerkarbeitsweise seit Aufnahme in den frühen 60-er Jahren bis zu den noch im Gange befindlichen Programmen erfasst und, in einigen Fällen, die zukünftige Entwicklungsrichtung einiger dieser Programme beschreibt. Unter diesen Programmen sind die zu nennen, die mit Unterstützung der US-Kriegsmarine, der Nationalen Luft- und Raumfahrtsadministration (NASA), und der US-Luftwaffe vorgenommen werden. Beschrieben werden Einlauf-, Brennkammer- und Düsenkomponentenversuche, Freistrahltriebwerksversuche, analytische Verfahren zur Analyse und Voraussage der Leistungen von Komponenten und Triebwerk, sowie fluggewichtsgemässe Geräteentwicklung, zusammen mit einer Erörterung der jeweiligen Versuchsergebnisse. Aus den letzteren geht klar hervor, dass ein effektiver Staustrahlantrieb mit Uberschallverbrennung in einer Vielfältigkeit von Flugkonfigurationen mit einer grossen Auswahl an Treibstoffen zu erzielen ist und dass ein derartiger Antrieb, da er die bei Hyperschallflug innerhalb der Atmosphäre wirksamste Arbeitsweise darstellt, für Antriebspläne der Zukunft ernst in Erwägung gezogen werden soll.

---

* Operating under Contract N00017-72-C-4401 with the Dept. of the Navy
† Section Supervisor, Supersonic Combustion, Propulsion Group
†† Head, Components Section, Hypersonic Propulsion Branch
††† Chief, Ramjet Technology Branch, Ramjet Engine Division

RÉSUMÉ

On présente un exposé sommaire du développement aux Etats Unis du réacteur à combustion supersonique qui comprends le développement de ce cycle unique de moteur depuis son début dans les premières années 60 jusqu'aux programmes divers d'aujourd'hui et qui, dans certains cas, décrit la direction future de plusieurs programmes. Parmis ces efforts de développement sont à nommer ceux de la Marine Militaire des Etats Unis, de l'Administration Nationale Aérospatiale (NASA), et de l'Armée de l'Air des Etats Unis. On donne des descriptions des essais des composantes de la prise d'air, de la chambre de combustion et de la tuyère, des essais jet-libre du moteur, des techniques analytiques élaborées pour l'analyse et la prédiction des performances des composantes et du moteur, et du développement poids-en-vol des appareils de bord, ainsi qu'une discussion des résultats correspondants. Il en résulte manifestement qu'une propulsion effective par statoréacteur à combustion supersonique s'avère accessible à une variété de configurations de vol avec un grand nombre de combustibles et, une telle propulsion représentant le cycle de moteur le plus efficace pour vol hypersonique dans l'atmosphère, elle doit être pris sérieusement en considération pour projets futurs de propulsion.

РЕЗЮМЕ

В работе дается обзор достигнутого в США развития ПВРД со сверхзвуковым сгоранием, охватывающий разработку этого своеобразного двигателя с его начала в ранних шестидесятых годах до разных еще выполняющихся программ и, в некоторых случаях, описывающий будущее направление развития некоторых из этих программ. К этим относятся программы, реализующиеся с поддержкой ВМС США, Национального управления по аэронавтике и исследованию космического пространства, и ВВС США. Описываются испытания элементов воздухозаборника, камеры сгорания и сопла, свободноструйного двигателя, аналитических методов, разработанных для анализа и предсказания эксплуатационных качеств составных частей и двигателя, и развитие оборудования полетного веса, вместе с обсуждением соответствующих результатов. Из последних видно, что достижимо эффективное движение при помощи ПВРД со сверхзвуковым сгоранием в разных конфигурациях полета и, так как такой способ движения является самым эффективным циклом работы при гиперзвуковом полете в атмосфере, он должен быть предметом серьезного рассмотрения в будущих проектах движения.

INTRODUCTION

The superiority of supersonic combustion ramjet (scramjet) engines over other, more conventional, types of engines, viz., rockets, turbojets, and subsonic combustion ramjets, for hypersonic flight within the atmosphere has been recognized since the early 1960's (Refs. 1-3). Although no United States developed scramjet has been flight tested to date, the performance potential and general behavior of this engine with a variety of fuels, and in a variety of configurations applicable to manned and unmanned vehicles, are well understood due to the numerous programs that have been supported by the United States Navy, the National Aeronautics and Space Administration (NASA), and the United States Air Force.

The reasons for the scramjet's superiority at hypersonic speeds within the atmosphere can be appreciated by considering Fig. 1. Here, engine fuel specific impulse, Isp, is shown as a function of flight Mach number, M_0, for a rocket using liquid hydrogen and oxygen, a turbojet using kerosene, and a ramjet and a scramjet using either hydrogen or a storable liquid borane fuel. For speeds up to about Mach 3, the turbojet (including fanjet variations) produces more thrust per pound of fuel burned than does the ramjet due to its inherently higher thermal efficiency. However, at about this speed the turbine inlet temperature, which has increased steadily with M_0, reaches its maximum allowable level for structural integrity. At higher speeds this constraint means that the combustor must be operated so lean that cycle efficiency drops rapidly. Then the ramjet, which contains no temperature-limited turbomachinery, becomes more efficient. Improved techniques for turbine blade cooling and/or the addition of hardware to cool the air between compressor stages may permit somewhat higher turbojet speeds, but the gains will come slowly and at considerable cost and weight penalties compared to the simple ramjet engine.

In the M_0 = 3-5.5 range, the ramjet is more efficient than the scramjet because the smaller loss in total pressures in its combustor more than compensates for the loss caused by the strong "normal" shock wave in its inlet. However, the normal-shock loss increases with flight speed, and at M_0 = 5.5-6.0 these losses tend to balance in the two cycles. For $M_0 \geq 6$, the scramjet is clearly superior. The lower static temperatures and pressures in the scramjet combustor are beneficial not only from a chemical kinetics viewpoint (greater oxidation, as opposed to molecular dissociation, occurs) but also because heat transfer and structural loads are greatly reduced.

Because a ramjet or scramjet requires a high forward speed to compress the air and produce net thrust, an auxiliary propulsion system must be provided to accelerate it to its minimum operating speed. For use in an aircraft a scramjet would be complemented by a turbojet, which might be a completely separate engine or might share some components (e.g., the air inlet, fuel system, and nozzle) with the scramjet. In missile applications the boost

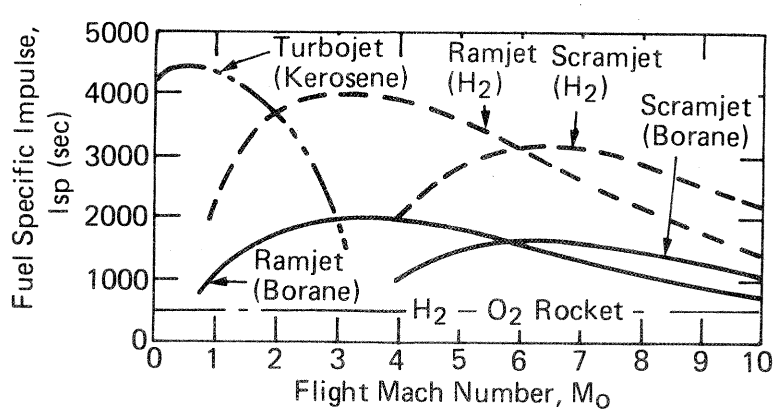


Fig. 1 Typical Engine Fuel Specific Impulses

engine would be a rocket, either as a separate system or integrated with the scramjet. (Many variations of these combined engine systems have been proposed; see, e.g., Ref. 4). In either case it is advantageous to begin the scramjet operation at the lowest possible speed (due to the relatively low top speed of the turbojet or the low specific impulse of the rocket). Fortunately, an appropriately designed scramjet (see Refs. 5 and 6) can also operate efficiently as a (subsonic combustion) ramjet in the lower Mach 3-6 speed range by having a normal shock located at the combustor entrance, so that the combustion process begins in subsonic flow. Since the scramjet engine generally does not have an area restriction (geometric throat) downstream of the combustor (as contrasted to the conventional subsonic combustion ramjet), the flow in the combustor accelerates continuously, passing through the sonic point before entering the nozzle. At intermediate speeds ($M_0 \simeq 6$ to 8) the precombustion shock weakens to oblique waves, the heat release begins at a Mach number of 2 or more, and the flow remains supersonic through the combustor and nozzle. At some higher speeds, the effects due to heat release may be insufficient to support a precombustion shock, and the combustion process will be "shock-free" and supersonic throughout (Refs. 5 and 6). An engine which combines these processes (i.e., subsonic and/or supersonic combustion) is commonly called a dual-mode scramjet.

With this introduction, the rest of this paper will be devoted to summarizing the scramjet development work supported either by the U. S. Navy, NASA, or U. S. Air Force and to projecting, in some instances, the future direction of these programs.

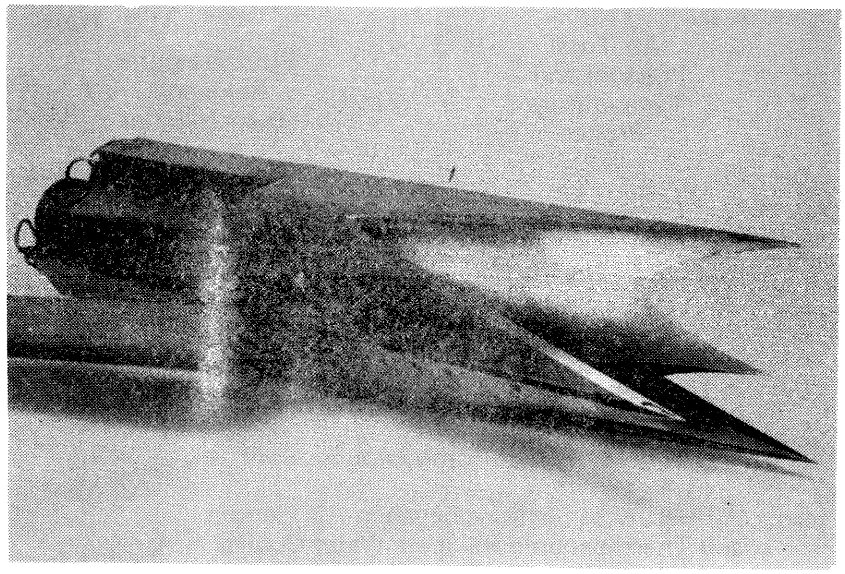
U.S. NAVY SCRAMJET DEVELOPMENT

The U. S. Navy scramjet development program has been carried out under the direction of the Johns Hopkins University Applied Physics Laboratory (APL) since its inception in the early 1960's. Unlike the NASA work and most of the U. S. Air Force programs, in which development of propulsion systems for manned earth-to-orbit shuttle or hypersonic transports were and still are the primary objectives, the U. S. Navy work has been directed toward development of unmanned, volume-limited systems, i.e., missiles, compatible with shipboard handling and launching. Because the desired system is volume-limited, storable liquid fuels, rather than hydrogen, are necessary, and the propulsion system/airframe should be passively, rather than regeneratively, cooled.

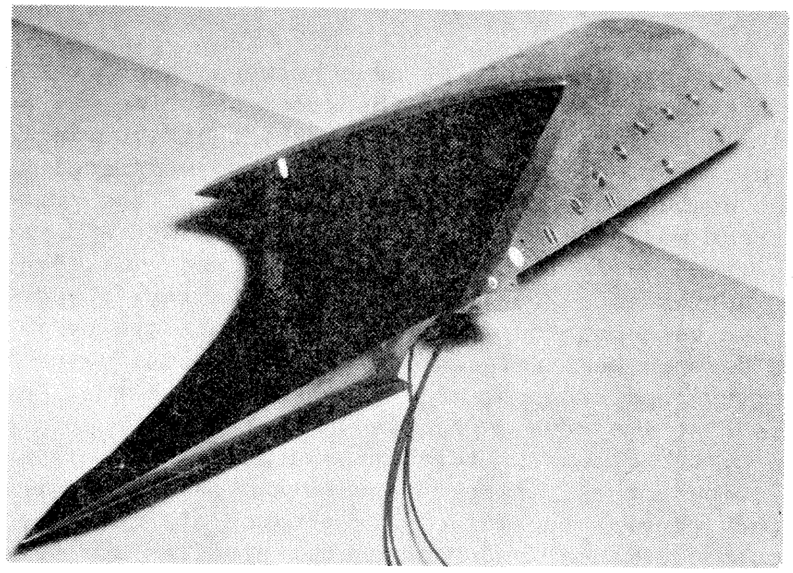
The scramjet propulsion work at APL grew out of earlier work on external burning, where net positive thrust on a small "external ramjet" model in a Mach 5 air stream was first demonstrated in 1959 (Ref. 7). After demonstrating, theoretically, the potential of storable-liquid-fueled scramjets in 1960 (Ref. 1), work was directed toward development of individual components, i.e., inlets, fuel injectors, and combustors, and the experimental and/or analytical techniques and hardware necessary to determine their respective performance. After initial component development, a heavyweight free-jet engine was designed and built and testing of it initiated in 1968. Since then, the APL program has benefited from a continuity of effort, enabling continual refinements or improvements in component design, testing, and analysis to be made and incorporated and tested in the free-jet engine design. The remainder of this section summarizes the experimental and analytical techniques developed, results obtained to date, and some problems that remain.

Component Development and Analysis

Inlet Development: The design criteria used to develop the inlet in this volume-limited system are: a) to capture the maximum amount of air possible for a given frontal area (which is necessary to achieve reasonable acceleration capabilities since the difference between inlet and exit stream thrusts at hypersonic speeds is small); b) to have no moving parts, e.g., a translating centerbody (throat); c) to minimize leading edge heat transfer (no active cooling) and drag; and d) to channel the flow coming out of the inlet in such a manner as to provide sufficient room for internal stores. One design which fulfills these requirements is an inverted internal axisymmetric compression flowfield split into three (or four) quadrants (modules) as shown in Fig. 2. The particular inlet shown in Fig. 2 is a scale model of the three-module (trifurcated) inlet design used in wind tunnel tests to obtain performance between Mach 4 and Mach 8. A four-module (quadrifurcated) model has also been tested from Mach 4 to Mach 10. In both cases, the measured performance exceeded the design goals initially set at each of the conditions tested. Also, in each of these designs, the "external compression" of each module allows the inlet to start over a wide range of conditions without changing its throat area and, except for the tips and "crotch" regions, the highly swept leading edges minimize heat transfer and drag. Furthermore, since the flow is split into three (or four) modules, sufficient space is left underneath and between the inlet, combustor, and nozzle modules for internal stores.



(a) 3-Module Inlet Model



(b) 4-Module Inlet Quadrant Model

Fig. 2 Photographs of APL Scramjet Inlets

Combustor Development: The work at APL has included development of the connected-pipe experimental techniques and hardware necessary to accurately measure combustion efficiency (η_c), wall skin friction (C_f), heat transfer (Q_w), and pitot pressure and chemical composition profiles in the exit plane of the combustor, as described in Ref. 8. The most important of these, the determination of η_c by steam calorimetry, is schematically illustrated in

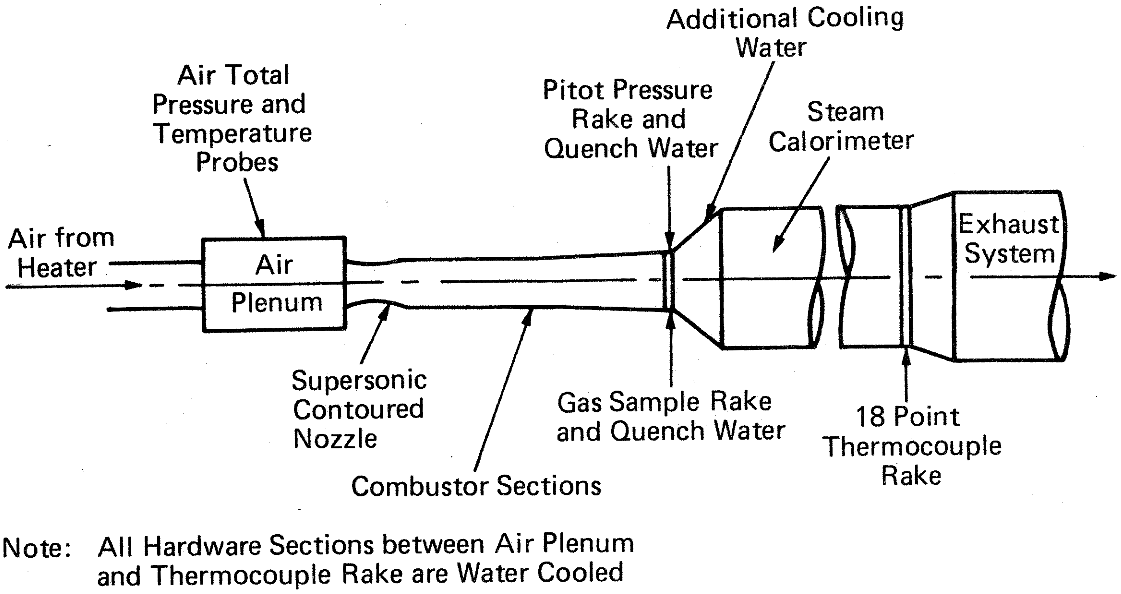


Fig. 3 Schematic of APL Connected-Pipe Combustor Apparatus and Steam Calorimeter

Fig. 3. An energy balance is made between the throat of the supersonic air supply nozzle and the exit of the steam calorimeter, where a multipoint (usually 18 distributed on equal areas) thermocouple rake is located. The supersonic nozzle, combustor sections, and steam calorimeter are water-cooled. The combustion reactions are quenched at the exit plane of the combustor by instream water injection from pitot-pressure and gas sample rakes and the flow is given sufficient time to diffuse and equilibrate in the calorimeter. All but approximately $\pm$ 3% of the energy actually added to or lost from the system can be accounted for. Combustion efficiency is determined by dividing the deduced heat release by the theoretical value for combustion to equilibrium conditions at the combustor exit plane. An accurate knowledge of $\overline{C}_f$ is also important since wall shear losses represent a significant portion of the combustor total pressure (p_t) loss in a supersonic flow. Accurate combustor-exit pitot-pressure and gas sample profiles are necessary since they provide local fuel-air/product-air composition distributions (fuel injector performance) and Mach number profiles and serve as redundant measurements to steam calorimetry for determining combustor performance. They also provide initial profiles for the exit-nozzle optimization techniques discussed later.

A number of axisymmetric combustor geometries, lengths, inlet-to-exit area ratios and fuel injector arrangements have been tested at combustor inlet Mach numbers (M_{ci}) of 1.62, 1.98, and 3.24 and stagnation temperatures (Tt_a) between 1100°K and 2500°K, all at simulated flight altitudes above 18 km. The configuration shown in Fig. 4 has given the best overall performance using autoignitable liquid fuels. It has a step increase in area just downstream of the fuel injector, followed by a constant area cylinder and a conical section, with an overall area ratio of 2.0. This is not necessarily the best configuration when a gaseous fuel such as hydrogen is used (see, e.g. Ref. 9). Fuel injection is from ten 0.762-mm-dia. holes equally spaced circumferentially and normal to the air stream and provides an adequate fuel-air distribution for reasonable combustion without the necessity of in-stream injection, at least for combustors with an inlet diameter similar to that tested (6.96-cm-dia.). Typically, η_c's in excess of 80% at a number of fuel-air equivalence ratios (ER's) have been achieved with fuels such as triethyl-

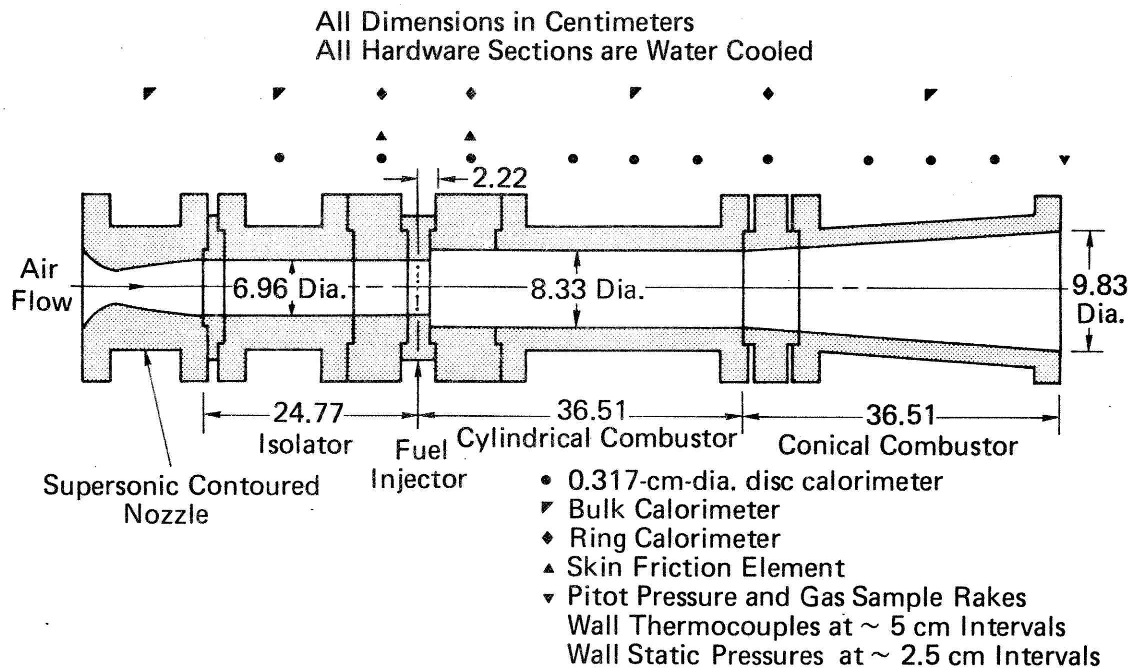


Fig. 4 Schematic of APL Direct Connect Combustor Hardware and Instrumentation

aluminum (TEA) and HiCal 3-D (principally ethyldecaborane). The step, in addition to acting as a flameholder, helps to isolate or limit the interaction of the precombustion shock system with the inlet flowfield of the engine by reducing the maximum pressure rise along the wall at the combustor entrance, thereby reducing the length of air duct needed to separate the combustor and inlet. An extensive amount of work has been completed in this area and published in Refs. 10-13.

Extensive wall heat transfer measurements, both local and bulk, have been made in these combustors. The data have been correlated to obtain bulk combustor heat flux as a function of heat release for both liquid fuels and hydrogen. A modified Reynold's Analogy is then used to obtain a bulk combustor wall skin friction coefficient, $\overline{C}_f$, as a function of heat release. A typical example is shown in Fig. 5. The curves shown are specific for the test conditions indicated, but the characteristic of increasing $\overline{C}_f$ with increasing heat release is characteristic of supersonic combustors and is indicative of the large combustor total pressure loss associated with the heat release. A more detailed description of the experimental data and data reduction technique is given in Ref. 8.

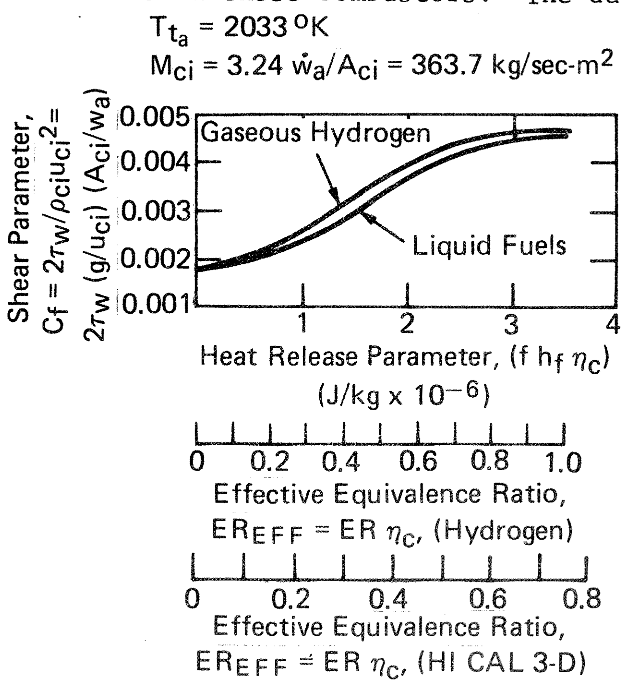


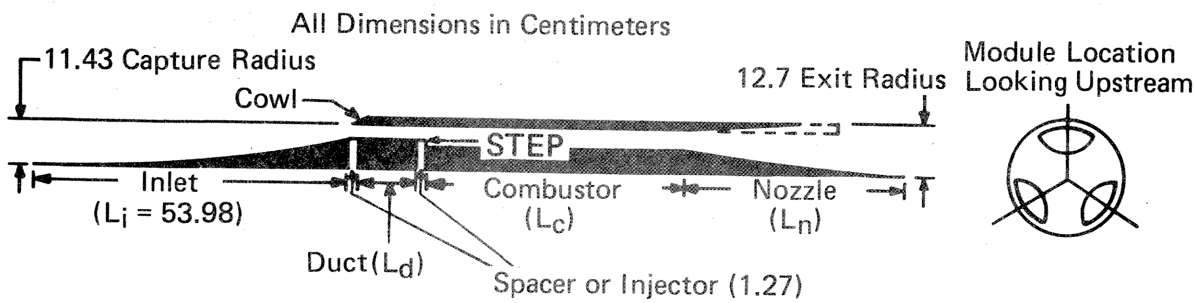
Fig. 5 Deduced Combustor Shear Parameters

Combustor Analyses: The pseudo-one-dimensional analysis of supersonic combustors is more complex than for conventional ramjet or rocket combustors because there is no exit nozzle throat (sonic point for one-dimensional calculations). One can integrate the one-dimensional conservation equations between the combustor inlet and exit assuming that the combustor wall pressure-area distribution for any particular set of conditions is known. However, only the special cases of a constant-area combustor or a combustor shaped for constant-pressure combustion can be treated without knowing from experiments what the wall pressure-area relationship is, which limits its usefulness for general design purposes. A more useful method is to assume a Crocco-type wall pressure-area relationship, i.e., pA^{ϵ} = constant, where ϵ is determined from entropy constraints, and integrate the one-dimensional equations of motion with heat addition between the combustor inlet and exit. This method, developed by Billig at APL (Refs. 5 and6), provides a means of determining combustor performance (i.e., combustion efficiency, wall pressure-area distributions, and/or combustor exit conditions) without an a priori knowledge of the wall pressure-area relationship for a given set of combustor inlet conditions and given amount of heat addition, and also prescribes the strength of the precombustion shock system present at the entrance of the combustor. Agreement of this method with experiments using a variety of combustors and fuel injectors has been very good (see, e.g., Refs. 6, 10, and 14) to date and the method has proven to be a powerful tool in the design of scramjet propulsion systems and interpretation of free-jet experimental data at APL. In addition to this simplified approach, other more sophisticated, i.e., three-dimensional analyses are being formulated but are still in the developmental stages (see e.g., Refs. 15-17) because of the phase changes, discontinuities (shock waves), three-dimensional aerodynamics with heat addition, equilibrium and non-equilibrium chemical kinetics, turbulent transport properties with heat addition, and numerical complexity involved. Ultimately, these techniques may evolve to the point where they are useful tools in the design of scramjet components.

Nozzle Development and Analysis

Development of the analytical techniques needed to optimize the exit-nozzle design for missiles has been in a continuing state of development since the late 1960's. Most of the work has been carried out by Propulsion Sciences Inc. (Refs. 18 and 19) with direction from APL, and owing to the significant computational effort involved, only planar or axisymmetric solutions have been completely developed to date. (The three-dimensional version is currently being checked out.) Some of the features of this analysis are that it accounts for real gas (in thermochemical equilibrium) and viscous effects, allows either uniform or non-uniform initial conditions with or without particles (fuel droplets) to be used, and, for a given set of initial conditions and nozzle inlet-to-exit area ratio, will compute the nozzle contour and length which will yield the maximum thrust (not necessarily uniform exit conditions).

Preliminary testing of an axisymmetric contoured nozzle, designed with the analysis of Refs. 18 and 19, and tested in the connected-pipe combustor set up just described (Fig. 3) using the M_{ci} = 3.24 air supply nozzle ($T_{t_a} \simeq$ 2200°K, $P_{t_a} \simeq$ 3.0 MN/m²) and HiCal 3-D fuel, has recently been completed, and comparisons with the theoretical predictions initiated. Without burning, the measured and predicted difference between inlet and exit stream thrusts are within ± 6%. With burning, large gradients due to combustion-induced compression and expansion waves are present and the analysis is being modified to handle these. Another design based on this analysis has been tested in the free-jet engine and is discussed in the following section.



Model	L_d	L_c	L_n	Exit Nozzle	A_{ex}/A_{in}*
Taper	0.00	55.88	22.85	15° Conical	1.0000
Step	0.00	55.88	22.85	15° Conical	1.0000
Long-Isolator-Taper	34.29	55.88	22.85	15° Conical	1.0000
Long-Isolator-Step	34.29	55.88	22.85	15° Conical	1.0000
Short-Isolator-Step	8.89	46.36	38.10	Contoured	1.2346

*Model Exit-to-Inlet Area Ratio

Dashed Lines Indicate Conical Nozzle

Taper — Tapered Combustor
Step — Step Combustor

Fig. 6 Schematic of APL Free-Jet Engines

Engine Model Tests in Free-Jet Facilities

Engine models of the type shown in Fig. 6 have been tested since 1968 at nominal free-stream Mach numbers, M_0, of 5.0, 5.8, and 7.0. The engine comprises a three-module, 22-cm-capture-diameter, inlet (discussed previously) three isolator air ducts (except for the step and taper configurations) and combustor modules (either tapered or stepped) which are semi-elliptical in cross section, and a three-module conical or contoured exit nozzle. The model is an uncooled thick-walled, heat-sink structure designed for ground test purposes. Except for the inlet's stagnation regions, the walls are 1.11-cm-thick and made of nickel 200. The spikes are molybdenum-disilicide-coated molybdenum and crotch region columbium-disilicide-coated columbium. The maximum outside diameter of the engine, including a 0.16-cm-thick stainless steel skin, is 25.4 cm.

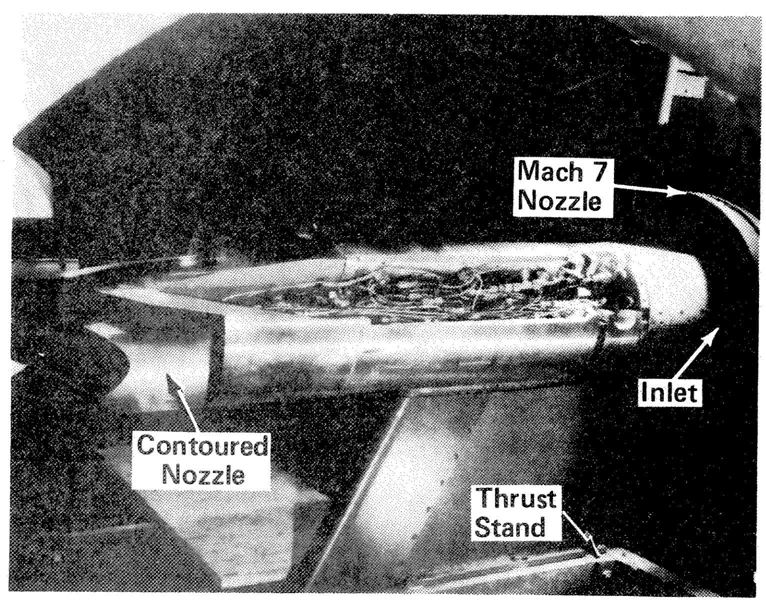


Fig. 7 Scram Engine with Contoured Exit Nozzle Installed on Thrust Stand in APL Free-Jet Facility with External Skin Removed

In each case, the engine is mounted on a sting support which resides on a thrust balance carriage as shown in Fig. 7. An independently mounted windscreen surrounds the sting and the fuel, water, and instrumentation lines to the engine, so that aerodynamic forces associated with this hardware are not transmitted to the force balance, allowing resolution of the mean thrust (or drag) to within ± 0.5 kg. A water-cooled rake (not shown) containing cone-static and pitot-pressure probes and gas sample probes is mounted in the nozzle exit plane, and numerous static pressure taps and thermocouples are located along the inner surfaces of the model along with a few water cooled, 0.317-cm-dia., disc-type heat flux calorimeters.

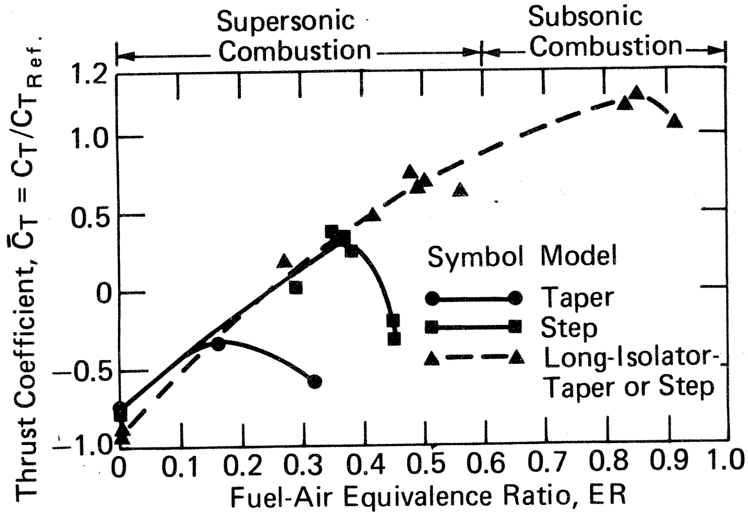


Fig. 8 Thrust Coefficient of APL Free-Jet Engine at Mach 5.0

The Mach 5.0 and 5.8 free-jet tests were run at the Ordnance Aerophysics Laboratory in Dangerfield, Texas before it shut down. In these tests, the air was preheated by a storage heater to 1000-1100°K and then topped with a hydrogen vitiation heater with oxygen makeup to 1333°K for the $M_0 = 5.0$ tests and 1922°K for the $M_0 = 5.8$ tests. In both cases, P_{t_a} was limited to 2.07 MN/m², limiting the tests to high altitude flight conditions.

Four configurations and a number of fuel injector arrangements were tested at these conditions. As demonstrated in the direct-connect tests, normal wall injection (in this case seven 0.58-mm-dia. holes), again proved to be superior to other injector arrangements. Figures 8 and 9 present typical results of engine thrust coefficient (normalized by a reference value), $\bar{C}_T$, as a function of equivalence ratio, ER, using HiCal 3-D fuel for the M_0 = 5.0 and 5.8 conditions, respectively. In Fig. 8 (M_0 = 5.0), the relatively strong effects of inlet-combustor coupling, are readily apparent. Disturbances caused by the combined effects of injection and combustion increase in strength as ER is increased, causing a pressure rise on the inlet compression ramp and a corresponding drop in $\bar{C}_T$. Each change made in the combustor geometry to alleviate this problem resulted in extending the permissible ER. Transition from supersonic combustion to subsonic combustion at ER ≃ 0.6 for the "long-isolator-taper and step" designs proceeded without any observable difficulties. At M_0 = 5.8 (Fig. 9), the disturbances at the combustor inlet are considerably weaker, and no adverse inlet conditions were experienced with any of the

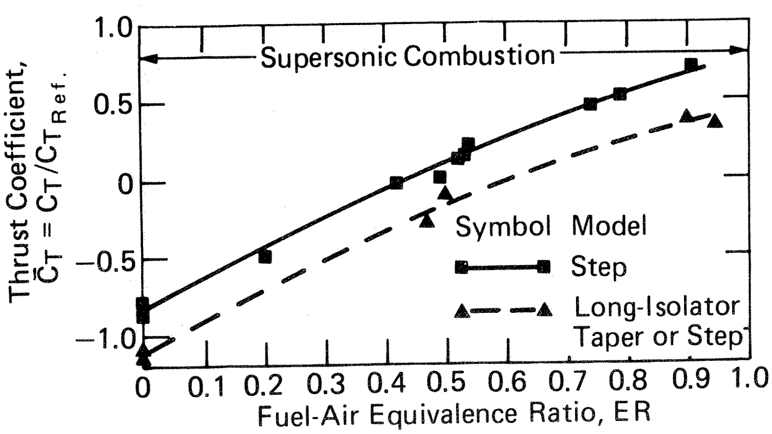


Fig. 9 Thrust Coefficient of APL Free-Jet Engine at Mach 5.8

combustor configurations.

The $M_0 = 7.0$ tests were made in the free-jet facility at the APL's Propulsion Research Laboratory, a portion of which is shown in Fig. 7. In these tests, the air is heated by a DC-electric-arc heater and the resulting stagnation conditions of $T_{t_a} \simeq 2200^\circ K$ and $P_{t_a} \simeq 3.8\ MN/m^2$, again, are representative of high altitude flight. A thorough description of the free-jet facility is given in Ref. 10. Only the step configurations (step, long-isolator-step, and short-isolator-step) were tested at these conditions because of their effectiveness in reducing the inlet-combustor interaction problem and because the step acts as a flame-holding device in cases where fuels other than pure boranes are used.

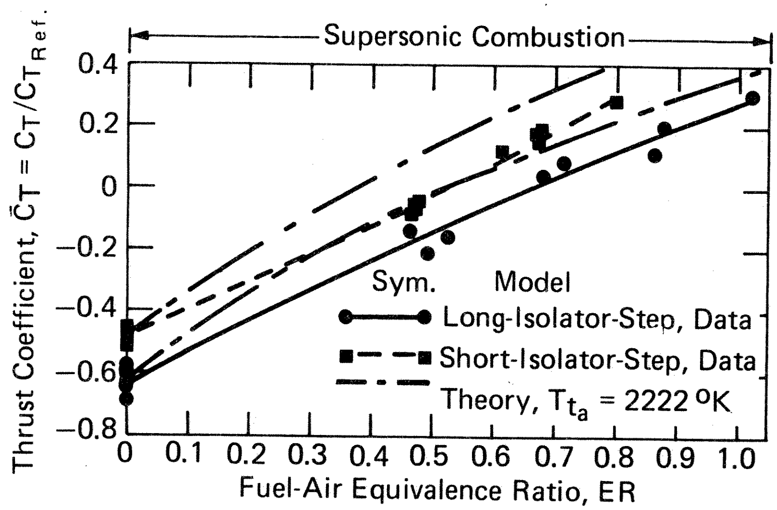


Fig. 10 Thrust Coefficient of APL Free-Jet Engine at Mach 7.0

Figure 10 presents $\bar{C}_T$ versus ER for the long-isolator-step and short-isolator-step configurations using HiCal 3-D fuel and is representative of results obtained in all of the tests at these conditions. The particular points to be made in this figure are that a) in the short-isolator-step configuration, the length of the isolator duct has been minimized for these particular test conditions based on the analysis developed in Refs. 10-13 and the exit nozzle contoured (based on the analysis of Ref. 18 and 19) and area ratio increased to optimize the performance of the free-jet engine, and b) the engine cycle and component analyses have been refined to the point where fuel combustion efficiencies and, therefore, overall engine performance, can be deduced by comparing theoretical predictions with experimental results. The solid curves in Fig. 10 are the theoretical predictions including heat loss to the engine which can be substantial with a heat-sink type engine. The close correspondence of the calculated and measured $\bar{C}_T$'s at ER = 0 for both configurations clearly indicate that the assumptions made in the theoretical engine cycle analysis are correct. In addition, nearly all of the expected rise in net force with the "optimized" engine, due mainly to increasing the exit nozzle area ratio and contouring it, was realized, verifying the nozzle optimization techniques developed at least at the conditions tested. For example, the fuel required for $\bar{C}_T = 0$ in the long-isolator-step configuration was reduced by approximately 27% from an ER of 0.650 to 0.475, in the short-isolator-step configuration, versus 28% predicted by the theoretical analysis.

Future Development

Although much has been accomplished in the design and testing of this volume-limited hypersonic propulsion system, there are a number of areas where further development is needed. These include, but are not limited to, testing the combustor, contoured nozzle, and free-jet engine at low altitude/Mach number flight conditions where high performance is essential; developing lower cost, less toxic alternatives to the borane fuels and fuel blends; and developing uncooled liner materials compatible with the conditions present in the combustor and nozzle. Some preliminary work has already been completed in the fuels area (Ref. 20) where it is expected that ignition aids, such as ignitors and/or fuel or oxidizer pilots, will be necessary to achieve reasonable fuel performance with principally heavy hydrocarbon fuels. Some materials testing has also been initiated, and to date, co-deposits of silicon carbide and pyrolytic graphite appear to be promising. However, an extensive amount of testing, as well as development of fabrication techniques (bonding, etc.), is still needed.

NASA SCRAMJET DEVELOPMENT

Hypersonic Research Engine Project

In 1965, NASA undertook an ambitious step to advance scramjet technology as a means of propulsion for manned vehicles with the Hypersonic Research Engine Project (HRE) which was aimed at testing a complete, regeneratively cooled, flightweight scramjet engine on the X-15 research airplane. The opportunity for flight test was lost when the X-15 program was terminated in 1968, but significant advances in flightweight engine structures and systems were accomplished by the HRE Project because of the initial goal of flight tests on the X-15.

The general concept of the HRE is shown in Fig. 11. The engine was designed and developed by H. Lopez and others of Garrett Air Research. A Structural Assembly Model (SAM) which was a complete flightweight, regeneratively cooled scramjet was successfully ground tested in the Langley 2.44 m, High Temperature Structures Tunnel (Ref. 21 and 22). Aerodynamic surfaces of the engine consisted of a Hastelloy skin backed by an integral offset-fin ring-stiffened heat exchanger through which the hydrogen fuel was circulated to provide cooling of the structure before injection in the combustor. The cowl leading edge, with a 0.762-mm radius, was cooled to survice a maximum heat flux of 20.5 MN/m^2 by hydrogen flowing perpendicular to the leading edge from the outer to the inner surface. The tests of the SAM verified the structural and cooling design of the engine for conditions simulating Mach 7 flight and included operation with reduced cooling flow to achieve structure temperatures appropriate for even higher flight speeds.

Key components of the engine systems necessary for flight tests were also developed during the HRE project (Ref. 23). These included hot fuel control valves, a fuel pump, a digital engine control system, and so on. Many of these components were built and bench tested before termination of the X-15 program. The design of the SAM, the development of practical manufacturing techniques, and success-

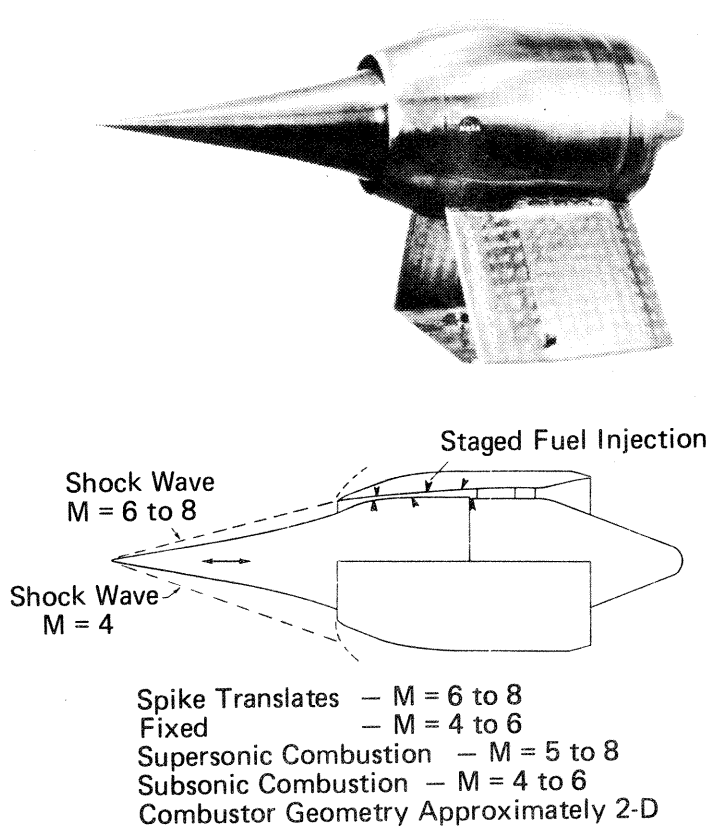


Fig. 11 Photograph and Schematic of NASA HRE.

ful ground tests of the SAM plus the development of major flight system components represent a major advance in technology toward practical application of scramjet propulsion.

Thrust performance of the HRE concept was measured with a separate water-cooled heavy-walled engine called the Aerothermodynamic Integration Model (AIM). The AIM engine was tested in the Hypersonic Tunnel Facility at the Plum Brook Station of the Lewis Research Center under conditions simulating flight at Mach 5, 6, and 7 (Refs. 24 and 25). Fuel specific impulse based on internal thrust measured at the different Mach numbers of these tests for stoichiometric combustion is shown in Fig. 12 and compared with the performance goal for the HRE. The measured performance is quite close to the performance goal. The upper edge of the shaded band indicates the level of performance that might be achieved by a series of different engines designed for operation at only one flight Mach number and without performance penalties for inlet, combustor, and nozzle losses. Measured AIM performance approached 70 percent of this ideal level and constitutes an important demonstration of the feasibility of producing good thrust over a range of flight speed with both subsonic and supersonic combustion operation in a single engine.

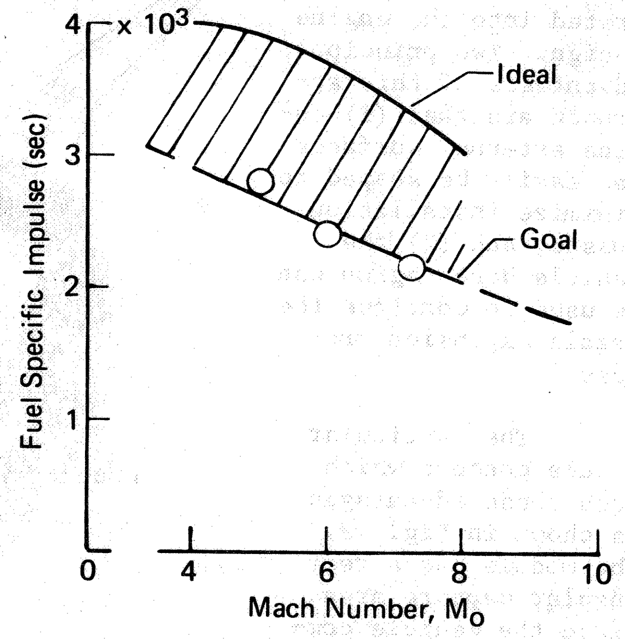


Fig. 12 Specific Inpulse of NASA HRE Based on Internal Thrust

Airframe-Integrated Scramjet Research

The successful AIM and SAM tests amply fulfill important parts of the original HRE program goals, that is, to demonstrate good internal thrust over a range of flight speed and to develop practical engine structures, regenerative cooling, and manufacturing techniques. The experience gained through the HRE project provided a firm basis for the current NASA effort in hypersonic propulsion. While internal thrust performance was the principal consideration in selecting the HRE configuration, integration of the engine with the vehicle is also necessary to achieve high installed performance (internal thrust minus external drag), particularly as flight speed increases. The current NASA program is focused on defining practical airframe-integrated scramjet concepts (see e.g., Ref. 26). The next section describes engine-airframe integration for hypersonic flight and shows how integration can improve vehicle performance.

Scramjet Module Concept: The first fact apparent for an airbreathing vehicle at M_0 = 10 to 12 is that the engine needs to use nearly all the airflow between the undersurface of the vehicle and the vehicle shock wave. The requirement for maximum airflow means an inlet capture area with an annular shape. If this annular area is split into smaller rectangular units or modules, the engine becomes a number of identical pieces which are the right shape to fit in a ground facility. The whole engine is made by placing several of the modules side by side. When the engine is treated in this way, the vehicle forebody performs part of the engine inlet compression process, the aftbody takes over part of the nozzle expansion process, and the entire undersurface of the vehicle is integrated into the engine design. Two principal advantages of this approach are that (1) engine external surfaces can easily be shaped to minimize installation losses, and (2) the vehicle base region can be used to continue the nozzle expansion process.

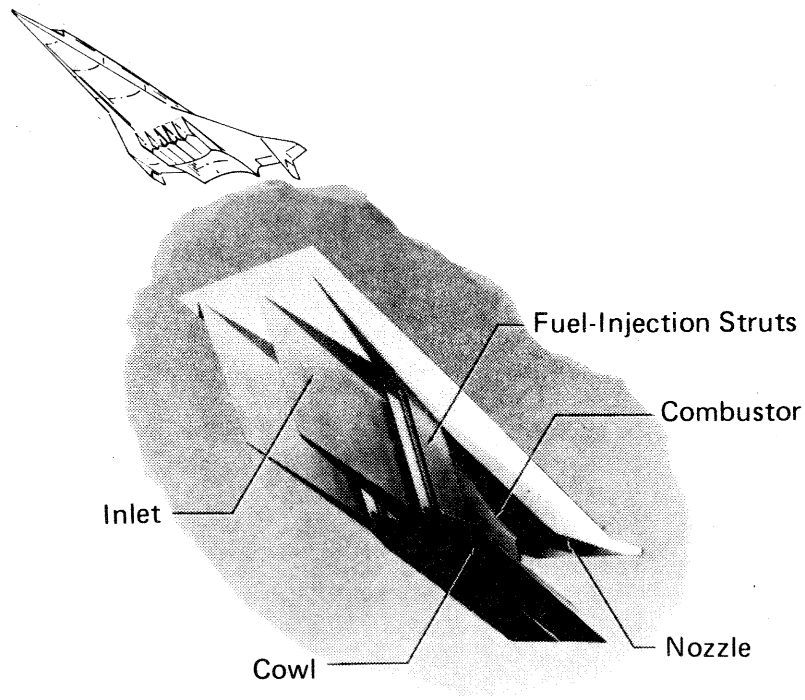


Fig. 13 Airframe-Integrated Supersonic Combustion Ramjet (NASA)

The particular module concept which uses these advantages is shown in Fig. 13. The module has a rectangular capture area. Since the vehicle compresses flow in the vertical direction, the module inlet has wedges to compress the flow horizontally. This approach reduces the degree of change in the inlet flow field that occurs with changing flight speed and makes fixed geometry feasible. Sweep of the compression wedges and a cutback cowl provide spillage which allows the inlet to start at low flight speeds. The inlet compression process is completed by wedge-shaped struts located at the throat which block about 60 percent of the flow cross section. The struts make the inlet shorter, and also provide multiple planes for fuel injection and thus shorten the combustor. The combustor is diverging and employs varying amounts of parallel and perpendicular fuel injection from the struts to control the distribution of heat released in the combustor as flight speed changes. These features should combine to give good performance with a significant reduction in the cooling requirement and the wall pressure level compared with those of a constant-area combustor with wall fuel injection.

The Langley Research Center is currently engaged in a program to establish the performance potential of this scramjet module concept. The program includes research on the module inlet and combustor to establish background for the design and testing of complete subscale scramjet engine modules in ground facilities. In the subscale engine work, maximum advantage is taken of heat sink cooling, single purpose design, and other techniques to limit the cost and fabrication time for hardware. Analysis of the flight engine structure is under way in parallel with this effort to apply HRE technology to the integrated scramjet module configuration. The goal of the combined subscale engine testing and flight structure effort is to define an integrated scramjet module configuration with attractive performance potential which can be fabricated with a flight-weight regeneratively cooled structure and performance tested in ground facilities. This engine will have a 45.72- to 50.8-cm-high cowl to allow realistic flight structures and fabrication techniques and to represent an engine of appropriate size for flight demonstration on a research airplane.

Inlet Development: Figure 14 shows an aerodynamic test model of the scramjet module inlet. The foreplate and trips ahead of the inlet partially simulate the boundary layer on the underside of the vehicle. One sidewall of the inlet has been removed to show the struts and rake to measure flow profiles at the throat. This model is the third in a series of models tested to establish inlet performance (Refs. 27-29). Tests of this model have been completed with Mach numbers ahead of the inlet from 2.3 to 6 and unit Reynolds numbers in the range from 9.8×10^6 to 3.3×10^6 m^{-1}. These conditions at the inlet entrance correspond to vehicle flight speeds from M_0 = 3 to 8 at altitudes of 18,000 to 25,000 m.

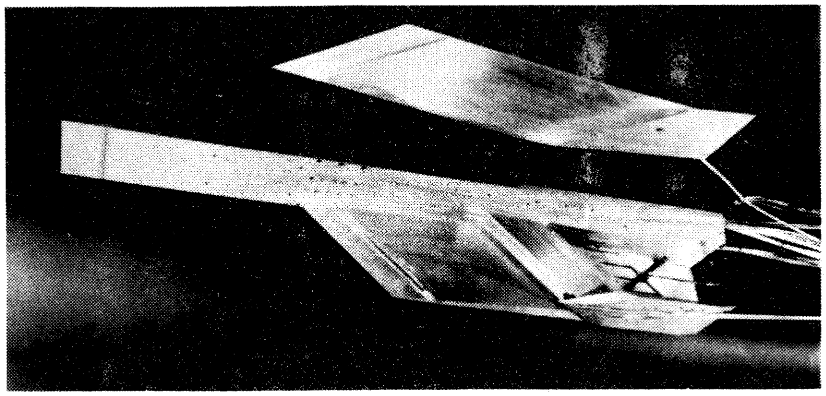


Fig. 14 NASA-Scramjet Inlet Model

Figure 15 presents four parameters deduced from these tests which give an overall measure of the inlet performance as a function of the flow field Mach number. The throat Mach number variation shows that the inlet is self-starting at the lowest Mach number tested. Sweep of the compression surfaces and the cutback cowl produce variations in inlet spillage and aerodynamic contraction that combine to give this fixed-geometry inlet characteristics which are similar to those of variable-geometry inlet. Pressure recovery is adequate for good engine performance, and the inlet is able to ingest the large boundary layer generated by the foreplate without separation or other problems, at least at the conditions tested. Contouring the inlet upper surface allows considerable latitude in spreading out the pressure rise that this boundary layer undergoes, and it is expected that the inlet can ingest the vehicle boundary layer without requiring bleed.

Combustor Development: Of course, fuel injector and combustor designs are high risk elements in the development of any new configuration. In recognition of this fact, considerable effort has been expended to evolve analytical and experimental tools for development of new high performance combustor concepts. The analytical approach adopted is based on estimating the distribution of mixing and heat release with length expected from a given fuel injector design. Both detailed theoretical prediction

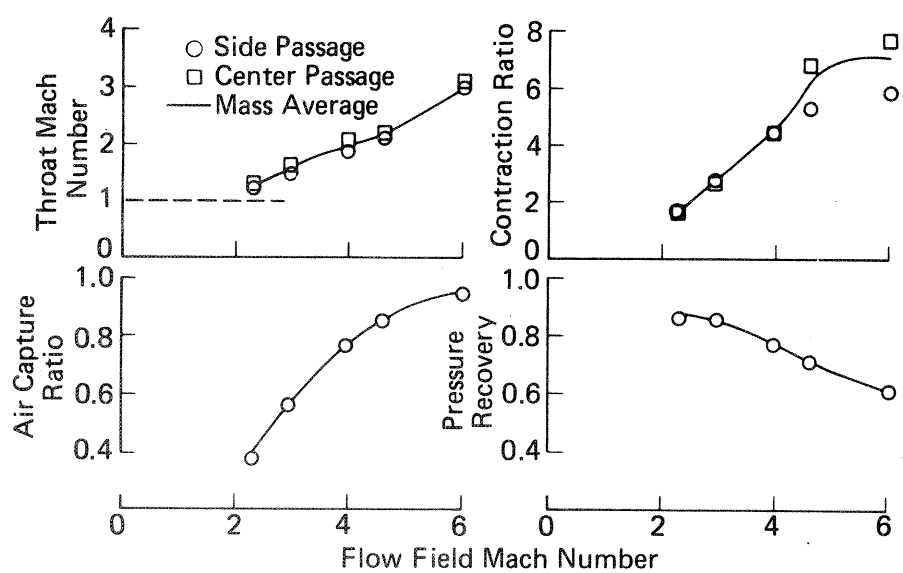


Fig. 15 NASA-Scramjet Module Inlet Performance

and empirical correlations provide useful input to this estimate. Pressure and other flow properties are calculated from combustor geometry and entrance conditions with a one-dimensional analysis which includes wall friction losses and heat transfer (see Ref. 30 for a detailed discussion). The ability to predict the overall combustor wall pressure force, friction force, and heat transfer that result for a given injector design and combustor geometry is an important advantage of this approach, since these parameters are necessary for overall engine performance evaluation. Good agreement with measured combustor performance has been found for a variety of injector/combustor designs including those with injection perpendicular to the flow. An exception are designs which produce a large pressure rise and separation on the combustor walls which dominates the flow pattern; this is not a serious shortcoming, since the high heat transfer and pressure loads which result from this type of flow are not desirable for regeneratively cooled engine structure requiring long cycle life.

Experimental evaluation of fuel injector and combustor designs are carried out at Langley in a direct-connect, combustion heated facility (Refs. 31-33). The facility burns a mixture of hydrogen, oxygen, and air to produce a high-temperature test gas with the same unburned oxygen mole fraction as the atmosphere. A maximum stagnation pressure of about 3.45 MN/m^2 can be produced with stagnation temperatures up to $2610^{\circ}K$, values which simulate Mach 8 flight speed. Figure 16 represents results from one of the experiments conducted in this facility that is typical of the work underway to establish combustor performance. Imagine a cross section of the fuel injector struts in the plane of the engine cowl. This cross section is shown by the shaded shapes in the upper portion of Fig. 16. The combustion heater and a supersonic nozzle are used to simulate the flow between the center and outer struts of the engine. The upper wall of the combustor model is shaped to represent the engine centerline, and the lower wall is shaped to represent the dividing streamline for the flow past the outer strut. The length of the combustor model is about one-third of the length of the engine combustor. Fuel is injected perpendicular to the flow from both walls downstream of small rearward-facing steps. The wall pressure distribution at conditions simulating Mach 7 flight and stoichiometric fuel injection is shown in the lower portion of Fig. 16. The increase in static pressure near the injector shows the effectiveness of the wall step and divergence downstream of the injectors in controlling combustion-generated pressure rise. Heat release maintains the static pressure nearly constant as the flow expands in the diverging duct downstream of the struts.

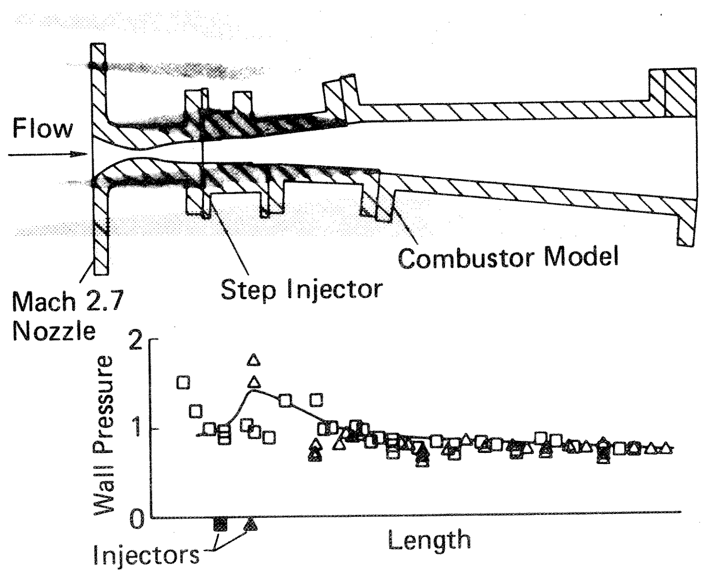


Fig. 16 **Strut Injector Simulation in NASA Scramjet Combustor Model and Wall Pressure Distribution for ER = 0 and 1**

In addition to surveys of wall static pressure distribution, surveys of gas

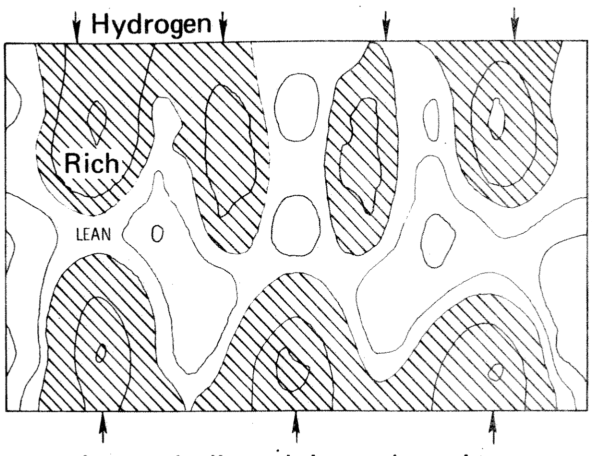


Fig. 17 NASA Combustor Exit Fuel Distribution with Stoichlometric Injection

composition and pitot pressure are made at the combustor exit to provide a direct means of determining where the injected fuel goes and how much reacts. Fuel distribution contours constructed from the gas sample surveys are shown in Fig. 17. A rich region corresponding to each injector can be identified. The overall amount of fuel injected is stoichiometric, and integration of these composition contours with the local mass flux shows that at the exit of the combustor model about two-thirds of the injected fuel has reacted. In other words, two-thirds of the fuel reacts in a length corresponding to one-third of the engine combustor length. By comparison with multiple jet mixing calculations for this geometry, this amount of reaction indicates combustor efficiencies of 90 to 95 percent for the engine. The kind of fuel distribution map shown represents an important capability that will be applied to optimizing fuel injector design in multi-strut injector tests which are now being prepared which provide complete simulation of the combustor flow field.

<u>Subscale Engine Tests</u>: The work on the inlet and combustor components is intended to provide the information necessary for the design and testing of complete subscale scramjet engine modules. Figure 18 shows the facility and some of the hardware being prepared for this work. The upper photograph shows (from left to right) the new arc heater, nozzle, and a test section with calibration probes. The arc heater uses a 20-megawatt direct-current power supply to produce air at 2200°K and up to 40 atmospheres simulating Mach 7 flight at high altitudes. The rectangular exit nozzle produces Mach 6 flow to represent the conditions behind the vehicle bow shock at Mach 7 flight speed, and includes an unheated supersonic shroud flow on three sides of the arc-heated core to allow maximum engine model size. Details of the facility design and operation are presented in reference 34. The engine model shown in the lower photograph has a 20.32-cm-high cowl and is constructed mostly of copper for heat sink cooling. Some water cooling is used in local areas in the combustor region to limit thermal stress due to temperature gradients. The arc heater has been run successfully

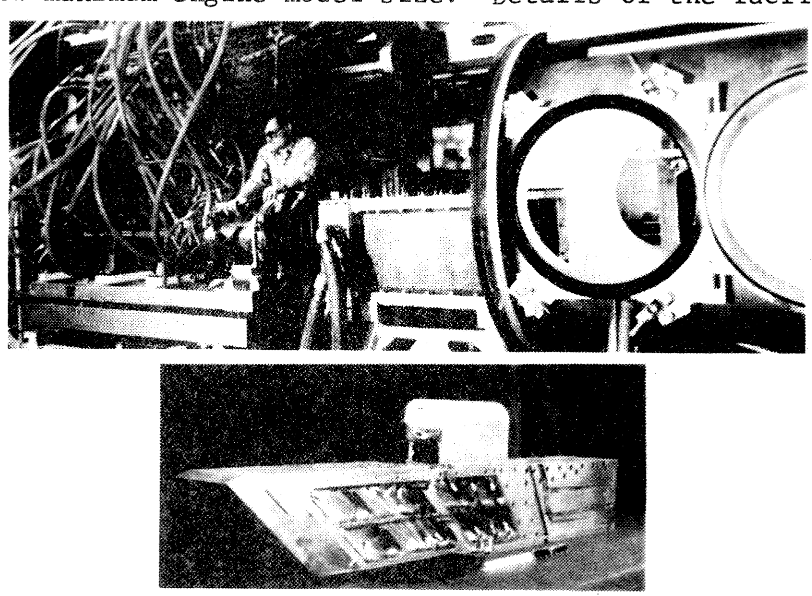


Fig. 18 NASA Mach 7 Subscale Engine Test Facility and Engine Model

at full power, and calibration runs with the test section and probe rake have been completed. The engine model is installed in a separate test cabin which has a model insertion and thrust measuring capability. The first tests of the engine are in progress.

U.S. AIR FORCE SCRAMJET DEVELOPMENT

Air Force interest in hypersonic propulsion began in the late 1950's under exploratory development programs conducted at Marquardt, initially on external burning systems in which thrust and augmented lift were obtained by combustion beneath a wing (Refs. 35 and 36). Interest in a supersonic combustion engine was intensified when a single stage earth-to-orbit vehicle (Aerospace Plane) was conceived. Two airbreathing propulsion schemes were of primary interest: namely, the air collection system requiring a subsonic ramjet to power the vehicle during the air collection and oxygen storage phase of the flight, and the supersonic combustion ramjet engine. Hydrogen fuel was selected for this application because of its intrinsic cooling capability and its high specific impulse. Both propulsion approaches were pursued by vigorous component development programs and ultimately led to the development of a subsonic combustion thrust chamber capable of hypersonic flight, and several scramjet engines. During the past decade the USAF has sponsored a number of scramjet engine programs (Ref. 37). The following engines are representative of the different types of scramjet engines developed and ground tested in these programs:

(a) United Aircraft Research Laboratory Variable Geometry Scramjet
(b) General Electric Component Integration Model (CIM) Scramjet
(c) General Applied Science Laboratory Low Speed Fixed Geometry Scramjet
(d) Marquardt Dual Mode Scramjet

These engines, shown in Figs. 19 through 22 respectively, were hydrogen fueled and achieved performance levels which, in general, substantiated theoretical predictions. Experience was gained regarding potential problem areas such as unfavorable combustor-inlet interactions leading to inlet unstart, and reduced combustion efficiency in divergent combustors. Although most of these engines were aerodynamically designed to operate over a wide range of hypersonic speeds and were substantiated by component tests conducted over a wide Mach number range, ground testing of the entire engine was restricted to a narrow Mach number range because of facility limitations. Hence, the full potential of these engines was never documented. A brief description of the first three engines will be given, followed by a more detailed discussion of the Marquardt Dual Mode Scramjet, which has undergone a relatively extensive testing program and is representative of an attractive concept for high speed aircraft (Ref. 38).

UARL Variable Geometry Scramjet

A 45.72-cm-dia. water-cooled variable geometry scramjet engine was developed and tested at $M_0 = 5$ by United Aircraft Research Laboratory in the 1965-1968 time period. It was designed to operate over a wide Mach number speed range (up to Mach 12) with all supersonic combustion (Ref. 39). The engine (see Fig. 19) is axisymmetric incorporating a translating cowl which slides on three support fins. The translation of the cowl provides a variable inlet capture area and contraction ratio in order to obtain higher compression at the high Mach numbers, and more air flow at the low Mach numbers. At the same time, the cowl translation increases the combustor area ratio at the low flight Mach numbers to alleviate the problem of thermal choking, and also changes the nozzle area ratio in such a manner as to reduce the over and under expansion losses. The engine has four fuel injection stations, three on the center body and one on the cowl, and a gas generator ignition system. Over twenty free-jet tests were performed at the Ordnance Aerophysics Laboratory (OAL) in which inlet performance, pilot ignition, and engine performance under various injector configurations were investigated.

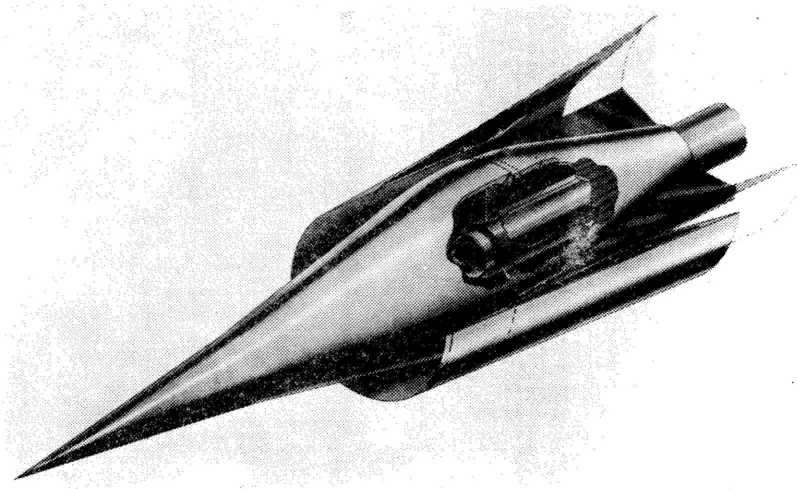


Fig. 19 United Aircraft Research Lab Variable Geometry Scramjet

GE Component Integration Model Scramjet

Two 22.86-in.-dia. water-cooled variable geometry scramjet engines were designed and tested at $M_0 = 7$ by the General Electric Company in the 1966-1969 time period. The first engine, CIM-I, provided an evaluation of a combined set of scramjet components designed for operation up to Mach 8 (Ref. 40). CIM-I, constructed of chrome copper, had an axisymmetric mixed compression inlet with a movable centerbody, an annular combustor and a fixed annular plug nozzle. Two independent stages of normal injection were employed downstream of a small rearward-facing step to prevent propagation of combustion pressure rise from inducing separation in the inlet throat region. The combustor consisted of a constant area section followed by an 8^o divergent section. Upon completion of testing in the General Electric Hypersonic Arc Tunnel, CIM-I was subsequently modified by replacing the cowl section with one having a smaller cowl lip angle to reduce external drag, and contouring some of the internal lines to increase performance. Extensive performance tests were conducted on CIM-II, see Fig. 20, to obtain the effects of varying inlet contraction ratio (13 to 25), equivalence ratio, fuel injector location, free stream Reynolds number and total enthalpy.

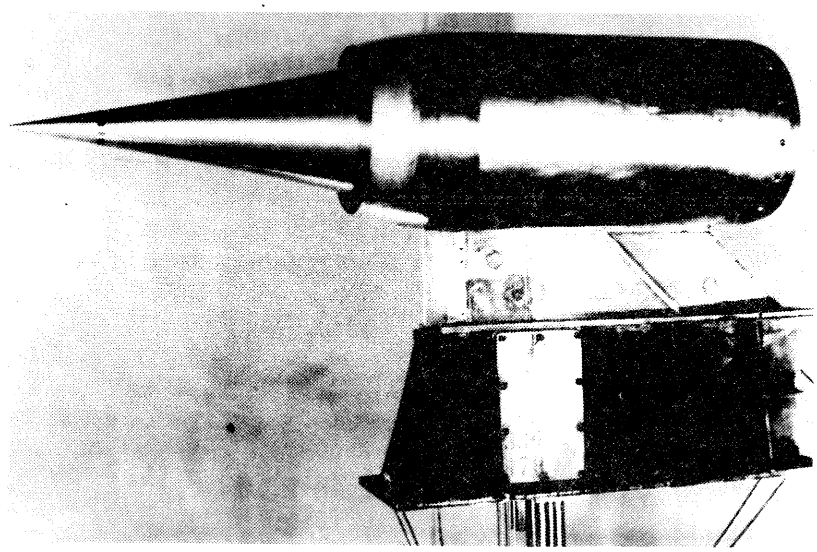


Fig. 20 General Electric Component Integration Model Scramjet

GASL Low Speed Fixed Geometry Scramjet

A Mach 3-12 engine concept, involving a series of heat sink engine models of approximately 194 to 226 cm^2 of capture area, was developed and tested by the General Applied Science Laboratories under the late Dr. Ferri in the 1964-1968 time period (Ref. 41). This concept employs a fixed geometry closely integrated inlet-combustor design with low overall geometric contraction (< 4), utilizing three-dimensional and combustor induced compression effects (sometimes referred to as thermal compression) to obtain an aerodynamic contraction ratio which varies with flight Mach number. At low Mach numbers, where flow disturbances propagate at large angles laterally, the swept back three-dimensional design permits large mass flow capture while preventing choking because of the large geometric flow area available. At high Mach numbers, where shock waves are highly swept, the stream tubes entering the inlet do not experience much lateral relief and thus are highly compressed in the local regions of large contraction. The resulting nonuniform combustor entrance flow is then diffused to relatively uniform conditions by utilizing combustion induced compression obtained from the proper placement of fuel injectors. Engine models demonstrating this concept have been tested at Mach = 2.7, 4 and 7 with inlet component tests covering Mach numbers from 2.7 to 11.3. Modifications to these designs were incorporated into a later engine model shown in Fig. 21 and tested at M = 7.4 in the GASL combustion heated high enthalpy blowdown tunnel under a wide variety of fuel injector patterns and fuel flow schedules.

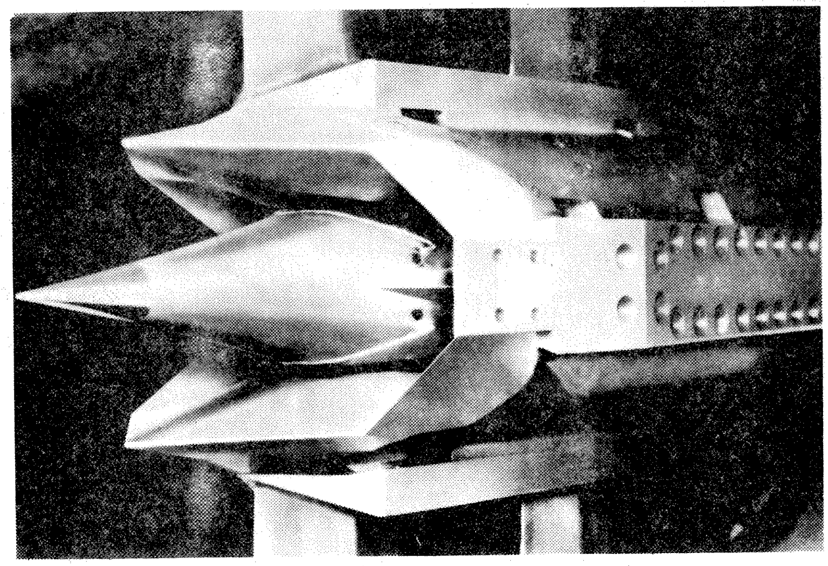


Fig. 21 General Applied Science Lab Low Speed, Fixed Geometry Scramjet

Dual Mode Scramjet

An attractive approach for the supersonic/hypersonic speed regime is the dual mode engine (Ref. 42) which combines the advantages of subsonic combustion at the lower flight speeds with supersonic combustion in the hypersonic regime. The main feature of this concept is that in principle the combustor operates in two modes: one for supersonic combustion and the other for subsonic operation. This can be accomplished by providing fuel injection at different axial locations within a common duct. The supersonic combustion section proceeds the subsonic one and acts as the subsonic diffuser of the inlet

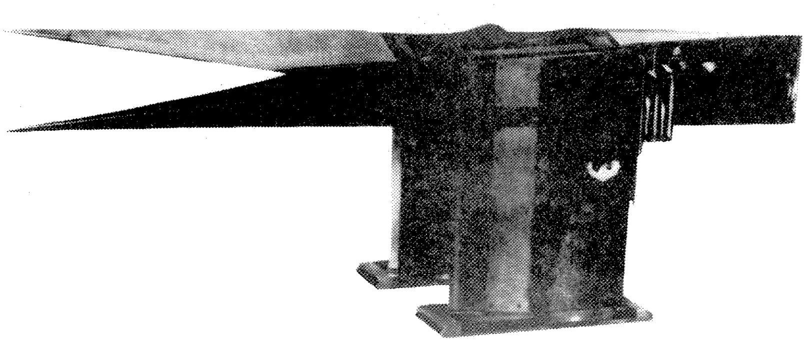
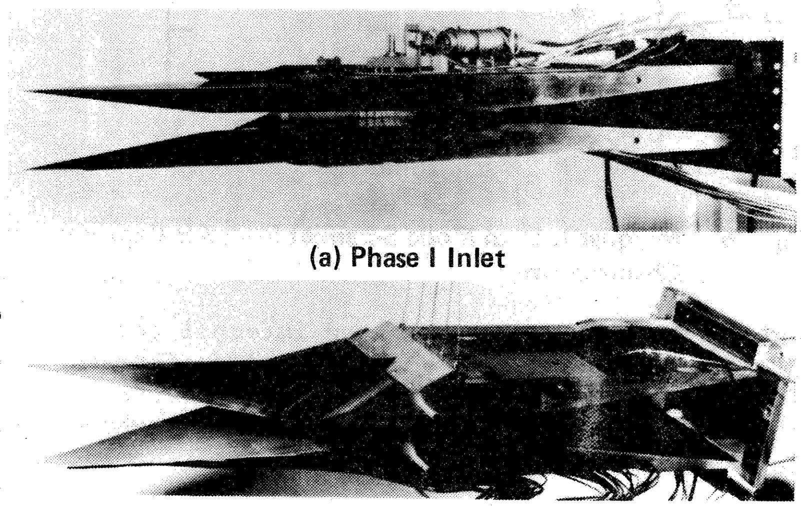


Fig. 22 Marquardt Dual Mode Scramjet

during the subsonic mode. An extensive component and engine development program was conducted by Marquardt in the 1964-1968 time period to develop this approach.

Inlet Development: Phase I analytical and experimental evaluations of a fixed geometry inlet which could satisfy both the low speed and high speed requirements of the Dual Mode Scramjet were conducted in mid 1965. This inlet, featuring highly swept leading edges (see Fig. 23) was tested at AEDC at Mach numbers from 2 to 6. A larger scale inlet was also tested in a free-jet cell at OAL at Mach 3 and 5 in combination with a combustor. Tests results indicated that this type of inlet had the overall desired characteristics, but that nonuniform compression was occurring in the throat region which resulted in a low critical pressure recovery for the subsonic mode of operation.



(a) Phase I Inlet

(b) Phase II Inlet

Fig. 23 Marquardt Dual Mode Scramjet Inlet Designs

The Phase I inlet was redesigned in Phase II to improve the throat mass flow distribution and capture area characteristics. This effort was conducted in two test programs involving four-tenth scale models of the proposed engine. The first was a pilot test series conducted over Mach numbers from 3 to 6 at AEDC. During these tests, geometric contraction ratio, leading edge contour, side wall configuration and boundary layer control were investigated over a range of values. As a result of these tests, a constant flow area section was added to the inlet throat. Documentation testing of the Phase II inlet (Fig. 23) followed covering the Mach number range of 2 to 10. A number of configurations were investigated in arriving at the final configuration in which all boundary layer bleed was eliminated. During these tests a traversing static/pitot rake was used to measure the performance of the inlet in the subsonic and supersonic modes of operation. Figures 24 and 25 show the pressure recovery and capture area characteristics, respectively, of this inlet. The pressure recovery in the subsonic mode corresponds to the critical pressure recovery.

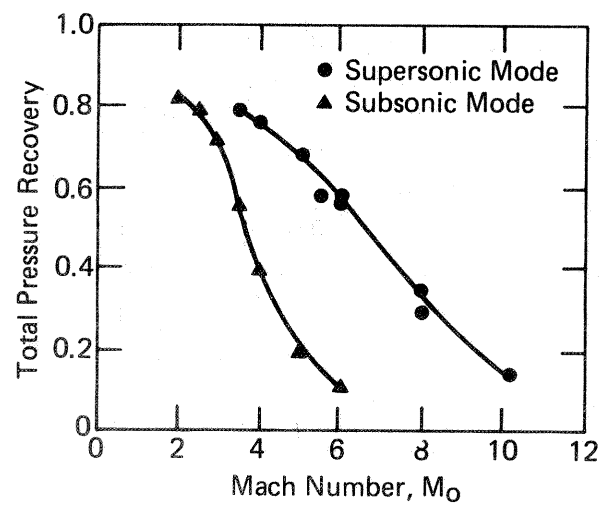


Fig. 24 Marquardt Dual Mode Scramjet Inlet Pressure Recovery

Combustor/Nozzle Development: Phase I inlet/combustor tests conducted in the free-jet cell at OAL indicated the need for additional experimental tests to provide a reliable ignition source and possible piloting system for low Mach number operation. As a result, a series of full scale, direct-connect combustor tests was conducted at Mach 3 and 5 simulated freestream conditions at the Marquardt Research Field Laboratory. Ignitors evaluated included H_2-air, pentaborane, and fluorine. Fluorine was shown to offer a positive and reliable ignition source under all test conditions. It was determined that piloting devices were not required in the low flight speed regime for the hydrogen fueled Dual Mode combustor. In addition to establishing ignition and piloting requirements, these tests investigated internal combustor contours and fuel injection patterns for maximum combustor performance in the subsonic and supersonic combustion modes. The ability to position the normal shock system by fuel modulation, while maintaining stable combustor performance during transition, was also demonstrated.

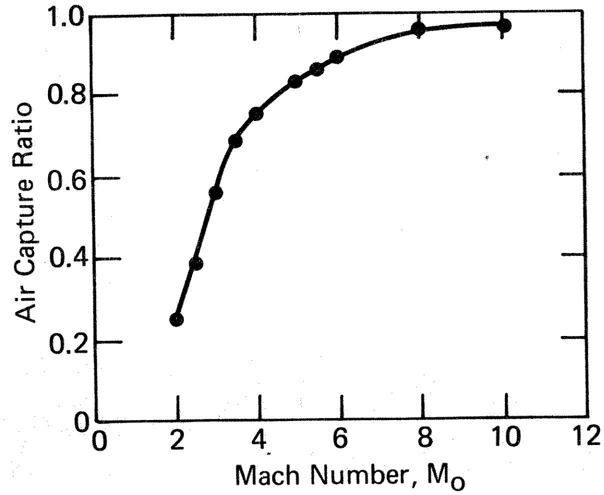


Fig. 25 Marquardt Dual Mode Scramjet Inlet Air Capture Characteristics

Free-Jet Engine Tests: Based upon the results of the preceeding Phase II inlet and combustor/nozzle tests, a water-cooled Dual Mode Scramjet Engine was fabricated and tested in 1967 (Ref. 43). The engine is a fixed geometry, two-dimensional configuration incorporating highly swept back features (see Fig. 22) and designed to operate over a wide Mach number speed range by using subsonic and supersonic modes of combustion. Its basic nominal dimensions consist of height = 24.89 cm, width - 38.61 cm and overall length - 222.25 cm. With an inlet contraction ratio of 5.62, the capture area is 619.38 cm^2. Nozzle exit to cowl area ratio is 1.44. The fuel injection system, consisting of nine axial fuel injector locations along with two fluorine ignitors, allows for stable combustion mode transitions to be made with high overall engine efficiency. The engine model was fabricated from Inconnel 718 and Hastelloy X and structurally designed for Mach 3-8 test conditions.

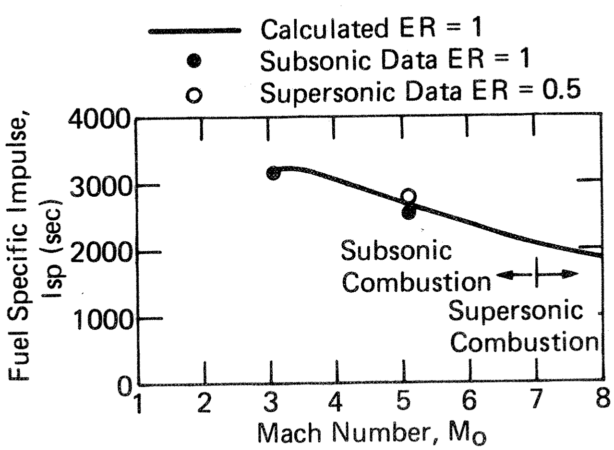


Fig. 26 Marquardt Dual Mode Scramjet Fuel Specific Impluse

The Dual Mode Scramjet Engine was free-jet tested at OAL at Mach 3 and Mach 5 in the subsonic combustion mode and at Mach 5 in the supersonic combustion mode. A total of 38 runs were made with the engine accumulating more than one hour of combustion operation. Theoretical performance was achieved in these tests based upon the individual component efficiencies achieved earlier in the detailed performance tests. Figures 26 and 27 show the specific impulse and thrust coefficient, respectively, obtained from this engine. For optimum performance the dual mode engine would not normally convert from subsonic to supersonic combustion until about Mach 7. However, due to facility limitations the engine was tested in the supersonic mode at Mach 5. Operation at ER = 1 in the supersonic mode was not possible at such a low Mach number due to thermal choking. This results in a low thrust coefficient for the supersonic combustion mode as can be seen in Fig. 27. Nine single and five dual combustion mode transitions were demonstrated during the testing of this engine. All of these transitions were made with smooth and stable engine operation. No combustion instability was observed during these runs or detected in an examination of the recorded data. These tests confirmed that components and their individual performance can be successfully integrated into an engine capable of supersonic/hypersonic flight with a high degree of confidence in achieving predicted performance levels.

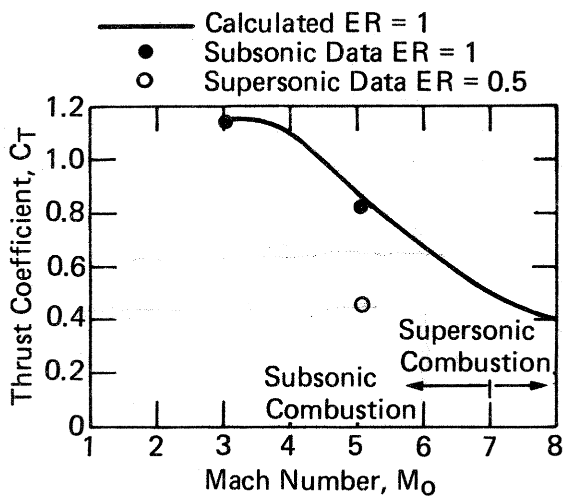


Fig. 27 Marquardt Dual Mode Scramjet Thrust Coefficient

Hydrocarbon Fueled Hypersonic Engines

At the conclusion of the dual mode scramjet program and the other successful scramjet engine programs, USAF interest shifted from large scramjet vehicles to smaller missile systems leaving the hydrogen scramjet area to NASA with their HRE and other programs. As a result, attention focused on the hydrocarbons and fuels with a high density impulse. Ignition delay and reaction times for gaseous hydrocarbons are much longer than for hydrogen, hence the problem of achieving high combustion efficiencies using these fuels proved more difficult than for hydrogen. Initially, attempts were made to simply modify the existing hydrogen scramjet engines by lengthening the combustor section and using gaseous fuels such as methane and ethylene, but these met with only limited success. Extensive effort has been devoted to the development of piloting systems for use in scramjet engines employing liquid hydrocarbon fuels, and is the approach employed in the Dual Mode Hydrocarbon Scramjet. The concept of a pilot is to provide a high temperature gas source along with a large concentration of free radicals. Good supersonic combustion efficiencies have been obtained using liquid hydrocarbon fuels in tests where suitable fuel injector piloting systems have been developed.

PROPOSED USAF/NASA X-24C RESEARCH VEHICLE

In light of technical interest and engineering activity relative to a variety of propulsion concepts as well as other interests existing in technical domains of structures, subsystems and miscellaneous components including avionics, efforts have been made by a joint USAF/NASA ad hoc group to describe the performance and design requirements for a low cost research vehicle. In essence, the aim of this group has been to provide a "flying wind tunnel" that would be free of some of the encumberances encountered in ground facilities, provide a capability for demonstrating large scale propulsion and structures in the actual environment, provide data by which to correlate ground facility results, and provide an insight into synergistic effects on various systems.

A major goal of the study was to determine minimum achievable vehicle acquisition costs through the application of three basic principles: The basic vehicle should be a small, flexible, low risk carrier for a variety of major technology demonstrations and experiments rather than a system dependent on advanced concepts and requiring costly new developments; existing subsystems available in government inventories as well as the experience gained from the X-15, X-24A, and X-25B programs should be utilized wherever possible; and maximum use should be made of government engineering and testing facilities. The X-24A and X-24B, two different aerodynamic classes of vehicle, were both designed to perform to a maximum Mach number of about 1.6. The next speed increment originally planned for the growth version of the X-24B class (X-24C) was Mach 5.0; however, the study group expanded considerations to Mach 7.0.

Within this framework, the performance requirements of the airplane were set by three categories of important technology demonstration experiments: configuration technology, research ramjet and scramjet propulsion system tests with representative engine/airframe integration, and demonstration of advanced structural concepts.

The planned X-24C shown in Fig. 28 is a versatile research tool, in that it can accommodate a number of major flight system demonstrations including scramjet propulsion. The strakes and fins are removable and can be replaced with a variety of actively cooled or hot structural test items. A removable/replaceable 10-foot payload bay section is incorporated into the forward section of the fuselage aft of the pilot. This entire section is available for cryogenic hydrogen tankage (an integral tank) or other experiment payloads. The vehicle can reach speeds to Mach 6 with the scramjet experiment installed and has enough rocket fuel and thermal protection for 40 seconds of cruise at Mach 6 using the throttled rocket in combination with the scramjet experiment. If pursued, this new research airplane could begin flights in the early 1980's. For more details of this system, the reader is referred to Ref. 38.

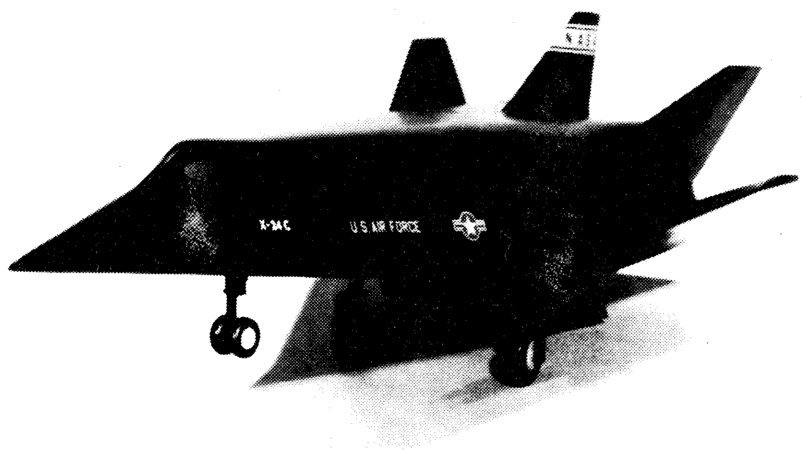


Fig. 28 Proposed USAF/NASA X-24C Research Vehicle

CONCLUDING REMARKS

Over the past 15 years, considerable advancement in the understanding and development of supersonic combustion ramjet propulsion systems has been achieved under the three programs just described. These include but are not limited to development of the experimental apparatus and techniques needed to provide basic component design data, development of the analytical techniques for designing and analyzing the various components and overall engine cycle, proof-of-concept and feasibility by free-jet engine testing, and development of regeneratively and passively-cooled thermal protection systems. Although much has been accomplished, continued experimentation and analysis are needed to refine the various component and overall designs as well as integrate the propulsion system with the specific type of vehicle or mission requirement before actual flight tests begin.

REFERENCES

1. G. L. Dugger, "Comparison of Hypersonic Ramjet Engines with Subsonic and Supersonic Combustion," Fourth AGARD Colloquium, Milan, Italy; also Combustion and Propulsion-High Mach Number Air Breathing Engines, Edited by Jaumotte, Rothrock, and Le Febvre, Pergamon Press, New York, 1960.

2. A.Ferri, "Supersonic Combustion Progress," Aeronautics and Astronautics, Vol. 2, No. 8, 1964, pp. 32-37.

3. A.Ferri, "Review of Scramjet Technology," Journal of Aircraft, Vol. 5, No. 1, Jan. 1968, pp. 3-10.

4. A.Brema (The Marquardt Co.), "Recent Advances in Mixed Cycle Engine Design and Application," 1st International Symposium on Air Breathing Engines, June 1923, 1972, Marseille, France.

5. F. S. Billig, "Design of Supersonic Combustors Based on Pressure-Area Fields," Eleventh Symposium (International) on Combustion, The Combustion Institute, 1967, pp. 755-769.

6. F. S. Billig and G. L. Dugger, "The Interaction of Shock Waves and Heat Addition in the Design of Supersonic Combustors," Twelfth Symposium (International) on Combustion, The Combustion Institute, 1969, pp. 1125-1139.

7. G. L. Dugger, B. Deklau, F. S. Billig, and S. Mathews, "Summary Report on External Ramjet Program," TG-419, The Johns Hopkins University Applied Physics Laboratory, Laurel, Md., Oct. 1961.

8. R. C. Orth, F. S. Billig and S. E. Grenleski, "Measurement Techniques for Supersonic Combustion Testing," AIAA Progress in Astronautics and Aeronautics, Vol. 34, Jan. 1974.

9. F. S. Billig, R. C. Orth, and J. A. Funk, "Direct-Connect Tests of a Hydrogen-Fueled Supersonic Combustor," NASA-CR-1904, August 1971.

10. F. S. Billig, G. L. Dugger and P. J. Waltrup, "Inlet-Combustor Interface Problems in Scramjet Engines," 1st International Symposium on Air Breathing Engines, June 19-23, 1972, Marseille, France, APL/JHU, preprint June 1972.

11. P. J. Waltrup and F. S. Billig, "Prediction of Precombustion Wall Pressure Distributions in Scramjet Engines," Journal of Spacecraft and Rockets, Vol. 10, No. 9, September 1973, pp. 620-622.

12. P. J. Waltrup and F. S. Billig, "The Structure of Shock Waves in Cylindrical Ducts," AIAA Journal, Vol. II, No. 10, Oct. 1973, pp. 1404-1408.

13. P. J. Waltrup and J. M. Cameron, "Wall Shear and Boundary Layer Measurements in Shock Separated Flow," AIAA Journal, Vol. 12, No. 6, June 1974, pp. 878-880.

14. F. S. Billig and S. E. Grenleski, "Heat Transfer in Supersonic Combustion Processes," paper presented at Fourth International Heat Transfer Conference (Versailles, France), August 1971.

15. A. M. Agnone, "Development of a Two-Dimensional Tangential Fuel Injector with Constant Pressure at the Flame," NYU 72-25, New York University, September 1972. (Available as NASA CR-112302).

16. S. Dash, "An Analysis of Internal Supersonic Flows with Diffusion Dissipation and Hydrogen-Air Combustion," ATL TR 152, Advanced Technology Laboratory, Inc., May 1970. (Available as NASA CR-111783).

17. S. V. Parankar and D. B. Spalding, "A Calculation Procedure for Heat, Mass and Momentum Transfer in Three-Dimensional Parabolic Flows," International Journal of Heat and Mass Transfer 15, 1787-1806, October 1972.

18. V. Agosta and S. Hammer, "Scramjet Nozzle Analysis, PSI Report 70-1, Propulsion Sciences, Inc., Melville, N. Y., February 1970.

19. D. Migdal, "Supersonic Annular Nozzles," AIAA 9th Aerospace Sciences Meeting, January 1971 Preprint 71-43.

20. R. C. Orth and J. M. Cameron, "Flow Immediately Behind a Step in a Simulated Supersonic Combustor," AIAA Journal, Vol. 13, No. 9, Sept. 1975, pp. 1143-1148.

21. HREP - Phase II - "Structures Assembly Model Test Report," NASA CR-111993, Sept. 1971.

22. A. R. Wieting, "Aerodynamic and Thermal Analysis of Results of Tests of a Hydrogen Cooled Scramjet Engine at Mach 6.3," NASA TM X-2767, May 1973.

23. "Hypersonic Research Engine Project Technological Status, 1971," NASA TM X-2572, 1972.

24. "Hypersonic Research Engine Project - Phase II, Aerothermodynamic Integration Model Development - Final Technical Data Report." (AP 75-11133, AiResearch Manufacturing Company of California; NAS1-6666) NASA CR-132654, 1975.

25. W. L. Andersen and L. Kado, "Hypersonic Research Engine Project - Phase II, Aerothermodynamic Integration Model (AIM) Test Report." (AP-74-10784, AiResearch Manufacturing Company of California; NAS1-6666) NASA CR-132655, 1975.

26. J. R. Henry and G. Y. Anderson (NASA Langley Research Center), "Design Considerations for the Airframe-Integrated Scramjet," 1st International Symposium on Air Breathing Engines, June 19-23, 1972, Marseille, France.

27. C. A. Trexler, "Performance of an Inlet for an Integrated Scramjet Concept," J. Aircraft, Vol. 11, No. 9, Sept. 1974, pp. 589-591.

28. C. A. Trexler, "Design and Performance at a Local Mach Number of 6 of an Inlet for an Integrated Scramjet Concept," NASA TN D-7944, 1975.

29. C. A. Trexler, "Inlet Performance of the Integrated Langley Scarmjet Module," AIAA Paper No. 75-1212, Sept. 1975.

30. G. Y. Anderson, "An Examination of Injector/Combustor Design Effects on Scramjet Performance," 2nd International Symposium on Air Breathing Engines, March 1974, Sheffield, England.

31. General Applied Sciences Labs., Inc., "Engineering Analyses and Design Calculations of NASA - Langley Research Center Hydrogen-Air-Vitiated Heater with Oxygen Replenishment," GASL-8669, Nov. 1968.

32. W. R. Russin, "Performance of a Hydrogen Burner to Simulate Air Entering Scramjet Combustors," NASA TN D-7567, Feb. 1974.

33. J. M. Eggers, "Composition Surveys of Test Gas Produced by a Hydrogen-Oxygen-Air Burner," NASA TM X-71964, June 1974.

34. W. B. Boatright, A. P. Sabol, D. I. Sebacher, S. Z. Pinckney, Z. Shimer, and R. W. Guy, "Langley Facility for Tests at Mach 7 of Subscale, Hydrogen Burning, Airframe-Integratable, Scramjet Models," AIAA Paper No. 76-11, Jan. 1976.

35. "The Performance of Supersonic Burning Engines at Various Equivalence Ratios and Its Use as a Powerplant for Auxiliary Jet Lift," Marquardt Report MR 20,006, November 1958.

36. A. Mager, "Supersonic Airfoil Performance with Small Heat Addition," J. of the Aero/Space Sciences, Vol. 26, Feb. 1959, pp. 99-107.

37. E. T. Curran and F. D. Stull, "Ramjet Engines: Highlights of Past Achievements and Future Promise," 2nd International Symposium on Air Breathing Engines, March 1974, Sheffield, England.

38. F. D. Stull, R. A. Jones and W. P. Zima, "Propulsion Concepts for High Speed Aircraft," SAE Preprint 751092, Nov. 1975.

39. C. E. Kepler and A. J. Karanian, "Hydrogen-Fueled Variable-Geometry Scramjet," United Aircraft Research Laboratories, AFAPL, TR-67-150, Jan. 1968.

40. D. L Harshman, "Design and Test of a Mach 7-8 Supersonic Combustion Ramjet Engine," AIAA Propulsion Specialist Meeting, July 1967, Washington, D. C.

41. "Investigations of the Low Speed Fixed Geometry Supersonic Combustion Ramjet," General Applied Science Laboratories, AFAPL-TR-66-139, March 1967.

42. E. T. Curran and F. D. Stull, "The Utilization of Supersonic Combustion Ramjet Systems at Low Mach Numbers," RTD-TDR-63-4097, AIAA Summer Meeting, June 1963, Los Angeles, California.

43. T. D. Burnette, "Dual Mode Scramjet," The Marquardt Corporation, AFAPL-TR-67-132, June 1968.