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DESIGN AND TESTING OF A CASELESS SOLID-FUEL INTEGRAL-ROCKET RAMJET ENGINE FOR USE IN SMALL TACTICAL MISSILES

by

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An investigation was conducted to determine the feasibility of a low-cost, caseless, solid-fuel integral-rocket ramjet (IRSFRJ) that has no ejecta. Analytical design of a ramjet powered air-to-ground missile capable of being fired from a remotely piloted vehicle or helicopter was accomplished using current JANNAF and Air Force computer codes. The results showed that an IRSFRJ powered missile can exceed the velocity and range of current systems by more than a two to one ratio, without an increase in missile length and weight. A caseless IRSFRJ with a non-ejecting port cover was designed and tested. The experimental results of the static tests showed that a low-cost, caseless IRSFRJ with a non-ejectable port cover is a viable design. Rocket-ramjet transition was demonstrated and ramjet ignition was found to be insensitive to the booster tail-off to air-injection timing sequence.

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Design and Testing of a Caseless
Solid-Fuel Integral-Rocket Ramjet Engine
for use in Small Tactical Missiles

by

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Submitted in partial fulfillment
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ABSTRACT

An investigation was conducted to determine the feasibility of a low-cost, caseless, solid-fuel integral-rocket ramjet (IRSFRJ) that has no ejecta. Analytical design of a ramjet powered air-to-ground missile capable of being fired from a remotely piloted vehicle or helicopter was accomplished using current JANNAF and Air Force computer codes. The results showed that an IRSFRJ powered missile can exceed the velocity and range of current systems by more than a two to one ratio, without an increase in missile length and weight. A caseless IRSFRJ with a non-ejecting port cover was designed and tested. The experimental results of the static tests showed that a low-cost, caseless IRSFRJ with a non-ejecting port cover is a viable design. Rocket-ramjet transition was demonstrated and ramjet ignition was found to be insensitive to the booster tail-off to air-injection timing sequence.

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I cannot fully express the thanks and appreciation deserved by Professor Dave Netzer. His quest for knowledge and his honest concern for his students were the inspiration which started me in this project and guided me to its completion. I would also like to thank the propulsion team of Mr. Pat Hickey, Mr. Don Harvey and Mr. Harry Conner for their hard work and technical assistance in the construction of the solid-fuel integral-rocket ramjet apparatus. Most of all, I wish to thank my wife Karen for her care and understanding during this time period.

I. INTRODUCTION

The purpose of this study was to investigate the feasibility of using a solid-fuel integral-rocket ramjet engine (IRSFRJ) to power an air-to-ground missile capable of being launched from a remotely piloted vehicle (RPV) or a helicopter. The general design goal for the missile was to double the range and velocity of current missile systems while not exceeding the current size and weight of these systems.

There are several reasons for wanting to extend the range and velocities of current air-to-ground missiles capable of being fired from RPV's and helicopter platforms. Target detection capabilities of RPV's and attack helicopters are rapidly increasing and in the near future, systems will be fielded which can acquire and track targets 10 to 20 miles in distance. Additionally, the capabilities of missile seekers and for autonomous missile operation are advanced enough for a missile to be launched in the general direction of enemy targets and automatically acquire, select, and engage these targets. Consequently, the ability of potential foes to engage our systems at extended ranges will also be increasing. Thus, a missile with extended range for standoff purposes and high velocity to overcome enemy attempts of electronic or physical evasion and enhance penetration abilities is a valid requirement for the near term.

Historically, air-to-ground missiles have been powered by solid propellant rocket motors. These motors are simple, reliable and inexpensive and have been quite capable of performing up to the level required for the missile's mission. However, the relatively low I_{sp} of these motors prevents the range or velocities to be increased significantly without incurring too great a penalty in weight gain to be used in missiles fired from RPV or helicopter platforms. Additionally, a large part of the missile's flight path occurs after engine burn out, thus the missile generally is coasting during the final engagement phase, which enhances the enemy's ability to evade being hit. Table 1 below describes representative air-to-ground missiles in use today. Additionally, the AIM-9 Sidewinder is listed for comparison purposes since the IRSFRJ is quite capable of powering an air to air missile of this performance level and size.

TABLE 1

CURRENT AIR-TO-GROUND MISSILE CHARACTERISTICS

MISSILE NAME	LENGTH (IN)	DIAMETER (IN)	WEIGHT (LBS)	VELOCITY (MACH)	RANGE (NMI)
TOW II BGM-71	55	6	47.4	<1	2.3
HELLFIRE AGM-114	64	7	100	1.1	<10
SIDEWINDER AIM-9L	120	5	180	2.5	9-11

Although no current U.S. RPV's carry a weapons payload, the new generation of RPV's will have the payload capacity to carry at least a limited amount of ordinance.

The IRSFRJ has several attractive features which makes it a very desirable device for powering an air-to-ground missile of the type under consideration. Typically, one can expect an increase in range of 200 - 400 % over a comparable size and weight solid propellant rocket motor [Ref. 1:p. 1]. This characteristic of the IRSFRJ is due to it being an air breathing device, thus the IRSFRJ's specific impulse (I_{sp}) is significantly higher than for a solid propellant rocket as is shown in Figure 1. Simplicity of construction and little need for esoteric materials allows the IRSFRJ to compete cost-wise very favorably with solid rockets. They are easily tailored for rugged handling and are readily storable with long shelf lives. Additionally, the self-throttling with air flow variations capability allows the IRSFRJ to have a relatively wide operating envelope. Figure 2 depicts a typical solid propellant rocket powered tactical missile and Figure 3 depicts a typical IRSFRJ powered missile.

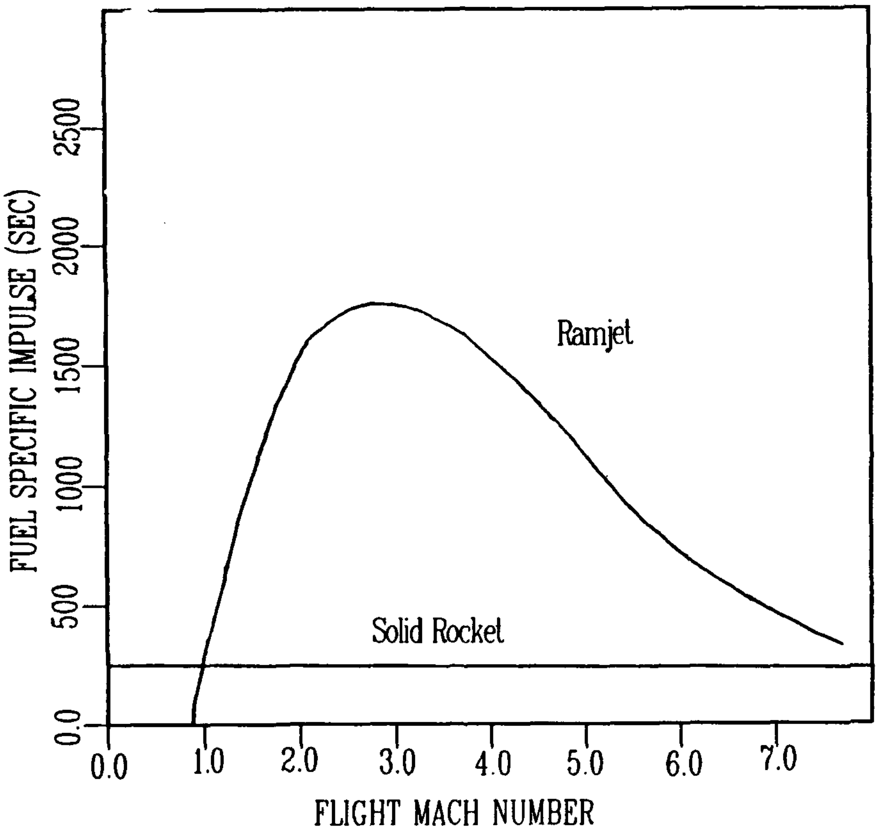

Figure 1. Theoretical performance envelopes.
[Adapted from Ref. 2:p. 144]

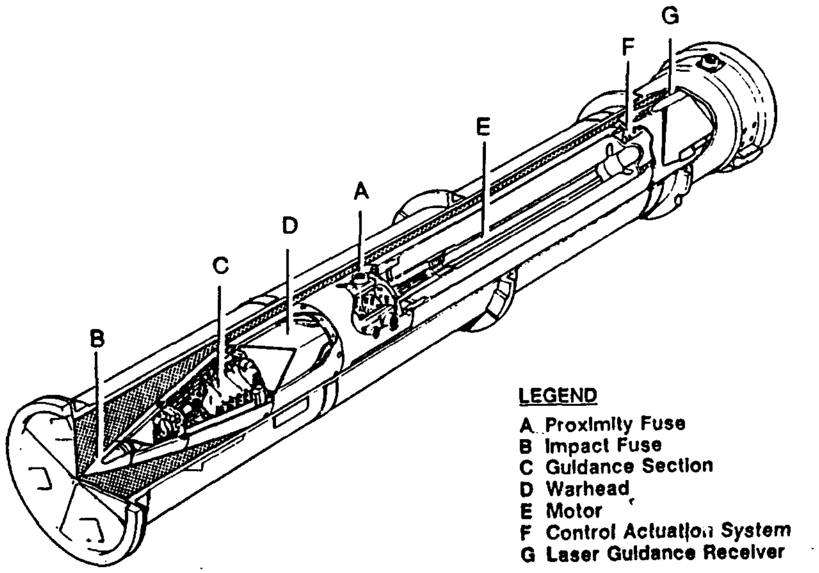

Figure 2. Solid propellant rocket powered missile. [Adapted from Ref. 3:p. 5]

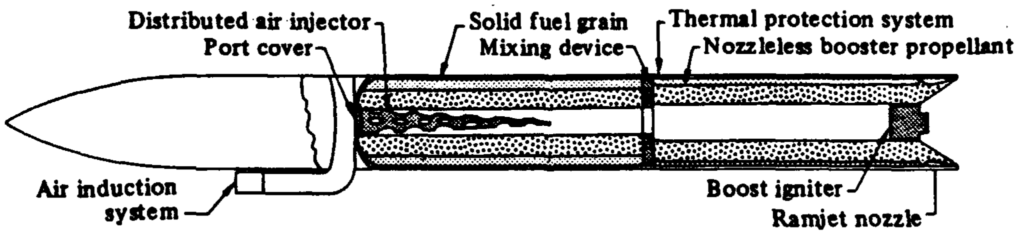

Figure 3. Solid-fuel integral-rocket ramjet powered missile. [Adapted from Ref. 4:p. 39]

II. DESIGN METHODOLOGY

The goal of this study was to determine the feasibility of using a small, low cost IRSFRJ suitable for powering air-to-ground missiles which are capable of being launched from RPV's or helicopters. Several additional requirements were also placed upon the design with the emphasis placed on the RPV mission. There were to be no ejectables and the outside diameter was not to exceed 5 inches. The range requirement was 10 - 20 nautical miles with the launch velocity at Mach 0.3 and the cruise velocity between Mach 2.0 - 2.5. The missile would nominally be launched from and cruise at an altitude of 20,000 ft. The length and weight were to be kept to a minimum while still meeting the performance requirements.

A two level approach was used to solve this design problem. Initially, an analytical approach based on SFRJ cycle analysis was used to obtain a missile configuration which met the design criteria. Concurrent with and following the analytical design, a IRSFRJ engine was designed and manufactured for evaluation on a test stand. Additionally, static firings were conducted to determine the feasibility of using an IRSFRJ design with no ejectables, where the solid ramjet fuel also functioned as the engine casing, in an attempt to minimize engine weight and cost and maximize manufacturing simplicity.

III. METHOD OF ANALYTICAL INVESTIGATION

Current JANNAF and Air Force computer codes were used to produce a conceptual design for the IRSFRJ powered missile. A missile with a 5 inch outside diameter was selected as a good compromise between warhead effectiveness against armored targets and missile size and drag characteristics. Typically, a shaped charge is used as the warhead on air-to-ground missiles and its penetrating ability is related to the diameter of the shaped charge. To insure the missile's ability to penetrate modern armored vehicles with its above the target attack trajectory, it was determined that a minimum warhead diameter of 5 inches was adequate. A forward-twin cheek-mounted, two dimensional inlet was selected because it afforded low drag and weight with good performance and compatibility with the non-ejectable port cover IRSFRJ engine being developed for this study. To maintain simplicity of design and to minimize weight and cost, a non-bypass engine was chosen. Although this prevented the design from achieving the highest possible combustion efficiencies attainable, it was deemed that an acceptable level of performance would result from the non-bypass configuration.

The booster chosen for the IRSFRJ engine was of a nozzleless design, utilizing a reduced smoke composite propellant. The reduced smoke characteristic of the booster enhances the non-detection of both the launch vehicle and the missile itself. The nozzleless booster design was chosen to meet the requirement for no missile ejectables. Although the nozzleless design degrades the Isp of the booster by 20-25%, several benefits are immediately gained by its use. First, there are no ejecta resulting from the discarding of a booster nozzle, thus greatly reducing hazard risk to the launch vehicle. Secondly, the relative simplicity of the grain design and the elimination of the costly nozzle allows for a 10 - 20 % production cost savings [Ref. 5:p. 193]. Also, careful propellant packaging in the volume previously filled by the nozzle can closely match the velocity increment provided by a nozzled booster [Ref. 5:p. 193].

The software allowed two choices for the solid ramjet fuel. These were UTX 18188 (hydrocarbon) and UTX 14660 (boron/HTPB). Missile designs using both fuels were generated, however, neither of these fuels has the structural strength and rigidity required for a caseless motor design. Software limitations prevented a caseless engine design. Thus, titanium was used in the computer design as the missile casing in order to minimize weight. In order to compare the performance of the fuels utilized by the software and a fuel which has the desired caseless motor characteristics, a performance comparison was made between the UTX 14660 fuel and a fuel composed of 40 % Plexiglas and 60 % boron. The performance comparison was based on their equilibrium, adiabatic combustion performance characteristics calculated using the Naval Weapons Center (NWC) Propellant Evaluation Program, NEWPEP [Ref. 6].

IV. EXPERIMENTAL APPARATUS AND PROCEDURE

The main thrust of the experimental aspect of the missile design was to test the feasibility of using a non-ejectable port cover. This port cover design must be simple and inexpensive to manufacture, rugged and capable of being sized for a small diameter missile. The second major thrust of the design was to test the feasibility of a caseless engine design by having the solid ramjet fuel also function as the outer casing of the engine. This requires the use of a fuel which is rigid and which has good structural characteristics. To minimize volume and provide high performance, a fuel with a high energy density is also desired. Another desired aspect of the fuel is that it be opaque in order to prevent subsurface heating by radiation effects from the flame and to reduce the visible signature. This opaque characteristic is especially critical in low light conditions, since there is no outer casing to block the visible or infrared emissions of the combustion process. In addition, it is required that the fuel pyrolysis temperature be high enough to prevent significant external erosion due to aerodynamic heating.

A potential candidate for a fuel which can meet these requirements is one which combines Plexiglas with a high energy metal. The Plexiglas would function as the fuel binder and provide the required structural strength and stiffness. To enhance the energy output of the fuel, particles of a high energy metal, such as aluminum or boron would be combined with the Plexiglas. Plexiglas contains 32 % oxygen by weight. The oxidizer should enhance surface ignition of the boron and provide improved combustion efficiency.

To enhance the structural strength of the fuel, nylon fibers or other appropriate fibers could possibly be added in small amounts without significantly affecting the combustion characteristics of the fuel.

To test the feasibility of having the IRSFRJ fuel also function as the outer casing of the motor, a simulated solid-fuel integral-rocket ramjet motor case was constructed for use on a test stand. Figure 4 describes this apparatus. The fuel grain consisted of a cylinder of Plexiglas and a metalized (4.7 % aluminum) composite propellant with a burning rate of 0.673 in/sec at 500 psi was used for the booster. Because of the limited propellant available an end-burning booster grain design was utilized, and the exhaust nozzle was sized to provide a nominal chamber pressure of 500 psi versus 1000 - 1500 psi in an actual motor design. The propellant was ignited using a pyrotechnic and a pressure-time trace of the run was recorded on an analog recorder. This test was conducted to determine the behavior of the Plexiglas when exposed to the high temperature propellant combustion products.

The non-ejectable port cover apparatus was constructed of stainless steel and was mounted on a static ramjet test stand. A schematic of the apparatus with the port cover closed is shown in Figure 5 while Figure 6 shows the port cover in the open position. The overall design of the port cover apparatus was uncomplicated and rugged, with emphasis placed on reliability and reproducibility of operation. A center dump design was utilized and the air flows from the twin inlets of the test stand were joined together at 45 degrees, upstream of the dump inlet.

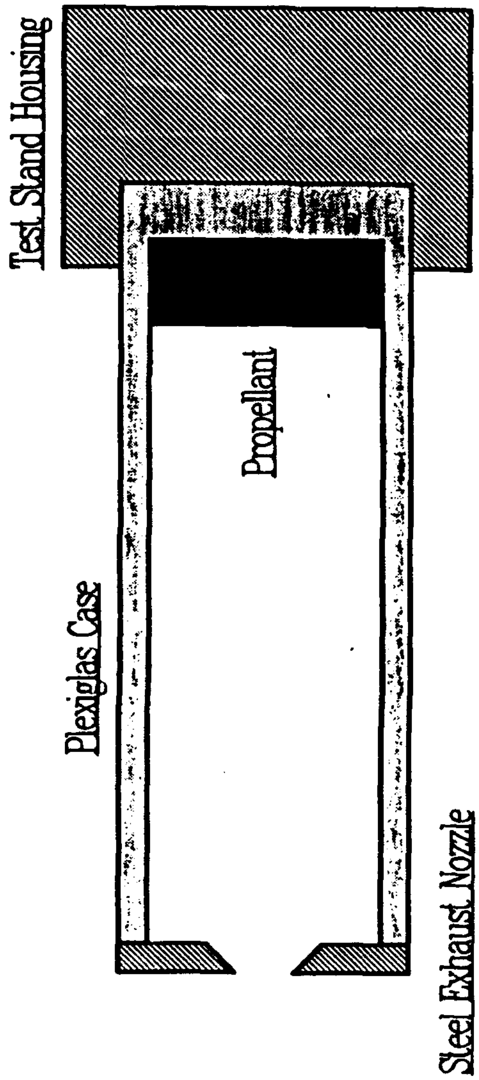

Figure 4. Apparatus for testing an IRSFRJ engine design.

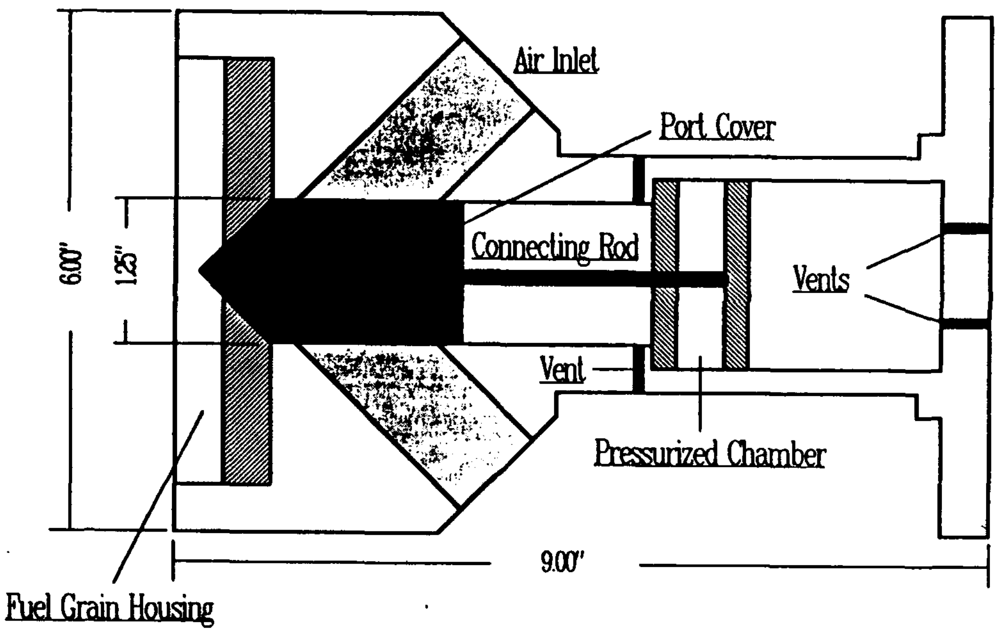

Figure 5. Non-ejectable port cover apparatus in the closed position.

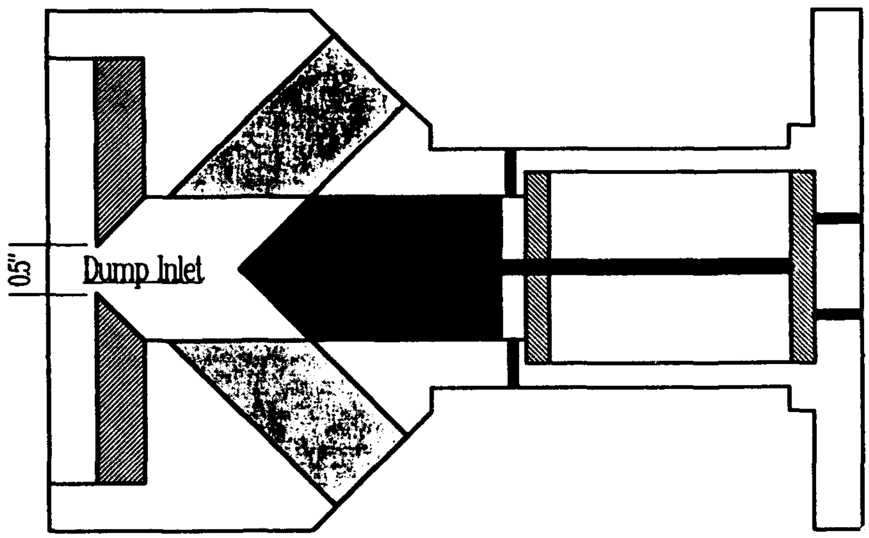

Figure 6. Non-ejectable port cover apparatus in the open position.

A sliding cylinder functioned as the port cover. Opening of the port cover, to allow for transition from the boost-phase to the sustain-phase, was activated by loss of boost pressure in the combustion chamber. The port cover was held in the closed position by a steel pin which could be retracted by a pneumatic device as shown in Figure 7. Port cover actuation was accomplished by withdrawing the pin and permitting the pressurized piston chamber to pull the port cover rearward. High temperature O-rings were used throughout the apparatus to withstand the high operating temperatures of the booster and sustainer and high inlet temperatures encountered at Mach 2.5.

Initial functional testing of the apparatus consisted of mounting it to the test stand and attaching a sealed fuel grain. The ability of the apparatus to maintain prescribed inlet pressures without leakage into the motor cavity was validated using pressurized air. The sealed fuel grain was then replaced with an open grain to allow for testing the proper mechanical operation of the apparatus with air flowing through the system. Controlling of the test sequence and of the actuating port cover was handled by a Hewlett Packard automatic data acquisition and control system. Repeated tests were conducted using cold air to properly sequence the opening of the port cover and related functions and to insure consistent and correct operation of the apparatus before conducting an actual test firing.

The goal of next phase of the testing of the apparatus was to determine the chamber pressure characteristics of the boost-phase. Limitations of the test facility and the available propellants prevented the casting of a propellent grain of a true nozzleless booster design. A nozzleless booster design allows for a propellant grain geometry which provides initial high booster chamber pressures, though the pressure drops off rapidly.

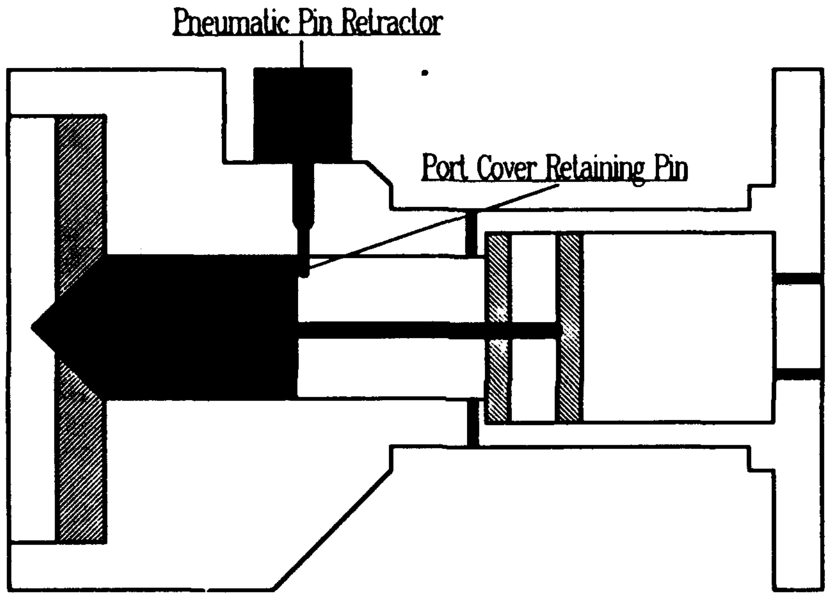

Figure 7. Port cover retaining pin and retracting mechanism.

It also makes it physically possible to use a single exhaust nozzle sized for the lower chamber pressures encountered during sustainer operation, without having to use an ejectable booster nozzle. Two modifications were made to simulate a nozzleless booster design in this situation. First, the operating chamber pressures of the booster and the sustainer phases were closely matched. Secondly, the nozzle was constructed of Plexiglas and was designed to erode during the booster phase. This allowed the nozzle to be initially sized for proper rocket motor booster-phase pressure and then increase in area enough to allow for correct chamber pressure and high enough flow rates during operation of the ramjet sustainer.

The boost propellant grain was a centered-perforated design and measured 2 inches in length with a web of 0.25 inches. A 0.25 in deep and 0.25 in length sliver of the Plexiglas fuel grain was removed from a recess at the forward end of the combustor, and the vacated space was filled with propellant. This was done to insure that once the booster propellant grain was consumed and the port cover was opened, that there would still be some propellant burning in the chamber to assist in the ignition of the sustainer fuel. The propellant was bonded to the Plexiglas fuel grain with RTV. Figure 8 shows the booster and sustainer grain configuration. A test run was conducted by igniting the solid propellant with a pyrotechnic and recording the chamber pressure trace on an analog recorder. The behavior of the eroding Plexiglas nozzle was also to be recorded. With this knowledge, the proper sequencing of the port cover opening was determined and preparations were made to perform the full transition test run.

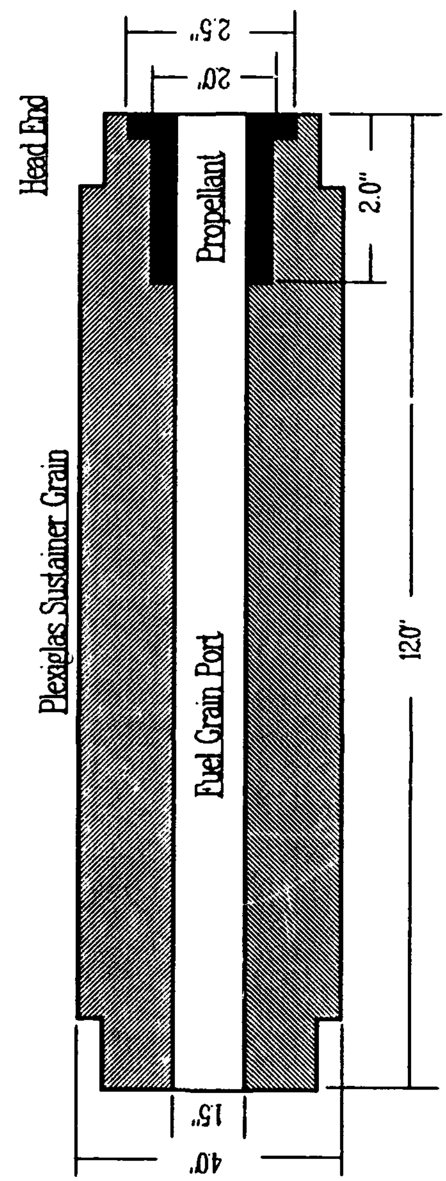

Figure 8. Caseless booster and sustainer grain configuration.

To perform the full transition run, inlet air conditions were simulated for a missile velocity of Mach 2.5 at an altitude of 20,000 ft. The inlet air was allowed to rest against the side of the closed port cover and an upstream bypass valve on the test stand allowed the hot, high pressure air to be vented to the atmosphere until the port cover was opened. The geometry of the fuel and propellant grains and the nozzle were kept the same as in the earlier boost-only run. The Hewlett Packard automated data acquisition and control system was programmed to monitor the inlet air temperature and pressure and the combustion chamber pressure. The system also controlled the operation of the bypass valve and port cover. Pre-run activities involved the calibration of pressure transducers and thermocouples and setting the correct flow rates of air, heater fuel and make up oxygen. The nominal mass flow rate of the air was set for 0.25 lbm/sec and the temperature of the air was set at 1020 degrees Rankine. The air mass flux (G) in the chamber port was expected to be 0.142 lbm/in^2 sec and the fuel-air ratio at these conditions was to be 0.054. The initial boost pressure was designed to be 170 psi with progressive burning tempered by the effects of the eroding exhaust nozzle throat area. The initial sustainer chamber pressure was designed to be 150 psi.

The full transition test began by initiating the program which controlled the operation of the Hewlett Packard data acquisition and control system. The ignition of the boost propellant was accomplished manually. The data acquisition and control system monitored the chamber pressure rise, and upon reaching 90 psi, the system then began monitoring for a pressure drop down to 75 psi. At this pressure, the commands were given by the control system to extract the port cover retaining pin, and simultaneously close the bypass valve. The hot air entering the combustion chamber, in combination with the small sliver of propellant still burning, was to ignite the Plexiglas and enable the initiation of the sustainer phase of the run. After a short sustainer run time, the combustion process was halted by opening the bypass valve and purging the combustion chamber with nitrogen. Figure 9a and 9b show photographs of the test apparatus complete with the fuel grain connected to the static test stand.

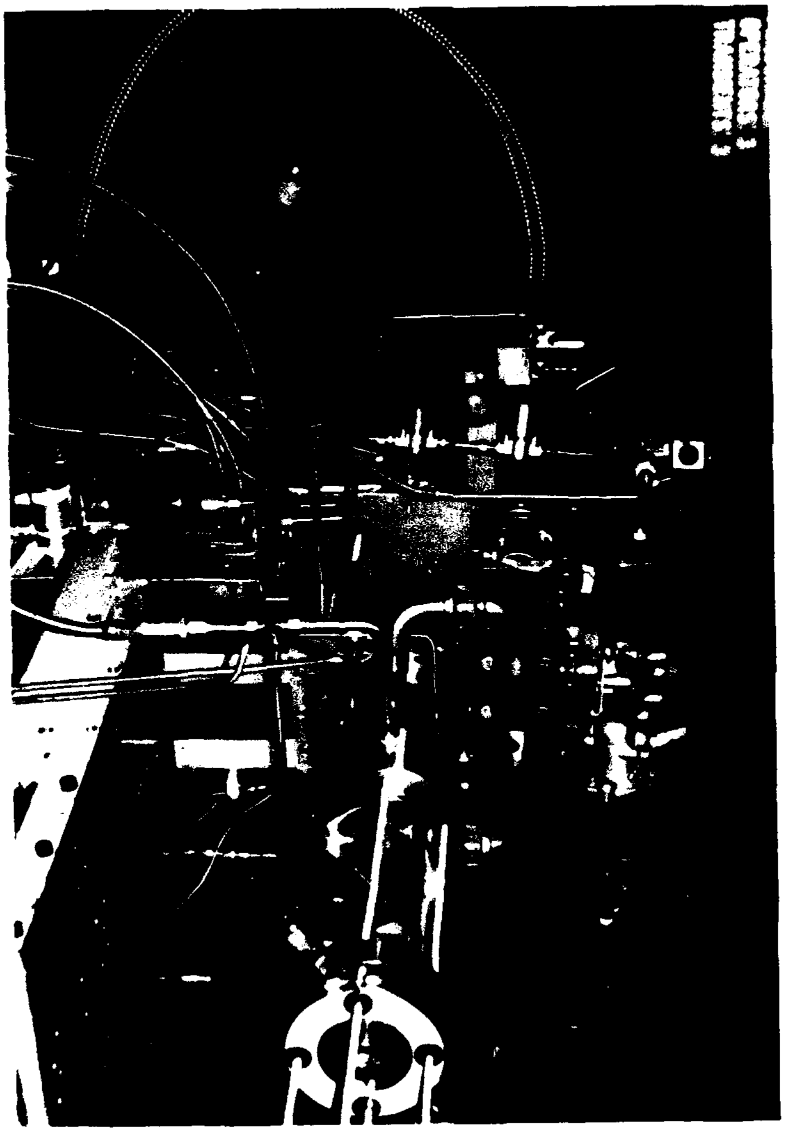

Figure 9a. Photograph of the IRSFRJ test apparatus and static test stand.

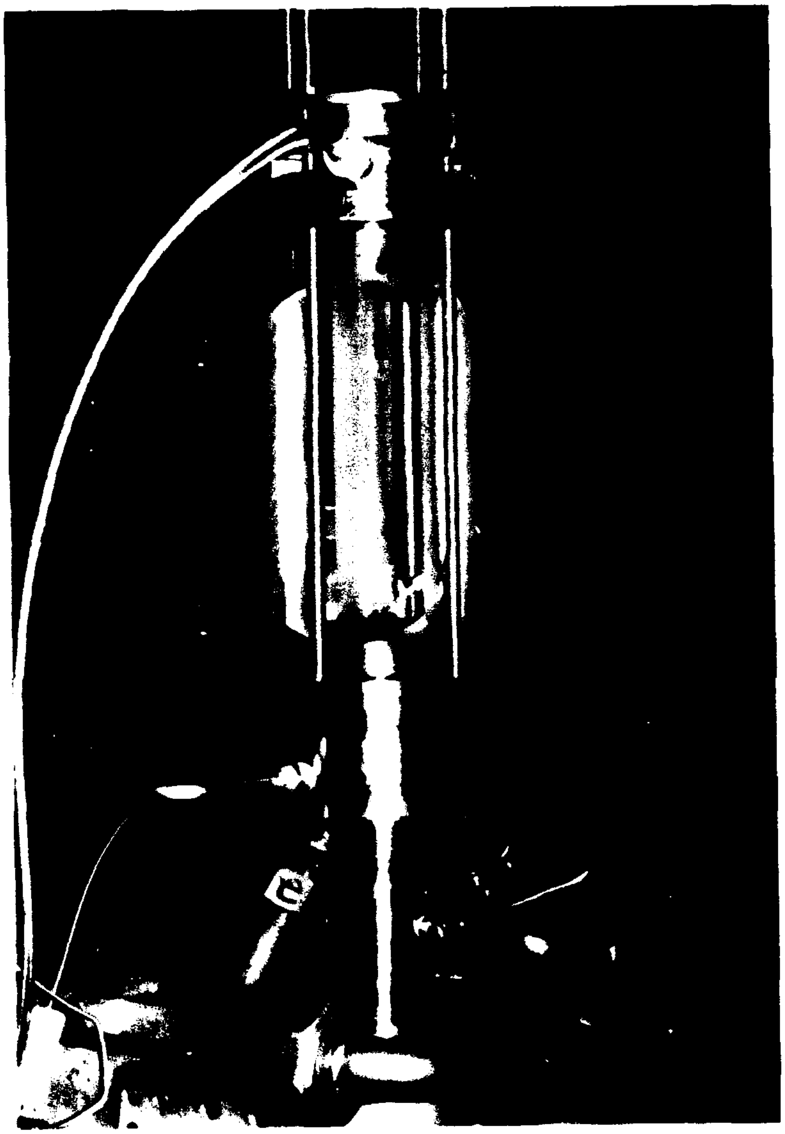

Figure 9b. Photograph of the IRSFRJ test apparatus.

V. DISCUSSION OF RESULTS

A. THEORETICAL STUDY

A total of six final missile configurations were generated. Three flight mission profiles were considered and missiles using the two types of fuel available in the software were generated for each flight profile. Table 2 below describes the three flight profiles.

TABLE 2

MISSILE FLIGHT PROFILES

FLIGHT PROFILE	ALT (FT)	LAUNCH (MACH)	CRUISE (MACH)	RANGE (NMI)	PURPOSE
1	20,000	0.3	2.3	20	High Altitude, RPV Mission
2	10,000	0.3	2.3	20	Mid Altitude, RPV Mission
3	2,000	0.3	2.2	12	Helicopter Mission

For each missile considered, the design software generates nine different engine configurations based on the input data. The selection criteria used to choose one of these nine configurations was to minimize missile weight with a fixed range restraint. Appendix A contains the complete printout of the results for the six missile configurations and Figure 13 shows engine station definitions.

Some of the characteristics common to all of the generated missiles are listed below:

- Nonpropulsion Weight - 22.5 lbs
- Wing Planform Area - 0.50 ft^2
- Tail Planform Area - 0.25 ft^2
- Nose Shape - Tan Ogive
- Inlet - Twin 2-D Cheek Mounted-Forward

Certain characteristics specific to each missile are shown in Table 3 below.

TABLE 3

MISSILE SPECIFIC OUTPUT DATA

MISSILE	**1**	**2**	**3**	**4**	**5**	**6**
FUEL TYPE	Boron	Boron	Boron	H/C	H/C	H/C
RANGE (NMI)	20.6	20.2	12.2	20.2	20.2	12.2
ALTITUDE (FT)	20K	10K	2K	20K	10K	2K
LENGTH (IN)	52.5	63.2	57.2	58.9	74.1	67.6
WEIGHT (LBS)	76.9	87.4	81.5	77.2	87.5	84.3
TAKE-OVER MACH	2.17	2.13	2.00	2.17	2.16	2.07
CRUISE MACH	2.25	2.20	2.15	2.30	2.30	2.20
ISP (SUSTAIN)	999	935	798	1088	1011	895
PROPULSION WT (LBS)	54.4	64.9	59.0	54.7	65.0	61.8
CASE WT (LBS)	6.15	8.20	7.05	7.35	10.21	9.00

The results indicated clearly that the design goals of the missile were met. In all cases, the boron based fuel provided equal missile performance with the hydrocarbon based fuel, but with less fuel required and a reduced fuel grain length, which consequently, reduces the overall length of the missile. One item to note is the weight of the required motor case. The case weight for all configurations was approximately 10 % of the total missile weight. Elimination of this motor casing by utilizing a caseless design would reduce the overall weight of the missile.

The comparison of the boron and HTPB based fuels utilized in the missile design software and the proposed caseless fuel composed of Plexiglas and boron was based on the grain geometry generated for missile number one detailed above. Comparisons were made for three performance parameters; jet specific impulse, net fuel specific impulse and specific thrust. Curves for each parameter, as it varied with the fuel-air ratio, were produced and the results for the two fuels were overlaid onto each other. The resulting graphs are contained in Appendix B. The jet specific impulse is defined by:

$$\frac{F_{jet}}{\dot{m}_e} = \frac{\dot{m}_e u_e}{\dot{m}_e g_c} = \frac{u_e}{g_c} \tag{1}$$

This value is obtained directly from PEPCODE. Figure 14, Appendix B shows the variation of jet specific impulse with changing fuel-air ratio. The fuel-air ratio is defined as:

$$f=\frac{\dot{m}_{fuel}}{\dot{m}_{air}} \tag{2}$$

The specific thrust is defined by:

$$\frac{F_{net}}{\dot{m}_o}=(1+f)\,\frac{u_e}{g_c}-\frac{u_o}{g_c} \tag{3}$$

The result is shown varying with the fuel-air ratio in Figure 15, Appendix B. The last performance parameter to be analyzed was the net fuel specific impulse. This is obtained by dividing the specific thrust by the fuel-air ratio and is given by:

$$I_{sp_{fuel}}=\frac{F_{net}}{\dot{m}_{fuel}} \tag{4}$$

The change in fuel specific impulse with the fuel-air ratio is shown in Figure 16, Appendix B.

The first performance comparison between the fuels was made by assuming the grain geometries, as defined in missile one above, were the same for both fuels and the regression rates for both fuels were also equal. The fuel-air ratio was 0.131 for the boron/HTPB fuel but changed to 0.160 for the boron/Plexiglass fuel and the fuel regression rate for both was taken as 0.0153 in/sec. The results for the three performance parameters described above are shown below in Table 4.

TABLE 4

FUEL COMPARISON WITH CONSTANT REGRESSION RATES

FUEL TYPE	JET SPECIFIC IMPULSE (sec)	SPECIFIC THRUST (sec)	FUEL SPECIFIC IMPULSE (sec)
BORON/HTPB	183	130	1,000
BORON/PLEXIGLAS	184	135	950

In this case, the fuels performed quite similarly with the boron/Plexiglas fuel providing slightly better specific thrust. The boron/HTPB fuel had an advantage in the range parameter of fuel specific impulse. The more dense boron/Plexiglas fuel would cause a 1.7 lb increase in fuel weight.

In the second performance comparison, the fuel grain geometry and the fuel-air ratio were kept the same for both fuels. This results in a different fuel regression rate and burn time for the two fuels. The burn time increased from 50.7 seconds for the boron/HTPB fuel to 62.2 seconds for the boron/Plexiglas fuel, about a 23 % increase. The performance parameters also changed and are shown below in Table 5.

TABLE 5

FUEL COMPARISON WITH CONSTANT FUEL-AIR RATIO

FUEL TYPE	JET SPECIFIC IMPULSE (sec)	SPECIFIC THRUST (sec)	FUEL SPECIFIC IMPULSE(sec)
BORON/HTPB	183	130	1,000
BORON/PLEXIGLAS	185	132	1,020

Once again, the values of the performance parameters of the two fuels were closely matched. If the regression rate, thus burn time of the two fuels were also kept the same, then the use of the boron/Plexiglas fuel would allow the fuel grain length to be reduced by 16.5 % while maintaining the same performance level as the boron/HTPB fuel.

One possible problem area involved in using a caseless fuel grain with Plexiglas as a major ingredient, is that its pyrolysis temperature is relatively low (600-650 degrees Kelvin). At a flight regime of Mach 2.5 at an altitude of 20,000 feet, the missile would experience a stagnation temperature of 559 degrees Kelvin. Thus, at these flight conditions, external erosion of the fuel grain due to aerodynamic heating should not be a problem. One possible solution for higher Mach number or lower altitude conditions would be to bond a light-weight insulator to the outside of the fuel grain. This would add very little to the weight of the missile, while only slightly increasing the cost and complexity of producing the missile.

The above calculations have shown that a 60 % boron and 40 % Plexiglas fuel grain should be able to provide the needed performance while also providing adequate material properties for a caseless motor design. Of course, the latter will have to be substantiated with structural testing.

B. EXPERIMENTAL RESULTS

The initial test to determine the compatibility with the booster combustion product gases was successful. One test firing was made and Figure 10 shows the chamber pressure-time trace. Peak pressure reached approximately 550 psi and the total burn time was about one second. Examination of the Plexiglas fuel grain after the test firing showed that the structural integrity of the grain was maintained and very little ablation of the Plexiglas occurred when exposed to the high temperature propellant combustion products.

The next series of test firings was made to determine the chamber pressure characteristics of the booster propellant grain and to determine the required sequence for full transition to ramjet combustion. Two test firings were conducted. Figure 11 shows the chamber pressure-time trace for one of the runs. The pressure-time trace showed that the sliver of propellant, which was design to enhance ignition of the sustainer fuel grain following the boost-phase, probably was consumed during booster tail-off. However, visual study of the test firing with the aid of a video camera showed that the Plexiglas grain remained very hot for several seconds after booster burnout. The results of the two firings also showed that the reproducibility of the chamber pressure characteristics was very good. The performance of the Plexiglas nozzle was as expected, with the nozzle area increasing approximately 30 % during the booster operation. Thus, the final sequencing of the port cover opening and related test stand activities could be made with a high confidence level in the proper functioning of the apparatus.

The full transition test firing was successful and the chamber pressure-time trace results are shown in Figure 12. The transition between the boost-phase and the sustain-phase occurred as anticipated. The initial bump in the chamber pressure during the sustainer phase can be attributed to the sliver of boost propellant still burning and/or to the rapid combustion of a hot, fluid layer of Plexiglas along the combustion chamber surface formed during the boost-phase. The regressive ramjet chamber pressure characteristic was due to the increasing exhaust nozzle throat area caused by the burning of the Plexiglas nozzle. One very encouraging result was that the booster-sustainer transition sequence was not critical. Booster chamber pressure completely decayed before the air was introduced, and yet no difficulties were observed for ignition of the hot Plexiglas grain surface. The integrity of the Plexiglas fuel grain and nozzle was maintained throughout the run. Lack of time prevented additional test runs to be conducted, but all indications seem to show that a high reproducibility of the results would be expected.

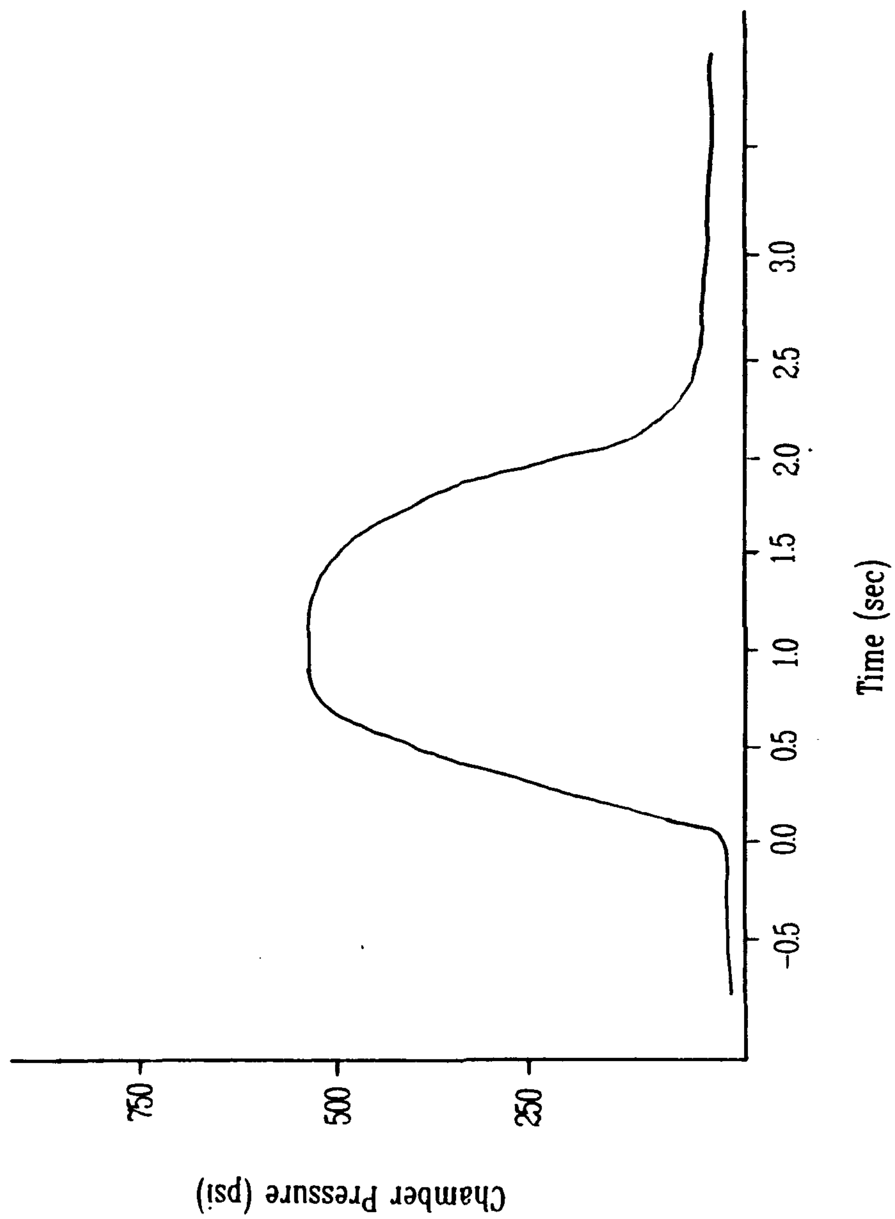

Figure 10. Chamber pressure-time trace for the caseless IRSFRJ feasibility test.

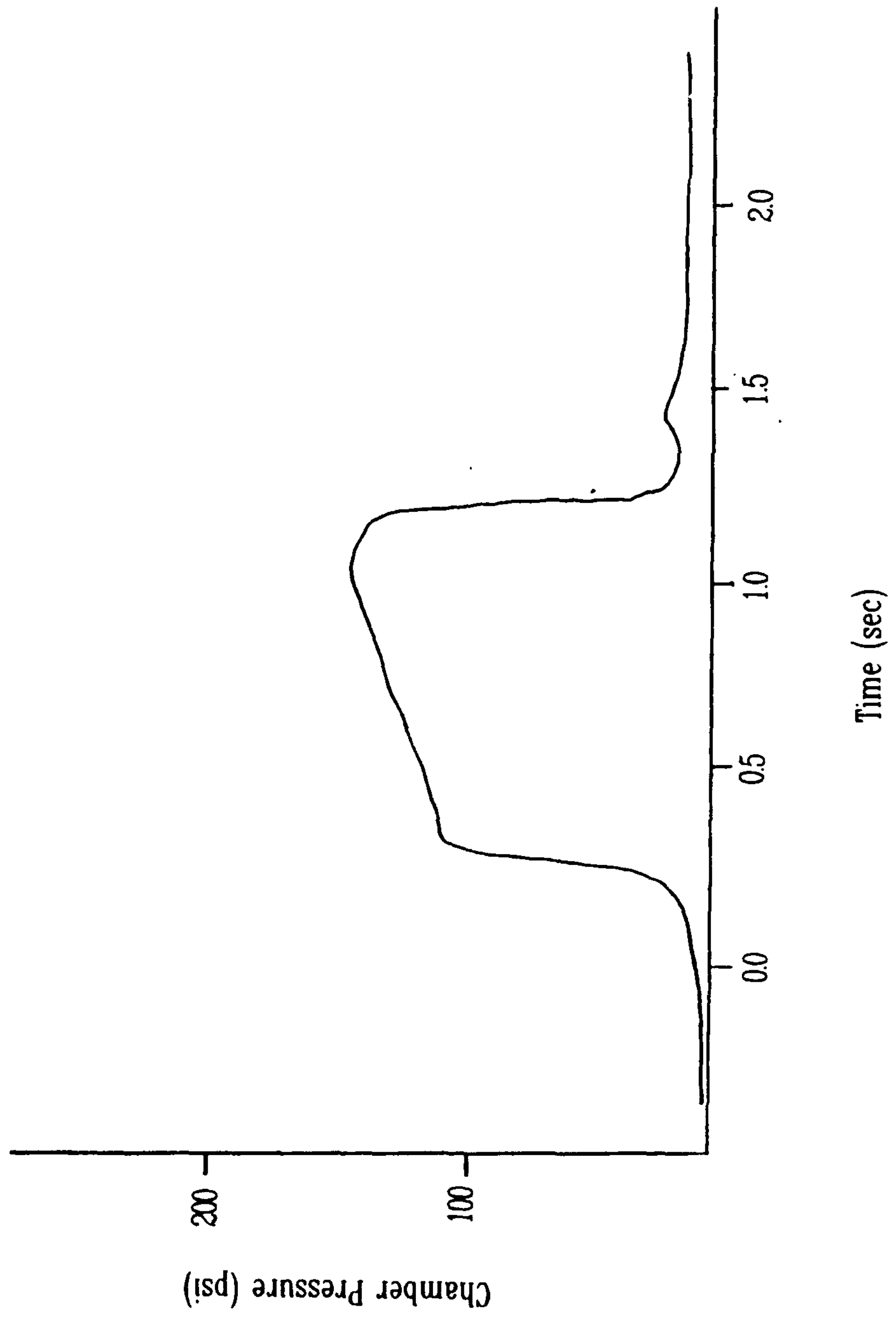

Figure 11. Chamber pressure-time trace of the boost-phase of the IRSFRJ test apparatus.

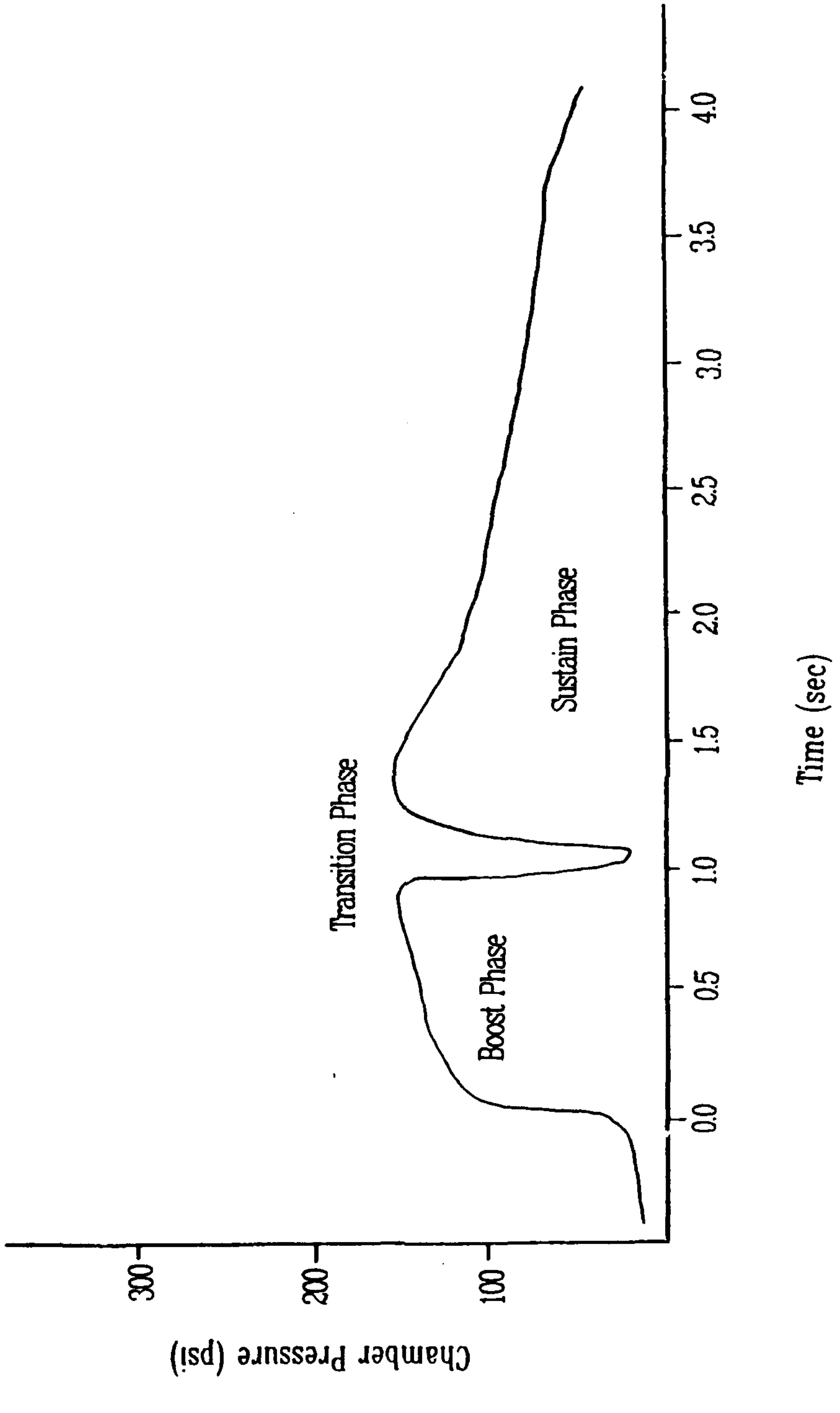

Figure 12. Chamber pressure-time trace for the full transition test.

VI. CONCLUSIONS

The results of both the analytical study and the experimental investigation appeared to validate the feasibility of using a solid-fuel integral-rocket ramjet to power a light-weight air-to-ground missile. Excellent booster-sustainer transition characteristics were obtained. Obviously, more detailed design and testing are required to validate the concept, especially concerning the structural capabilities of a caseless grain design which must operate at high booster pressures. However, it appears that a small, low cost, solid-fuel ramjet powered missile without ejecta can be designed and built which would provide a significant performance increase over current small air-to-ground missiles capable of being fired from RPV's or helicopters.

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5. Procinsky, I. M. and McHale, Catherine A., *Nozzeless Boosters for Integral-Rocket-Ramjet Missile Systems*, J. Spacecraft Vol. 18, No. 3, May - June 1981.

6. Cruise, D. R., *Theoretical Computations of Equilibrium Compositions, Thermodynamic Properties, and Performance Characteristics of Propellant Systems,* Naval Weapons Center Report NWC TP 6037, April 1979.

APPENDIX A - MISSILE DATA

TABLE 6

VARIABLE DEFINITIONS

VARIABLE NAME	DEFINITION	UNITS
CT5	Sustain Thrust Coefficient	
WF	Sustainer Fuel Flow Rate	lbm/sec
CD	Drag Coefficient	

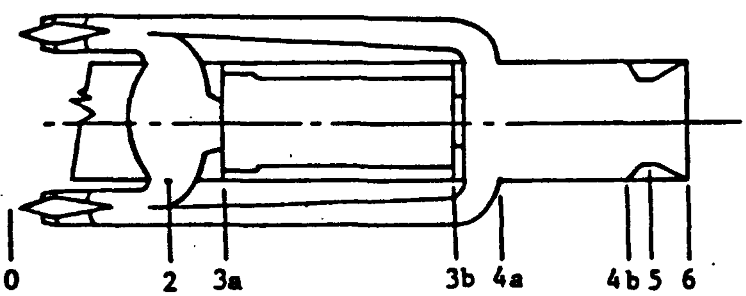

Figure 13. Engine station definitions (including bypass air).

* * * MISSILE 1 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 20.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:
WEIGHT= 76.9 LENGTH= 52.50 DIAMETER= 5.00 RANGE= 20.2(NMI)
LAUNCH MACH= .3 LAUNCH ALT= 20000.0
CRUISE MACH=2.3 CRUISE ALT= 20000.0
EOB VELOCITY=2712.6 AVG. VELOCITY=2284.3
TAIL CHASE RANGE= 20.06(NMI) MAX F-POLE=17.5(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(37.5)	(54.4)
IRR COMBUSTOR + NOZZLE	35.0	9.1
INLET+AFT FAIRING	(42.5)	5.5
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		24.2
FUEL		9.1
TOTAL	52.5	76.9

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .400(FT2)
TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER

NOZZLELESS
THRUST= 1587.8(LBS)
ISP= 209.9(SEC)

BURN TIME= 3.20
ATHROAT= .76(IN2)
RANGE= .80(NMI)

SUSTAINER ENGINE CHARACTERISTICS		AREAS IN (IN2)	
AC=	8.154	ACA3=	.4360
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.167	AT MDES= 2.250	AT MCR= 2.250
CT5	.6170	.5530	.5530
CD	.3428	.3514	.3514
WF	.1735	.1735	.1735
ISP	1034.0510	999.2504	999.2504
COMB.EFF	.8500	.8500	.8500

SUSTAIN BURN TIME= 50.66(SEC) SUSTAIN RANGE= 19.44(NMI)

SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.250
SREF= .1364(FT2)

TOTAL= .3540
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD= .3096 V=3500 FPS

* * * MISSILE 2 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 20.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:
WEIGHT= 87.4 LENGTH= 63.21 DIAMETER= 5.00 RANGE= 20.2(NMI)
LAUNCH MACH= .2 LAUNCH ALT= 10000.0
CRUISE MACH=2.2 CRUISE ALT= 10000.0
EOB VELOCITY=2567.2 AVG. VELOCITY=2304.3
TAIL CHASE RANGE= 20.05(NMI) MAX F-POLE=18.3(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(48.2)	(64.9)
IRR COMBUSTOR + NOZZLE	45.7	11.1
INLET+AFT FAIRING	(53.2)	5.6
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		27.9
FUEL		13.9
TOTAL	63.2	87.4

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .400(FT2)

TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1666.8(LBS)
ISP= 204.0(SEC)

BURN TIME= 3.41
ATHROAT= .79(IN2)
RANGE= .78(NMI)

SUSTAINER ENGINE CHARACTERISTICS AREAS IN (IN2)

AC=	7.888	ACA3=	.4218
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.133	AT MDES= 2.200	AT MCR= 2.200
CT5	.6141	.5623	.5623
CD	.3412	.3484	.3484
WF	.2696	.2696	.2696
ISP	960.0958	934.9776	934.9776
COMB.EFF	.8500	.8500	.8500

SUSTAIN BURN TIME= 49.94(SEC) SUSTAIN RANGE= 19.44(NMI)

```
            SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.200
                        SREF=    .1364(FT2)
TOTAL CD=  .3384
INCLUDES 1.1 FACTOR FOR PROTUBERANCES, ETC

BOOST CD=  .3037 V=3500 FPS
```

* * * MISSILE 3 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 12.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:

WEIGHT= 81.5 LENGTH= 57.23 DIAMETER= 5.00 RANGE= 12.2(NMI)
LAUNCH MACH= .2 LAUNCH ALT= 2000.0
CRUISE MACH=2.2 CRUISE ALT= 2000.0
EOB VELOCITY=2546.5 AVG. VELOCITY=2283.8
TAIL CHASE RANGE= 12.07(NMI) MAX F-POLE=11.0(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(42.2)	(59.0)
IRR COMBUSTOR + NOZZLE	39.7	10.0
INLET+AFT FAIRING	(47.2)	4.7
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		26.6
FUEL		11.2
TOTAL	57.2	81.5

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .400(FT2)

TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1635.5(LBS)
ISP= 199.1(SEC)

BURN TIME= 3.24
ATHROAT= .78(IN2)
RANGE= .74(NMI)

SUSTAINER ENGINE CHARACTERISTICS		AREAS IN (IN2)	
AC=	6.724	ACA3=	.3596
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.000	AT MDES= 2.150	AT MCR= 2.150
CT5	.6181	.5129	.5129
CD	.3434	.3373	.3373
WF	.3719	.3719	.3719
ISP	832.4511	798.1639	798.1639
COMB.EFF	.8500	.8500	.8500

SUSTAIN BURN TIME= 29.18(SEC) SUSTAIN RANGE= 11.44(NMI)

```
                SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.150
                              SREF=     .1364(FT2)
TOTAL CD=    .3357
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD=   .2891 V=3500 FPS
```

* * * MISSILE 4 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 20.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:

WEIGHT= 77.2   LENGTH= 58.93   DIAMETER= 5.00   RANGE= 20.2(NMI)
LAUNCH MACH= .3   LAUNCH ALT= 20000.0
CRUISE MACH=2.3   CRUISE ALT= 20000.0
EOB VELOCITY=2676.5   AVG. VELOCITY=2306.8
TAIL CHASE RANGE= 20.05(NMI)   MAX F-POLE=17.5(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(43.9)	(54.7)
IRR COMBUSTOR + NOZZLE	41.4	10.2
INLET+AFT FAIRING	(48.9)	5.6
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		24.0
FUEL		8.4
TOTAL	58.9	77.2

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .500(FT2)

TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1578.6(LBS)
ISP= 209.9(SEC)

BURN TIME= 3.20
ATHROAT= .75(IN2)
RANGE= .79(NMI)

SUSTAINER ENGINE CHARACTERISTICS

AC= 7.992
A5= 13.090
A6= 19.635
A3= 18.700

AREAS IN (IN2)

ACA3= .4274
A5A3= .7000
A6A3= 1.0500
A2A3= .6000

SUSTAINER PERFORMANCE

	AT MTO= 2.167	AT MDES= 2.250	AT MCR= 2.300
CT5	.6390	.5757	.5406
CD	.3550	.3633	.3588
WF	.1629	.1629	.1629
ISP	1140.7050	1108.7260	1087.9210
COMB.EFF	.9295	.9307	.9313

SUSTAIN BURN TIME= 50.11(SEC)   SUSTAIN RANGE= 19.44(NMI)

SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.300
SREF= .1364(FT2)

TOTAL CD= .3567
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD= .3192 V=3500 FPS

* * * MISSILE 4 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 20.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:

WEIGHT= 77.2 LENGTH= 58.93 DIAMETER= 5.00 RANGE= 20.2(NMI)
LAUNCH MACH= .3 LAUNCH ALT= 20000.0
CRUISE MACH=2.3 CRUISE ALT= 20000.0
EOB VELOCITY=2676.5 AVG. VELOCITY=2306.8
TAIL CHASE RANGE= 20.05(NMI) MAX F-POLE=17.5(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(43.9)	(54.7)
IRR COMBUSTOR + NOZZLE	41.4	10.2
INLET+AFT FAIRING	(48.9)	5.6
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		24.0
FUEL		8.4
TOTAL	58.9	77.2

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .500(FT2)
TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1578.6(LBS)
ISP= 209.9(SEC)

BURN TIME= 3.20
ATHROAT= .75(IN2)
RANGE= .79(NMI)

SUSTAINER ENGINE CHARACTERISTICS		AREAS IN (IN2)	
AC=	7.992	ACA3=	.4274
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.167	AT MDES= 2.250	AT MCR= 2.300
CT5	.6390	.5757	.5406
CD	.3550	.3633	.3588
WF	.1629	.1629	.1629
ISP	1140.7050	1108.7260	1087.9210
COMB.EFF	.9295	.9307	.9313

SUSTAIN BURN TIME= 50.11(SEC) SUSTAIN RANGE= 19.44(NMI)

SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.300
SREF= .1364(FT2)

TOTAL CD= .3567
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD= .3192 V=3500 FPS

* * * MISSILE 5 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 20.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:

WEIGHT= 87.5 LENGTH= 74.13 DIAMETER= 5.00 RANGE= 20.2(NMI)
LAUNCH MACH= .3 LAUNCH ALT= 10000.0
CRUISE MACH=2.3 CRUISE ALT= 10000.0
EOB VELOCITY=2542.1 AVG. VELOCITY=2391.8
TAIL CHASE RANGE= 20.04(NMI) MAX F-POLE=17.5(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(59.1)	(65.0)
IRR COMBUSTOR + NOZZLE	56.6	13.1
INLET+AFT FAIRING	(64.1)	6.0
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		26.7
FUEL		12.7
TOTAL	74.1	87.5

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .500(FT2)

TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1669.7(LBS)
ISP= 204.0(SEC)

BURN TIME= 3.26
ATHROAT= .80(IN2)
RANGE= .77(NMI)

SUSTAINER ENGINE CHARACTERISTICS

		AREAS IN (IN2)	
AC=	8.066	ACA3=	.4313
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.167	AT MDES= 2.250	AT MCR= 2.300
CT5	.6299	.5662	.5310
CD	.3499	.3583	.3539
WF	.2572	.2572	.2572
ISP	1064.8560	1032.5740	1011.8540
COMB.EFF	.9213	.9223	.9228

SUSTAIN BURN TIME= 48.11(SEC) SUSTAIN RANGE= 19.44(NMI)

SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.300
SREF= .1364(FT2)

TOTAL CD= .3384
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD= .3110 V=3500 FPS

* * * MISSILE 6 * * *
* SOLID FUEL RAMJET *
* RANGE OBJECTIVE= 12.0 *
MISSILE SELECTION CRITERIA IS MIN WEIGHT

MISSILE PARAMETERS:

WEIGHT= 84.3 LENGTH= 67.55 DIAMETER= 5.00 RANGE= 12.2(NMI)
LAUNCH MACH= .2 LAUNCH ALT= 2000.0
CRUISE MACH=2.2 CRUISE ALT= 2000.0
EOB VELOCITY=2517.6 AVG. VELOCITY=2308.5
TAIL CHASE RANGE= 12.09(NMI) MAX F-POLE=11.0(NMI)

MISSILE COMPONENTS:

COMPONENT	LENGTH(IN)	WEIGHT(LBS)
NONPROPULSION	15.0	22.5
PROPULSION	(52.5)	(61.8)
IRR COMBUSTOR + NOZZLE	50.0	11.9
INLET+AFT FAIRING	(57.5)	5.2
MISCELLANEOUS		5.0
PLENUM	2.5	1.4
PROPELLANT		27.4
FUEL		10.9
TOTAL	67.5	84.3

NOSE

SHAPE= TAN OGIVE
FINENESS RATIO=2.5
BLUNTNESS RATIO= .2200

CONTROL SURFACES

WING PLANFORM AREA= .500(FT2)
TAIL PLANFORM AREA= .250(FT2)

PROPULSION:

BOOSTER
NOZZLELESS
THRUST= 1636.1(LBS)
ISP= 199.1(SEC)

BURN TIME= 3.33
ATHROAT= .78(IN2)
RANGE= .75(NMI)

SUSTAINER ENGINE CHARACTERISTICS		AREAS IN (IN2)	
AC=	7.109	ACA3=	.3802
A5=	13.090	A5A3=	.7000
A6=	19.635	A6A3=	1.0500
A3=	18.700	A2A3=	.6000

SUSTAINER PERFORMANCE

	AT MTO= 2.067	AT MDES= 2.100	AT MCR= 2.200
CT5	.6421	.6154	.5412
CD	.3567	.3605	.3447
WF	.3666	.3666	.3666
ISP	936.6643	927.0051	894.6385
COMB.EFF	.8999	.9005	.9021

SUSTAIN BURN TIME= 28.77(SEC) SUSTAIN RANGE= 11.44(NMI)

SUSTAIN DRAG COMPONENTS AT CRUISE MACH= 2.200
SREF= .1364(FT2)

TOTAL= .3364
INCLUDES 1.1 FACTOR FOR PROTUBERANCES,ETC.

BOOST CD= .3091 V=3500 FPS

APPENDIX B - PERFORMANCE CURVES

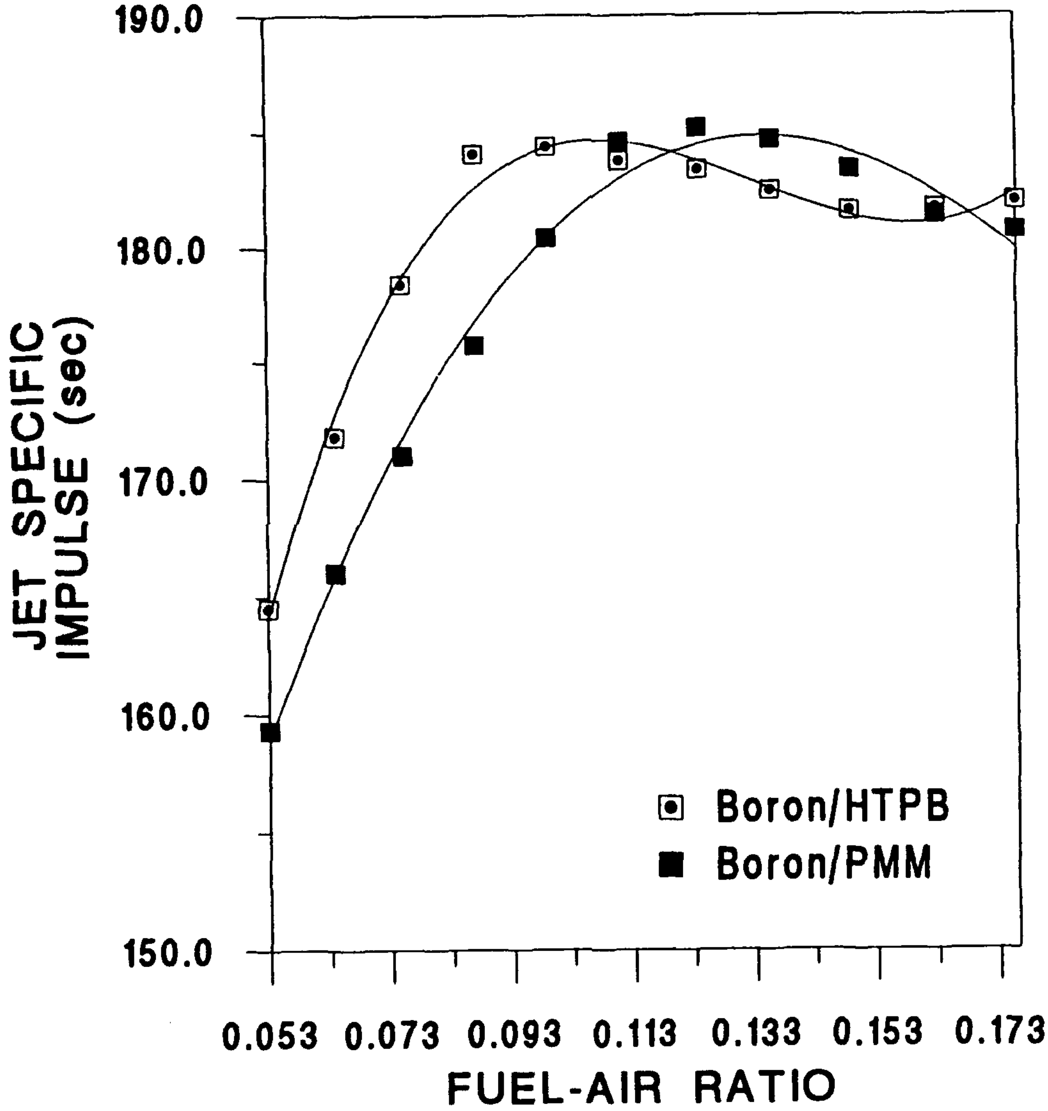

Figure 14. Jet specific impulse variation with fuel-air ratio.

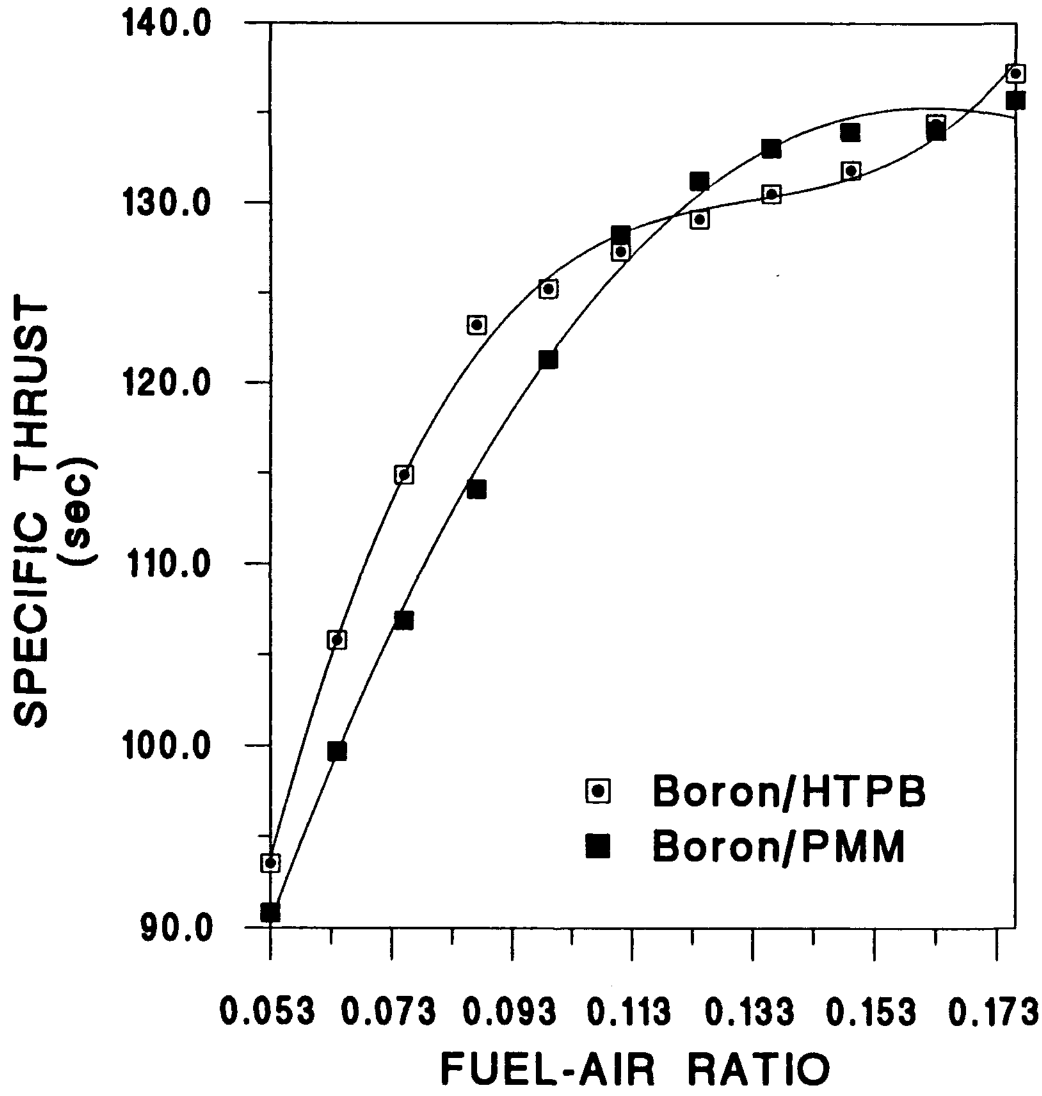

Figure 15. Specific thrust variation with fuel-air ratio.

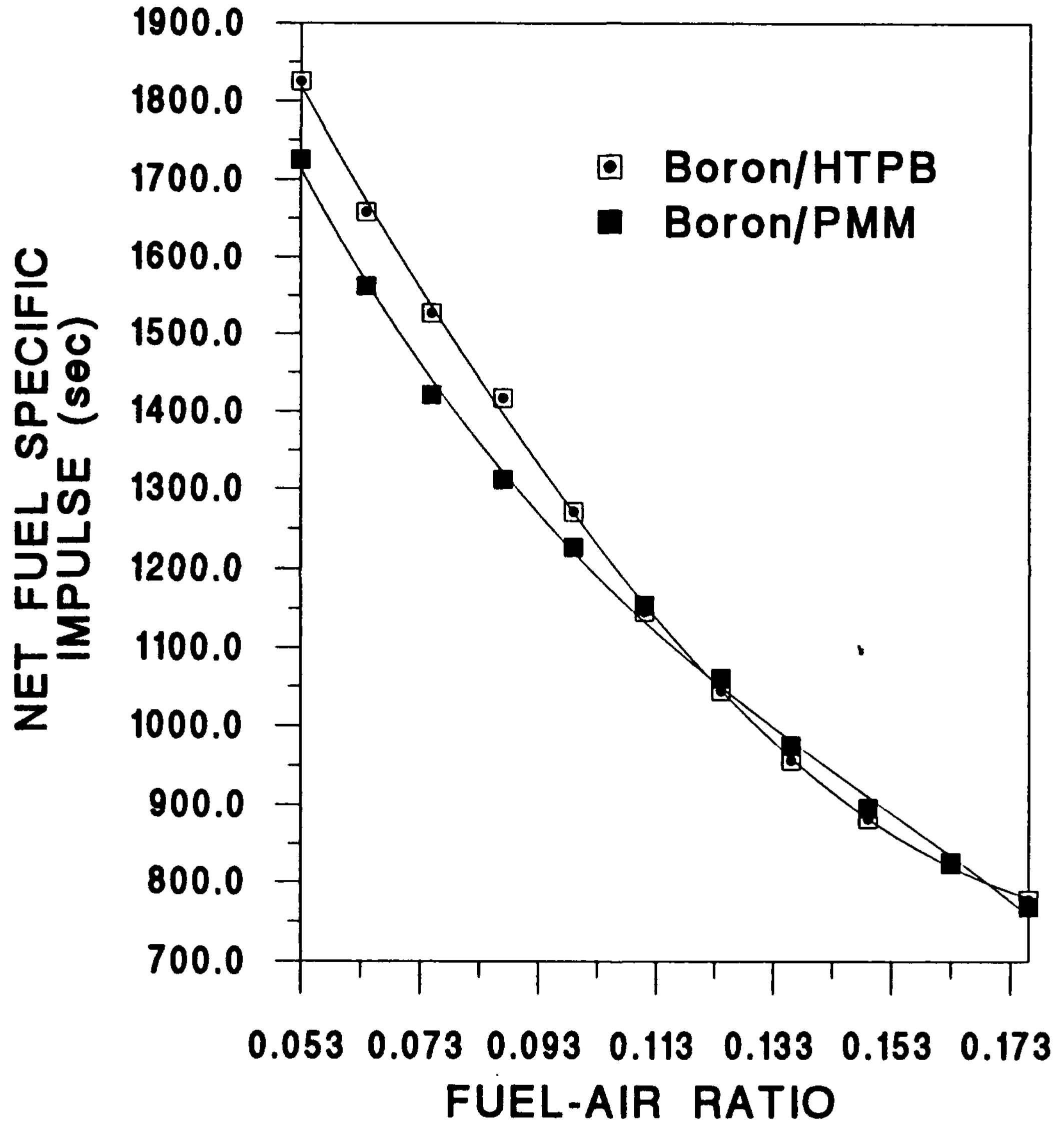

Figure 16. Fuel specific impulse variation with fuel-air ratio.

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