IN-29192

NASA Technical Memorandum 88822

Proven, Long-Life Hydrogen/Oxygen Thrust Chambers for Space Station Propulsion

G. Paul Richter and Harold G. Price
Lewis Research Center
Cleveland, Ohio

Prepared for the
1986 JANNAF Propulsion Meeting
sponsored by the JANNAF Interagency Propulsion Committee
New Orleans, Louisiana, August 26–28, 1986

NASA

E-3171

PROVEN, LONG-LIFE HYDROGEN/OXYGEN THRUST CHAMBERS FOR SPACE STATION PROPULSION

G. Paul Richter and Harold G. Price
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

The development of the manned Space Station has necessitated the development of technology related to an onboard auxiliary propulsion system (APS) required to provide for various Space Station attitude control, orbit positioning, and docking maneuvers. A key component of this onboard APS is the thrust chamber design. To develop the required thrust chamber technology to support the Space Station Program, the NASA Lewis Research Center has sponsored development programs under contracts with Aerojet TechSystems Company and with Bell Aerospace Textron Division of Textron, Inc.

During the NASA Lewis sponsored program with Aerojet TechSystems, a 25 lb_f hydrogen/oxygen thruster has been developed and proven as a viable candidate to meet the needs of the Space Station Program. Likewise, during the development program with Bell Aerospace, a 50 lb_f hydrogen/oxygen Thrust Chamber has been developed and has demonstrated reliable, long-life expectancy at anticipated Space Station operating conditions.

Both these thrust chambers were based on design criteria developed in previous thruster programs and successfully verified in experimental test programs. Extensive thermal analyses and models were used to design the thrusters to achieve total impulse goals of $2x10^6$ lb_f-sec.

Test data for each thruster will be compared to the analytical predictions for the performance and heat transfer characteristics. Also, the results of thrust chamber life verification tests will be presented.

INTRODUCTION

Of the many systems onboard the manned Space Station, one of the most important will be the Auxiliary Propulsion System, or APS. This system will provide the propulsion for various Space Station attitude control, orbit positioning, and docking and avoidance maneuvers.

Key components in this onboard APS are the thrust chambers, which must be highly reliable, long-life expectancy units, with reasonable performance, and available at low cost.

One of the candidate systems being considered for the Space Station includes gaseous hydrogen (GH_2) and gaseous oxygen (GO_2) as propellants for the APS. A program to develop and demonstrate the required thrust chamber technology to support the Space Station Program was initiated by NASA Lewis early in 1985. One of the objectives of this program was to consider various thrust chamber concepts, and to "fine-tune" the most promising approach prior to the initiation of the Space Station design and fabrication efforts.

Separate programs were conducted at Aerojet TechSystems Company (NAS3-24398) and Bell Aerospace Textron (NAS3-24656), sponsored by NASA, to design, fabricate and test GH_2/GO_2 thrust chambers utilizing each Contractor's unique concept.

This paper presents the results of the analytical, design, and experimental test efforts to meet this objective.

AEROJET TECHSYSTEMS THRUSTER

Considering first the Aerojet TechSystems 25 lb_f thruster, the following Design Requirements were established:

TABLE I. - THRUSTER DESIGN REQUIREMENTS

PROPELLANTS	GO_2/GH_2
MIXTURE RATIO, O/F	4.0±1.0
SPECIFIC IMPULSE	≥400 LBF-SEC/LBM
FUEL INLET TEMPERATURE	≥200 °R
OXIDIZER INLET TEMPERATURE	≥300 °R
TOTAL IMPULSE	$2x10^6$-LBF SEC

Based on system considerations, the thruster was designed to operate at a thrust of 25 lb_f and a chamber pressure of 75 psia.

Several programs (Refs. 1 to 3) conducted by Aerojet for NASA in the early 1970's; and a program sponsored by Jet Propulsion Laboratory (JPL) in the early 1980's (Refs. 4 and 5) provide the basis and test experience for the Space Station Thruster design. Figure 1 shows the complete Space Station thruster assembly.

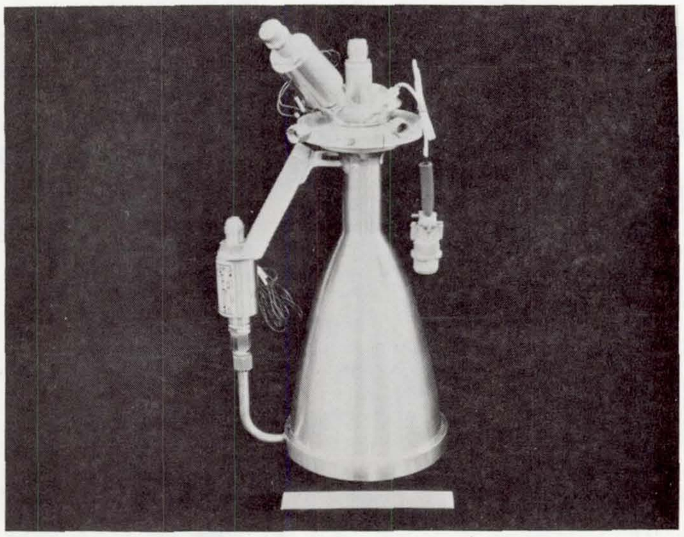


FIGURE 1. - AEROJET SPACE STATION THRUSTER ASSEMBLY.

The thruster assembly, as shown in the cutaway view of Fig. 2, consists of a combination (integral) igniter and injector, a regeneratively cooled thrust chamber, fuel balancing orifices, and a sleeve insert at the chamber forward end. Integral, directly-actuated poppet-type valves control the flow of propellants to the thruster. GH_2 fuel is used to regeneratively cool the thrust chamber, flowing counter to the combustion gases. As the fuel exits the chamber coolant channels at the forward end, it is collected in an annular manifold formed at the interface between the integral igniter/injector and the thrust chamber. This annular manifold feeds two sets of radial flow passages. One set supplies hydrogen to the injector where it impinges on the spark-energized oxygen, causing ignition in the oxidizer-rich (O/F = 16) core. The second set directs the remainder of the hydrogen into slotted passages within the sleeve, Fig. 3. This hydrogen is injected as film-cooling along the inner wall of the thrust chamber. The film coolant and injector core streams progressively mix as the core flow is entrained into the film coolant. Design and fabrication details are covered in Ref. 10.

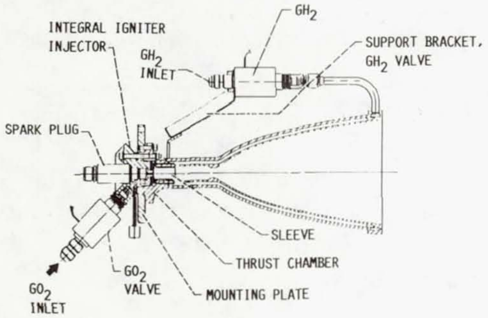


FIGURE 2.- CUTAWAY VIEW OF THRUSTER ASSEMBLY.

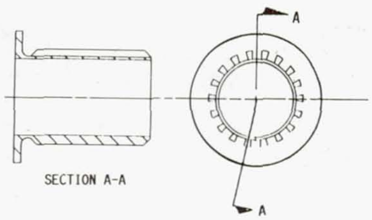


FIGURE 3.- SLEEVE INSERT.

ANALYSIS

Ignition Analysis. An extensive data base and analytical/design capability have been established for spark-initiated igniters, highlighted by Figs. 4 and 5. The Space Station thruster spark ignition parameters are based on and consistent with these figures and the historical experience for this igniter concept (Refs. 1 to 3).

Thermal Analysis. The JPL test experience, Ref. 5, indicated that the mixing efficiency, E_m is 1.0 (100 percent) with 60 percent fuel film cooling (FFC), as shown by the performance data of Fig. 6, with the corresponding maximum specific impulse of 438 $lb_f\text{-sec}/lb_m$. However, since the thruster sleeve cooling was predicted to be marginal at 60 percent FFC, 75 percent FFC was selected for the Space Station thruster baseline design.

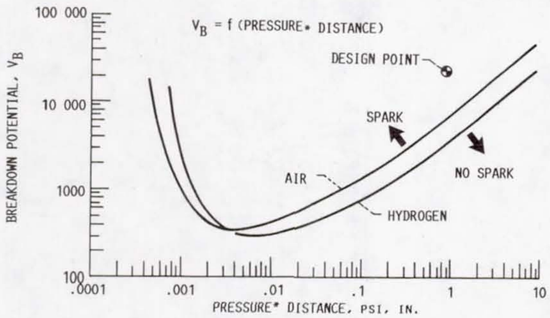


FIGURE 4.- PASCHEN'S LAW CURVE FOR AIR.

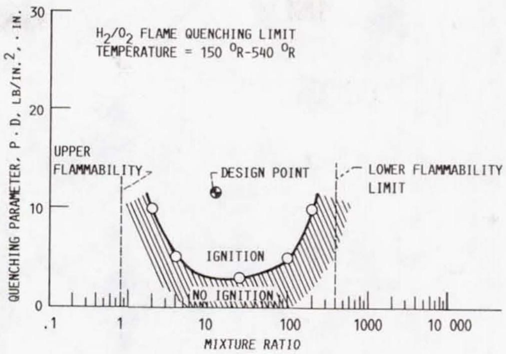


FIGURE 5.- FLAME QUENCHING LIMIT FOR O_2/H_2.

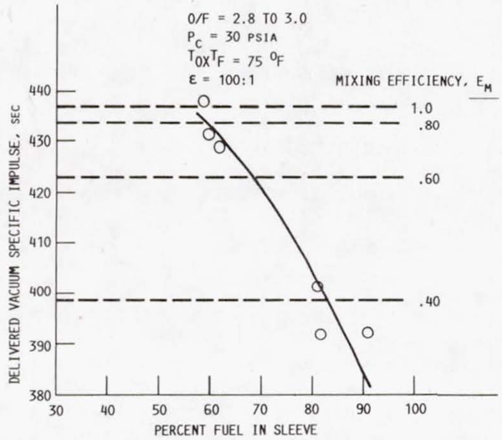


FIGURE 6.- EXPERIMENTALLY DETERMINED EFFECT OF FUEL FILM COOLANT ON DELIVERED SPECIFIC IMPULSE AND MIXING EFFICIENCY.

A thermal model of the thruster has been developed with the computer code HOCOOL (Refs. 6 and 7) which is used for determining cooling requirements and for predicting thermal response of rocket thrust chambers, specifically those with hydrogen film and regenerative cooling. HOCOOL requires the input of two empirical constants: the heat transfer correlation coefficient, Cg_n, and the entrainment fraction, KSO. The first constant, Cg_n, accounts for flow acceleration effects and injector characteristics on chamber heat transfer. The second constant, KSO, defines the rate of entrainment of the core gases into the mixing layer, made up of the film coolant and the core gases.

The Cg_n values and the KSO were based on the JPL thruster test program, during which a thin-walled 100:1 area ratio rhenium thruster was fired over a range of chamber pressure from 30 to 90 psia and mixture ratios from 2.0 to 3.4. By performing transient wall analyses with a one-dimensional heat conduction code, temperature transients from the test data have been matched to determine heat transfer coefficients and adiabatic wall temperatures. A typical comparison of the empirical adiabatic wall temperatures with HOCOOL calculations is shown in Fig. 7. HOCOOL very accurately predicts the throat adiabatic wall temperature but, as noted, somewhat overpredicts divergent nozzle temperatures. Appropriate modifications, based on test data, have been made to the predictions for design purposes.

Chamber Life Analysis. The 2.0×10^6 lb_f-sec total impulse requirement equates to a total firing duration of 22.2 hr at a thrust of 25 lb_f. During the Space Station 10-year lifetime design for RCS thrusters, the actual duty cycle is expected to comprise about 500 deep thermal cycles and possibly 100 000 impulse bits. The estimated cycle life for the thruster is based on the Manson-Halford method of universal slopes. The thermal strain in the chamber wall is determined at the point of maximum gas side temperature, using finite element thermal and structural models. The calculated strain range, $\Delta\varepsilon$ is used, as shown in Fig. 8, to obtain the predicted cycle life for a given temperature and time at temperature, t. Chamber life expectancy for the Space Station thruster was estimated to exceed 500 deep thermal cycles, and based on wall temperatures being lower than originally predicted, the updated life prediction is about 2000 deep thermal cycles. Impulse bit capability is effectively infinite, because very little thermal strain is developed during short firings.

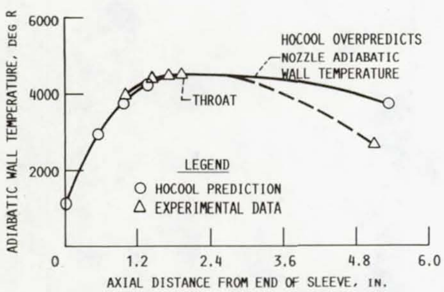


FIGURE 7.- COMPARISON OF CALCULATED THRUSTER ADIABATIC WALL TEMPERATURE PROFILE WITH HOCOOL PREDICTION.

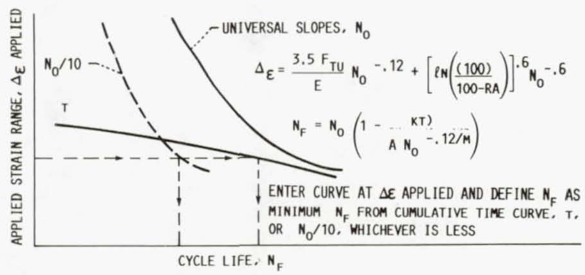


FIGURE 8.- MANSON-HALFORD METHOD OF UNIVERSAL SLOPES FOR LOW CYCLE FATIGUE.

Performance Analysis. Combustion performance of GO_2/GH_2 has been evaluated parametrically over broad ranges of design points and operating conditions using the TDK and TBL computer programs (Refs. 8 and 9). The results of these parametric studies have been incorporated into a performance prediction model called ROCKET. To account for the impact of incomplete mixing, ROCKET utilizes a mixing efficiency parameter, Em, which is defined for the simplified two stream tube flow characterization.

Performance predictions were based on thermal analysis of the JPL test data which characterized the mixing between the core and the film coolant streams in terms of E_m values. Subsequent analysis using ROCKET and these E_m values determined the delivered specific impulse for various operating points being evaluated.

TEST OBJECTIVES

The primary objective of the original Aerojet test program was to demonstrate $2x10^6$ lb_f-sec total impulse over a mixture ratio (r) range of 3.0 to 5.0. High performance was not considered an objective, however, 400 lb_f-sec/lb_m Isp was defined as a minimum goal. As the program progressed, a wider range of desired mixture ratios was incorporated, to demonstrate that the thruster could operate successfully on the electrolysis products of water, i.e., at r = 8.0 Potentially, only 20 percent of the Space Station total impulse will be generated at r = 8.0, which will be approximately 200 000 lb_f-sec, thus establishing a revised goal at that level.

TEST SETUP

The Space Station thruster has been tested in the altitude test facility shown in Figure 9, which is equipped with a hardware test cell, an 11 000 ft^3 altitude chamber capable of maintaining >100 000 ft simulated altitude, and the necessary auxiliary instrumentation, controls, and data recording equipment.

The thruster hardware mounts to a test stand designed to measure thrust, as shown in Fig. 10. The thruster and stand are installed as a subassembly into the test cell, as shown in Fig. 11.

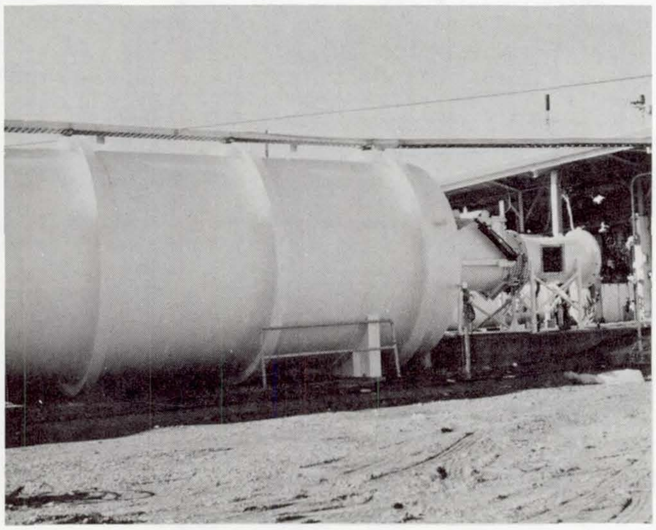


FIGURE 9. - AEROJET ALTITUDE TEST FACILITY.

ORIGINAL PAGE IS
OF POOR QUALITY

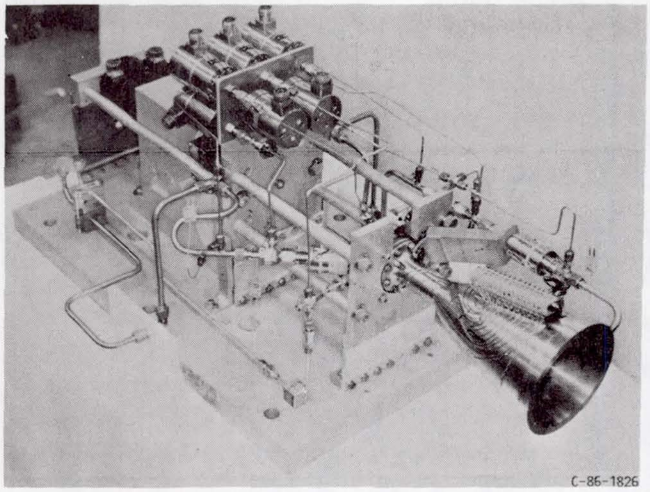


FIGURE 10. - AEROJET THRUSTER MOUNTED TO TEST STAND.

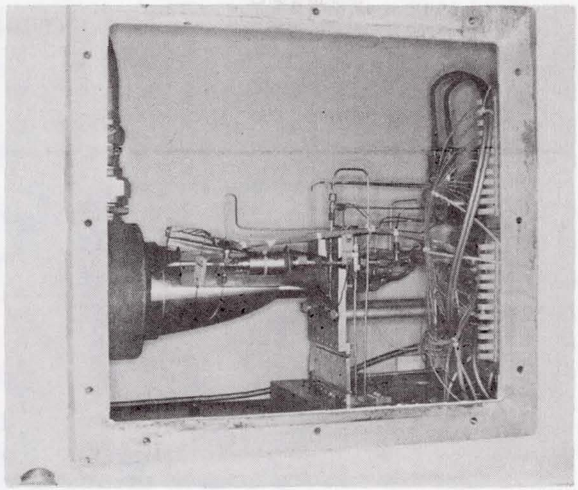


FIGURE 11.- THRUSTER ON STAND MOUNTED IN TEST CELL.

Propellant supply lines are connected to the test cell from a standard GH_2 trailer and a 50 ft^3, 6000 psi gaseous oxygen supply facility.

The assembly is instrumented to measure and record thrust, propellant flow rates, inlet pressures and temperatures, coolant bulk temperature rise, combustion pressure, igniter cold-flow pressures, and chamber backside wall temperatures. Test data are recorded with both digital and analog devices.

EXPERIMENTAL RESULTS AND DISCUSSION

An extensive test program has been run, accumulating in excess of 500 000 lb_f-sec. of impulse. The Space Station thruster has been successfully fired over a range of mixture ratios from 2.2 to 8.1, which far exceeds the original design range of 3.0 to 5.0. Total impulse accumulated at various mixture ratios is summarized in Table II. As shown, most of the thruster impulse (432 000 lb_f-sec) has been obtained at mixture ratios from 7.0 to 8.0, with the longest firing duration being 2200 sec at a 7.5 mixture ratio.

TABLE II. - SPACE STATION THRUSTER TEST SUMMARY

MIXTURE RATIO, O/F	TOTAL DURATION, SEC	TOTAL IMPULSE, LBF-SEC
2	60	1 302
3	240	5 107
4	3 735	89 526
5	224	5 526
6	221	4 728
7	17 563	428 997
8	155	3 221
TOTAL	22 198	538 457

CD-86-21275

The Space Station thrust chamber shows absolutely no sign of any degradation from the testing.

Thermal data agree very well with prediction for the thruster design point, indicating that the thermal model is satisfactory. Measured and predicted backside wall temperature profiles are compared at r = 4.0 in Figs. 12 and 13 for 60 and 75 percent fuel film cooling, respectively. An excellent correlation exists between predicted and measured values for the diverging section of the chamber. For the converging and cylindrical sections, it appears that axial conduction averages the highs and lows predicted by the one-dimensional HOCOOL model. In Figs. 12 and 13, the maximum measured backside temperature occurring at approximately 2.4 in. forward of the throat (for this mixture ratio) is within a few percent of the average predicted chamber values. Likewise, measured coolant bulk temperature increases are within 10 percent of predicted values.

Additional thermal data are given in Figs. 14 and 15. Figure 14 shows maximum backside temperature variations with percent fuel film cooling. Maximum backside temperatures decrease linearly with increasing film cooling and appear to be much more sensitive to mixture ratio. Figure 15 supports the conclusion of a stronger dependence on mixture ratio than on film cooling. The high mixture ratio (7 to 8) tests are also indicated in Figs. 14 and 15, where it is apparent that the wall temperatures are higher at these off-design operating points.

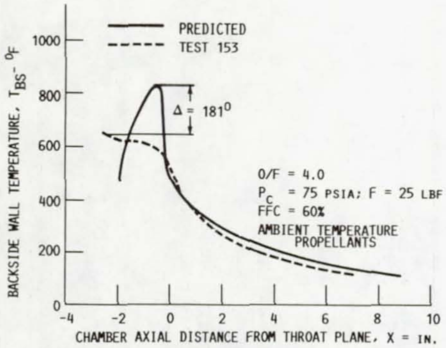


FIGURE 12.- BACKSIDE TEMPERATURE PROFILE WITH 60 PERCENT FFC.

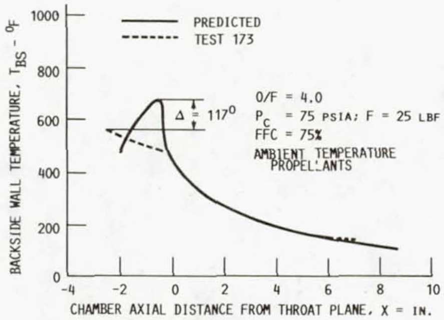


FIGURE 13.- BACKSIDE TEMPERATURE PROFILE WITH 75 PERCENT FFC.

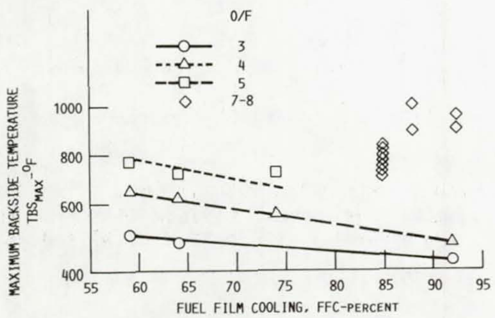


FIGURE 14.- MAXIMUM BACKSIDE TEMPERATURE VARIATION WITH FFC.

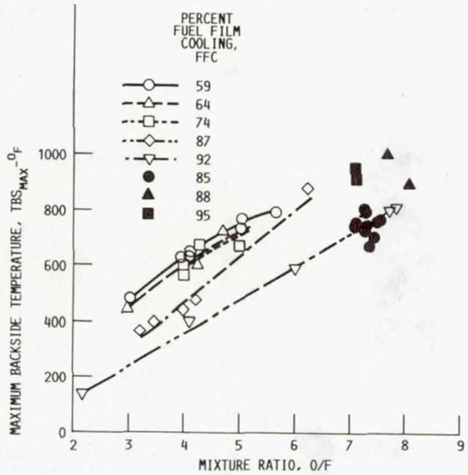


FIGURE 15.- MAXIMUM BACKSIDE TEMPERATURE VARIATION WITH MIXTURE RATIO.

The igniter body, oxidizer valve body and thruster mounting plate temperatures were also monitored during testing, never exceeding values of 200, 75, and 250 °F, respectively. These values were maintained regardless of operating point or test duration, even for the 2200 sec test previously mentioned. Such low temperatures assure minimum heat rejection to the vehicle.

Performance varies widely with film cooling and mixture ratio, as indicated in Figs. 16 and 17. Predicted and measured values for performance do not agree as well as for temperature, although the

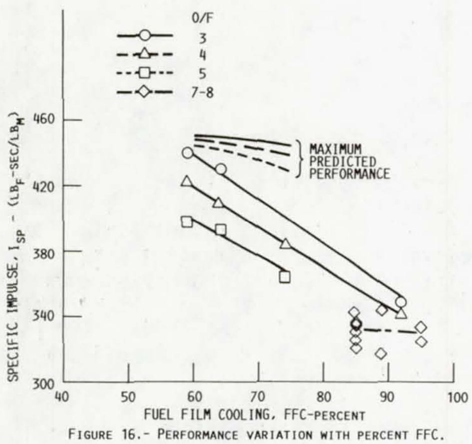


FIGURE 16.- PERFORMANCE VARIATION WITH PERCENT FFC.

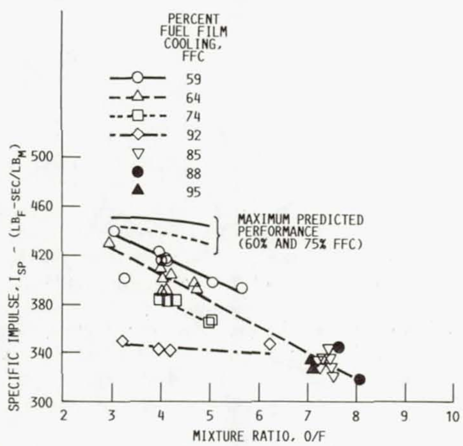


FIGURE 17.- PERFORMANCE VARIATION WITH MIXTURE RATIO.

trends are predicted correctly. It appears that the larger the proportion of hydrogen flowing through the injector, the better the prediction. This condition is attributed to a momentum ratio effect and is influenced by injector hydraulics. The ratio of the hydrogen momentum between the main injector and the fuel film cooling is selected to maximize performance at the design condition of r = 4.0. At the off-design condition of r = 7.0 to 8.0 the hydrogen injector momentum decreases resulting in a loss of performance. This degradation can be recovered by optimizing the injector hydraulics for the higher mixture ratios.

BELL AEROSPACE THRUSTER

Turning to the Bell Aerospace Thruster, the same design requirements were established:

TABLE III. - THRUSTER DESIGN REQUIREMENTS

PROPELLANTS	GO_2/GH_2
MIXTURE RATIO, O/F	4.0±1.0
SPECIFIC IMPULSE	≥400 LBF-SEC/LBM
FUEL INLET TEMPERATURE	≥200 °R
OXIDIZER INLET TEMPERATURE	≥300 °R
TOTAL IMPULSE	$2x10^6$-LBF SEC

CD-86-21282

In this case, a thrust level of 50 lb_f was established for demonstrating their concept.

THE REVERSE FLOW THRUSTER

The reverse flow concept is based on an unconventional use of gas vortex mixing to create a simplified combustor for use with GH_2 and GO_2 propellants. The reverse flow pattern is created when hydrogen is injected as a sheet at a station in the nozzle convergent section, flows toward the front of the spherical combustor where the flow is reversed, and mixes with a vortexing stream of oxidizer. The concept thus combines the reverse flow principle of fuel injection with vortex oxidizer gas injection, forming large chamber mixing vortices and an exposed cooling zone along the chamber wall (Fig. 18).

Experiments with this type of combustor have been conducted since 1958, initially at the Air University, Institute of Technology, Wright-Patterson Air Force Base (Ref. 11) and later at Bell Aerospace Textron. Later in the decade, initial interest in hydrogen and oxygen for the space shuttle spurred a number of developments with the most refinement of the technology displayed at the 1500-lb_f thrust level (Refs. 12 to 15). The 1500-lb_f thruster demonstrated the technology for a qualifiable chamber prior to NASA's decision to eliminate GO_2 and GH_2 as Shuttle propellants. In the absence of identifiable requirements, interest in both gaseous propellant injection and the reverse flow concept lay dormant for more than a decade until recently revived for the Space Station and related applications.

The technology developed for the 1500-lb_f thrust engine was to be translated directly to the smaller size with the objective of reducing the program risk, and to minimize the time and cost of demonstrating the technology maturity. A total impulse of 500 000 lb_f-sec was established as a viable level for demonstration purposes.

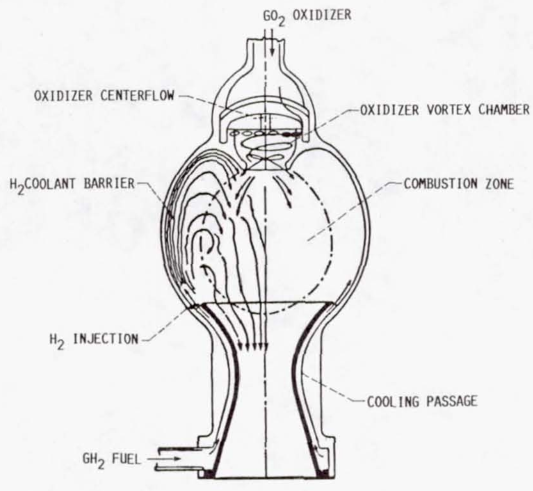


FIGURE 18.- THE REVERSE FLOW THRUSTER.

Bell Aerospace TEXTRON

The reverse flow thruster designed for this application is shown in Fig. 19. A heavyweight, boltup configuration was chosen to facilitate hardware testing and modification. The basic components of this thruster are the spherical chamber (combustor), the vortex oxidizer swirl cup, the nozzle (including the regen-cooled throat and the fuel inlet) and the nozzle extension. The other components are the spark plug igniters (the exciter and lead are not shown) with auxiliary oxidizer cooling and the propellant valves. Photographs of the test hardware in Fig. 20 show both the components and the chamber assembly.

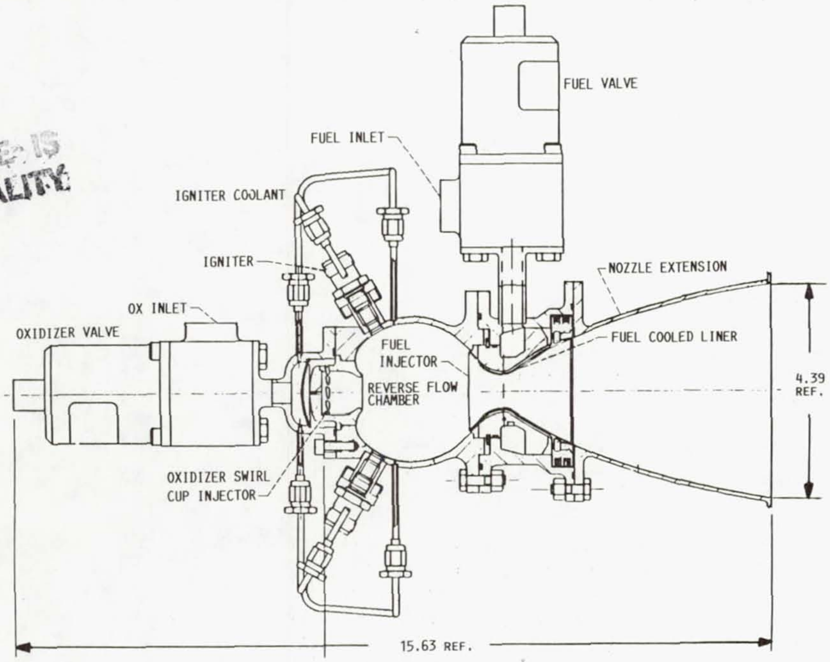
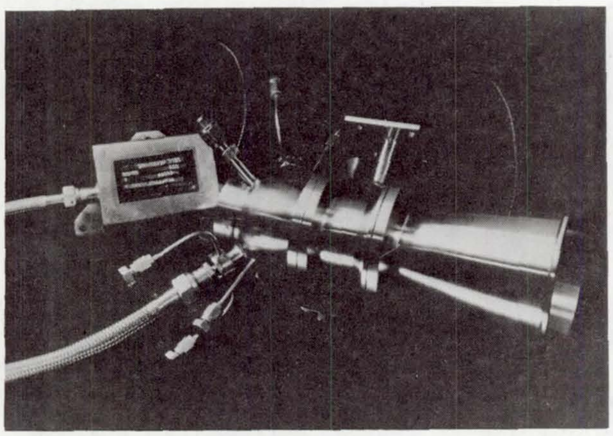
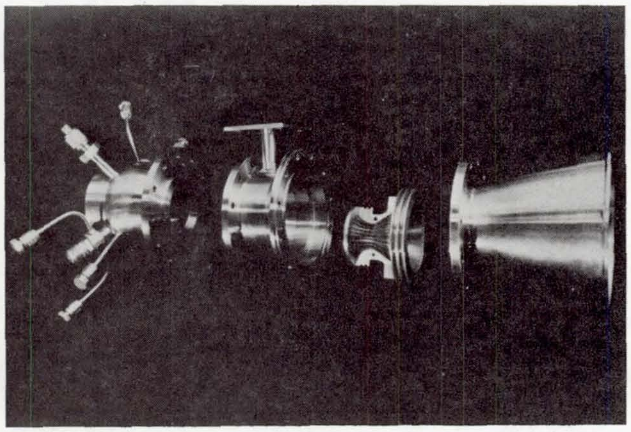


FIGURE 19.- BELL MODEL 8911 THRUST CHAMBER.



(A) THE 8911 THRUSTER. (ASSEMBLED).

FIGURE 20.



(B) THE 8911 THRUSTER. (EXPLODED VIEW).

FIGURE 20. - CONCLUDED.

The fuel inlet and nozzle design is shown in Fig. 21. The propellant enters the nozzle at midsection and is routed aft to enter both the divergent nozzle film coolant manifold and the nozzle regeneratively-cooled passages. H_2 flow is through these cooling passages and out the fuel injection orifices, as indicated in Fig. 21 and 22. The fuel then passes openly along the chamber wall until turned into the oxidizer stream at the head of the chamber.

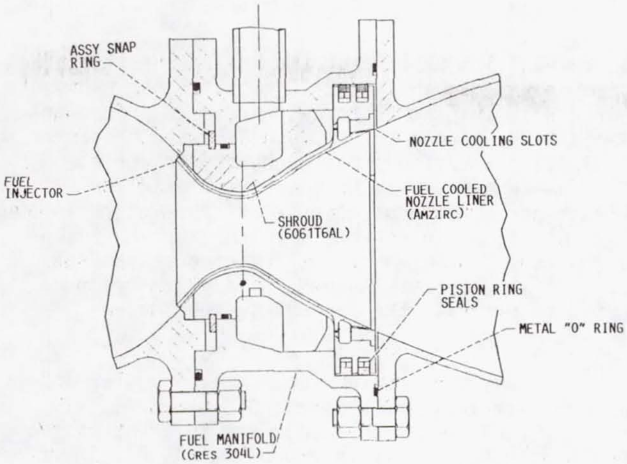


FIGURE 21.- MODEL 8911 REGEN-COOLED NOZZLE.

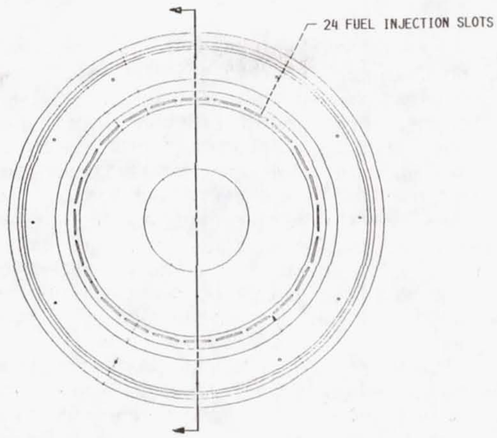


FIGURE 22.- MODEL 8911 FUEL INJECTOR.

The oxidizer flows into the chamber from the valve to the inlet of the vortex cup, through a distribution baffle, and then enters the vortex cup through the swirl orifices and the centerflow orifice. A small amount of oxidizer is drawn from the vortex cup inlet as a spark plug coolant and auxiliary ignition propellant (1/2 percent each igniter).

The construction materials used for this thruster reflect the objective of incorporating readily available materials throughout. The 50-lb_f thruster has a Type 304 stainless steel combustion chamber, oxidizer injector and nozzle holder. The throat section (nozzle liner) is fabricated from Amzirc copper and the nozzle shroud (coolant passage closeout) is a wrap-around two-piece Type 6061 aluminum part. The thruster nozzle extension was fabricated from Hastelloy X.

The thruster design parameters are listed in Table IV.

TABLE IV - MODEL 8911 THRUSTER DESIGN PARAMETERS

THRUST	50 LBF
P_C	75 PSIA
ε	40:1
DIVERGENT NOZZLE COOLANT	6% OF THE FUEL
OXIDIZER COOLANT FOR SPARK PLUGS	1/2% EACH
% BELL (NOZZLE)	80%
CHAMBER L^*	30 IN.
IGNITION	~ 60 SPARKS/SEC
	~ 70 MINIJOULES
TYPE IGNITION	CAPACITIVE DISCHARGE
SPARK PLUG	CHAMPION FHE 297-1
VOLVE	WRIGHT PN 12350

CD-86-21270

One further design criterion was used, which included all the technology developed for the 1500-lb_f chamber. The review of this prior program data indicated that a 1600 °F wall temperature in the chamber could be maintained for the 50 lb_f design, assuming no adverse scale factor was encountered and that a fuel injection velocity of Mach 0.5 was incorporated. As many of the 1500-lb_f engine features as possible were retained to create a mini-50-lb_f version of the engine. The 1500-lb_f engine features were primarily used to minimize the risk in obtaining a reasonable performance (better than 400 sec I_{sp}).

THERMAL ANALYSIS

The challenge facing the thermal designer was analyzing the large vortex which produces the combustion, and the outer fuel "film" which protects the wall from this highly turbulent zone. The heat transfer in this region is extremely difficult to model and even more difficult to document adequately when attempting to verify any model developed. As a consequence, an "experience factor" was used along with the basic assumption that the 50-lb_f thruster would operate at the same wall temperatures as the 1500-lb_f thruster.

This assumption proved to be inaccurate during the initial testing of the 50-lb_f thruster and led to the immediate recognition that the "scale" factor was probably tied directly to the hydrogen film thickness and that a more complete model was needed if this combustor was to be described analytically. Development of such a model was considered to be beyond the resources of the present program, so a test program of methodical oxidizer injection variations was conducted to achieve chamber operation in the desired temperature range. The technique was to adjust the oxidizer centerflow to allow a decreased vortex combustion ratio and consequent combustion temperature in the region where the oxidizer vortex impinges on the cooling film. In effect, a third zone was introduced where the three zones are: (1) outer H_2 reverse flow film, (2) the vortex combustion area consisting of the fuel and the oxidizer vortex flow, and (3) the central zone of oxidizer-rich injection. This general combustor model was then used to evaluate the various changes made, including the rather extreme case of operation at $r = 8$.

The balance of the thrust chamber yielded to analysis by more conventional heat transfer models. The regeneratively cooled nozzle was examined by methods attribute to Eckert and Drake (Ref. 10) resulting in the wall temperature prediction shown in Fig. 23. A thermocouple was inserted in the test hardware at approximately the maximum predicted temperature location approximately 0.3 in. forward of the throat and the nozzle metal temperature recorded at that station.

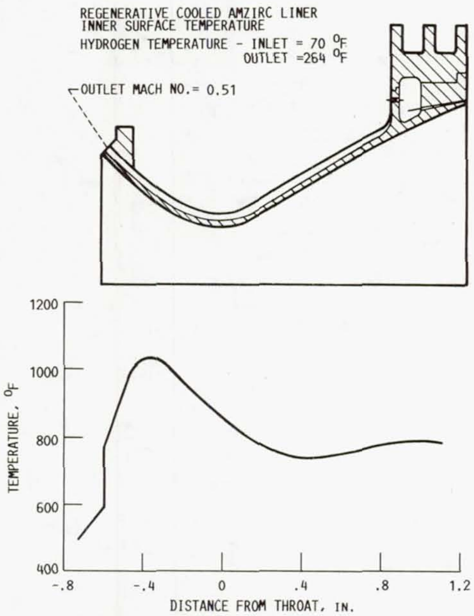


FIGURE 23.- REGENERATIVE-COOLED AMZIRC LINER INNER SURFACE TEMPERATURE.

The nozzle extension was also examined with operating conditions similar to the 1500-lb_f engine. This extension used a small amount of fuel film cooling to reduce the temperature at the aft flange seal area and on the extension itself. The results are presented parametrically in Fig. 24. As a result of this analysis, a design which incorporated 6 percent of the fuel as nozzle cooling was selected to keep the expected nozzle extension temperature to less than 2000 °F.

PERFORMANCE ANALYSIS

Performance for this thrust chamber was estimated via the Standard JANNAF One-Dimensional Equilibrium Methods, assuming that the combustion efficiency would be approximately 96 percent. The resultant parameters are as shown in Fig. 25. No attempt was made to predict the effects of propellant interactions in the mixing region other than to assume the 96 percent combustion efficiency. This assumption proved to be valid in subsequent testing during which the efficiency was measured, as the initial test configuration exceeded the predicted I_{sp} number by approximately 1 percent.

TEST OBJECTIVE

The primary objective of the original Bell Aerospace test program was to demonstrate 500 000 lb-sec total impulse at a mixture ratio of 4.0. As outlined for the Aerojet program, high performance was not considered a prime objective, but 400 lb_f-sec/lbm I_{sp} was defined as a minimum goal.

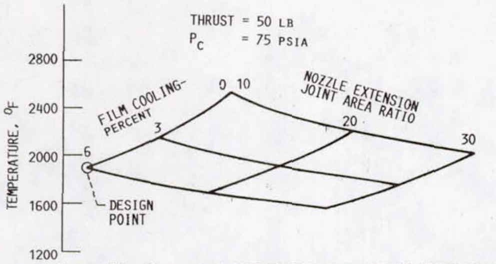


FIGURE 24.- FILM-COOLED NOZZLE EXTENSION MAXIMUM TEMPERATURE.

ENGINE PARAMETERS	
THRUST	50 LB_F
CHAMBER PRESSURE	75 PSIA
NOZZLE AREA RATIO	40:1
LOSSES	
DIVERGENCE LOSS	1.2%
KINETIC LOSS	1.2%
BOUNDARY LAYER LOSS (DRAG + DISPLACEMENT)	2.9%
DUMP-COOLANT LOSS	0.6%
ENERGY RELEASE LOSS	4.0%
	9.9%
THEORETICAL I_{SP}, LB_F-SEC/LB_M	473.3
EXPECTED I_{SP}, LB_F-SEC/LB_M	426.8

FIGURE 25.- MODEL 8911 PERFORMANCE PREDICTION.

One of the design/test program objectives was to understand and define a method of managing wall temperatures to produce an indefinite chamber life using standard construction materials. As the program progressed, the major emphasis was placed on achieving the limited wall temperatures required for long life. Consistent with the Space Station interest in operating the APS on GO_2/GH_2 at mixture ratios up to 8.0, the Bell Aerospace program was also modified to provide hardware and test results to meet this objective.

The thruster design parameters shown in Table IV were established for the mixture ratio 4 design, and required re-evaluation to permit operation at r = 8. The difference in the design at r = 8 was an increase in chamber pressure and thrust to allow the same fuel flow in the nozzle for cooling purposes. During testing the chamber pressure and thrust did increase to values of approximately 96 psia and 72 lb_f, respectively, with the increase in oxidizer flow. The modified Bell test program had the same objective of demonstrating 200 000 lb_f-sec total impulse at r = 8. This was accomplished during a one and one-half month effort, which included design, fabrication, and testing of the modified thruster.

TEST CELL AND OPERATION

All fire-testing of the Space Station Auxiliary Thruster was conducted in the Bell altitude facility A-2. The test cell used has a nominal altitude capability of 120 000 feet with a duration capability in excess of 1000 sec. The Bell altitude facility is operated by a dedicated steam generation system tied in with a factory power plant, providing low-cost operations of almost unlimited duration. The general arrangement of the facility is shown in Fig. 26.

PROCEDURE

Operation of the thruster is accomplished by a timer panel. The start and shutdown sequence of events to the igniter and valve systems are preplanned and operate in an automatic sequence. For these tests, the fuel valve was sequenced to open one millisecond ahead of the oxidizer valve, although no confirmation measurements were attempted to ascertain the propellant chamber entry sequence.

Ignition was accomplished with the use of the exciter, having an approximate frequency of 60 sparks per sec, operating a spark plug installed in the combustor wall. The start sequence was programmed by the automatic operating panel with the ignition system started and then valves opened. Examination of the start traces showed positive and immediate starts as soon as positive oxidizer pressure was identified.

INSTRUMENTATION

Normal performance measurement parameters, including thrust, chamber pressure and propellant flow rates, were measured for all tests. Flow rates were measured using temperatures and sonic orifices. Cell instrumentation includes a thrust line load cell arrangement where the thrust chamber is mounted vertically and fired in a downward direction. Three stabilizing webs were used on the chamber mount so that thrust alignment was maintained without influencing the thrust measurement.

Temperatures were measured with thermocouples placed at various positions on the chamber. Since there has been very little precedent for failure criteria for this type of thrust chamber, thermocouples were placed at various positions on the chamber to establish criteria for the formulation of more complete heat monitoring arrangement. Thermocouples were placed on the nozzle extension, at the nozzle flange, on one of the lands in the copper nozzle liner, in a coolant passage and on the combustion chamber at a variety of positions. The initial test results showed that the high temperatures were at the midpoint of the spherical chamber. To monitor subsequent results or design changes, four thermocouples were continuously monitored at the chamber midpoint. These chamber temperatures were used to determine the magnitude of temperature changes as well as circumferential temperature distribution.

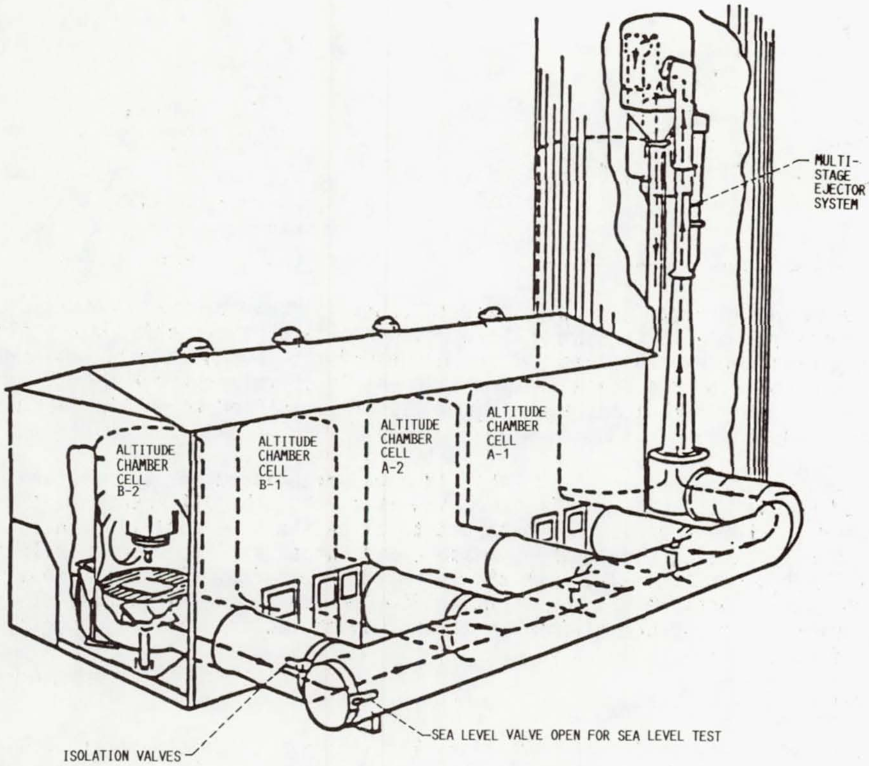


FIGURE 26.- BELL AEROSPACE ALTITUDE TEST COMPLEX.

Bell Aerospace TEXTRON

TEST LIMITATIONS

Initial testing indicated that almost any sequence of tests could be readily accomplished within the test cell, until long durations were attempted. The initial long duration tests showed that the ejector system was not adequate for complete purging of exhaust gases, and recirculation back into the cell resulted. This recirculation resulted in some overheating which eventually affected instrumentation. After an initial 1000-sec test, a supplementary exhaust duct with a duration limit of 300 sec was installed and used for all subsequent long duration tests. Even with this modification, testing of the mixture ratio 8 hardware resulted in cumulative heating in the duct when rapid repeats of long duration tests were made. This heating did not appear to affect the thruster operation in any way but it did result in some questionable test measurements.

EXPERIMENTAL RESULTS AND DISCUSSION

All of the initial testing was conducted with the chamber material described, although a Hastelloy X chamber was tested when the program objective mixture ratio was changed to 8. The rest of the chamber used the same materials; in fact, the tests used the same nozzle throughout. Thruster operation with stable wall temperatures was demonstrated in a 1000-sec duration firing with the hardware in excellent condition post-test.

The results of the initial testing were quite satisfying in that the performance met or exceeded the values originally predicted. This achievement was particularly satisfying due to the concern that the design might not be correct when reducing the features of the 1500-lb_f thruster.

While measured performance was as predicted, chamber heating was substantially higher than anticipated and high enough to compromise operation with the "uncooled" stainless steel hardware. The 1500-lb_f chamber had produced a maximum wall temperature of approximately 1600 °F and that value was used as a basis for the 50-lb_f thrust design. Initial test results indicated an equilibrium temperature of 1995 °F was more probable and approximately 2400 °F was anticipated, operating at a mixture ratio of 5. The initial test results compared to predicted are as shown in Table V.

TABLE V - INITIAL THRUSTER TEST RESULTS

	PROGRAM REQUIREMENT	R = 4 PREDICTED	TEST RESULTS	R = 5 EXPECTED
F, LB	50	50	48.65	-
P_C, PSIA	75	75	70.8	-
I_{SP}, SEC	400	426.8	430.3	-
METAL REGEN. TEMP. MAX. °F	-	1030	1018	-
CHAMBER TEMPERATURE, °F	-	-	1584*	-
PREDICTED CHAMBER TEMPERATURE AT STABILIZATION, °F	-	1600	1995	2400**

* NOT TESTED TO STABILIZATION

** ESTIMATED ON BASIS OF R = 4 TEST

CD-86-21269

One obvious result of the initial testing was the need to reduce the chamber wall temperature without losing any more performance than necessary. The method selected was to modify the orifices in the swirl cup and the centerflow in the oxidizer cup. Changing the swirl cup orifices changes the oxidizer swirl emission angle from the oxidizer cup according to the relation:

$$\alpha = f \frac{D_1 D_2}{A_s} = f(K)$$

where α is the cone angle, (D_1) the swirl cup diameter, (D_2) the swirl cup exit diameter and A_s the tangential flow injection area (Fig. 27). It was assumed that a decrease in cone angle would decrease the degree of interaction between the vortex and fuel film, hence increasing its integrity on the wall. Since α decreases as the value of A_s increases, the first modification to the oxidizer vortex cup was to increase the number of vortex orifices. On this first modification (Fig. 27, interim ox cup), no change was made to the centerflow orifice.

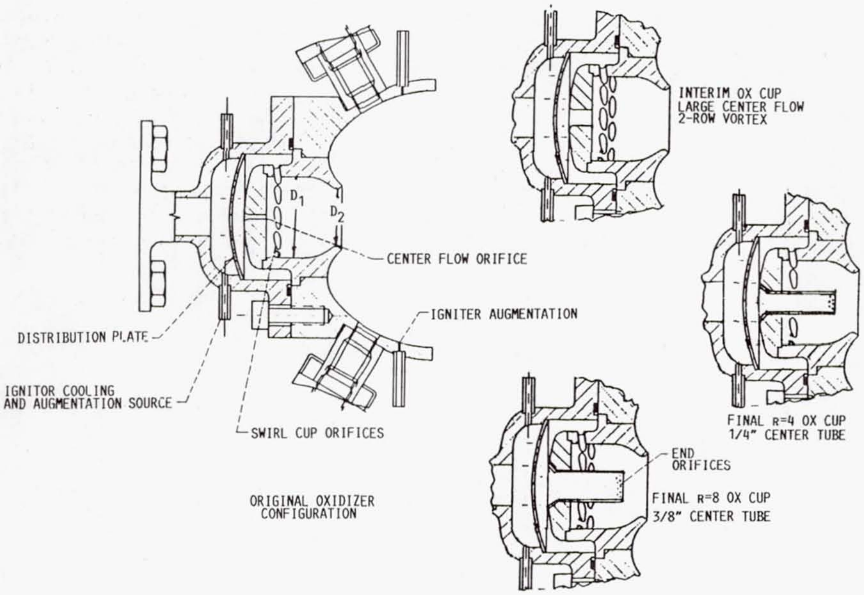


FIGURE 27.- MODEL 8911 OXIDIZER VORTEX CUP CONFIGURATIONS.

Bell Aerospace TEXTRON

The improvement in wall temperature from this first modification was modest at best with only a few degrees improvement in wall temperature. Further changes in A_s would have reduced the pressure drop in the oxidizer cup below the limits for stability. As a consequence, the next variable investigated was the centerflow.

Three increases in size to the centerflow orifice were made with significant results. The first change was to increase the centerflow to 10 percent of the oxidizer with a resultant performance and wall temperature decrease. This success led to two more immediate modifications, including a 25 percent centerflow mod and further to a 33 percent centerflow. The 25 percent centerflow showed a significant improvement but the 33 percent centerflow change did not result in the expected improvement (Fig. 28).

MODIFICATION	CENTERFLOW, %	NO. OF VORTEX ORIFICES	I_{SP}-SEC	WALL TEMPERATURE, ^0F	TIME TO TEMPERATURE, SEC
1	3.46	10	430.3	1584	10
2	2.53	20 ORIG.	433.8	1648	11
3	10.3	20	436.9	1629	12
4	25.1	20	417.3	1667	30
5	33.0	20	413.1	1648	30
6	43.9	10	410.2	1661	30
7	41.4	10 0.8" TUBE	398.3	1491	30
8	41.4	10 0.55" TUBE	397.9	1572.3	30
9	41.4	10 0.6" TUBE	399.5	1588	1000-SEC RUN
10	47.4	20 0.6" TUBE	376	1277	STABILIZATION

FIGURE 28. TEST RESULTS: OXIDIZER VORTEX CUP CONFIGURATION MODIFICATIONS.

The temperature results of the 33 percent centerflow tests showed that the temperature control effects of increasing the centerflow were limited and ineffective over 25 percent. Apparently, the temperature control was negated by the increased interactions between the oxidizer flow and centerflow. The oxidizer from the centerflow orifice may have been expanding into the vortex at the minimum section (D_2), thus reducing the separation of the two flow regions.

To provide the required greater oxidizer centerflow without interactions with the vortex flow, a 1/4-in. tube with a 0.020-in. wall was installed to contain and direct the centerflow. The first tube evaluated was 0.8 in. long and extended to the minimum section of the oxidizer vortex cup. This modification was tested and the results were significant in that performance and temperatures were close to anticipated values and the temperature "distribution" of the wall was improved. This tube burned back approximately 0.13 in. during the tests, indicating a hot gas circulation in the region and that the tube length could not be maintained. Two additional tube lengths were investigated, both shorter; the first with no end orifices, and the second a 0.6-in. long tube with twenty-four 0.020-in. diameter holes to protect the end from hot gas-induced deterioration. This second tube version (Fig. 27 final r = 4 ox cup) was so successful that it was selected as the configuration which was installed and operated for 1000 sec continuously. The performance (I_{sp}) related to the changes made in the oxidizer cup can be seen in Fig. 29. Also included in this chart are the data taken using the mixture ratio 8 hardware when tested at r = 4 (Fig. 27 Final r = 8 ox cup).

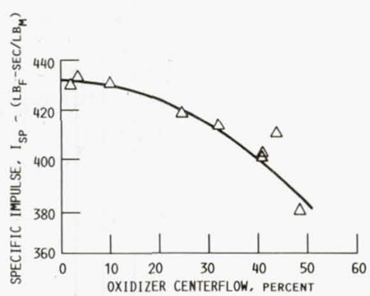


FIGURE 29.- SPECIFIC IMPULSE VERSUS OXIDIZER CENTERFLOW.

The final version of the mixture ratio 4 test hardware is shown firing in the test cell in Fig. 30. The data from this final test configuration were recorded as shown in Table VI:

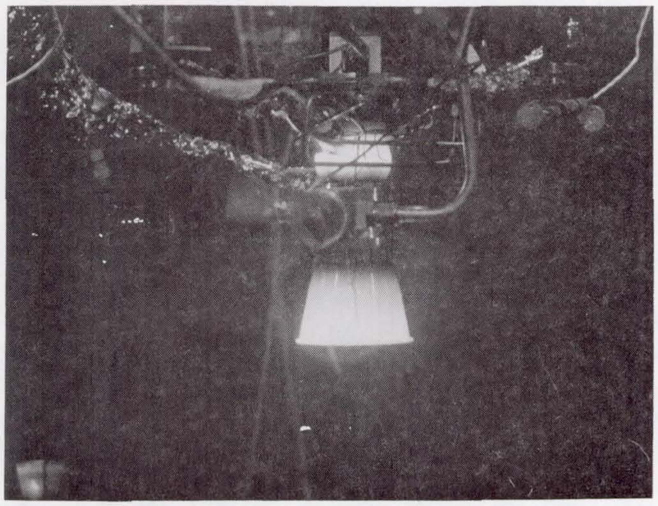


FIGURE 30. - MODEL 8911 THRUSTER FIRING IN TEST CELL A-2.

TABLE VI. - TEST DATA FOR THRUSTER OPERATING AT R = 4

F	51.48 LBF
P_C	74.9 PSIA
I_{SP}	397.2 LBF-SEC/LBM
METAL REGENERATIVE TEMPERATURE MAXIMUM	885 °F
CHAMBER TEMPERATURE	1581 °F

CD-86-21280

The experimental program was then redirected to the demonstration of a thruster at a mixture ratio of 8. New hardware was fabricated: the oxidizer cup, cap and new chamber. The original nozzle was retained with the ground rule that the fuel flow would remain constant to cool the regen. nozzle. The proper mixture ratio was obtained by increasing the oxidizer flow (and total weight flow).

Testing of the mixture ratio 8 hardware was conducted within an extremely short time span with the entire program priority devoted to the demonstration of 200 000 lb/sec impulse at a mixture ratio of 8. This objective was accomplished in several days of testing with the hardware remaining intact through the repetitive 300-sec tests.

In all, ten 300-sec tests were conducted with several shorter tests as preliminary evaluation items. The performance for these tests remained constant as indicated in Table VII.

TABLE VII - TEST DATA FOR THRUSTER OPERATING AT R = 8
(PERFORMANCE DATA TAKEN AT 29.4 SEC)

RUN NO.	R	F (LB)	I_{SP} (LBF-SEC/LBM)	WALL TEMPERATURE, T.C. #11 (°F) AT 300 SEC	TEST DURATION (SEC)
4368	7.924	76.58	346.0	1397	300
4369	7.920	75.24	340.1	1498	300
4370	7.912	76.93	347.7	1499	300
4371	8.024	76.87	339.7	1499	300
4372	8.074	78.91	346.5	1458	300
4373	7.972	76.81	340.3	1481	300
4374	7.940	76.26	340.2	1472	300
4375	7.971	78.14	345.3	1500	300
4376	7.957	76.87	342.2	1496	300
4377	8.034	78.15	346.7	1511	300

CD-86-21281

Once started, this test series was conducted without incident. In fact, operation of the thruster could almost be called casual in that it could be run, shut down or restarted at will with totally uncomplicated procedures.

Again, the thruster outlasted the test cell instrumentation; a new exhaust duct was installed in the test cell between the 1000-sec mixture ratio 4 test and the installation of the new r = 8 hardware. While the new test duct helped immensely, some blowback still existed and heating in the cell occurred. This test cell heating had no discernible effect on the operation of the hardware.

After completion of the long duration testing, a second series of tests were made to operate over the projected possible operating range of r = 3 to r = 8. The results of this testing are shown in Fig. 31 and 32.

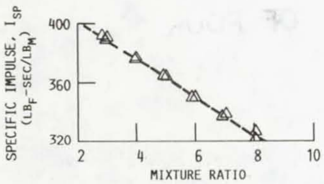


FIGURE 31.- PERFORMANCE VS MIXTURE RATIO.

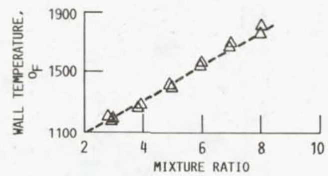


FIGURE 32.- WALL TEMPERATURE VS MIXTURE RATIO.

In essence, this data graphically illustrates the problem in design for the high mixture ratios where a loss of approximately 50 sec I_{sp} is accompanied by a 500 °F increase in wall temperature going from a mixture ratio of 4 to a mixture ratio of 8. On the other hand, an extreme range of capability is illustrated for the Bell thruster in operating over this entire range.

The extensive life test program of the Bell reverse flow concept thruster has successfully demonstrated the ability to operate over a wide range of mixture ratios, with long-life expectancy. The total impulse accumulated at various mixture ratios is summarized in Table VIII.

TABLE VIII. - BELL MODEL 8911 THRUSTER
TEST RESULTS

MIXTURE RATIO, O/F	TOTAL DURATION, SEC	TOTAL IMPULSE, LBF-SEC
3	275	13 470
4	1619	79 637
5	124	6 123
6	83	4 367
7	65	3 449
8	3116	225 607
TOTAL	5282	332 653

CD-86-21276

Performance of the Space Station thruster varies widely with oxidizer centerflow, as shown in Fig. 29, and with mixture ratio, as indicated in Fig. 31. Predicted and measured values for performance are shown in Fig. 33, which reflects the loss in performance with increasing oxidizer centerflow at higher mixture ratios, as the operation gets further from the design conditions. This loss can be recovered by optimizing the flow interaction effects for the higher mixture ratio.

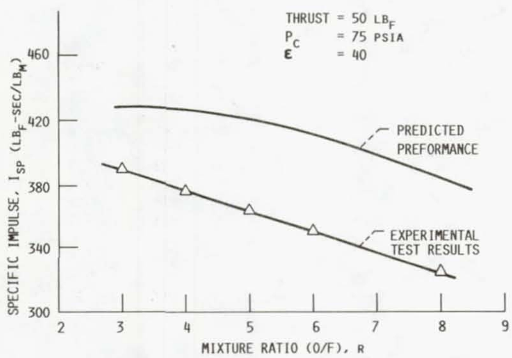


FIGURE 33.- COMPARISON OF PREDICTED AND EXPERIMENTAL PERFORMANCE.

SUMMARY OF RESULTS

This technology demonstration program has provided a number of encouraging results. Based on previously proven concepts, thruster designs have been identified and demonstrated to provide extended firing times using GO_2 and GH_2 as propellants. These thruster designs combined good performance with low wall temperatures - and thus long life - over a wide range of mixture ratios.

Existing analytical models for predicting life, performance, and wall temperature have been utilized and can be further refined with the test data obtained. Also, an additional combustion model has been suggested for the unique reverse flow thruster concept.

The capability of controlling wall temperatures by varying the fuel film cooling for the Aerojet thruster, and the central oxidizer core flow for the Bell thruster, was demonstrated with substantial reductions in temperatures through systematic increases in the respective flows.

Simple, proven ignition systems have been utilized, and appear to be more than adequate for Space Station Auxiliary propulsion.

Finally, the thruster designs were fabricated from "non-strategic" (i.e., readily available, low cost) materials.

CONCLUSIONS

The technology for long-life GO_2/Gh_2 thrusters for Space Station Auxiliary propulsion has been successfully demonstrated.

1. Long firing times, consistent with Space Station goals have been achieved.

2. The thrusters are capable of operating over a wide rage of mixture ratios with acceptable performance.

3. The ability to trade thruster life and performance by making simple engineering modifications has been demonstrated.

Improvements in the analytical models for these thrusters, through continuing technology efforts, would undoubtedly allow recovery of some performance losses.

With this key technology in place, the straight-forward development and operational success of GO_2/GH_2 auxiliary propulsion for the Space Station is assured.

REFERENCES

1. Rosenberg, S.D., et al.: Ignition Systems for Space Shuttle Auxiliary Propulsion System. (ALRC-1678-26-F, Aerojet Liquid Rocket Co.; NASA Contract NAS3-14348) NASA CR-72890, 1972.

2. Hydrogen-Oxygen Auxiliary Propulsion for the Space Shuttle. Vol. 1-High Pressure Thrusters, NASA CR-120895, 1973.

3. Blubaugh, A.L.; and Schoenman, L.: Extended Temperature Range ACPS Thruster Investigation. NASA CR-134655, 1974.

4. Berkman, D.K.: Oxygen/Hydrogen Thrusters for Space Station Auxiliary Propulsion Systems. NASA CR-175691, 1984.

5. Dowdy, M.W.; and Appel, M.A.: High-Temperature Gaseous Oxygen/Hydrogen Thrusters for Space Station. 1984 Advances in Aerospace Sciences and Engineering: Structures, Materials, Dynamics and Space Station Propulsion, U. Yuceoglu and R. Hesser, eds., ASME, 1984, pp. 179-183.

6. Ewen, R.L., et al: Combustion Effects on Film Cooling, HOCOOL Users Manual. Aerojet Liquid Rocket Co., NASA Contract NAS3-17813, July 1975.

7. Rousar, D.C.; and Ewen, R.L.: Combustion Effects on Film Cooling. NASA CR-135052, 1977.

8. Nickerson, G.R.; Coats, D.E.; and Bartz, J.L.: Two-Dimensional Kinetic (TDK) Reference Computer-Engineering and Programing Manual. NASA CR-152999, 1973.

9. Weigold, H.D.: ICRPG Turbulent Boundary Layer Nozzle Analysis Computer Program. NASA CR-100653, 1968.

10. Robinson, P.J.; and Rosenthal, S.E.: A Proven 25-lbF H_2/O_2 Thruster for Space Station Auxiliary Propulsion. AIAA Paper 86-1560, June 1986.

11. James, R.N.: An Evaluation of a Reverse Flow Film-Cooled Rocket Engine. Air Force Institute of Technology, Wright-Patterson Air Force Base, 1960.

12. Senneff, J.M: High Pressure Reverse Flow APS Engine. (BELL-8636-950004, Bell Aerospace Co.; NASA Contract NAS3-14353) NASA CR-120881, 1972.

13. Drexhage, M.; and Safeer, N.: Reverse Flow Thrust Chamber Developments. Twelfth JANNAF Liquid Propulsion Meeting, CPIA-PUB-201-Vol-1, 1970, pp. 479-489.

14. Schorr, C.J.; Berman, K.; and Worner, G.: Modeling of High Energy Gaseous Combustors for Performance Prediction. 8th JANNAF Combustion Meeting, CPIA-PUB-220-VOL-1, T.W. Christian, ed., 1971, pp. 639-657.

15. Berman, K.; and Ferger, T.M.: A Note on Compressible Flow Through a Vortex Swirl Cup. J. Appl. Mech., vol. 40, no. 2, June 1973, pp. 619-620.

16. Eckert, E.R.G.; and Drake, R.M., Jr.: Heat and Mass Transfer. 2nd ed., McGraw-Hill, 1959.

1. Report No. NASA TM-88822	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Proven, Long-Life Hydrogen/Oxygen Thrust Chambers for Space Station Propulsion		5. Report Date
		6. Performing Organization Code 481-01-02
7. Author(s) G. Paul Richter and Harold G. Price		8. Performing Organization Report No. E-3171
		10. Work Unit No.
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

Prepared for the 1986 JANNAF Propulsion Meeting, sponsored by the JANNAF Interagency Propulsion Committee, New Orleans, Louisiana, August 26-28, 1986.

16. Abstract

The development of the manned Space Station has necessitated the development of technology related to an onboard auxiliary propulsion system (APS) required to provide for various Space Station attitude control, orbit positioning, and docking maneuvers. A key component of this onboard APS is the thrust chamber design. To develop the required thrust chamber technology to support the Space Station Program, the NASA Lewis Research Center has sponsored development programs under contracts with Aerojet TechSystems Company and with Bell Aerospace Textron Division of Textron, Inc. During the NASA Lewis sponsored program with Aerojet TechSystems, a 25 lb_f hydrogen/oxygen thruster has been developed and proven as a viable candidate to meet the needs of the Space Station Program. Likewise, during the development program with Bell Aerospace, a 50 lb_f hydrogen/oxygen Thrust Chamber has been developed and has demonstrated reliable, long-life expectancy at anticipated Space Station operating conditions. Both these thrust chambers were based on design criteria developed in previous thruster programs and successfully verified in experimental test programs. Extensive thermal analyses and models were used to design the thrusters to achieve total impulse goals of 2×10^6 lb_f-sec. Test data for each thruster will be compared to the analytical predictions for the performance and heat transfer characteristics. Also, the results of thrust chamber life verification tests will be presented.

17. Key Words (Suggested by Author(s)) Space Station auxiliary propulsion; Hydrogen/oxygen; Thrust chamber; Total impulse		18. Distribution Statement Unclassified - unlimited STAR Category 20	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of pages	22. Price*

*For sale by the National Technical Information Service, Springfield, Virginia 22161