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DESIGN, DEVELOPMENT, AND TESTING OF A 1000 POUND (4450 N) THRUST FLOX-PROPANE ABLATIVE ROCKET ENGINE

by Donald A. Peterson, Jerry M. Winter, and Albert J. Pavli

Lewis Research Center
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ABSTRACT

A pressure-fed liquid FLOX-liquid propane rocket engine was designed and test fired at 100 psia (690 kN/m^2) chamber pressure and 1000 pounds (4450 N) thrust. A 137 element oxidant-fuel-oxidant triplet injector produced a characteristic exhaust velocity efficiency of 95.2 percent (96.1 percent corrected) at an O/F of 4.5. The injector was successfully scaled from a 150-pound thrust design. A long run capability was demonstrated by firing an ablative graphite cloth/phenolic thrust chamber for about 200 seconds total time with a throat area change less than ±4 percent.

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SUMMARY

A pressure-fed liquid FLOX-liquid propane rocket engine was designed and test fired at 100 psia (690 kN/m^2) chamber pressure and 1000 pounds (4450 N) thrust. A characteristic exhaust velocity efficiency of 95.2 percent (96.1 corrected) was obtained at an O/F of 4.5. The injector was a 137 element oxidant-fuel-oxidant triplet with the elements mutually perpendicular. The hole diameters selected resulted in an oxidant-to-fuel velocity ratio of 0.67. The design of the injector was successfully scaled from a 1.2-inch (3.05-cm) throat 150-pound (667-N) thrust design which had a characteristic exhaust velocity efficiency of 95.5 percent (97.6 percent corrected) and an oxidant-to-fuel velocity ratio of 0.61.

A long term run capability was demonstrated by firing an ablative graphite cloth/phenolic engine for approximately 200 seconds total time.

INTRODUCTION

Early attempts to demonstrate a successful FLOX-LPG (liquid fluorine and oxygen - liquid petroleum gas) engine design met with some difficulty in achieving desired performance levels (ref. 1). Accordingly, NASA undertook design and test of a 150-pound (667 N) thrust FLOX-propane engine which operated at 100 and 200 psia (690 and 1380 kN/m^2) chamber pressure with a nozzle throat diameter of 1.2 inches (3.05 cm) (ref. 2). Based on earlier experience with earth storable propellants, it was believed that the technique of arranging triplet elements to give mutually perpendicular spray fans could be used to provide improved mixing and high performance with space storable propellants. Using this principle in reference 2, a characteristic exhaust velocity efficiency in excess of 95 percent of shifting equilibrium was obtained over a range of oxidant-to-fuel ratios. An ablative chamber and nozzle of graphite cloth/phenolic material was tested at 100 psia (690 kN/m^2) and at an O/F of 4.5 for 300 seconds continuously with no increase in the throat area.

Due to the initial success experienced in the 1.2-inch (3.05-cm) throat diameter engine, and also because of NASA interest in FLOX-LPG combinations as space storable propellant systems for larger size engines, it was considered desirable to design and test an engine with a 3.0-inch (7.62 cm) throat diameter. This program was initiated primarily to determine if the general design of the injector, chamber, and nozzle material established in the small-scale program would be applicable to larger size engines.

The objectives of the program were to demonstrate stable combustion at a characteristic velocity efficiency level of 95 percent or better and also to demonstrate a long term run capability with the use of ablative materials for the chamber and nozzle.

The design operating conditions were chosen as an oxidant-to-fuel ratio of 4.5 to 1 (theoretical peak specific impulse), chamber pressure of 100 psia (690 kN/m^2) (pressure fed engine), and a throat diameter of 3.0 inches (7.62 cm) for 1000 pounds (4450 N) thrust. Test conditions included an O/F range of 3.5 to 5.0 and an L* variation from 23 to 53 inches (58.4 to 134.5 cm).

APPARATUS

Engine Design

Successful engine design for ablative chamber and nozzle materials requires not only careful choice and design of the ablative, but also requires design of the injector for a propellant distribution which provides high characteristic velocity efficiency and an acceptable environment for the ablative.

Injector. - Figure 1 is a sketch of the injector that was used for the 1000-pound (4450-N) thrust engine. The design was an attempt to draw on the experience of reference 2, which used a successful 37-element oxidant-on-fuel triplet. Experience gained with larger injectors for use with Earth storable propellants was also used in the design (ref. 3). Table I compares the basic design parameters of the 150-pound (667-N) thrust injector of reference 2 to the 1000-pound (4450-N) thrust engine used in this program. Differences between the large and small scale injector design parameters were due primarily to modifications required by size. The combination oxidant manifold and face cooling passages in the large scale design were of such length and diameter that fewer elements per unit face area could be used in the large scale injector. The increase in element spacing led to an increase in propellant flowrate per element which also required that the hole sizes be increased to keep injector pressures at levels consistent with a pressure-fed engine system. As described above, the face coolant passages were necessarily farther apart than in the injector of reference 2. Therefore, it was decided

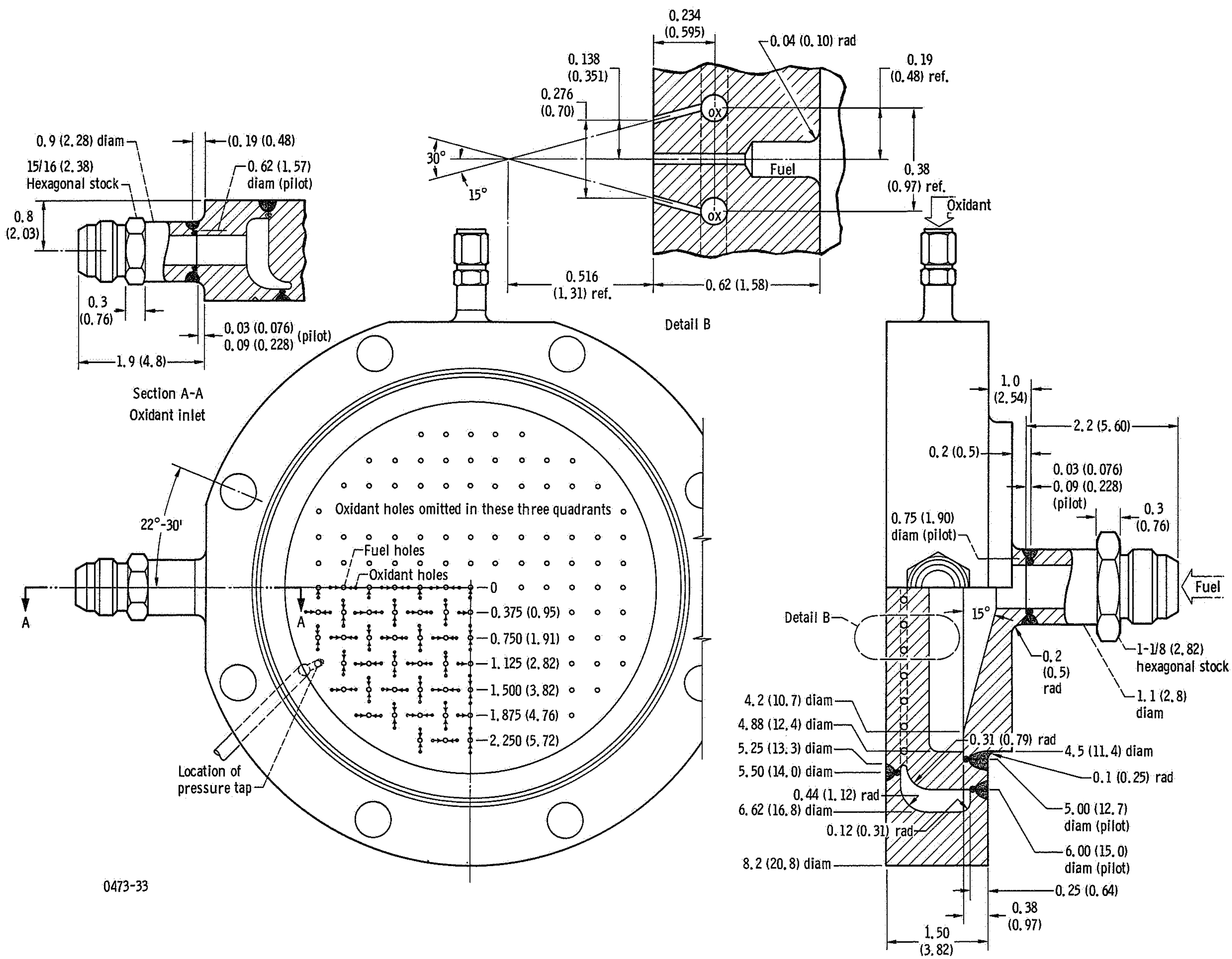


Figure 1. - Injector; 1000-pound (4440-N) force engine; grid pattern; oxidant on fuel triplet. (All linear dimensions are in inches (cm).) Material-6061-T6 aluminum.

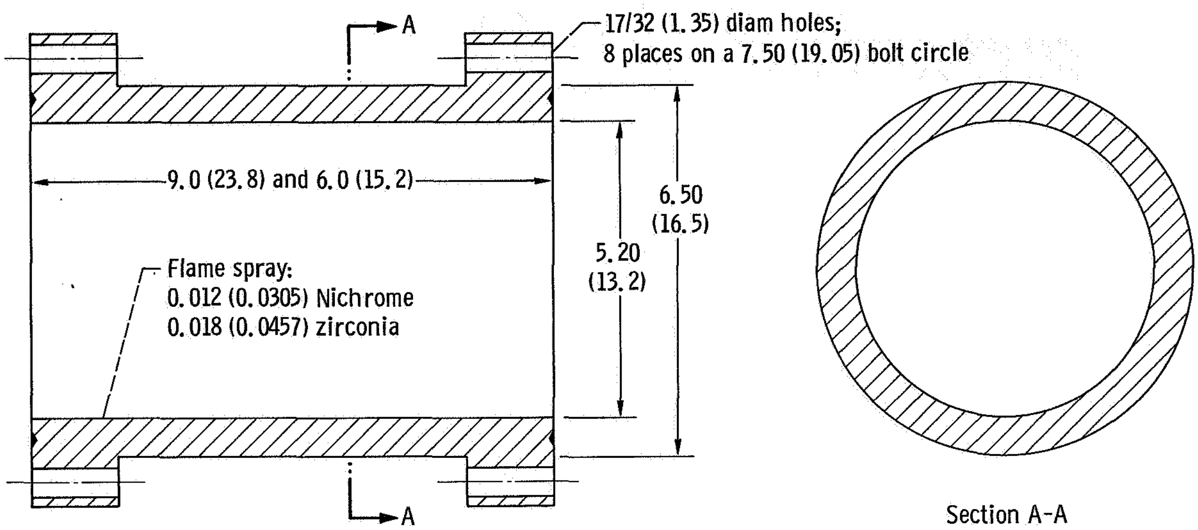
TABLE I. - INJECTOR DESIGN PARAMETER COMPARISON

Parameter	Engine throat diameter, in. (cm)	
	3.0 (7.62)	1.2 (3.05)
Number of elements	137	37
Element spacing, in.; cm	0.375; 0.953	0.250; 0.635
Element density (element/in.^{2} face area)	6.450	10.890
Flow per element, lbm/(sec)(element); kg/(sec)(element)	0.0253; 0.0115	0.015; 0.007
Oxidant diameter, in.; cm	0.020; 0.051	0.018; 0.056
Oxidant injection ΔP (measured), psi; kN/m^2	64; 441	68; 469
Oxidant velocity, ft/sec; m/sec	52; 16	38; 12
Fuel diameter, in.; cm	0.0156; 0.040	0.0135; 0.034
Fuel injection ΔP (measured), psi; kN/m^2	66; 445	37; 255
Fuel velocity, ft/sec; m/sec	78; 24	62; 19
Ratio of oxidant velocity to fuel velocity	0.666	0.613
Included impingement angle, deg	30	47
Impingement distance, in.; cm	0.516; 1.311	0.10; 0.254

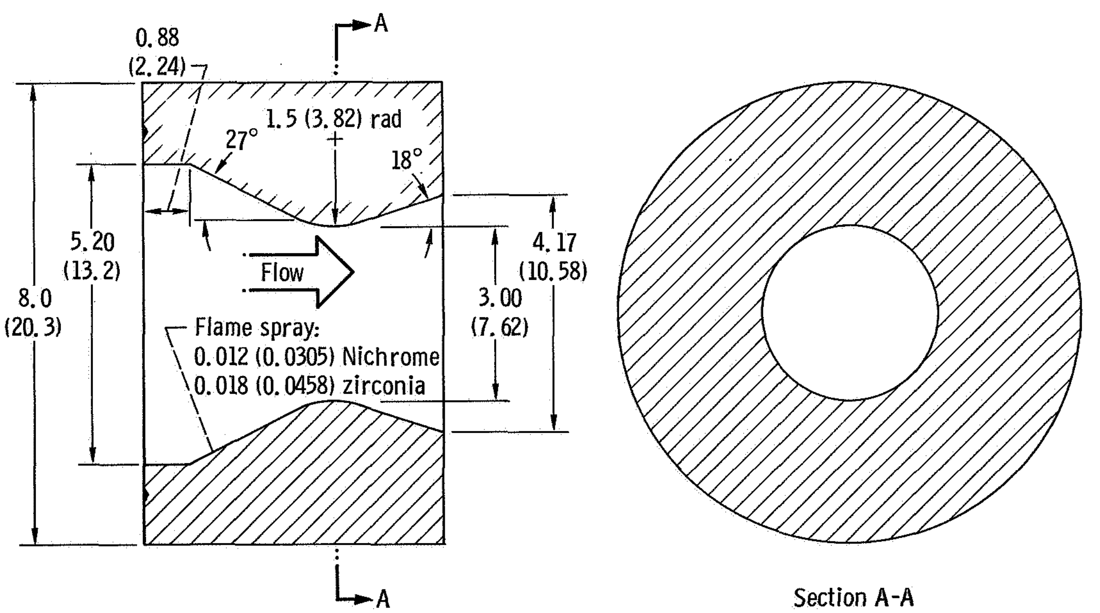
to prevent overheating of the injector face by moving the flame front aft. The impingement distance and angle used were the same as that of previous Earth storable injectors. Mutually perpendicular triplet elements were used to provide good propellant mixing. The design was intended to provide uniform propellant distribution with no O/F zoning.

Thrust chambers. - Two types of thrust-chamber construction were used to evaluate characteristic exhaust velocity efficiency. The zirconia-coated steel thrust chamber design is shown in figure 2. The design provided a simple, economical method for injector testing. The water-cooled thrust chamber is shown in figure 3. This was used to test injector durability over long firing times before comitting the ablative chamber to test. The two thrust chamber constructions also enabled measurement of the performance difference, if any, between water-cooled and uncooled thrust chambers. The L* variation during injector performance testing was achieved by using a 6-inch (15.24-cm) chamber length (L* = 23 in. or 58.4 cm), a 9-inch (22.8-cm) chamber length (L* = 33 in. or 83.8 cm), and the two together as a 15-inch (38.1-cm) chamber length (L* = 53 in. or 134.6 cm).

Figure 4 illustrates the design of the ablative thrust chamber. The material used was 65 percent graphite cloth/35 percent phenyl aldehyde resin (FM5064), the same as was used successfully in the 150-pound (667-N) thrust engine of reference 2. The smaller engine utilized a reinforcement orientation of 60° to the centerline. Although fabrication of the smaller thrust chamber at 60° orientation was not a problem, the larger thrust chamber was fabricated with a 90° orientation to decrease delamination



(a) Chamber diameter, 5.20 inches (13.2 cm).



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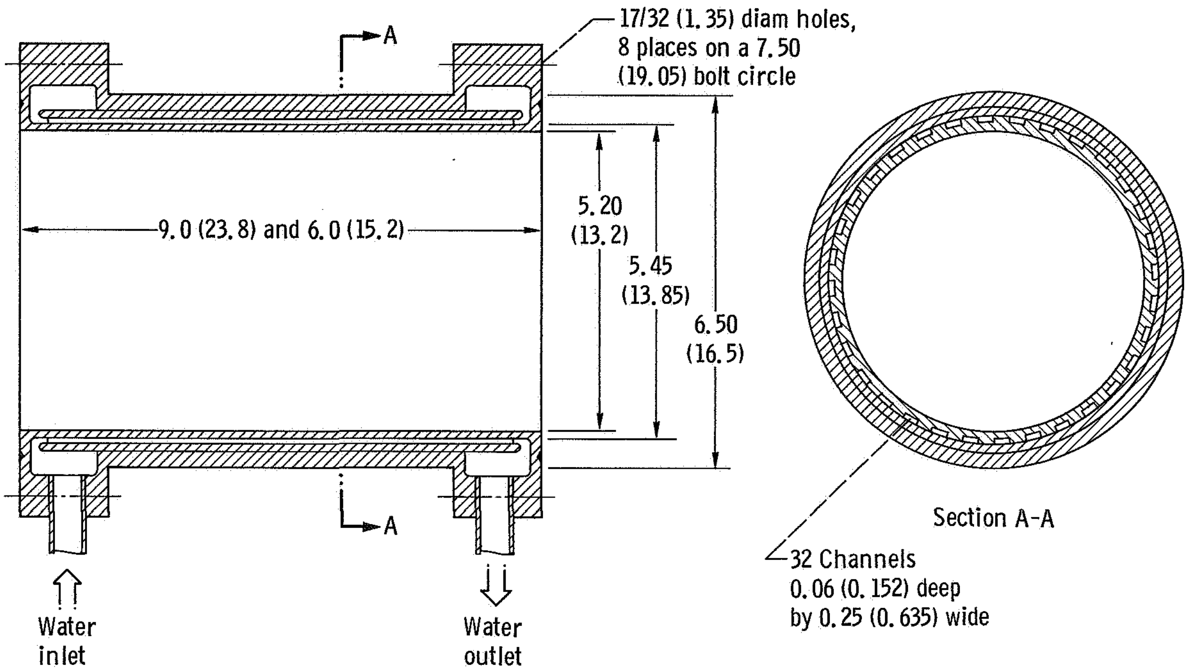
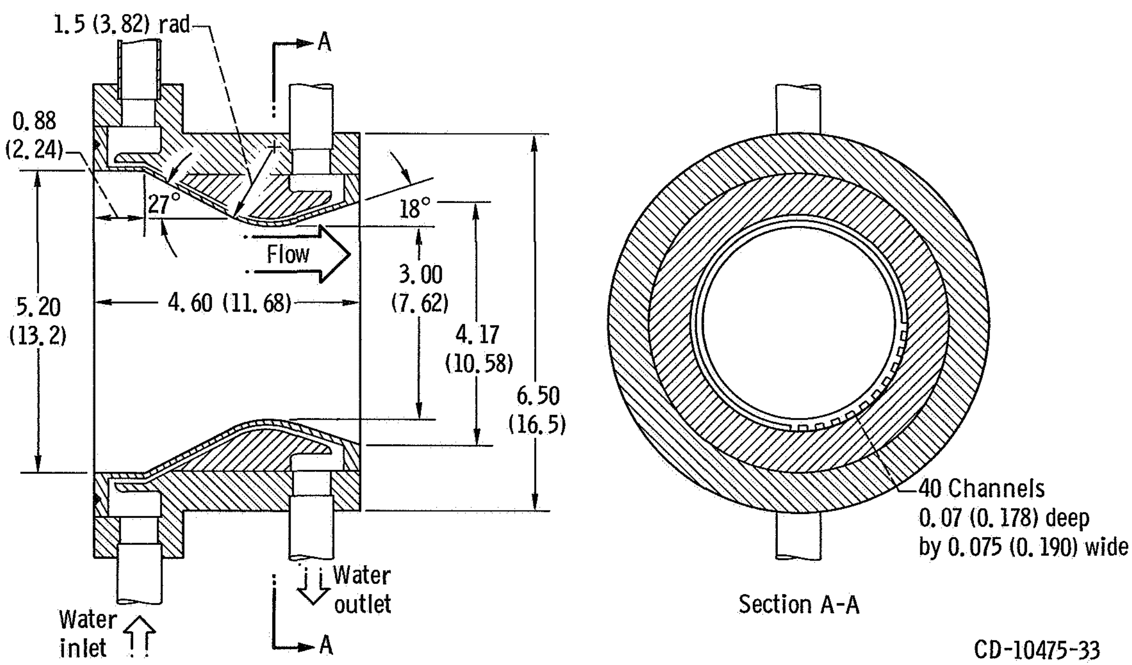
(b) Nozzle throat diameter, 3.0 inches (7.62 cm).

Figure 2. - Heat sink configuration. Material, mild steel. (All linear dimensions are in inches (cm).)



(a) Chamber diameter, 5.20 inches (13.2 cm).



(b) Nozzle throat diameter, 3.0 inches (7.62 cm).

Figure 3. - Water-cooled configuration. Material, 5083 aluminum. (All linear dimensions are in inches (cm).)

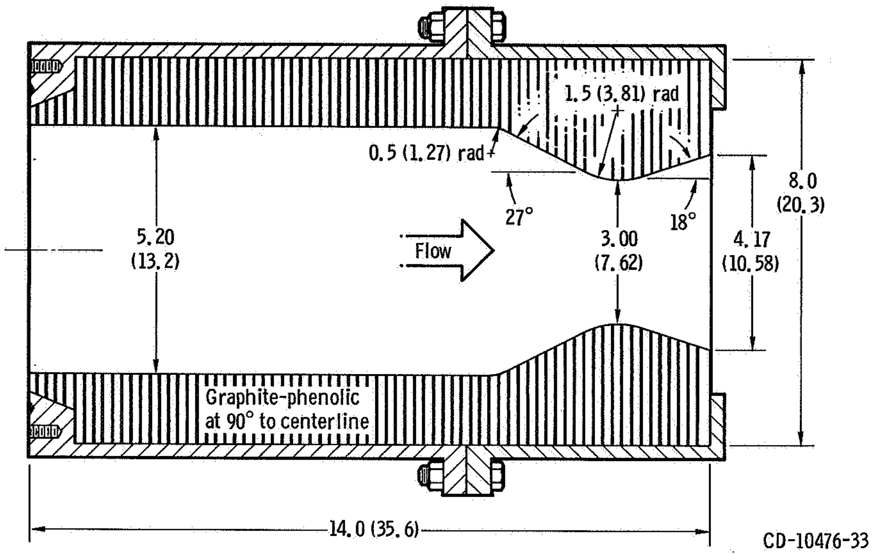


Figure 4. - Ablative engine. Throat diameter, 3.0 inches (7.62 cm). (All linear dimensions are in inches (cm).)

and cracking tendencies. An ablative chamber wall thickness of 1.4 inches essentially duplicated the 150-pound (667-N) thrust engine. No asbestos-phenolic overwrap was used on the 1000-pound (4450-N) thrust engine, however. The contraction ratio of 3.0 had been tested in the 150-pound (667-N) thrust engine size without excessive chamber erosion.

FACILITY

Test cell. - The experimental test firings were conducted in a small rocket engine test facility as shown in figure 5. A schematic diagram of the installation is given in figure 6. The propane tank was enclosed in a liquid nitrogen bath pressurized at 100 psia (690 kN/m^2) to provide a propellant temperature of 180° R (100 K) in the tank. Some pressure was necessary to keep the liquid nitrogen above the freezing temperature of propane (152° R) (84.5 K) and 180° R (100 K) was selected as a value that allowed relatively stable fuel injection temperatures during a long ablative firing.

FLOX was prepared by mixing liquid oxygen and liquid fluorine in a liquid nitrogen jacketed weigh tank. The average value attained by the weighing was 76.2 percent F_2 with a standard deviation of ±0.8 percent F_2. The actual percentages obtained during each mixing operation were used to calculate oxidant density and for theoretical values.

Instrumentation. - The combustion chamber pressure was measured by strain gage type pressure transducers. Two transducers were installed on a single hole in the injector face (see fig. 1, p. 3). The average value was used for performance calculations.

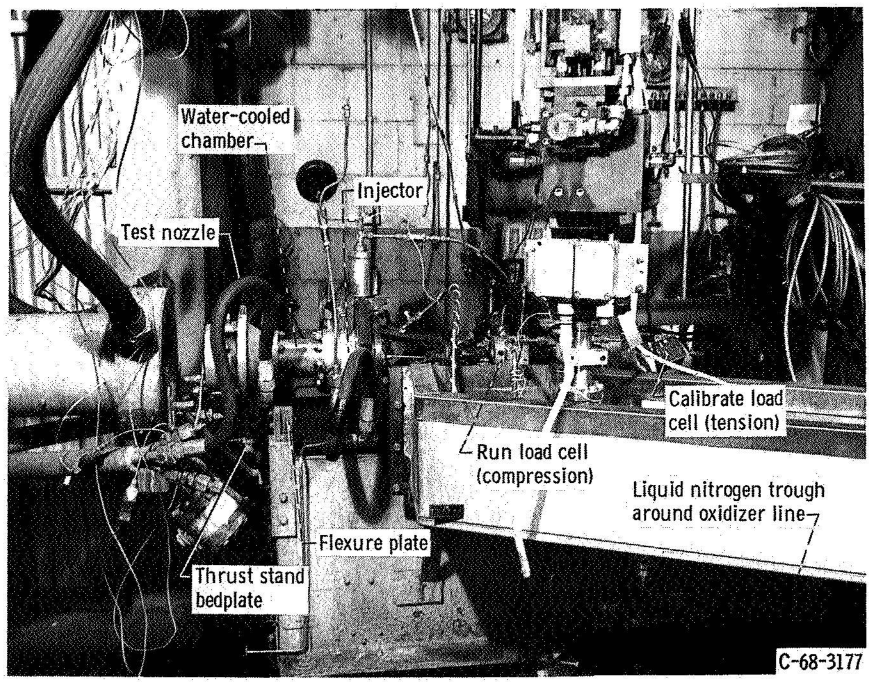


Figure 5. - Test facility.

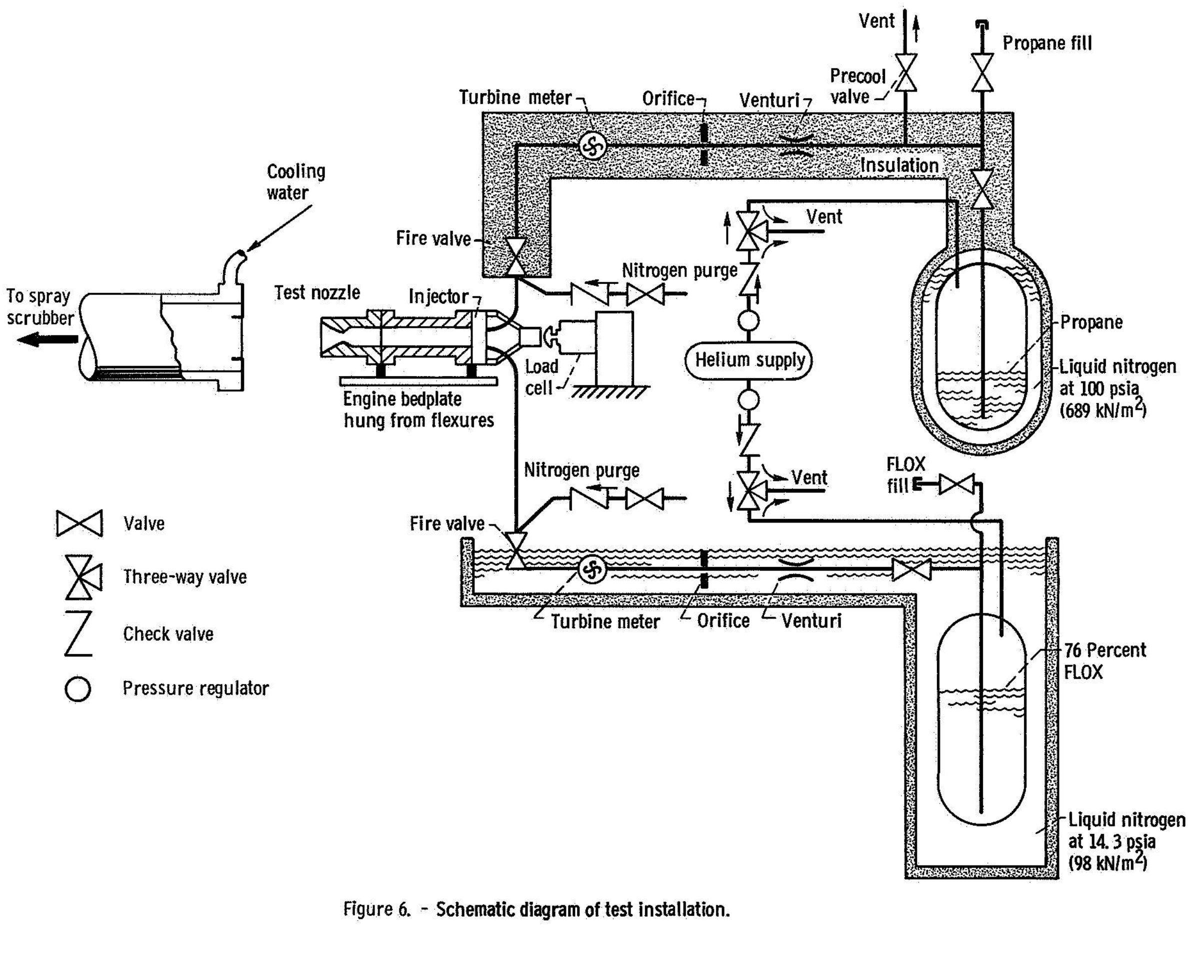


Figure 6. - Schematic diagram of test installation.

A high frequency flushmounted transducer mounted in the water-cooled and steel chamber walls was used to monitor combustion instability.

The flow rate of each propellant was measured three different ways: (1) flow orifice, (2) venturi, and (3) turbine flowmeter. Three meters were used to provide redundancy of measurement and a method for checking flow rates against one another. The meters were calibrated separately with water. Corrections were calculated to account for density and thermal contraction differences when the meters were used with the propellant. The three fuel flowmeters agreed within ±2 percent of one another while the oxidizer flowmeters agreed to within ±1.5 percent of one another. The final propellant weight flow was obtained by averaging the three values for each propellant. No analytical method was thought sufficient to discriminate between the three types of flowmeters. Flow calibration with the actual propellants is required to determine the most accurate type of flowmeter.

Thrust was measured with a double-bridge strain gage type load cell.

PROCEDURE

Data Recording and Processing

All electrical sensor outputs were digitized and converted into calculated values by use of a digital computer. Selected electrical outputs were also recorded on a multichannel oscillograph for control room examination and processing.

Test Procedure

Precooling of about three-fourths of the length of the insulated propane line was accomplished by running liquid propellant through the line prior to firing. The instrumentation was electrically calibrated prior to each run. A sequence timer automatically activated appropriate valves, data acquisition equipment, and propellant line purges. A high frequency response flushmounted pressure transducer located in the water-cooled and steel thrust chambers was monitored with an oscilloscope to check for combustion instability during the firings. Two separate automatic closed loop controllers were used to maintain chamber pressure and O/F constant during each test firing.

Engine Assembly

Two basic engine assemblies were used. Figure 7 shows the water-cooled engine

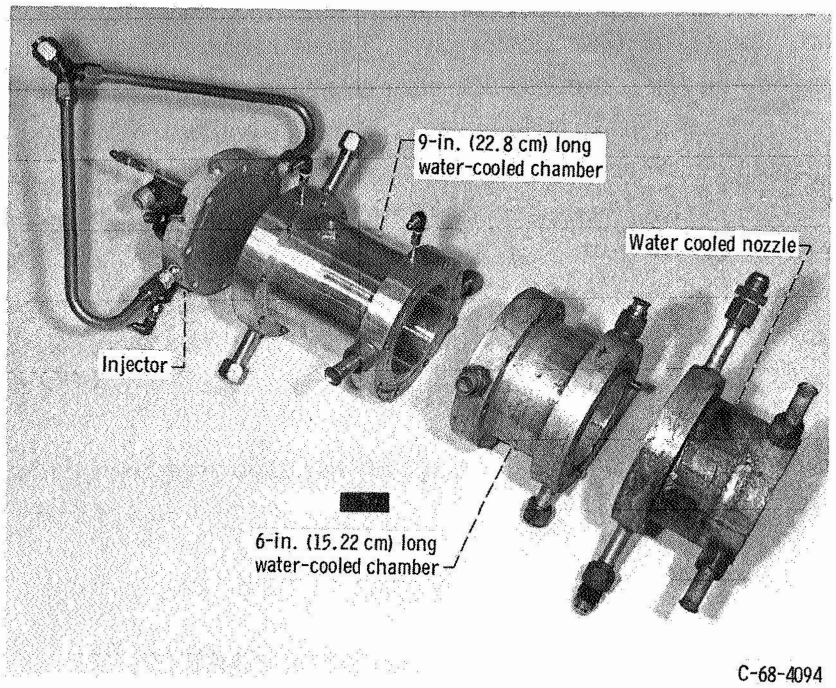


Figure 7. - 1000-Pound (4450-N) thrust water-cooled engine, disassembled.

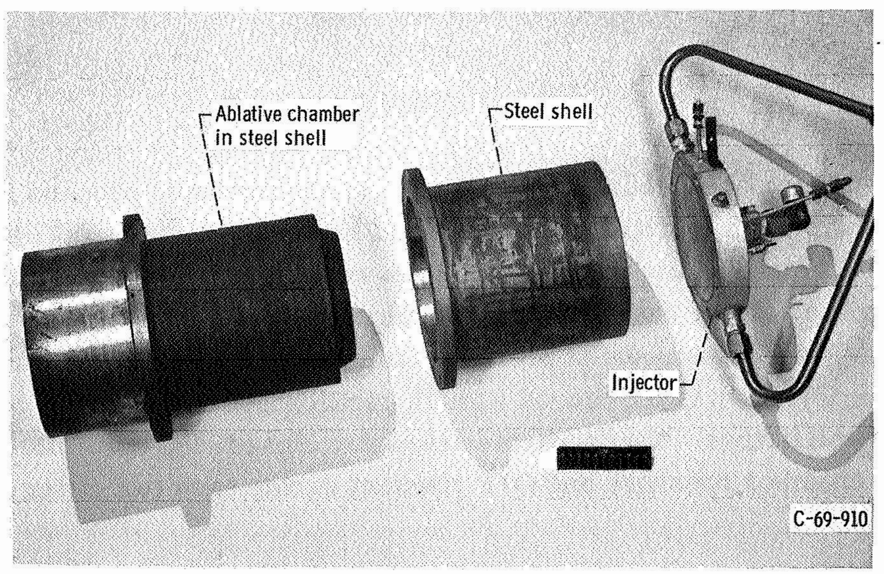


Figure 8. - 1000-Pound (4450-N) thrust ablative engine assembly.

assembly used for the characteristic velocity efficiency test firings. Figure 8 is the engine assembly used for the long duration test firings of the ablative thrust chamber.

Calculations

All theoretical values were calculated using the one-dimensional isentropic expansion method of Sanford Gordon (ref. 4). Theoretical values for the actual fluorine percentages obtained from each mixing operation were used.

A post-test measurement was also made by taking a photograph of the rocket throat plane. The photograph was enlarged to twice the size and the area of the throat measured with a planimeter. The area was then converted to an effective throat radius to give an independent measure of throat erosion.

RESULTS AND DISCUSSION

Characteristic Velocity Efficiency

Calculations made by the method of reference 4 indicate a performance loss of 0.87 percent if heat absorbed by the water-cooled engine is subtracted from the combustion process. The calculated loss was not added to the values reported, however. Neither was any correction made for thermal effects on the throat area due to heating during a test firing. It was felt the two corrections would be compensating so that the corrected value would be within the uncertainty band of the ηC^* data (±0.9 percent for one standard deviation).

Effect of oxidant-to-fuel ratio on injector performance. - Figure 9 is a plot of the

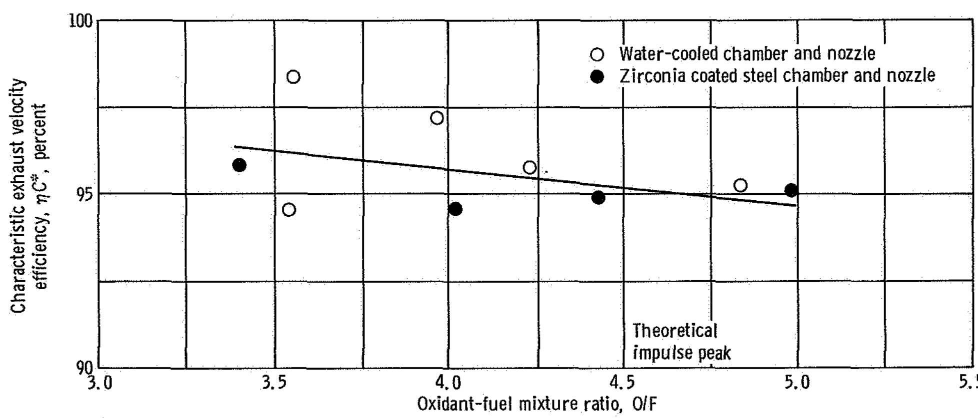


Figure 9. - Injector performance for 1000-pound (4450-N) thrust engine. Chamber pressure, 100 psia (690 kN/m^2); throat diameter, 3.0 inches (7.62 cm); propellant, FLOX-propane; characteristic chamber length, 33 inches (83.8 cm).

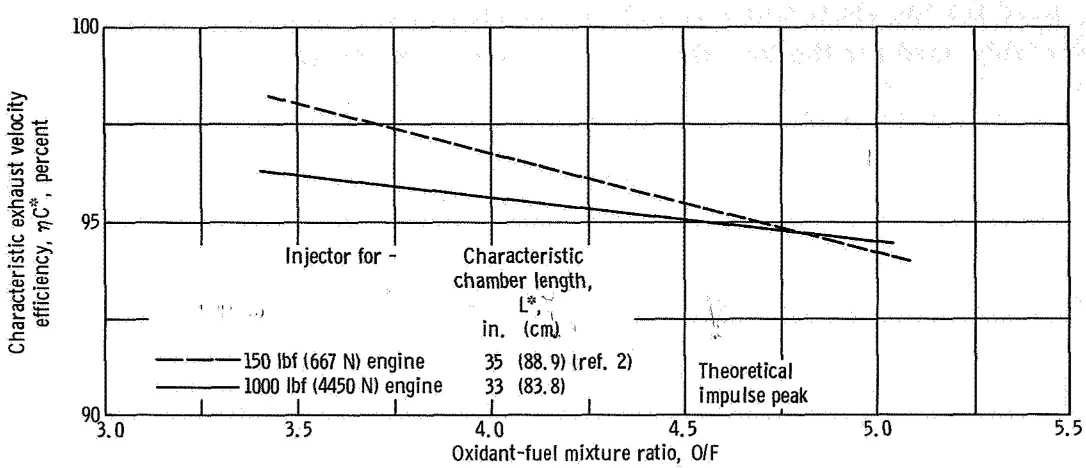


Figure 10. - Injector performance comparison for constant characteristic length. Chamber pressure, 100 psia (690 kN/m^2); propellants, FLOX-propane.

firing data for the short duration firings. Plotted is the characteristic exhaust velocity efficiency as a function of the oxidant to fuel ratio over a range of 3.5 to 5.0. No significant difference in efficiency level was measured between the water-cooled engine firings and those made with ZrO_2-coated steel engines.

The general trend of increasing performance with decreasing O/F agrees with that found earlier in references 1 and 2. The efficiency level reported here and in reference 2 was higher than that of reference 1, however. The higher efficiency was most likely due to the use of mutually perpendicular triplet elements which provide better mixing than the parallel triplets used in reference 1.

Figure 10 compares the efficiency levels for the injector (150 lbf or 667 N) from reference 2 and the injector described herein (1000 lbf or 4450 N). The performance values were close enough to demonstrate successful scaling. The performance of the smaller injector was somewhat higher, which we believe was due to improved propellant vaporization from the smaller droplet size distribution produced by smaller hole diameters and higher impingement angle.

Effect of characteristic chamber length on injector performance. - Figure 11 compares the effect of characteristic chamber length (L*) on injector performance at an O/F of 4.5 for both engine sizes. For the 1000-pound (4450-N) thrust engine, an L* value of 33 inches (83.8 cm) was required to give 95.2 percent C* and an increase to 53 inches (134.6 cm) increased the C* efficiency to only 95.5 percent. Calculations made by the method of reference 5 indicate the propellants were completely vaporized at an L* of 33 inches (83.8 cm). Therefore, with the propellant droplet size distribution present, the process was most likely mixing limited in efficiency level at the 4.5 O/F.

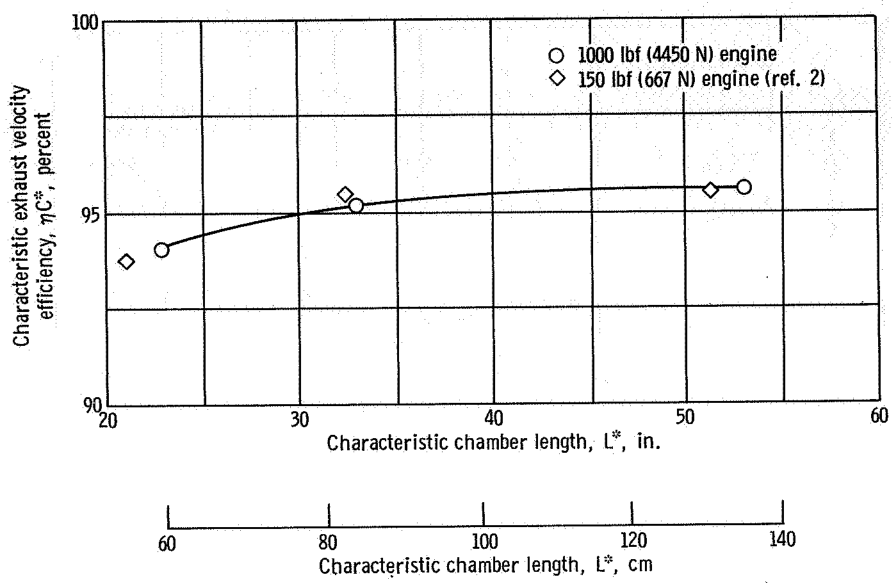


Figure 11. - Injector performance comparison for constant oxidant-fuel mixture ratio of 4.5. Chamber pressure, 100 psia (690 kN/m^2); propellants, FLOX-propane.

Combustion stability. - During the performance testing, no predominant high frequency was detected during steady-state engine operation. It was therefore concluded that stable combustion was attained. Normal minor fluctuations in chamber pressure during steady-state engine operation were generally 2 to 3 percent of chamber pressure in magnitude.

Ablative Engine Firings

A long term run capability was demonstrated using an ablative thrust chamber with an L* of 33 inches (83.8 cm). The ablative engine was manufactured using 65 percent graphite fabric and 35 percent of a highly substituted ring structure phenyl aldehyde condensation resin. The firing results are shown plotted in figure 12. The two short firings were caused by faulty low flow abort signals. Complete charring of the ablative leading to excessive outside surface temperatures necessitated termination after a third firing of 155 seconds duration. The charring was accentuated by the two 20-second firings which resulted in added amounts of heat soak during shutdown. The rapid char-through was also partially due to the limited wall thickness and to the material being oriented 90° to the nozzle centerline instead of 60° used in the small-scale testing (ref. 2). The 90° orientation was chosen for the larger size engine to lessen delaminations and cracking during fabrication. The test results indicate a thicker ablative wall coupled with 60° centerline orientation would be more desirable if crack-free parts can be fabricated as in reference. 6

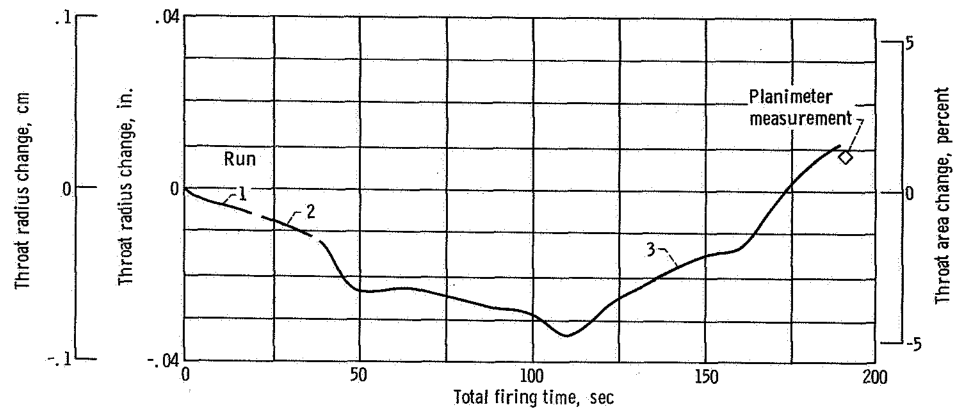


Figure 12. - Ablative engine performance. Graphite-phenolic liner oriented 90° to centerline. Nozzle throat diameter, 3.0 inch (7.62 cm); propellants, FLOX-propane; chamber pressure, 100 psia (690 kN/m^2); oxidant-fuel mixture ratio, 4.5.

Adding asbestos phenolic insulation as was done in reference 2 would have decreased outside temperatures but probably would not have changed the erosion due to char-through. Another alternative would be to use a more insulating material (carbon-phenolic) as a second layer around the graphite-phenolic. Scaling of the ablative thrust chamber, however, was generally successful.

The reduction in nozzle throat area during the early portion of the tests was probably due to graphite deposits on the throat. The surface temperature at this time was conducive to deposition. When char-through caused an increase in the surface temperature, oxidation predominated, resulting in the erosion shown. A post-test photograph of the assembled ablative engine is shown in figure 13.

Figure 13 also shows the graphite deposits on the injector end to illustrate a potential problem with these propellants such as injector blockage or flow disturbances. The performance was not significantly decreased during the 155-second run, however. These deposits were not observed during the tests of reference 2. In reference 2, however, a water-cooled chamber was adjacent to the injector. A thin deposit of soot or amorphous carbon was deposited on the cold walls. For the 1000-pound (4450-N) thrust engine, the ablative walls adjacent to the injector provided a hot surface for the deposition of graphite. Deposits shown in figure 13 grew radially inward from the ablative chamber wall rather than from the cooled injector face. The 1000-pound (4450-N) thrust engine also had more widely spaced injection elements with a longer impingement distance. These differences would have allowed more recirculation with the larger engine.

A photograph of the cross section of the ablative engine (fig. 14) illustrates holes where the decomposing resin escapes, delaminations, and loss of material at the throat due to oxidation and chunking, the primary failure mode. Oxidation of the chamber wall near the injector, probably due to the recirculation pattern, can also be seen in figures 13 and 14.

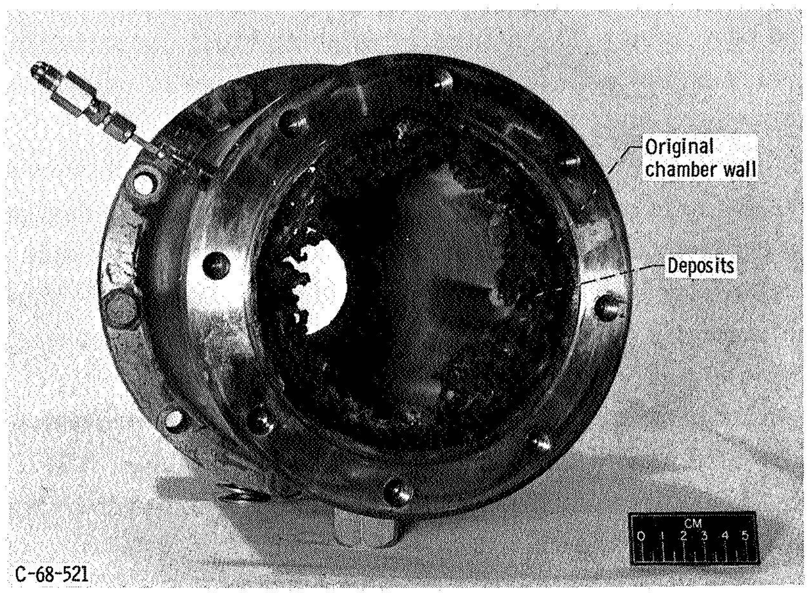


Figure 13. - 1000-Pound (4450-N) thrust rocket engine. (View form injector end.)

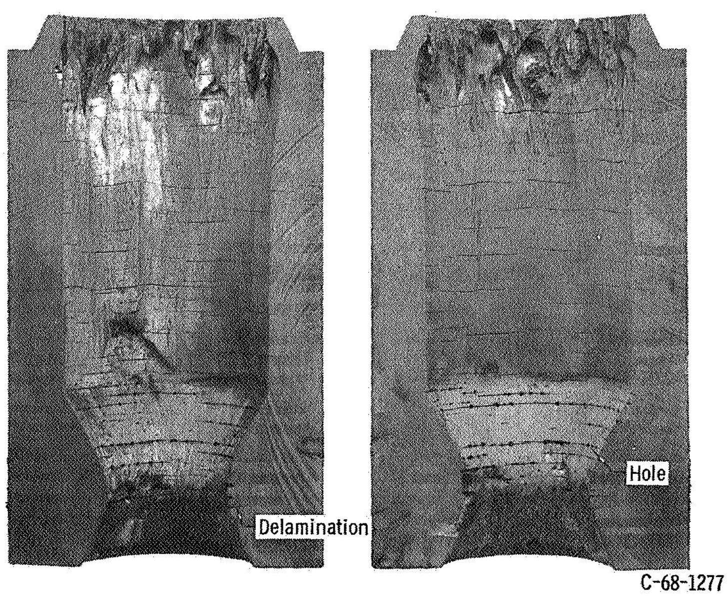


Figure 14. - 1000-Pound (4450-N) thrust ablative engine post-test.

SUMMARY OF RESULTS

A 3.0-inch (7.62-cm) throat, 1000-pound (4450-N) thrust, FLOX-propane engine was designed and test fired. The nominal design and test operating conditions included a chamber pressure of 100 psia (690 kN/m^2) over a range of oxidant to fuel ratios. The results of the program are as follows:

1. A FLOX-propane injector was designed and tested which showed stable combustion at an efficiency level of approximately 95 percent. The design was successfully scaled from a previously demonstrated 150-pound (667-N) thrust size to the 1000-pound (4450-N) thrust size reported. The principles of using small hole diameters for good vaporization and mutually perpendicular triplet elements for good mixing were found valid in both engine sizes.

2. An L* value of 33 inches (83.8 cm) was required to provide 95.2 percent characteristic velocity efficiency. An increase in L* to 53 inches (134.6 cm) provided only a slight increase in efficiency to 95.5 percent. These results indicate that the process was most likely mixing limited.

3. A run duration capability of 190 seconds total was demonstrated using a graphite cloth/phenyl aldehyde resin ablative thrust chamber. The test was terminated because of ablative char-through which led to excessive outside surface temperatures. Longer run durations would require a thicker ablative wall, a different ablative reinforcement orientation, an insulative overwrap, or a combination of these.

4. Throat erosion was prevented initially by the ablative process coupled with carbon deposition. After ablative char-through, oxidation began, resulting in about 1 percent throat area increase after 190 seconds total firing time.

5. Carbon deposits near the injector can be of such magnitude as to block or deflect the injection streams and, therefore, are a potential problem. A shorter impingement distance and higher impingement angle may reduce these carbon deposits.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, May 22, 1969,
128-31-36-02-22.

APPENDIX - SYMBOLS

A_e	rocket nozzle exit area, in.^{2}; cm^2
A_f	fuel injection area, ft^2; m^2
A_{ox}	oxidant injection area, ft^2; m^2
A_t	throat area, in.^{2}, cm^2
C_d	nozzle discharge coefficient (0.994)
C^*_{theor}	theoretical (shifting equilibrium) characteristic exhaust velocity, ref. 4, ft/sec; m/sec
F	measured thrust, lbf; N
F_{vac}	vacuum thrust, $F + P_o A_e$; lbf; N
g	gravitational constant, 32.174 ft/sec^2; 9.81 m/sec^2
I_{vac}	vacuum specific impulse, F_{vac}/W_p, sec
$I_{vac\ theor}$	theoretical vacuum specific impulse, ref. 4, sec
L*	characteristic chamber length, V_c/A_t, in.; cm
O/F	oxidant to fuel ratio, W_{ox}/W_f
P_c	chamber pressure measured at injector, psia; kN/m^2
$P_{c_{corr}}$	total pressure at rocket throat, φP, psia; kN/m^2
P_o	ambient pressure surrounding engine, psia; kN/m^2
R_i	initial throat radius measured, in.; cm
R_t	throat radius at any time, $\sqrt{\dfrac{W_p \eta C^*_K C^*_{theor}}{\pi g P_{c_{corr}} C_d}}$, in.; cm
V_c	chamber volume, in.^{3}; cm^3
V_{ox}	oxidant injection velocity, $\dfrac{W_{ox}}{\rho_{ox} A_{ox}}$, ft/sec; m/sec
V_f	fuel injection velocity, $\dfrac{W_f}{\rho_f A_f}$, ft/sec; m/sec
W_f	fuel weight flow, lbm/sec; kg/sec
W_{ox}	oxidant weight flow, lbm/sec; kg/sec

W_p total propellant weight flow, lbm/sec; kg/sec

ΔR_{eff} effective throat radius change, in.; cm, $R_t - R_i$

ρ_f fuel density, lbm/ft^3, kg/m^3

ρ_{ox} oxidant density, lbm/ft^3, kg/m^3

ηC^*_{exp} experimental characteristic exhaust velocity efficiency, $\frac{\eta I_{sp}}{\eta C_{Fvac}}$

ηC_{Fvac} vacuum thrust coefficient efficiency (determined as in ref. 3), 0.972

ηC^*_K constant characteristic exhaust velocity efficiency (determined from calibration firings)

ηI_{sp} vacuum specific impulse efficiency, $\frac{I_{vac}}{I_{vac\ theor}}$

φ momentum pressure loss correction, 0.98 (ref. 7)

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