NASA Technical Memorandum 81503

COOLING OF HIGH PRESSURE ROCKET THRUST CHAMBERS WITH LIQUID OXYGEN

H. G. Price
Lewis Research Center
Cleveland, Ohio

Prepared for the
Sixteenth Joint Propulsion Conference
cosponsored by the AIAA, ASME, and SAE
Hartford, Connecticut, June 30-July 2, 1980

COOLING OF HIGH PRESSURE ROCKET THRUST
CHAMBERS WITH LIQUID OXYGEN

H. G. Price*
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

Abstract

An experimental program using hydrogen and oxygen as the propellants and supercritical liquid oxygen (LOX) as the coolant was conducted at 4.14 and 8.274 MN/m^2 (600 and 1200 psia) chamber pressure. The objective was to demonstrate and provide experimental data on the following:

(1) The effect of LOX leaking into the combustion region through small cracks in the chamber wall.

(2) Verification of the supercritical oxygen heat transfer correlation developed from heated tube experiments.

Four thrust chambers with throat diameters of 0.066 m (2.6 in.) were tested. Three of these were cyclically tested to 4.14 MM/m^2 (600 psia) chamber pressure until a crack developed. One had 23 additional hot cycles accumulated with no apparent metal burning or distress. The fourth chamber was operated at 8.274 MM/m^2 (1200 psia) pressure to obtain steady state heat transfer data. Wall temperature measurements confirmed the heat transfer correlation.

Introduction

Preliminary design studies by NASA and its contractors for vehicles such as the mixed-mode single-stage-to-orbit (SSTO) and the heavy lift launch vehicle (HLLV) have indicated that high pressure (>27.58 MN/m^2 (4000 psia) chamber pressure) booster engines using a hydrocarbon fuel with oxygen for propellants may be needed.[1] These vehicles would be used to provide a truly economical means of accomplishing many of the space missions envisioned in the 1990 time period.

It has been shown that the hydrocarbon fuel (RP-1) is limited in its cooling capability, and thus, may not be able to cool the thrust chamber at these high chamber pressures. This may necessitate the use of the oxygen as the regenerative coolant. Analysis[1] using a heat transfer correlation like that developed from heated tube tests[2] has indicated that supercritical oxygen is capable of cooling such engines to chamber pressures in excess of 27.58 MN/m^2 (4000 psia) with reasonable pressure budgets. Experimental verification of this heat transfer correlation is needed before serious consideration could be given to this alternative cooling technique. In addition, experimental verification is needed that a small leak of the coolant oxygen into the combustion chamber would not cause a catastrophic failure where ignition of the metal thrust chamber wall would occur.

*Aerospace engineer.

Testing[3] with kerosine and LOX has demonstrated the feasibility of using LOX as a coolant at 8.481 MN/m^2 (1230 psia) chamber pressure and 50 KN (11 000 lbf) thrust. That program did not attempt to demonstrate the effect of a coolant leak into the combustion chamber.

This paper summarizes a program whose objective was to demonstrate that a small leak of the coolant oxygen into the thrust chamber during a firing would not result in a catastrophic failure, and to provide experimental verification of the supercritical oxygen heat transfer correlation under actual rocket firing conditions. Four thrust chambers were tested with hydrogen/oxygen as the propellants and supercritical liquid oxygen as the coolant. All of these chambers had identical cooling channel designs. Of the four, three were cyclically tested at 4.14 MN/m^2 (600 psia) chamber pressure until a crack developed from the coolant channel through the hot gas wall into the combustion chamber. These short duration cyclic tests were conducted to demonstrate that there would be no catastrophic failure if a crack developed in the thrust chamber during a firing. On one of the thrust chambers, 23 additional hot cycles were applied after a crack was first observed. The fourth thrust chamber was used to obtain steady state heat transfer data at 8.274 MN/m^2 (1200 psia) chamber pressure.

Heat Transfer Correlation

Heat transfer to supercritical oxygen has been investigated[2] with a series of heated tubes at high pressures ranging from 17 to 34.5 MN/m^2 (2500 to 5000 psia) and bulk temperature of 96 to 217 K (173° to 391° R). From this test data and previously existing data[4,5] a multiple regression analysis was conducted as part of the work done in Ref. 2 which led to the following design correlation for calculating supercritical oxygen heat transfer coefficients:

$$Nu_b = 0.0025\, Re_b Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-1/2} \left(\frac{K_b}{K_w}\right)^{1/2} \times \left(\frac{\overline{Cp}}{Cp_b}\right)^{2/3} \left(\frac{P}{P_{cr}}\right)^{-1/5} \left(1 + \frac{2}{L/D}\right)$$

where

- Cp constant pressure specific heat
- $\overline{Cp}$ integrated average specific heat from T_W to T_b
- D inside tube diameter
- h heat transfer coefficient
- K thermal conductivity
- L heated tube length
- Nu Nusselt number, hD/K

- P pressure
- Pr Prandtl number, $C_p\mu/K$
- Re Reynolds number, $\rho DV/\mu$
- T temperature
- V fluid velocity
- μ viscosity
- ρ density

Subscripts:

- b evaluated at bulk temperature
- cr critical state
- w evaluated at wall temperature

The thrust chambers used in this investigation were designed and fabricated using this coolant side heat transfer correlation. Refer to Ref. 6 for design details.

Apparatus and Procedure

Injectors

A typical face plate view of the injectors used in this program is shown in Fig. 1. In this design, all of the hydrogen was injected through the porous metal (Rigimesh) faceplate and the oxygen was injected through 91 showerhead tubes. Two grades of porosity of Rigimesh were used; 26.9 SCMM at 0.0138 MN/m^2 (950 SCFM at 2 psid) for testing at the 4.14 MN/m^2 (600 psia) chamber pressure and 39.6 SCMM at 0.0138 MN/m^2 (1400 SCFM at 2 psid) for testing at the 8.274 MN/m^2 (1200 psia) chamber pressure. The diameter of the oxidizer tubes was 1.321 mm (0.052 in.) for the 4.14 MN/m^2 (600 psia) injectors and 1.829 mm (0.072 in.) for the 8.274 MN/m^2 (1200 psia) injectors.

The 4.14 MN/m^2 (600 psia) injector type was used extensively on previous experimental programs[7] and was selected for this program because of its uniform circumferential heat flux profile, durability, and high energy release efficiency. The energy release efficiency of the injectors as measured during this program was greater than 98 percent.

Combustion Chambers

The thrust chambers were of a milled slot, electroform nickel closeout construction. The chamber liner which was made from oxygen-free, high-conductivity (OFHC) copper had 100 axial coolant channels. The details of the channel dimensions are given in Ref. 6. The inside dimensions of the thrust chambers were:

	Diameter
Chamber	0.122 m (4.8 in.)
Throat	.066 m (2.6 in.)
Exit	.159 m (6.26 in.)
	Length
Injector to Throat	0.254 m (10 in.)
Overall	.432 m (17 in.)

Four thrust chambers were used during this program. A photograph of a completed thrust chamber is shown in Fig. 2. The thrust chambers were instrumented with Chromel/Constantan thermocouples imbedded in the rib between coolant channels approximately 1.27 mm (0.05 in.) from the hot gas wall as described in Ref. 7. Two of the chambers, S/N 1 and 2, had four thermocouples evenly spaced circumferentially at the throat plane as shown in Fig. 2.

The other two, S/N 3 and 4, had 16 thermocouples evenly spaced circumferentially in four axial planes, two upstream of the throat at 0.127 and 0.222 m (5 and 8.75 in.) from the injector, one at the throat 0.254 m (10 in.) from the injector, and one 0.279 m (11 in.) from the injector. Each of the chambers also had a corresponding Chromel/Constantan thermocouple attached to the outside wall in the same position as the rib thermocouple. Figure 3 shows one of the thrust chambers with four planes of instrumentation as it was installed in the test facility. The thrust chamber fired vertically downwards.

Test Facility and Procedures

This investigation was conducted in a 222 410 N (50 000 lbf) thrust, sea-level rocket test stand equipped with an exhaust-gas muffler and scrubber. The facility used pressurized propellant storage tanks to supply the propellants to the combustion chamber. The propellants were liquid oxygen and ambient-temperature gaseous hydrogen. LOX was used as the coolant. Details of the installation can be seen in Fig. 3.

Two types of tests, cyclic and steady state, were preformed during this program. In the cyclic tests, the chamber was brought up to pressure and maintained at that pressure for 0.8 second and then the propellant valves were closed for a duration of 2 seconds (the LOX valve was closed first). This was followed immediately by a second cycle to the same operating condition. As many as 30 cycles at a time were performed in this manner. The LOX coolant flow continued during both firing and nonfiring portions of the cycle. This type of test was used to first produce a crack into the combustion chamber and then to investigate the effect of a LOX leak on thrust chamber wall integrity.

In the steady state tests, the pressure was brought up in the chamber and maintained at the desired level for a duration from 3 to 10 seconds. The heat transfer information was obtained from this type of test. The thermocouples imbedded in the channel ribs reached steady values in approximately 2 seconds and remained constant while the data were recorded.

Test cycles were programmed into a solid-state timer that was accurate and repeatable to within ±0.001 second. Fuel and oxidizer flows were controlled by fixed-position valves and propellant tank pressure. Coolant inlet pressure was controlled by coolant tank pressure. Coolant exit pressure was kept constant by a closed-loop controller modulating a backpressure valve. With this arrangement, the coolant flow rate started high and decreased to the desired value as the final combustion conditions were reached.

Control room operation of the test included monitoring of the test hardware by means of three closed-circuit television cameras and one cell microphone to aid in detecting cracks. The output of the microphone and one television camera were recorded on magnetic tape for later playback. The cell microphone did not prove to be as valuable a tool for detecting cracks in a chamber wall as it did when hydrogen was the coolant.[7] With hydrogen there was a distinctive sound made by the hydrogen escaping when a crack developed.

With LOX this change in sound was not nearly as noticeable. It took very careful monitoring of the tape to detect a LOX leak. Therefore, all of the leaks were found by observation after the tests. Data were recorded every 0.02 second, averaged over five recordings, and the average reported every 1/10 second.

Test Results

Test Conditions

Four thrust chambers were tested during this program. The conditions for these tests are shown in Table I. Chambers S/N 1 and 4 were operated at 4.14 MN/m^2 (600 psia) chamber pressure. Chamber S/N 3 was operated at 8.274 MN/m^2 (1200 psia) chamber pressure and chamber S/N 2 was operated at both pressures. Three of these thrust chambers (S/N 1, 2, and 4) were cyclically tested until a crack through the cooling channel to the combustion chamber was observed. The fourth chamber, S/N 3, was not cracked.

Steady state heat transfer information was obtained at a nominal chamber pressure of 4.14 MN/m^2 (600 psia) and mixture ratios of 4, 5, and 6 with thrust chamber S/N 4. Steady state heat transfer data was also obtained at a chamber pressure of 8.274 MN/m^2 (1200 psia) and a mixture ratio of 4 with thrust chambers S/N 2 and 3.

Heat Transfer Results

Figure 4 is a history of the rib temperatures measured at the throat of thrust chamber S/N 4 by the thermocouples imbebbed in the hot-gas wall. These thermocouples are located within 1.27 mm (0.05 in.) of the combustion surface. The operating conditions of this test were 4.151 MN/m^2 (602 psia) chamber pressure and a mixture ratio of 6.02. Also included in this figure are the outer or back side wall temperatures as measured by the thermocouples attached to that surface.

The SINDA computer program, as described in Ref. 8, using a cross-sectional model of the coolant channel, was used to predict the following:

(1) the hot-gas wall temperature
(2) the rib temperature
(3) the outer or back side wall temperature

These temperatures are also plotted in Fig. 4. The predicted temperatures were based on the hot gas heat transfer correlation as obtained previously[7,9] with hydrogen/oxygen propellants in this size thrust chamber and the same type of injector. The coolant side heat transfer coefficient was obtained from the supercritical liquid oxygen correlation as described in the section "Heat Transfer Correlation." From the plot it can be seen that the temperature predictions followed the measured temperatures very closely at the rib temperature location. The back side predicted wall temperature was higher than any of the measured temperatures. This is probably due mostly to axial heat transfer effects and some convective heat loss to the surroundings which are not accounted for by the computer model.

Figure 5 is a histogram of measured rib and back side temperatures also in the throat plane except in this case the thrust chamber was S/N 2 and the operating conditions were 8.067 MN/m^2 (1170 psia) chamber pressure and a mixture ratio of 4. For this test it was necessary to increase the coolant side heat transfer coefficient by 30 percent to obtain agreement between the predicted rib temperature and the measured rib temperature. Both the original predicted temperatures and the temperatures obtained by increasing the coefficient by 30 percent are plotted. This increase is within the data scatter band of the original heat transfer correlation development.

Figures 6(a) to (d) are histograms obtained at the four instrumented planes with thrust chamber S/N 3. The operating conditions were 8.315 MN/m^2 (1206 psia) chamber pressure and a mixture ratio of 4.1. In Fig. 6 data are presented at planes of 0.127 m (5 in.) (Fig. 6(a)), 0.222 m (8.75 in.) (Fig. 6(b)), 0.254 m (10 in.) (Fig. 6(c)), and 0.279 m (11 in.) (Fig. 6(d)) from the injector. Examination of the figures shows a large amount of circumferential variation in wall temperatures. This resulted from a slightly damaged injector which produced some high temperature combustion gas streaks. Again the predicted temperatures of Fig. 6 were obtained with a coolant side heat transfer coefficient which was increased by 30 percent. Because of the large amount of scatter in the experimental data, it is impossible to draw any conclusions on the agreement between experimental and predicted results. In general, the experimental data scattered around the predicted rib temperature with a 30 percent increase in the coolant side heat transfer coefficient.

With this limited amount of data of 4.14 MN/m^2 (600 psia) and 8.274 MN/m^2 (1200 psia) chamber pressure, it is not known if the trend of 30 percent increase to the coolant side heat transfer coefficient at 8.274 MN/m^2 (1200 psia) will continue as chamber pressure increases. It is therefore recommended that the supercritical liquid oxygen correlation be used in its original form. This would give a conservative design until more data at the higher pressures can be obtained.

Effect of LOX Leaks on Thrust Chamber Integrity

One of the objectives of this experimental program was to determine what effect a crack in the combustion wall would have if it allowed oxygen to enter the combustion zone. It was postulated that there would be no effect if the metal wall were maintained below its ignition temperature. From Table II it can be seen that three of the chambers were operated until a crack developed. Leakage through these cracks was very evident by observing the large amounts of vapors leaving the chamber between cycles. This was particularly true in the case of thrust chamber S/N 2 which was operated 23 cycles after a crack (leakage) was evident. Figure 7 shows a number of the cracks in the chamber wall. Chambers S/N 1 and 4 were also operated a few cycles after a crack was formed in the chamber wall. None of these chambers showed any signs of apparent metal burning or distress. The cracks acted like the cracks that developed when hydrogen was used as the coolant. There were no catastrophic failures during the test program.

Concluding Remarks

The present work has demonstrated that supercritical LOX is capable of cooling thrust chambers using hydrogen/oxygen as the combustion propellants. However, engines operating with RP-1/O_2 as the propellants may present a more severe operating environment if a small crack develops in the chamber wall. With these propellants a carbon film would be produced on the hot gas wall which would operate at a higher temperature than the metal wall. The LOX, entering the combustion chamber through the crack, could oxidize the carbon film which in turn would heat the chamber wall. If the metal ignition temperature was reached, the metal would oxidize and there would be a catastrophic failure. The LOX entering through the crack could also film cool the carbon layer with no oxidation of either the carbon film or the metal wall. Which of these events takes place has to be determined. This will be the subject of further experiments that are scheduled to be performed later this year. Since the hot side heat transfer coefficient is smaller when using RP-1/O_2, it will be possible to operate to a higher chamber pressure with the same thrust chamber design. Therefore, the RP-1/O_2 tests will be performed up to 13.79 MN/m^2 (2000 psia) chamber pressure.

Summary of Results

Four thrust chambers with identical designs were tested with hydrogen/oxygen as the propellants and LOX as the coolant. Three of these thrust chambers were tested at 4.14 MN/m^2 (600 psia) chamber pressure and over a mixture ratio range of 4 to 6. The fourth thrust chamber was tested at 8.274 MN/m^2 (1200 psia) chamber pressure at a mixture ratio of 4. The results of these tests were as follows:

1. Successful cooling with LOX was demonstrated.

2. Three chambers were cyclically tested until cracks occurred in the hot-gas wall that permitted oxygen to flow into the combustors with no catastrophic failures.

3. On one chamber more than 20 cyclic tests were made after the first through crack was observed with no apparent metal ignition or distress.

4. Confirmation of a LOX heat transfer correlation was obtained within the limits of the correlation.

5. The thrust chamber wall cracks that formed as a result of the cyclic testing with LOX as the coolant appeared to have similar characteristics as those with liquid hydrogen as the coolant.

References

1. Luscher, W. P. and Mellish, J. A., "Advanced High Pressure Engine Study for Mixed-Mode Vehicle Applications," Aerojet Liquid Rocket Company, Sacramento, CA, Jan. 1977. (NASA CR-135141)

2. Spencer, R. G. and Rousar, D. C., "Supercritical Oxygen Heat Transfer," Aerojet Liquid Rocket Company, Sacramento, CA, Nov. 1977. (NASA CR-135339)

3. Dederra, H. and Kirner, E., "High Pressure Rocket Engine Liquid Oxygen Technology," Paper No. IAF-76-174, Oct. 1976.

4. Rousar, D. and Miller, F., "Cooling with Supercritical Oxygen," AIAA Paper 75-1248, Sept. 1975.

5. Powell, W. B., "Heat Transfer to Fluids in the Region of the Critical Temperature," Jet Propulsion Laboratory, Pasadena, California, Progress Report No. 20-285, 1956.

6. Spencer, R. G., Rousar, D. C., and Price, H. G., "LOX-Cooled Thrust Chamber Technology Developments," AIAA Paper 78-1035, July 1978.

7. Hannum, N. P., Kasper, H. J., and Pavli, A. J., "Experimental and Theoretical Investigation of Fatigue Life in Reusable Rocket Thrust Chambers," NASA TM X-73413, 1976.

8. Smith, J. P., "Systems Improved Numerical Differencing Analyzer (SINDA): User's Manual," TRW Systems Group Redondo Beach, CA, TRW-14690-H001-R0-00, Apr. 1971. (NASA CR-134271)

9. Schacht, R. L., Quentmeyer, R. J., and Jones, W. L., "Experimental Investigation of Hot-Gas Side Heat-Transfer Rates for a Hydrogen-Oxygen Rocket," NASA TN D-2832, 1965.

TABLE I. - TEST CONDITIONS

Chamber S/N	Nominal chamber pressure, MN/m^2 (psia)	Nominal mixture ratio	Nominal coolant flow rate, kg/sec (lb/sec)	Nominal coolant inlet pressure, MN/m^2 (psia)	Nominal coolant outlet pressure, MN/m^2 (psia)
1	4.14 (600)	6	6.804 (15)	14.13 (2050)	10.34 (1500)
2	8.274 (1200) 4.14 (600)	4 6	9.072 (20) 7.257 (16)	22.00 (3200) 14.13 (2050)	13.51 (1960) 10.34 (1500)
3	8.274 (1200)	4	9.072 (20)	22.41 (3250)	13.17 (1910)
4	4.14 (600)	4-6	7.257 (16)	14.13 (2050)	10.34 (1500)

TABLE II. - TEST CONDITIONS AND HISTORY

Chamber S/N	Nominal chamber pressure, MN/m^2 (psia)	Number of cycles	Nominal mixture ratio
1	4.14 (600) 4.14 (600)	1-36 Found crack 37-41	6 6
2	8.274 (1200) 4.14 (600)	1-4 5-144 Found crack 145-176 Found 7 cracks	4 6
3	8.274 (1200)	1-13 No cracks	4
4	4.14 (600)	1-81 Found crack	4-6

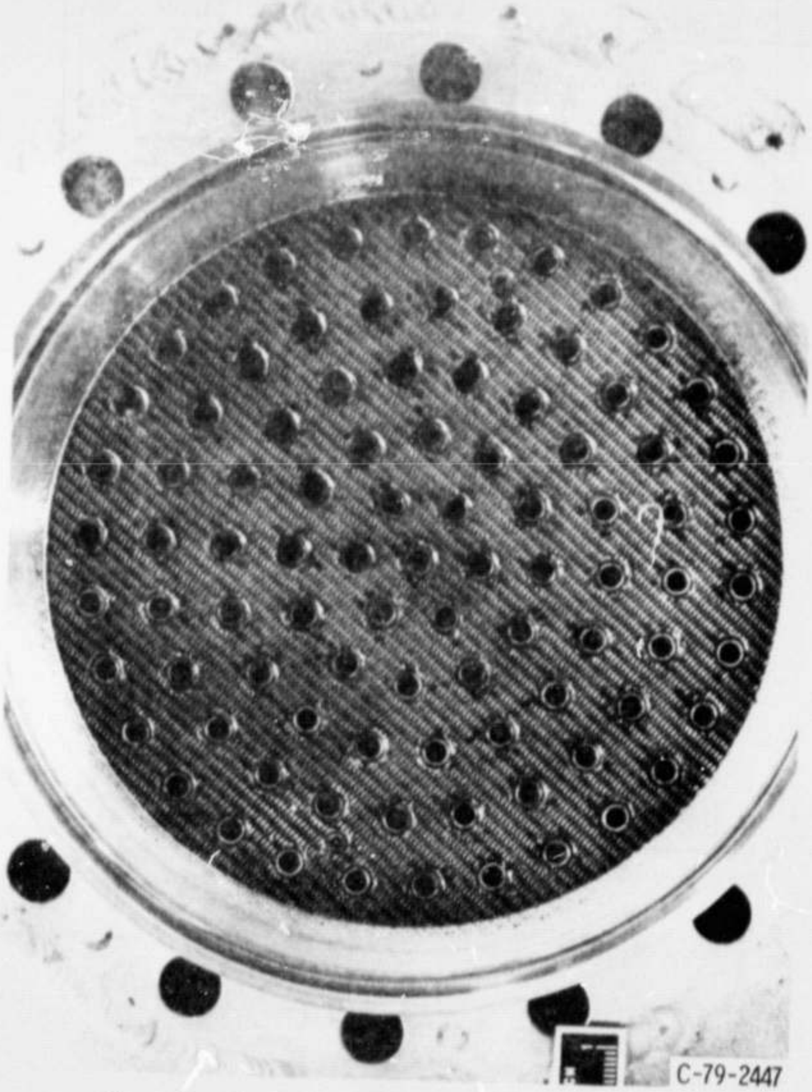


Figure 1. - Typical face plate of experimental injector.

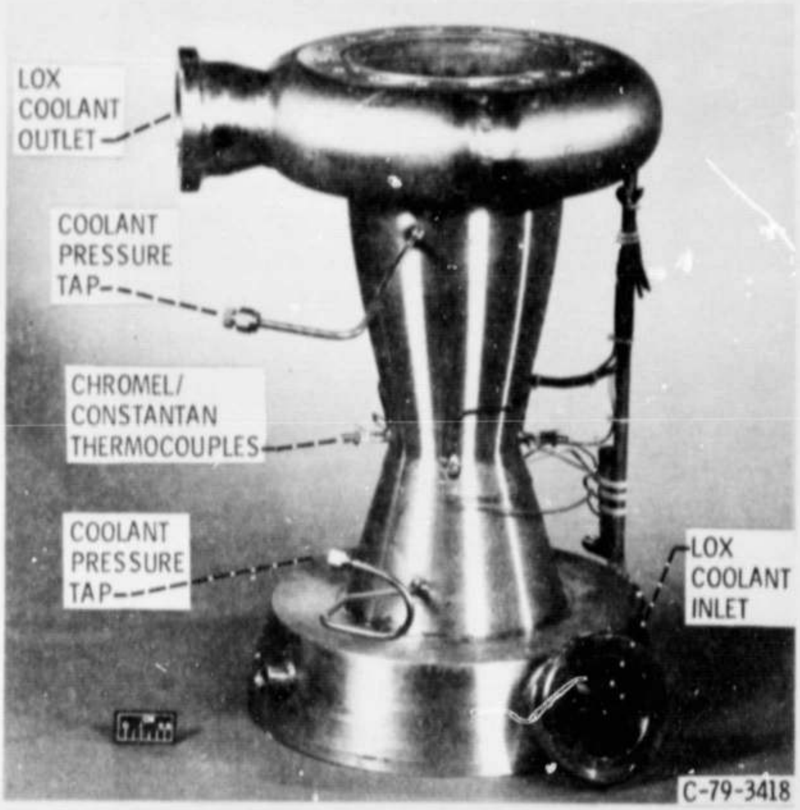


Figure 2. - Thrust chamber S/N 2.

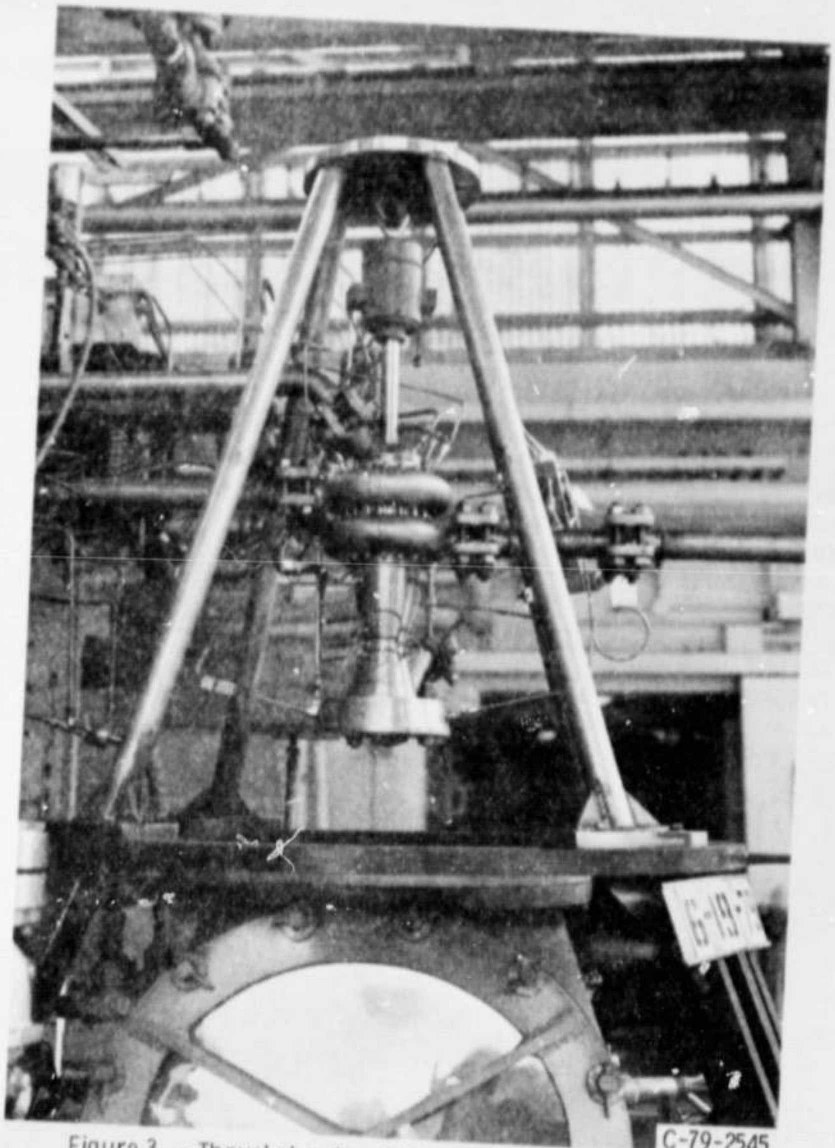


Figure 3. - Thrust chamber S/N 4 installed in test facility.

ORIGINAL PAGE IS
OF POOR QUALITY

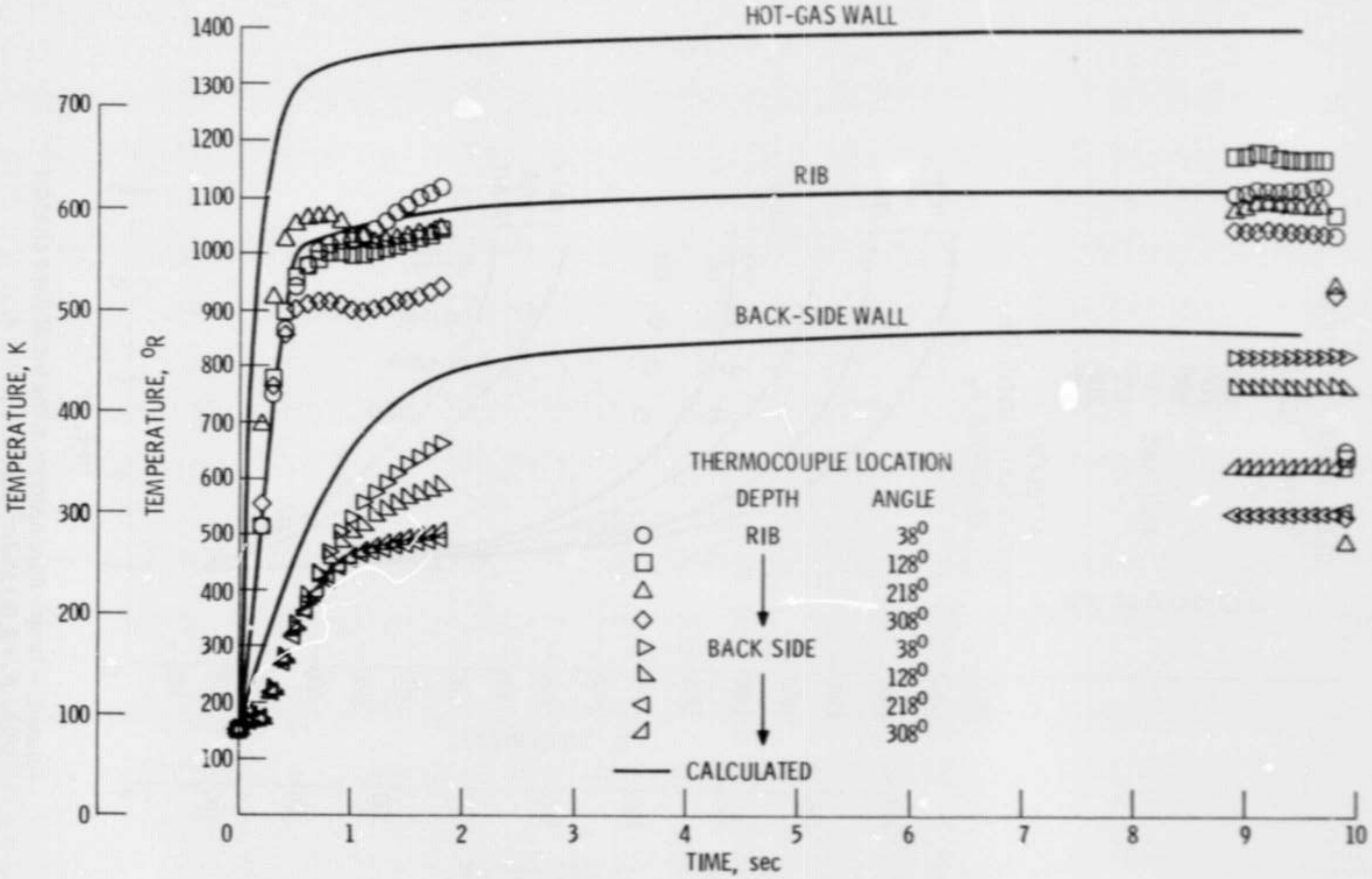


Figure 4. - Temperature histories at the throat plane chamber S/N4: P_c = 4.151 MN/m^2 (602 psia); O/F = 6.02.

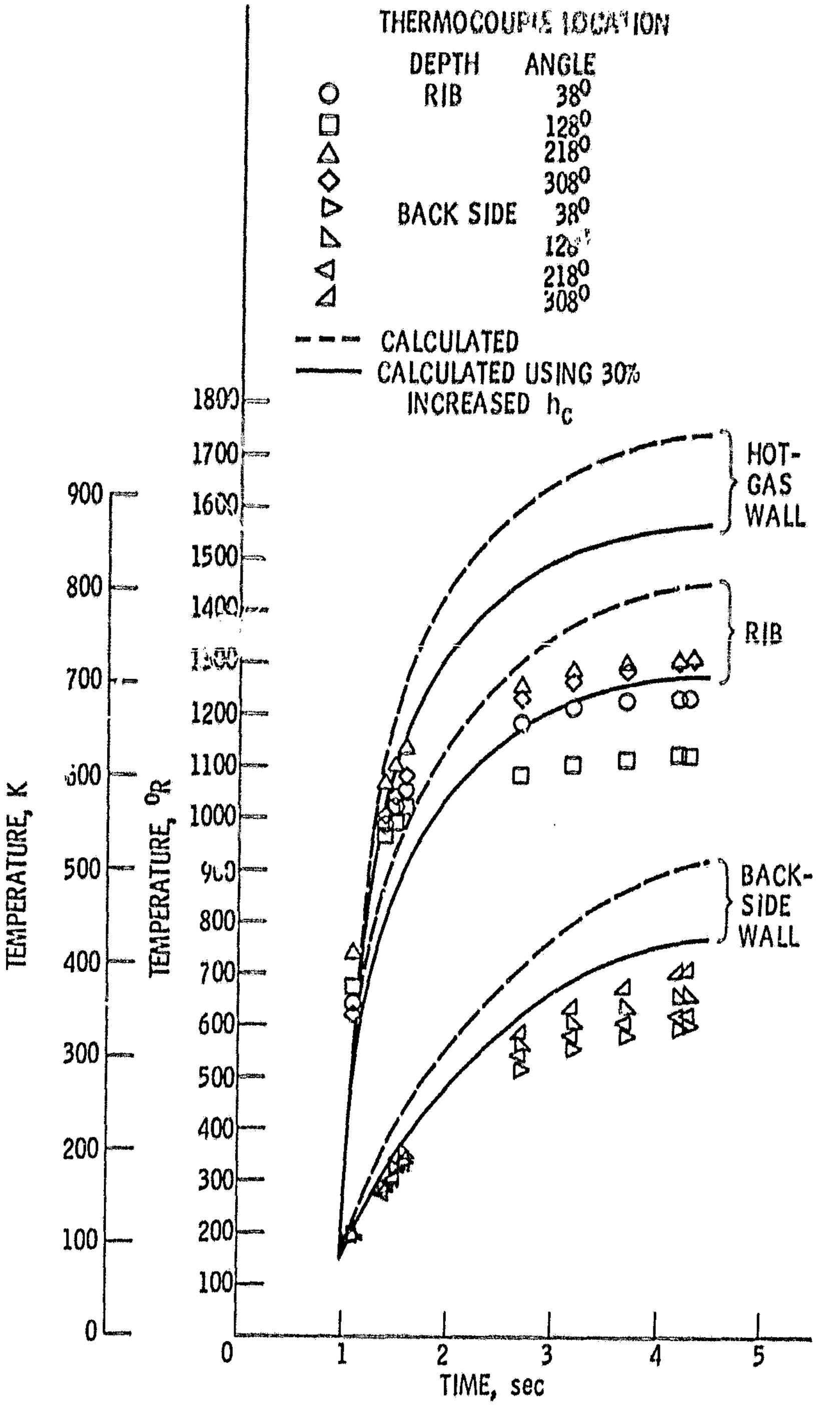
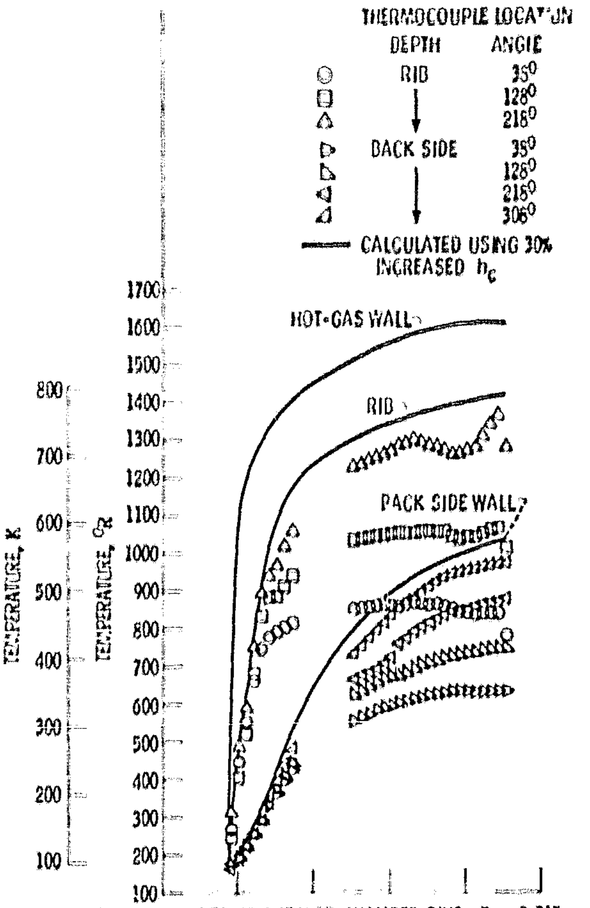


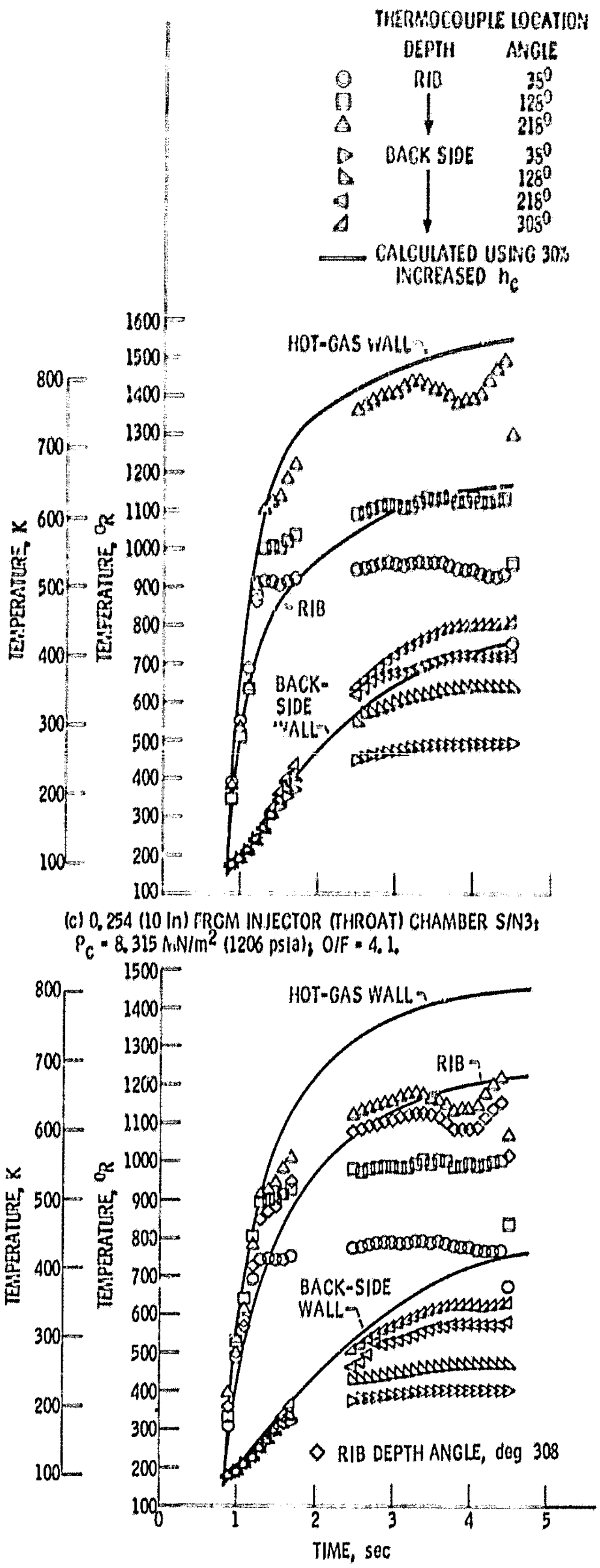
Figure 5. - Temperature histories at the throat plane chamber S/N2: P_c = 8.067 MN/m^2 (1170 psia); O/F = 4.



(a) 0.127 m (5 in) FROM INJECTOR CHAMBER S/N3; P_c = 8.315 MN/m² (1206 psia); O/F = 4.1.

(b) 0.222 m (8.75 in) FROM INJECTOR CHAMBER S/N3;
P_c = 8.315 MN/m² (1206 psia); O/F = 4.1.

Figure 6. - Temperature histories.



(c) 0.254 (10 in) FROM INJECTOR (THROAT) CHAMBER S/N3; P_c = 8.315 MN/m^2 (1206 psia); O/F = 4.1.

(d) 0.279 m (11 in) FROM INJECTOR CHAMBER S/N3; P_c = 8.315 MN/m^2 (1206 psia); O/F = 4.1.

Figure 6. - Concluded.

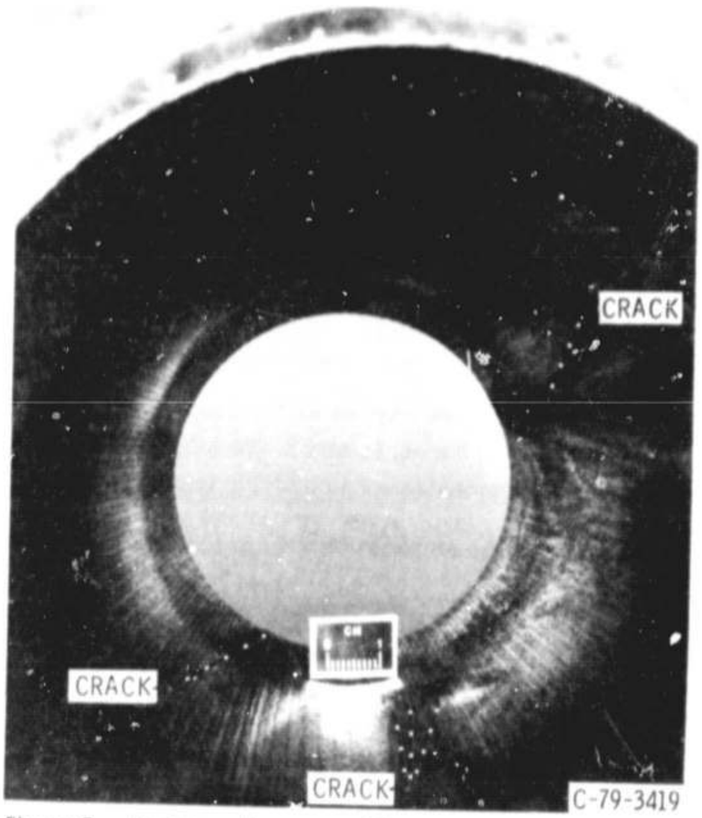


Figure 7. - Inside wall surface of thrust chamber S/N 2 after 176 hot fire cyclic tests.