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LOX-Cooled Thrust Chamber Technology Developments

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An experimental liquid oxygen (LOX) heat transfer study and a LOX-cooled thrust chamber demonstration program are summarized. Heat transfer to supercritical oxygen was investigated at 17-34.5 MPa (2460-5000 psia) and heat fluxes up to 90 MW/m^2 (55 Btu × in.^{2} s). Experimental data obtained previously were correlated along with these recent data and a design equation was derived which correlates 95% of the data within ±30%. LOX-cooled thrust chambers have been designed using this correlation and are currently being fabricated. A test program is planned for evaluating the LOX-cooled design concept. The purpose of these tests is to verify the LOX cooling correlation in a high-pressure liquid rocket engine and to determine the effects of a LOX coolant leak.

Nomenclature

Cp = constant pressure specific heat
$\bar{C}p$ = integrated average specific heat from T_w to T_b
D = inside tube diameter
h = heat-transfer coefficient
hg = hot-gas-side heat-transfer coefficient
k = thermal conductivity
L = heated tube length, in.
Nu = Nusselt number = hD/k
P_c = chamber pressure
O/F = mixture ratio
Pr = Prandtl number = $Cp\mu/k$
Re = Reynolds number = $\rho DV/\mu$
T = temperature
V = fluid velocity
$\dot{w}$ = coolant flow rate
μ = dynamic viscosity
ρ = density
ϕ = heat flux

Subscripts

b = evaluated at bulk temperature
cr = critical state
w = evaluated at wall temperature
GW = hot-gas-side wall
CW = coolant-side wall

Introduction

RECENT studies of advanced high-pressure rocket engines for future space transportation have shown that high-pressure booster engines will be needed that burn a hydrocarbon fuel with liquid oxygen.[1] Because of the generally poor cooling capabilities of the fuel, regenerative cooling with supercritical pressure oxygen is being considered for these engine systems.

Initial studies of supercritical oxygen chamber cooling indicated that it was feasible, but the results were considered preliminary since the cooling capability of high-pressure oxygen was unknown at the time.[2] Oxygen thrust chamber cooling was actually demonstrated in an 8.5 MPa (1230 psia) chamber pressure, 50 kN (11,000 lb$_f$) thrust LOX/kerosene engine at heat fluxes up to 35.6 MW/m^2 (21.8 Btu/in.^{2} s).[3] Subsequently, a preliminary design correlation for high-pressure oxygen heat-transfer coefficient was developed from a limited amount of heated tube test data.[4]

This paper summarizes two related supercritical oxygen cooling technology programs: 1) an experimental study of supercritical oxygen heat transfer characteristics which was recently completed and 2) an oxygen-cooled thrust chamber demonstration program which is currently being conducted.

The first study described in this paper was conducted on contract NAS 3-20384.[5] Heat transfer to supercritical oxygen was experimentally measured in electrically heated tubes. Experimental data were obtained for pressures ranging from 17 to 34.5 MPa (2460 to 5000 psia) and heat fluxes from 2 to 90 MW/m^2 (1.2 to 55 Btu/in.^{2}-s). Bulk temperatures ranged from 96 to 217 K (173 to 391 R). Experimental data obtained previously by Aerojet Liquid Rocket Company (ALRC)[4] and the Jet Propulsion Laboratory (JPL)[6] were correlated along with the recent data, with a total of 28 different correlating equations evaluated. A design equation was established which predicts 95% of all the available LOX data within ±30%.

Heat transfer analyses have indicated that oxygen is capable of cooling high-pressure LOX/hydrocarbon engines at chamber pressures in excess of 4000 psia with reasonable pressure budgets and power margins.[1] Experimental verification in an actual regeneratively cooled engine is required, however, before serious consideration can be given to this alternative cooling technique.

Another concern with oxygen-regenerative cooling is that a small crack may behave like an oxy-acetylene cutting torch and cause a catastrophic failure. This is not expected to occur, because in a typical regeneratively cooled design the wall temperature is below its kindling temperature and thus cannot be ignited by the leaking oxygen. Again, this model needs an experimental demonstration in an actual rocket thrust chamber.

The objective of the second study described in this paper is to experimentally demonstrate LOX cooling in high-pressure

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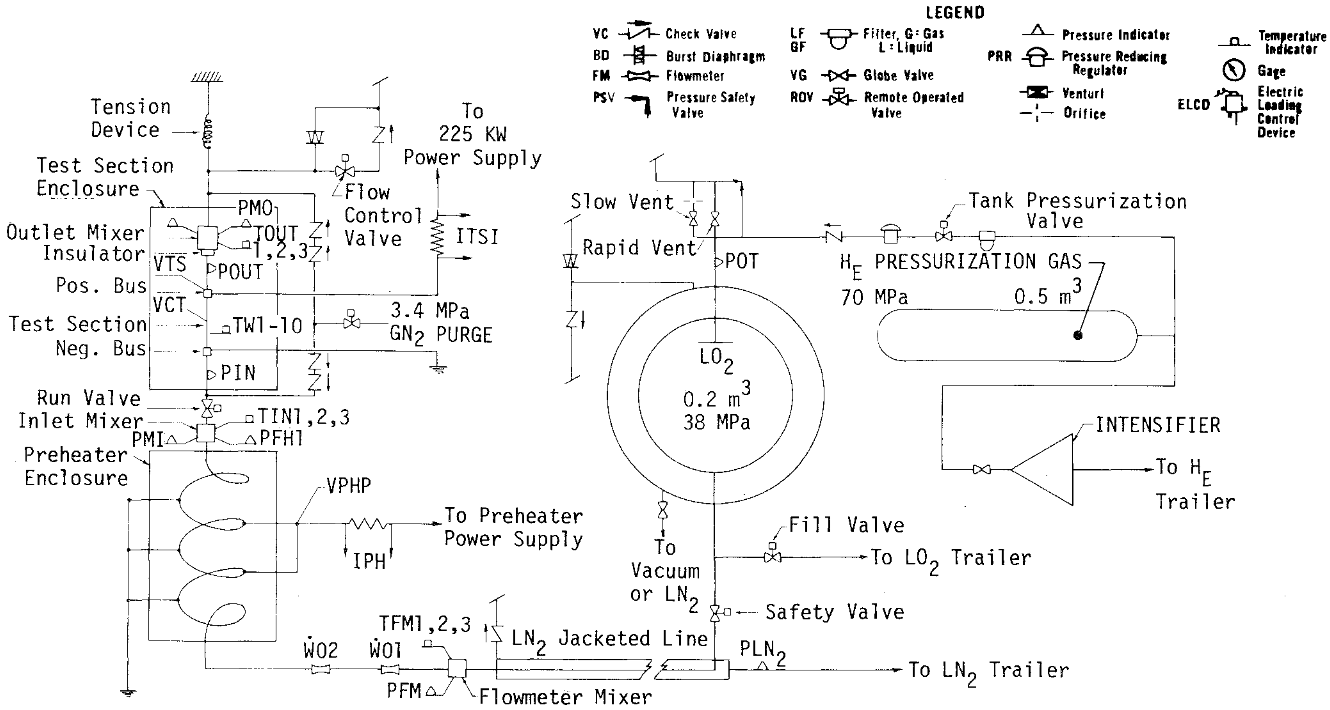

Fig. 1 Supercritical oxygen heat transfer heat schematic.

Table 1 Test data conditions

Data source	Pressure, MPa (psia)	T_{bulk}, K (R)	T_{wall}, K(R)	Heat flux, MW/m^2 (Btu/in.^{2} s)	$Re \times 10^{-6}$	L/D	No. of measurements
ALRC/NASA[5]	17-34.5 (2000-5000)	96-217 (173-391)	122-952 (220-1714)	2-90 (1.2-55)	0.15-3.2	7-63	472
ALRC[4]	24.2-34.5	98-162	150-989	2.1-64	0.21-1.3	4-40	91[a]
JPL[6]	1.75-7.41 (254-1075)	106-566 (191-1019)	333-1000 (600-1800)	0.3-14.2 (0.2-8.7)	0.15-2.1	5-167	81[b]

[a] Forty measurements correlated in Ref. 4 were not used to develop the recommended correlation because of the approximate method used to calculate local test section pressure (Tests HT-14-104, HT-14-105).
[b] "Interpolated data" of Ref. 6 (Table 20).

liquid engines. LOX-cooled thrust chambers have been designed using the recent correlation and are currently being fabricated. A test program is planned for evaluating the LOX-cooled design concept. The purpose of these tests is to verify the LOX cooling correlation in an actual thrust chamber and to demonstrate the effects of a LOX coolant leak.

Experimental Heat Transfer Study

Experimental Approach

Heat transfer to supercritical oxygen flowing through electrically heated tubes was measured, using the apparatus shown schematically in Fig. 1. The test sections were fabricated from Monel K-500 and Inconel 625 tubes with outside diameters of 3.18 and 4.76 mm (0.125 and 0.188 in) and heated lengths ranging from 51 to 254 mm (2 to 10 in.). All test sections had 0.38 mm (0.015 in.) thick walls. The test sections were heated directly with a 225 kW dc power supply. A separately powered preheater was used to control the oxygen temperature at the test section inlet. The test sections were instrumented to measure the oxygen temperature, pressure, and flow rate at the inlet, the temperature and pressure at the outlet, and the temperature along the outer wall.

The tests were conducted by flowing oxygen through the test section at a predetermined rate and pressure. The heat input was increased in steps and data recorded at each step until the oxygen supply was consumed or the tube failed.

Test Results

Figure 2 shows the range of pressures and heat fluxes tested. For each point on Fig. 2, outer tube wall temperature was measured at up to five places along the tube, resulting in 472 independent heat-transfer coefficient measurements at the range of test conditions noted on Table 1 and Fig. 3. Detailed tabulations of the test results are presented in Ref. 5.

Previous Test Results

A simular but less extensive series of high-pressure tests was conducted previously using the same test facility.[4] Heat transfer coefficient measurements were obtained in the 24.2-34.5 MPa (3510-5000 psia) pressure range at heat fluxes up to 64 MW/m^2 (39 Btu/in.^{2} s). Lower pressure data were obtained at the Jet Propulsion Laboratory at pressures up to 7.41 MPa (1075 psia) and heat fluxes up to 14.2 MW/m^2 (8.7 Btu/in.^{2} s).[6] The test conditions for these previous results are noted on Table 1 and Fig. 3. Detailed tabulations of the Ref. 4 and 6 data are presented in Ref. 5.

In the previous study,[4] a linear regression technique was used to correlate 90% of the low-pressure JPL data within ± 30% with the following equation:

$$Nu_b = 0.0054\, Re_b^{0.95} Pr_b^{0.4} \left(\frac{\mu_b}{\mu_w}\right)^{1.216} \left(\frac{k_b}{k_w}\right)^{-0.746}$$

$$\times \left(\frac{\rho_b}{\rho_w}\right)^{-588} \left(\frac{\bar{C}_p}{Cp_b}\right)^{0.514} \tag{1}$$

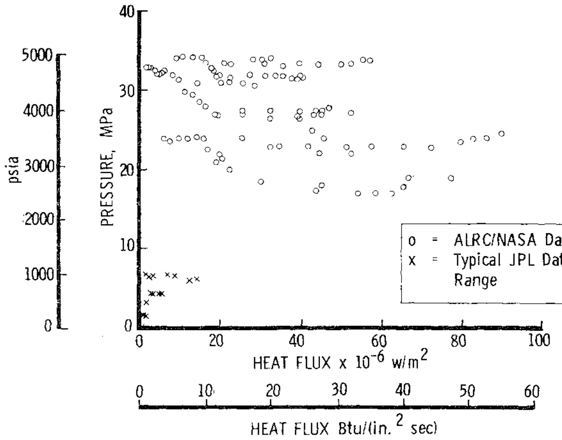

Fig. 2 Range of pressure and heat flux tested.

Attempts to correlate the high-pressure data in this manner yielded a bulk temperature effect prediction which appeared peculiar. Consequently, a correlation for high pressures was devised by empirically correcting the JPL data correlation with an F_p factor to better predict the limited high-pressure data:

$$F_p = 4.66(P/P_{cr})^{-1.06}$$

This preliminary design correlation is now superceded by the correlation recommended in the following section.

Data Correlation

To develop a heat transfer correlation, an equation of the following form was assumed:

$$Nu = KRe^a Pr^b \left(\frac{\rho}{\rho_w}\right)^c \left(\frac{\mu}{\mu_w}\right)^d \left(\frac{k}{k_w}\right)^e \times \left(\frac{\bar{C}p}{Cp}\right)^f \left(\frac{P}{P_{cr}}\right)^g \left(1 + \frac{2}{L/D}\right) \tag{2}$$

Equation (2) is an expanded version of the correlation approach originally published in Ref. 7 and subsequently applied to supercritical ClF_5 and O_2 at ALRC. The critical pressure ratio term was added, based on the previous supercritical oxygen work.[4] The entrance effect term was suggested during recent work with supercritical nitrogen data.[8]

A multiple regression computer program was used to find the coefficient and exponents for Eq. (2) which best fit the data. A total of 27 correlating equations were evaluated.

The major conclusions reached on the basis of these correlating equations and the corresponding statistical results are summarized by the following statements:

1) Bulk temperature properties correlate the data better than film temperature properties.
2) The P/P_{cr} term increases data correlation.
3) The L/D term increases data correlation.
4) The viscosity ratio term is not statistically significant for correlating supercritical oxygen data.
5) The specific heat ratio is not statistically significant for correlating the high-pressure oxygen data alone, i.e., $P > 24$ MPa (3500 psia). However, it is an important parameter if the lower-pressure data are included.
6) A Reynolds number exponent near 1.0 correlates the supercritical oxygen data better than the traditional value of 0.8.

The Reynolds number exponent was initially fixed at 0.8 and the Prandtl number exponent at 0.4 so that the resulting equation would approach the standard Dittus-Boelter equation as wall temperature approached the bulk temperature. A Reynolds number exponent of 0.95 suggested by Hines[9] was also tried and found to give a better fit to the data. When the exponent was a variable determined by the multiple regression computer program, the low-pressure data obtained by Powell was fit best with an exponent of 0.93. When the high-pressure ALRC data were included, the best fit was obtained with an exponent of 1.03. For the final correlation, the Re exponent was rounded off to 1.0. No attempt was made to improve the data correlation through variation of the Pr exponent. The final recommended correlation based on all of the Table 1 data is

$$Nu_b = 0.0025\, Re_b Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-1/2} \left(\frac{k_b}{k_w}\right)^{1/2} \times \left(\frac{\bar{C}p}{Cp_b}\right)^{+} \left(\frac{P}{P_{cr}}\right)^{-1/5} \left(1 + \frac{2}{L/D}\right) \tag{3}$$

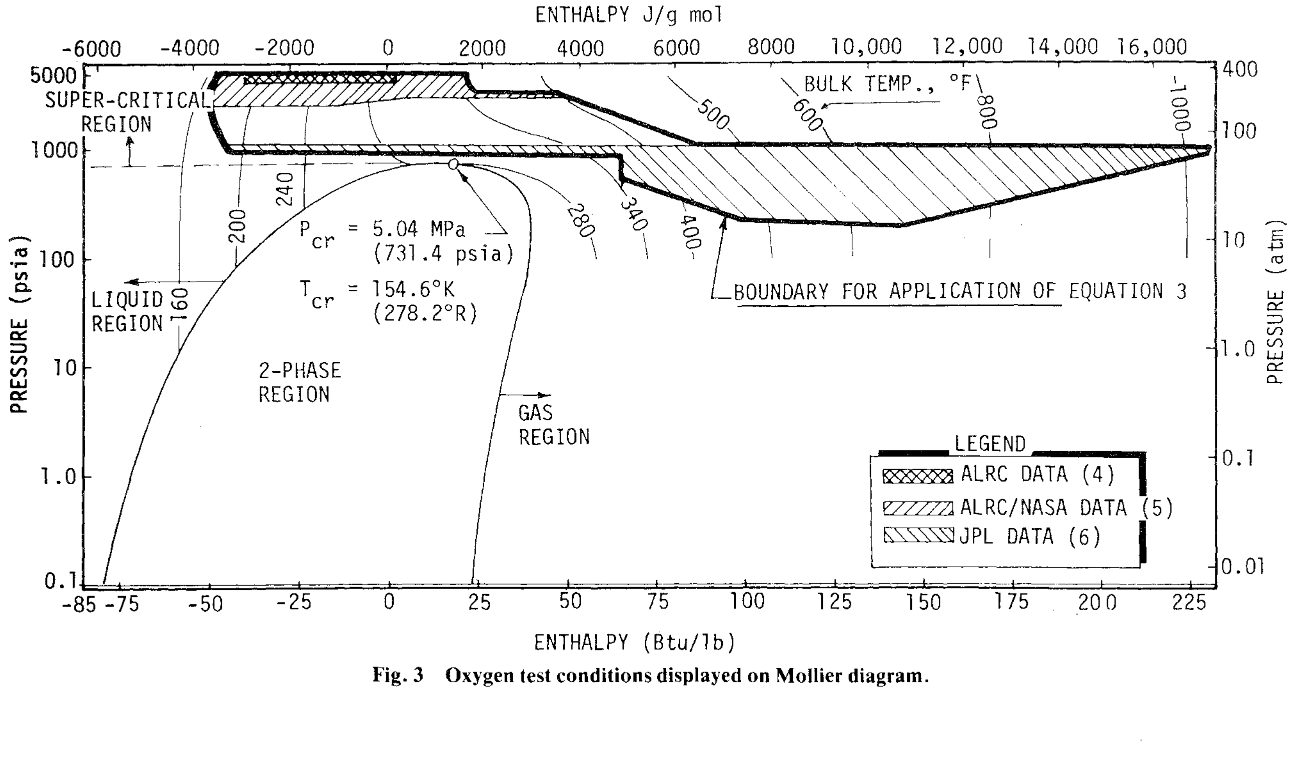

Fig. 3 Oxygen test conditions displayed on Mollier diagram.

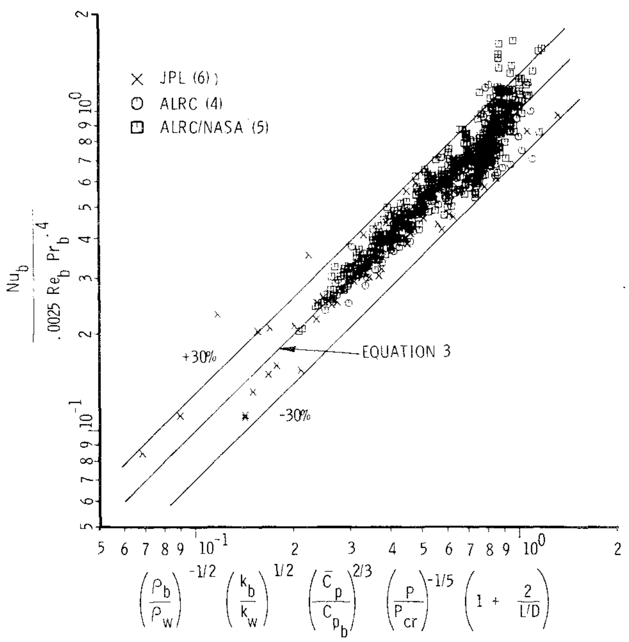

Fig. 4 Recommended correlation.

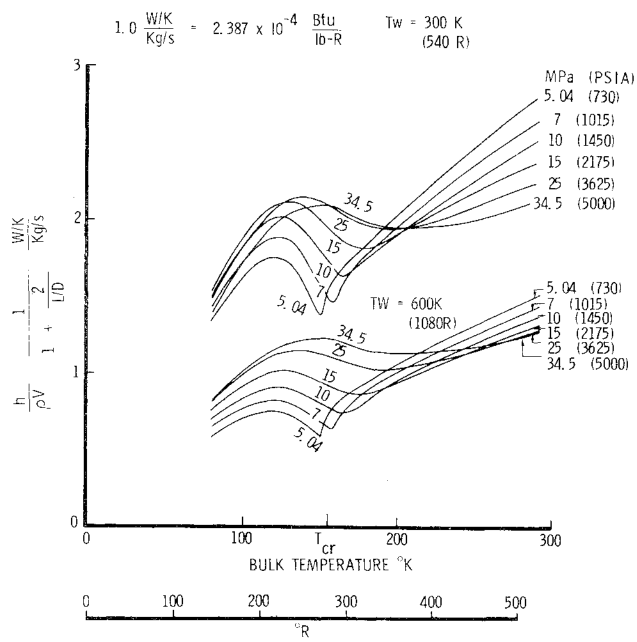

Fig. 6 Predicted heat-transfer coefficient variation.

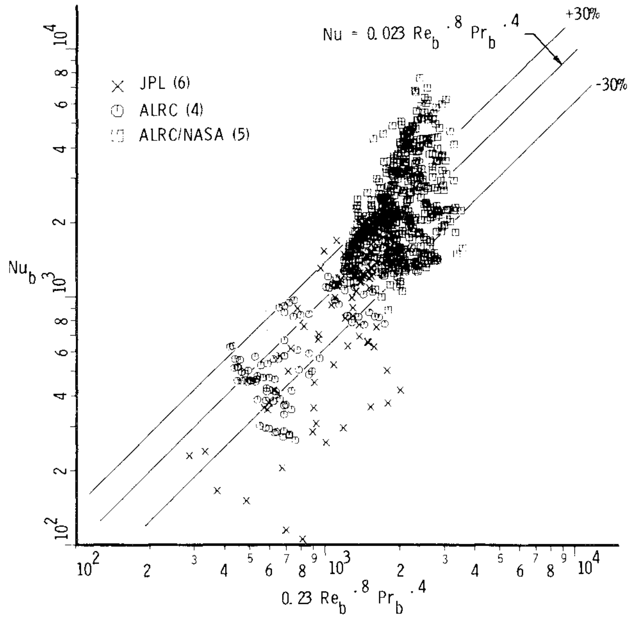

Fig. 5 Comparison of oxygen data and Dittus-Boelter correlation.

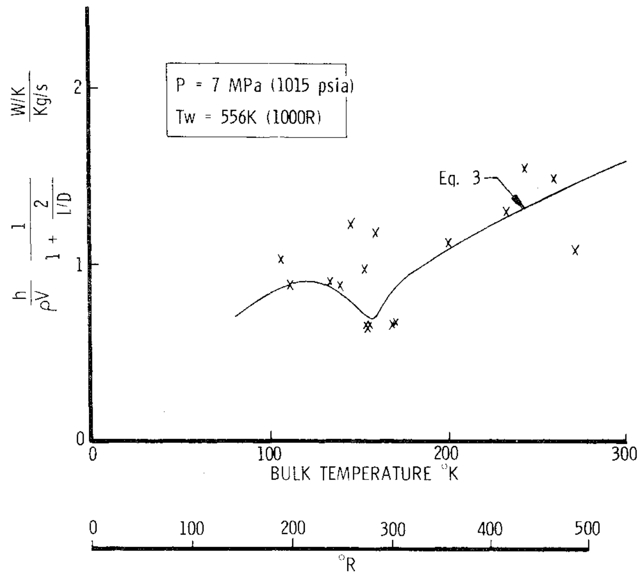

Fig. 7 Predicted vs measured heat transfer at low pressure.

Table 2 NASA Lewis test plan

Test series	Objective	Injector propellants	Chamber pressure, MPa (psia)	Coolant
I	Verify Eq. (3) in rocket engine cooling channels at low heat flux	O_2/H_2	4.137 (600)	Supercritical LOX
II	Develop cooling channel crack, observe effect	O_2/H_2	4.137 (600)	Supercritical LOX
III	Verify Eq. (3) in rocket engine cooling channels at high heat flux	O_2/H_2	13.79 (2000)	Supercritical LOX
IV	Determine local *hg* and effect of O/F on carbon layer development	O_2/RP-1	up to 13.79 (2000)	Supercritical LOX
V	Calorimetric measurement of heat flux profile	O_2/RP-1	4.137+ (600+)	Water

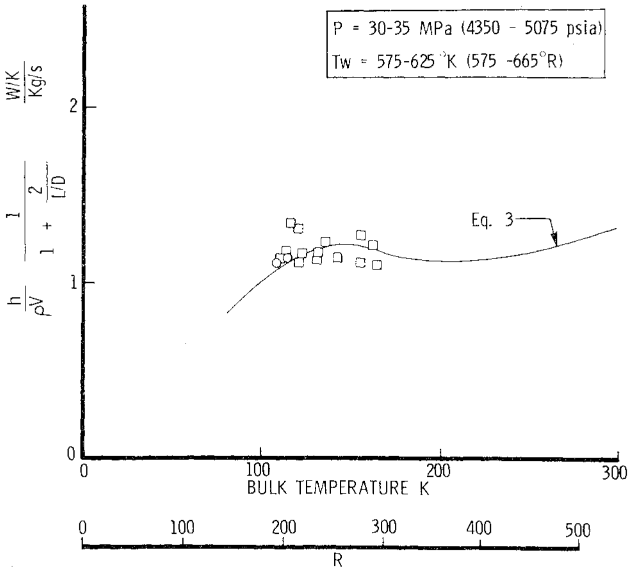

Fig. 8 Predicted vs measured heat transfer at high pressure.

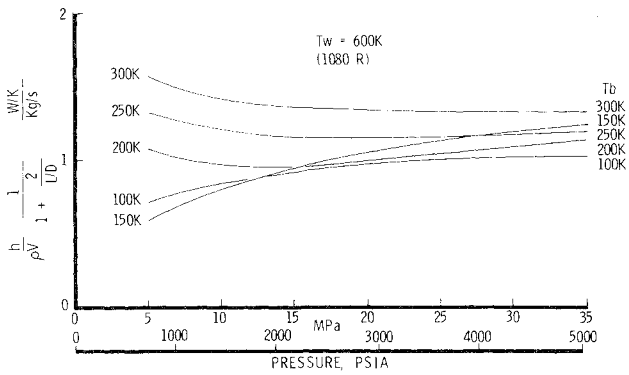

Fig. 9 Predicted heat transfer coefficient variation with pressure.

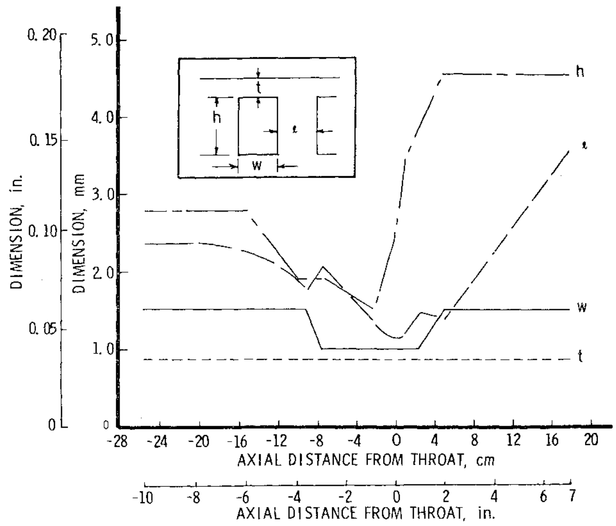

Fig. 10 Combustor wall and channel dimensions.

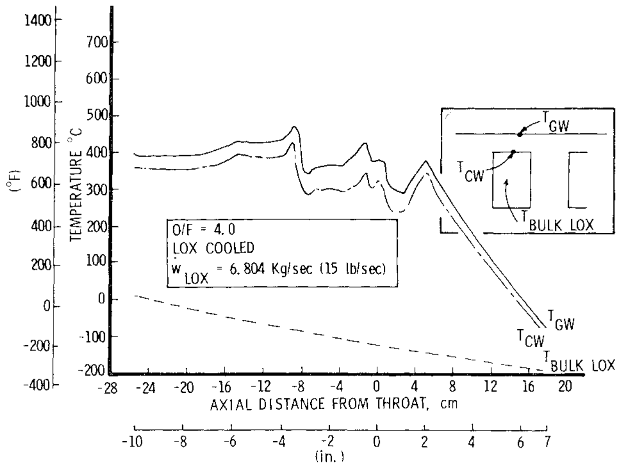

Fig. 11 Temperature distribution with O_2/H_2 propellants at 4.137 MPa (600 psi) P_c.

This equation predicts over 95% of all the experimental data within ±30%. The accuracy of the prediction is shown in Fig. 4, which is a plot of all the 644 individual measurements that were correlated. The range of pressure and bulk temperature conditions where Eq. (3) is applicable is indicated on Fig. 3.

Equation (3) is not a simple equation to apply. Its use is justified simply because it predicts the data better than the more standard forced convection correlations such as the Dittus-Boelter equation which is compared to the oxygen data on Fig. 5. The Dittus-Boelter equation predicts only 50% of the data within ±30% and deviations on the order of +100-−70% are not uncommon. Incorporation of the L/D term of Eq. (3) does not appreciably change this comparison.

Equation (3) can be rewritten to put all the fluid properties on the right side, leaving the heat transfer coefficient, mass flux, and entrance term on the left:

$$\frac{h}{\rho V}\left(1+\frac{2}{L/D}\right)^{-1}=0.0025\left(\frac{k_b}{\mu_b}\right)^{0.6}Cp_b^{0.4}\left(\frac{\rho_b}{\rho_w}\right)^{-1/2}$$
$$\times\left(\frac{k_b}{k_w}\right)^{1/2}\left(\frac{\bar{C}p}{Cp_b}\right)^{2/3}\left(\frac{P}{P_{cr}}\right)^{-1/5} \qquad (4)$$

This can be plotted to show the variation in heat transfer as a function of bulk temperature, wall temperature, and pressure, as in Fig. 6. Figure 6 indicates a sharp drop in heat transfer near the critical pressure and temperature. The JPL data indicate this general trend, although there is considerable data scatter near the critical point, as shown in Fig. 7. The scatter may be due to the rapidly changing properties near the critical point which makes accurate measurements difficult. At higher pressures, the data are more tightly grouped, as in Fig. 8.

Figure 6 also indicates a decrease in heat-transfer coefficient below 100 K (180°R) bulk temperature. There is insufficient data in this region to substantiate this trend, however, and more tests are required in this temperature range.

Equation (4) is plotted to show the variation in heat transfer as a function of pressure in Fig. 9. As the pressure increases, the heat transfer coefficient appears to approach a constant. On this basis, extrapolation to higher pressures appears justified; however, additional higher-pressure data is needed to verify this indication.

Figure 3 shows that a "data gap" exists in the 7.6-17.2 MPa (1100-2500 psia) pressure range. It is expected that the Eq. (3) predictions are reasonable for this pressure range; however, additional data is also recommended for this region if a high heat flux cooling application is ever contemplated.

LOX-Cooled Thrust Chamber Demonstration

Scope and Objectives

The scope of the test program is summarized in Table 2. The objective of this program is to provide experimental data on the following: 1) LOX-cooling feasibility, 2) coolant-side

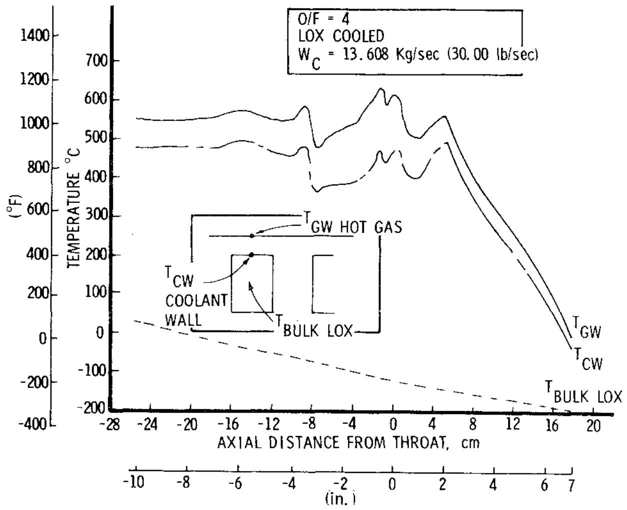

Fig. 12 O_2/H_2 temp. distribution at 10.342 MPa (1500 psi) P_c.

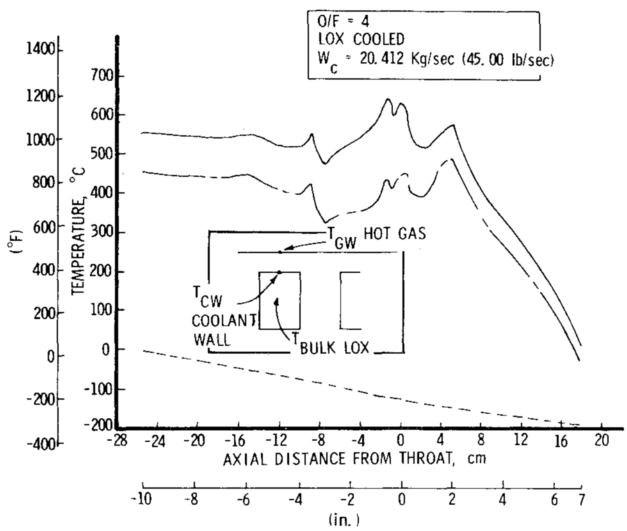

Fig. 13 O_2/H_2 temp. distribution at 13.790 MPa (2000 psi) P_c.

LOX heat-transfer coefficient in an actual rocket engine thrust chamber, and 3) effects of a small crack from the cooling channel into the combustion region.

Although LOX cooling is intended for hydrocarbon-fueled systems, the initial tests of this program will be conducted with an O_2/H_2 thrust chamber because the hot-gas-side heat transfer coefficient is better characterized in the type of hardware that will be used. After the initial tests to verify the coolant-side heat transfer coefficient of LOX, testing will proceed until a cooling channel develops a through crack into the combustion zone. This latter test series will be performed to determine if a catastrophic failure of the thrust chamber will result because of the uncombusted LOX passing into the combustion zone. It is not anticipated that such an incident will happen, because as long as the wall material is maintained at a temperature below its kindling temperature, 825°C (1517°F) at 10.1 MPa (100 atm) O_2 pressure, rapid material oxidation cannot take place. It is hypothesized that the oxygen flowing through the crack will provide film cooling which will keep the wall temperature below its kindling temperature.

Upon completion of this portion of the program, tests will be conducted with O_2/RP-1 as the propellants and LOX as the coolant. New chambers of the same coolant passage design will be used; however, the length of combustion chamber will be increased to obtain high performance with

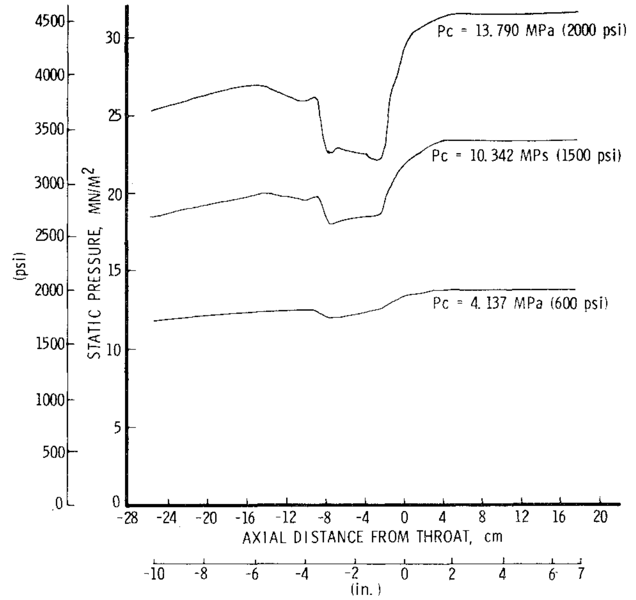

Fig. 14 Coolant static pressure distribution, O_2/H_2 propellants, O/F = 4, LOX-cooled.

the LOX hydrocarbon propellant combustion. The purpose of the O_2/RP-1 tests are to determine the hot-gas-side heat transfer coefficient and carbon layer buildup in a LOX/hydrocarbon thrust chamber operating at chamber pressures up to 13.79 MPa (2000 psia).

Thrust Chamber Design

The design of the thrust chamber cooling passages is shown in Fig. 10. This figure shows the cooling channel dimensions, gas-side hot-wall thickness, and metal thrickness between channels as functions of the axial distance from the chamber throat. Figures 11-13 show the calculated combustor wall temperatures and LOX coolant temperature distributions at 4.137, 10.34, and 13.79 MPa (600, 1500, and 2000 psia) chamber pressure at a mixture ratio of 4 with O_2/H_2 as the propellants. Calculated combustor coolant static pressure distribution with respect to axial length is shown in Fig. 14. The wall temperature calculations were made using the Eq. (3) correlation for LOX heat transfer coefficient.

All of the preceding figures are plotted for a chamber length from injector face to throat of 0.254 m (10 in.). These chambers will be used in tests conducted with O_2/H_2 propellant combinations. If it is necessary to use longer chambers to achieve high performance with O_2/RP-1 propellants, the cylindrical portion of the chamber will be extended 0.1016 m (4 in.). The cooling channel dimensions (passage height and width) will remain constant in this extension.

Concluding Remarks

Supercritical LOX Heat Transfer Correlation

Heat transfer to supercritical oxygen has been investigated in a series of heated tube heat transfer tests at high pressure. Evaluation of these data[5] and the previously existing data[4,6] using a multiple regression analysis computer program has led to the following design correlation which is recommended for calculating supercritical oxygen heat transfer coefficients:

$$Nu_b = 0.0025\, Re_b Pr_b^{0.4} \left(\frac{\rho_b}{\rho_w}\right)^{-1/2} \left(\frac{k_b}{k_w}\right)^{1/2} \times \left(\frac{\bar{C}p}{Cp_b}\right)^{2/3} \left(\frac{P}{P_{cr}}\right)^{-1/5} \left(1+\frac{2}{L/D}\right) \tag{5}$$

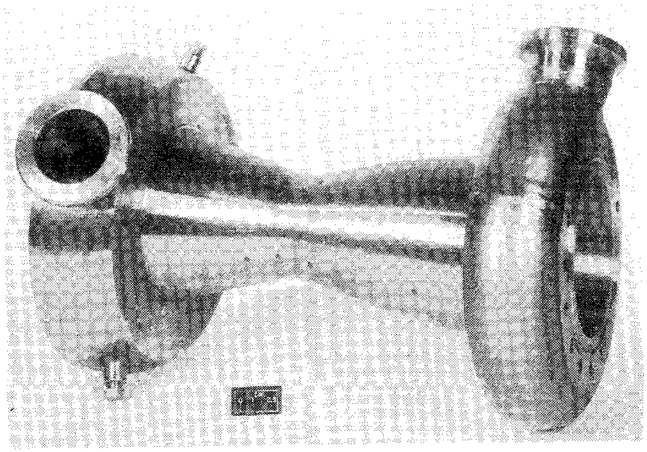

Fig. 15 LOX-cooled thrust chamber.

This correlation is applicable over the range of pressure and bulk temperature conditions indicated on Fig. 3.

LOX-Cooled Thrust Chamber Demonstration

The primary objective of the NASA Lewis test program is to demonstrate the feasibility of using supercritical LOX as a rocket engine regenerative coolant. The recommended LOX coolant-side heat transfer coefficient correlation will be verified in actual thrust chamber hardware instrumented with thermocouples as described in Ref. 10. In addition, the program will demonstrate the effect of a small through crack from the cooling channel into the combustion zone, and its effect on the structural integrity of the thrust chamber.

LOX regenerative cooling will be demonstrated with two different propellant combinations, O_2/H_2 and O_2/RP-1, and at two chamber pressure operating conditions, 4.137 and 13.79 MPa (600 and 2000 psia). The six thrust chambers which will be used for the O_2/H_2 portion of the program are in the final stages of preparation. One of these chambers is shown in Fig. 15. Testing of these chambers at the Lewis Research Center was initiated in 1979. Following the completion of this phase of the program, a series of tests will be conducted with O_2/RP-1 as the propellants and LOX as the coolant. The first phase was completed in September. A total of five chambers will be tested with O_2/RP-1. In addition to LOX regenerative cooling data, this second test series will provide data on the hot-gas-side heat transfer coefficient and carbon layer development at two different chamber pressures and over a range of mixture ratios.

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