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COMBUSTION PERFORMANCE AND HEAT TRANSFER CHARACTERIZATION OF LOX/HYDROCARBON TYPE PROPELLANTS

FINAL SUMMARY REPORT

September 1983

Prepared For:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Lyndon B. Johnson Space Center
Contract NAS-9-15958

Prepared By:

AEROJET LIQUID ROCKET COMPANY

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Prepared For

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FOREWORD

The Aerojet Liquid Rocket Company (ALRC) submits this summary report as a part of the contract NAS 9-15958, Combustion Performance and Heat Transfer Characterization of LOX/Hydrocarbon Type Propellants. It is a condensation of the program final report, Reference 1. The program was also documented while in progress by means of three comprehensive data dumps for each of the three tasks and by monthly progress reports.

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ABSTRACT

This program, Combustion Performance and Heat Transfer Characterization of LOX/Hydrocarbon Type Propellants, Contract NAS 9-15958, was undertaken to evaluate liquid oxygen and various hydrocarbon fuels as low cost alternative propellants suitable for future space transportation system applications. The emphasis of the program is directed toward low earth orbit maneuvering engine and reaction control engine systems.

The feasibility of regeneratively cooling an orbit maneuvering thruster was analytically determined over a range of operating conditions from 100 to 1000 psia chamber pressure and 1000 to 10,000-lbF thrust, and specific design points were analyzed in detail for propane, methane, RP-1, ammonia, and ethanol; similar design point studies were performed for a film-cooled reaction control thruster.

Heat transfer characteristics of propane were experimentally evaluated in heated tube tests. Forced convection heat transfer coefficients were determined over the range of fluid conditions encompassed by 450 to 1800 psia, -250 to +250°F, and 50 to 150 ft/sec, with wall temperatures from ambient to 1200°F, and heat fluxes to 10 Btu/in.^{2}sec. Nucleate boiling and coking were also evaluated.

Seventy-seven hot firing tests were conducted with LOX/propane and LOX/ethanol, for a total duration of nearly 1400 seconds, using both heat sink and water-cooled calorimetric chambers. Combustion performance and stability and gas-side heat transfer characteristics were evaluated. Four injectors were tested: two with conventional like-on-like doublet and OFO triplet elements, and two with unconventional platelet elements. Film cooling was also assessed. The combustion chamber was sized for a nominal thrust of 1000-lbF at 300 psia chamber pressure, and testing spanned a significant range of chamber pressure and propellant mixture ratio conditions.

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I. INTRODUCTION

A. PROGRAM OBJECTIVES

The objectives of this program were to evaluate and characterize candidate liquid oxygen/hydrocarbon fuel combinations, and to establish a technology base for these propellants that would guide the selection of hydrocarbon fuels in future space transportation system applications.

While the program results are pertinent to any size liquid rocket engine, the program was directed toward that thrust range representative of the current Reaction Control System (RCS) and Orbit Maneuvering System (OMS) engines on the Space Shuttle.

The current RCS and OMS propellants -- nitrogen tetroxide and monomethyl hydrazine -- have several drawbacks: high cost, potential unavailability due to limited manufacture, formation of carcinogenic intermediates during manufacture, toxicity, handling difficulties, and associated handling requirements.

The current storable propellant combination was selected over liquid oxygen/liquid hydrogen, which offered much higher performance but was constrained by the volume requirements of the fuel, as well as over liquid oxygen/hydrocarbon fuel alternatives, for which the technology base was generally lacking. The storable propellants had a large technology base, and the simple pressure-fed engine systems promised high reliability and minimal development cost.

Engine development cost and recurring operational costs are key factors in the overall cost of a space transportation system. Low-cost easily handled propellants, typified by oxygen/hydrocarbons, and reusable engine systems combine to minimize operational costs. Development costs can, in part, be minimized by the judicious selection of the propellants; that selection presupposes a substantial technology base. The intent of this program is to contribute to such a base.

B. PROGRAM SUMMARY

The program was conducted over a forty month period, beginning in October 1979. It consisted of three major task areas; as described below. These task areas are documented in three comprehensive data dumps, References (2), (3), and (4).

TASK I - REGENERATIVE COOLING CHARACTERIZATION

This task comprised two subtasks. First, forced convection and nucleate boiling heat transfer data and correlations available in the literature for candidate hydrocarbon fuels were reviewed. Those candidates included propane, methane, RP-1, and ammonia. Regenerative chamber cooling analyses were then conducted to compare the cooling capabilities of each fuel and determine the operating point (thrust and chamber pressure) limits imposed thereby. Second, heated tube tests were performed to determine the heat transfer characteristics and the coking behavior of propane, both commercial grade and instrument grade.

TASKS II AND IV - SUBSCALE INJECTOR CHARACTERIZATION

Tasks II and IV involved the design, fabrication, testing and data analysis of subscale hardware, i.e., nominal thrust of 1000-lbF, to evaluate the combustion performance, stability, and gas-side heat transfer characteristics of liquid oxygen/hydrocarbon propellants. Four injector patterns were tested, including conventional OFO triplets and like-on-like doublets, and unconventional platelet patterns in which fuel swirler elements were located within pairs of drilled orifice or splashplate oxidizer elements. Heat sink and water-cooled calorimeter chambers were utilized, and a removable chamber section was used with the former to allow evaluation of chamber length effects. A fuel film coolant ring was used in conjunction with the triplet and platelet injectors. An adjustable accoustic cavity section provided combustion stability.

Seventy-seven tests were conducted, with a total duration of approximately 1370 seconds. Both propane and ethanol were tested, the latter with gaseous as well as liquid oxygen. Chamber pressure and mixture ratio were varied widely to assess operating point effects.

TASK III - PRELIMINARY ENGINE SYSTEM CHARACTERIZATION

In Task III numerous engine operating points were analyzed to determine engine performance and weight figures for orbit maneuvering and reaction control system thrusters. The work built upon the regenerative cooling studies of Task I, updated for the propane heat transfer correlation derived empirically in that task, and extended to include turbomachinery for pump-fed systems, alternative chamber materials for the orbit maneuvering thruster, and film cooling for the reaction control thrusters. Thruster envelopes were defined by the current engines on the Space Shuttle.

C. PROGRAM CONTRIBUTIONS TO NASA OBJECTIVES

This program significantly enlarges the technology base for LOX/hydrocarbon propellants and is an important step towards a LOX/hydrocarbon auxiliary propulsion system. A number of additional steps is obviously necessary for that system to become a reality.

I, C, Program Contributions to NASA Objectives (cont.)

Specific results and conclusions developed in the program are summarized below. The extensive experience gained in the design, analysis, and testing of hardware for these propellants also contributes to the technology base but cannot be readily quantified.

Hot fire testing went smoothly and was quite successful. High combustion performance was achieved with conventional as well as unconventional injector elements and stable combustion was readily obtained with acoustic cavities. However, chamber gas-side heat fluxes were considerably higher than values based on standardized predictive methods. Apart from this, there were no big surprises, and the design of high performance, stable, regeneratively-cooled thrust chambers does not appear to present any unusual or insurmountable difficulties.

Perhaps the biggest disapppointment -- in terms of using LOX/hydrocarbon propellants for the APS was the low wall temperature threshold determined for coking of propane. This, combined with propane's incompatibility with copper, the material of choice for high pressure regeneratively cooled chambers because of its high thermal conductivity, may eliminate propane as a candidate propellant. This would be unfortunate, because propane otherwise offers a desirable combination of high combustion performance and high mass density.

On the analytical side, the engine point designs generated in this program, in conjunction with the system point design studies conducted in Reference (5) -- to which the Task III results were input -- strongly support any future selection of propellant, operating point, engine cycle, and degree of system integration. The approach here was to first consider the flow and pressure drop requirements of the thrust chamber and injector and then work upstream to the turbopump requirements and/or tank conditions, overall engine performance and weight, and finally in the Reference (5) program to system optimization.

II. RESULTS AND CONCLUSIONS

A. TASK I - REGENERATIVE COOLING CHARACTERIZATION

1. The parametric regenerative cooling analysis showed the following for the four candidate fuels:

(a) Methane: either vapor phase or supercritical pressure fluid is an acceptable coolant at higher thrust levels over the entire range of chamber pressure without the need for additional film-cooling. Subcritical pressures are unacceptable because of the limited subcooling.

(b) Propane: either vapor phase or supercritical pressure fluid is acceptable at higher thrust levels without additional film cooling. Subcritical pressures are unacceptable because of low burnout heat flux.

(c) RP-1: because of low coking temperature, RP-1 is not a satisfactory coolant.

(d) Ammonia: either liquid (nucleate boiling) or vapor phase is acceptable.

2. Sufficient heat can be picked up in the nozzle to vaporize the fuel -- in the case of methane and propane only -- to allow vapor-phase cooling of the combustion chamber.

3. Heated-tube testing of propane resulted in a forced convection correlation that grouped 95% of the data within $\pm 24\%$. Limited film and nucleate boiling data were obtained; burnout heat flux was found to be considerably higher than an extrapolation of available low flux data would predict.

4. Coking in the heated tube tests occurred at wall temperatures less than 500°F; coking rate was comparable to published data for RP-1. Propane purity affected the rate but not the threshold temperature of coking.

B. TASKS II AND IV - SUBSCALE INJECTOR CHARACTERIZATION

1. The like-on-like injector pattern was fired with LOX/propane in a heat-sink chamber and found to be low-performing, as a result of both poor atomization and poor mixing. The combustion was bomb-stable.

2. The OFO triplet injector was fired with both LOX/propane and LOX/ethanol in both heat-sink and water-cooled calorimeter chambers. In the calorimeter chamber it was tested with and without fuel film-cooling. Performance was very high with LOX/propane, for which the unit was designed, and slightly lower with LOX/ethanol due to non-optimum propellant momentum match. Combustion was stable with both propellant combinations.

3. One platelet injector was designed for liquid-phase injection of LOX/ethanol; the injector pattern consisted of a swirler fuel element within two splashplate oxidizer elements. Although this unit achieved high performance, propellant blowapart apparently occurred, causing the outer periphery to be oxidizer-rich. The addition of fuel film-coolant increased the gas-side heat flux as well as injector performance.

4. The other platelet injector was designed for gaseous oxygen (GOX)/ethanol injection. The pattern consisted of a fuel swirler element within two drilled oxidizer orifices. This injector achieved high performance with ambient temperature propellants and slightly reduced performance at low (-130°F) temperature.

5. Throat heat fluxes experienced with ethanol were considerably higher than would be predicted with the standardized pipe-flow correlation. The inferred correlating coefficient (Cg) was approximately 70% higher than would be expected for storable propellants. The correlating coefficient for ethanol was found to be extremely sensitive to mixture ratio.

6. Carbon deposition in the acoustic cavities with LOX/propane was extensive to the point that acoustic damping capabilities could be lost. Film-coolant injection from the forward end of the cavities reduced the amount of carbon deposition within the cavities.

7. Carbon deposition on the chamber wall occurred only with LOX/propane and was largely lost during the start and/or shutdown transients. Engine restart was marked by a return to clean-wall heat flux conditions, followed by a progressive decay as the deposition layer increased. As a result, the thermal resistance of the deposition layer cannot be assumed for design purposes to limit gas-side wall temperatures to less than clean-wall values.

8. Carbon deposition was negligible with LOX or GOX/ethanol. The exhaust plume was clear whereas with LOX/propane it was not.

C. TASK III - PRELIMINARY ENGINE SYSTEM CHARACTERIZATION

1. Design point analyses for ten different concepts (propellant combinations and operating points) involving a pressure-fed regeneratively-cooled orbit maneuvering engine showed the following:

(a) Methane, with vapor-phase cooling, offers the highest specific impulse.

(b) Propane performance, with vapor-phase cooling, is nearly as high but is severely degraded with liquid-phase cooling due to high film-cooling requirements.

(c) Ethyl alcohol requires no film cooling but the performance is lower than that of liquid propane.

2. Analyses of twenty-eight concepts involving a pump-fed, regeneratively-cooled orbit maneuvering engine showed the following:

(a) The highest performance is again obtained for methane.

(b) Performance with propane is slightly lower.

(c) Performance of all twelve methane and propane concepts is within a range of 10 sec Isp, over a large range of thrust and chamber pressure.

(d) Ethyl alcohol performance is lower than that of methane or propane, and the performance of ammonia is only slightly higher than that of a pressure-fed storable propellant engine.

(e) In light of the propane/copper compatibility issue, nickel was examined as an alternative (to copper) chamber wall material and is found suitable to about 400 psia chamber pressure without the use of film-cooling.

(f) Regenerative cooling with liquid oxygen is feasible at high chamber pressures, if required because of fuel-cooling limitations.

(g) Subcooling the propane could eliminate the need for boost pumps.

3. Analyses of twelve concepts for the film-cooled reaction control engine and vernier engine showed the following:

(a) The trend of performance for the candidate fuels is similar to that for regeneratively cooled thrusters: methane, propane, ethyl alcohol, and ammonia.

(b) Film-coolant requirements center around 20% of the fuel for the reaction control thruster regardless of fuel or chamber pressure.

III. RECOMMENDATIONS

A. Investigate the causes of propane coking -- impurities, catalytic effects, etc.

B. Develop solutions to the incompatibility of propane and copper, such as coatings, alloys, fuel additives, etc.

C. Characterize coking thresholds and heat transfer of methane and ethanol.

D. Develop correlations for gas-side soot formation of LOX/methane and LOX/propane.

E. Characterize gas-side heat transfer for these propellants (typically higher heat transfer rates are measured than would be predicted with standard formulations). Also, characterize film-cooling behavior.

F. Address fuel-rich combustion behavior as applicable to gas generator and turbopump devices.

G. Evaluate the cost aspects and systems issues (handling, etc.) associated with LOX/hydrocarbon propellants.

H. Pursue the explanation for anomalous behavior observed during testing: (1) the requirement for higher oxidizer-to-fuel momentum ratios to achieve optimum performance in hot-fire tests than would be predicted on the basis of cold-flow test results; (2) the exceptionally high throat heat fluxes observed in the ethanol firings; (3) the increased carbon deposition effect noted with LOX/propane at higher mass flux (chamber pressure).

IV. TECHNICAL OVERVIEW

A. TASK I - REGENERATIVE COOLING CHARACTERIZATION

Task I consisted of two major subtasks: (1) Cooling Correlation and Comparison; (2) Experimental Heat Transfer Investigation.

The most important aspect of the first subtask was the determination of regenerative cooling feasibility over the range of operating conditions encompassing 1000 to 10,000 lbF thrust and 100 to 1000 psia chamber pressure for the four candidate fuels (propane, methane, RP-1, and ammonia). A total of seventy-four design points was investigated to characterize cooling feasibility.

The analyses were based on a 15 hour operating life in conjunction with a 2000 start cycle life requirement. Minimum coolant channel dimensions were limited to near-state-of-the-art manufacturing capability. The gas-side heat transfer coefficient formulation was based on laminar, transition, or turbulent flow correlations as pertinent. Carbon deposition on the gas-side was accounted for -- in the calculation of coolant bulk temperature rise only -- by the application of a multiplication factor to the clean wall heat flux; the following factors were considered: methane 0.765; propane 0.42; RP-1 0.25; ammonia 1.00. Film-cooling in addition to regenerative cooling was not considered. Coolant-side heat transfer was based on the ALRC oxygen correlation (Ref. 6) for supercritical pressure propane and methane; all other forced convection modes were represented by the Hines Equation (Ref. 7). Figure 1 shows the feasibility prediction for the four propellant combinations.

In the second subtask the heat transfer characteristics of propane were investigated. The objectives were to correlate the forced convection behavior at sub- and supercritical pressures, determine the nucleate boiling and burnout heat flux characteristics, and evaluate coking behavior at elevated wall temperatures. Twelve tests were conducted, exceeding 18,500 sec. duration and accumulating 840 individual data points.

Forced convection heat transfer coefficients were measured over the following ranges:

Pressure:	450 to 1800 psia
Bulk Temperature:	-250 to +250°F
Velocity:	50 to 160 ft/sec
Heat Flux:	0.2 to 10 Btu/in.^{2}sec

Nucleate boiling coefficients and critical heat fluxes were determined over the following ranges:

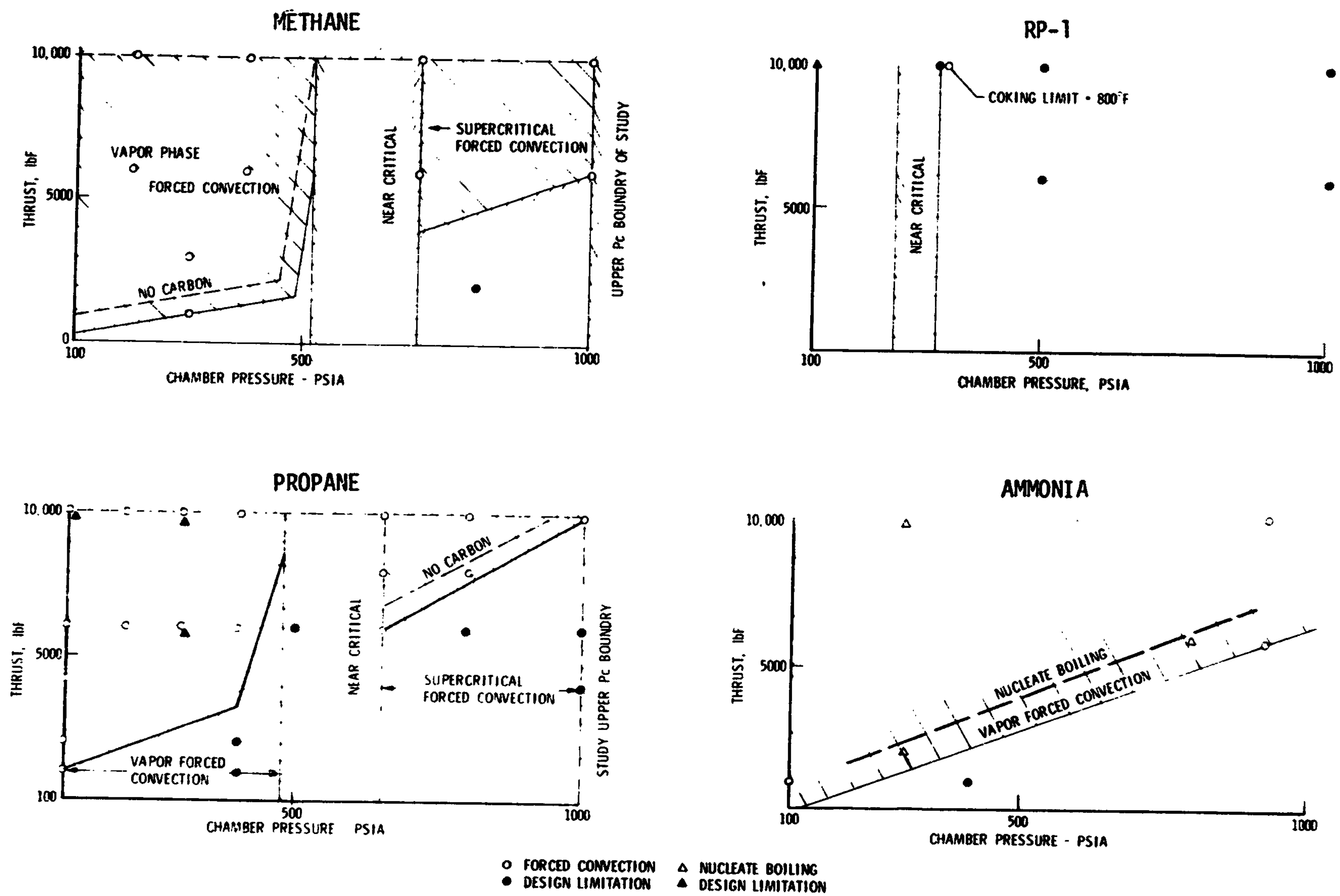

Figure 1. Cooling Feasibility Map

IV, A, Task I - Regenerative Cooling Characterization (cont.)

Pressure: 450 to 500 psia
Bulk Temperature: -240 to - 12°F

$V\Delta T_{sub}$: 20,000 to 40,000 F ft/sec

Coking was evaluated over the following ranges:

Pressure:	1800 psia
Bulk Temperature:	70 to 230°F
Wall Temperature:	350 to 1000°F
Velocity:	50, 150 ft/sec
Grade:	Instrument (99.5% purity) Natural (96% purity)

Test sections consisted of 5 to 10.5 in. Monel K500 tubes of 0.125 or 0.1875 in. O.D. and 0.015 in. thick wall. The tubes were electrically heated by means of a 225 KW DC power supply. Five spring-loaded thermocouples were located along the tube length and insulated from it by thin pieces of mica; the configuration had previously been calibrated against measured tube wall temperatures and the data were corrected accordingly. Figure 2 shows the test section installation.

Forced convection heat transfer data were correlated by using the following equation:

$$Nu_b = (K)\,(Re_b)^a\,(Pr)^c\left(\frac{\rho_b}{\rho_w}\right)^d\left(\frac{\mu_b}{\mu_w}\right)^e\left(\frac{k_b}{k_w}\right)^f\left(\frac{\overline{Cp}}{Cp_b}\right)^g\left(\frac{P}{P_{crit}}\right)^h\left(1+\frac{2}{L/D}\right)$$

where:

Nu	=	Nusselt number
Re	=	Reynolds number
Pr	=	Prandtl number
ρ	=	Density
μ	=	Viscosity
k	=	Thermal conductivity
Cp	=	Specific heat
K	=	Experimental determined constant
P	=	Pressure
P_{crit}	=	Critical pressure
L/D	=	Length/diameter from initiation of heating

and subscripts:

b	=	denotes property evaluated at bulk temperature
w	=	denotes property evaluated at wall temperature

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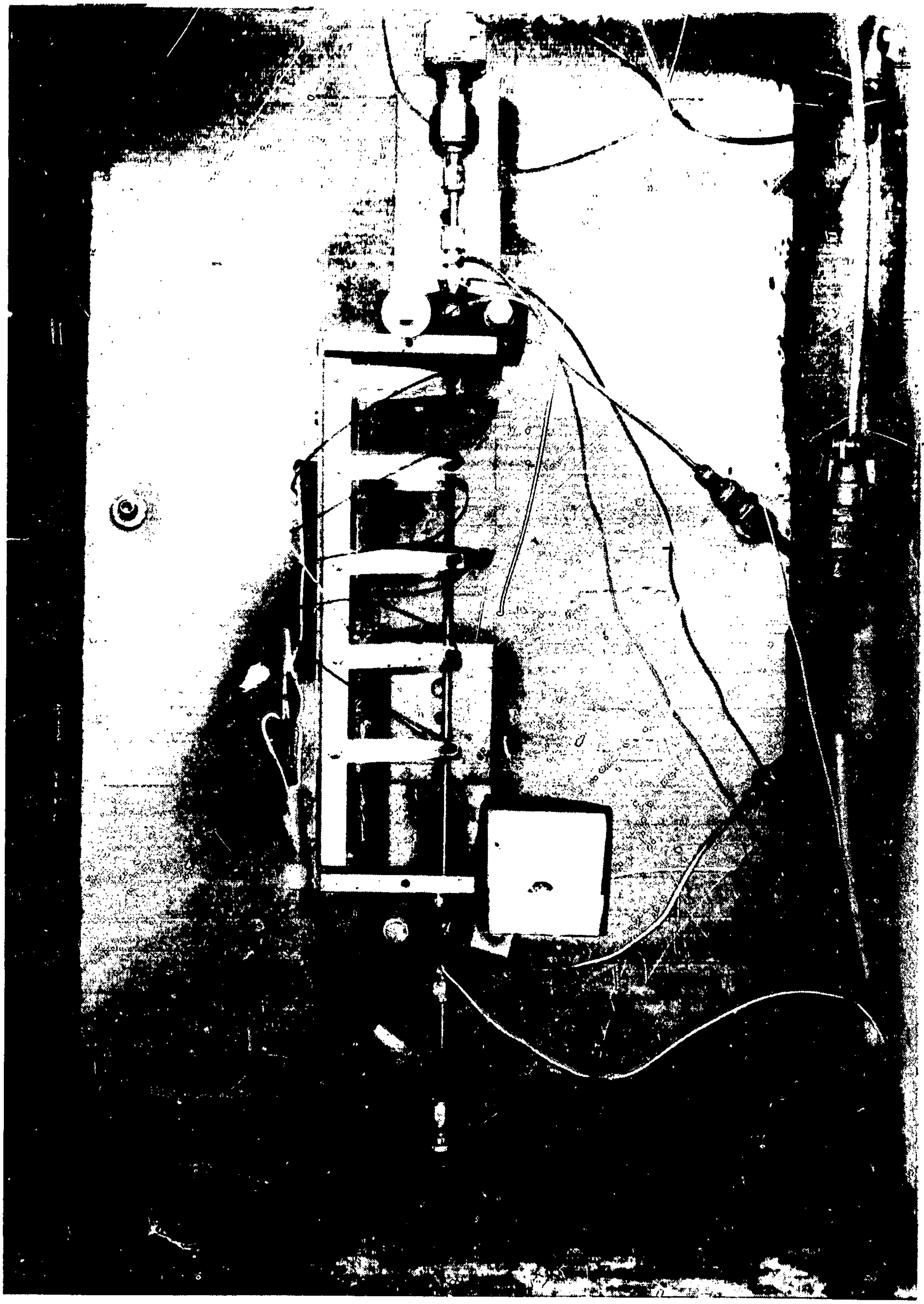

Figure 2. Test Section Installation

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The constants k, a, c, d, e, f, g, and h were determined from the forced convection data by using a multiple regression analysis computer program.

Five cases were analyzed, as follows:

Case Number	Coefficients / Exponents								STD Deviation	Comments
	K	a	c	d	e	f	g	h		
1	.00538	.90	.4*	-.125	.242	.193	-.395	-.024	.130	All forced convection data
2	.00145	1.0*	.4*	-.227	.357	.069	-.299	-.037	.136	All forced convection data Reynolds number fixed
3	.00545	.898	.4*	-.114	.228	.267	-.526	0*	.130	All forced convection data (P/P_{crit}) removed
4	.00532	.889	.4*	-.129	.351	.0995	-.432	0*	.127	Supercritical data (P/P_{crit}) removed
5	.00568	.876	.4*	.120	-.142	.828	-.368	.254	.121	Supercritical data with (P/P_{crit}) term

*Denotes exponent held constant in analysis

Figures 3 and 4 plot the recommended forced convection correlations based on all data and supercritical data only (cases 3 and 5).

Burnout heat flux data were correlated by:

$$\emptyset_{BO} = 0.5 + 0.00027 \quad V \Delta T \text{ sub}$$

where:

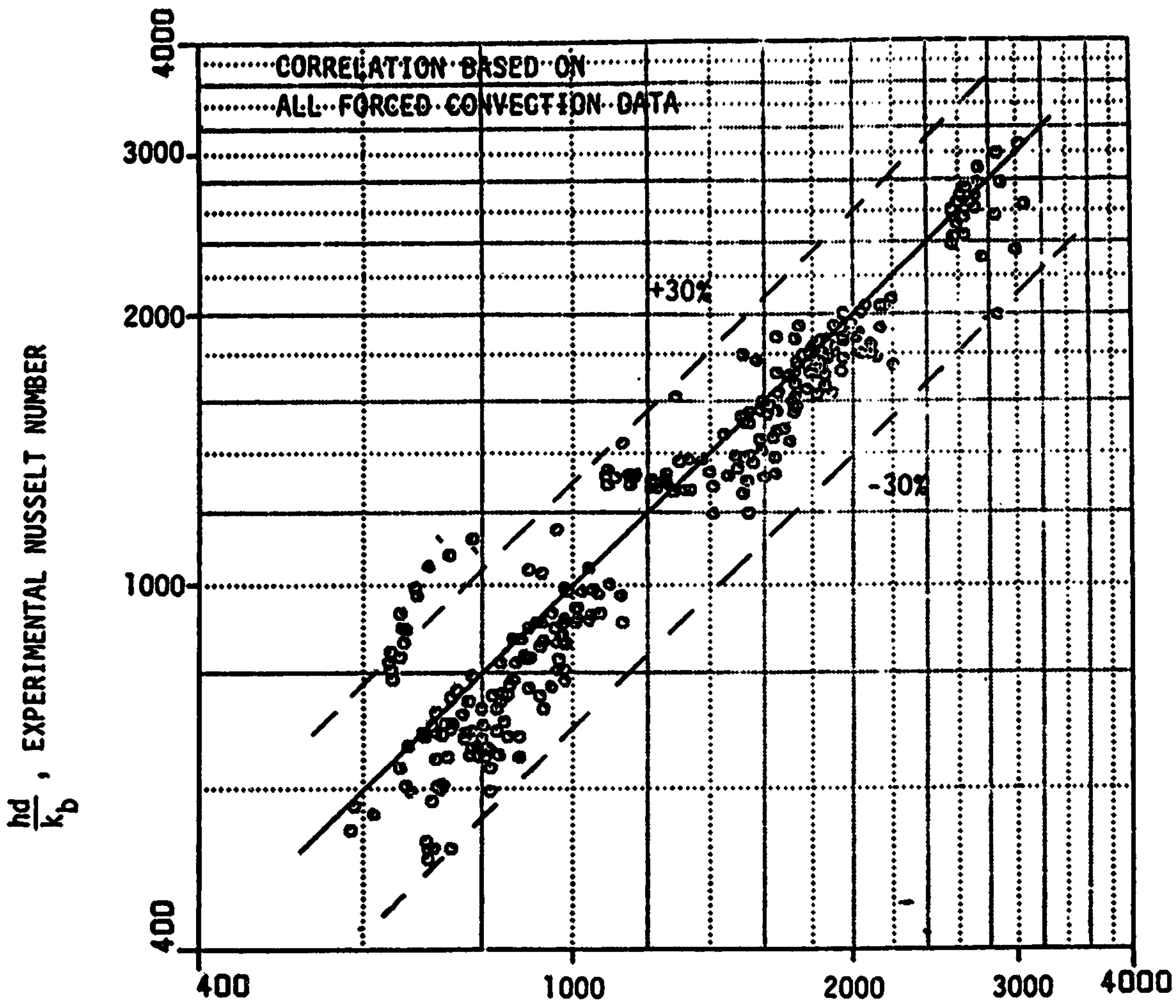
$\emptyset_{BO}$ = Burnout heat flux - Btu/in.^{2}sec
V = Velocity - ft/sec
ΔT_{sub} = (T saturation - T bulk) - °F

Coke buildup rate, defined as the change in coke thermal resistance per unit time, was correlated against reciprocal temperature as shown in Figure 5. The results for propane are not significantly different from those for RP-1, as given in Reference 8, which is considered to be a very "sooty" fuel.

B. TASKS II AND IV - SUBSCALE INJECTOR CHARACTERIZATION

The objectives of Tasks II and IV was to etablish a data base characterizing LOX/Hydrocarbon combustion, in particular the influence of injector pattern, acoustic cavity configuration, chamber length, operating point, and film-cooling, on performance, heat transfer, and stability.

Seven test series comprising 77 tests and a total duration of 1367 sec were conducted. Table I summarizes the test series. Major variables of the test series are listed below.



$$.00545\ Re^{.90}\ Pr^{.4}\left(\frac{\rho b}{\rho w}\right)^{-.11}\left(\frac{\mu b}{\mu w}\right)^{.23}\left(\frac{kb}{kw}\right)^{.27}\left(\frac{\overline{Cp}}{Cp_b}\right)^{.53}\left(1+\frac{2}{L/D}\right)$$

Figure 3. Forced Convection Correlation Based on All Data

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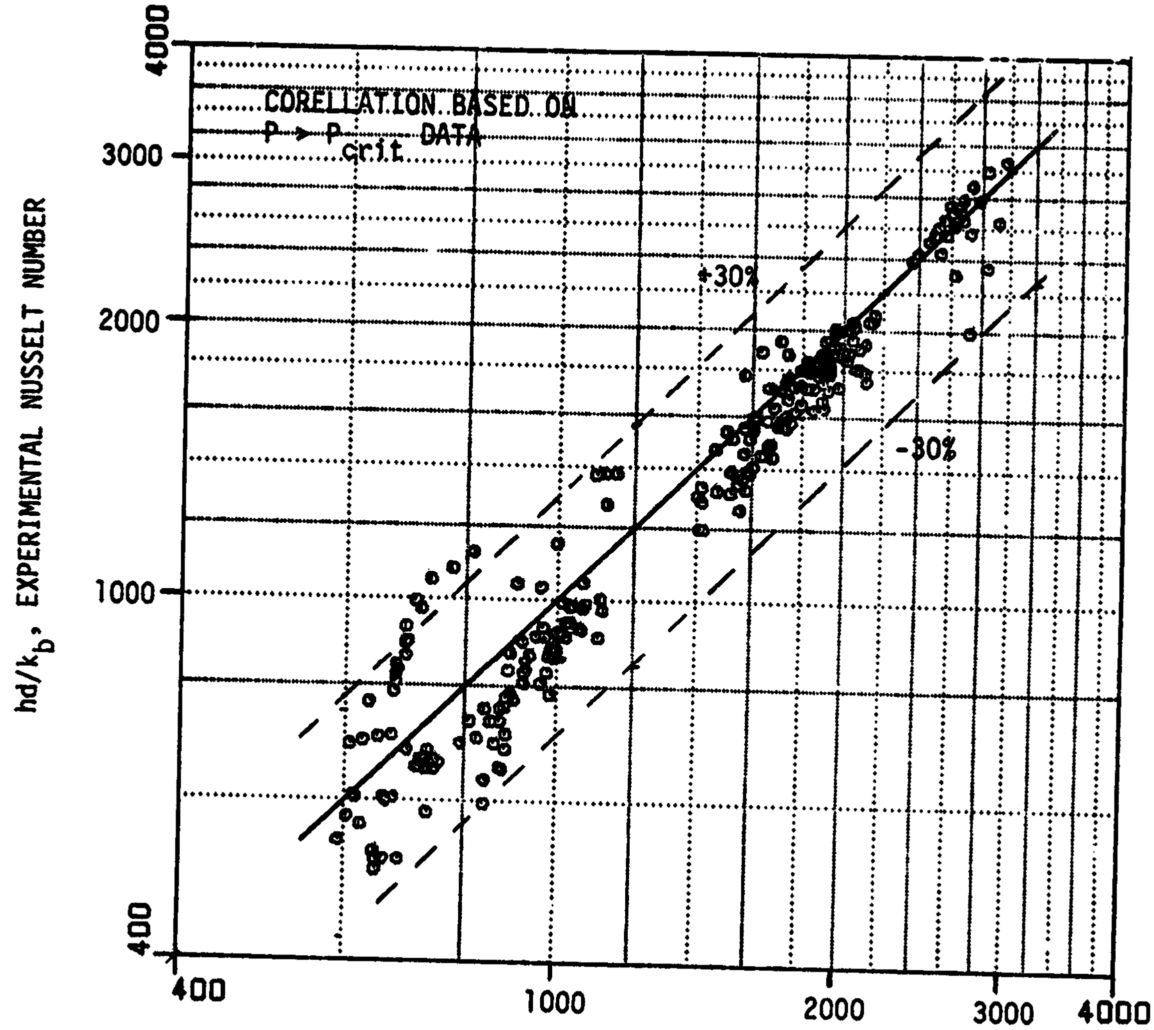

$$.00569\ Re_b^{.88}\ Pr_b^{.4}\left(\frac{\rho b}{\rho w}\right)^{.12}\left(\frac{\mu b}{\mu w}\right)^{-.14}\left(\frac{kb}{kw}\right)^{.83}\left(\frac{\overline{Cp}}{Cp_b}\right)^{-.37}\left(\frac{P}{Pcr}\right)^{.25}\left(1+\frac{2}{L/D}\right)$$

Figure 4. Forced Convection Correlation Based on Supercritical Pressure Data

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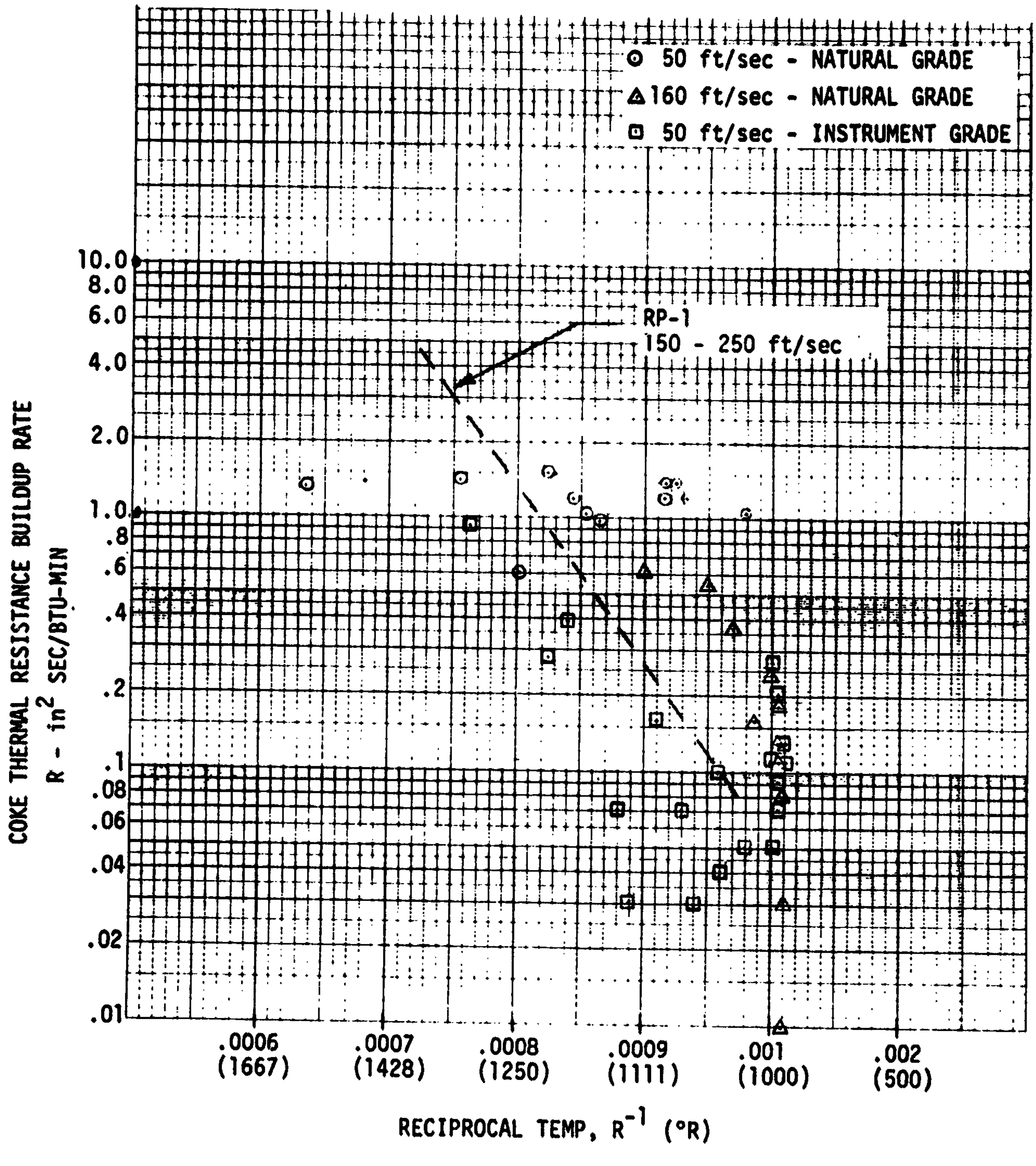

Figure 5. Propane Coking Rates

TABLE I - TEST SERIES SUMMARY

Test Series	Injector *	Propellant	Chamber	Pc (psia)	O/F (-)	Film Coolant	No Tests	Duration (sec)
1	LOL	LOX/C_3H_8	4, 8 in. L' Heat Sink	100-400	2.0-4.0	W/O	18	16
2	OFO	LOX/C_3H_8	8 in. L' Heat Sink	200-400	2.1-4.1	W/O	11	8
3	OFO	LOX/C_3H_8	8.0 in. L' Calorimeter	200-400	1.8-4.3	W/O	10	578
4	OFO + FFC	LOX/C_3H_8	8.7 in. L' Calorimeter	190-420	1.6-4.0	W & W/O	9	322
5	OFO + FFC	LOX/C_2H_5OH	8.7 in. L' Calorimeter	270-350	1.2-2.1	W & W/O	7	147
6	SSS + FFC	LOX/C_2H_5OH	8.7 in. L' Calorimeter	200-400	1.5-2.5	W & W/O	9	185
7	OSO + FFC	LOX/C_2H_5OH (Ambient and low temperature)	8.7 in. L' Calorimeter	90-300	1.0-2.3	W & W/O	13	111
							77	1367

* LOL = Like-on-like, EDM orifices
OFO = Triplet, EDM orifices
SSS = Splashplate-Swirler-Splashplate, platelet OFO
OSO = Orifice-Swirler-Orifice, EDM platelet OFO
FFC = Film-Coolant injector

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Propellant combinations:

LOX/Propane
LOX/Ethanol

Injectors:

Like-on-like, conventional EDM-drilled orifices

OFO triplet, conventional EDM-drilled orifices

Splashplate-Swirler-Splashplate, platelet preatomized triplet OFO

Orifice-Swirler-Orifice, platelet OFO triplet with EDM-drilled oxidizer orifices (gaseous oxidizer)

Film-Coolant injector (ring)

Chambers:

4 in. L' heat sink
8 in. L' heat sink
8 in. L' water-cooled calorimeter chamber

Other Variables:
Oxygen state (liquid and gas)
Propellant temperature
Film-cooling percentage
Chamber pressure
Mixture ratio

Testing began with a like-on-like injector and heat sink chamber which could be extended to a longer length by means of a heat sink barrel section. Figure 6 shows the heat sink hardware and like-on-like injector. In the second test series the injector was replaced with an OFO triplet pattern unit. Figure 7 shows this injector and the five ring manifold injector core.

The water-cooled calorimeter chamber contained nine separate flow sections ganged to twenty-four circumferential coolant channels. Figure 8 shows the copper liner and several split rings, which form the coolant passages in the throat section, prior to brazing into the surrounding cylinder. Figure 9 shows the completed assembly and one of its two external manifolds. External manifolding allowed the possibility of replumbing the coolant flow through the chamber.

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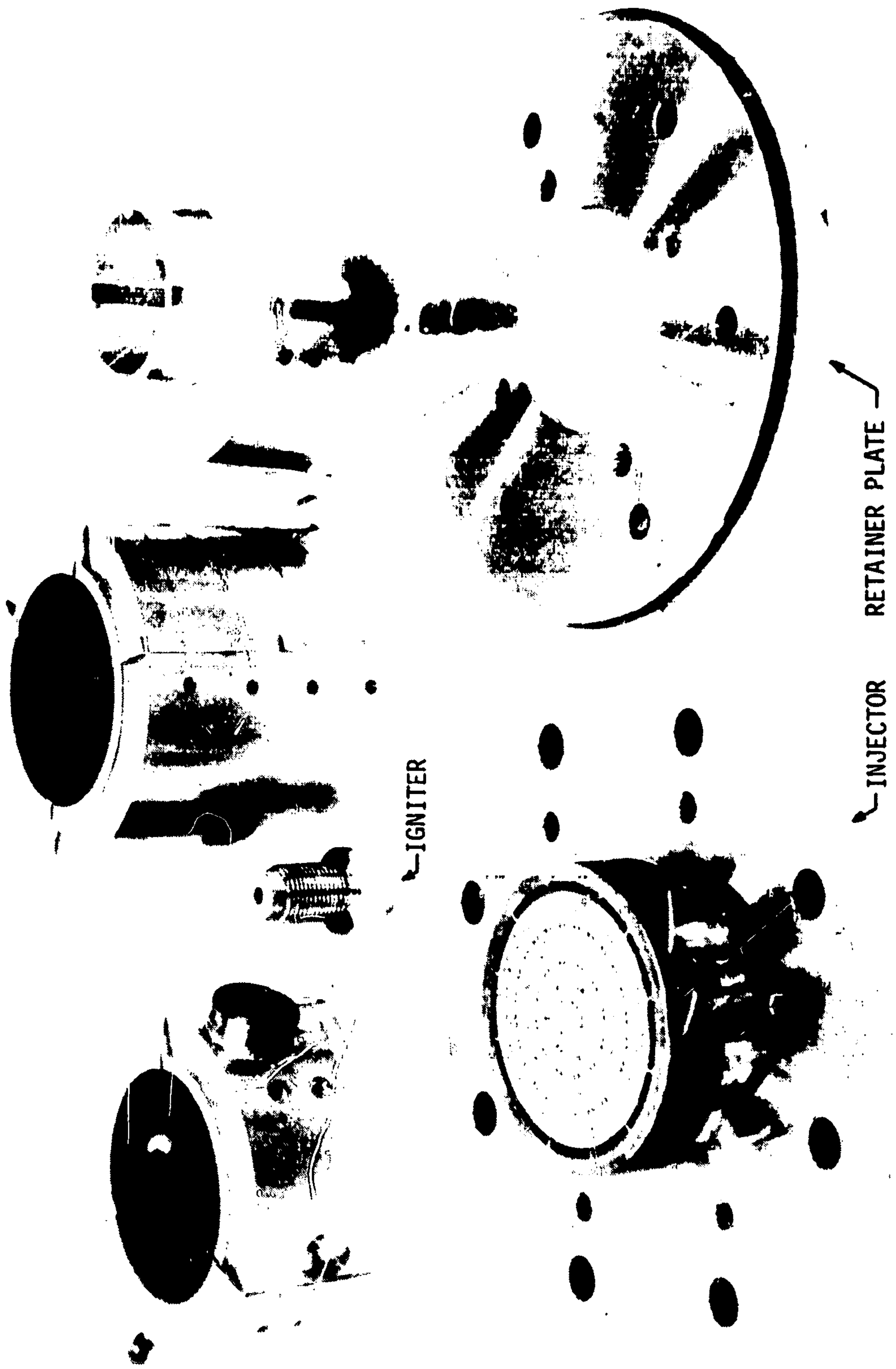

Figure 6. LOX/Propane Thrust Chamber Assembly (Pictorial View)

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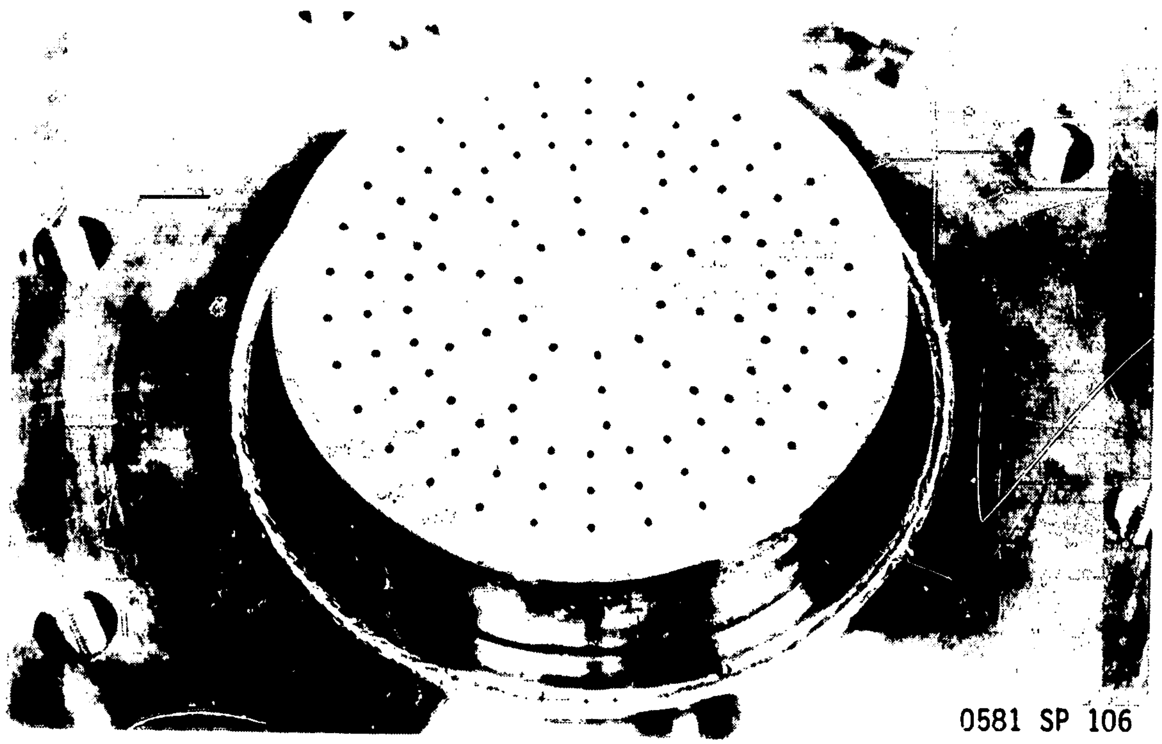

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Figure 7. OFO Triplet Injector

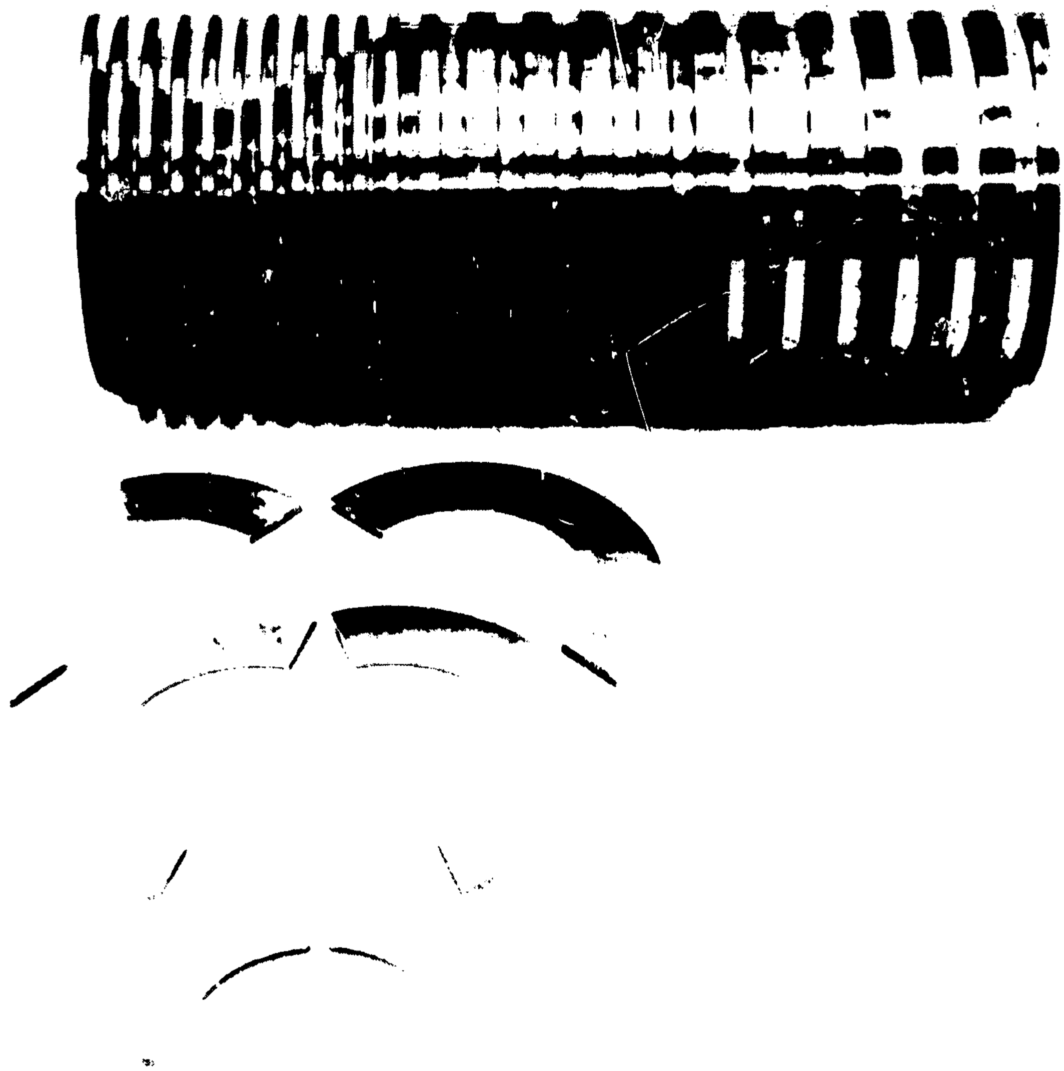

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Figure 8. Copper Liner Prior to Braze Assembly

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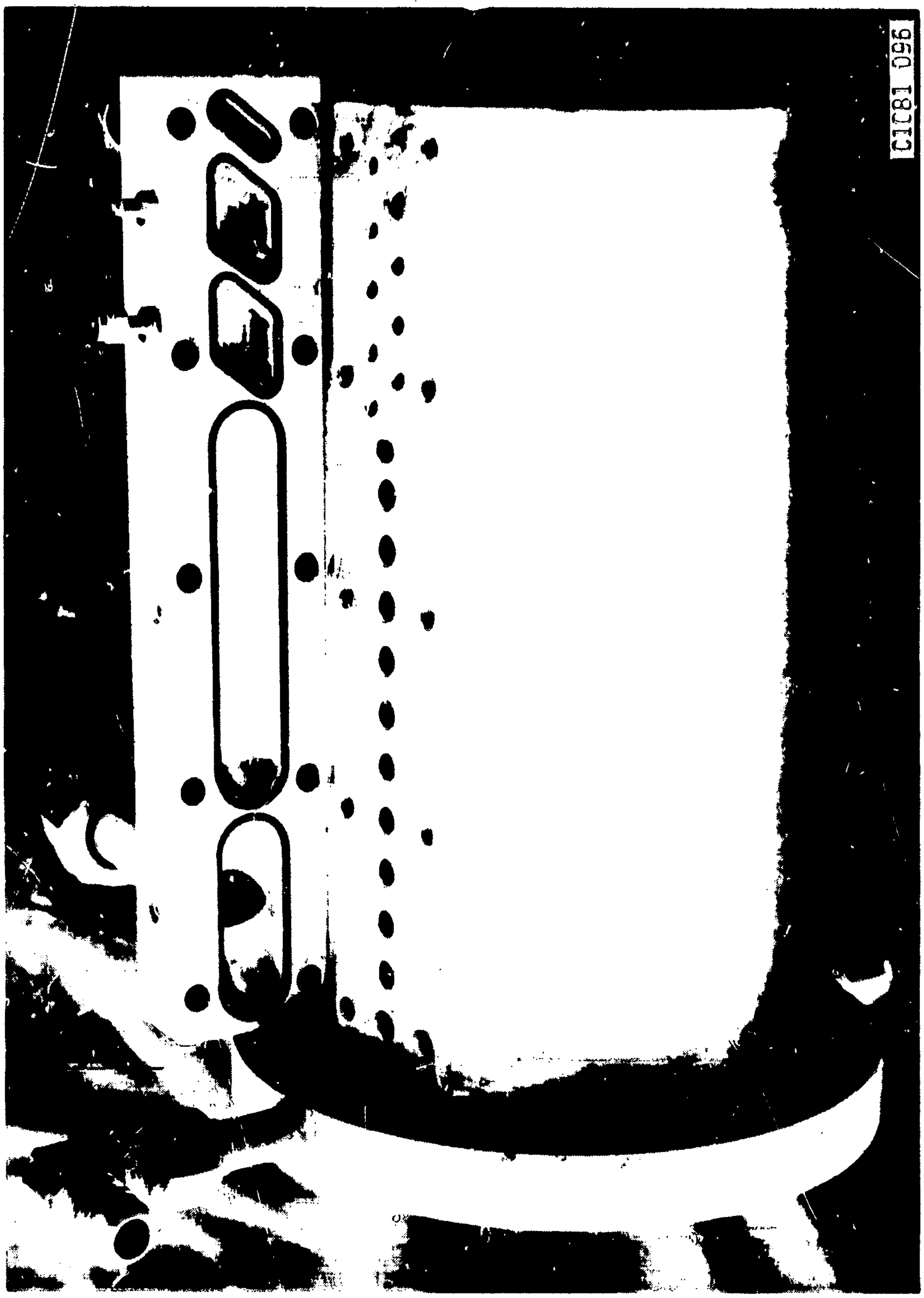


Figure 9. Final Chamber Assembly

IV, B, Tasks II and IV - Subscale Injector Characterization (cont.)

The film-coolant injector (ring) served also as the acoustic cavity section, with the film-coolant being injected from the forward side of the cavity and mechanically atomized by impingement against the injector periphery. This approach worked very well and as an added benefit, kept the cavities free of soot which otherwise was deposited. Figure 10 shows the hydrotest of the film-coolant ring, without the main injector.

Two platelet injectors were fabricated, both OFO-type patterns in which the liquids were mechanically atomized prior to injection. In both units the fuel element was a swirler type which formed a spray cone. The oxidizer element in the first unit was a splashplate, and in the second unit -- which was designed for gaseous oxygen -- a simple EDM-drilled orifice. The injector platelet stacks consisted of 15 or 16 nickel sheets bonded together to a total thickness of approximately 0.125 inch. Figure 11 shows the hydrotest of the liquid-liquid injector.

The calorimeter chamber as mounted on the test stand is illustrated by Figure 12.

Test Series 1. The first test series involved 18 short duration tests of the like-on-like injector in 4 and 8 inch heat sink chambers. Ignition on all tests was reliable and smooth. Stability bombs produced chamber overpressures of 100% which were damped in less than 4 msec through the use of acoustic cavities. Performance was low: 85% energy release efficiency with the 4 inch chamber and 93% with the 8 inch chamber.

Test Series 2. Eleven short firings with the OFO triplet injector and 8 inch heat sink chamber were conducted in the second test series. The combustion was bomb-stable with acoustic cavities but could be bombed unstable without cavities. Energy release efficiency was 4.5% higher, 97.5% at the nominal operating point.

Test Series 3. The calorimeter chamber was used for the first time in this series, in conjunction with the OFO triplet injector. Ten tests were conducted. Firing durations of 20 to 60 seconds were required before the full heat flux reduction due to carbon deposition on the gas-side chamber surface was realized. Cumulative test-to-test heat reductions due to carbon deposition were not experienced, and in fact the shutdown and/or startup transient removed most of the carbon deposit, such that in effect each firing started from a clean-wall condition. The acoustic cavities were progressively being filled with soot, which would ultimately cause loss of damping effectiveness. The highest heat fluxes were recorded at low mixture ratio (O/F), and the highest carbon buildup rates at high mixture ratio. This effect is believed to be the result of a fuel-rich wall environment produced by the OFO

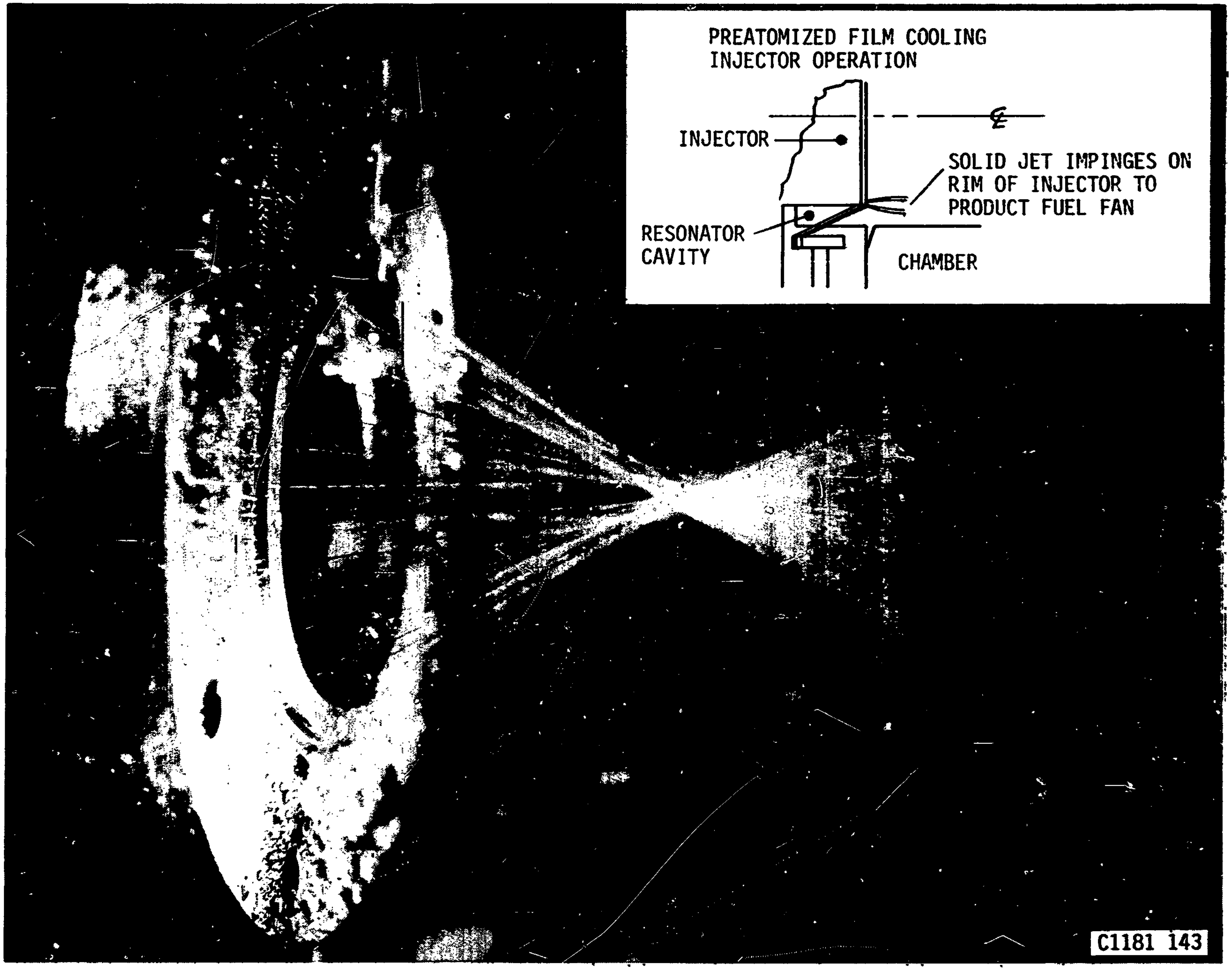
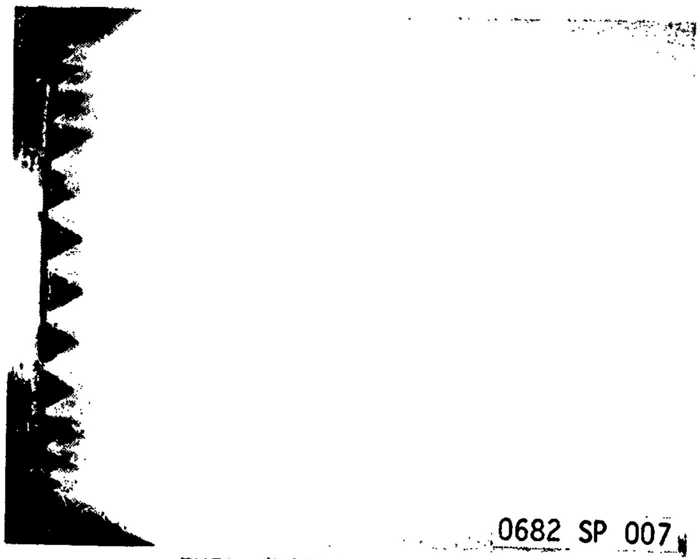
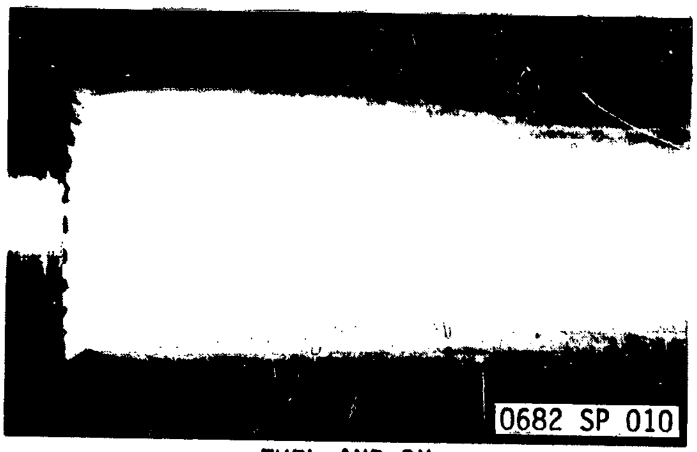


Figure 10. Cold Flow of FFC Ring

OX SPLASH PLATE ONLY



FUEL SWIRLER ONLY



FUEL AND OX

Figure 11. Preatomized Triplet (PAT) Injector Cold Flow

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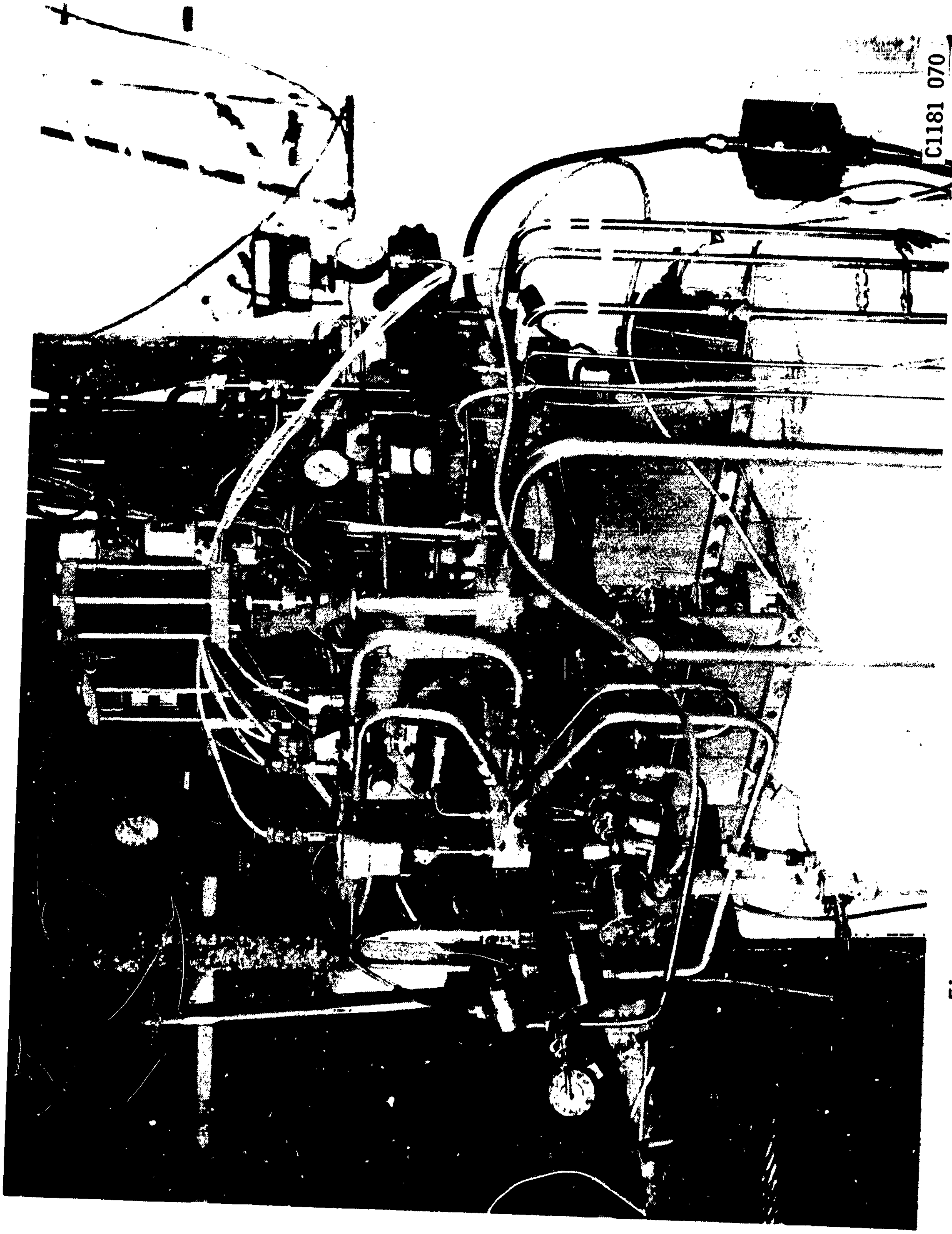

Figure 12. Bay 6 Test Facility with Water-Cooled Chamber In Place

element when the oxidizer-to-fuel momentum ratio is high. Combustion performance in the long duration tests was slightly higher (1.5%) than in the short duration heat sink chamber tests. The OFO triplet injector achieved 99% efficiency at all mixture ratios above 2.7.

Test Series 4. In test series 4 the fuel film-coolant ring (injector) was added to the test configuration of the previous series. Performance and heat transfer data were obtained with 0, 9, and 14% fuel film-cooling. Nine tests were conducted. Significant reductions (approximately 50%) in both barrel and throat heat fluxes were experienced at the highest film-coolant rate. In the last five seconds of each firing, the fuel film-cooling was discontinued; the wall heat flux gradually returned to the values measured without film-cooling in Series 4, indicating that the thermal resistance of the carbon deposit may be a unique function of operating point and not of previous operating history.

Test Series 5. In this series the above hardware was tested with LOX/Ethanol. All prior testing was with LOX/Propane. Combustion performance was slightly degraded (approximately 1.5%) due to non-optimum propellant momentum ratio. In contrast to the LOX/Propane firings, there was no chamber sooting and the exhaust plume was clear. The throat heat flux was somewhat higher (approximately 20%) than the peak values for LOX/Propane and approximately 60% higher than steady-state values with full carbon buildup.

Test Series 6. The platelet splashplate-swirler-splashplate injector was tested with LOX/Ethanol in nine hot firings in this test series. In the absence of film-cooling, head-end heat fluxes were down and combustion efficiency was about 1.5% lower than measured with the OFO triplet. Both head-end heat flux and performance increased with small amounts of fuel film-cooling, indicating an oxidizer-rich boundary condition which is attributed to propellant blowapart. With film-cooling the platelet injector achieved slightly higher performance and lowr wall fluxes than the OFO triplet injector.

Test Series 7. In the last test series, another platelet injector was tested thirteen times with GOX/Ethanol. Eleven tests were conducted with ambient temperature propellants and two with cold (-130°F) propellants. Performance was slightly higher than measured with the other platelet injector, with a nominal point energy release efficiency of 99%. Cold propellants caused a 2% performance loss. There was no evidence of blowapart.

Experimental results are highlighted in Figures 13 through 15. Figures 13 and 14 display engine specific impulse versus mixture ratio, chamber pressure, and percent film-cooling for LOX/Propane and LOX/Ethanol respectively, for the four injectors tested. Figure 15 compares the measured throat heat flux for the three OFO-type injectors and two propellant combinations.

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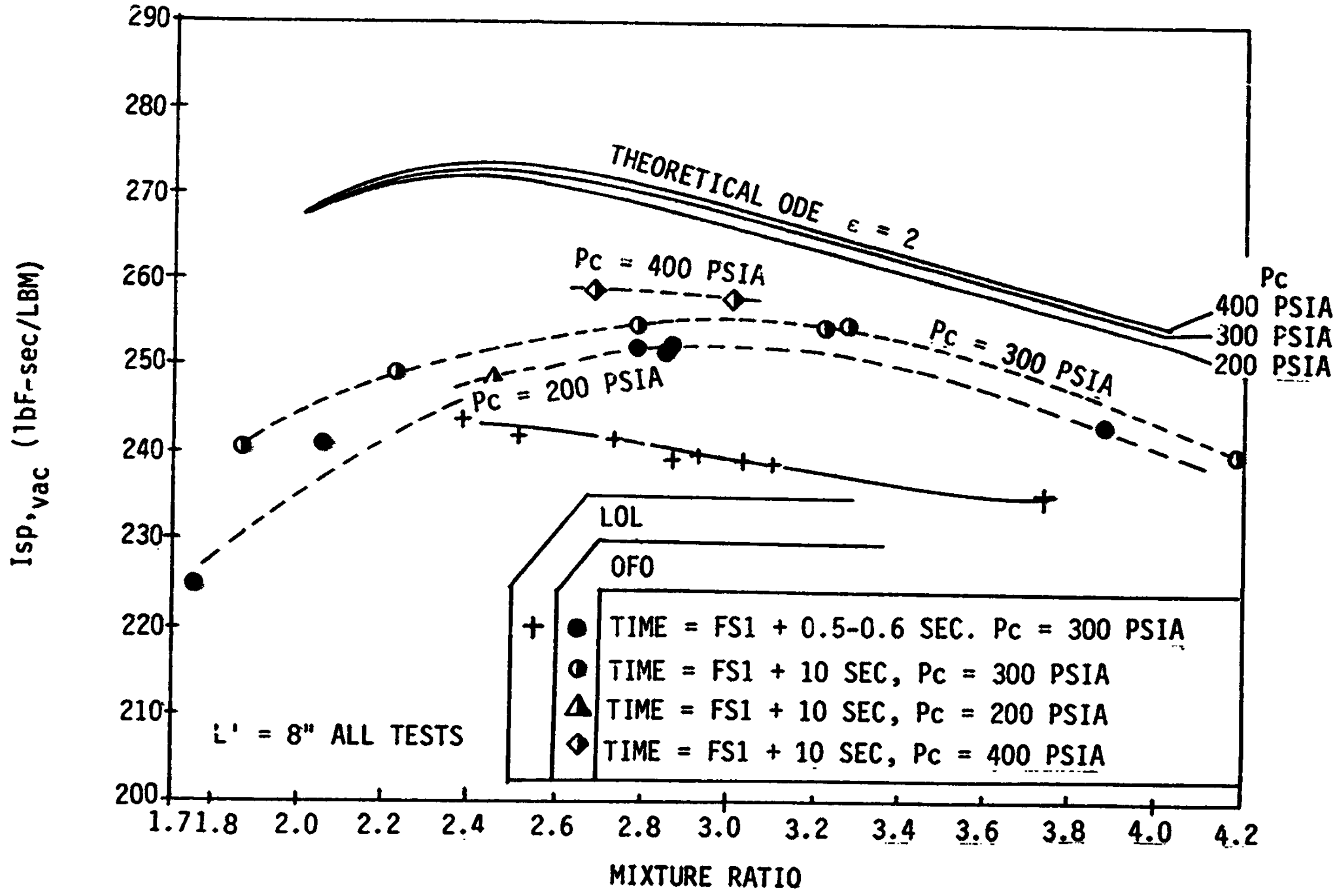

Figure 13. Measured Performance in LOX/Propane Testing

LOX/ETHANOL		GOX/ETHANOL
PLATELET (SSS)	OFO TRIPLET	PLATELET (OSO)
△ 0% FFC	⬡ 0% FFC	○ 0% FFC
▲ 17% FFC	⬢ 14% FFC	● 20% FFC
	◐ 7.5% FFC	

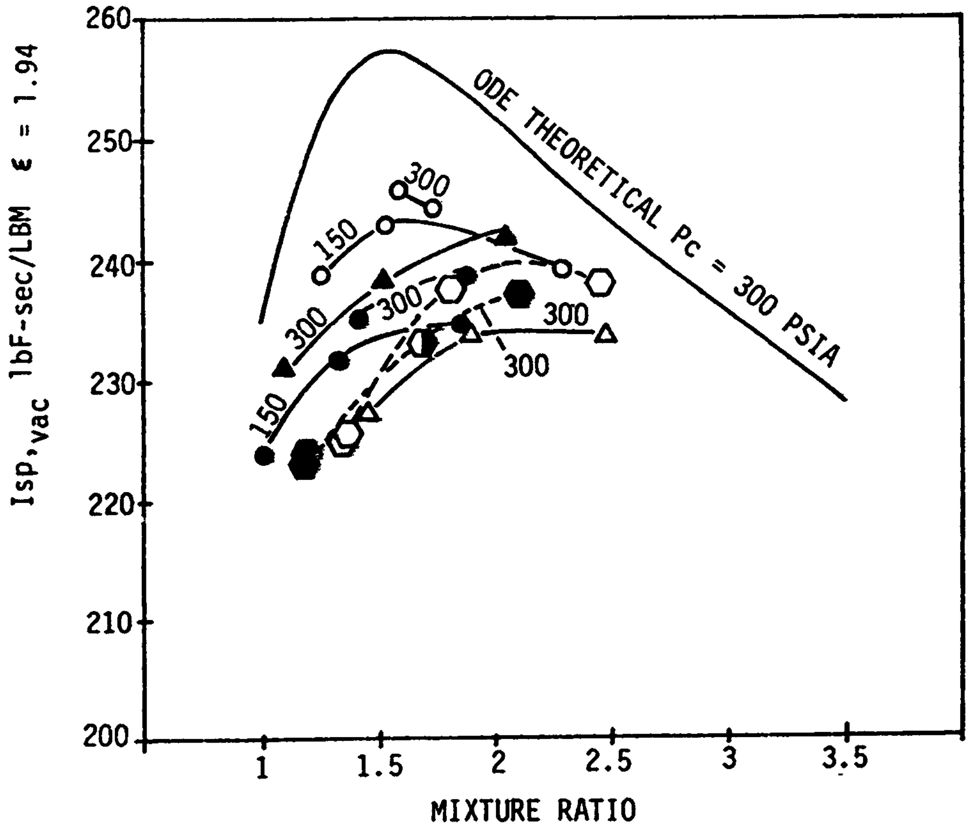

Figure 14. Measured Performance in LOX or GOX/Ethanol Testing

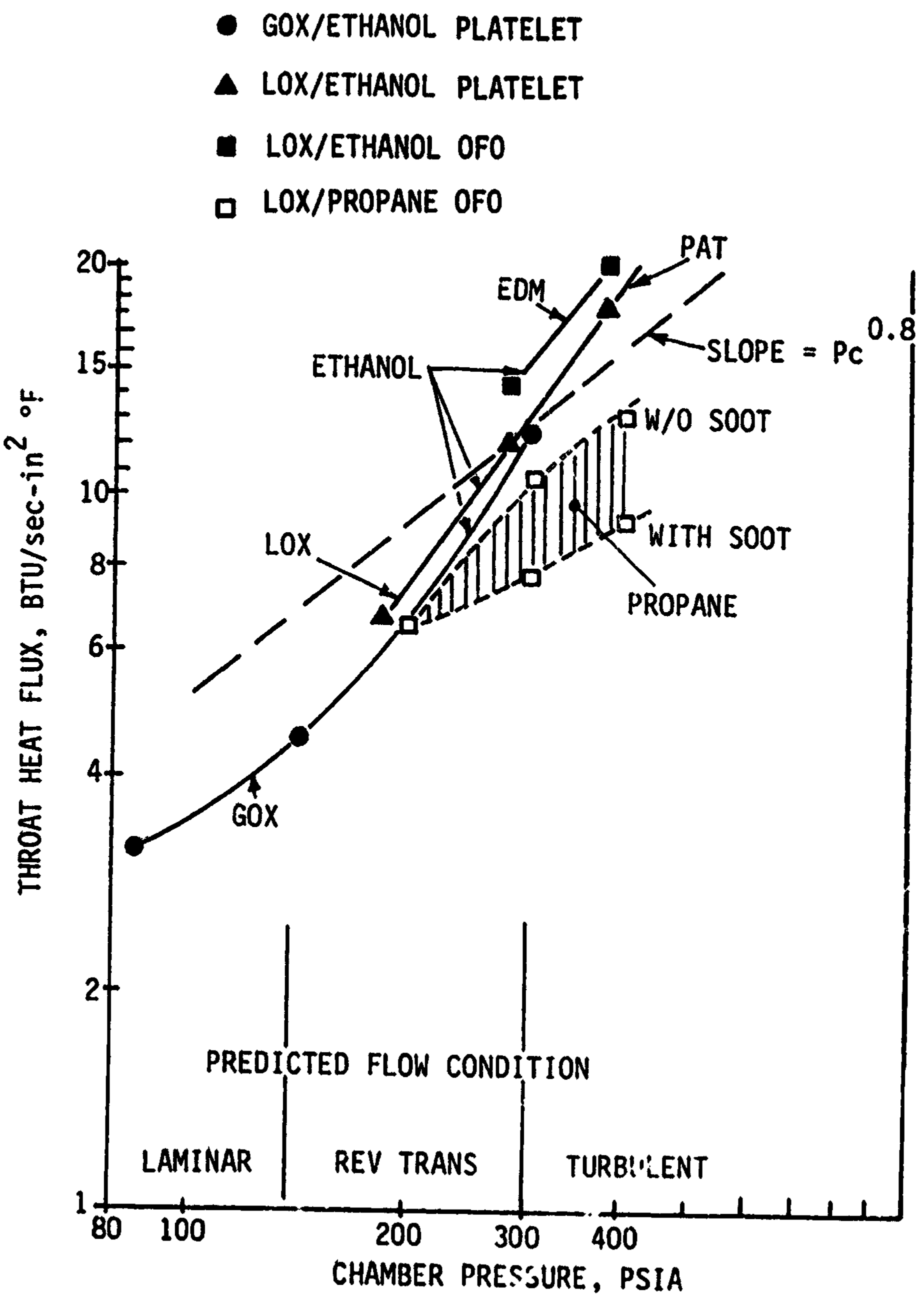

Figure 15. Maximum Throat Heat Flux vs Chamber Pressure

IV, Technical Overview (cont.)

C. TASK III - PRELIMINARY ENGINE SYSTEM CHARACTERIZATION

The objective of Task III was to characterize LOX/Hydrocarbon engine system parameters, in particular performance and weight for orbit maneuvering and reaction control system thrusters. Task III results formed a basis for a related contract, LOX/Hydrocarbon Auxiliary Propulsion System Study (Ref. 5), conducted by McDonnell Douglas Astronautics Company to characterize the entire pod system. ALRC also supported this program under subcontract to provide additional parametric data.

Thirty-eight OME and twenty RCE design points were analyzed on the two contracts. Four fuels were considered: propane, methane, ethyl alcohol, and ammonia. Of the OME design points, twenty-eight were pump-fed systems and ten were pressure-fed. The pump-fed systems were primarily gas generator cycles in which the fuel-rich gas was used to drive separate turbopumps for the two propellants; common shaft concepts were also investigated. Several expander cycles were investigated. Figure 16 illustrates the OME cycles schematically. All twenty of the RCE designs were treated as pressure-fed. Twelve vernier engine design points were also analyzed in a cursory manner.

The analytical approach for a given design point was to first determine the chamber coolant needs; these in turn determined turbopump requirements or tank pressure requirements in pressure-fed systems. Turbopump requirements dictated gas generator requirements. The engine components were thus analyzed sequentially and thereupon the overall engine weight and performance figures could be calculated.

In general the groundrules and assumptions that guided the analysis were consistent with good design practice and the requirements of the current engine specifications. Current engine envelopes were maintained. The propane heat transfer results obtained in Task I were utilized for both the propane and methane design cases. The higher-than-normal gas-side heat fluxes observed with hydrocarbon fuels in NAS 3-21030 (Ref. 9) as well as in Task II were accounted for.

Key results of the parametric study are presented in Table II for all cases analyzed under both contracts.

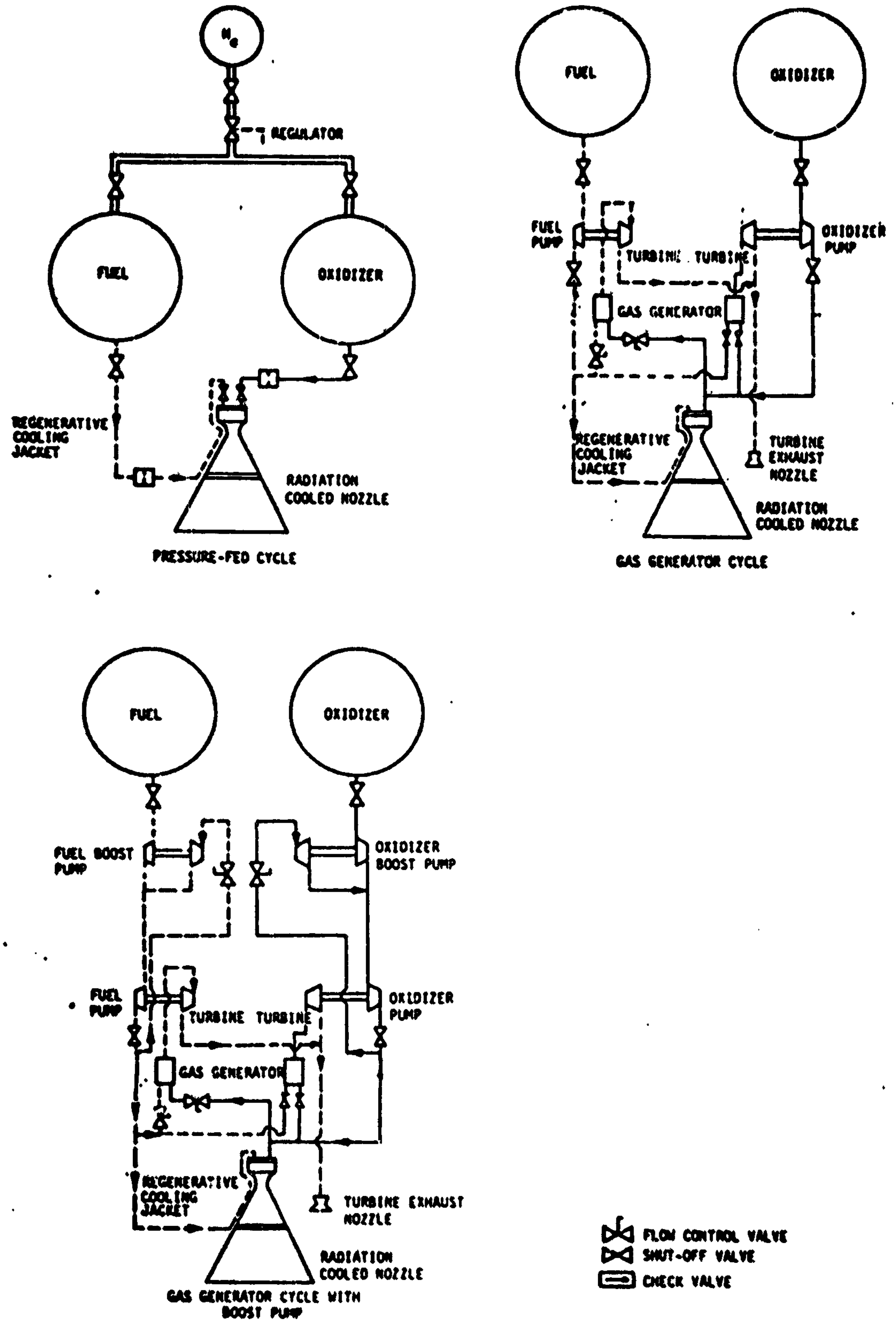

Figure 16. Candidate OME Cycles

TABLE II

LOX/HYDROCARBON APS PARAMETRIC STUDIES

PRESSURE-FED OME (F_v = 6K lbF)

PROPELLANTS	Pc, Psia	CHAMBER MAT'L	TCA MR	TCA Isp_v, Sec	%FFC	INTERFACE PRESS: OX/F	ENGINE* WEIGHT	COMMENTS
LOX/C_3H_8 (NASA)	100	Zr-Cu	1.92	324.3	30 (10.3)**	143/183	323.1	LIQUID REGEN
	100	Zr-Cu	2.75	337.0	Ø	143/157	322.1	AMBIENT C_3H_8 VAPOR REGEN
	100	Ni	2.75	337.0	Ø	143/161	318.0	AMBIENT C_3H_8 VAPOR REGEN
	150	Zr-Cu	1.79	326.7	35 (12.5)	209/381	285.8	LIQUID REGEN
LOX/CH_4 (NASA)	100	Zr-Cu	3.00	343.2	Ø	143/147	318.0	VAPOR REGEN
	100	Ni	3.00	343.2	Ø	143/150	318.0	" "
	150	Zr-Cu	3.40	346.2	Ø	209/231	289.4	" "
LOX/NH_3 (N.SA)	100	CRES	1.25	318.8	11 (4.9)	143/159	317.7	LIQUID REGEN
LOX/C_2H_5OH (MDAC)	100	Zr-Cu	1.60	319.9	Ø	143/206	313.7	" "
	150	Zr-Cu	1.60	326.4	Ø	209/467	286.1	" " 2 PASS REGEN

*VAPOR REGEN ENGINE WGTS DO NOT INCLUDE ALLOWANCE FOR POTENTIAL HEAT EXCHANGER NOZZLE EXTENSION
**%FFC OF TOTAL ENGINE FLOW

TABLE II (cont.)

● PUMP-FED OME

PROPELLANTS	F_V/Pc	CHAMBER MAT'L	ENGINE MR	ENGINE Isp_V	%FFC	PUMP DISCHARGE PRESSURE: OX/FUEL	ENGINE WEIGHT	COMMENTS
LOX/C_3H_8 (NASA)	6K/400	Ni	2.58	354.9	3 (0.8)*	535/980**	328.3	NOM Cg
	"	Ni	2.66	355.3	Ø	535/980	328.3	FLAT Cg
	"	CRES	2.02	343.3	25 (7.9)	535/980	325.3	
	6K/800	Zr-Cu	2.81	368.7	Ø	1020/1196	327.9	
	10K/400	"	2.69	351.7	Ø	535/980	393.4	
	10K/800	"	2.83	364.2	Ø	1040/1155	395.6	W/O BOOST PUMPS
	"	"	2.82	363.8	Ø	1040/1155	408.5	WITH BOOST PUMPS
LOX/CH_4	6K/400 (NASA)	Ni	3.41	360.8	Ø	535/1080	326.9	
	" (MDAC)	Ni	3.50	363.7	Ø	535/1150	317.2	EXPANDER CYCLE
	" (MDAC)	Zr-Cu	3.50	363.7	Ø	535/1150	317.2	EXPANDER CYCLE
	6K/600 (MDAC)	Ni			NOT FEASIBLE			EXPANDER CYCLE
	" (MDAC)	Zr-Cu			NOT FEASIBLE			EXPANDER CYCLE
	6K/800 (MDAC)	Ni			NOT FEASIBLE			EXPANDER CYCLE
	" (MDAC)	Zr-Cu			NOT FEASIBLE			EXPANDER CYCLE
	10K/400 (NASA)	Zr-Cu	3.44	356.0	Ø	535/980	382.4	
	10K/800 (NASA)	Zr-Cu	3.43	366.0	Ø	1040/1123	382.6	FUEL REGEN
	" (NASA)	"	3.39	365.3	Ø	1308/1020	382.1	OXID REGEN
	" (NASA)	"			NOT FEASIBLE			EXPANDER CYCLE
LOX/NH_3 (NASA)	10K/400	CRES	1.21	326.5	13 (5.8)	533/614	376.8	
	10K/800	"	0.93	317.4	33 (17)	1040/1330	383.8	
LOX/C_2H_5OH (MDAC)	6K/400	Ni	1.59	334.4	10 (3.8)	535/1094	329.7	NO SOOT DEPOSITION
	"	Zr-Cu	1.77	339.6	Ø	535/1036	"	"
	6K/600	Ni	1.33	324.2	25 (10.6)	770/1084	325.4	"
	"	Zr-Cu	1.76	346.1	Ø	770/1066	"	"
	6K/800	Ni	1.24	318.7	30 (13.3)	1020/1128	323.0	"
	"	Zr-Cu	1.72	346.3	2 (0.7)	1020/1150	"	"
	10K/400	Zr-Cu	1.77	335.5	Ø	555/1121	404.1	2 PASS REGEN
	10K/800	"	1.75	344.1	Ø	1040/1338	389.4	

*%FFC OF TOTAL ENGINE FLOW

**ALL C_3H_8 & CH_4 DESIGNS HAVE SUPERCRITICAL REGEN COOLING

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TABLE II (cont.)

550-LBF AND 870-LBF RCE

PROPELLANTS	Fv/Pc	PROP. INLET* STATE: OX/F	MR	I_{SPV}	% FFC	PROP. INLET PRESSURE: OX/F	TCA WGT
LOX/C_3H_8 (NASA)	870/100	L/L	2.24	297.9	18.6 (5.7)	177	23.1
	870/150	L/L	2.23	305.4	19 (5.9)	256	22.0
	870/150	G/L	2.23	305.6	18.8 (5.8)	202/256	22.9
	870/250	G/L	2.20	315.5	20 (6.3)	316/414	21.8
	870/300	L/L	2.19	318.8	20.5 (6.4)	492	20.6
LOX/CH_4	870/100 (MDAC)	L/L	2.49	302.8	17.1 (4.9)	177	23.1
	870/150 (NASA)	L/L	2.49	313.7	17 (4.9)	256	22.0
	870/150 (NASA)	G/G	2.48	313.9	17.2 (5.0)	204	24.8
	870/250 (MDAC)	L/L	2.47	318.8	17.6 (5.1)	420	20.9
	870/250 (NASA)	G/G	2.43	322.2	19 (5.5)	316	23.7
	870/400 (MDAC)	L/L	2.69	326.7	20.8 (5.6)	659	20.1
LOX/NH_3 (NASA)	870/150	L/L	1.12	292.4	20 (9.4)	246	21.8
	870/250	G/L	1.12	301.6	20 (9.4)	316/398	21.5
LOX/C_2H_5OH (MDAC)	870/150	L/L	1.30	288.3	18.9 (8.2)	252	21.8
	550/100	G/L	1.29	281.5	19.4 (8.5)	143/173	21.8
	550/250	G/L	1.40	300.0	22.5 (9.4)	326/403	20.0
	550/400	G/L	1.39	308.0	22.7 (9.5)	508/632	19.4
	870/100	G/L	1.30	275.8	18.7 (8.1)	143/173	23.7
	870/250	G/L	1.40	294.5	22.2 (9.2)	326/403	21.5
	870/400	G/L	1.40	303.6	22.3 (9.3)	508/632	20.7

*USED FOR INTERFACE PRESSURE CALCULATIONS. PERFORMANCE ANALYSES ARE BASED ON NBP PROPELLANT TEMPERATURES AND THERMAL ANALYSES ON FILM COOLANT INJECTED AS A SATURATED VAPOR.

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TABLE II (cont.)

25-LBF RCE

PROPELLANTS	Pc, PSIA	PROP. INLET STATE: OX/F	MR	Isp_v	% FFC	PROP. INLET PRESSURE: OX/F	TCA WGT
LOX/C_3H_8 (NASA)	100	L/L	2.75	223.4	23	SAME AS 870-LBF RCES	8 ± 1 LBM
	150	L/L	"	229.0	"		
	150	G/L	"	229.2	"		
	250	G/L	"	236.6	"		
	300	L/L	"	239.1	"		
LOX/CH_4 (NASA)	150	L/L	3.00	235.3	21		
	150	G/G	"	235.4	"		
	250	G/G	"	241.6	"		
LOX/NH_3 (NASA)	150	L/L	1.4	219.3	39		
	250	G/L	"	226.2	"		
LOX/C_2H_5OH (MDAC)	150	L/L	1.6	216.2	36		
	250	G/L	1.8	225.8	33		

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