

FINAL REPORT

**GAS-LIQUID SPACE STORABLE
PROPELLANT PERFORMANCE**

BY
W. A. CARTER



PREPARED FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

A 10-month analytical and experimental technology program was conducted to develop design criteria for a high performance coaxial (pintle) injector using gas-liquid space storable propellants. The propellant combination employed was gaseous methane and liquid FLOX (82.6% F_2). The overall program was comprised of five tasks: Task I: Injector and Thrust Chamber Analysis and Design; Task II: Injector and Thrust Chamber Fabrication; Task III: Injector Characterization Tests; Task IV: Injector Performance Evaluation Tests; and Task V: Data Evaluation. Nominal engine design/operating conditions were 3000 lbf (13345 N) sea-level thrust at a chamber pressure of 500 psia (3440 kN/m²) and mixture ratio of 5.75. Experimental variables evaluated included several injection orifice geometries, engine mixture ratio (5.0 to 6.5) and thrust chamber characteristic length, 15 inches (38.1 cm) to 60 inches (152.4 cm). Demonstrated performance with the nominal 30 inch (76.2 cm) characteristic length chamber at the design 5.75 mixture ratio was 97 percent of the theoretical equilibrium characteristic velocity. Engine dynamic stability was demonstrated with a pulse gun induced excitation test, and a single extended duration test was conducted. The program provided fundamental coaxial (pintle) injector design criteria for the gas-liquid space storable propellants.

FOREWORD

This report was prepared for Lewis Research Center of the National Aeronautics and Space Administration, Cleveland, Ohio by the Science and Technology Division of TRW Systems Group. The program was conducted in accordance with Contract NAS3-13307.

Mr. L.H. Gordon, NASA Lewis Research Center, served as NASA Program Manager. The TRW Program Manager was Mr. W.A. Carter. Technical guidance was provided by Dr. H. L. Burge and S.J. Van Grouw.

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1. SUMMARY

Presented in this report are the analytical and experimental results obtained during a 10-month technology program conducted to investigate high performance coaxial pintle injector designs using gaseous fuel/liquid oxidizer space storable propellants. Liquid FLOX (82.6% F₂ and 17.4% O₂) and gaseous methane were the propellants used during the program. The primary objective of this program was to extend previous FLOX/LPG liquid/liquid coaxial injector design criteria to liquid oxidizers/gaseous fuels operating at a nominal chamber pressure of 500 psia (3447 kN/m²). Secondary program objectives included experimental evaluations of injector dynamic stability and resultant thrust chamber wall environment.

The initial phase of this program was directed toward the analytical definition of the gaseous fuel jet velocity distribution, the atomization of the liquid streams by the gaseous jet and the resultant effect on combustion performance. The gaseous methane jet characteristics were defined based upon the work of Abramovich and others. The atomization model for the gas-liquid impingement was based upon the work of Ingebo at NASA and the combustion model used was an extension of the vaporization model presented by Priem and Heidmann.

Finally, an injector model based upon hydraulic characteristics was developed to define the injection properties (geometry and velocity ratios) which would deliver optimum performance.

Nominal design and operating conditions for the test effort were as follows:

Sea-level thrust	3000 lb _f (13345 N)
Chamber pressure	500 psia (3447 kN/m ²)
Mixture ratio	5.0 to 6.5
Characteristic length, L*	15, 30 and 60 inches (38.1, 76.2 and 152.4 cm)
Chamber contraction ratio, ϵ_c	4.0
Nozzle expansion ratio, ϵ	5.0

The experimental phase of this program consisted of liquid-liquid cold flow tests of each injector configuration and 27 hot firing tests. The cold flow tests using water as the propellant simulants were conducted in an effort to obtain a qualitative ranking of each injector configuration. The hot firing experiments included streak tests of each injector geometry. Combustion performance over a mixture ratio range of 5.0 to 6.5 was determined for each configuration in a baseline 30-inch (76.2 cm) thrust chamber. The most promising element was then selected for evaluation in chambers of 15- (38.1 cm) and 60-inch (152.4 cm) characteristic length (L*). The engine was then subjected to a dynamic stability test using a

pulse gun as the excitation device, and, finally, a single planned extended duration test was conducted to demonstrate injector durability. Combustion efficiency and heat flux characteristics of the injector were shown to be strong functions of injector geometry. It was also shown that performance for this injector was not vaporization limited. The injector exhibited high combustion performance (approximately 97 percent of theoretical shifting equilibrium C^* at $O/F = 6.3$) in the baseline 30-inch (76.2 cm) L^* thrust chamber along with very smooth and stable combustion characteristics. Combustion "noise" or roughness was less than 3 percent peak-to-peak and the engine recovered from a 300 psi (2028 kN/m²) over-pressure in less than 15 msec, demonstrating its inherent dynamic stability.

This program demonstrated that an inherently stable high performing coaxial pintle injector with compatible wall heat fluxes could be developed for use with gaseous methane and liquid FLOX at a chamber pressure of 500 psia (3447 kN/m²). Fundamental design criteria for gas-liquid injectors using space storable propellants was obtained.

2. INTRODUCTION

The Gas-Liquid Space Storable Propellant Performance Program, Contract NAS3-13307, was a 10-month effort to extend the design criteria developed by TRW for liquid-liquid injectors under a previous contract (Reference 1) to gas-liquid propellants. This effort marked the second research program on the coaxial injector for primary space engine propulsion. The specified space storable propellants were liquid 82.6 percent FLOX and gaseous methane. The chamber pressure was also extended from 100 (689 kN/m²) to 500 psia (3447 kN/m²). The major objectives of the program were to:

- Design a coaxial pintle injector for high performance (a goal of 98 percent of theoretical shifting characteristic exhaust velocity)
- Evaluate resultant wall heat fluxes
- Prove inherent injector dynamic stability
- Develop design criteria which would allow subsequent extrapolation of the results of this program to other space storable propellant combinations and operating conditions.

To accomplish the objectives of the contract, a five-task program was established which involved analysis, design, fabrication and testing of the TRW coaxial pintle injector. The nominal design conditions for the injector were a sea-level thrust of 3000 lbf (13345 N) and a chamber pressure of 500 psia (3447 kN/m²) at a mixture ratio of 5.75.

The Task I effort consisted of a design analysis of the coaxial pintle injector for use with the gaseous methane-liquid FLOX (82.6% F₂) propellant combination. An evaluation of the data for liquid-liquid, coaxial pintle injectors and its application to the gas-liquid injector was conducted. The analysis included hydraulic/combustion analysis of gas-liquid injectors and thermal analyses of the injector and the thrust chamber to be used for experimental purposes.

Task II covered the fabrication of an injector assembly with interchangeable oxidizer orifice configuration in accordance with the design approved in Task I. Thrust chamber assemblies were fabricated with the capability of varying the characteristic length (L*) and incorporating streak sections. An extended duration passively cooled graphitic thrust chamber was also fabricated.

Initial injector characterization was accomplished during Task III by a series of cold-flow tests and a hot-firing streak test series.

Task IV covered injector performance evaluation for three oxidizer element geometries as a function of mixture ratio in the baseline 30-inch (77.2 cm) L* chamber. One oxidizer element configuration was then selected and the injector performance was evaluated as a function of

characteristic length (L^*) over the range of 15 to 60 inches (38.1 to 152.4 cm) and mixture ratio from 5.0 to 6.5. The engine dynamic stability was demonstrated in the baseline 30-inch (76.2 cm) L^* chamber using a pulse gun excitation device. The injector extended duration capability was also demonstrated in an all-graphite, passively cooled thrust chamber.

Evaluation of experimental data collected during Tasks III and IV was accomplished in Task V. Data acquired during all of the previous experimental tasks were evaluated to provide design criteria for the coaxial pintle injector, using liquid FLOX and gaseous methane as propellants. The hot firing results of Task IV were correlated with the cold flow results of Task III and the analytical predications of Task I.

3. ANALYSIS

The injector design selected for performing the tasks of this program was based upon the use of the TRW single element coaxial injector. The coaxial injector design is a single central element configuration wherein the gaseous fuel is injected as a continuous cylindrical sheet which impinges with liquid oxidizer streams injected radially outward from a series of uniformly spaced orifices as illustrated in Figure 1. Part of the fuel strikes the top surface of the individual oxidizer streams and part of the fuel penetrates between the oxidizer streams. The initial combustion products of the hypergolic reaction drive the propellants into each other, promoting secondary mixing. Performance is controlled through oxidizer element sizing and orifice spacing as well as fuel sheet thickness. Additional performance tuning can be accomplished by the addition of secondary oxidizer elements between the major oxidizer slots.

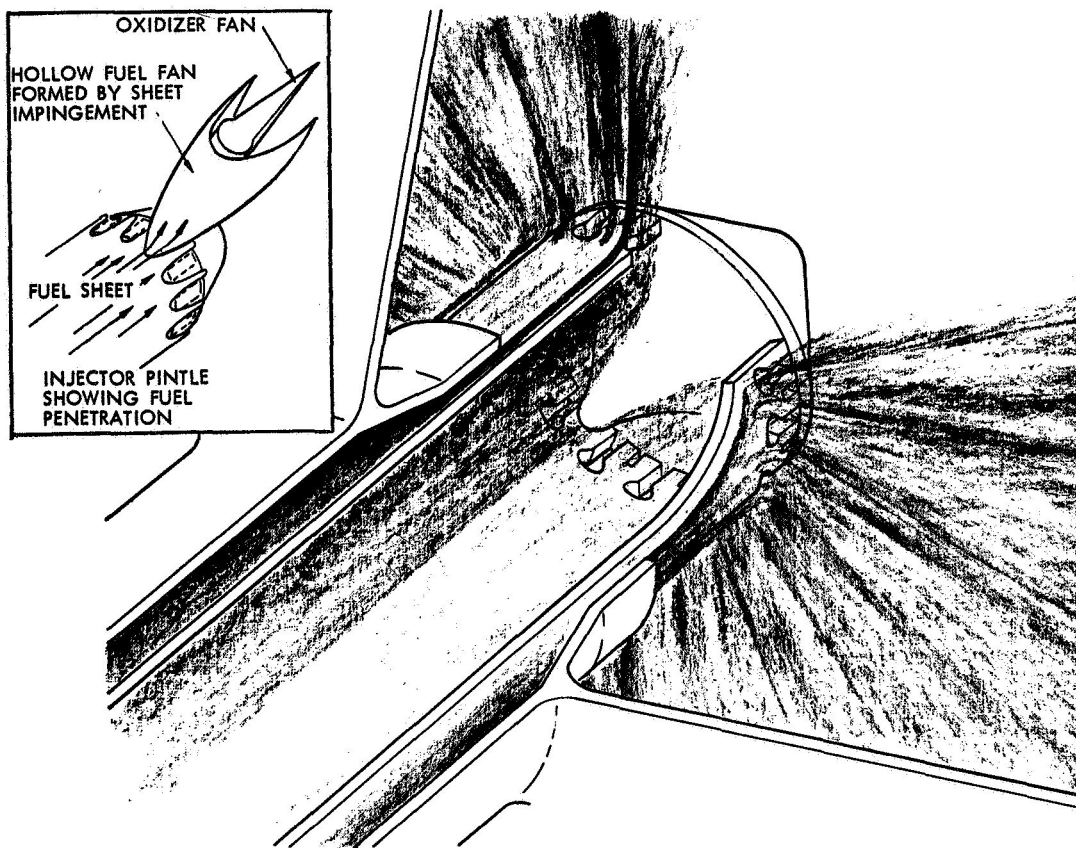


Figure 1. Single Element Coaxial Pintle Injector Schematic

3.1 INJECTOR COMBUSTION ANALYSIS

Analyses were conducted to determine the effects of geometry, momentum ratio, velocities, impingement angles and pressure drop on combustion performance of the injector. For gas-liquid injector designs (in absence of gross propellant maldistribution), propellant atomization, by its control of vaporization rate, is the controlling process in determining the degree of combustion efficiency.

Analyses of the atomization of the liquid stream by the gas stream were performed for use in conjunction with the combustion analysis. In order to define the atomization process between the liquid-gas impingement, a knowledge of the gaseous jet characteristics was necessary. This section presents fundamentals of gaseous jets as they affect performance of the injector.

3.1.1 Gaseous Jet Theory

The theory presented here provides a basis for geometrical layout of an injector employing gaseous jets; the theory is for a two-dimensional jet but can be extended to axisymmetric jets. Specific attention is directed toward a two-dimensional gaseous methane jet, but is generally applicable to any gaseous injection.

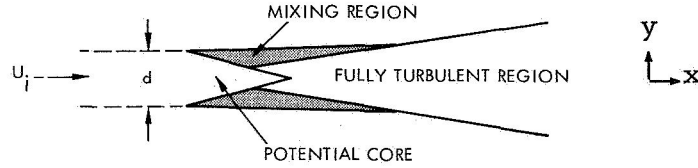
A free turbulent flow is one whose motion is not directly affected by the presence of solid boundaries so that turbulence can spread laterally into the surrounding fluid. In consequence, these flows exhibit inhomogeneity in the direction of mean motion. A common example is the jet produced by flow through an orifice at sufficiently large Reynolds numbers, into a region of stationary or slowly moving fluid. Because of the high injection velocities of the methane gases relative to the chamber gases at the injection face, it can be assumed that the jet flow is turbulent. The jet from the annular fuel orifice approximated as a two-dimensional jet.

Certain simplifying properties are required for the analytical solution of jet flow. The mainstream flow is in the direction of the jet axis and the mean velocity component at right angles to the axis is found to be very small by comparison with the mainstream velocity itself and can sometime be neglected. At the same time, gradients measured at right angles to the axis are greater than those along the axis, similar to the boundary layer approximation. Turbulent jets show a small but significant decrease in mean static pressure relative to that in the surrounding fluid, because of the turbulent velocity fluctuations.

The variation of mean velocity across sections of fully developed jet flow is usually found to exhibit near geometrical similarity. This implies that the mean velocity distribution and the turbulent structure on which it depends are completely determined by a velocity scale and a length scale. These scales are the maximum mean velocity in a section, and the local "half-width" given by the distance from the axis at which the mean velocity falls to half its maximum value in the section. A further assumption which has experimental support is that of self-preservation which implies a continuing similarity in turbulent structure during the decay or development of turbulent flow. Such a situation is to be expected in a steady-state

brought about by the equilibrium between viscous decay and the generation of turbulent energy through turbulent shear stresses. These shear stresses depend on mean velocity gradients which are determined by the turbulent structure upstream which is then transported in the mainstream direction. Such an argument supposes the local Reynolds number remains sufficiently large for direct viscous effects on the mean velocity to be negligible.

When a fluid of uniform flow discharges through a two-dimensional or circular orifice into a fluid at rest, an initial core of potential flow is formed. This core is separated from the surrounding fluid near the orifice by two vortex sheets in two-dimensional flow or by a cylindrical vortex sheet in axially symmetric flow. These vortex sheets diffuse and become unstable in the axial direction forming free mixing layers which penetrate further and further into the potential core. As a result, the core of a two-dimensional jet persists only for about six orifice diameters downstream. The fully turbulent region of a jet is separated from the surrounding fluid by a viscous superlayer whose erratic movements are caused by the large-scale eddies in the turbulent motion. The mean position of the various boundaries in a turbulent jet are shown in the sketch below.



The equations of motion (Reference 2) for a two-dimensional incompressible jet are

$$U \frac{\partial u}{\partial x} + V \frac{\partial u}{\partial y} = \frac{1}{\rho} \frac{\partial \tau}{\partial y} \quad (1)$$

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \quad (2)$$

and for conservation of momentum flux,

$$\rho \int_{-\infty}^{+\infty} U^2 dy = \text{constant} \quad (3)$$

Assuming $U_{\max}$ denotes the mean velocity at the axis of the jet and b a measure of its width such as the half width, the similarity assumption gives the equations

$$\frac{U}{U_{\max}} = f\left(\frac{y}{b}\right), \quad \frac{\tau}{U_{\max}^2} = g\left(\frac{y}{b}\right), \quad (4)$$

where f and g are unknown functions. Writing

$$U_{\max} \sim x^{-n} , \quad b \sim x^m \quad (5)$$

then

$$\frac{\partial U}{\partial x} \sim x^{-n-1} , \quad \frac{\partial \tau}{\partial y} \sim x^{-2n-m} \quad (6)$$

Equations (1), (2) and (3) are then independent of x provided $m = 1$, $n = 1/2$, so that $U_{\max} \sim x^{-1/2}$, $b \sim x$ and $U_{\max} b / \nu \sim x^{1/2}$. These results hold only in the fully developed region beginning slightly downstream from the end of the potential core.

The transverse distribution of mean velocity is in good agreement with a Gaussian law with the exception of discrepancies near the edges of the jet. The transverse distribution of the mean velocity is plotted in Figure 2 taken from Reference 3. As shown above, the maximum centerline velocity, $U_{\max}$, is proportional to $x^{-1/2}$. This result is important in that it indicates how rapidly the velocity of a free jet can decay. The proportionality constant has been determined by Tollmien (Reference 4) based on Prandtl's mixing length theory. The "constant" is not truly a constant and depends upon density, thermal and velocity differences; however, these refinements are not important in a primary sense. Of primary interest in the injector design is the $U(x, y)$ distribution of the jet. The maximum centerline velocity is given by

$$U_{\max} = 2.37 \frac{U_j}{\sqrt{n}} \quad (7)$$

where

$$x = n d_j$$

It is apparent from this relation that decay of the centerline velocity, $U_{\max}$, does not begin until $n > 5.6$. The $u(x, y)$ distribution for the two-dimensional methane jet is shown in Figure 3 based upon the generalized curve presented in Figure 2.

The methane jet velocity at the orifice exit is a function of the ratio of injection pressure and chamber pressure. Figure 4 illustrates the effect of injection pressure ratio on methane injection velocity, U_j .

The above results were used to determine the velocity of the gaseous methane as a function of distance from the injector orifice and for several injection velocities. The velocity distribution is required to find the effect of injection parameters on atomization.

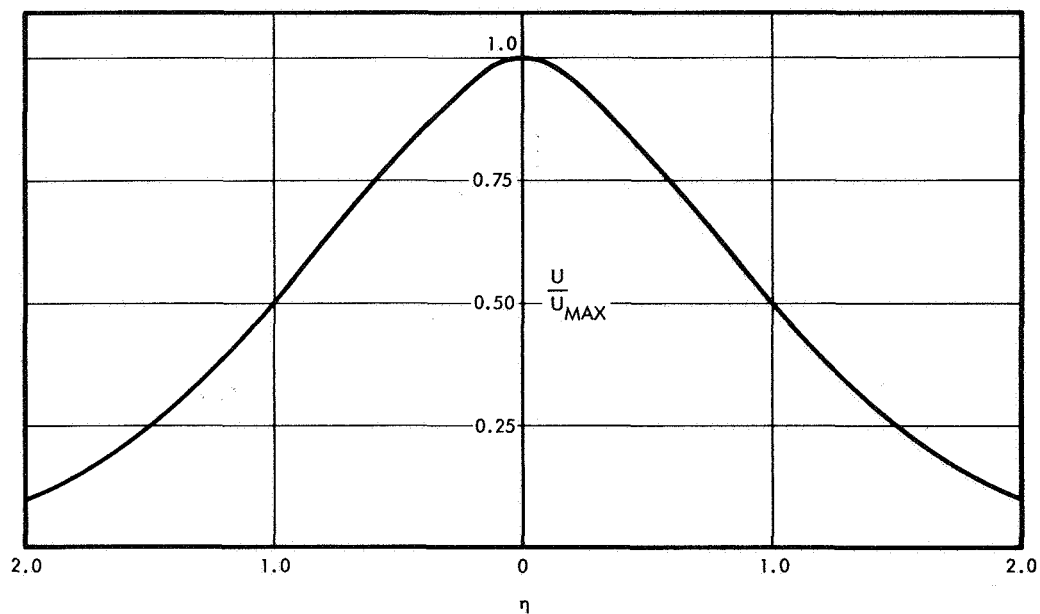


Figure 2. Universal Velocity Distribution for Incompressible, Constant-Pressure Turbulent Jets

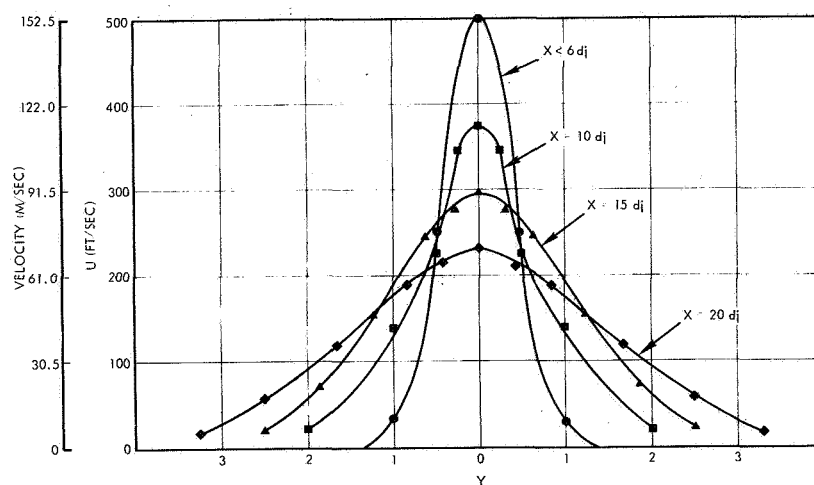


Figure 3. Axial and Longitudinal Velocity Distribution of Methane Jet

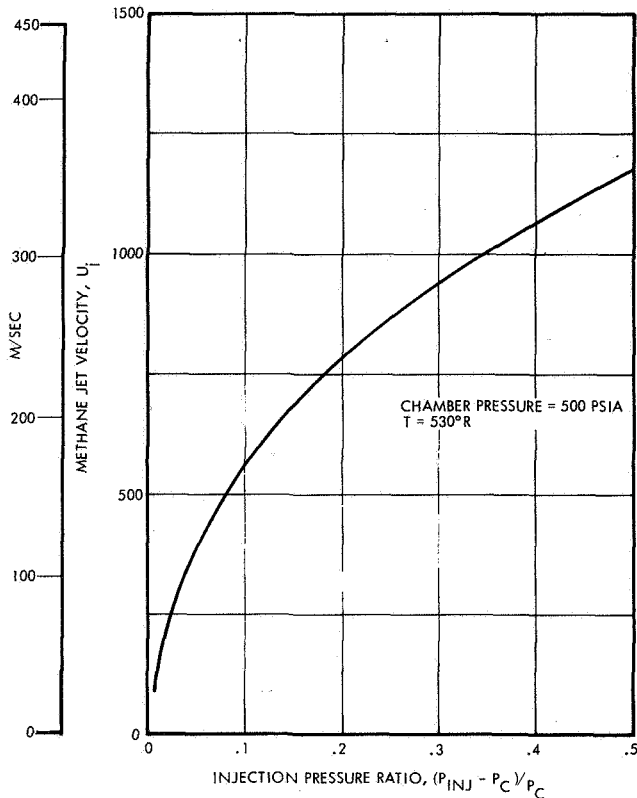


Figure 4
Effect of Injection Pressure Ratio on Injection Velocity for a Methane Jet

3.1.2 Propellant Atomization

Propellant atomization, through its control of the vaporization rate, is a major factor in injector performance. It was, therefore, necessary to investigate the effect of the injector design parameters (velocity, geometry, etc.) on FLOX atomization prior to conducting a combustion analysis.

The impingement of the radial liquid FLOX streams with the axial gaseous fuel sheet was analyzed using the methods of Ingebo (References 5 and 6) and Clark (Reference 7). Both Ingebo and Clark have conducted experiments to determine drop size distributions for cross-current breakup of jets. The most suitable data for application to the single element coaxial injector is probably that of Ingebo (Reference 6). In this work, Ingebo developed an empirical equation for correlating the liquid drop size with the orifice diameter. The equation developed by Ingebo is:

$$\frac{D_{30}}{D_o} = 3.9 \left(\frac{We}{Re} \right)^{.25} \quad (8)$$

where

D_{30} = volume-median droplet diameter

D_o = injector orifice diameter

We = Weber number, $\sigma / \rho_g V_g^2 D_o$

Re = Reynolds number based on orifice diameter, $D_o V_g / \nu$

Equation (8) represents the volume number mean drop diameter which should occur in a uniform gaseous CH_4 velocity duct with no chemical reactions. This equation represents the smallest mean drop size which could be obtained with the given velocity and enough gas flow to complete atomization without depletion of gas stream momentum.

Figure 5 is a plot of volume median droplet diameter as a function of gas Mach number and axial distance from the fuel orifice for liquid FLOX-gaseous methane at 500 psia (3447 kN/m) chamber pressure. The abscissa of this figure is the injection Mach number at the fuel orifice exit. The data shown in this figure are for an oxidizer orifice diameter of 0.050 inch (0.127 cm). The effect of the attenuation of the free jet velocity downstream of the injection orifice is also illustrated. Use of Equation (8) to estimate the volume number mean droplet diameter results in estimates of D_{30} which are less than approximately 25 microns for the oxidizer orifice diameter and injection Mach numbers of interest.

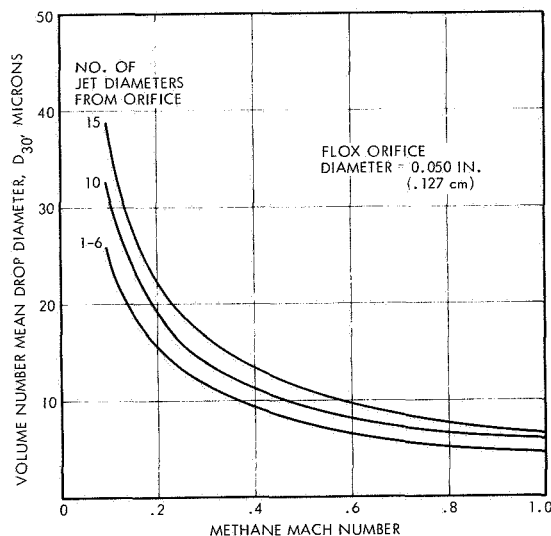


Figure 5

Effect of Methane Injection Mach Number and Longitudinal Distance on Volume Number Mean Drop Diameter

The fuel is injected axially through an annular orifice and therefore considered a two-dimensional jet. It has been shown in this section that the jet velocity is attenuated in proportion to the inverse of the square root of the distance from orifice. Since the centerline jet velocity decays in proportion to $x^{-1/2}$ rather than x^{-1} as with round jets, the effect on atomization of moving the impingement further from the orifice exit is much less. Figure 5 shows that the effect of distance from fuel orifice on droplet diameter is quite small in the region of interest. As is also evident from this figure, increasing the injection Mach number above about 0.4 does not appreciably decrease the oxidizer droplet diameter. The mean droplet diameter predicted by Equation (8) and shown in Figure 5 results in estimates of D_{30} which are less than approximately 40 microns at gaseous methane Mach numbers of 0.1 or higher. Experimental efforts (Reference 8) have indicated larger droplet diameters than those predicted by Equation (8). This is caused by the limited gas momentum which is

rapidly diminished through interchange with the radial stream, so that some of the liquid is atomized in a significantly weakened gas flow region. This results in larger droplets than for the experimental conditions where the gas stream flow rate is quite large compared to the liquid flow rate.

The range of droplet diameters calculated using the empirical relations shown in Figure 5 were then used to determine the effect on combustion performance. A combustion analysis was conducted based upon the model outlined by Priem and Heidmann (Reference 9). This model is a vaporization-limited combustion model in which the combustion efficiency is a function of the percent of the liquid which is vaporized. Figure 6 presents the theoretical relationship between combustion efficiency and the percent of FLOX vaporized. The percent of liquid vaporized is related to the initial drop size as shown in Figure 7. As combustion chamber length (the length downstream of the initial drop size) is decreased from 3 inches (7.62 cm) to 1 inch (2.54 cm) at a chamber pressure of 500 psia (3447 kN/m²), the percent of FLOX which is vaporized is predicted to decrease from 100 to approximately 25 percent for an initial drop diameter of 250 microns. For drop diameters below approximately 65 microns, all of the liquid is vaporized within chamber lengths of 1 inch (2.54 cm) or longer. Therefore, if combustion performance is limited by vaporization of the FLOX, a unique relationship should exist between characteristic velocity efficiency and some measure of drop size (volume mean, mass mean, etc.) at some distance from the injector face for a particular chamber configuration and propellant combination.

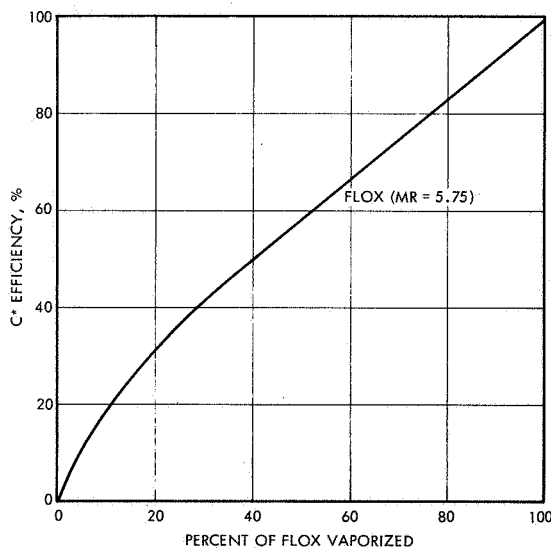


Figure 6. Theoretical Relationship Between C* Efficiency and Percent of FLOX Vaporized

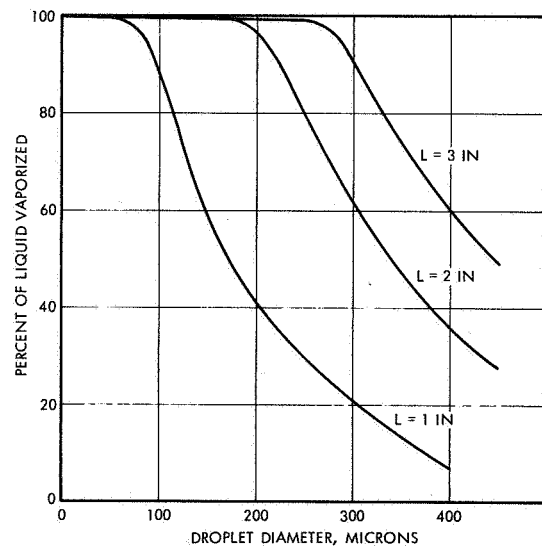


Figure 7. Effect of FLOX Droplet Size and Combustion Chamber Length on Percent of FLOX Vaporized

The calculated mean droplet diameter as calculated from Equation (8) was used to determine the combustion efficiency using the vaporization model described. Thus, if a liquid-vaporization-limited combustion model is assumed, and if maximum drop diameter is 65 microns or less, then a combustion efficiency approaching 100 percent would be obtained if a combustion length of at least 1 inch (2.54 cm) downstream of this initial drop diameter were available.

Under this assumption, these estimates indicated that:

- (1) If sufficiently small (less than about 65 microns) maximum drop diameters can be obtained using CH₄ gas to atomize the liquid, then C* efficiencies approaching 100 percent (neglecting friction and heat transfer loss to the walls) can be achieved in combustion chamber lengths of a few inches.
- (2) The TRW gas-liquid injector design provided for maximum liquid atomization.
- (3) Primary factors which contributed to losses in C* efficiency were mass and mixture ratio maldistribution.

If the droplet diameters as calculated by Equation (8) were larger by a factor of 3 as indicated in Reference 8, then nearly 100 percent C* efficiency could still be achieved in a chamber length of approximately 2 inches (5.08 cm).

3.1.3 Injector Design Model

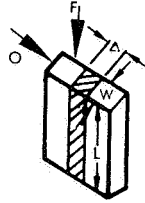
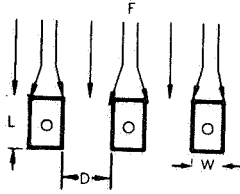
This section discusses the approach to coaxial injector optimization through the control of the hydraulic design parameters. The single element coaxial pintle injector concept, unique for the elimination of damaging instability through geometry, embodies many of the features of more conventional injectors. The injector design has performance characteristics dependent upon momentum ratios quite similar to conventional injectors.

The basic atomization and mixing control for the pintle design is in the control of the annular fuel sheet and the radial oxidizer element sizes along with the use of kinetic energy of jet interaction. A similar process occurs for multielement impinging jet injector designs.

An effective approach to this control is obtained through an examination of the gross dynamics of the interaction of fuel and oxidizer. Maximum mixing can be achieved only by optimum use of the available stream momenta (the product of density, area and the square of the velocity). It has been shown by Rupe (Reference 10) that for optimum mixing of two impinging streams, the ratio of fuel force to oxidizer force is unity or

$$\frac{F_f}{F_o} = 1$$

For the central element coaxial injector configuration, the proper areas for the fuel and oxidizer are obtained through an examination of the initial interaction mechanics. The following basic interaction model was developed (Reference 11):



The fuel force on the oxidizer is taken as

$$F_f = \rho_f V_f^2 [\Delta W + 2 \Delta LC]$$

where the symbols are defined in the foregoing sketch and C is a cross influence term to account for side interaction between the oxidizer streams.

On the basis of a single reflective interaction between elements for the fuel, a first approximation to C is taken as

$$C = \left(\frac{L}{V_f} \right) \left(\frac{a}{D} \right) \left[1 - \frac{ND}{\pi D_i} \right]$$

which is seen to be an estimate of the ratios of time of flight of the fuel to the time of dispersion of the two-phase reactants. The value of a, which is a measure of the velocity of a disturbance, is a weak function of mixture ratio and varies from about 600 (183 m/sec) to 800 ft/sec (244 m/sec) for combustible mixtures.

The oxidizer force on the fuel sheet is taken as

$$F_o = \rho_o V_o^2 WL$$

The final momentum ratio for optimal momentum utilization of the slotted configuration becomes

$$\frac{\rho_f V_f^2 [\Delta W + 2 \Delta LC]}{\rho_o V_o^2 WL} = 1$$

Central element pintle injector designs based upon the above momentum ratio design criteria provide peak performance at the optimum momentum ratios as defined above. That the central element pintle design will in fact show peak performance for a given design obeying this type of behavior is illustrated in the program results (Section 5). The ratio of oxidizer slot width, W, to unit width, W + D, is commonly called the blockage ratio.

It has been demonstrated that the blockage ratio is an important parameter in the design of the coaxial pintle injector, affecting both performance and resultant chamber wall environment (Reference 12).

3.2 THERMAL ANALYSES

Thermal analyses were conducted to determine the pretest injector temperatures and conditioning necessary to assure unrestricted operation at nominal full thrust conditions. The injector design was analyzed to determine the steady-state temperatures at the injector face and pintle tip during engine firing. Analyses of the thrust chambers were also conducted to determine theoretical thrust chamber wall heat flux as a function of engine operating conditions. The thermal analyses were used to define limits for operation with the heat sink chamber hardware.

3.2.1 Injector Thermal Analysis

A 55-node thermal analysis program (TAP) model was developed to investigate the potential problem areas of methane condensation and FLOX boiling in the 3K injector assembly during prefiring chilldown and engine firing conditions. Steady-state temperature distributions were determined for the following three cases:

- (1) LN_2 in the oxidizer side with stagnant air in the fuel side — effect of chilldown
- (2) LN_2 in the oxidizer side with warm GN_2 in the fuel side — effect of chilldown
- (3) FLOX and methane — effect of engine firing

In order to check the integrity of the pintle tip and the occurrence of FLOX boiling behind the tip, a 9-node model of the tip was formulated and run on ITAP (Implicit Thermal Analyzer Program) (steady-state) and SNAP (Systems Network Analyzer Program) (transient).

3.2.2 Thermal Model

The thermal model of the 3K injector assembly is shown in Figure 8. The injector volume was divided into 44 nodes (numbered 1-44). Six fluid nodes (70-75) were assigned to the gas in the fuel side, number 70 being a constant temperature entrance node. The cryogenic liquid in the oxidizer passages (node 60) was assumed to remain at a constant temperature. Two constant temperature nodes were assigned to the chamber wall, one as a radiation source from the throat area (node 80) and one for the cooled section near the injector (node 81). Other constant temperature nodes were used for the ambient air external to the injector (node 51) and for the flame temperature (node 50).

Flame temperature for combustion of FLOX and methane at a chamber pressure of 500 psia (3477 kN/m^2) was assumed to be 7720°F (4271°C). The chamber wall near the throat was assumed to be a radiation heat source operating at 4400°F (2427°C). Convection from the flame to

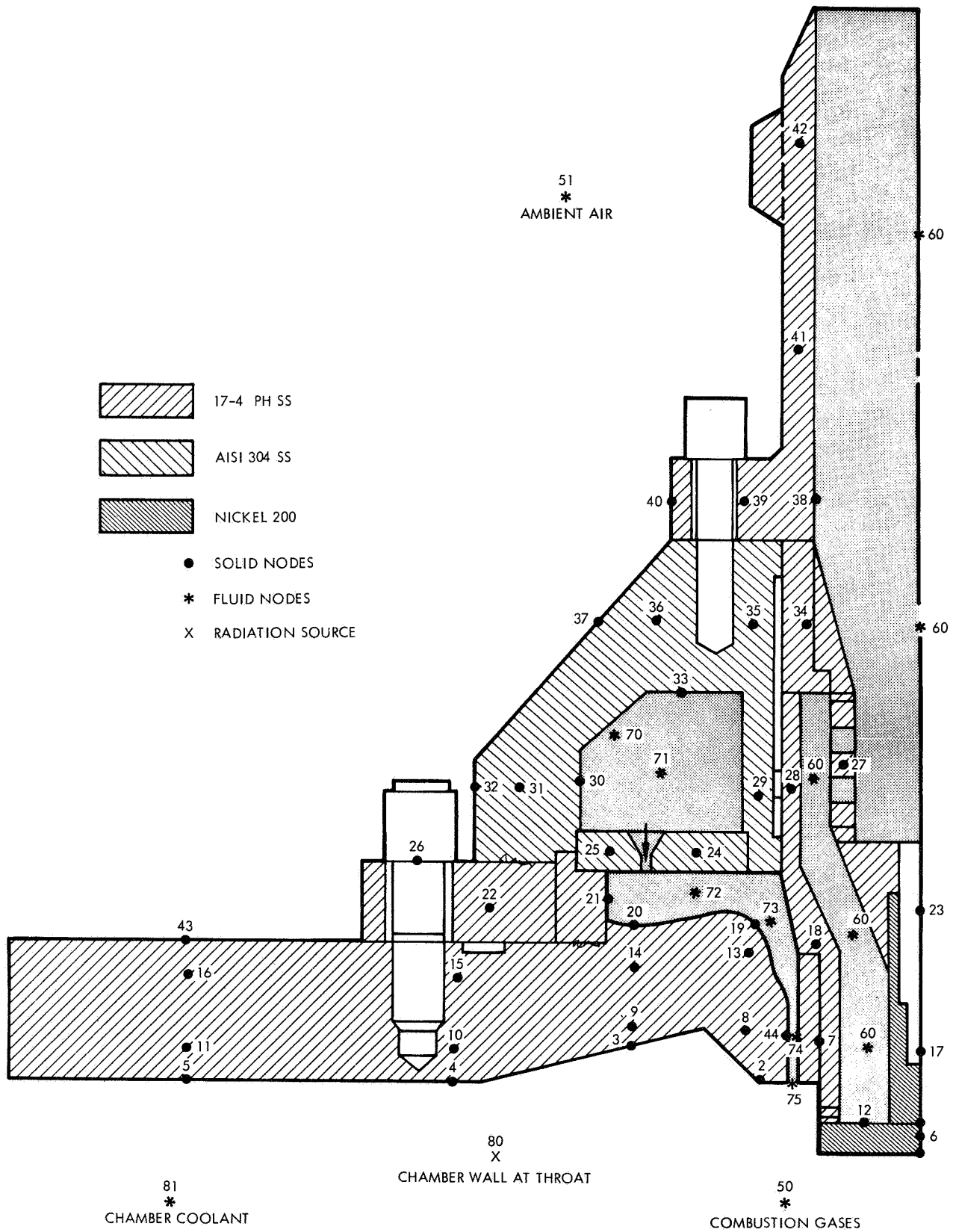


Figure 8. Thermal Model (TAP) of the 3K FLOX/CH₄ Injector Assembly

the pintle tip was simulated with the heat transfer coefficient assumed to be 1/4 that at the throat. Convection to the injector face was not considered.

The first case, chilldown with LN₂ and stagnant air, showed excessively low temperatures around the fuel flow path. The inner half of the manifold would be below -180°F (-118°C) and the inside of the fuel orifice would be nearly -320°F (-196°C). Since methane at 550 psia (3792 kN/m²) condenses at -116°F (-82°C), these temperatures would result in excessive methane condensation during start-up. Temperatures of the principal nodes for this case are shown in Figure 9.

Chilldown with LN₂ and ambient GN₂ showed low temperatures along only a small section of the fuel flow path. The manifold will be above 5°F (-15°C), while the inside of the orifice will be -137°F (-94°C). This would cause condensation of negligible amounts of methane. Principal temperatures are shown in Figure 10 for this case.

For the case of steady-state engine firing with FLOX and methane, the analysis indicated practically no methane condensation. Temperatures around the manifold will range from 45°F (7°C) to 108°F (42°C). The inside of the fuel orifice was calculated to about -119°F (-84°C). A minor amount of FLOX boiling could occur on the back side of the pintle tip. Principal temperatures for this case are shown in Figure 11.

The results showed that warm GN₂ is necessary on the fuel side during chilldown and that there will be negligible amounts of FLOX boiling and methane condensation during firing.

In the detailed analysis of the pintle tip, it was found that film boiling will occur on the back side of the tip, but the heat transfer caused by film boiling is negligible compared to that caused by forced convection.

Detailed analysis of the pintle tip (Figure 12) showed a maximum tip temperature of about 1450°F (788°C) during FLOX/methane steady-state engine firing at a chamber pressure of 500 psia (3447 kN/m²). For this case, the effective FLOX temperature was assumed to be at the boiling point of -200°F (-129°C) at 500 psia (3447 kN/m²).

Injector materials used in the analysis were AISI304 and 17-4PH stainless steels with a nickel 200 pintle tip. The fluid entering temperatures, pressures and flow rates were as follows:

	Temperature		Pressure		Flow Rate	
	(°F)	(°C)	(psia)	(kN/m ²)	(lb/sec)	(kg/sec)
FLOX	-320	-196	500	3447	7.9	3.58
Methane	70	21	550	3792	1.37	0.62
LN ₂	-320	-196	30	207	2.0	0.91
GN ₂	70	21	125	862	0.756	0.34

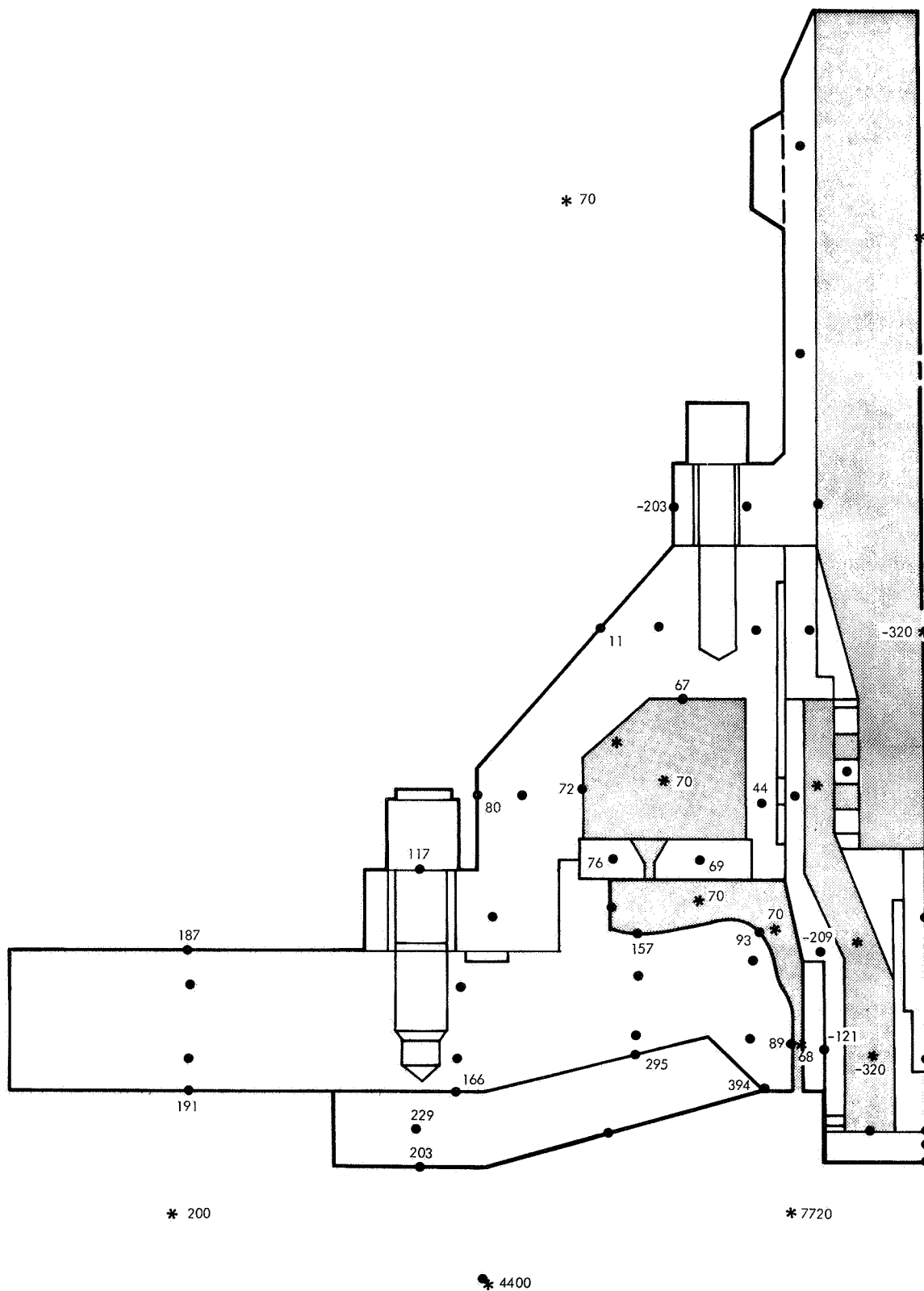


Figure 11. Injector Temperature Distribution –
FLOX/CH₄ Engine Firing

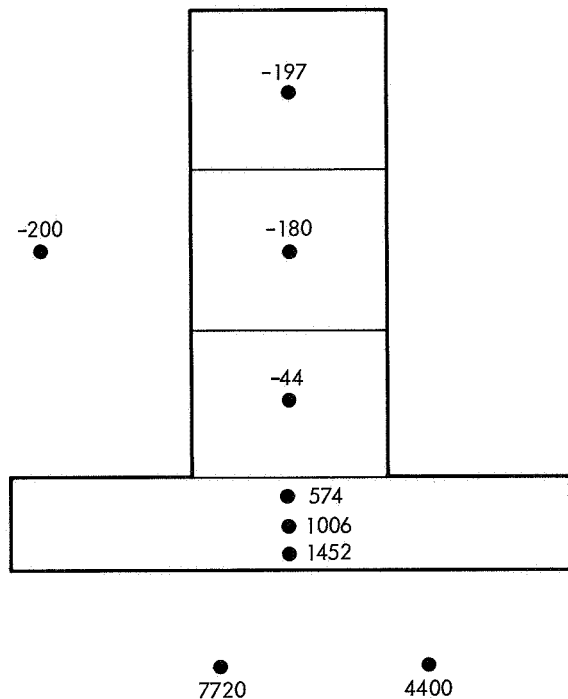


Figure 12. Thermal Model of
Injector Pintle Tip

Heat transfer by forced convection to the cryogenic liquids (FLOX or LN_2) was assumed to be to a constant liquid bulk temperature. The heat transfer coefficient was calculated using equivalent hydraulic diameters at each station from the relationship:

$$N_{\text{NU}} = \frac{hD_e}{k} = .023 \left(\frac{UD_e \rho}{\mu} \right)^{.8} \left(\frac{\mu C_p}{k} \right)^{.4}$$

Convection to or from the gases (methane or GN_2) was computed by using the forced convection function in TAP, thus accounting for the change in gas temperature as it flows through the injector. Heat transfer coefficients for these cases were calculated from the relationship:

$$N_{\text{NU}} = \frac{hD_e}{k} = .023 \left(\frac{UD_e \rho}{\mu} \right)^{.8} \left(\frac{\mu C_p}{k} \right)^{.3}$$

Free convection from the injector to the ambient air was estimated from the relationship:

$$h = 5.6 \times 10^{-7} \left(\frac{\Delta T}{L} \right)^{.25}$$

3.2.3 Thrust Chamber Thermal Analysis

To satisfy the conditions of short duration performance evaluation tests and long duration firings, two thrust chamber concepts were employed.

The program utilized a multisection instrumented heat sink thrust chamber for the performance evaluation tests. The chamber was designed so that characteristic length (L^*) variations could be achieved by interchanging the cylindrical sections with a common nozzle section. The streak tests were also accomplished by replacing the nominal 30-inch L^* (76.2 cm) heat sink section with an ablative section.

An additional thrust chamber fabricated entirely of G-90 graphite was designed to accomplish the long-duration run. This chamber did not have the instrumentation which the performance evaluation chambers had.

Thermal analyses of the thrust chambers were conducted to evaluate wall operating temperatures as a function of the theoretical heat flux. The thermal analyses defined duration limits for operation with the copper cylindrical sections and graphite nozzle. The heat sink cylindrical chamber sections were fabricated from OFHC copper and the nozzle section from G-90 graphite.

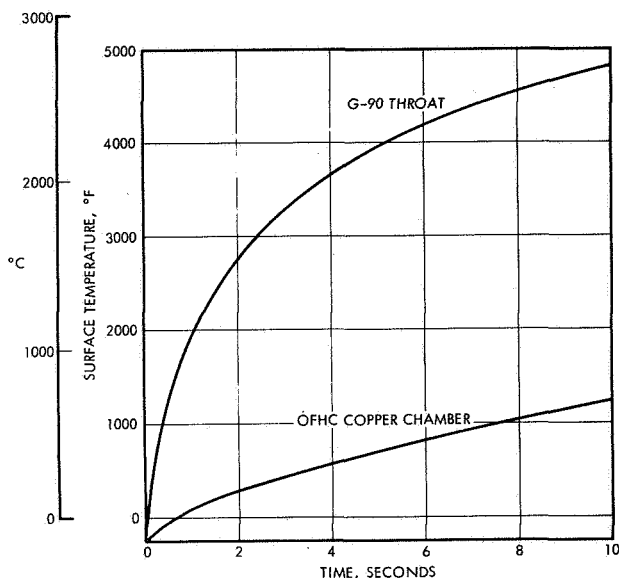


Figure 13. Effect of Firing Duration on Surface Temperature for the Graphite Throat and Copper Cylindrical Section

The OFHC copper cylindrical section/graphite nozzle and all graphite chamber configurations were evaluated under "worst" case (from a thermal standpoint) conditions. The worst case condition is operation at full theoretical combustion temperature with no carbon deposition from the methane. The resulting inner wall thermal response for the copper cylindrical section and the graphite nozzle is shown in Figure 13 (assuming an initial hardware temperature of -250°F (-157°C)). Figure 13 indicated that both the copper cylindrical section and the graphite nozzle were adequate for the short duration performance evaluation tests under worst case conditions. For the long duration firing (up to 60 seconds), only the all G-90 thrust chamber was analyzed.

The G-90 thrust chamber assembly was analyzed considering both the case with and without carbon deposition. A substantial reduction in heat flux caused by carbon deposition using LPG fuels was reported by Pratt & Whitney Aircraft Co. in Reference 13. Carbon deposition and corresponding heat flux reduction was verified by TRW Systems on the Space Storable Propellant Performance program (NAS3-11200). The correlation for carbon deposition developed in Reference 13 was used to predict the throat surface temperature for the thrust chamber conditions of this program. The G-90 throat surface temperatures obtained from this analysis are shown in Figure 14 at the end of a 60- and a 10-second firing for combustion gas temperatures from 75 to 100 percent of the theoretical combustion temperature. Also shown in Figure 14 is the effect of carbon deposition on throat surface temperature.

3.3 STABILITY CONSIDERATIONS

The application of the space storable propellants to primary propulsion engine design results in combustion chamber sizings which may support acoustic modes of instability. The tangential, radial, and

longitudinal modes are of concern; the tangential modes are most important (Reference 14). The single-element coaxial injector provides a fundamental approach for theoretical elimination of these modes.

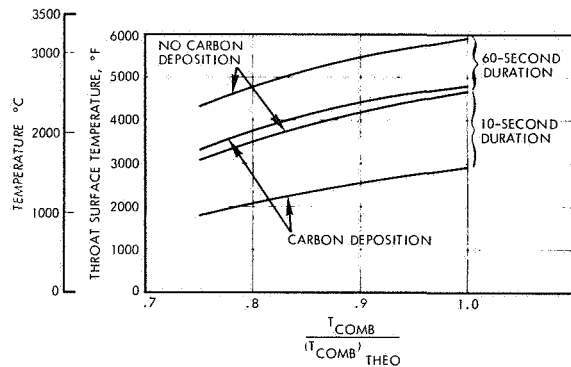


Figure 14
Effect of Recovery Temperature and Duration on Throat Surface Temperature for a Graphite Chamber

To arrive at a means to theoretically eliminate combustion instability it is necessary to examine the relationship between the nodes and antinodes of the acoustic vibrations associated with various instability modes and the location of possible abnormally high energy release zones which could result in sustaining combustion instability. These are shown in Figure 15 indicating comparisons between distributed injection and centralized injection.

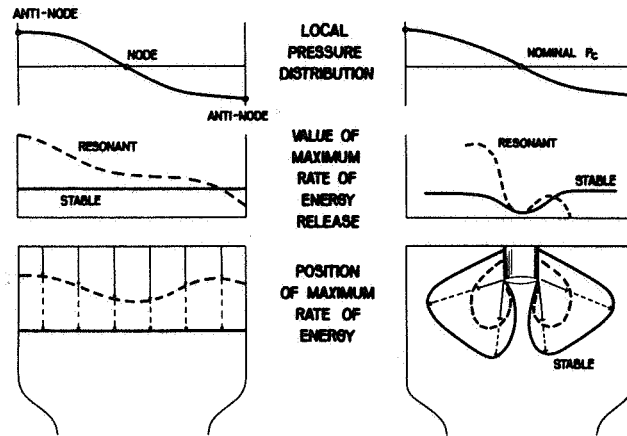
The figure shows that the possible locations for abnormally high energy release in the chamber can have a dominant effect on whether or not the engine will be dynamically stable (without resorting to mechanical damping). For example, if a large local pressure pulse is generated in a region in which the maximum compression (pressure antinode) would occur for a given acoustical mode, then that mode will probably be initiated. Inert gases, the energy supplied by the pulse will be dissipated in a few cycles through various damping processes such as turbulence, wall drag, droplet drag, and possible interference from other vibration modes which may also have been excited. However, if unburned propellant is introduced into the pressure antinode regions, the energy required to sustain the oscillations can be supplied through a rapid increase in the normal rate of combustion caused by pressure-sensitive gas reactions and increased mixing, resulting from high-velocity gases just behind the pressure front.

An injector design which introduces propellant near the outer periphery of the combustion chamber can be expected to be very susceptible to the tangential modes of instability, whereas an injector having a single central element would be highly resistant to the tangential modes if fast mixing and reaction are assured by proper interlocking of oxidizer and fuel.

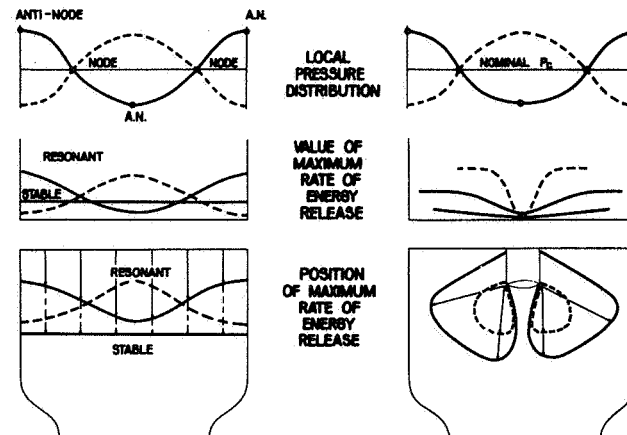
In Figure 15 conventional multiple-orifice injection is illustrated on the left side. Under stable operation, the maximum rate of energy release zone will be somewhat as shown by the solid line. Under the influence of

Distributed Injection

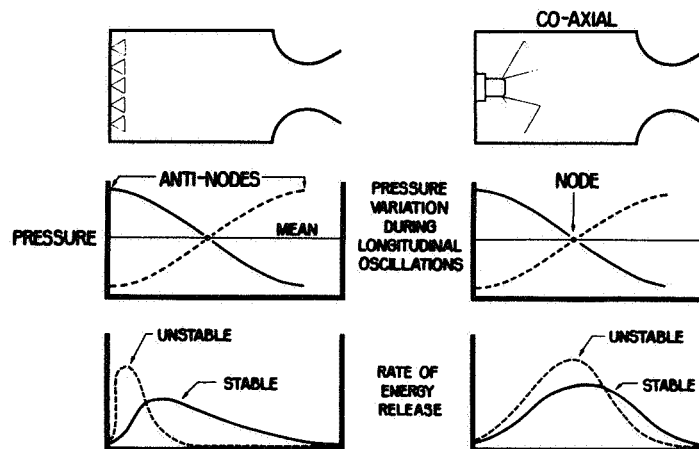
Central Injection



Tangential



Radial



Longitudinal

Figure 15. Comparison of Stability Modes

a large pressure pulse at one side of the injector (or subsequently during unstable operation when the pressure front has just passed the left side), the energy release pattern will approximate that shown by the dotted line. The zone of maximum energy release rate moves generally toward the injector face and even closer at the peak pressure region, thus sustaining the instability.

A schematic diagram of the coaxial injection scheme is shown on the right side of Figure 15, where it can be seen that under the influence of a large local pressure pulse, the zone of maximum energy release must again move toward the incoming propellants to increase the combustion rate above its normal value. In this case, it moves both toward the injector end of the chamber and inward toward the center line of the engine. It is apparent that, under these conditions, the energy would be released near the pressure node and would be very ineffective in sustaining the spinning or tangential pressure front. Therefore, any disturbance which initiates the tangential modes would be expected to damp out within a few cycles.

The very high injection density near the center of the injection in the TRW design also tends to prevent the establishment of the radial mode of instability. On the right side of Figure 15 it can be seen that for the coaxial element, the normal maximum energy release zone is near the pressure node of the radial mode, and therefore does not tend to contribute effectively toward sustaining the mode. With a distributed injection some incoming propellant is at the antinode of the radial mode as shown on the left side of Figure 15. However, in any case the radial mode is not easy to sustain.

With respect to the longitudinal mode of instability, it is seen in Figure 15 that energy added close to the injector face will be most effective in sustaining the mode. For a given energy release rate, there is a maximum chamber length below which the energy is, in effect, being released at a pressure node, and therefore will not sustain longitudinal instability. However, the energy release rate is not constant, but is usually a strong function of the pressure disturbance. A high pressure front can "drive" the combustion zone back toward the injector face until the propellant injection density and/or lack of mixing is sufficient to prevent any further increase in combustion rate. For injectors using a distributed propellant source such as the one shown on the left in Figure 15, the propellant density does not rapidly increase until very close to the injector face. Therefore, a large pressure disturbance could drive the energy release zone close to the head end of the chamber or into the antinode region and thus establish a longitudinal instability.

The coaxial injector resists the longitudinal mode of instability because the propellant injection density increases much more rapidly and to much higher values at a given distance from the injection ports than in injectors with a distributed source. Furthermore, the injection point is physically adjustable farther into the chamber. Thus, for a given chamber length, the stable maximum energy release occurs nearer the pressure node region and cannot be driven nearly as far toward the head end of the chamber (see the right side of Figure 15). Since it cannot couple efficiently in this region, the engine remains stable.

4. EXPERIMENTAL HARDWARE

The nominal engine design conditions for the test hardware were a sea-level thrust of 3000 pounds (13345 N), a chamber pressure of 500 psia (3447 kN/m²), and a mixture ratio of 5.75. Propellants were liquid FLOX (82.6% F₂) supplied at liquid nitrogen temperature and gaseous methane supplied at ambient temperatures.

4.1 INJECTOR DESIGN

The specific injector configuration was a "building block" version of a flight-type injector employing replaceable oxidizer injection elements. The building block approach provides flexibility to accommodate hardware modifications with minimum schedule/cost impact. The oxidizer injection metering slots were located in a replaceable ring structurally held between the pintle tip and the pintle sleeve which are bolted together. Oxidizer injection geometry variations can be effected by replacing metering (orifice) rings. Five variations of the oxidizer geometry were designed. The fuel metering orifice is an annular gap whose metering area can be varied by changing the pintle sleeve outside diameter or the bore diameter in the injector faceplate.

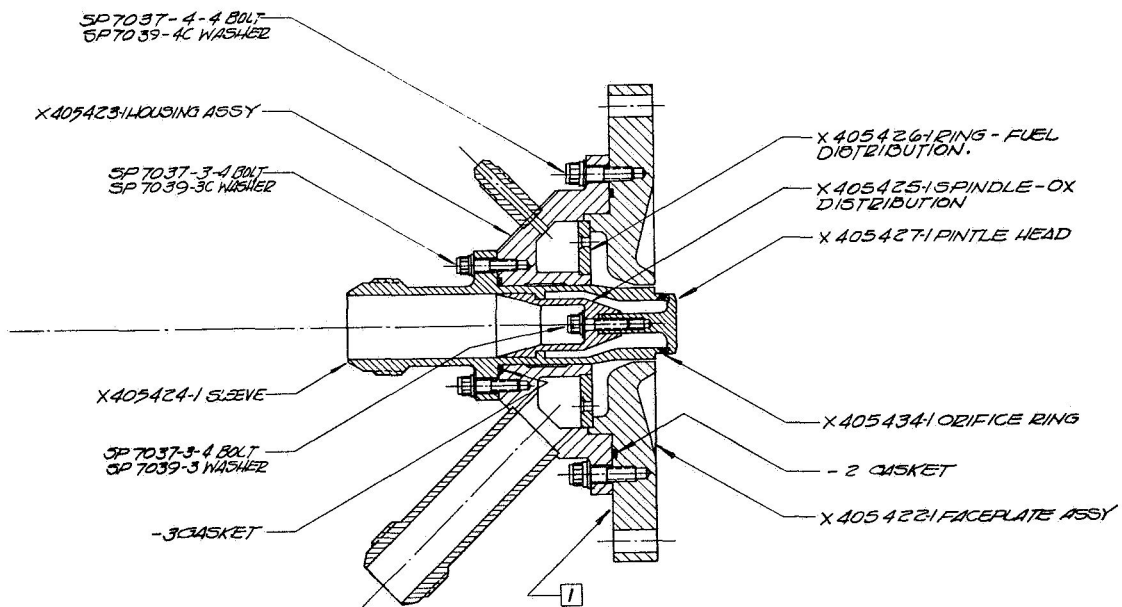
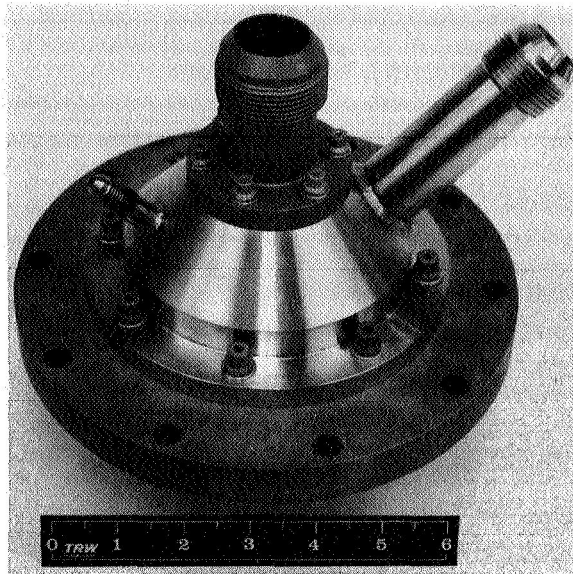
The injector is a bolt-up assembly using aluminum (1100-0) crush seals on both the oxidizer and fuel sides. Assembly and disassembly was accomplished rapidly by means of this bolt-up design. Figure 16 is an illustration of the basic injector assembly; Figure 17 is a disassembled view of the injector; and a drawing of the injector assembly is presented as Figure 18. Provision was made for measuring static (head-end) chamber pressure through a post located in the injector face.

The injector assembly consisted of two basic subassemblies; the fuel and oxidizer subassemblies. The oxidizer feed subassembly consisted of the oxidizer sleeve, the oxidizer distribution spindle, the orifice ring and the pintle tip. The retaining bolt was inserted through the distribution spindle and threaded into the pintle tip as illustrated in Figure 19. The oxidizer was distributed radially outward from the distribution spindle through 36 holes. The maximum flow velocity in the oxidizer feed subassembly was 40 ft/sec (12 m/sec).

The fuel feed subassembly consisted of the faceplate and the housing with the fuel distribution ring clamped between them. The fuel was distributed from the single inlet within the injector by means of a fuel distribution ring. The distribution was accomplished by 36 equally spaced holes. Water flow tests provided evidence of uniformity of fuel distribution.

The gaseous methane is injected into the chamber through an annular gap having a nominal width of 0.087 inch (0.22 cm).

The oxidizer orifice elements were rings fabricated from nickel 200. Figure 20 illustrates a typical oxidizer orifice ring.



① INJECTOR ASSEMBLY

Figure 16. 3K FLOX/Methane Injector Assembly (X 405421)

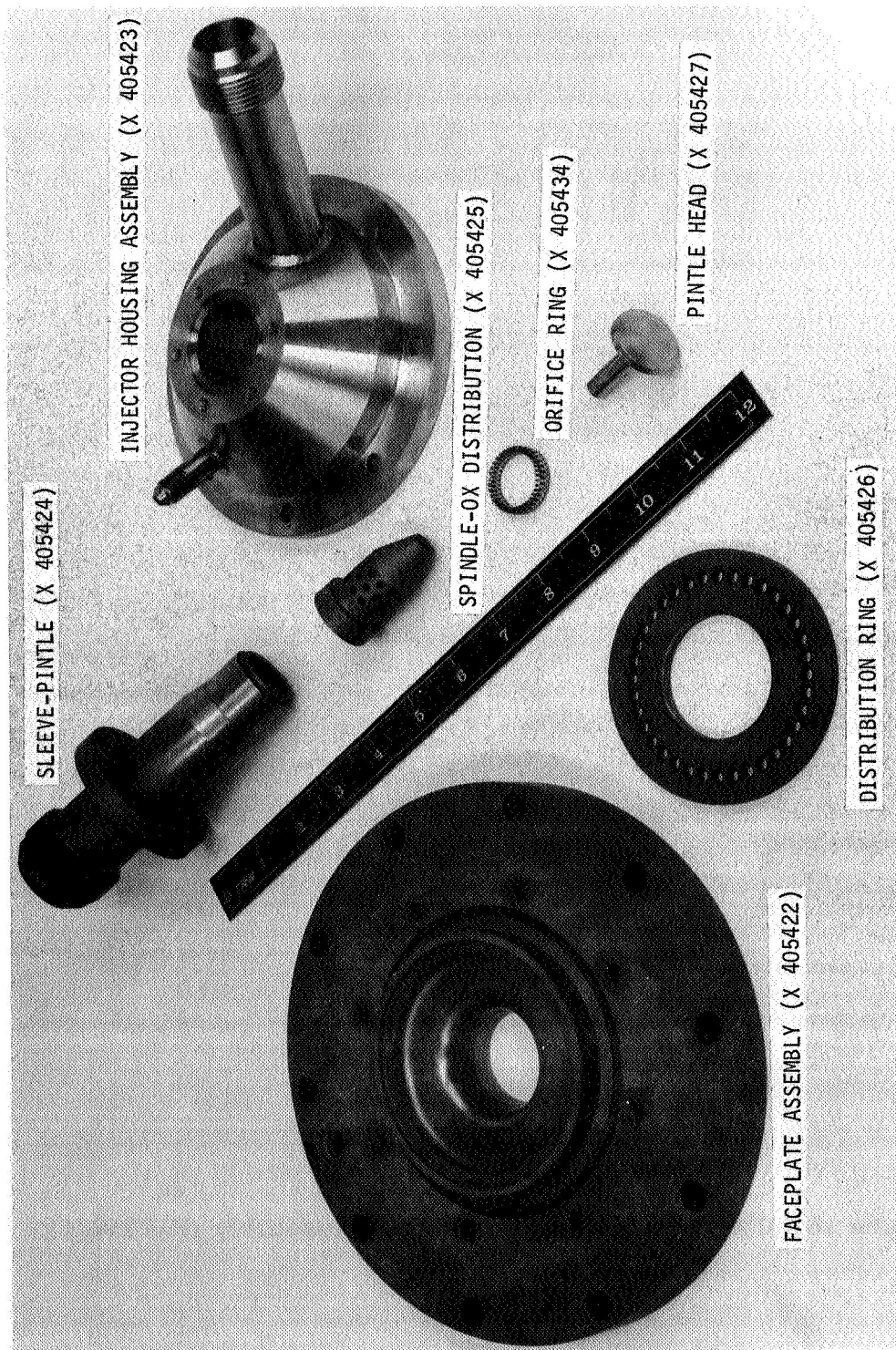
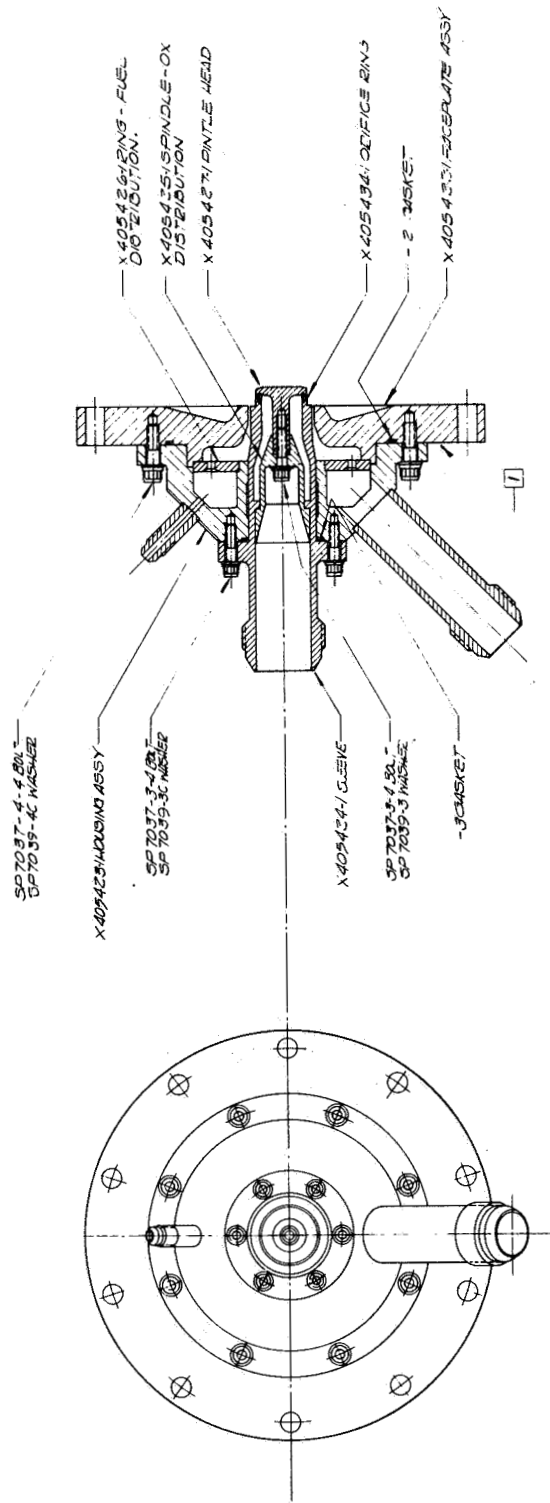


Figure 17. Disassembled View of 3K FLOX/Methane Injector



① INJECTOR ASSEMBLY

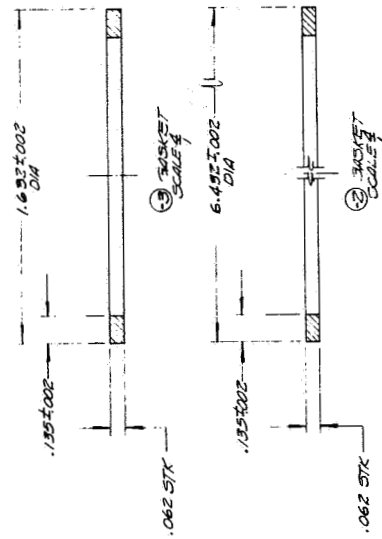


Figure 18. 3K FLOX/Methane Injector

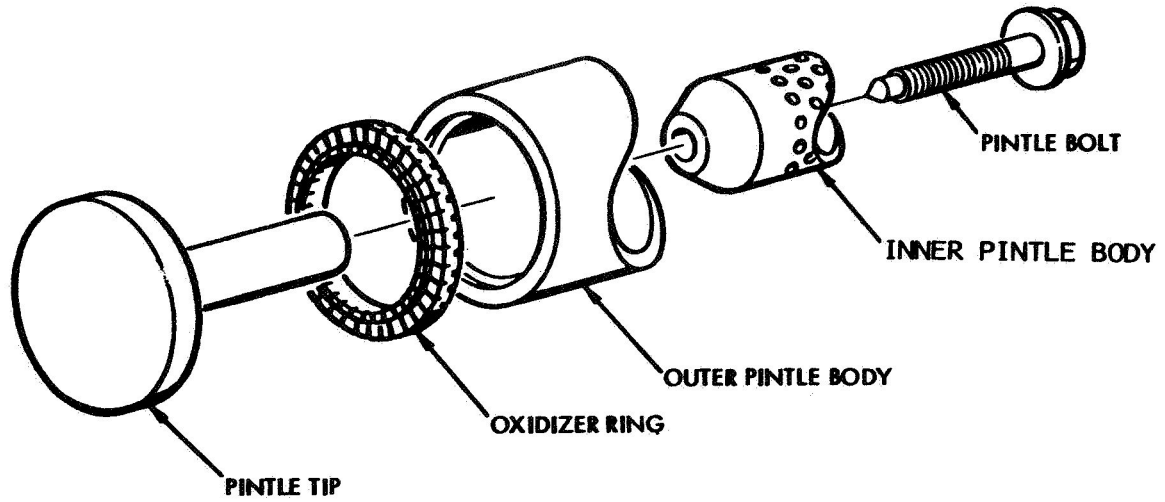


Figure 19. Pintle Tip - Oxidizer Orifice Ring Boltup Arrangement

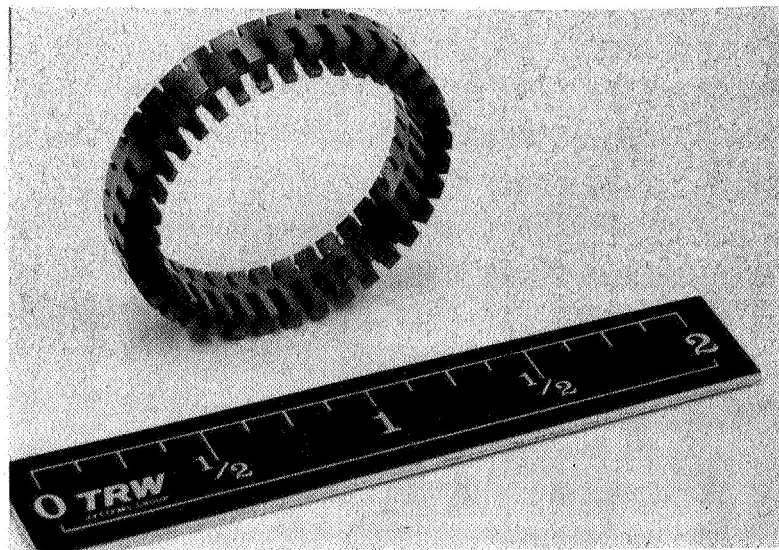
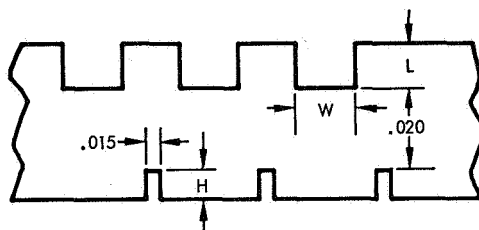


Figure 20. Oxidizer Orifice Ring for 3K FLOX/CH₄ Injector

Various slot geometries are accomplished by simply cutting slots of different dimensions in the ring. Table 1 lists the pertinent dimensions of the various oxidizer elements tested during the program. The oxidizer orifice ring could be changed by removing the oxidizer feed subassembly from the injector assembly, removing the pintle retaining bolt and withdrawing the pintle tip and oxidizer orifice ring and then reversing the process.

Table 1. Oxidizer Orifice Element Geometrical Characteristics



INJECTOR CONFIG.	NOMINAL PRESSURE DROP		RING DIAMETER		UNIT SPACING		W		L		H		[SLOT WIDTH UNIT WIDTH*] × 100
	LB/IN ²	kN/m ²	IN	CM	IN	CM	IN	CM	IN	CM	IN	CM	
-1 RADIAL	200	1379	1.00	2.54	.087	.22	.036	.091	.072	.183	.043	.109	50
-2 CANTED 30°	200	↓	1.00	2.54	.087	.22	.026	.066	.098	.249	↓	↓	40
-3 RADIAL	200	↓	1.14	2.88	.099	.25	.020	.051	.130	.330	↓	↓	34
-4 RADIAL	200	↓	↓	↓	↓	↓	.036	.091	.072	.183	↓	↓	50
-5 RADIAL	150	1034	↓	↓	↓	↓	.025	.064	.117	.297	.049	.124	39
-6 CANTED 30°	200	1379	↓	↓	↓	↓	.025	.064	.104	.264	.043	↓	39
-7 RADIAL	200	↓	↓	↓	↓	↓	.016	.041	.162	.411	.043	↓	30

*UNIT WIDTH AT IMPINGEMENT.

The oxidizer orifice ring could be changed with the injector mated to the thrust chamber on the test stand. This provided injector flexibility by enabling rapid oxidizer orifice changes while maintaining overall firing readiness of the hardware and the test facility.

4.2 INJECTOR FABRICATION

The injector assembly was fabricated of 304 stainless steel. Provision for thermally protecting the injector face from the hot combustion gases was accomplished by installing an MX 2625 silica phenolic liner against the injector face.

The injector fuel subassembly was sealed using 1100-0 aluminum crush seals between the faceplate and housing and between the housing and pintle sleeve.

The oxidizer feed subassembly was assembled from the oxidizer sleeve, distribution spindle, orifice ring and pintle tip. The sleeve and spindle were fabricated from 17-4 PH CRES while the orifice ring and

pintle tip were fabricated from nickel 200. The pintle sleeve utilized 17-4 PH CRES to eliminate galling when the oxidizer feed subassembly was inserted into the fuel feed subassembly.

The oxidizer orifice rings were fabricated from nickel 200. The basic blank ring was fabricated to size and then using a precision slotting saw, the slots were added in accordance with the desired configuration. For the canted ring configuration, the slots were formed by electrical discharge machining (EDM).

4.3 THRUST CHAMBER DESIGN

The thrust chambers to perform the experimental phase of the program were designed to the following conditions:

Chamber pressure	500 psia (3447 kN/m^2)
Thrust (sea level)	3000 lbf (13345 N)
Total propellant flow rate at $O/F = 5.75$	$9.3 \text{ lb}_m/\text{sec}$ (4.2 kg/sec)
Contraction ratio	4
Expansion ratio	5
Nozzle configuration	Conical - 15 degree half angle

The Task III thrust chambers were designed to give qualitative information on injector streaking while the Task IV performance evaluation chambers were primarily to give quantitative data on injector performance and chamber heat flux distribution as well as dynamic stability. A single additional chamber was designed to provide an extended duration capability.

4.3.1 Task III Thrust Chamber

The uncooled thrust chamber design utilized for Task III injector characterization tests is shown in Figure 21. The chamber was designed to deliver 3000 pounds (13345 N) sea-level thrust while operating at 500 psia (3447 kN/m^2) chamber pressure. The Task III thrust chamber assembly consisted of a silica phenolic cylindrical streak chamber contained within a steel shell and a graphite convergent-divergent nozzle section also contained within a steel shell. The nominal characteristic length (L^*) was 30 inches (76.2 cm). The inside diameter of the chamber was 4.53 inches (11.51 cm) which resulted in a contraction ratio of 4.0.

The nozzle incorporated a 30-degree convergence angle and a full radius throat blended into a 15-degree half angle nozzle. The nozzle expansion was truncated at an area ratio of 5.0.

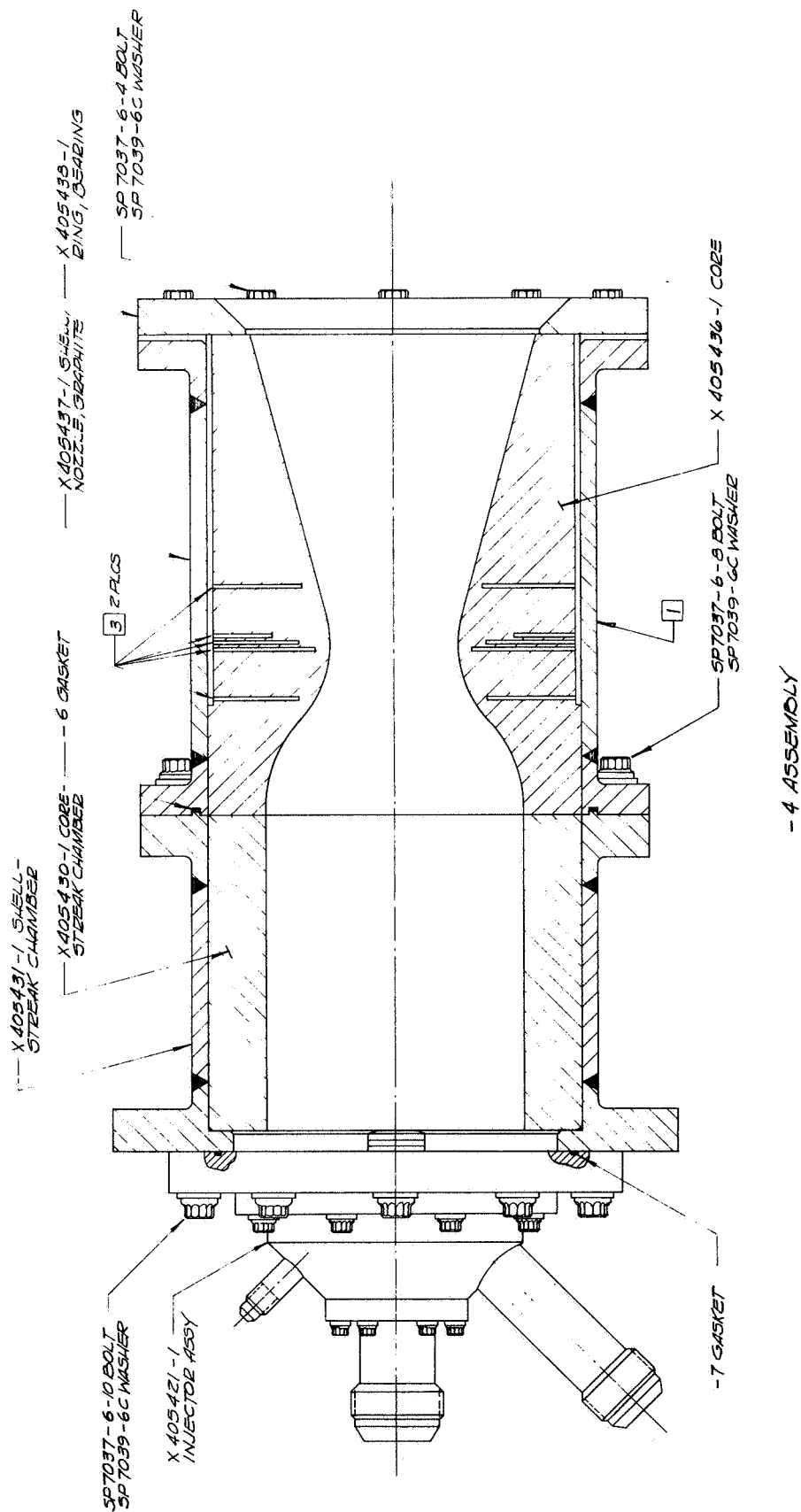


Figure 21. Engine Assembly—Streak Test

The injector was mated to the thrust chamber flange by means of ten 3/8-inch (0.95 cm) bolts. The injector/chamber alignment is assured by use of a 1/4-inch (0.64 cm) dowel pin in the injector and a mating hole in the chamber flange. Sealing at the injector/chamber flange was through use of 1100-0 aluminum crush seals. The seals were made in the form of flat rings which slip into grooves. The mating section had a serrated tongue which compressed the aluminum ring forming the seal.

The chamber cylindrical section and the nozzle section were jointed together by twelve 1/2-inch (1.27 cm) bolts. Sealing at the chamber/nozzle interface was by means of 1100-0 aluminum crush seals.

4.3.2 Task IV Thrust Chamber

For the Task IV performance evaluation tests the uncooled thrust chamber illustrated in Figure 22 was utilized. The chamber design employed a copper cylindrical section and the same nozzle section used for the Task III testing. The chamber characteristic length (L^*) could be varied by simply changing the cylindrical spool section. The nominal L^* for the engine was 30 inches (76.2 cm). A shorter cylindrical section which reduced the L^* to 15 inches (38.1 cm) and a longer section which increased the L^* to 60 inches (152.4 cm) were also fabricated and tested.

The copper cylindrical sections were instrumented for static and high frequency pressure measurements. Dual static pressure measurements were provided at the start of nozzle convergence. High frequency pressure measurements were achieved through the use of a Photocon water-cooled high frequency transducer. The copper cylindrical sections also included calorimetric plug thermocouples to measure heat flux. The graphite nozzles were instrumented with ten thermocouples imbedded in the graphite. The thermocouples were located in two groups of five thermocouples 180 degrees apart.

4.3.2.1 Pulse Gun Assembly

The pulse gun used to demonstrate the engine dynamic stability was a design which has been extensively used at TRW (Reference 12). Operation of the pulse gun requires a 28-volt signal to a small detonating primer containing two grains of explosive. The resulting explosion forces the firing pin to strike the center-fire primer of a cartridge contained in the gun breech. This percussion primer in turn ignites the gun powder charge. When sufficient pressure has been built up in the breech, the burst diaphragm ruptures and the relief of the breech pressure forms a shock which propagates down the barrel. In this application, a .300 Magnum cartridge loaded with 50 grains of Hercules Bullseye gun powder was used in combination with a 20,000 psi (1379 MN/m²) burst disc. Figure 23 presents disassembled and assembled views of the pulse gun.

4.3.2.2 Extended Duration Thrust Chamber

The thrust chamber designed to demonstrate the extended duration capability of the injector was an all-graphite configuration. The nominal characteristic length (L^*) of this chamber was 30 inches (76.2 cm) and the inside dimensions were identical to the performance evaluation chamber.

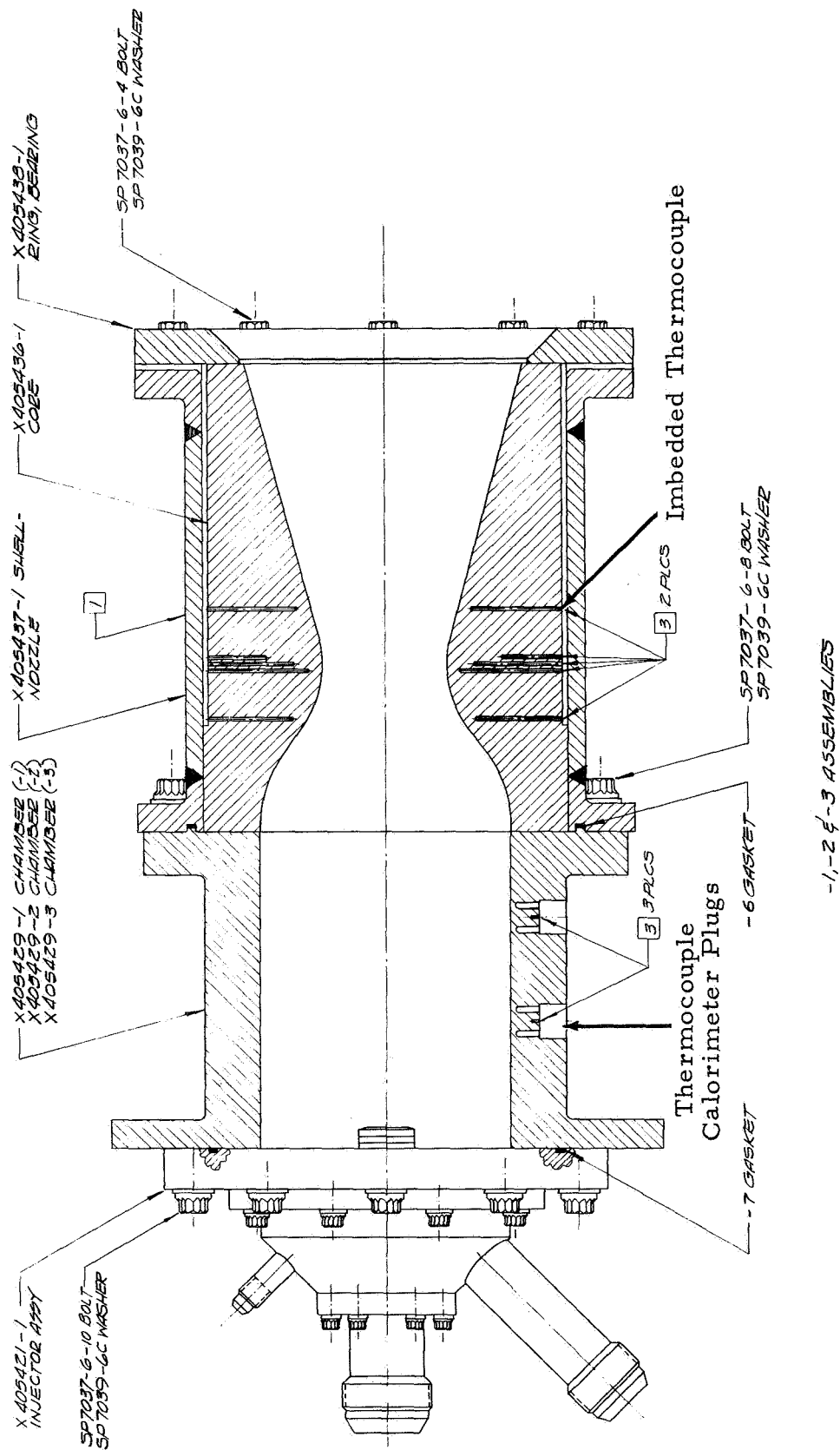


Figure 22. Engine Assembly - Performance Evaluation Tests

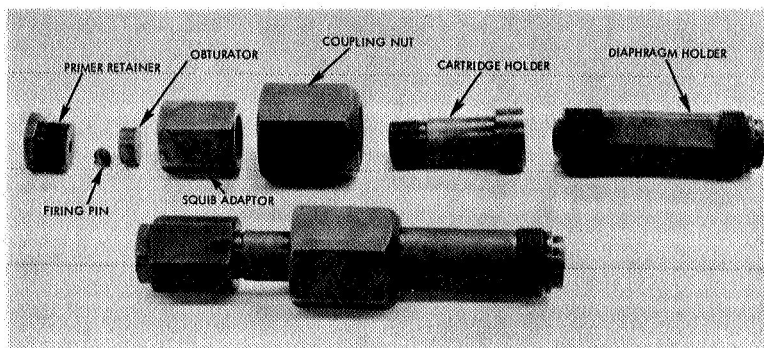


Figure 23. Disassembled and Assembled Views of the Pulse Gun Gun Assembly Used for Dynamic Stability Tests

The graphite chamber was contained within a water-cooled steel jacket. The thrust chamber assembly is illustrated in Figure 24.

4.4 THRUST CHAMBER FABRICATION

The cylindrical sections for the Task III injector streak tests were ablative liners contained within a stainless steel shell. The ablative liner was MX 2600 silica phenolic material with the laminations oriented 90 degrees to the chamber center line. Figure 25 is a photograph of the streak chamber cylindrical section showing the ablative liner and the steel container.

The nozzle section, which bolted directly to the cylindrical section, was a graphite core contained within a stainless steel shell. The nozzle section was common to both the streak test chamber and the performance evaluation chambers. The nozzle cores were machined from both G-90 and ATJ graphite. The shell was fabricated from stainless steel seamless tubing to which the flanges were welded. The steel shell was heated and shrunk-fit over the graphite core. The nozzle core and container are pictured in Figure 26.

For the Task IV performance evaluation tests, the thrust chamber assembly consisted of copper cylindrical sections corresponding to L^* 's of 15, 30 and 60 inches (38.1, 76.2 and 152.4 cm), shown in Figure 27 in combination with the graphite nozzle section (Figure 26) used for the Task III streak tests.

The 30-inch (76.2 cm) L^* heat sink performance evaluation chamber was also used for the injector/facility checkout tests prior to the Task III streak tests. Figure 28 shows the 30-inch (76.2 cm) L^* copper chamber section with the calorimetric plugs and Photocon transducer.

Subsequent to the performance evaluation tests, a fixture was brazed to the 30-inch (76.2 cm) L^* chamber for installing the pulse gun. The stainless steel fixture was brazed into the copper cylindrical section and a tangential entry port drilled into the chamber.

The cylindrical sections were fabricated from hot rolled OFHC copper billets. Thermocouple calorimetric plugs were machined into the cylindrical section at six locations on the 30-inch (76.2 cm) L^* chamber. Provision for installing a Model 5327 Photocon was machined into the cylindrical section. Steel inserts were installed in all the bolt holes on both flanges of the copper sections.

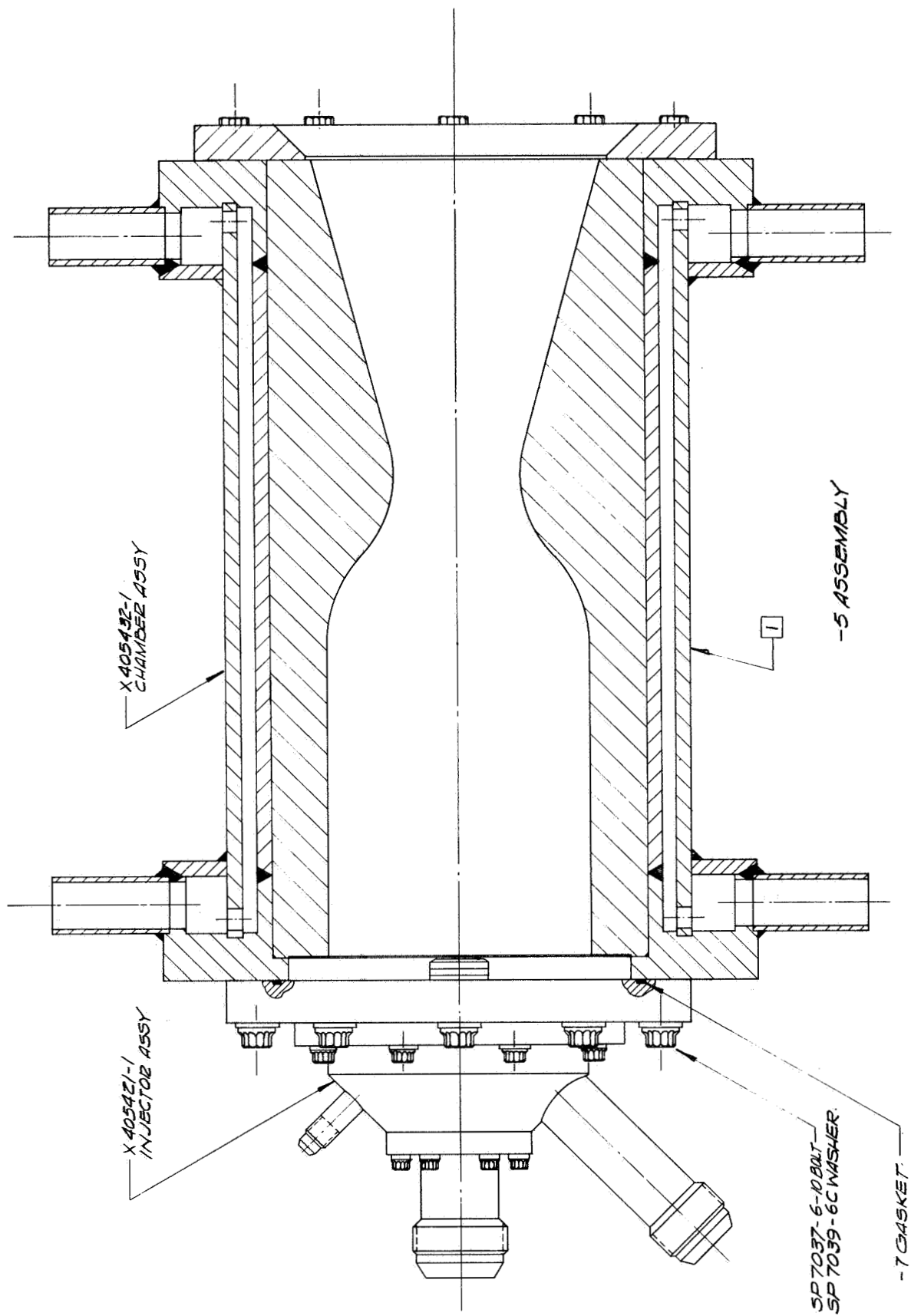


Figure 24. Engine Assembly — Extended Duration Test

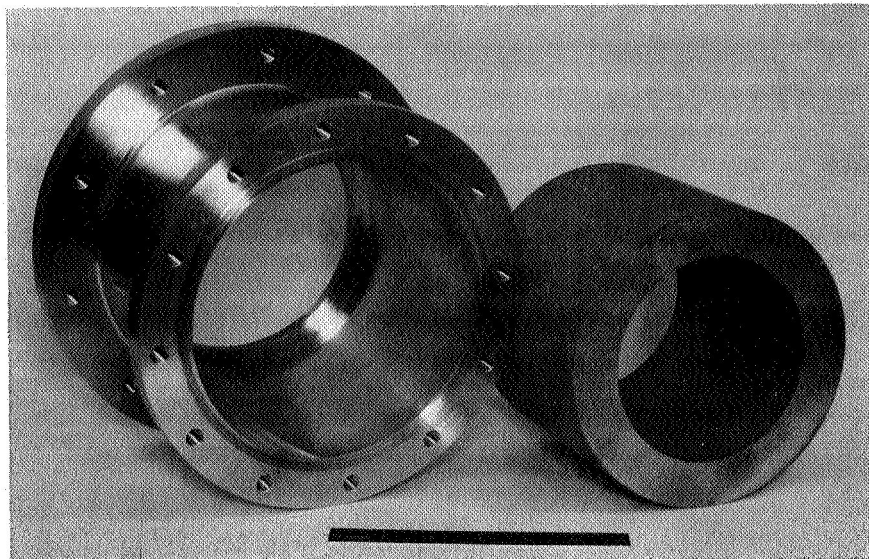


Figure 25. Streak Chamber Assembly

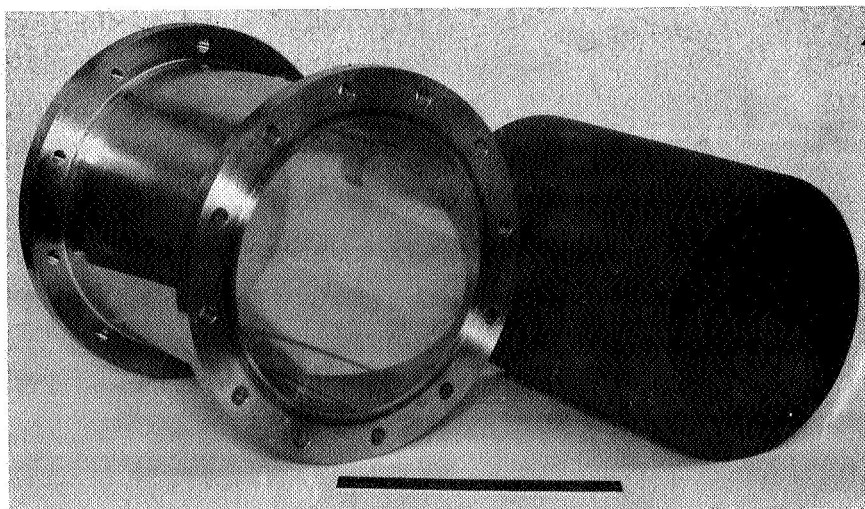


Figure 26. Graphite Nozzle (X405436) and
Steel Container (X405437)

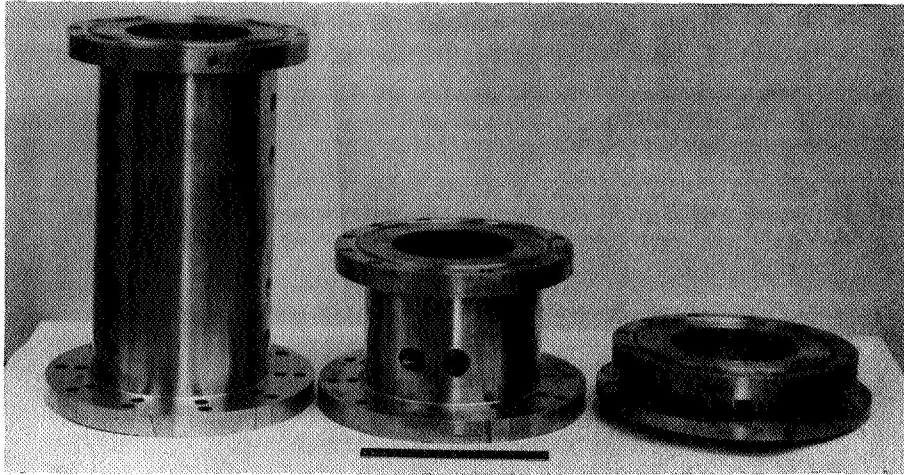


Figure 27. Copper Heat Sink Chamber Sections

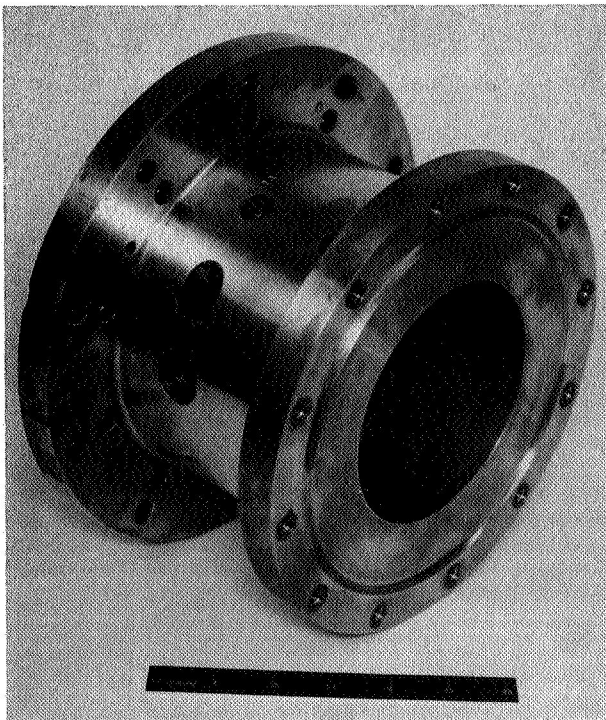


Figure 28
30-inch L* Copper
Chamber Section

The extended duration chamber, consisting of a graphite core shown in Figure 29 and water-cooled jacket, was fabricated from G-90 graphite and 1020 mild steel, respectively. The water-cooled jacket pictured in Figure 30, was shrunk-fit around the graphite core.

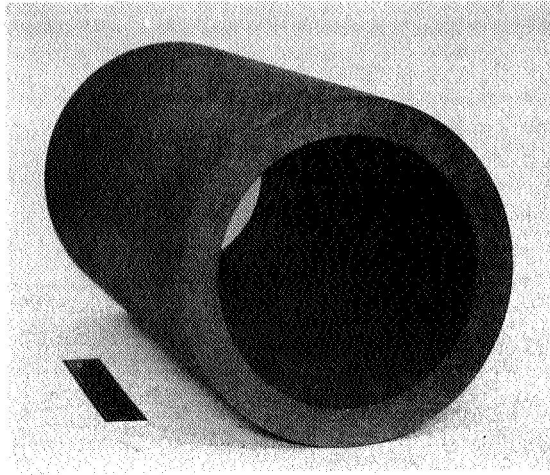


Figure 29. Graphite Chamber for
Extended Duration
Test

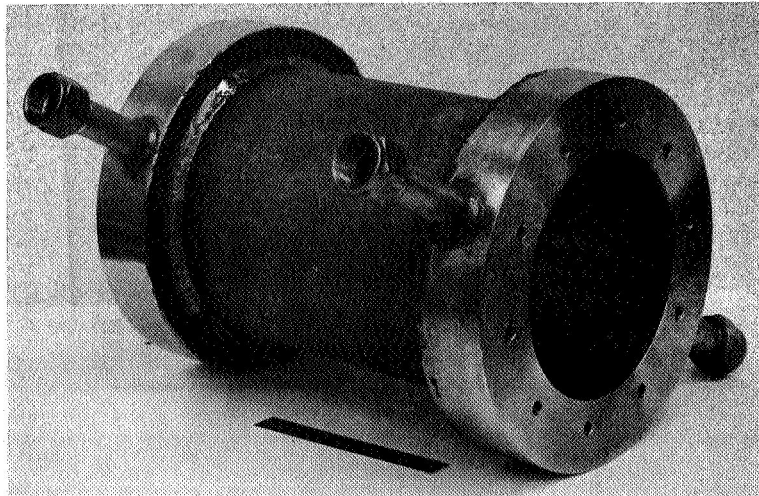


Figure 30. Water-Cooled Jacket for Extended
Duration Graphite Chamber

5. EXPERIMENTAL RESULTS

An injector assembly and sea-level expansion thrust chamber were fabricated to provide test hardware for obtaining injector performance, streaking characteristics, and chamber heat transfer data. The injector assembly was capable of providing injection geometry variations. As described earlier, the injector was designed to provide capability for hardware modification while maintaining test stand readiness.

The experimental effort was conducted in two phases. Task III consisted of a brief series of cold flow tests and injector streak tests to characterize each of the oxidizer elements selected from the Task I analyses. Because of the limited scope of this program, only liquid-liquid cold flow tests were conducted. The injector performance was evaluated during Task IV for three oxidizer injection elements. Each oxidizer element was tested over the mixture ratio range of 5.0 to 6.5. An oxidizer element was then selected and the performance was determined as a function of both characteristic length and mixture ratio. The engine dynamic stability was demonstrated and an extended duration test was conducted.

5.1 TASK III — INJECTOR CHARACTERIZATION TESTS

The purpose of the Task III test effort was to characterize the injector by means of a series of cold flow liquid tests followed by a series of hot firings. Four oxidizer orifice geometries were evaluated by both cold flow and hot firing tests. The specific objectives of the test series were:

- Conduct a series of liquid-liquid propellant simulant cold flow tests to determine the mass and mixture ratio distribution for each oxidizer configuration selected in Task I.
- Perform a series of hot firings with liquid FLOX-gaseous methane to determine the streaking characteristics of the injector with the four selected oxidizer elements.

5.1.1 Injector Cold Flow Tests

The injector cold flow effort was conducted in two phases using the test apparatus described in Appendix B. The first phase of the effort consisted of defining the hydraulic characteristics of the injector assembly and each of the oxidizer orifice configurations. The pressure drop versus flow rate was determined, using water as the propellant simulant, for (1) the fuel distribution ring separately, (2) the total fuel circuit, (3) the oxidizer distribution spindle separately, and (4) the total oxidizer circuit. Once the pressure drop versus flow rate was determined for water, it was converted to equivalent pressure drops for the propellants. The pressure drop through the oxidizer distribution spindle and fuel distribution ring were required in order to determine the pressure drop across the metering orifices only. Figure 31 shows the pressure drop as a function of flow rate for the fuel distribution ring. This figure shows both the measured

(water flow) and calculated (methane) data. The measured and calculated pressure drop versus flow rate are shown in Figure 32 for the oxidizer distribution spindle.

After the injector hydraulic characteristics were determined, a series of liquid-liquid cold flow tests were conducted to define the mass and mixture ratio distribution of each of the oxidizer elements selected.

These cold flow tests were conducted using water as a simulant for both propellants at the values of momentum equivalent to the nominal engine mixture ratio, 5.75. Mixture ratio was determined from these tests by inserting a dye (methylene blue) in one propellant and analyzing the mixture. The absorbency of the water containing the dye was measured and compared with the absorbency of the mixture by means of a Beckman Model B spectrophotometer. Since the absorbency of the solution is linear with concentration, a direct reading of the percentage of one propellant is possible.

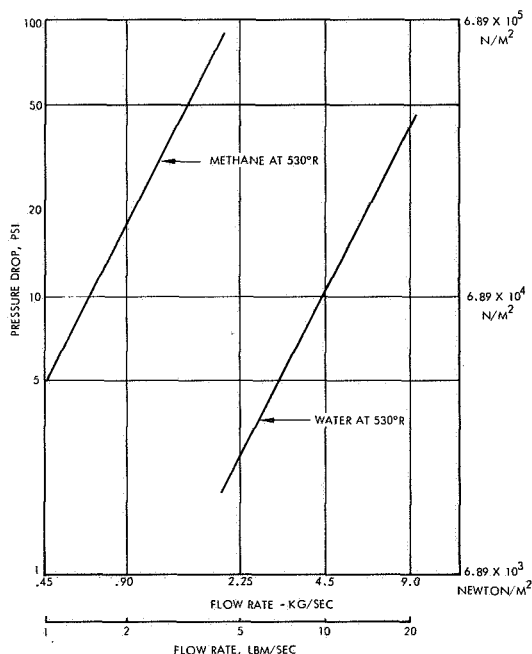


Figure 31

Pressure Drop as a Function of
Flow Rate for the Fuel
Distribution Ring

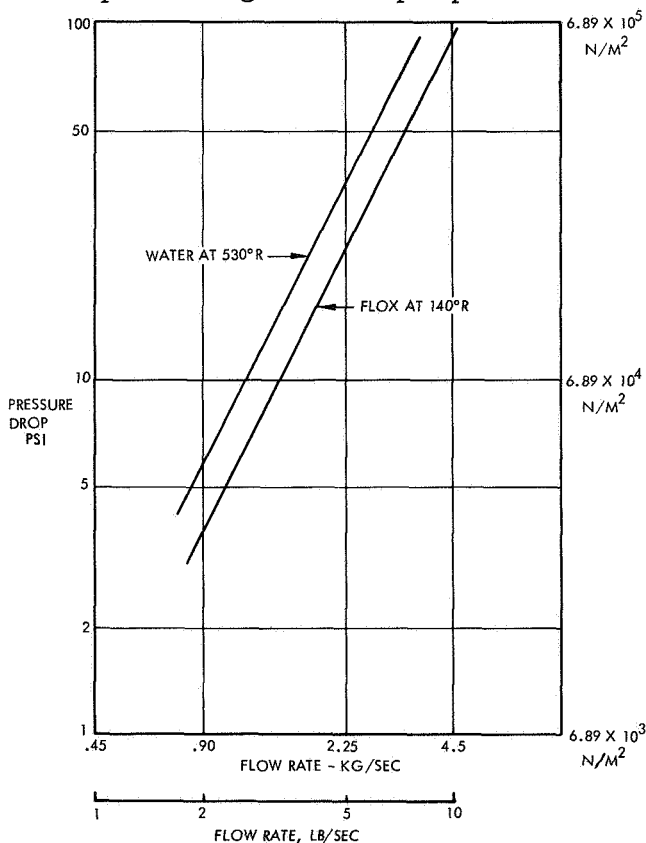


Figure 32

Pressure Drop as a Function of
Flow Rate for the Oxidizer
Distribution Spindle

The mass fraction and mixture ratio distribution as determined from these liquid-liquid cold flow tests are shown in Figures 33 through 35. While these data are not directly applicable to liquid-gas propellants they give a qualitative ranking for the various injector configurations by indicating relative mass and mixture ratio distribution.

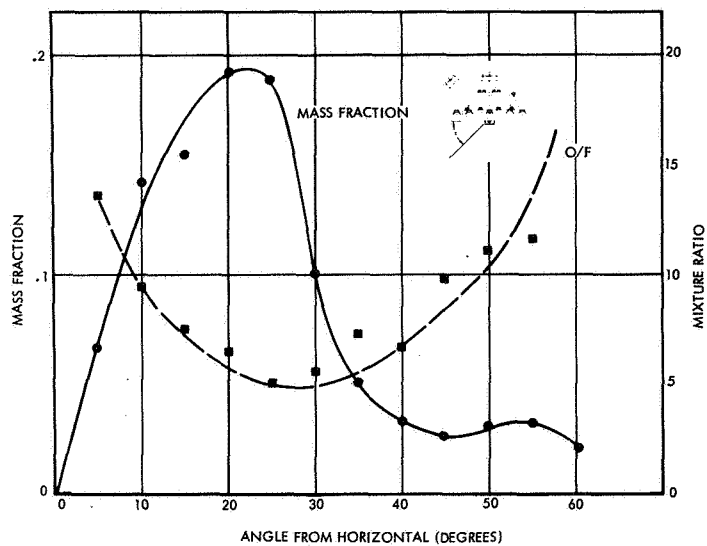


Figure 33
Mass and Mixture Ratio
Distribution for Injector
Configuration Number 1

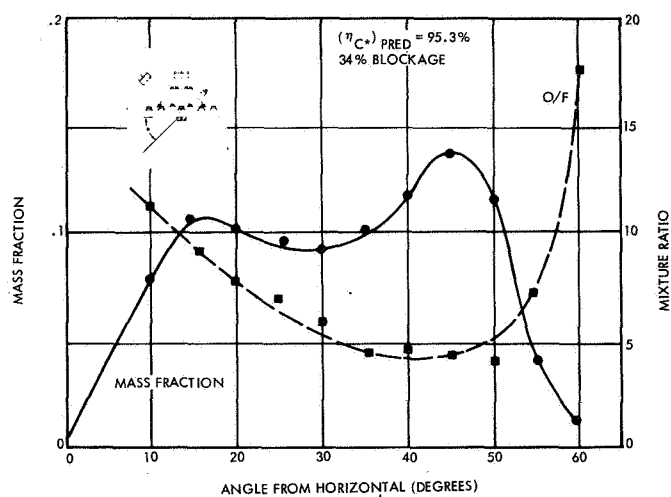


Figure 34
Mass and Mixture Ratio
Distribution for Injector
Configuration Number 3

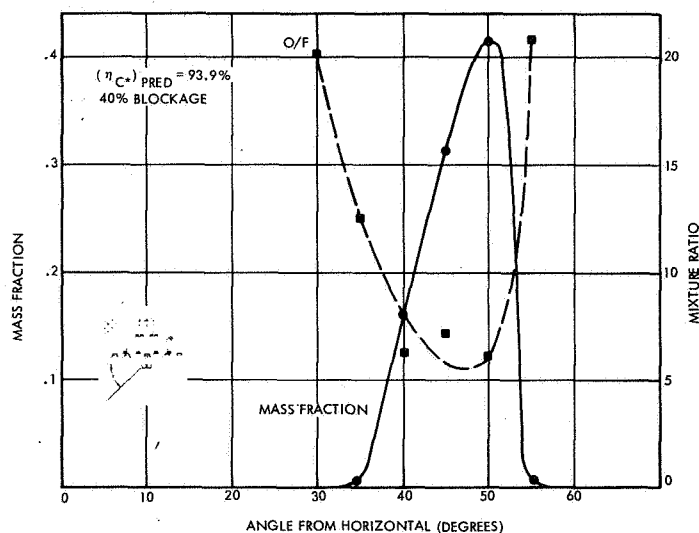


Figure 35
Mass and Mixture Ratio
Distribution for Injector
Configuration Number 6

5.1.2 Injector Hot Firing Tests

Subsequent to the injector cold flow tests, a series of hot firing tests were performed to determine the streaking characteristics of the injector incorporating the selected oxidizer configurations. The geometry of all oxidizer elements tested is tabulated in Table 1. Prior to the streak tests, several injector/facility checkout tests were conducted. A total of ten hot firing tests were conducted during the Task III experimental effort. The results of the Task III experimental effort are summarized in Table 2.

The initial test conducted during the program (HB2-170) was an injector/facility checkout test of approximately 2 seconds duration with a target mixture ratio of 5.75. The injector assembly incorporated the 50 percent blockage oxidizer configuration (X405434-1) and was assembled with the 30-inch (76.2 cm) L* chamber. Figure 36 shows the 3K FLOX/Methane engine assembly mounted on the test stand. The ignition and shutdown transients of this test and all subsequent tests was very smooth. The actual mixture ratio of the test (6.32) was higher than anticipated because the sonic orifice in the fuel system was too large and not choked during the test. Post-test inspection of the injector revealed that the pintle tip had eroded.

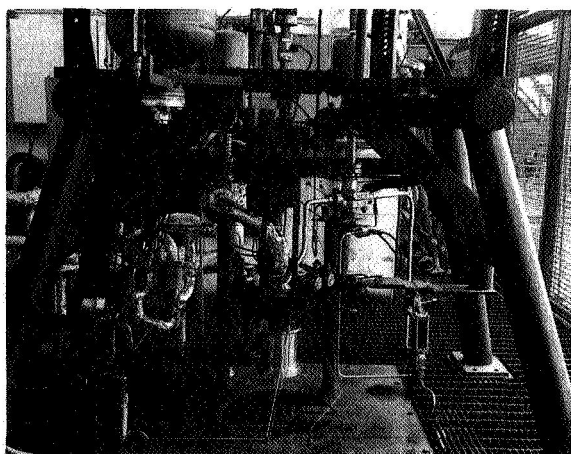


Figure 36

3K FLOX/Methane Engine Installed
on HEPTS B-2 Position

injection velocity increases fuel penetration and is thus beneficial from the pintle erosion standpoint by providing a fuel rich region at the pintle face.

The next test (HB2-171) was conducted using the oxidizer configuration in which slots were canted 30 degrees in a downstream direction rather than in radial streams. The test target conditions were the same as the initial test. The test duration was 2 seconds; however, the pintle tip eroded and failed at approximately 1 second after initial chamber pressure rise.

Because of the pintle tip erosion on the first two tests, it was apparent that a high heat flux/oxidizer rich region was occurring at the pintle face because of a lack of fuel penetration through the oxidizer fan. Previous work at TRW indicated that lowering the fuel

Following test -171, the injector was modified so that the pintle tip and oxidizer orifice ring diameter were enlarged. This eliminated the step or void space between the orifice ring and the pintle sleeve.

Table 2. Summary of Task III Test Results

Test Number	Injector Configuration	Stagnation Chamber Pressure		Mixture Ratio	Total Propellant Flow Rate		Measured Thrust F_{meas}		Vacuum Thrust F_{vac}		C^*_{theo}		Corrected C^* Based on P_c (percent)	Corrected C^* Based on F (percent)	Remarks
		psia	kN/m ²		lbm/sec	Kgm/sec	lb _f	N	lb _f	N	ft/sec	m/sec			
HB2-170	1	375	2586	6.32	8.78	.398	2094	9314	2386	10613	6990	213	80.4	79.1	Injector/facility checkout tests ↓
-171	2	365	2516	5.72	8.93	.405	1990	8852	2283	10155	7147	218	75.9	72.7	
-172	3	468	3227	6.10	8.96	.406	--	--	--	--	7084	216	96.8	--	
-173	↓	464	3199	6.25	9.05	.410	--	--	--	--	7042	215	95.5	--	
-174		458	3158	5.56	9.31	.422	--	--	--	--	7149	218	95.5	--	
-175	↓	429	2958	4.80	9.53	.432	2532	11263	2824	12562	7132	217	86.8	84.9	Injector streak tests ↓
-176		483	3330	5.77	9.43	.428	2841	12637	3135	13945	7133	217	98.8	94.7	
-177	4	415	2861	5.82	9.21	.418	2374	10560	2668	11868	7112	217	82.9	82.7	
-178	5	432	2978	5.88	9.21	.418	2468	10978	2761	12282	7101	216	86.4	85.8	
-179	6	415	2861	5.92	9.21	.418	2354	10471	2647	11774	7088	216	83.1	82.4	

NOTE: All Task III tests were conducted using the 30-inch (76.2 cm) L* chamber.
No thrust data obtained on Tests HB2-172, 173, & 174.

The modified injector was then installed on the test stand preparatory to test HB2-172. The oxidizer configuration used in this assembly was a 34 percent blockage ring. The target test conditions were again an O/F of 5.75 and a 2-second duration. The engine again exhibited a smooth start/shutdown transient and smooth combustion. Injector performance was significantly increased and there appeared to be no problem with pintle durability.

Tests -173 and -174 were virtual repetitions of test number -172 except that the duration was extended to 3 seconds on test -174. Post-test inspection of the injector hardware showed no pintle erosion or other injector damage. The pintle tip had a heavy carbon deposit on the face indicating that the decrease in fuel velocity had resulted in a fuel rich region at the pintle face. The heavy carbon deposits on the pintle are shown in Figure 37. Following this test the streak test series began.

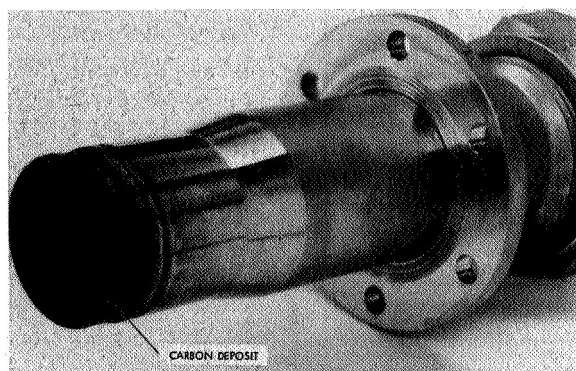


Figure 37

Oxidizer Element Assembly Showing
Carbon Deposition on Pintle

The initial streak test (HB2-175) used the injector as configured on the three previous tests and the streak sample thrust chamber described in Section 4 and shown in Figure 25. The target mixture ratio was 5.75, and the duration set at 4 seconds. The actual test mixture ratio was 4.8 because of an incorrect setting for the fuel regulator. The pintle tip failed because of erosion at approximately 3 seconds. Because of the low mixture ratio, the fuel velocity was excessively high and again resulted in an oxidizer rich region on the pintle face causing erosion and subsequent failure. Post-test inspection of the streak sample showed

no erosion at any point and a heavy carbon deposit over the entire inside wall surface.

Following the test, during the feed system venting operation, a fire occurred on the test stand which destroyed adjacent instrumentation and control cabling. The source of the fire was in the oxidizer system upstream of the flowmeters. The cause of the fire was determined to be a failure of a thermocouple probe in the FLOX line. As shown in Figure 39, this caused a burn-through between the FLOX line and the surrounding LN₂ jacket causing dilute FLOX to flow through the LN₂ jacket. Damage to the plumbing was confined to the single section of line where the burn-through occurred. The damaged section of jacketed FLOX line and the thermocouple probe are shown in Figures 38 and 39.

Four streak tests (HB2-176 through -179) were conducted using the injector assembly with each of the four selected oxidizer configurations. All tests were of 4-second durations with a target mixture ratio of 5.75. The performance of the injector, assembled with each of four oxidizer configurations, is presented in Table 2.

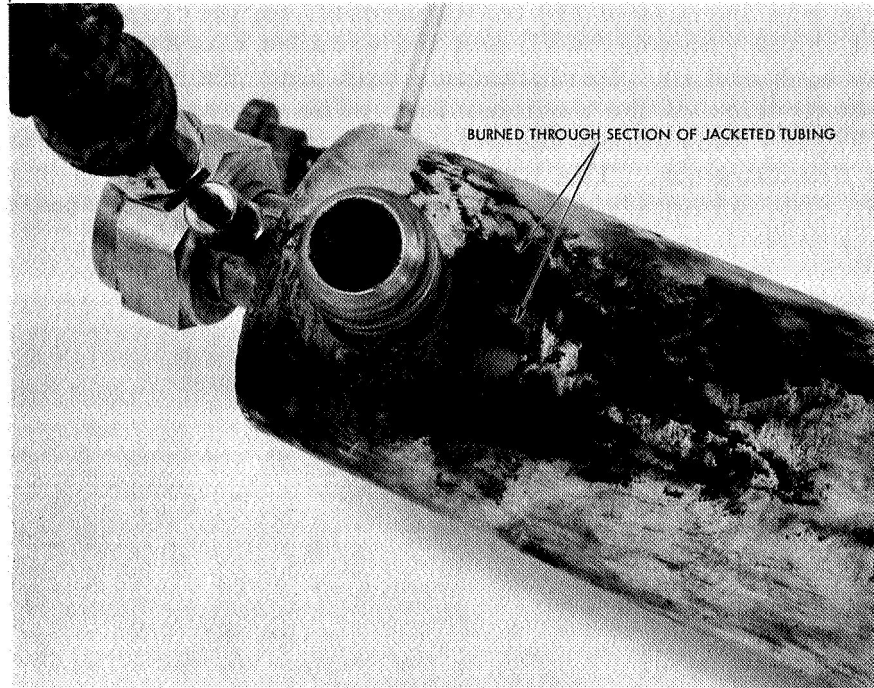


Figure 38. Damaged LN₂ Jacketed FLOX Line



Figure 39. FLOX Line Thermocouple Probe

The highest performing element (X405434-5) exhibited some local erosion at the injector end of the chamber as shown in Figure 40a. As the blockage was increased (34 to 50 percent) an increased fuel-rich region was evident at the chamber wall and the erosion was virtually eliminated (Figures 40 a, b and c). Performance was also diminished because of less fuel penetration of the oxidizer fan. The streak chamber for test -179 (Figure 40d) shows the effect of canting the oxidizer stream 30 degrees downstream. The impingement point on the wall was moved downstream approximately 1 inch (2.54 cm) but still exhibited the local erosion corresponding to the 36 primary oxidizer streams.

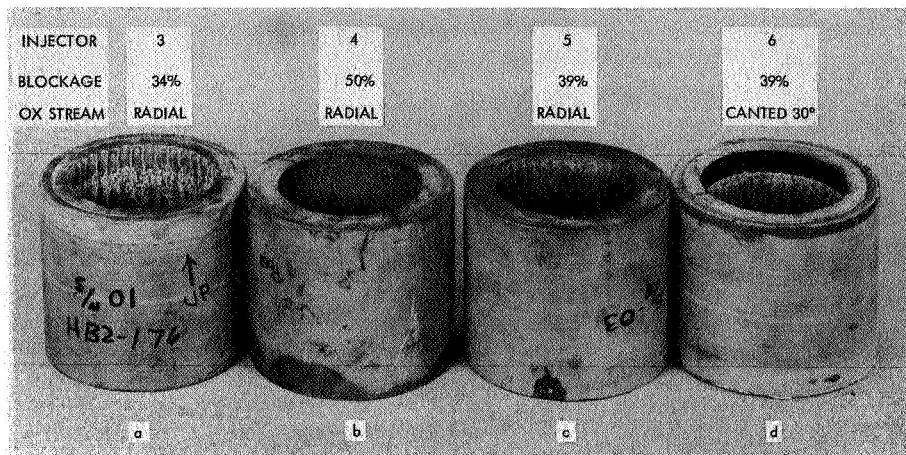


Figure 40. Post-Test View of Task III Injector Streak Samples

Test number HB2-179 concluded the experimental effort of Task III. Ten tests were conducted during this task; five were injector/facility checkout tests and five were injector streak tests. The checkout tests used heat sink thrust chamber hardware while the streak tests incorporated silica phenolic cylindrical sections with the same graphite nozzle used for the heat sink chamber. The results of the Task III tests are summarized in Table 2.

5.2 TASK IV - INJECTOR PERFORMANCE EVALUATION TESTS

The purpose of the Task III test series was to evaluate the injector performance by a series of hot firing tests. Three injector orifice ring configurations were tested. Following the performance screening of the three oxidizer elements in the baseline thrust chamber, one configuration was selected for further characterization. The specific objectives of the test series were:

- Conduct a series of hot firings with each of three oxidizer orifice rings to evaluate the injector in terms of characteristic velocity efficiency, heat flux, and dynamic stability. Each oxidizer configuration was tested over a

mixture ratio range of 87 to 110 percent of the optimum mixture ratio (5.75). The nominal engine characteristic length for these tests was 30 inches (76.2 cm).

- Following the above screening tests, one oxidizer configuration was selected based upon the results of the streak tests and the performance evaluation tests. The selected configuration was subjected to a series of hot firing tests over a mixture ratio range of 87 to 110 percent of the optimum mixture ratio. The characteristic length of the thrust chamber was varied to 15 inches (38.1 cm) and 60 inches (152.4 cm). A schematic representation of the thrust chamber used for the performance evaluation hot firing tests is shown in Figure 41.

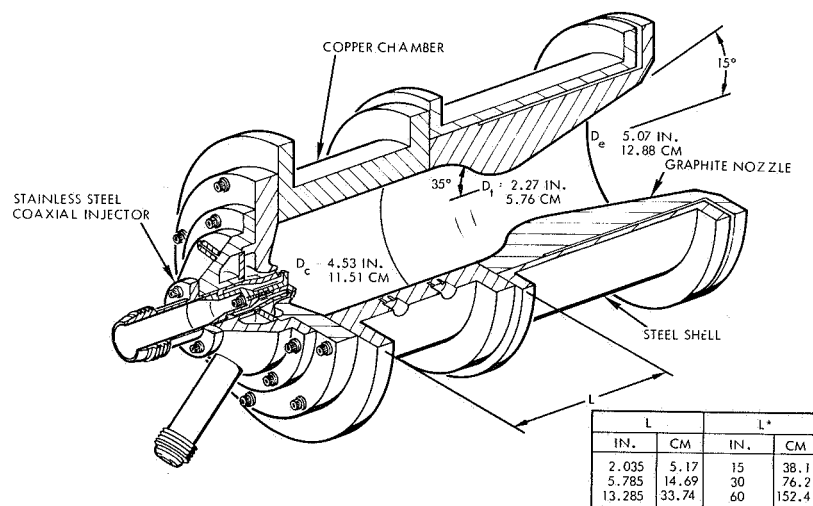


Figure 41. Cutaway Isometric View of Typical Assembled Engine Used in Performance Evaluation Experiments

Seventeen tests were conducted during the performance evaluation task of the program. The results of the Task IV tests are summarized in Table 3. Based upon the results of the Task III streak tests, an additional oxidizer element was fabricated and substituted for two of the lower performing oxidizer elements. The new oxidizer element further reduced the blockage to 30 percent.

All Task IV tests were conducted at a nominally constant total flow rate to keep injection parameters (at the same O/F) nearly constant. Therefore, the variations in throat area caused by erosion are reflected in changes in chamber pressure.

5.2.1 Oxidizer Element Evaluation

Three oxidizer configurations were tested over the mixture ratio range of 5.0 to 6.5 using the baseline thrust chamber. Each of the three injector configurations was tested at three mixture ratios. Tests HB2-180

Table 3. Summary of Task IV Performance Data

Test Number	Injector Configuration	Chamber L* in cm	Stagnation Chamber Pressure psia kN/m ²	Mixture Ratio	Total Propellant Flow Rate lbm/sec Kgm/sec	Measured Thrust lb _f N	Vacuum Thrust lb _f N	C* theo ft/sec m/sec	Corrected η_{C^*} Based on P _c (percent)	Corrected η_{C^*} Based on F _c (percent)	I _{vac} (sec)	$\eta_{I_{vac}}$ (percent)
HB2-180	3	76.2	478 3268	5.77	9.14 .414	2792 12419	3085 13723	7128 217	96.3	97.2	337.5	91.6
-181	↓	→	473 3234	6.30	9.36 .424	2787 12397	3080 13700	7029 214	97.7	98.3	329.0	90.9
-182	↓	→	463 3192	5.01	9.31 .422	2766 12304	3059 13607	7132 217	94.9	94.9	328.6	89.7
-183	4	→	388 2675	5.96	9.03 .410	2237 9951	2530 11254	7087 216	82.4	80.2	280.2	76.7
-184	↓	→	375 2586	6.76	9.20 .417	2132 9484	2425 10787	6944 212	80.2	78.3	263.6	74.2
-185	↓	→	381 2627	5.11	9.24 .419	2164 9626	2457 10929	7116 217	78.8	76.0	265.9	72.4
-186	7	→	438 3020	5.80	9.18 .416	2557 11374	2851 12682	7120 217	88.3	88.5	310.6	84.5
-187	↓	→	411 2834	6.52	9.43 .428	2487 11063	2781 12370	6993 213	86.4	86.0	294.9	82.3
-188	↓	→	414 2854	5.02	9.48 .430	2738 12179	3031 13482	7121 217	92.4	91.7	319.7	87.3
-189	3	152.4	477 3234	5.87	9.18 .416	2771 12326	3065 13634	7147 218	96.3	97.1	333.9	91.0
-190	↓	→	452 3089	6.76	9.25 .420	2681 11926	2974 13229	6962 212	97.3	97.4	321.5	90.4
-191	↓	→	456 3116	5.12	9.28 .421	2802 12464	3095 13767	7141 218	95.8	96.9	333.5	90.8
-192	15	38.1	426 2937	5.83	9.16 .415	2500 11120	2793 12424	7111 217	88.1	86.2	304.9	83.2
-193	↓	→	412 2841	6.59	9.36 .424	2473 11000	2767 12308	6900 210	86.4	87.0	295.6	82.7
-194	↓	→	422 2910	5.00	9.37 .425	2604 11583	2897 12886	7128 217	89.6	88.2	309.2	84.4
-195	↓	→	408 2813	5.99	9.27 .420	2666 11859	2960 13167	7087 216	92.4	93.8	319.3	87.5
-196	↓	→	462 3185	5.36	9.02 .409	2618 11645	2911 12949	7142 218	92.9	94.2	322.7	87.6

through -182 were conducted using the 34 percent blockage configuration (X405434-5) that had exhibited high performance during Task III testing. Target mixture ratios were 5.75, 5.0 and 6.5. The performance was relatively flat over the entire mixture ratio range—varying from 94.9 percent at an O/F of 5.01 to 97.7 percent at an O/F of 6.3.

The next series of three tests (HB2-183, 184 and 185) utilized a 50 percent blockage oxidizer element. The performance of this injector configuration dropped approximately 15 percent from the previous configuration. The drop in performance at high percent blockage was believed caused by a lack of penetration of the radial oxidizer fan by the fuel sheet. The lack of fuel penetration was also evident from the heavy carbon deposits on the wall of the streak sample and from the heat flux data, which showed reduced wall heat fluxes.

In order to more clearly define the effect of oxidizer geometry on performance, an element was fabricated which decreased the blockage below that of the initial element tested in this task. The geometry for this element had a slot width-to-unit width ratio of 0.3 with all other injection parameters (pressure drop, percent secondary flow, etc.) held constant. These tests, covering the O/F range of 5.0 to 6.5, were conducted using this element. The injector performance of the 30 percent element was intermediate to the other two injector configurations.

5.2.2 Characteristic Length Evaluation

Following test HB2-188, the -5 oxidizer element (34 percent blockage) was selected, with the concurrence of the NASA LeRC Project Manager, for evaluation in thrust chambers of 60- and 15-inch (152.4 and 38.1 cm) characteristic length. The L^* variation was made by interchanging cylindrical copper spool sections with the basic graphite throat. Three tests were conducted over the nominal O/F range of 5.0 to 6.5 with each L^* chamber. Characteristic velocity efficiency showed a slight increase with the 60-inch (152.4 cm) L^* chamber but dropped substantially (about 8 percent) when the L^* was decreased to 15 inches (38.1 cm).

5.2.3 Thrust Chamber Heat Transfer

One of the objectives of the Task IV test series was to evaluate the injector in terms of heat transfer characteristics to the thrust chamber.

The tests of Task IV, with the exception of the extended duration test, were conducted with the copper heat sink cylindrical sections and a graphite throat. The copper spool sections were designed to yield both axial and circumferential heat flux profiles in the thrust chamber. Additionally, the graphite nozzle was instrumented with thermocouples imbedded at various depths and axial stations to permit evaluation of throat heat flux levels. A schematic of the instrumentation locations is presented in Figure 42.

The copper chamber sections were instrumented using the calorimetric plug design which was used successfully to obtain heat flux data under Contract NAS3-11200. The data from these calorimetric plugs were analyzed by the methods outlined in Appendix A of Reference 1.

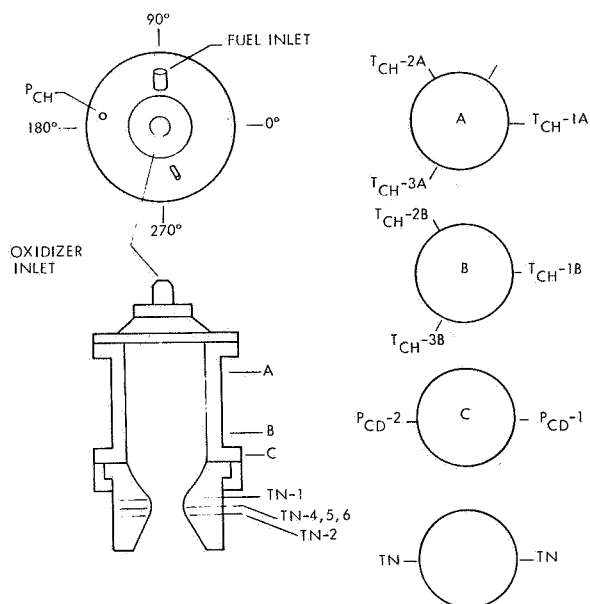


Figure 42

Schematic of Thrust Chamber
Assembly Instrumentation Location

the pulse gun installed in shown in Figure 43. A test (HB2-195) was conducted to demonstrate the dynamic stability of the engine. A charge of 50 grains of Hercules Bullseye powder in combination with a 20,000 psi (1379 MN/m^2) burst disc was used to provide the tangential pressure pulse.

5.2.5 Extended Duration Capability

The final test of the program was to demonstrate the capability of the injector to operate for extended durations. In order to demonstrate this, an all graphite thrust chamber was fabricated as described in Section 4. The test (HB2-196) had a target mixture ratio of 5.75; the anticipated duration was 50 seconds and was limited by the capability of the high pressure fuel tank. The thrust chamber assembly is shown in Figure 44 installed in HEPTS position B-2 prior to test -196.

Testing of the passively cooled graphitic chamber began at an O/F of 5.2. The test was terminated after 26 seconds because of a combustion gas leak which occurred at the injector/chamber flange.

5.3 DATA EVALUATION

This section presents the analyses of the Task IV experimental results. The hot firing data from this task were utilized to develop design criteria for gas-liquid single-element coaxial injectors.

The characteristic velocity efficiency of each of the three elements is plotted in Figure 45 as a function of mixture ratio. The data as plotted in Figure 45 show that injector performance was relatively flat over the

For each of the three selected Task IV oxidizer elements, the chamber heat flux was determined as a function of mixture ratio. Chamber heat flux variation with axial and circumferential position was also evaluated during this test series.

5.2.4 Combustion Stability

All tests of Task IV were instrumented with high frequency pressure transducer (Model 5327 Photocon) to determine combustion roughness and detect any instances of combustion instability. Following the C^* evaluation tests with the 30-inch (76.2 cm) thrust chamber, the chamber was modified to accept the pulse gun. A fitting was fabricated and brazed to the chamber to allow for a tangential entry to be bored into the chamber. The pulse gun was then threaded into this fitting. The baseline chamber with

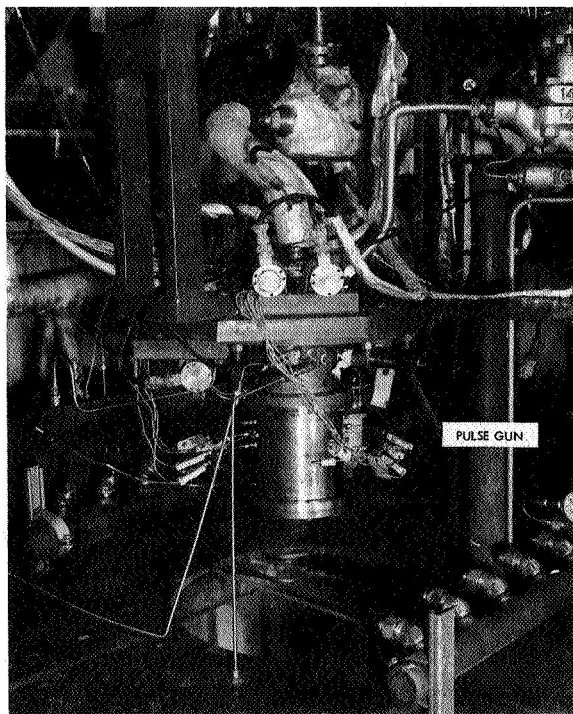


Figure 43

3K FLOX/Methane Engine with
Pulse Gun Installed in 30-Inch
L* Chamber

O/F range for both the 34 and 50 percent blockage elements; the peak performance occurred approximately at the design mixture ratio of 5.75. For the 30 percent blockage ring, however, the performance increased as O/F was lowered and the performance peak is at a lower mixture ratio. Because of the limited scope of the test effort, the element was not subjected to further testing to determine the peak performance.

The C* efficiency of the injector as a function of percent blockage is illustrated in Figure 46. These data indicate that for a fixed fuel momentum, the oxidizer element geometry exhibits an optimum ratio of oxidizer stream width to unit width (percent blockage). This is presented graphically in Figure 46.

Following the performance evaluation testing of the three oxidizer elements, injector element number three was selected to evaluate the effect of L* on characteristic

velocity efficiency. Figure 47 presents the effect of L* and mixture ratio on C* efficiency for the number three injector. This figure indicates that very little C* performance increase is realized above an L* of about 30 inches (76.2 cm) and that the injection process is probably not vaporization limited. Mass and mixture ratio distribution are assumed to be the limiting factors in attaining high C* efficiency with this injector design.

The agreement between C* efficiency based on measurement of thrust and chamber pressure is shown in Figure 48. As this figure illustrates, excellent agreement between the two calculation methods was demonstrated.

5.3.1 Thermal Characteristics

Detailed analyses of the engine thermal data were conducted for a series of firings encompassing a mixture ratio range of 5.00 to 6.5 at an operating chamber pressure of approximately 500 psia (3447 kN/m²). The analyses used heat sink test data reduction to determine heat flux profiles. The oxidizer orifice configurations analyzed were the configurations selected from Task III testing.

The heat sink engine was instrumented with thermocouples as shown previously in Figure 42. Copper calorimeter plugs were used in the copper chamber section to measure the heat flux while chromel-alumel

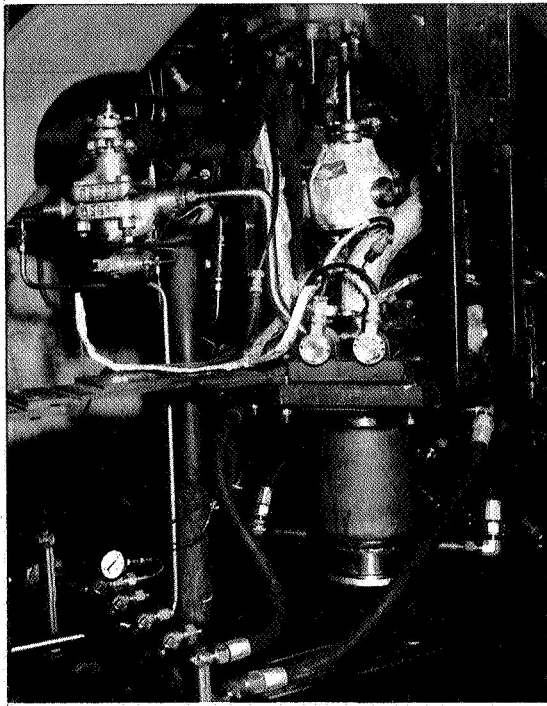


Figure 44
Water-Cooled Graphite Thrust
Chamber Installed in HEPTS
Position B-2

by placing calorimeter thermocouple plugs 120 degrees apart at each of the two axial positions in the 30-inch (76.2 cm) L^* chamber. The relative

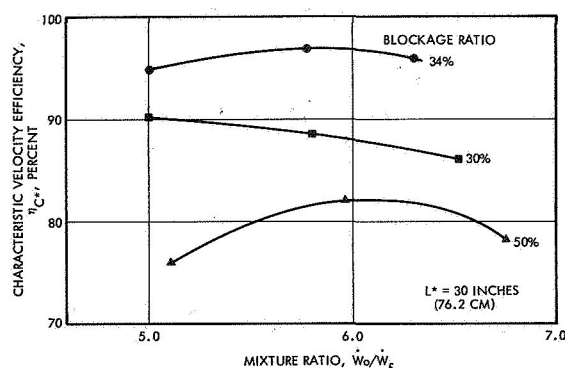


Figure 45
Effect of Mixture Ratio and Blockage
Ratio on Characteristic Velocity
Efficiency

lent to the Bartz prediction while throat heat flux is lower than predicted by about 30 percent.

thermocouples were installed in the graphite nozzle to measure the temperature gradients. The test data for the calorimeter plugs was reduced using a transient analysis technique developed for a two-dimensional calorimeter plug which accounted for the heat losses between the calorimeter plug web and the chamber wall. A summary of the heat flux data is presented in Table 4 for the various thrust chamber stations for each of the three injector configurations.

The heat flux in the cylindrical section of the 30-inch (76.2 cm) L^* engine is presented in Figure 49 as a function of mixture ratio for each of the three injector configurations tested during Task IV. Figure 50 presents the variation in chamber heat flux as the ratio of oxidizer slot width to unit width is varied. This curve shows the same trend as the combustion efficiency curve.

The circumferential variation in chamber heat flux was measured plugs 120 degrees apart at each of the two axial positions in the 30-inch (76.2 cm) L^* chamber. The relative position of the thermocouple plugs is shown schematically in Figure 42. Figure 51 presents the circumferential variation in heat flux for mixture ratios of 5.0, 5.75 and 6.5.

The heat flux distribution as a function of axial distance in the thrust chamber is presented in Figures 52 and 53 at a firing time of 2.0 seconds for injectors 3 and 7. The heat flux prediction based upon the Bartz equation predicted a 25 percent lower heat flux in the combustion chamber and approximately equivalent heat flux at the throat and in the convergent section compared to the measured values for injector number 3. For injector number 7, Figure 53 shows chamber heat flux approximately equivalent to the Bartz prediction while throat heat flux is lower than predicted by about 30 percent.

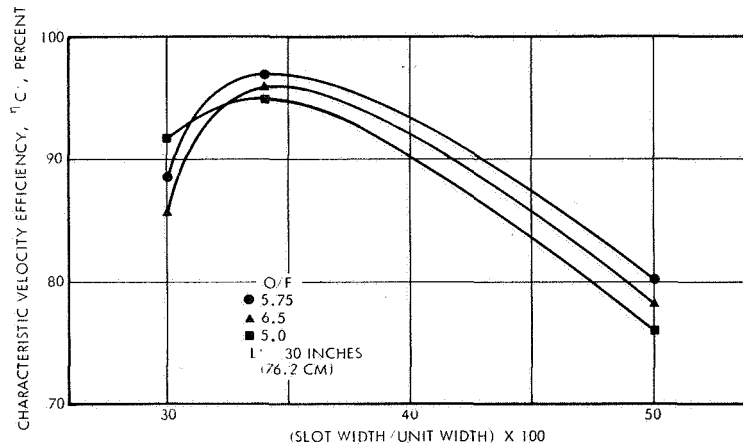


Figure 46
Effect of Oxidizer
Injection Geometry
and Mixture Ratio
on Characteristic
Velocity Efficiency

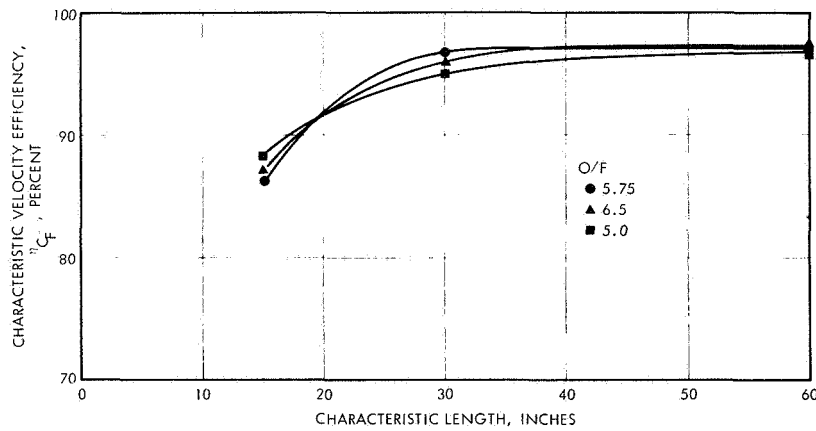


Figure 47
Effect of
Characteristic
Length (L^*) and
Mixture Ratio on
Characteristic
Velocity Efficiency

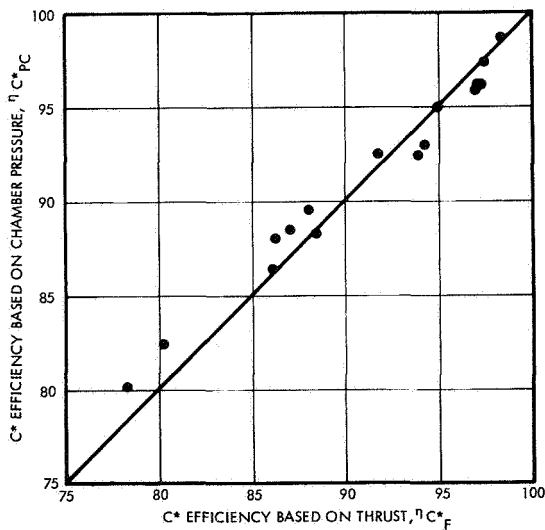


Figure 48
Comparison of Corrected
 C^* Efficiencies Based on
Measurement of Chamber
Pressure and Throat

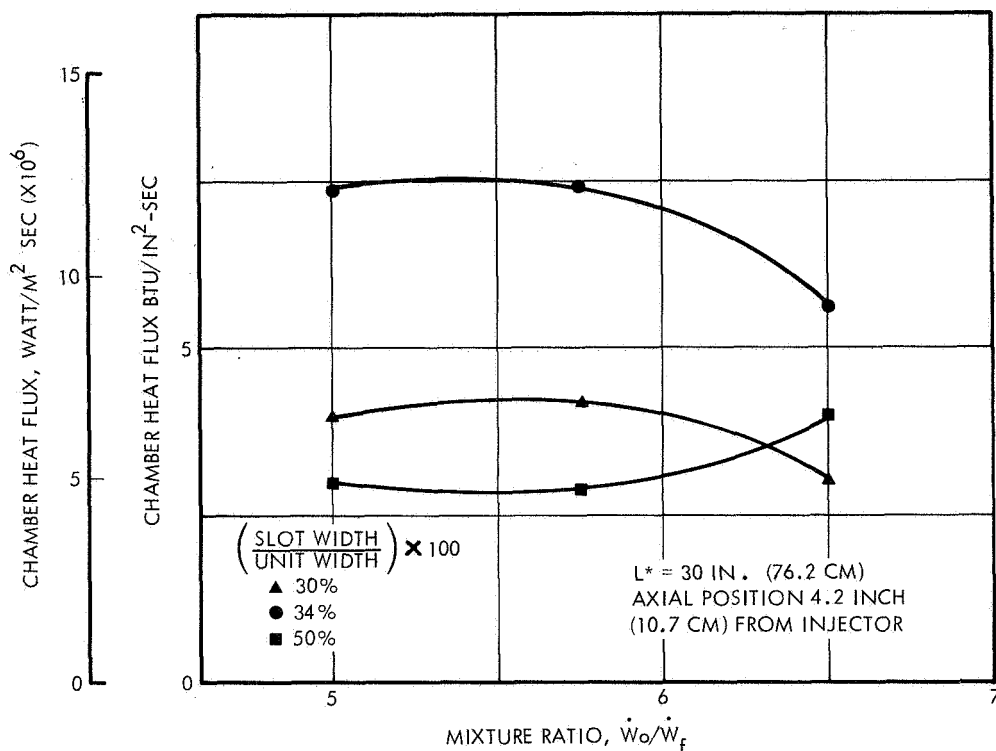


Figure 49. Effect of Mixture Ratio and Injector Geometry on Chamber Heat Flux

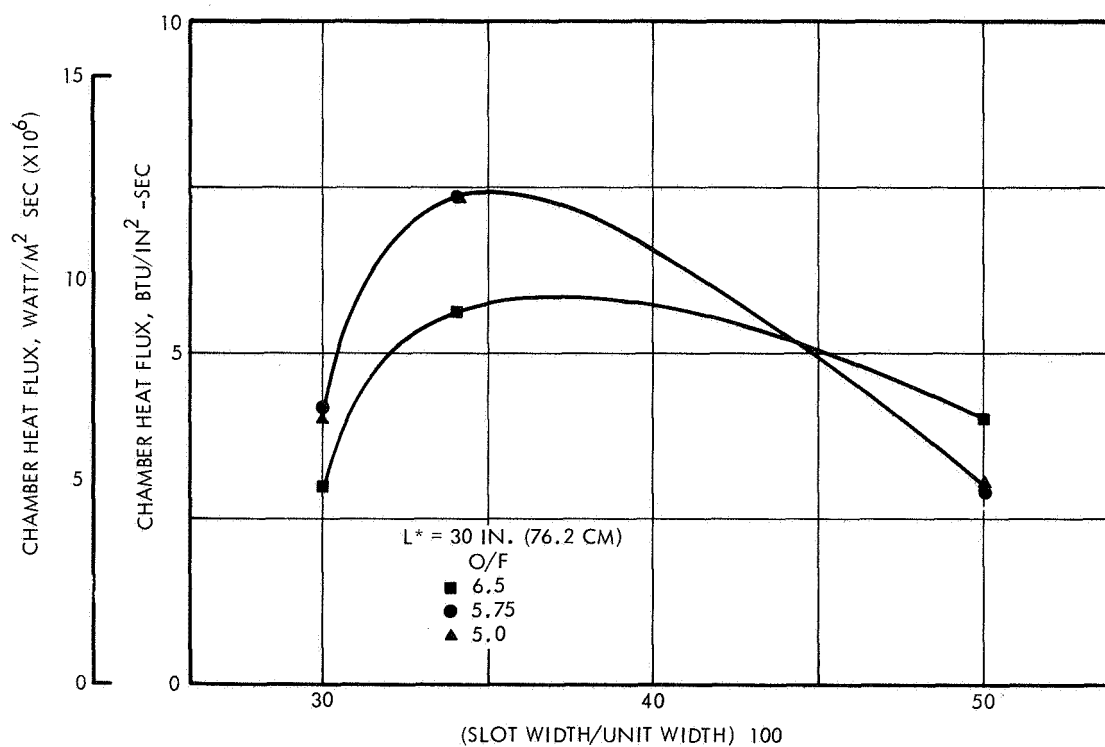


Figure 50. Effect of Oxidizer Injection Geometry and Mixture Ratio on Chamber Heat Flux

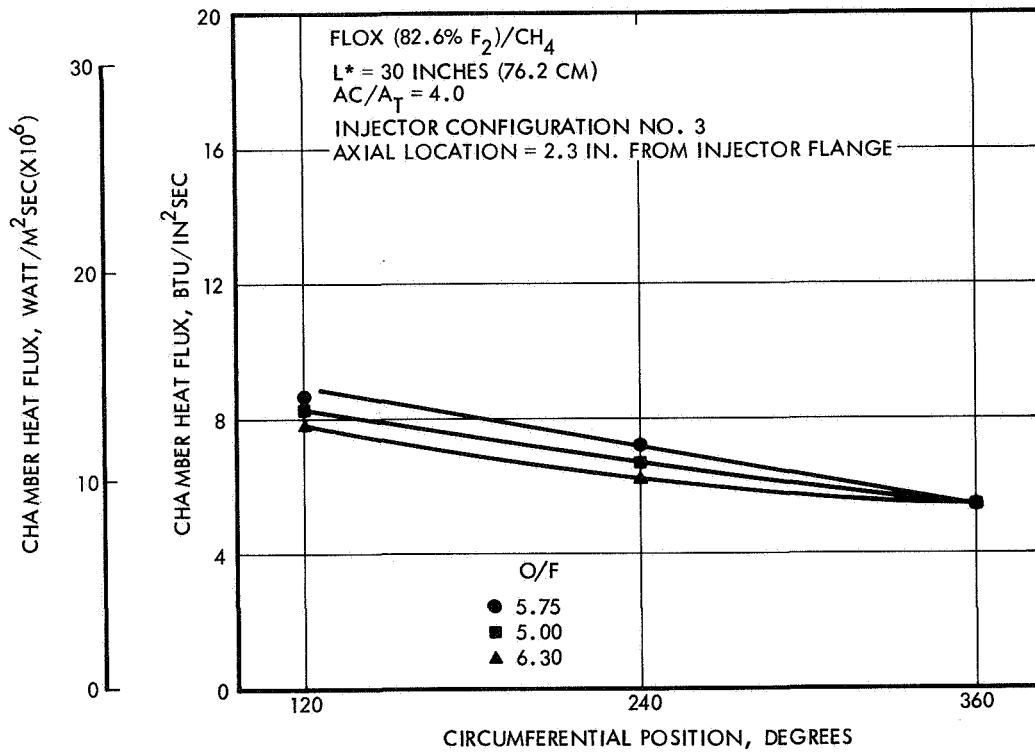


Figure 51. Thrust Chamber Heat Flux as a Function of Circumferential Position and Mixture Ratio

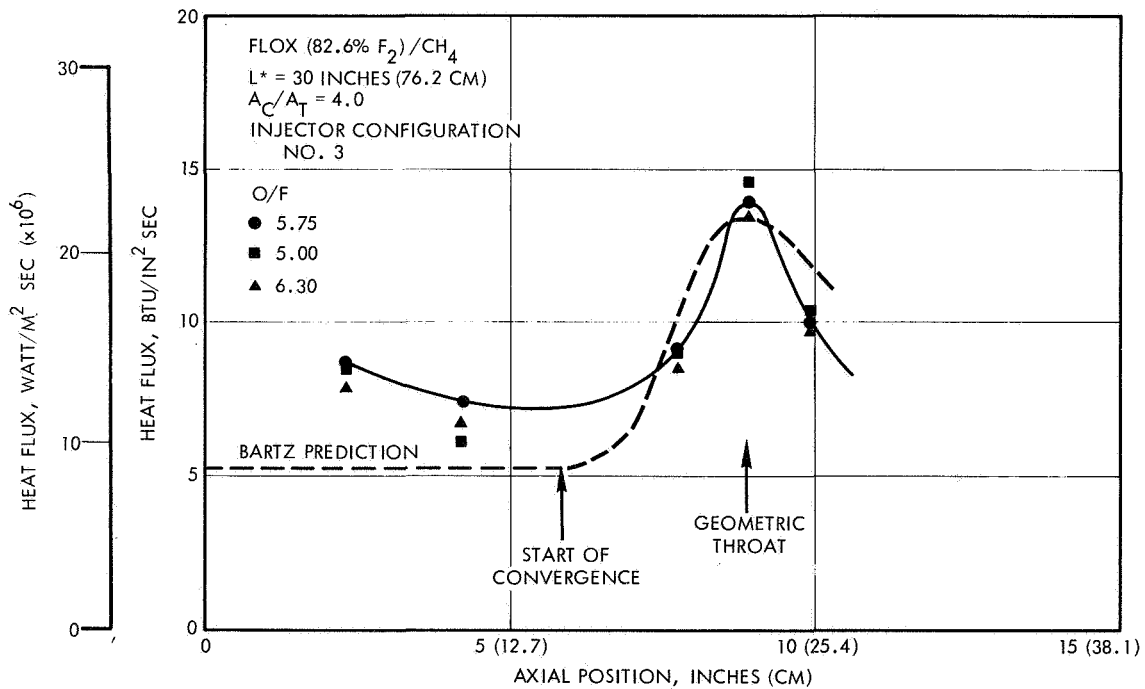


Figure 52. Thrust Chamber Heat Flux Distribution as a Function of Axial Position and Mixture Ratio for Injector Number 3

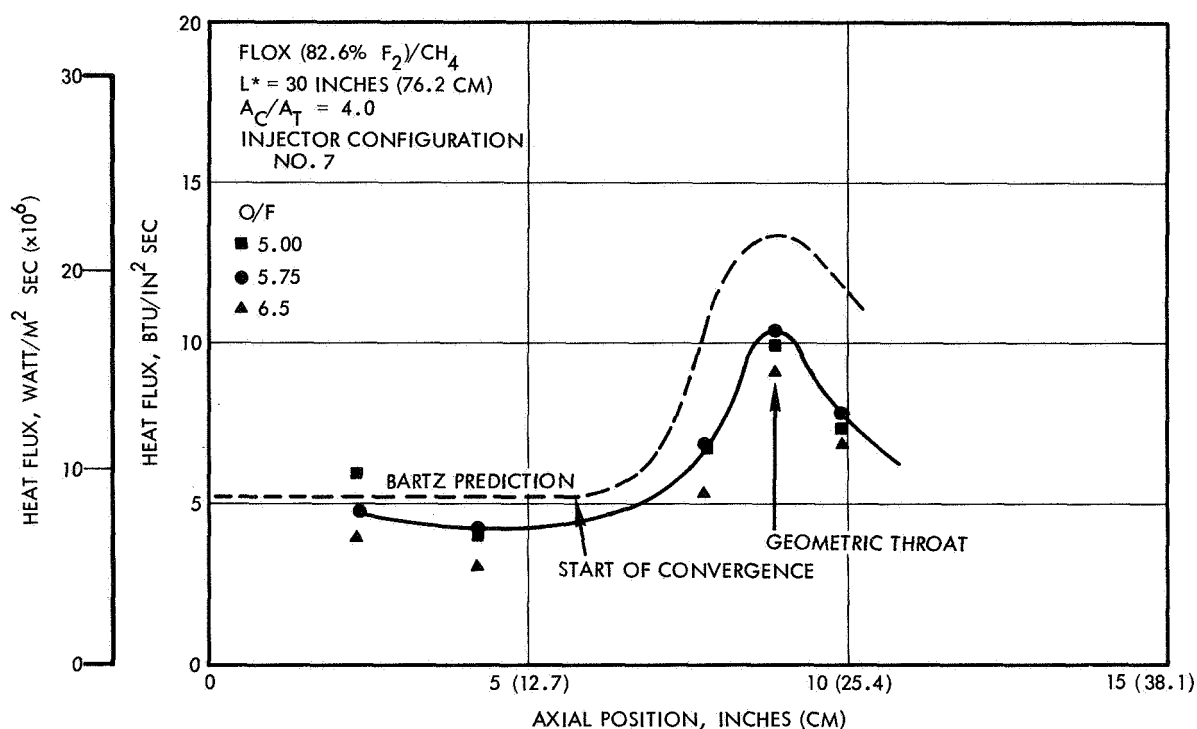


Figure 53. Thrust Chamber Heat Flux Distribution as a Function of Axial Position and Mixture Ratio for Injector Number 7

5.3.2 Dynamic Stability

During the Task IV testing, dynamic chamber pressure measurements were made on all tests using a Photocon high frequency transducer. In no instance was unstable combustion recorded. The start and shutdown transients were very smooth as illustrated in Figures 54 and 55. Combustion roughness or "noise" was less than 1 percent of chamber pressure peak-to-peak. The dynamic stability of the engine was demonstrated on test HB2-195 utilizing the previously described pulse gun hardware. The oscillograph trace of this test is shown in Figure 56. The engine was subjected to a 300 psi (2068 kN/m²) pulse overpressure and recovered in less than 15 msec.

5.3.3 Injector Model Correlation

The injector performance data taken in Task IV were correlated with the injector design model described in detail in Section 3.1. Using test data (density and velocity) and the actual injector geometric characteristics, the momentum ratios were determined. Figure 57 shows characteristic velocity efficiency (as a function of the ratio of fuel momentum to oxidizer momentum). Figure 57 clearly shows that the injector yields peak performance where the momentum ratio, as defined in Section 3.1, is unity. This curve represents three geometric configurations and a mixture ratio range from 5.0 to 6.5. The cross mixing index factor (C) as defined in Section 3.1 and Reference 11 ranges from 0.61 to 1.20 to correlate the data.

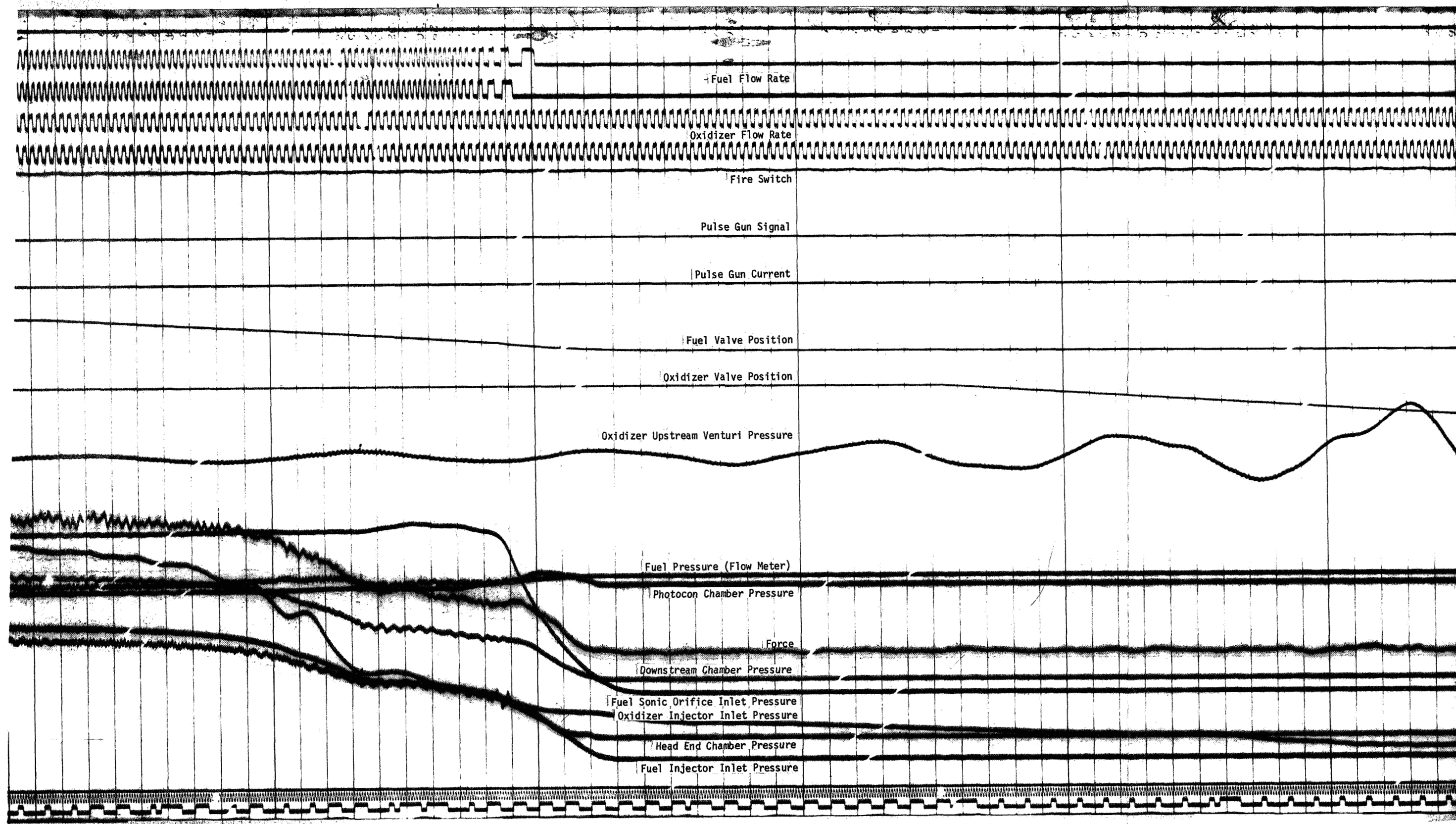


Figure 54. Oscilloscope Trace of
3K FLOX/CH₄ Start
Transient (HB2-195)

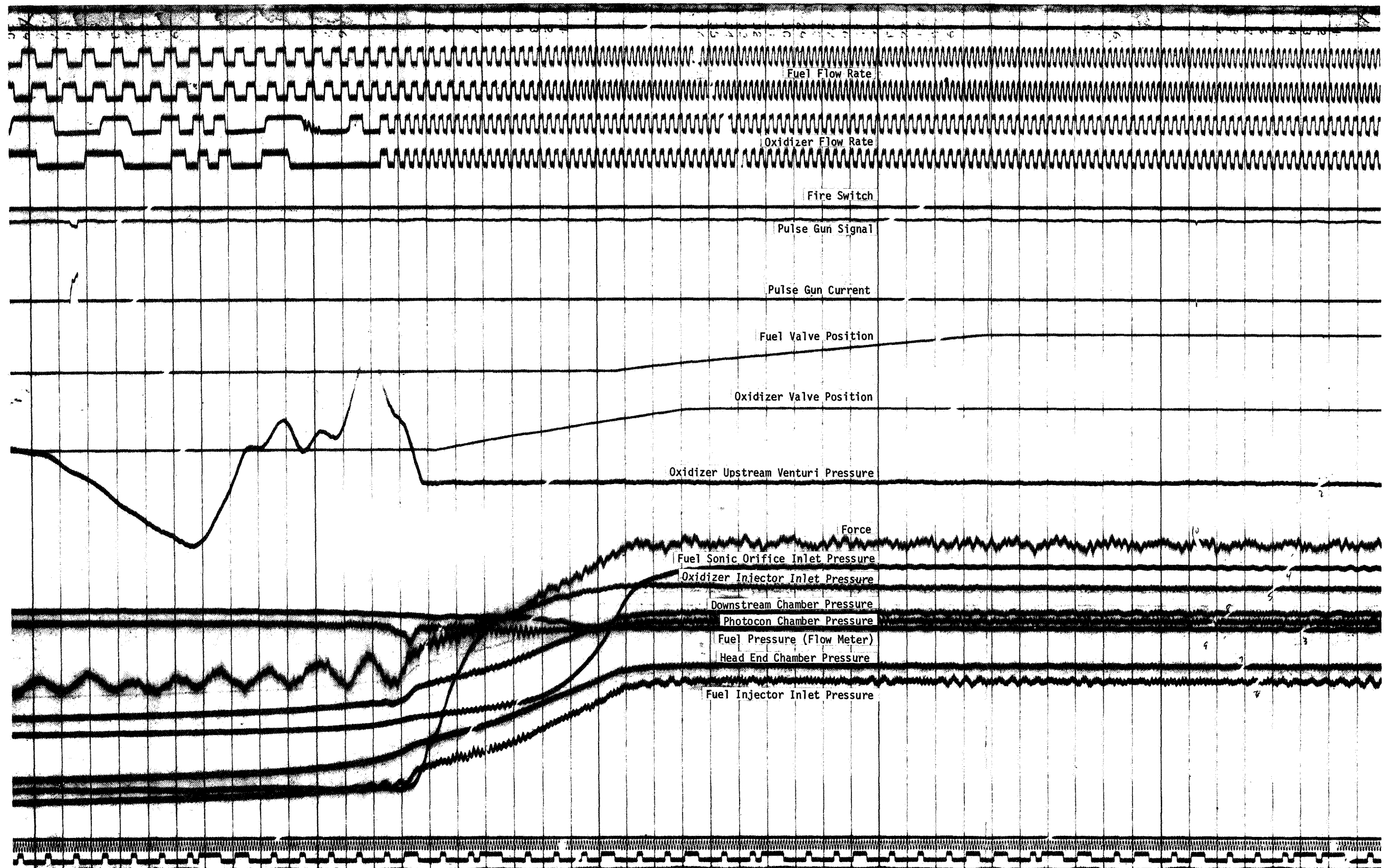


Figure 55. Oscillograph Trace of
3K FLOX/CH₄ Shutdown
Transient (HB2-195)

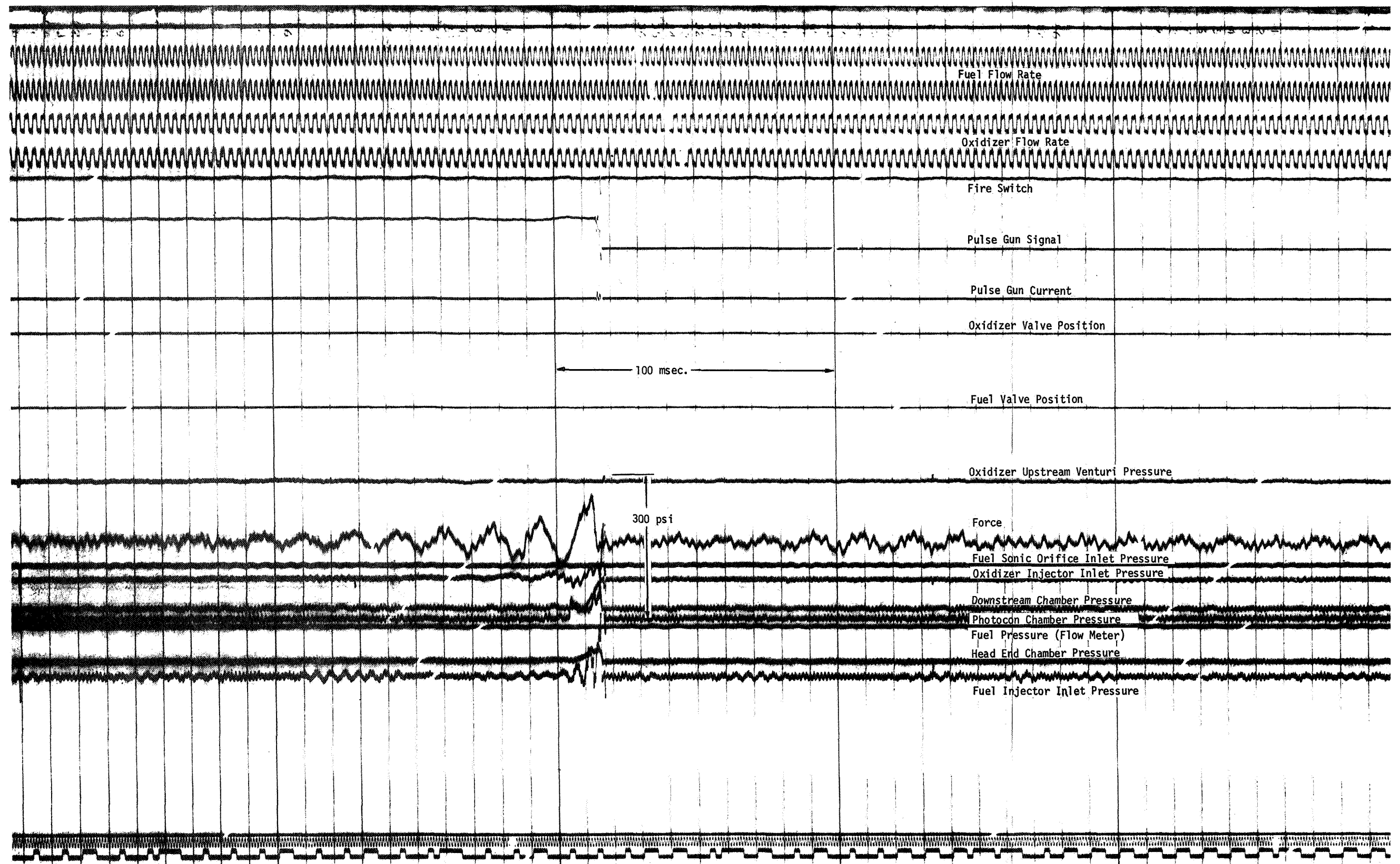


Figure 56. Oscillograph Trace of
Dynamic Stability Test
(HB2-195)

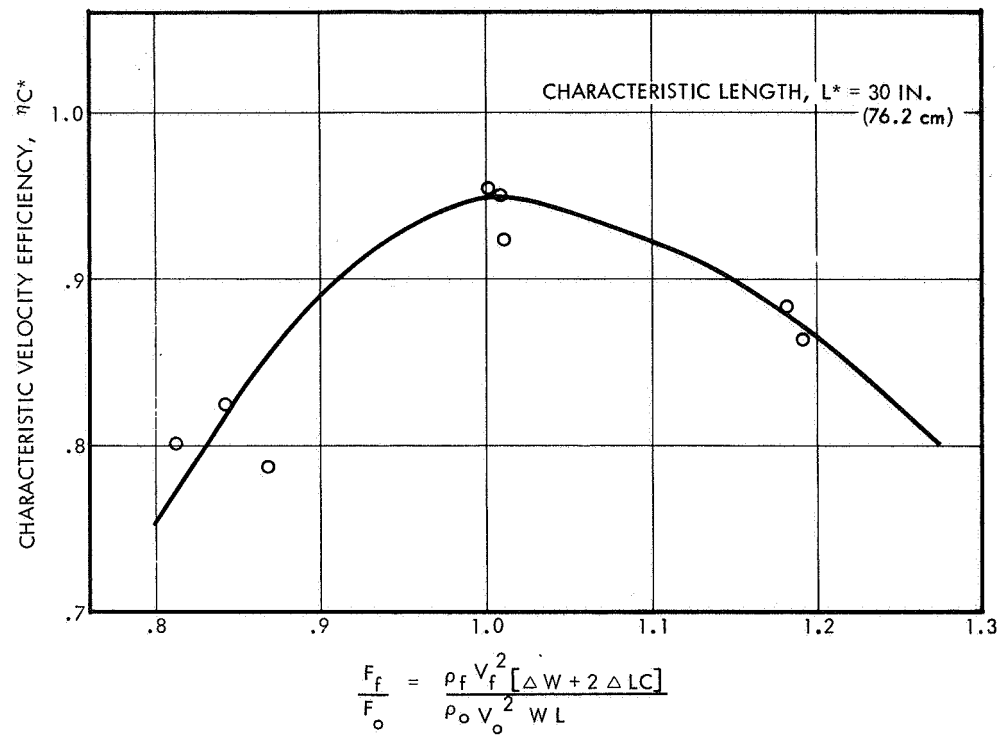


Figure 57. Effect of Propellant Force Ratio on Characteristic Velocity Efficiency

6. CONCLUDING REMARKS

An analytical, design, and experimental technology program has been conducted to obtain basic design criteria for high performance single-element coaxial pintle injectors using liquid FLOX/gaseous methane propellants. Nominal sea-level thrust and chamber pressure levels for these evaluations were 3000 lb_f (13345N) and 500 psia (34.5×10^5 N/m²), respectively. The major conclusions drawn from the Gas-Liquid Space Storable Propellant Performance Program are as follows:

- A high performance coaxial pintle injector can be designed to operate with liquid FLOX and gaseous methane at a chamber pressure of 500 psia (34.5×10^5 N/m²).
- The coaxial pintle injector demonstrated its stability by a complete absence of self-induced instability and combustion roughness of less than 1 percent of chamber pressure as measured by high frequency pressure transducers during 27 hot firing tests. The injector also demonstrated recovery in less than 15 msec from an induced tangential perturbation from a pulse gun excitation device.
- The injector hydraulic model formulated in Task I was able to predict peak performance when correlated with hot firing data.
- The thrust chamber wall environment can be predictably controlled through injector design/operating conditions.

The experimental results indicated that the liquid FLOX/gaseous methane injector performance and thrust chamber wall environment trends agree with previous data obtained on both earth and space storable liquid/liquid propellant coaxial injector designs. The performance model employed in Section 3 to correlate the gas-liquid data is identical to the formulation used on previous liquid-liquid injectors. Previous liquid-liquid injector data indicated that for a fixed injection momentum ratio (M_O/M_F) the resultant chamber heat flux levels were inversely proportional to percent blockage (i. e., higher percent blockage produced reduced wall heat flux). At fixed percent blockage, experimental wall heat flux data on liquid-liquid coaxial injectors have shown heat flux levels directly proportional to oxidizer/fuel injection momentum ratios. The experimental performance and chamber heat transfer data obtained during this program exhibited similar operating trends. Although the subject program scope did not permit optimization of the performance/resultant wall environment characteristics of the coaxial pintle injector, it did provide preliminary design criteria applicable to future gas-liquid space storable propellant engine designs.

NOMENCLATURE

A	area	c	specific heat, constant, characteristic velocity	<u>Subscripts and Superscripts:</u>	
C	discharge coefficient	d	differential	—	average
D	diameter	f	friction factor	'	mixed properties, constant
E	activation energy	g	gravitational constant	a	axial
F	force	h	film coefficient, specific enthalpy	ad	adiabatic
H	enthalpy parameter	k	thermal conductivity	c	cold, cross-section
I	specific impulse	l	distance	F	fuel
K	constant	m	mass	H	hydraulic
L	length	r	radial	f	fuel, fluid
M	molecular weight	t	time, tangential	g	gas
P	pressure	v	vapor	i	inside
Q	heat rate, evaporation rate (lbs/ft ² sec)	w	flow rate, distance	l	liquid
R	gas constant, radius, reaction rate	x	distance	m	mean average, mass
T	temperature	y	distance	o	stagnation, oxidizer, outside
U	velocity	α	thermal diffusivity	p	at constant pressure
V	velocity	γ	specific heat ratio	r	radial
W	weight rate of flow	ρ	density	s	surface
R_e	Reynolds number	ϵ	strain	t	tangential
P_r	Prandtl number	σ	surface tension, stress	w	wall
MR	mixture ratio	μ	viscosity	v	vapor
CR	contraction ratio	η	efficiency	tot	total
a	constant, axial	ν	Poissons ratio		
b	constant				

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APPENDIX A

CALCULATION OF C* EFFICIENCY

The index of injector performance for the experimental program is the corrected C* efficiency. This parameter is calculated by two independent methods, one based on measurement of chamber pressure and the other on measurement of thrust. Details of the computational procedures and of the applied corrections are given in the following sections. The procedures and nomenclature format are essentially those as developed in NASA sponsored programs.

1. CHAMBER PRESSURE TECHNIQUE

Characteristic velocity efficiency based on chamber pressure is defined by the following:

$$\eta_{C^*} = \frac{(P_c)_o (A_t)_{\text{eff}} g_c}{(\dot{w}_T) (C^*)_{\text{theo}}} \quad (\text{A. 1})$$

where

$(P_c)_o$ = stagnation pressure at the throat

$(A_t)_{\text{eff}}$ = effective thermodynamic throat area

g_c = conversion factor (32.174 lbf-ft/lbf-sec²)

$\dot{w}_T$ = total propellant weight flow rate

$(C^*)_{\text{theo}}$ = theoretical characteristic velocity based on shifting equilibrium

Values calculated from Equation (A.1) are referred to as "corrected" C* efficiencies, because the factors involved are obtained by application of suitable influence factor corrections to measured parameters. Stagnation pressure at the throat is obtained from measured static pressure at start of nozzle convergence by assumption of isentropic expansion, effective throat area is estimated from measured geometric area by allowing for geometrical radius changes during firing and for nonunity discharge coefficient, and chamber pressure is corrected to allow for energy losses from combustion gases to the chamber wall by heat transfer and friction. Equation (A.1) may therefore be written as follows:

$$\eta_{C^*} = \frac{P_c A_t g_c f_p f_{TR} f_{DIS} f_{FR} f_{HL} f_{KE}}{(\dot{w}_o + \dot{w}_f) (C^*)_{\text{theo}}} \quad (\text{A. 2})$$

where

P_c = measured static pressure at start of nozzle convergence, psia

A_t = measured geometric throat area, in²

- g_c = conversion factor (32.174 lbf-ft/lbf-sec²)
 w_o = oxidizer weight flow rate, lb/sec
 w_f = fuel weight flow rate, lb/sec
 $(C^*)_{theo}$ = theoretical C^* based on shifting equilibrium calculations, ft/sec
 f_p = influence factor correcting observed static pressure to throat stagnation pressure
 f_{TR} = influence factor correcting for change in throat radius during firing
 f_{DIS} = influence factor correcting throat area for effective discharge coefficient
 f_{FR} = influence factor correcting measured chamber pressure for frictional drag of combustion gases at chamber wall
 f_{HL} = influence factor correcting measured chamber pressure for heat losses from combustion gases to chamber wall
 f_{KE} = influence factor correcting C^* values to account for finite chemical reaction rates

Methods of estimation of the various correction factors are described in the following paragraphs.

1.1 Pressure Influence Factor (f_p)

Measured static pressure at start of nozzle convergence is converted to stagnation pressure at the throat by assumption of effectively no combustion in the nozzle and application of the isentropic flow equations, with contraction ratio (A_c/A_t) and shifting-equilibrium specific heat ratios (γ). Frozen-equilibrium specific heat ratios usually make the influence correction factor about 1/2 percent larger. Hence, the value employed with shifting-equilibrium is the more conservative. Figure A-1 shows the influence factor as a function of contraction ratio.

1.2 Throat Radius Influence Factor (f_{TR})

Temperature gradients produced at the solid nozzle wall result in thermal stresses which affect throat radius, with the result that the geometric throat diameter ambient measurement is not the same as that which exists during firing.

For certain types of nozzles thermal penetration of the nozzle wall at the initiation of firing is small with respect to the wall thickness, hence the outer wall diameter is unchanged. The inner wall material will therefore expand toward the center, resulting in a decrease in throat diameter. As heat penetrates throughout the nozzle wall, the outer diameter will also increase, allowing outward expansion of the inner portion and consequent

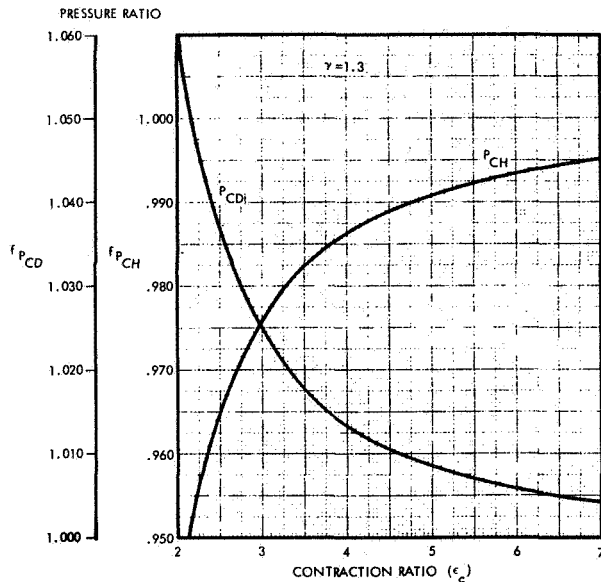


Figure A-1. Momentum Correction

for such nozzles can be estimated from the following for parabolic temperature distributions:

$$dr = \alpha \left[\frac{1 + \nu}{1 - \nu} \right] \frac{T_i - T_o}{(R_o - R_i)^2} \left[\frac{R_o^2 R_i}{2} - \frac{2}{3} R_o R_i^2 + \frac{R_i^3}{4} - \frac{R_o^4}{12 R_i} \right] \quad (A.3)$$

where

R_i = inner wall radius

R_o = outer wall radius

α = coefficient of linear expansion of wall material

ν = Poisson's ratio of wall material

T_i = temperature of inner wall

T_o = temperature of outer wall

The temperature distribution is given by

$$T = a + br + cr^2 \quad (A.4)$$

and is estimated by the method of Reference A-1. The throat shrinkage effects manifest themselves at the initiation of firing. For long steady-state firings, the throat size may actually increase, depending upon the

increase in throat diameter. Therefore, throat diameter during firing is a function of time, as well as of the physical properties of the throat material and the temperature and pressure of the combustion gases.

The actual computation is based upon integration of transient thermal stress equations for a hollow cylinder. A relatively simple expression results by assuming parabolic temperature distribution.

1.2.1 Thermal Effects

Since performance is influenced also by throat area changes, attention should be directed primarily to this zone of the thruster. With monomaterial construction, the throat effects can be analyzed with reasonable accuracy. It can be shown that throat shrinkage and expansion effects

temperature distribution and resultant stresses. Plastic as well as elastic deformations are readily included, as well as gas pressure effects.

For chambers employing thin wall throats the thermal growth is more easily obtained from the thermal expansion based on the temperature change from ambient temperature. The change in throat area can be written as:

$$\Delta A_{th} = \frac{\pi}{4} (2 + \alpha \Delta T)(\alpha \Delta T)D^2 \quad (A.5)$$

where

ΔA_{th} = change in throat area caused by thermal growth

α = average thermal expansion coefficient

ΔT = temperature rise from ambient conditions

D = throat diameter at ambient conditions

The throat area correction factor is as follows:

$$\begin{aligned} f_{TR} &= 1 + \frac{\Delta A_{th}}{A_{th}} \\ &= (1 + \alpha \Delta T)^2 \end{aligned} \quad (A.6)$$

The thermal expansion coefficient for graphite is $\alpha = 1.16 \times 10^{-6}$ in/in-°F. Assuming an ambient temperature of 70°F, the throat area correction factor becomes

$$f_{TR} = \left[1 + 1.16 \times 10^{-6} (T_{th} - 70) \right]^2 \quad (A.7)$$

T_{th} = throat temperature

1.3 Throat Discharge Coefficient Influence Factor (f_{DIS})

The discharge coefficient is defined as the ratio of actual flow rate through the throat to the theoretical maximum, based on geometric throat area and ideal, uniform, one-dimensional flow with no boundary layer. The discharge influence coefficient may be estimated in two ways: one based on calculations made from a theoretical, inviscid flow model of combustion products, and the other based on a correlation of results obtained in various experimental study results of air flow through nozzles of similar geometry.

1.3.1 Theoretical Model

Total mass flow rate is given by

$$m = \int_0^A \rho V dA \quad (A.8)$$

where

ρ = gas density

V = gas velocity

A = cross-sectional area

Theoretical maximum flow rate at the throat is

$$m_{\max} = \int_0^{A_t} \rho^* V^* dA \quad (A.9)$$

where

A_t = geometric area of the throat

ρ^* = sonic gas density

V^* = sonic gas velocity

For ideal, uniform, parallel flow, Equation (A.8) becomes

$$m_{\max} = \rho^* V^* A_t \quad (A.10)$$

The discharge coefficient is then

$$C_D = \frac{\dot{m}}{\dot{m}_{\max}} = \int_0^A \left(\frac{\rho}{\rho^*} \right) \left(\frac{V}{V^*} \right) \left(\frac{dA}{A_t} \right) \quad (A.11)$$

1.3.2 Empirical Value

Experimental conical nozzle discharge coefficients obtained with air by various investigators are plotted in Figure A-2 against the indicated geometric parameter. Data sources also are listed in Figure A-2.

The values obtained by both methods are found to be in excellent agreement.

1.4 Frictional Drag Influence Factor (f_{FR})

Calculations of C^* based on chamber pressure are concerned with chamber phenomena up to the nozzle throat. Drag forces to this point are generally small and can be estimated from the Blasius flat plate approximation for the local skin coefficient and integration of the local shear forces. The results are shown in Figure A-3 as a function of chamber length.

1.5 Energy Loss Influence Factor (f_{HL})

Chamber pressure and thrust are decreased by heat transfer from the combustion gases to the walls of a thrust chamber. This enthalpy

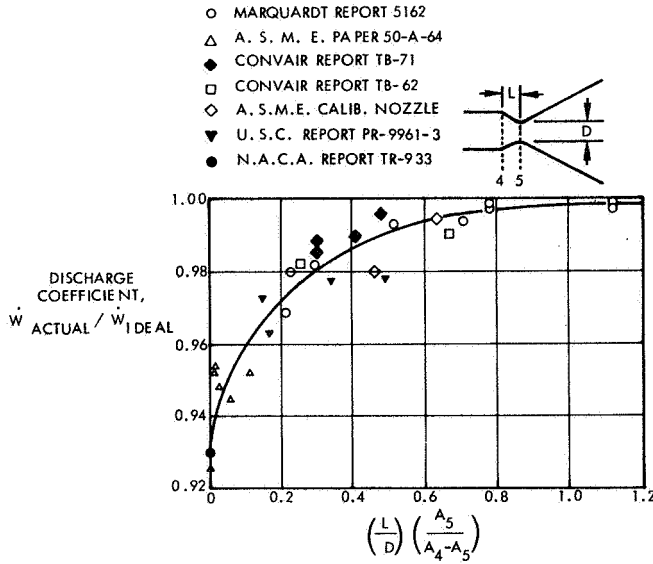


Figure A-2. Conical Nozzle
 Discharge Coefficient

where

V_c = gas velocity at chamber exit

V_t = gas velocity at nozzle throat

H_c = gas enthalpy at chamber exit

H_t = gas enthalpy at nozzle throat

$\dot{Q}_{conv}$ = heat loss in nozzle convergence

Velocity at the throat is given by:

$$V_t = \left[V_c^2 + 2(H_c - H_t - \dot{Q}_{conv}) \right]^{1/2} \quad (A.13)$$

With negligible nozzle inlet velocity

$$V_t = \left[2(H_c - H_t - \dot{Q}_{conv}) \right]^{1/2} \quad (A.14)$$

Logarithmic differentiation of Equation (A.14) gives

$$\frac{dV_t}{V_t} = 1/2 \frac{d(H_c - H_t - \dot{Q}_{conv})}{(H_c - H_t - \dot{Q}_{conv})} = 1/2 \left(\frac{dH_c - dH_t}{H_c - H_t - \dot{Q}_{conv}} \right) \quad (A.15)$$

loss is substantially reduced in ablative chambers and is effectively recovered in a regeneratively cooled chamber.

The effect on C^* of enthalpy loss by heat transfer can be estimated from a loss of chamber enthalpy. This is determined from a two station energy balance, one at the start of nozzle convergence and the other at the throat which

$$\begin{aligned} & 1/2 V_c^2 + H_c \\ &= 1/2 V_t^2 + H_t + \dot{Q}_{conv} \end{aligned} \quad (A.12)$$

Substitution of enthalpy definition into Equation (A. 15) gives

$$\frac{dV_t}{V_t} = 1/2 \left(\frac{c_{pc} dT_c - c_{pt} dT_t}{H_c - H_t - \dot{Q}_{conv}} \right) \quad (A. 16)$$

With constant C_p between the two stations,

$$\frac{dV_t}{V_t} = 1/2 \left(\frac{c_p dT_c}{H_c - H_t - \dot{Q}_{conv}} \right) \left(1 - \frac{dT_t}{dT_c} \right) \quad (A. 17)$$

If the specific heat ratio, γ , is assumed constant,

$$\frac{dT_t}{dT_c} = \frac{T_t}{T_c} \quad (A. 18)$$

Substituting Equation (A. 18) into Equation (A. 17), replacing differentials by incrementals, and noting that C^* is proportional to gas velocity at the throat gives:

$$\frac{\Delta V_t}{V_t} = \frac{\Delta C^*}{C^*} = 1/2 \left(\frac{c_p \Delta T_c}{H_c - H_t - \dot{Q}_{conv}} \right) \left(1 - \frac{\Delta T_t}{T_c} \right) \quad (A. 19)$$

Total heat loss to the chamber walls, in Btu per pound of propellant, is obtained by summation of observed heat fluxes over the appropriate areas:

$$\text{Heat loss} = \frac{\Sigma(q/A) A}{\dot{w}_T} \quad (A. 20)$$

where

q/A = experimentally observed heat flux

A = area applicable to each q/A value

$\dot{w}_T$ = total propellant flow rate

If this heat loss is equated to the change in enthalpy of the gas in the combustion chamber, $c_p \Delta T_c$, then substitution in Equation (A. 21) gives:

$$\frac{\Delta C^*}{C^*} = 1/2 \left[\frac{\Sigma(q/A)A}{\dot{w}_T} \right] \left[\frac{1 - (T_t/T_c)}{H_c - H_t - Q_{conv}} \right] \quad (A. 21)$$

The applicable influence factor is

$$f_{HL} = 1 + \frac{\Delta C^*}{C^*} = 1 + 1/2 \left[\frac{\Sigma(q/A)A}{\dot{w}_T} \right] \left[\frac{1 - (T_t/T_c)}{H_c - H_t - Q_{conv}} \right] \quad (A.22)$$

An alternate expression can be obtained from the basic C^* definition:

$$C^* = \frac{\sqrt{RT_c}}{\Gamma} \quad (A.23)$$

Logarithmic differentiation of this yields:

$$\frac{dc^*}{c^*} = \frac{1}{2} \frac{dT_c}{T_c} \quad (A.24)$$

Substituting incrementals from differentials in Equation (A.24) gives:

$$\frac{\Delta c^*}{c^*} = \frac{1}{2} \frac{\Delta T_c}{T_c} \quad (A.25)$$

Equating ΔT_c with the heat loss from Equation (A.20) results in the following:

$$\frac{\Delta c^*}{c^*} = \frac{1}{2} \left[\frac{\Sigma(q/A)A}{\dot{w}_t} \right] \left[\frac{1}{c_p T_c} \right] \quad (A.26)$$

The applicable influence factor is:

$$f_{HL} = 1 + \frac{1}{2} \left[\frac{\Sigma(q/A)A}{\dot{w}_t} \right] \left[\frac{1}{c_p T_c} \right] \quad (A.27)$$

where

c_p = specific heat at constant pressure

Although derived independently it can be shown that these two expressions, Equations (A.22) and (A.27), are nearly equivalent.

1.6 Influence Factor for Chemical Kinetics (f_{KE})

The effect of finite chemical reaction rates is to produce a C^* less than the corresponding theoretical equilibrium values. A TRW Systems developed one-dimension nonequilibrium reacting gas computer program is employed with reaction rate constants selected for the propellant system. The fluid mechanical and chemical equations are integrated from the inlet section by an implicity technique.

2. CALCULATIONS BASED ON THRUST

The alternate determination of C^* efficiency is based on thrust:

$$\eta_{C^*} = \frac{F_{vac} g_c}{(C_F)_{vac} \dot{w}_T C^*_{theo}} \quad (A. 28)$$

where

F_{vac} = measured thrust corrected to vacuum conditions
by the equation: $F_{vac} = F + P_a A_e$

F = measured thrust, lbf

P_a = ambient pressure, psia

A_e = area of nozzle exit, in²

g_c = conversion factor (32.174 lbf-ft/lbf-sec²)

$(C_F)_{vac}$ = theoretical shifting thrust coefficient (vacuum)

$\dot{w}_T$ = total propellant flow rate, lbf/sec

C^*_{theo} = theoretical shifting-equilibrium characteristic
velocity, ft/sec

Values of vacuum thrust are obtained by applying corrections to sea-level measurements. With these values, which include allowances for all important departures from ideality, theoretical thrust coefficients may be used for calculation of C^* . C_F efficiency is taken as 100 percent if there is no combustion in the nozzle, if chemical equilibrium is maintained in the nozzle expansion process, and if energy losses from the combustion gases are accounted for.

Applicable influence factors for measured thrust are specified in the following equation:

$$\eta_{C^*} = \frac{(F + P_a A_e) g_c \phi_{FR} \phi_{DIV} \phi_{HL} \phi_{KE}}{(C_F)_{theo} (\dot{w}_o + \dot{w}_f) (C^*)_{theo}} \quad (A. 29)$$

where

F = measured thrust, lbf

P_a = ambient pressure, psia

A_e = area of nozzle exit, in²

g_c = conversion factor (32.174 lbf-ft/lbf-sec²)

$(C_F)_{\text{theo}}$ = theoretical shifting thrust coefficient
(vacuum)

$\dot{W}_o$ = oxidizer weight flow rate, lbm/sec

$\dot{W}_f$ = fuel weight flow rate, lbm/sec

$(C^*)_{\text{theo}}$ = theoretical shifting equilibrium characteristic
velocity, ft/sec

ϕ_{FR} = influence for frictional losses

ϕ_{DIV} = influence factor for nozzle divergence

ϕ_{HL} = influence factor for heat losses to chamber and
nozzle walls

ϕ_{KE} = influence factor correcting C^* and C_F values to
account for finite chemical reaction rates

The influence factors in Equation (A. 29) are applied to vacuum thrust $(F + P_a A_e)$ instead of to measured site thrust (F) because, for convenience, the factors are readily calculated as changes in efficiency based on theoretical vacuum parameters. The total influence factor is then of the form $\Delta F/F_{\text{vac}}$.

Implicit in the use of theoretical C_F values are corrections to geometric throat area and to measured static chamber pressure at start of nozzle convergence. Therefore, calculation of corrected C^* efficiency from thrust measurement includes all the previously described corrections plus an additional one to account for nonparallel nozzle exit flow. However, because $(C_F)_{\text{theo}}$ is essentially independent of small changes to chamber pressure and contraction ratio which are involved in corrections to P_c and A_t , these corrections are of no practical significance in calculation of C^* from thrust measurements.

2.1 Influence Factor for Frictional Drag (ϕ_{FR})

This factor corrects for energy losses caused by viscous drag forces on the thrust chamber walls. Its magnitude is estimated by a boundary layer analysis utilizing the integral momentum equation for turbulent flow, which accounts for boundary layer effects from the injector to the nozzle exit by suitable description of the boundary layer profile and local skin friction coefficient. A computer program is used to carry out a numerical integration of the equation, including effects of pressure gradient, heat transfer, and surface roughness. The program requires a potential nozzle flow solution obtained from variable-property, axisymmetric method of characteristics calculation of the flow field outside the boundary layer; corresponding properties for the subsonic combustion chamber flow field are also calculated. The calculated values of the frictional influence factor are shown in Figure A-3.

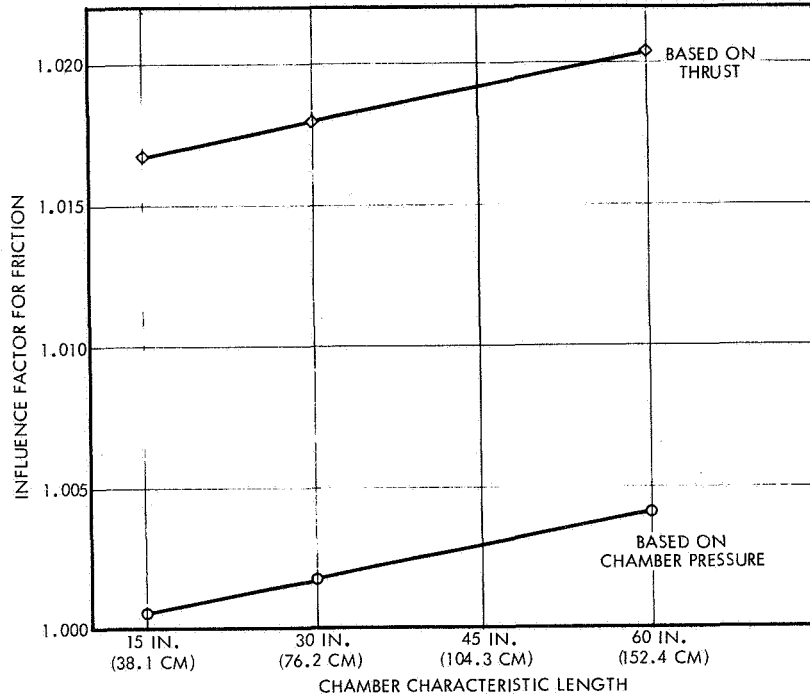


Figure A-3. Friction Effects Influence Factor Based on Thrust and Chamber Pressure as a Function of Characteristic Length

2.2 Influence Factor for Nozzle Divergence (ϕ_{DIV})

The one-dimensional theoretical performance calculations assume that flow at the nozzle exit is uniform and parallel to the nozzle axis. The influence factor, ϕ_{DIV} , allows for nozzle divergence (i. e., for nonaxial flow) and for nonuniformity across the nozzle exit plane. It is calculated by a computer program which utilizes the axisymmetric method of characteristics for a variable-property gas. Computation begins with a transonic input near Mach 1, providing a characteristic line for use in the analysis of the supersonic portion of the nozzle. The resulting pressures are integrated over the given geometry to give the geometric efficiency.

2.3 Influence Factor for Heat Loss (ϕ_{HL})

To obtain the heat loss influence factor from measured thrust the approach is identical to that taken previously from the pressure measurement, except that the nozzle losses must also be included. With constant specific heat and gamma from start of nozzle convergence to exit, Equation (A. 22) becomes

$$\phi_{HL} = 1 + \frac{1}{2} \left[\frac{\sum(q/A)A}{\dot{w}_T} \right] \left[\frac{1 - T_e/T_c}{H_c - H_e - Q_{nozzle}} \right] \quad (A. 30)$$

when "e" corresponds to the exit condition, and the summation occurs over the entire combustion.

An alternate can also be derived as in Equation (A.27). This equation becomes

$$\phi_{HL} = 1 + \frac{1}{2} \left[\frac{\sum(q/A)A}{\dot{w}_T} \right] \left[\frac{1}{c_p T_e} \right] \quad (A.31)$$

2.5 Numerical Example with Experimental Data

This section presents a numerical example of performance data reduction and correction. The data from test HB2-181 are analyzed in this example and are typical of all tests. A performance evaluation test in the 30-inch (76.2 cm) L^* chamber at nominal operating conditions was analyzed. Steady-state data from this test, summarized in Table A-1, were extracted from the digital data printout. Strip chart traces of the same parameters are shown as Figure A-4.

Table A-1. Test HB2-181 Data

Parameter	Numerical Value
Static Chamber Pressure*	$(466.18 + 466.45)/2 = 466.21$ psia (3210) KN/M ²
Oxidizer Flow Rate**	$(8.060 + 8.058)/2 = 8.059$ lb/sec (3.66) kg/sec
Fuel Flow Rate**	$(1.280 + 1.239)/2 = 1.260$ lb/sec (0.57) kg/sec
Measured Thrust	2790.9 lb _f (12400 N)
Throat Diameter	2.298 inch (5.84 cm)

* Measured at two locations 180 degrees apart at the start of nozzle convergence.

** Flow rates measured by two flow meters in series.

2.6 Performance Based on Chamber Pressure

Characteristic velocity efficiency based upon chamber pressure was calculated by Equation (A.2). Two chamber pressure measurements were taken at the injector face and two at a location 1/2 inch (1.27 cm) from the start of nozzle convergence. Calculation of the various correction factors to be applied to the raw data of test HB2-181 are described below.

2.6.1 Influence Factor for Pressure Correction, f_p

The pressure correction factor is a function of chamber contraction ratio and gas specific heat ratio. A correction factor can be applied to either the head end chamber pressure or the chamber pressure at the start of nozzle convergence. The correction factors for this program are shown in Figure A-1. The value of f_p for this program is 1.0135.

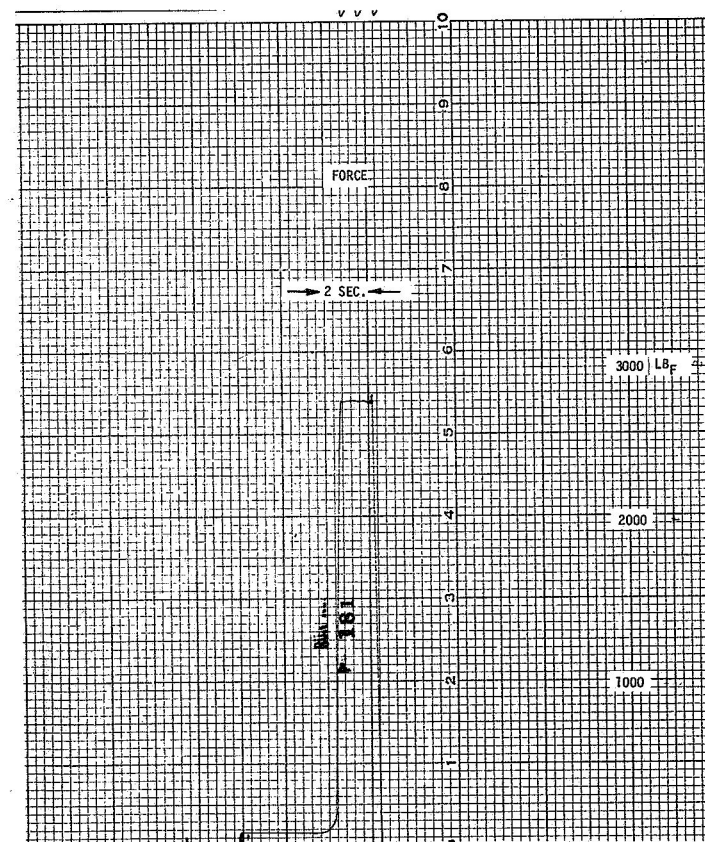
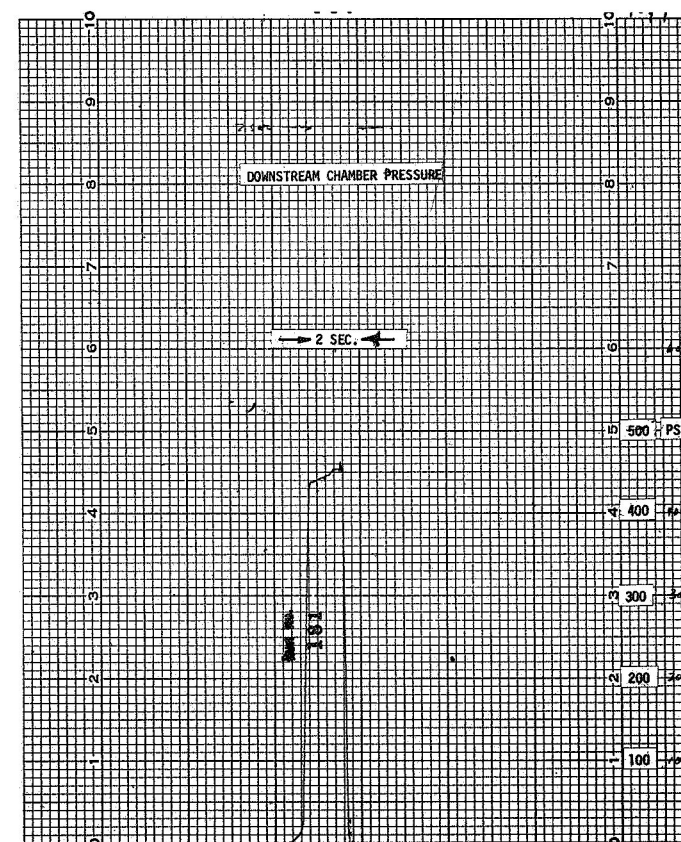
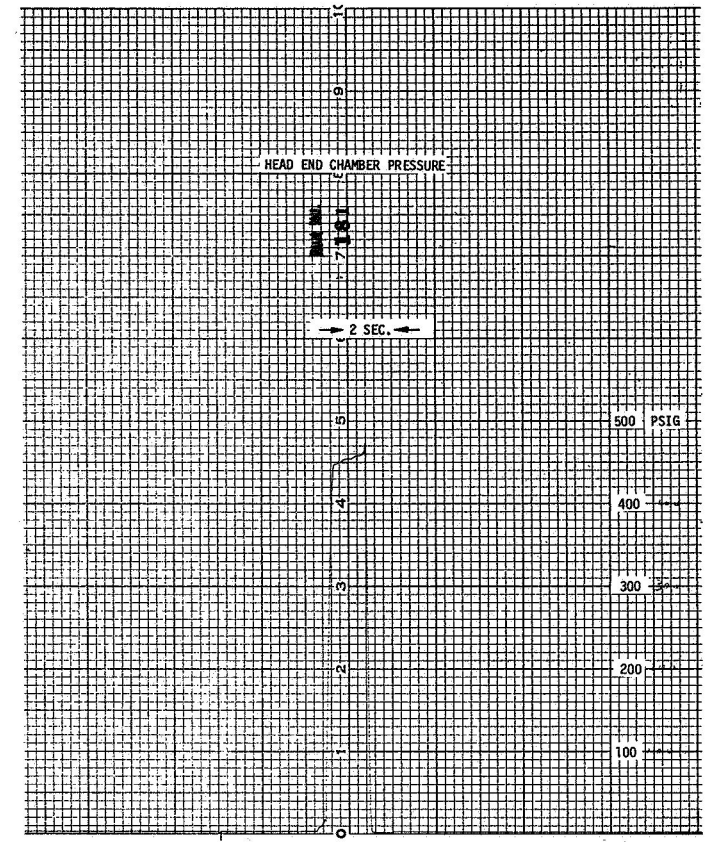
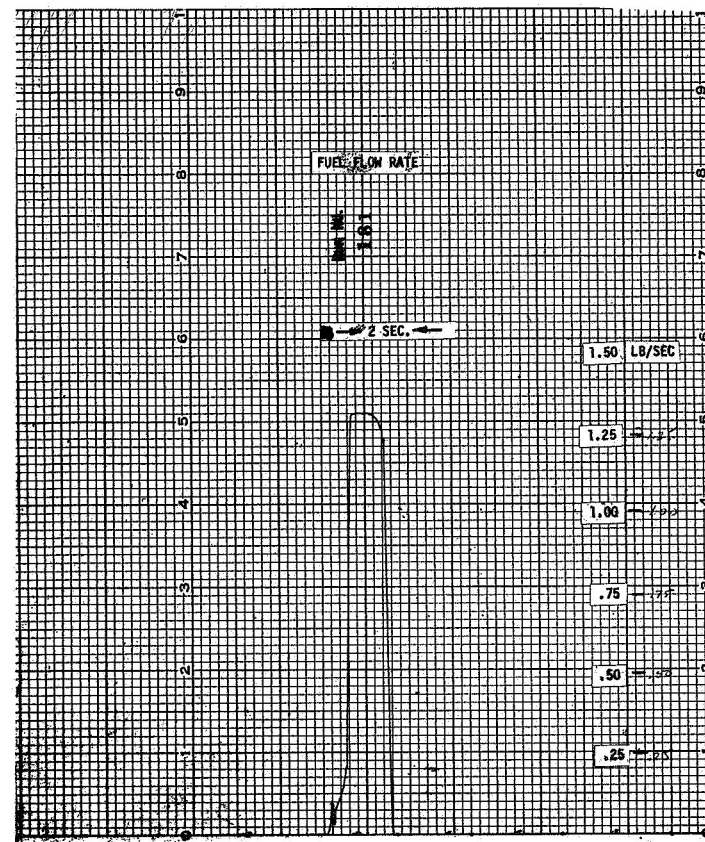
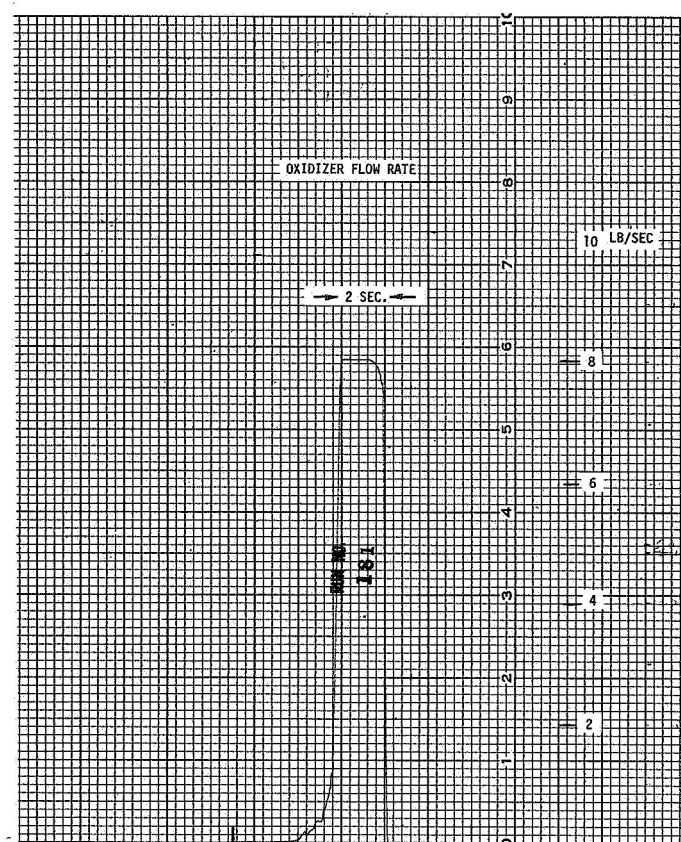


Figure A-4. Strip Chart Data from Test HB2-181

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2.6.2 Influence Factor for Friction, f_{FR}

The calculated influence factor for frictional losses was presented in Figure A-3. For the conditions of this test the influence factor was estimated to be 1.002.

2.6.3 Influence Factor for Throat Radius (f_{TR})

The influence factor for throat area changes was minor over the time interval of interest. Using Equation (A.3) the throat radius change was calculated at a throat wall temperature of 1000°F (538°C). The calculated throat radius change was 0.0015 inch (0.0038 cm). The influence factor was determined in terms of area changes by

$$f_{TR} = \frac{R^2}{R^2 - 2Rdr + dr^2}$$

The calculated influence factor, f_{TR} , was

$$f_{TR} = 0.999$$

2.6.4 Throat Discharge Coefficient Influence Factor, f_{DIS}

The throat discharge coefficient was taken from Figure A-2 as 0.995 for the hardware tested during this program.

2.6.5 Energy Loss Influence Factor, f_{HL}

The heat loss effect on the chamber pressure computation technique is readily accomplished once the axial heat flux profile has been established. For the conditions of this test the heat loss influence factor was computed from Equation (A.27) as

$$\begin{aligned} f_{HL} &= 1 + \frac{1}{2} \left[\frac{\Sigma(q/A)A}{\dot{w}_T} \right] \left[\frac{1}{C_p T_c} \right] \\ &= 1 + \frac{1}{2} \left[\frac{1035}{9.32} \right] \left[\frac{1}{(0.407)(2111)} \right] \\ &= 1.017 \end{aligned}$$

2.6.6 Corrected C* Calculation

The tabulated test data and calculated influence factors were substituted into Equation (A. 2). The corrected C* efficiency based on chamber pressure was calculated as follows:

$$(\eta_{C^*})_{PC} = \frac{P_c A_t g_c f_p f_{FR} f_{TR} f_{DIS} f_{HL}}{(\dot{w}_{ox} + \dot{w}_f) C_{THEO}^*}$$

$$(\eta_{C^*})_{PC} = \frac{(466.21)(4.15)(32.17)(1.0135)(1.002)(0.999)(0.995)(1.017)}{(8.059 + 1.269)(7020)}$$

$$(\eta_{C^*})_{PC} = 0.977$$

2.7 Performance Based on Thrust

Calculation of corrected C* efficiency based on thrust was by means of Equation (A. 29). The measured thrust was corrected to vacuum conditions as follows:

$$\begin{aligned} F_{vac} &= F_{meas} + P_a A_e \\ &= 2790.6 + (14.54)(20.2) = 3084.6 \text{ lbf} \end{aligned}$$

2.7.1 Influence Factor for Frictional Drag, ϕ_{FR}

The influence factor for frictional effects on thrust was calculated using a computer program described in paragraph 2.1 of this appendix and plotted in Figure A-3. For the test conditions presented in this example the influence factor, ϕ_{FR} , was estimated to be 1.018.

2.7.2 Nozzle Divergence Influence Factor, ϕ_{DIV}

The nozzle divergence influence factor was taken as 1.017 for all tests conducted on this program.

2.7.3 Heat Loss Influence Factor, ϕ_{HL}

The heat loss influence factor for performance based on measured thrust was calculated from Equation (A. 31). This method of calculation of ϕ_{HL} is similar to the calculation of the chamber pressure correction factor except that it is based upon total thrust chamber heat losses which include the heat loss in the divergent section of the nozzle. The heat loss

correction factor for thrust was calculated as follows:

$$\begin{aligned}\phi_{HL} &= 1 + \frac{1}{2} \left[\frac{\sum(q/A)A}{w_t} \right] \left[\frac{1}{C_p T_e} \right] \\ &= 1 + \frac{1}{2} \left[\frac{1179}{9.32} \right] \left[\frac{1}{(0.385)(4440)} \right] \\ &= 1.024\end{aligned}$$

2.7.4 Corrected C* Calculation

The measured test data and the calculated influence factors were substituted into Equation (A.29). The corrected C* efficiency based on thrust was then:

$$\begin{aligned}(\eta_{C^*})_F &= \frac{(F + P_a A_e) g_c \phi_{FR} \phi_{DIV} \phi_{HL}}{C_{F_{vac}} (\dot{w}_{ox} + \dot{w}_F) C_{theo}^*} \\ (\eta_{C^*})_F &= \frac{(2790.9 + 293.7)(32.17)(1.018)(1.017)(1037)}{(1.6568)(9.319)(7020)} \\ (\eta_{C^*})_F &= 0.983\end{aligned}$$

APPENDIX B

EXPERIMENTAL FACILITY AND PROCEDURES

This appendix describes the experimental facilities utilized during this program to characterize the 3K FLOX/methane injector. The cold-flow and hot-firing facilities are discussed individually.

1. COLD FLOW FACILITY AND PROCEDURES

The injector cold flow evaluation conducted during this program encompasses two parts. The basic hydraulic operating characteristics were determined from pressure drop, mass flow, and visual observations. Second, cold flow characterization for the purpose of correlating with hot firing results was attempted. With respect to the latter effort, it marked the second effort at TRW to provide such characterization for the coaxial injector. All cold flow work was conducted at TRW's Capistrano Test Site Hydraulic Laboratory.

The initial phase of the cold-flow experimental program was performed using the same apparatus as the second phase but without provision for collection and sampling.

An effort was made to correlate cold flow patternation data with hot firing results. These data are presented in Section 5.

In order to obtain meaningful coaxial pintle cold flow data to correlate with hot firing data, a different approach to collection of cold flow simulants is required. Both circumferential and longitudinal distributions must be determined. The mass and mixture ratio distributions were obtained with the use of the apparatus shown in Figure B-1. This water flow patternation fixture, developed by TRW, is located at the Capistrano Test Site's Hydraulic Laboratory. This fixture has the capability of measuring the mass and mixture ratio distribution of an injector by diverting the fluids into 1/4-inch tubes located about 6 inches from the injector. This equipment collects the flow from the injector in tubes spaced both circumferentially and longitudinally, resulting in a determination of the circumferential uniformity of the spray and in the mass and mixture ratio gradients through the spray fan. Mixture ratio was measured by flowing water through the oxidizer side and water dyed with methylene blue through the fuel side because without a pressurized cold flow apparatus it was impossible to simulate the actual fuel density.

Translation from simulant to actual propellants is made by:

$$(MR)_{\text{propellants}} = (MR)_{\text{sim}} \left[\left(\frac{\rho_o}{\rho_f} \right)_{\text{prop}} \left(\frac{\rho_f}{\rho_o} \right)_{\text{sim}} \right]^{1/2}$$

Circumferential distribution is found to be easily achieved with the radial flow coaxial injector and minor nonuniformities do not severely affect either the performance or wall thermal environment. Prior to hot firing,

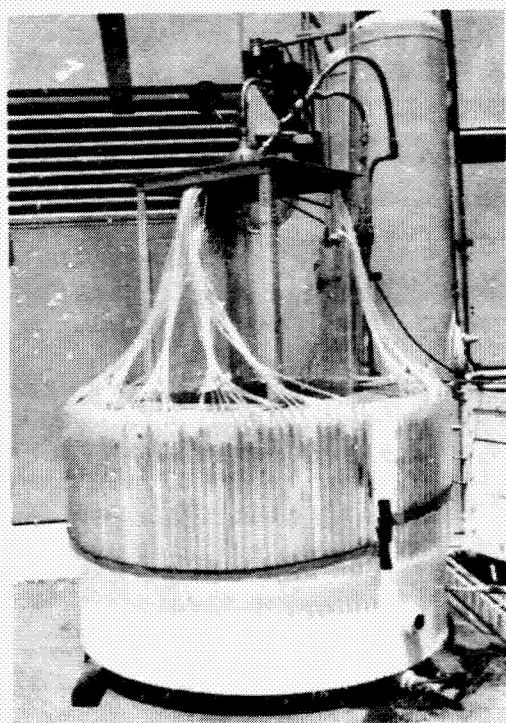
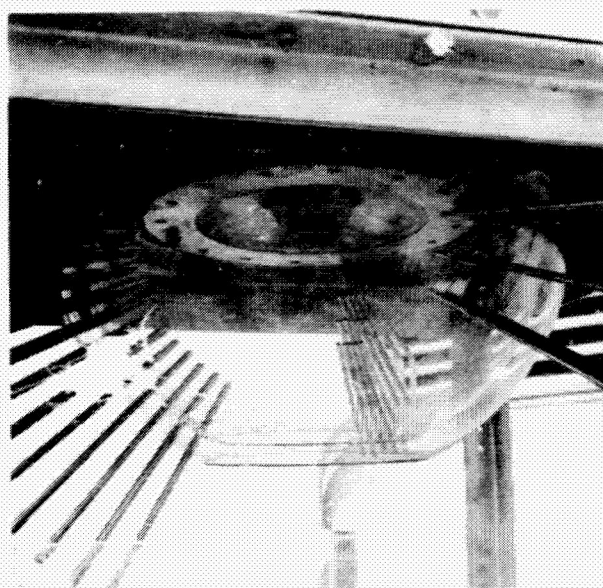


Figure B-1. Injector Patternation Fixture

each injector configuration was water-flowed to insure basic circumferential uniformity. However, the longitudinal distribution of both mass and mixture ratio is found to be of primary importance to both performance and thermal environment. The collector used has tubes spaced 5 degrees apart in the range from 30 to 90 degrees from the centerline. This provides for a detailed measurement of the gradients through the resulting spray fan.

A stream tube analysis can be used to determine the combustion efficiency of any injector design from the cold flow data. Several assumptions are implicit in the case of cold flow data to predict hot firing results:

- No reaction effects on mixing
- No propellant vaporization effects on mixing
- No secondary mixing downstream

In addition to these, if the ratio of specific heats variation is small for wide mixture ratio variation, the cold flow-hot firing correlation is quite simple. Each collector tube is taken as a stream tube having mass, $\dot{w}_i$, and mixture ratio, r_i . The mass in each tube is normalized with respect to the total mass sample collected, $\dot{w}_i/\dot{w}_t$. The tube mixture ratio can be equated to a C_i^* for that tube. The total predicted C^* is the sum of the predicted C_i^* 's from each tube, $\dot{w}_i/\dot{w}_t C_i^*$.

$$C^* = \sum_i \frac{\dot{w}_i}{\dot{w}_t} C_i^*$$

The predicted combustion efficiency is the ratio of the predicted C^* and the theoretical equilibrium C^* for the nominal injector mixture ratio.

$$\eta_{C^*} = C^*/C^*_{\text{theo}}$$

The oxidizer elements were cold-flow tested to determine the predicted combustion efficiency and expected wall environment. The mass and mixture ratio distribution for these elements, simulating a MR of 5.75, are shown in Section 5.

2. HOT FIRE TEST FACILITY, INSTRUMENTATION AND PROCEDURES

The experimental facility used for testing of the 3K FLOX/methane injector was the High Energy Propellant Test Stand (HEPTS) located on the northern boundary of TRW Systems Capistrano Test Site.

HEPTS is a multiposition test complex comprising a control center and two major test modules (A and B, each fitted with tank bays and two vertical open-air firing positions), plus two low-thrust altitude test positions and a complete propulsion system altitude test cell. The HEPTS B-2 cell, which was utilized for this program, contains a position for firing fluorine oxidized propellants in engines rated to 10,000 lbf (44.48 KN) thrust. The HEPTS B-2 cell propellant supply is a 150-gallon (.566 m³), 1000 psig (6.89 MN/m²) working pressure oxidizer tank and a 16.7 cubic

foot (.47 m³), 6000 (41.4 MN/m²) psig fuel tank. The oxidizer tank is a triple jacketed cryogenic tank. The FLOX is plumbed to the test stand through liquid nitrogen jacketed feed lines up to the fire valves. All the valves in the oxidizer system are rated for cryogenic service and cooled with liquid nitrogen. Oxidizer feed lines downstream of the fire valves are evaluated. A nitrogen cascade system pressurizes the oxidizer tankage and supplies engine purges. Figure B-2 illustrates the general arrangement of the HEPTS B module with the 3K FLOX/methane engine installed. Figure B-3 is a schematic of the feed system.

Engine test data are transmitted via twin conductor-shielded cables from the test position to the HETPS Control Center. An SDS 925 (and/or SDS 930) computer in the Main Control Center is programmed for post-test computation of most of the important rocket engine performance parameters. Input is prerecorded tape from the digital tape recorders. It also provides engineering units reduction of parameter data recorded on the digital tape. Primary output devices are a 120-character, 10 line-per-second printer, and an 8-channel digital-to-analog converter.

2.1 Thrust Measurement

The thrust measurement stand mounted in the HEPTS B-2 cell is a double A-frame configuration bolted directly to the stand structure. (See Figure B-2.) The engine bolts to a yoke which transmits the force through an Ormond flexure to a dual bridge load cell (measurement load cell) firmly fixed to the stand structure. Axial location of the yoke is maintained by two flexures at 90 degrees to one another. Original alignment of the yoke, accomplished by sighting with a transit, is maintained by two flexures at 90 degrees to one another. Remote calibration is achieved using an electric motor and reduction gear arrangement with a standard load cell mounted atop the thrust stand. The gear box converts the rotary motion of the motor into translational motion of a shaft which presses down on the standard cell putting it in compression. The plate upon which the gear box is mounted floats freely above the standard cell supported on three tie rods from the main thrust yoke. As force is applied to the calibration cell, tension is placed on the tie rods which draw up on the thrust yoke, placing the measurement cell in compression equal to that on the standard cell. Calibration force levels are set by reading the standard cell output on a digital voltmeter. This calibration method has demonstrated itself to be quick, reliable, and trouble-free, allowing actual force calibrations of the thrust system with the engine installed immediately prior to testing. This thrust stand and calibrator have been tested extensively for accuracy and repeatability. Numerous calibrations were run during which the thrust stand demonstrated very low nonlinearity. Repeatability was excellent. As a final check, an accurately known force was placed against the thrust yoke, simulating a firing. This force was reported by the system with an accuracy of better than 0.1 percent at the recorder.

2.2 Flow Measurement

Flow rate measurement of the oxidizer was achieved with Potter turbine flowmeters. The reliability of these meters has been such that

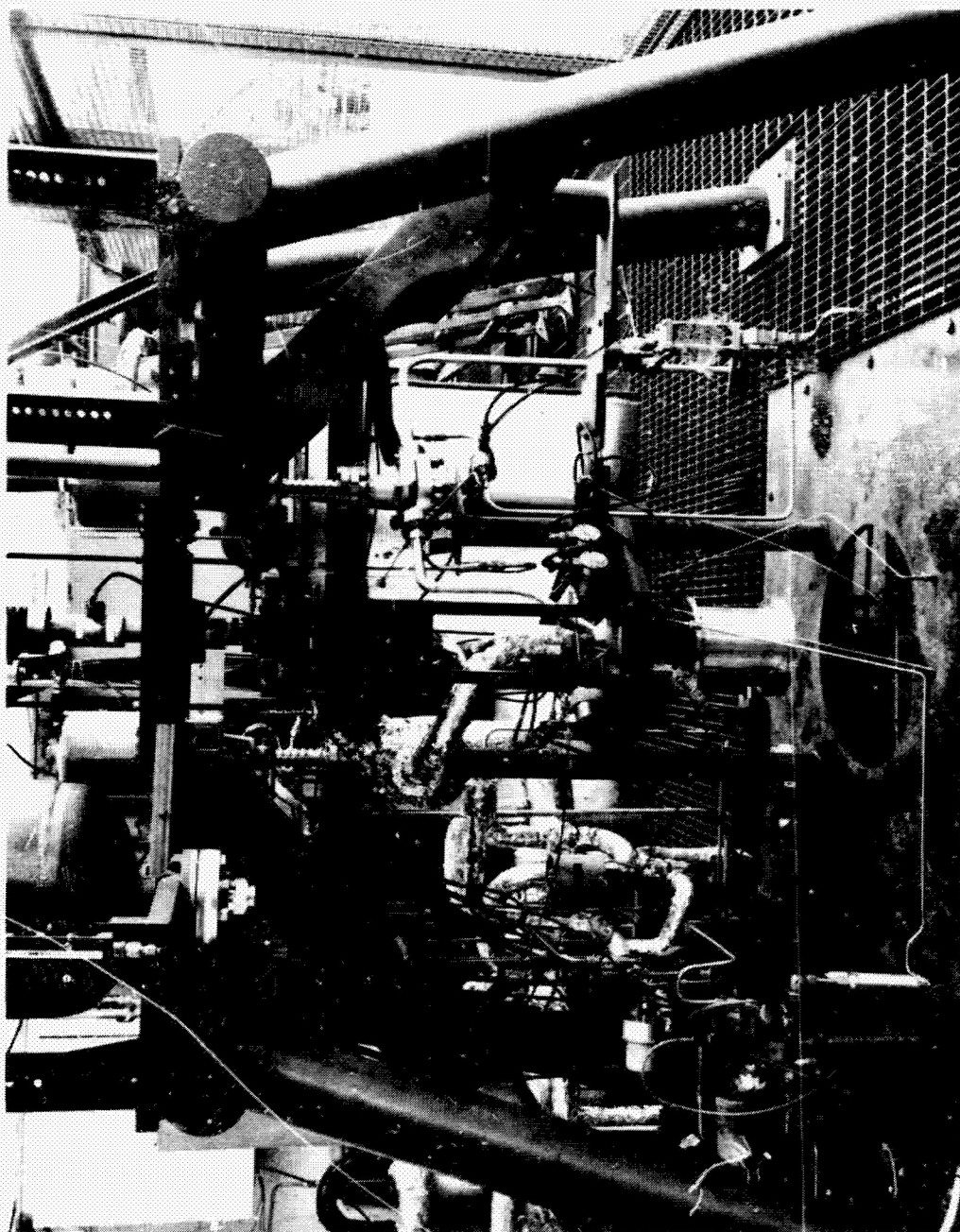


Figure B-2. 3K FLOX/Methane Engine Installed in HB-2 Test Stand

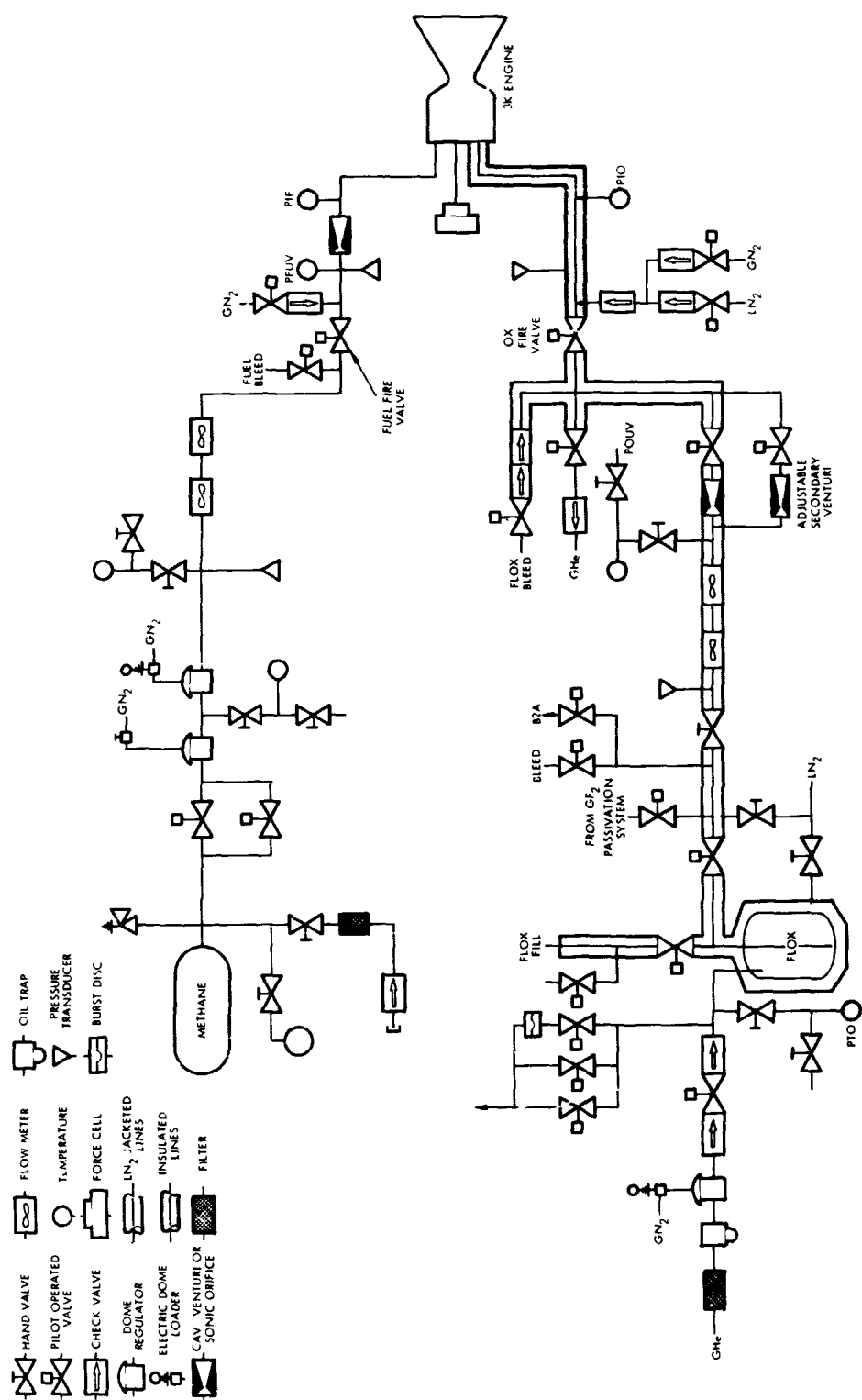


Figure B-3. Flow Schematic of HEPTS Position B-2

one pair in the FLOX line was used for the entire program with no problems. The oxidizer flowmeters were Cox turbine type, calibrated at the manufacturer's facility. Secondary fuel flow measurement was made by means of a sonic orifice and pressure transducer.

The flowmeter calibration facility at CTS is a closed loop system, incorporating cavitating venturis downstream of the flowmeters to prevent any effect on flow rate of receiver tank pressure rise. The system includes a 400-gallon run tank, a 300-gallon catch tank, and an electronic counter for determining the weight of propellant transferred into the receiver tank. This weight system is dead-weight calibrated prior to each flowmeter series. Error analysis of this system indicated accuracy in determination of meter constants on the order of 0.25 percent.

2.3 Pressure Measurement

All pressure transducers used on the program were Taber-Teledyne strain gage transducers with the exception of a Photocon Model 5327 transducery to measure high frequency chamber pressure fluctuations. These transducers were dead-weight calibrated on a regular 3-month basis or more frequently if there was evidence of erratic operation.

2.4 Injector Performance Data Reduction

All raw performance data were recorded on digital tape and reduced by means of the Capistrano Test Site SDS 925 digital computer on a run-to-run basis. The test firings are characterized by two performance parameters: delivered specific impulse (I_{sp}) and characteristic exhaust velocity (C^*). The characteristic exhaust velocity is computed by:

$$C^* = \frac{P_o A_T g_o}{\dot{w}_T}$$

where:

P_o = nozzle stagnation pressure, psia

A_T = nozzle throat area, in²

g_o = gravitational constant, 32.174 ft/sec²

$\dot{w}_T$ = total propellant weight flow, lb/sec

Each of the parameters within the C^* equation computed as follows. The nozzle stagnation pressure (P_o) computed from measured nozzle entrance chamber static pressure (P_{CD}) by $P_o/P_{CD} = 1.0135$. (The P_o/P_{CD} ratio determined from classical gas dynamic relationships for total to static pressure ratio as a function of contraction ratio and gas specific heat ratio.) The nozzle static pressure (P_{CD}) computed from the average of two measurements. The head end static chamber pressure, P_{CH} , also recorded to empirically define the P_{CD}/P_{CH} ratio to permit C^* determination in the ablative thrust chamber test where P_{CD} was not directly measured.

The propellant flow rates were computed from the average of a pair of turbine flowmeters. The actual flow rate was computed from the frequency output of the meter, the calibration factor, and the propellant densities based on measured propellant temperatures.

The nozzle throat area was computed from the average of three throat diameter measurements taken prior to each test.

The combustion efficiency (η_{C^*}) was determined by dividing the computed C^* by the theoretical equilibrium characteristic exhaust velocity at the appropriate run nozzle stagnation pressure and oxidizer-to-fuel mixture ratio. The variation of the theoretical equilibrium C^* with nozzle stagnation pressure and mixture ratio was provided as a tabular input to the data reduction program.

The engine I_{sp} was computed on the basis of the measured thrust and total propellant weight flow.

$$I_{sp} = \frac{F, \text{ measured thrust, lb}}{\dot{w}_T, \text{ measured total flow rate, lb/sec}}$$

The measured thrust is determined from the load cell output and the pre-run calibration. The measured specific impulse data were also corrected to vacuum conditions by adding to the measured thrust a correction equal to the product of the nozzle exit area times the ambient pressure.

All parameters affecting either hydraulic or combustion performance of the injector were measured redundantly to insure the validity of all measurements. The redundantly measured parameters were: oxidizer injector pressure, fuel injection pressure, head end chamber pressure, downstream chamber pressure, oxidizer flow rate, fuel flow rate, oxidizer temperature and fuel temperature. The oxidizer mixture (FLOX) was 82.6 percent liquid fluorine and 17.4 percent liquid oxygen.

3. FIRING PROCEDURES

3.1 FLOX System Passivation

All components of the FLOX system were cleaned prior to installation in the test stand. After the FLOX system was assembled it was leak checked, dried and passivated prior to use. Passivation was of the assembled system and was accomplished as follows: the system was purged with hot GN_2 for at least 1 hour and then evacuated by means of a high-capacity vacuum pump for at least 1 hour. The system was then filled with gaseous fluorine at 20 psig (138 kN/m²) for 10 minutes and inspected for hot spots. If no hot spots were found, the system pressure was increased to 100 psig (689 kN/m²) and maintained for at least 2 hours.

The feed system/thrust chamber downstream of the fire valve was passivated using gaseous fluorine exhausted through the injector by actuation of the oxidizer fire valve with low pressure gaseous nitrogen purge initiated on the fuel side.

3.2 Operating Procedure

All tests were conducted using automatic timers for all functions requiring precise timing. All remotely controlled valves, such as stand valves, purge and vent valves were operated from this console.

Prior to each test, the FLOX system was chilled by flowing LN₂ through the jacketed lines up to the oxidizer fire valve. System chilldown normally began 60 minutes prior to the test. Immediately prior to engine firing the inlet lines downstream of the oxidizer fire valve and the injector were chilled by flowing LN₂ through the injector assembly. The fuel side of the injector was maintained at near temperature flowing GN₂ through the injector. Following system and injector chilldown, the following sequence was performed:

- 1) LN₂ and GN₂ flow through the injector was stopped
- 2) Oxidizer injector purges actuated on then off
- 3) Console arm switch activated
- 4) Following a 15-second countdown, the fire switch was turned on and automatic timers fired the engine

All tests conducted using the heat sink hardware were nominal 2-second duration and streak tests were of 4 seconds duration. An oxidizer lead at startup and an oxidizer lag at shutdown were sequenced into the procedure. Primary engine performance parameters were recorded on strip charts in the control center. Critical temperatures were monitored on strip charts and direct reading voltmeters for engine redline temperatures. Motion picture coverage of the streak tests, stability tests and extended duration test utilized Bell and Howell cameras. A Fastax camera was used to film the dynamic stability test.

APPENDIX C

PHYSICAL PROPERTY AND THEORETICAL PERFORMANCE SUMMARY

The theoretical performance, combustion temperature and physical properties of the propellants used for this program are presented in this appendix.

1. THEORETICAL PERFORMANCE

Theoretical performance of the liquid FLOX/gaseous methane combination is presented in this section. Theoretical shifting equilibrium characteristic velocity (C^*) is shown in Figure C-1 as a function of mixture ratio and chamber pressure. The theoretical combustion temperature as a function of mixture ratio and chamber pressure are presented in Figure C-2. Optimum sea-level performance is presented in Figure C-3 where sea-level specific impulse as a function of mixture ratio is illustrated.

2. PROPELLANT PROPERTY DATA

The physical properties of the propellants used for this program are tabulated in Tables C-1 and C-2. The density and vapor pressure of FLOX (82.6% F_2) are presented in Figure C-4 and C-5, respectively.

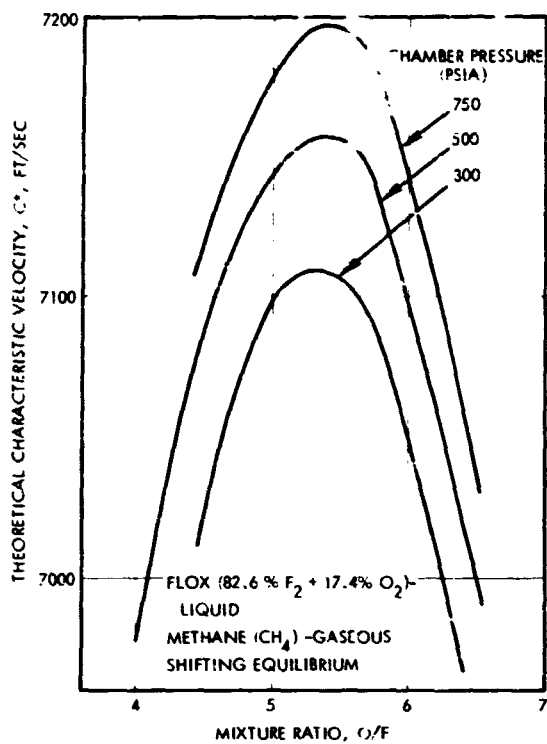


Figure C-1. Theoretical Characteristic Velocity as a Function of Mixture Ratio and Chamber Pressure

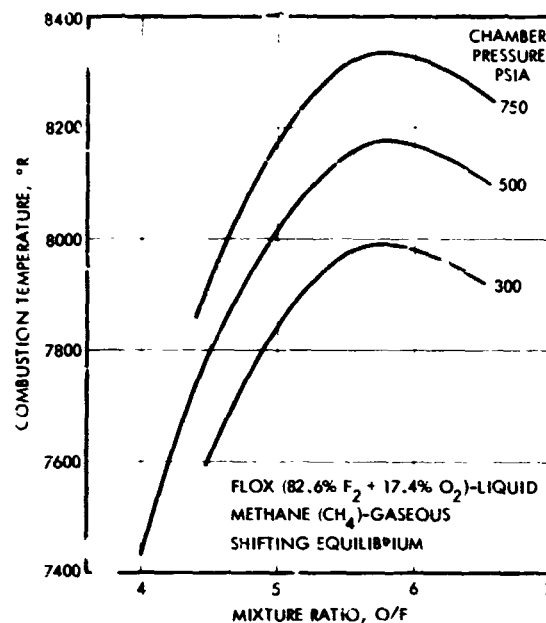


Figure C-2. Theoretical Combustion Temperature as a Function of Mixture Ratio and Chamber Pressure

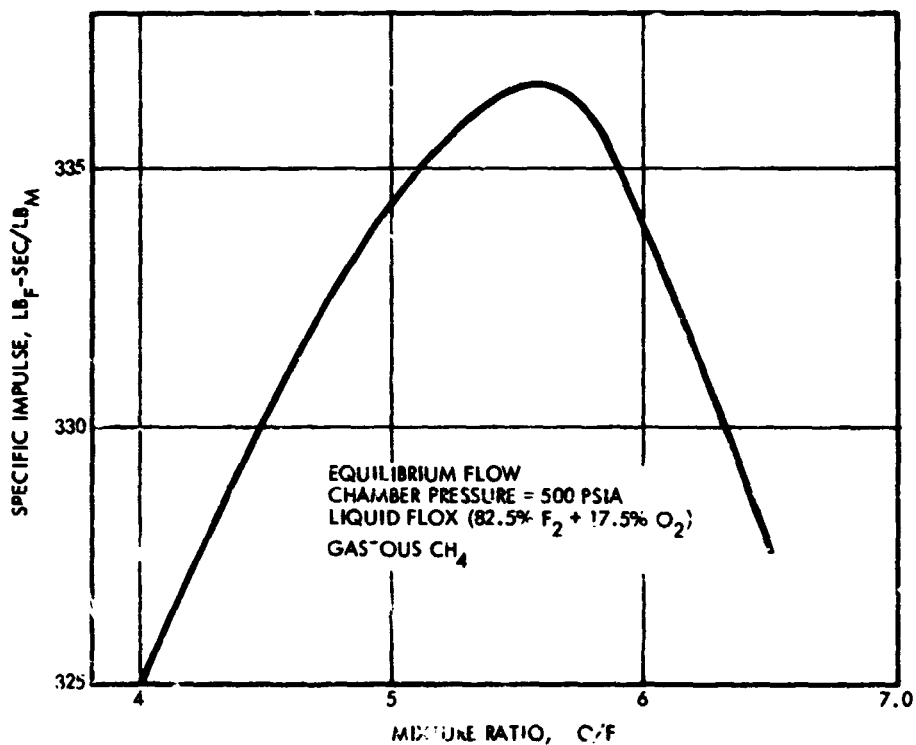


Figure C-3. Optimum Sea Level Expansion Performance for Liquid FLOX/Gaseous CH₄

Table C-1. FLOX Property Summary

Ref: NASA SP-3037

Fluorine and Fluorine-Oxygen Mixtures in Rocket Systems

Chemical Formula	82.6% F ₂ + 17.4 O ₂
Normal Freezing Point, °R	--
Normal Boiling Point, °R	154°R
Liquid Density at NBP, lb/ft ³	91
Critical Temperature, °R	--
Critical Temperature, °R	--
Critical Pressure, psia	--
Critical Volume, ft ³ /lb	--
ΔH (vaporization) at NBP, Btu/lb	74.5
ΔH (fusion), Btu/lb	--
Viscosity at NBP, lb/(sec)(ft ²)	5.0
Thermal Conductivity at NBP, Btu/ft-hr-°R	0.090
Specific Heat at NBP, Btu/lb-°R	0.372
Specific Heat, Gas at 60°F	
C _p , Btu/lb-°R	0.200
C _v , Btu/lb-°R	
Ratio, C _p /C _v	
Viscosity at 60°F, centipoise	0.0216
Viscosity, gas, 32°F, 1 atm, centipoise	0.0203
Gas Constant, R, ft-lb _f /lb _m -°R	40.8

Table C-2. Methane Property Summary

Reference: Pratt Whitney Aircraft PWA FR-1443

Chemical Formula	CH ₄
Molecular Weight	16.042
Normal Freezing Point, °R	163.2
Normal Boiling Point, °F	200.8
Liquid Density at NBP, lb/ft ³	26.48
Critical Temperature, °R	43.3
Critical Pressure, psia	673
Critical Volume, ft ³ /lb	0.0989
ΔH (vaporization) at NBP, Btu/lb	219.22
ΔH (fusion), Btu/lb	25.25
Viscosity at NBP, lb/ft-sec	7.0 x 10 ⁻⁵
Thermal Conductivity at NBP, Btu/ft-hr-°R	0.1075
Specific Heat at NBP, Btu/lb-°R	0.80
Specific Heat, Gas at 60°F	
C _p , Btu/lb-°R	0.5271
C _v , Btu/lb-°R	0.4032
Ratio, C _p /C _v	1.307
Viscosity at 60°F, centipoise	0.012
Viscosity, Gas, 32°R, 1 atm. centipoise	0.009
Gas Constant, R, ft-lbf/lbm-°R	96.31
Gas Density at 32°F, 1 atm	0.045 lb/ft ³
70°F, 1 atm	0.041
70°F, 100 psia	0.285
70°F, 500 psia	1.495
70°F, 1000 psia	3.17
70°F, 2000 psia	6.80*
70°F, 4000 psia	11.77*
70°F, 6000 psia	14.80*

* Calculated from theoretical compressibility curve.

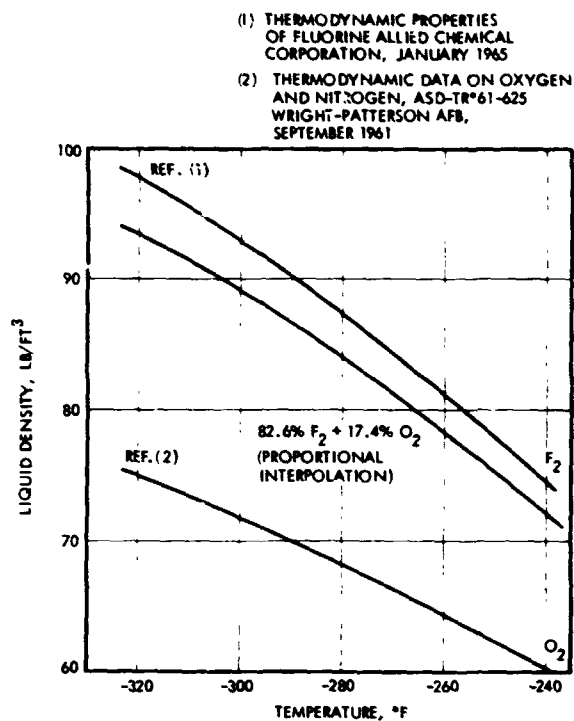


Figure C-4
Liquid Fluorine, Oxygen and
FLOX Density as a Function
of Temperature

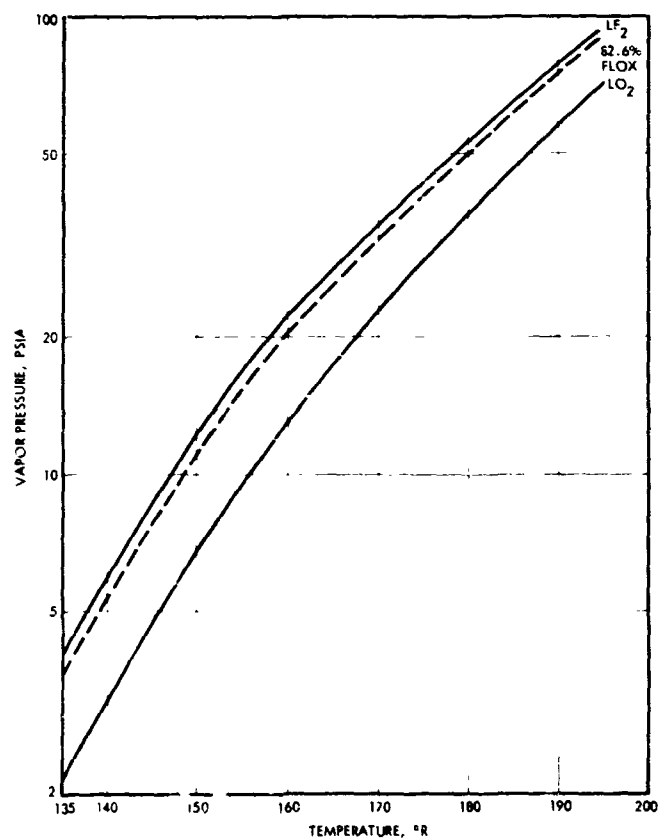


Figure C-5
Vapor Pressure of Liquid
Fluorine, Oxygen and FLOX
as a Function of Temperature