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Performance Comparison of Axisymmetric and Three-Dimensional Hydrogen Film Coolant Injection in a 110N Hydrogen/Oxygen Rocket

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Abstract

An experimental performance comparison of two geometrically different fuel film coolant injection sleeves was conducted on a 110 N gaseous hydrogen/oxygen rocket. One sleeve had slots milled axially down the walls and the other had a smooth surface to give axisymmetric flow. The comparison was made to investigate a conclusion in an earlier study that attributed a performance underprediction to a simplifying modeling assumption of axisymmetric fuel film flow. The smooth sleeve had higher overall performance at one film coolant percentage and approximately the same or slightly better at another. The study showed that the lack of modeling of three-dimensional effects was not the cause of the performance underprediction as speculated in earlier analytical studies.

Introduction

Low thrust propulsion is required on every launch vehicle, satellite, and spacecraft. Low thrust propulsion functions include apogee insertion, attitude control, stationkeeping, rendezvous, docking separation, midcourse correction and planetary retro. Currently, the bulk of low thrust propulsion functions are provided by small chemical rockets with thrust levels, depending on the function, ranging from 450 mN to 4500 N.

Small rocket flowfields differ from those of larger thrust class rockets in several ways. The flows are more strongly influenced by viscous effects because of the relatively small size and corresponding large surface-to-volume ratio in small rockets. Also, a substantial percentage of the fuel is usually required for film cooling of the walls. The large amounts of fuel film cooling and low Reynolds number flow can lead to significant mixing and boundary layer losses in small rockets. Furthermore, in the mixing layer, a secondary combustion zone can exist between the fuel film cooling flow and the core flow. This chemically reacting viscous dominated flowfield has resulted in considerable difficulty in the prediction of performance and thermal behavior in small rockets.[1]

To accurately model the mixing and heat transfer in small rocket flowfields, an effort was undertaken to apply numerical methods developed in the aeronautical community to rocket flowfield modeling. The RPLUS code,[2] originally developed to study supersonic combustion of hydrogen in air for ramjets and scramjets, was adapted to model small rocket flowfields. The RPLUS code numerically solves the coupled set of Navier-Stokes and species transport equations in axisymmetric coordinates, over the entire flowfield.

In the preliminary assessment of RPLUS, two versions[3,4] were developed that compared the measured performance from a 110 N, gaseous hydrogen/gaseous oxygen rocket to the performance calculated by RPLUS. In the code, the rocket geometry was simplified to fit the axisymmetric assumption. The actual rocket utilized a chamber sleeve insert that fit against the injector to divide the hydrogen flow. Part of the hydrogen was diverted into the injector for ignition with the oxygen, while the rest was ducted down milled slots in the chamber sleeve insert for wall film cooling. The exit of the sleeve, then, was composed of a core flow of hydrogen and oxygen combustion products and a coannular flow of hydrogen. By starting at the sleeve exit, the modeling was simplified to that of a single element coaxial injector, with a precombusted oxidizer-rich core, surrounded by an outer annular flow of gaseous hydrogen blanketing the wall.

The trends in performance were correctly predicted by both versions of RPLUS. However, the measured performance values were underpredicted by both versions by three to four percent (figure 1a,b[3]). This result was unexpected, since the analyses assumed equilibrium composition for the core flow and adiabatic walls, both of which have the effect of increasing the calculated performance values. The low predictions of performance seemed to indicate inaccurate modeling of the core flow/film flow mixing. Specifically, it was suspected that there were three possible causes for the underprediction. First, it was thought that the turbulence modeling in the combustion shear layer did not accurately represent the mixing between the core and film flows. However, when RPLUS was run with and without the turbulence modeling, little difference was found in the calculated flowfields. Second, the results of the turbulence modeling, coupled with an estimated shear layer Reynolds number between 400 to 600,[4] indicated an unsteady laminar flow. The Reynolds number was based on the width of the film cooling annulus. A laminar shear layer would provide less mixing than the measured performance numbers suggest. If laminar flow then prevails in the small rocket, the unaccounted for mixing could be due to unsteady effects not well predicted by turbulence models. Finally, it was proposed that the three-dimensional mixing effects at the film coolant injection point not modeled by RPLUS were a possible cause. This report describes an experimental evaluation of the last proposed cause, since the model results are insensitive to large variations in assumed turbulent mixing and unsteady laminar flow phenomena are extremely difficult to evaluated in real rockets.

This study compares the experimental performance of a three-dimensional fuel film injection geometry with an axisymmetric fuel film injection geometry. Performance tests were conducted with the same 110 N, gaseous hydrogen/oxygen rocket used in the previous studies. The slotted sleeve used previously and a second smooth sleeve of equivalent flow area were tested at the same operating conditions. The slotted and smooth sleeves correspond to the three-dimensional and axisymmetric injection of the fuel film coolant, respectively. An attempt was made to operate over the same range of fuel film coolant(FFC) percentages as used in previous studies. However, this could not be done due to problems with the thruster temperatures. This will be discussed below. While no direct performance comparisons could be made to previous studies, the results provide insight into the effect that fuel film injection geometry can have upon performance, and the requirements for accurately modeling low thrust rockets. The purpose of this study was to measure the difference in performance between an axisymmetric and a three-dimensional film coolant injection geometry to determine if the lack of modeling of three-dimensional effects was the cause of the performance underprediction. In the following sections the RPLUS code is discussed first, followed by a description of the thruster and test facility used to test the axisymmetric assumption of the RPLUS model. The test results were then discussed, followed by the concluding remarks.

RPLUS

The RPLUS code was originally developed at NASA Lewis Research Center to analyze supersonic combustion of scramjets and ramjets.[5] It was adapted to analyze subsonic-supersonic flows in rockets,[2,6] and applied to a 1000:1 area ratio nozzle[7] and to the 33:1 area ratio rocket that is the subject of this study.[3] Another version of the code was developed at Pennsylvania State University[4] for the same 33:1 area ratio rocket. That version included a modified turbulence model and a solution algorithm to speed up convergence in the subsonic region.

RPLUS solves the full Navier-Stokes equations and the species equations in a coupled manner, using the lower-upper symmetric successive over-relaxation (LU-SSOR) numerical scheme. The hydrogen-oxygen combustion process is modeled by an 8-species, 18-step finite-rate reaction mechanism. For the rocket used in this study, a grid composed of 202 axial and 60 radial lines was used which was clustered in the region of high gradients in the flow. Convergence was determined as the point when the mass flow rate was conserved to within two percent. A more detailed description of the code can be found in references 2-4.

The plane of the chamber sleeve exit serves as the inflow surface or starting point for the RPLUS calculation. The inflow surface is assumed to be composed of a pre-combusted, oxygen-rich core flow, with a coannular flow of hydrogen. The RPLUS input includes the Mach number, pressure, temperature and species mass fractions of the film and core flow streams. There are assumptions which are known to both over and under predict performance. The accuracies of the predictions were not quantified in the previous study.

The measured chamber pressure is assumed to be the pressure of the core and film flows. The core flow temperature and species mass fractions are found from the Chemical Equilibrium Composition (CEC) computer program[8], using the pressure and mixture ratio of the core as inputs. The use of CEC, which computes the equilibrium composition of combustion products, implies an 100 percent core combustion efficiency. This leads to an overestimation of the inlet enthalpy in the core which should result in an overprediction of performance.

The current version of RPLUS also assumes adiabatic walls. The measured film temperature accounts for the enthalpy that is added to the hydrogen from regenerative heating in the nozzle and combustion chamber, but does not account for heat losses from the thruster. The adiabatic wall assumption should result in higher computed performance values.

The film temperature is assumed to be the average of two thermocouples, located 180 degrees apart, at the sleeve exit (position TC1 on figure 2). The thermocouple tips are located flush with chamber wall. The film is assumed to be 100 percent hydrogen. The input value of the Mach number of the core flow is found from one-dimensional, isentropic relations. The current version of RPLUS assumes the same Mach number for the core and film flows to facilitate the calculation. This assumption could lead to an underestimation of the enthalpy in the sleeve and lower computed performance values.

Apparatus and Procedure

Thruster

The RPLUS code models the regeneratively cooled gaseous hydrogen/gaseous oxygen thruster built by the Gencorp Aerojet Propulsion Division under contract for NASA Lewis Research Center. The thruster is designed to run at a nominal chamber pressure of 517 kPa giving a nominal chamber thrust of 110 N with an overall mixture ratio of 8:1. The thruster has an overall length of 24.8 cm, a combustion chamber diameter of 2.54 cm and a throat diameter of 1.27 cm. The Rao optimized[9] nozzle is bell shaped with a 33:1 area ratio. A schematic of the thruster is shown on figure 3 and a photograph of the injector components is shown in figure 4.

Hydrogen enters the regeneratively cooled nozzle at the exit plane were it flows through the milled passages within the nozzle and chamber walls towards the injector. Within the injector, hydrogen enters a manifold where a flow splitting washer divides it into a core flow and film cooling flow. The slot dimensions in the washer determines the percentage of hydrogen used as fuel film coolant. The core hydrogen is injected radially just downstream of the spark plug tip. All of the oxygen is injected radially through a platelet stack upstream of the spark plug tip. The spark plug excites the oxygen as it flows past the tip where it mixes and ignites with the radially injected hydrogen to form an oxygen rich core flow. The film cooling hydrogen first travels down the sleeve insert between the combustion chamber and the regeneratively cooled wall. The hydrogen is then dumped into the combustion chamber, where it forms a film coolant which mixes with the core flow by shear layer interaction to increase the thruster performance.

The slotted sleeve insert originally designed for the Aerojet thruster, provided one of the test cases for the present study. The second sleeve, referred to as the smooth sleeve, was designed to have an equivalent coolant flow area without the slots, while having the same inside radius so as not to change the combustion chamber size. The motive for having the same flow area, was in theory to have the same film velocities in both cases. Geometries of the two sleeves are shown in figure 5 and both sleeves are pictured in figure 6. The thruster could not be tested at the lower FFC levels used in the previous studies because of higher temperature profiles that resulted in nozzle overheating. An injector modification was completed between the previous studies and this one which was suspected as the cause of the higher temperatures. The injector was modified by milling it out at the base of the spark plug to decrease the chance of the spark arcing at the base instead of the tip of the spark plug; thereby improving the number of successful ignitions.

The injector, nozzle, and combustion chamber are instrumented with static pressure taps and thermocouples. Pressures measured include chamber, hydrogen injection, and oxygen injection pressures. Thermocouples measure both internal and external wall temperatures. Internal wall thermocouple positions are shown in figure 3. The external wall temperatures are not reported in this work. The thruster is described in detail in reference 10.

Test Facility

The tests were conducted at the NASA Lewis Research Center in the low thrust propulsion test facility.[11] The facility was designed as a testbed for low thrust rockets that use gaseous hydrogen and gaseous oxygen for propellants. It became operational in March of 1989 and can be used for a range of testing programs from long duration steady state to short duration cyclic. The facility was designed to test the 22-220 N thrust class rockets. All data are recorded on the stand alone data acquisition system. During testing, real-time calculated performance parameters are available.

The rocket hardware was installed in the cylindrical test tank that maintains a pressure equivalent to an altitude of 36.6 km or a back pressure on the nozzle of 1.4 to 2.1 kPa. The pressure in the tank, is maintained by a two-stage air ejector system. The nozzle axis was oriented horizontally in the thrust stand. The thruster was mounted on flexible plates to insure freedom of movement along the thrust axis. Also, all the pressure and propellant lines were constructed from rigid tubing. The lines were mounted perpendicular to the thrust axis to minimize the effect on the thrust measurement. The plume from the nozzle was fired into a water cooled diffuser. After the diffuser, the exhaust enters a spray cart, where it is cooled by water spray prior to entering the air ejectors. The flow rate of water spray is controlled by the exiting exhaust temperature. The exhaust is finally vented through a pair of mufflers. A more detailed facility description is available in reference 10.

Measurement Uncertainties

Hydrogen and oxygen mass flow rates were calculated using the measured inlet pressures, the measured inlet temperatures and the discharge coefficient of the critical flow venturis with corrections for real gas effects.[12] Thrust was measured with a 440 N thrust measurement system. Thrust calibrations were performed in-situ, at altitude conditions and with pressurized propellant lines.

The measurement uncertainties in the performance values are determined using the JANNAF recommended procedure.[13] With each measured quantity (pressure, temperature, etc.), there is a random precision error and a bias error associated with the calibrations of the measuring instrument and with the data acquisition of the measurement. The uncertainty of a performance parameters, vacuum specific impulse, characteristic velocity and thrust coefficient, are a combination of the precision and bias errors of the measured quantities, propagated to the performance values. The largest contributors to the precision errors in this test series were the venturi inlet pressure and venturi discharge coefficient calibration errors used in the mass flowrate calculations. A zero drift in the thrust measurement load cell (probably due to bending of the thruster flange under thermal loading) caused some error is thrust measurements. This was reflected as a bias in the thrust measurement, calculated as the difference between the pretest and posttest zero readings of the thrust.

The uncertainty of the vacuum specific impulse values ranges from 1.7 percent to 2.5 percent in the positive direction (the direction of thrust bias) and typically 1.7 percent in the negative direction. The uncertainty of the characteristic velocity values are typically +/- 1.7 percent. The uncertainty for the thrust coefficient is 1.5 percent or less in positive direction, and 0.8 percent or less in the negative direction.

Test Program

During testing the chamber pressures were held at 535 +/- 7 kPa, and the hydrogen mass flow rates were approximately the same, for both the slotted and smooth sleeves. Two fuel film cooling washers, 74.7 and 79.1%, were used over a mixture ratio range of 3 to 5 for both sleeves. The test runs were conducted on consecutive test days. Changing the sleeve and %FFC washer required that only the injector be removed from the nozzle/chamber portion, which remained in the thrust stand. Tank pressures were generally between 1.4 and 1.8 kPa at the start of a test, then dropped to 1.3 kPa once the ignition and flow through the diffuser were established.

Results and Discussion

The performance test data are presented sequentially in Table 1. The characteristic velocity(often referred to as C*), specific impulse, and thrust coefficient for the 74.7 and 79.1% FFC for both smooth and slotted sleeves are plotted in figures 7-10. Multiple tests were run at each mixture ratio and the data were found to be repeatable.

Figures 7 and 8 show that the thrust coefficients are the same for the slotted and smooth sleeves for the 74.7% and 79.1% fuel film cooling cases. Since the nozzle performed the same over the different test cases, specific impulse and characteristic velocity can then be discussed interchangeably. These performance parameters are plotted in figures 9a and 9b for the 74.7% FFC case, and 10a and 10b for the 79.1% FFC case. Both performance parameter curves show the same trends at a given FFC percentage. The specific impulse and characteristic velocity graphs (figures 9a and 9b) show that the smooth sleeve had better performance than the slotted sleeve for the 74.7% FFC. At a mixture ratio of four the performance is greater by only two percent, but at a mixture ratio of five the performance was seven percent greater. The seven percent is well above the uncertainties for this experiment. For the 79.1% FFC case (figures 10a and 10b), the performance difference between smooth and slotted sleeves is within the range of the experimental uncertainty. At a mixture ratio of three, the slotted sleeve performs better by approximately two percent. At a mixture ratio of four and five, the smooth sleeve performs better by four and two percent, respectively. The results of this performance study show that the thruster with the three-dimen-sional sleeve did not have the greater measured performance. Therefore the underprediction of performance by the axisymmetric RPLUS code is not due to the axisymmetric "smooth" assumption for the FFC percentages tested.

Thermocouple measurements obtained inside the combustion chamber at the locations shown in figure 3 are shown in figures 11 and 12. For the 74.7% FFC case, the slotted sleeve is hotter at mixture ratios of four and five than the smooth sleeve. At a mixture ratio of three the temperature profiles are almost the same. For the FFC case of 79.1%, the slotted sleeve has higher temperatures only for a mixture ratio of five. At the mixture ratio of four the smooth sleeve experiences higher temperatures and, at a mixture ratio of three the profiles are almost the same. Temperature results, at the TC1 position, are used to calculate the hydrogen film coolant velocity to core velocity ratios, shown in Figures 13 and 14. Note that the velocity ratios are near unity and that they are generally higher for the higher film coolant percentages. Also, except for the mixture ratio of four, the smooth wall generally ran cooler than the slotted wall sleeve, making it difficult to correlate the temperature profiles and the performance.

The axisymmetric sleeve was built to match the film flow area of the one modeled in previous analytical studies. The intent was to compare two injection geometries of the same flow area. However, the results indicate that the edge thickness of the sleeves may play a more significant role in the mixing than the injector geometry. As shown in figure 6, in order to maintain the same flow area for both sleeves the edge or splitter plate thicknesses had to be different. At the sleeve edge or flow splitter where the film coolant flow and core flow meet, the slotted sleeve was relatively thin, i.e. 0.076 cm, in comparison to the smooth wall sleeve with an edge thickness of 0.132 cm. The thicker edge of the smooth sleeve may be creating larger recirculations zones between the film coolant and core flows, which enhances mixing and thus performance. The significance of this geometry change is shown in the ratio of the edge thickness to film coolant gap. This ratio is much greater for the smooth sleeve compared to the slotted sleeve, 1.21 and 0.46, respectively, allowing the unsteady flow or turbulence to interact with more of the film coolant. The effect of the taper on the end of the splines of the slotted sleeve is uncertain, because it should result in a thinning and slowing down of the film at the edge of the sleeve. This effect could be studied by tapering a smooth sleeve to the same dimension as a slotted sleeve, and conduct a performance study on both sleeves. In this way perhaps some of the difference in performance between the smooth and slotted sleeves could be explained.

Concluding Remarks

An experimental performance comparison was conducted on a 110 N(25 lbf) gaseous hydrogen oxygen rocket to establish the effect of hydrogen film coolant sleeve design on rocket performance. One sleeve had milled slots running axially down the surface, the other was designed with a smooth surface. Both were designed to give the same flow area and therefore, the same film velocities.

Previous work at Pennsylvania State University[4] compared the experimental performance characteristics of the same slotted sleeve and thruster combination with the analytical results obtained from the RPLUS code. In the code the film coolant sleeve was modeled as a smooth sleeve to simplify calculations. The RPLUS results underpredicted the experimental performance results by three to four percent. In the study, it was suggested that the three-dimensional effects of the slotted sleeve increased the mixing, which was not accounted for in the axisymmetric model.

Test were conducted over a mixture ratio range of 3 to 5 at FFC values of 74.7% and 79.1% to determine if the slotted sleeve was actually responsible for the increased performance observed in the model. Results showed that the smooth sleeve configuration had greater characteristic velocity and specific impulse, for the 74.7% FFC case and approximately the same or slightly better performance for the 79.1% FFC case. These results showed the three-dimensional slots were not responsible for the underprediction by the RPLUS code. Because of the difference in geometries between sleeves, the smooth sleeve having the thicker edge at the core and film coolant flow interface may enhance the mixing between the two flows, thus, improving the performance.

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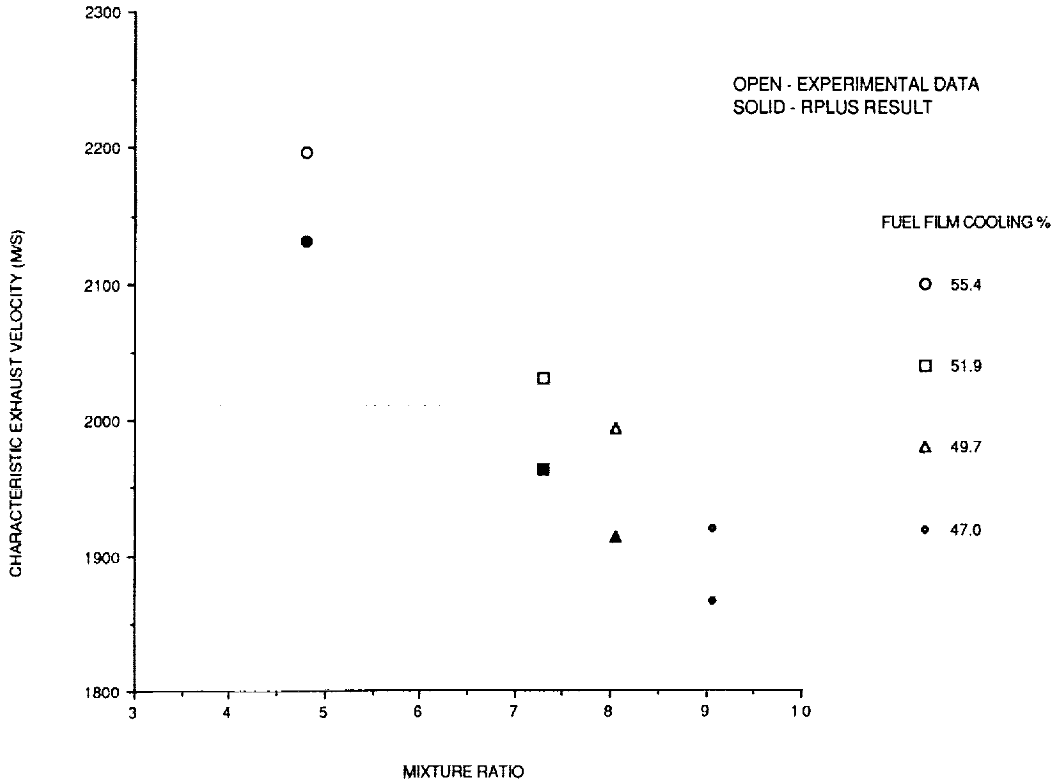

Figure 1a: Experimental and RPLUS Comparison of Characteristic Exhaust Velocity vs. Mixture Ratio

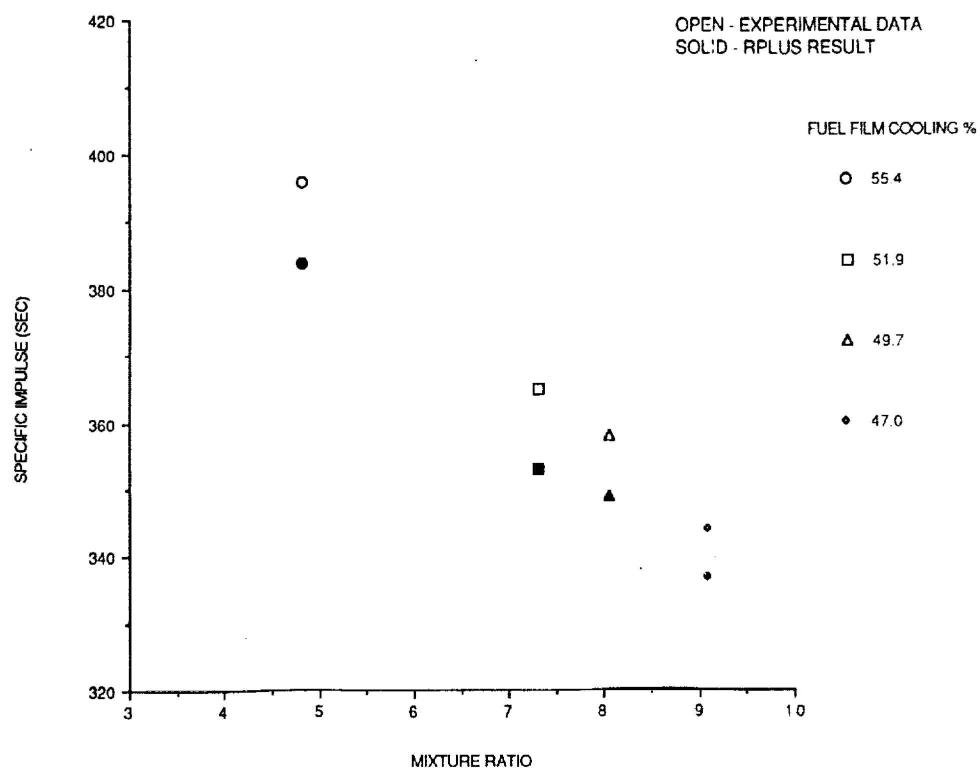

Figure 1b: Experimental and RPLUS Comparison of Specific Impulse vs. Mixture Ratio

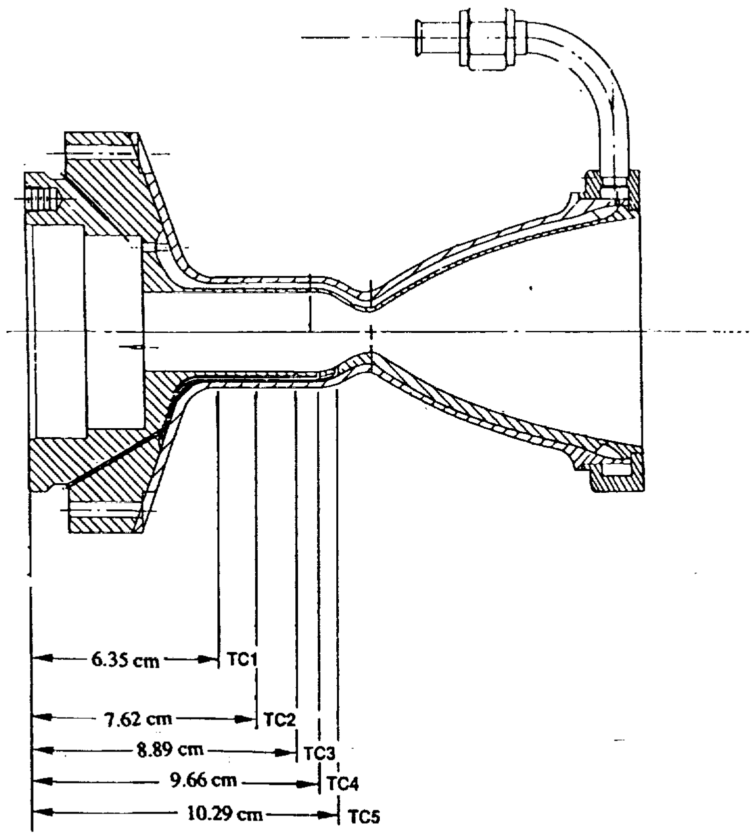

Figure 2. Axial Positions of Internal Thermocouples.

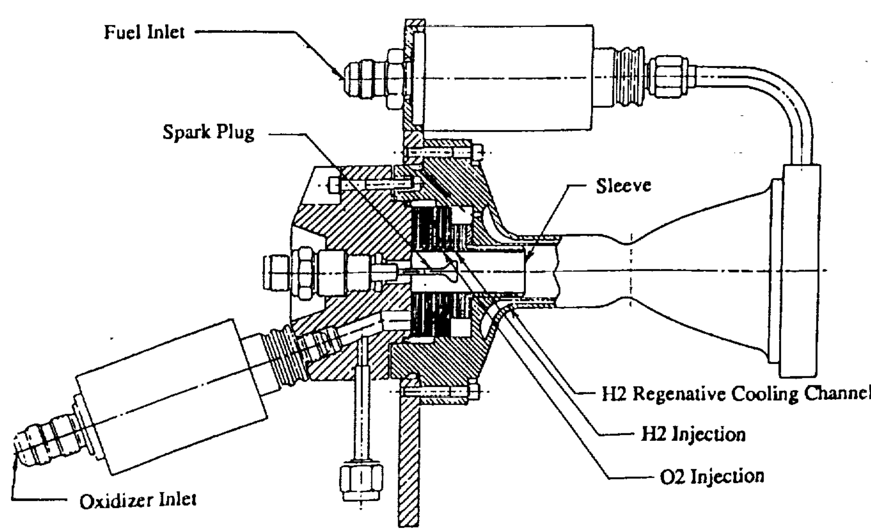

Figure 3. Aerojet 110 N(25 lbf) Gaseous H_2/O_2 Thruster.

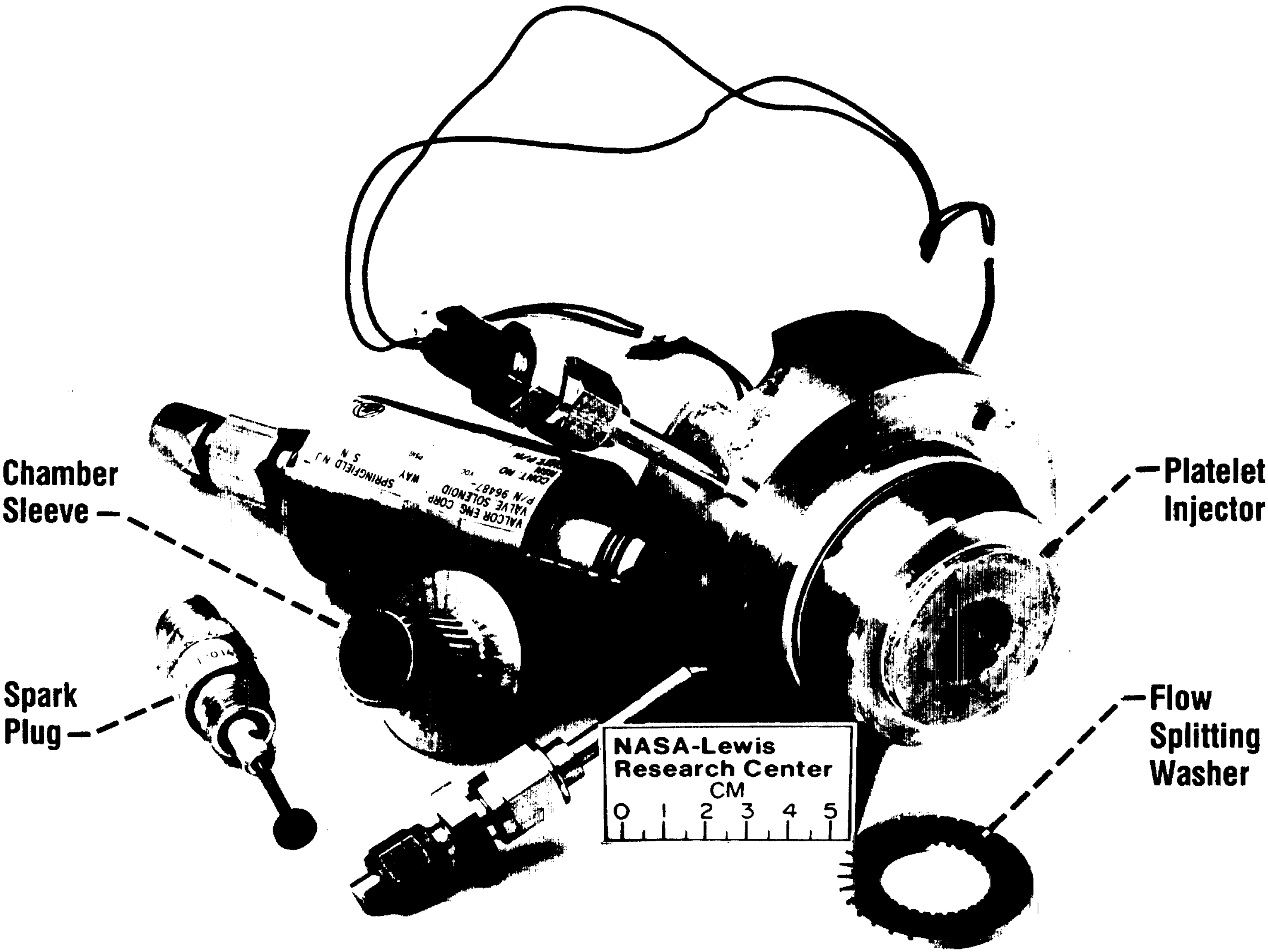

Figure 4. Injector Components of Aerojet Thruster.

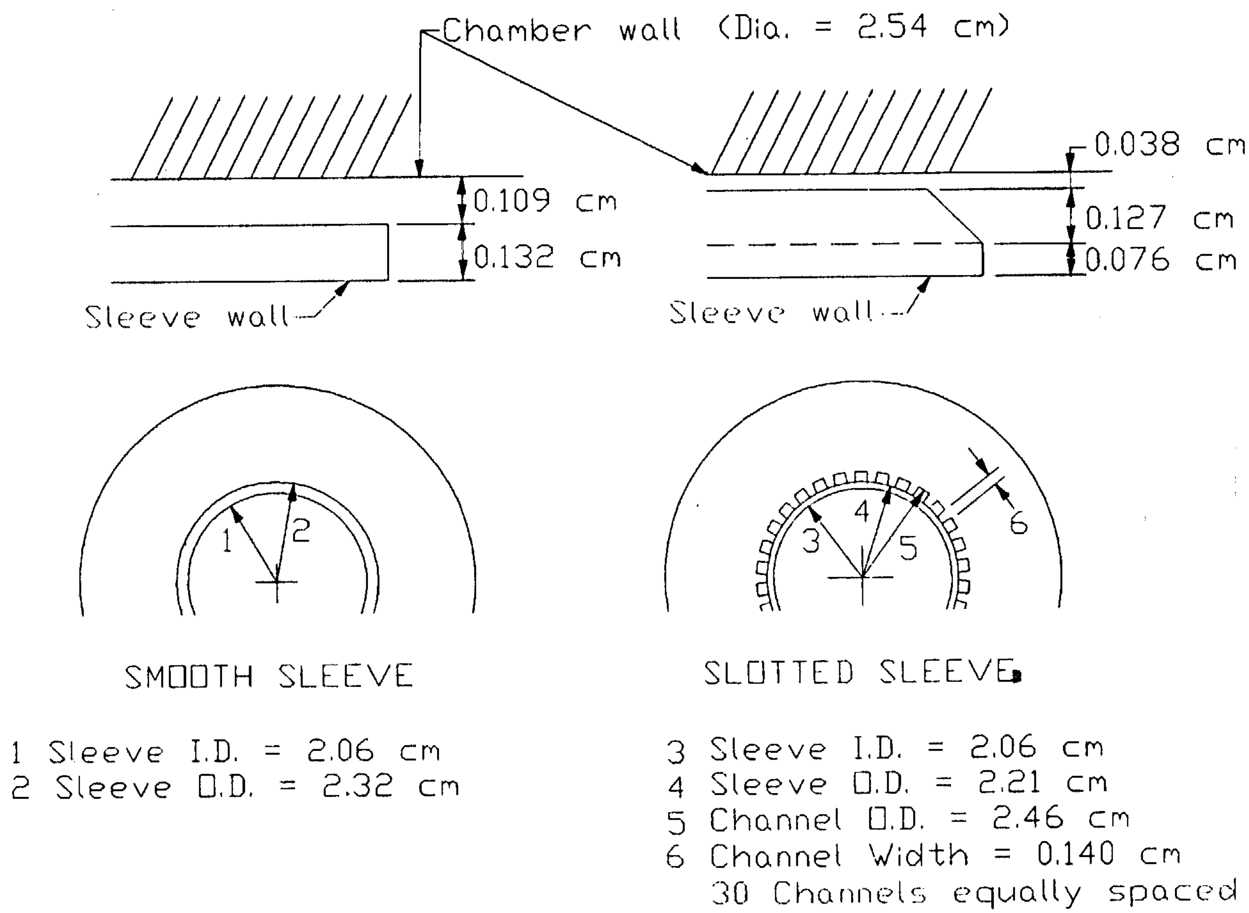

Figure 5. Dimensional Comparison of Smooth and Slotted Sleeves.

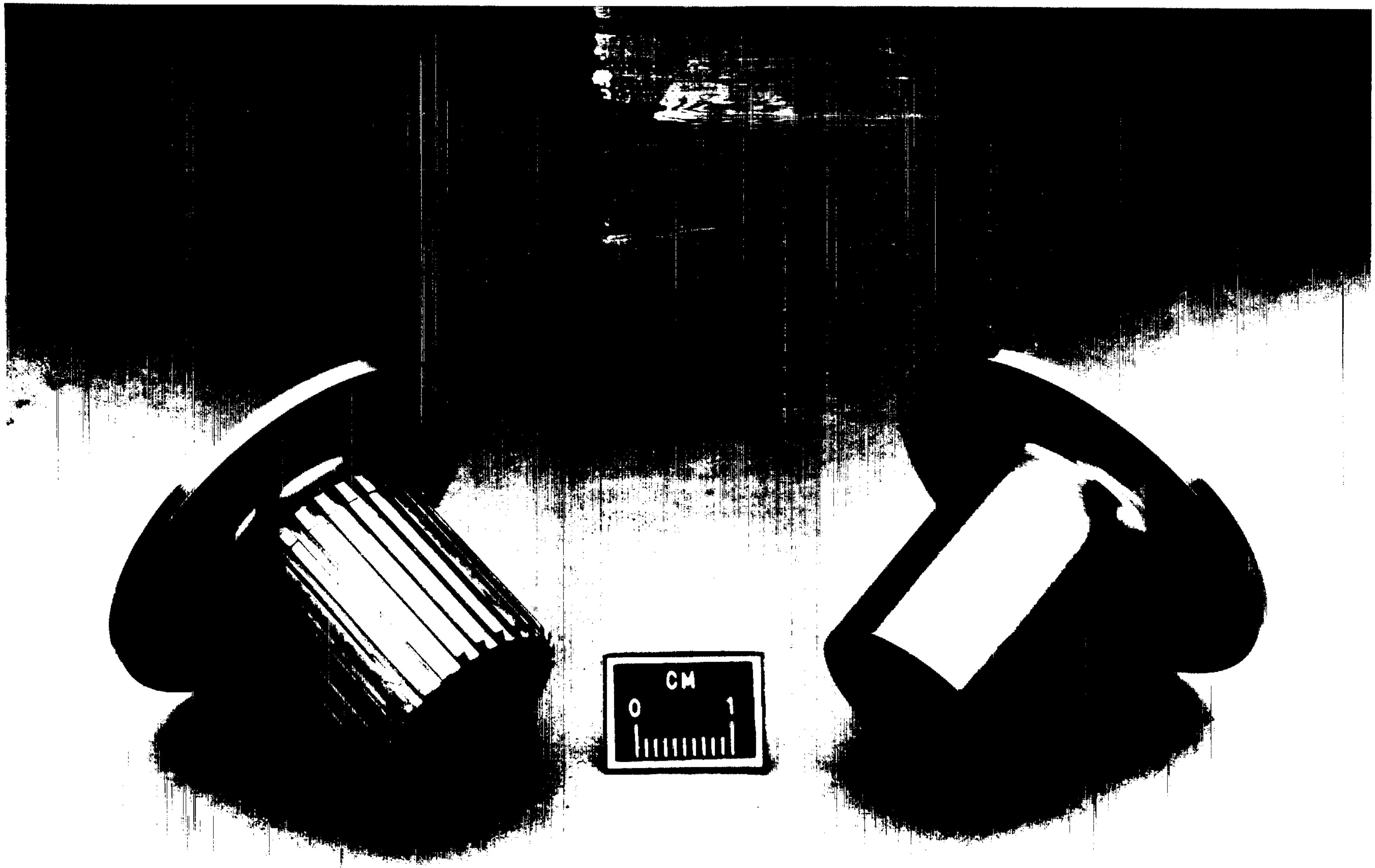

Figure 6. Slotted and Smooth Sleeves.

Table 1. Experimental Test Data

Test Run #	Mixture Ratio	Specific Impulse (sec)	Characteristic Exhaust Velocity (m/sec)	Thrust Coefficient	FFC (%)	Sleeve Configuration
4	3.33	380.9	2086	1.79	74.7	slotted
5	4.13	377.6	2117	1.75	74.7	slotted
7	5.26	359.5	2008	1.76	74.7	slotted
8	5.15	364.7	2017	1.77	74.7	slotted
12	4.15	380.3	2126	1.76	74.7	slotted
15	3.24	383.1	2108	1.78	74.7	slotted
16	4.20	378.5	2108	1.76	74.7	slotted
17	5.28	360.5	2000	1.77	74.7	slotted
18	5.30	362.1	2001	1.78	74.7	slotted
21	4.14	380.6	2104	1.78	74.7	slotted
22	4.17	379.7	2120	1.76	74.7	slotted
24	3.31	383.5	2123	1.77	74.7	slotted
25	3.28	381.3	2123	1.76	74.7	slotted
26	4.15	379.5	2101	1.77	74.7	slotted
27	5.33	361.3	2019	1.76	74.7	slotted
30	3.30	382.8	2114	1.78	74.7	slotted
32	3.23	403.1	2231	1.77	74.7	smooth
33	4.16	386.1	2138	1.77	74.7	smooth
34	4.19	387.1	2153	1.76	74.7	smooth
35	4.24	387.8	2162	1.76	74.7	smooth
36	4.15	386.5	2169	1.75	74.7	smooth
37	4.18	388.1	2159	1.76	74.7	smooth
41	3.18	404.9	2232	1.78	74.7	smooth
42	3.22	405.5	2239	1.78	74.7	smooth
44	3.25	402.7	2222	1.78	74.7	smooth
47	3.25	401.6	2235	1.76	74.7	smooth
48	5.19	391.8	2162	1.78	74.7	smooth
49	5.21	391.1	2156	1.78	74.7	smooth
50	5.25	393.1	2162	1.78	74.7	smooth
51	5.23	388.5	2147	1.78	74.7	smooth
55	4.17	376.7	2105	1.76	79.1	slotted
56	4.12	376.3	2093	1.76	79.1	slotted
57	4.13	376.5	2090	1.77	79.1	slotted
58	4.14	376.5	2087	1.77	79.1	slotted
59	4.13	378.0	2090	1.77	79.1	slotted
60	3.08	385.9	2150	1.76	79.1	slotted
61	3.05	383.4	2137	1.76	79.1	slotted
63	3.09	382.4	2131	1.76	79.1	slotted
64	3.09	382.8	2119	1.77	79.1	slotted
66	3.07	384.2	2128	1.77	79.1	slotted
67	3.07	381.9	2128	1.76	79.1	slotted
68	5.16	361.3	1998	1.77	79.1	slotted
69	5.18	361.9	2016	1.76	79.1	slotted
70	5.21	358.5	2005	1.75	79.1	slotted
71	5.21	359.6	2005	1.76	79.1	slotted
72	5.22	359.6	2003	1.76	79.1	slotted
75	4.06	388.5	2169	1.76	79.1	smooth
76	4.01	392.3	2169	1.78	79.1	smooth
77	4.06	388.8	2178	1.75	79.1	smooth
78	4.04	389.1	2172	1.76	79.1	smooth
79	5.06	367.0	2040	1.77	79.1	smooth
80	5.11	365.1	2014	1.78	79.1	smooth
81	5.12	373.8	2035	1.80	79.1	smooth
82	5.12	373.5	2035	1.80	79.1	smooth
84	5.15	370.1	2033	1.79	79.1	smooth
87	3.22	378.9	2117	1.76	79.1	smooth
88	3.27	379.4	2078	1.79	79.1	smooth
89	3.27	378.8	2078	1.79	79.1	smooth
90	3.28	376.7	2073	1.78	79.1	smooth
91	3.26	373.7	2070	1.77	79.1	smooth

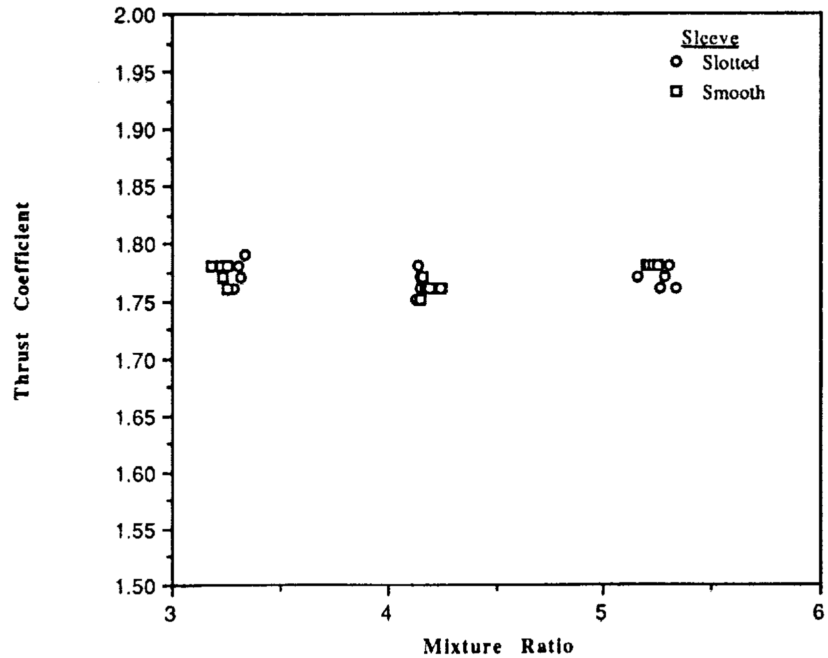

Figure 7. Comparison of Thrust Coefficient Versus Mixture Ratio for 74.7% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

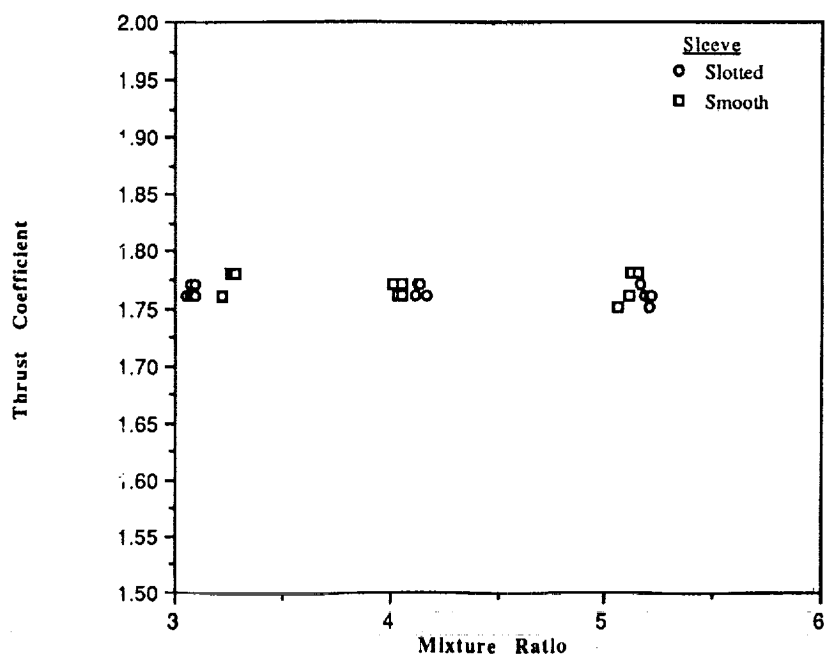

Figure 8. Comparison of Thrust Coefficient Versus Mixture Ratio for 79.1% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

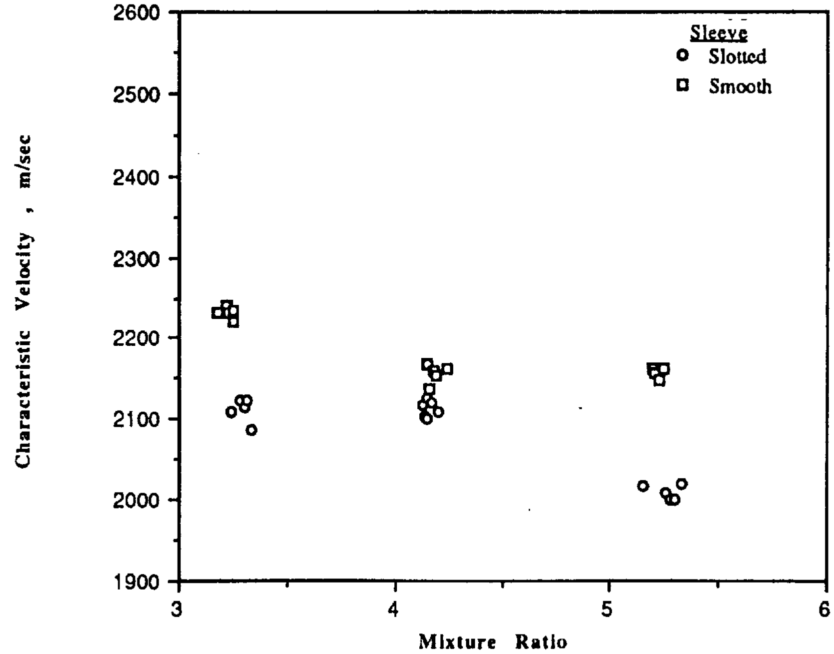

Figure 9a. Comparison of Characteristic Velocity Versus Mixture Ratio for 74.7% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

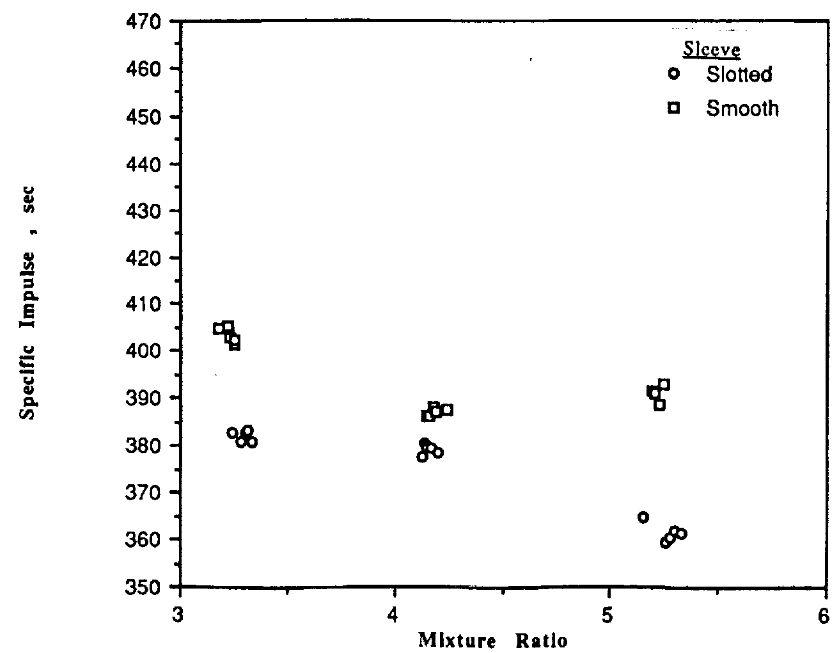

Figure 9b. Comparison of Specific Impulse Versus Mixture Ratio for 74.7% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

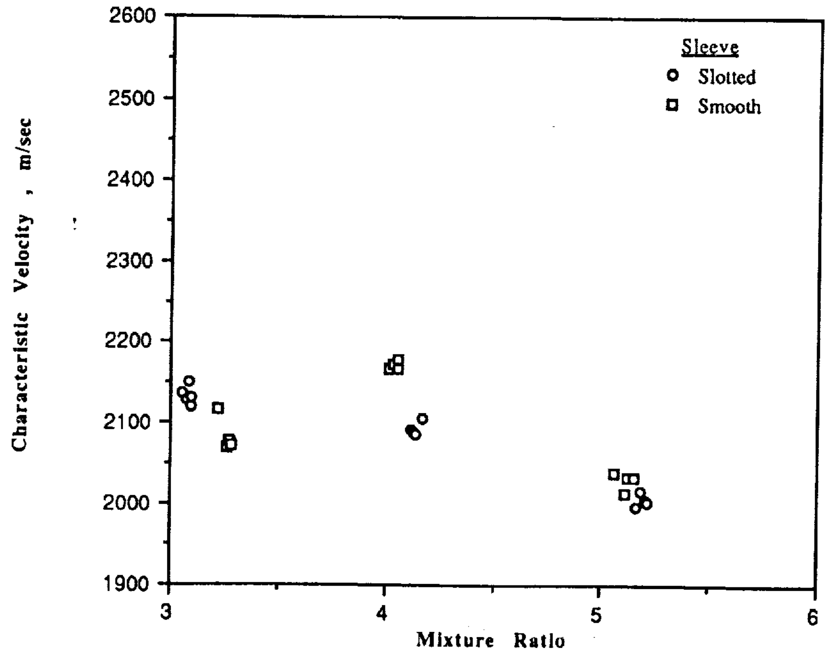

Figure 10a. Comparison of Characteristic Velocity Versus Mixture Ratio for 79.1% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

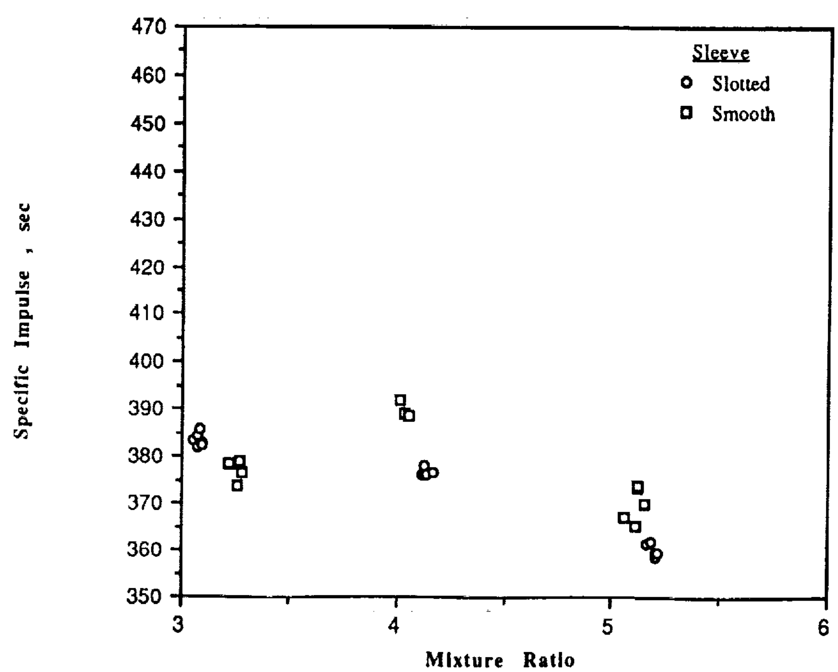

Figure 10b. Comparison of Specific Impulse Versus Mixture Ratio for 79.1% FFC at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

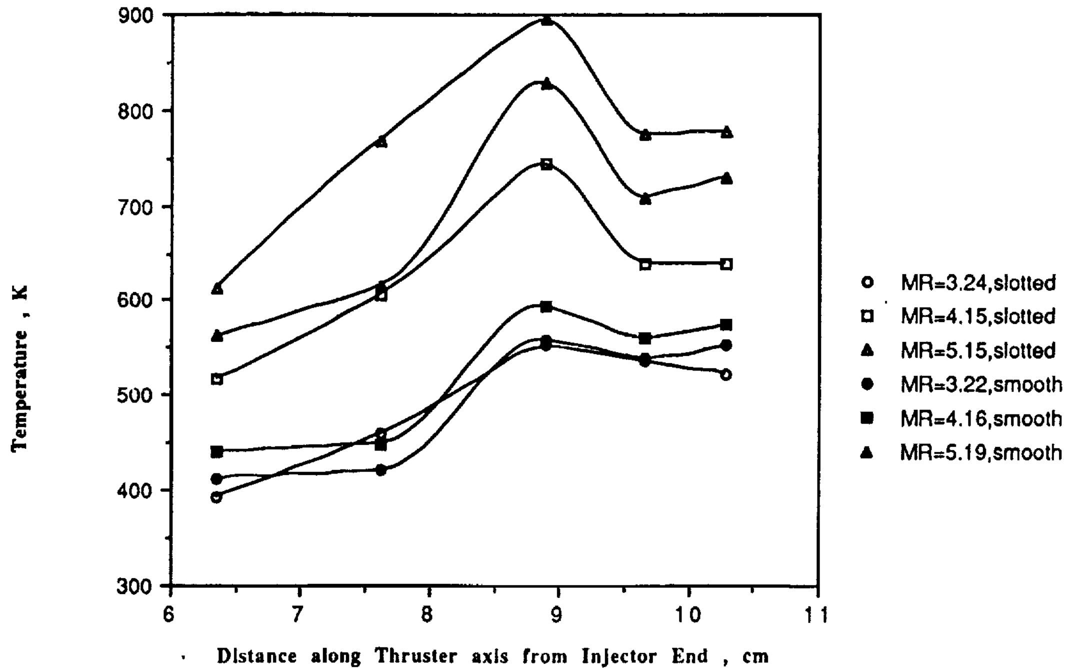

Figure 11. Internal Temperatures versus Axial Position from Injector End for 74.7% FFC.

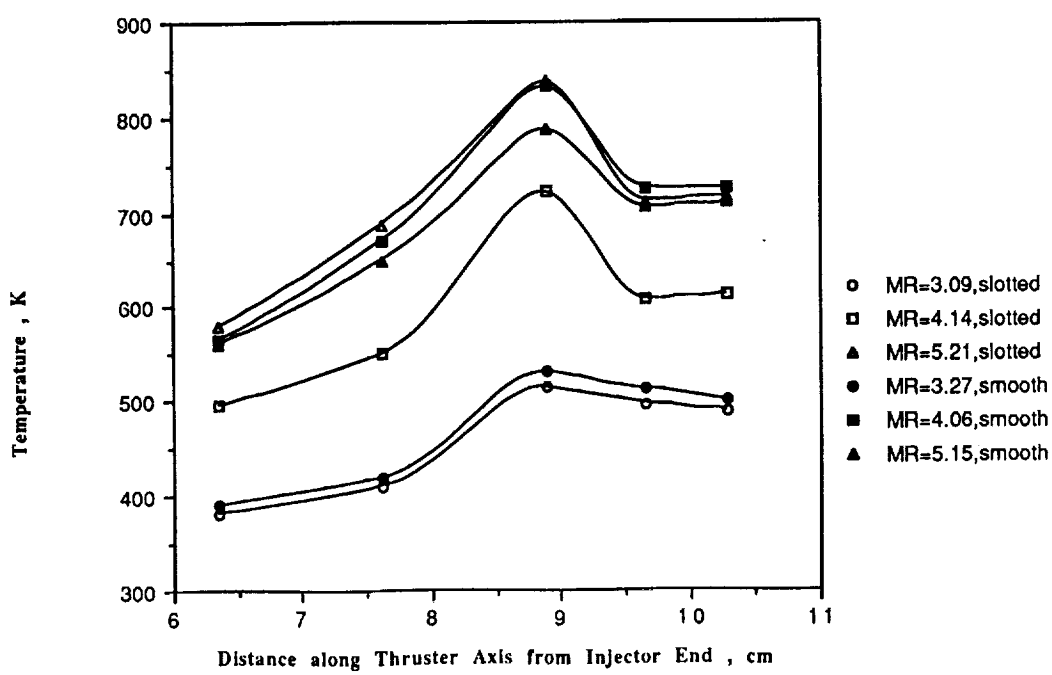

Figure 12. Internal Temperatures versus Axial Distance from Injector End for 79.1% FFC.

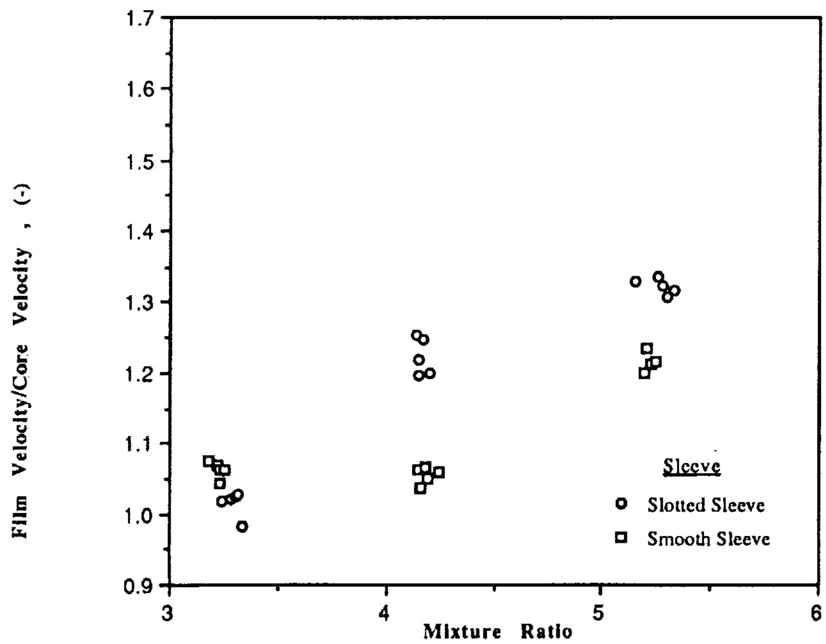

Figure 13. The Ratio of Film Velocity to Core Velocity versus Mixture Ratio for 74.4% Film Flow Cooling at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

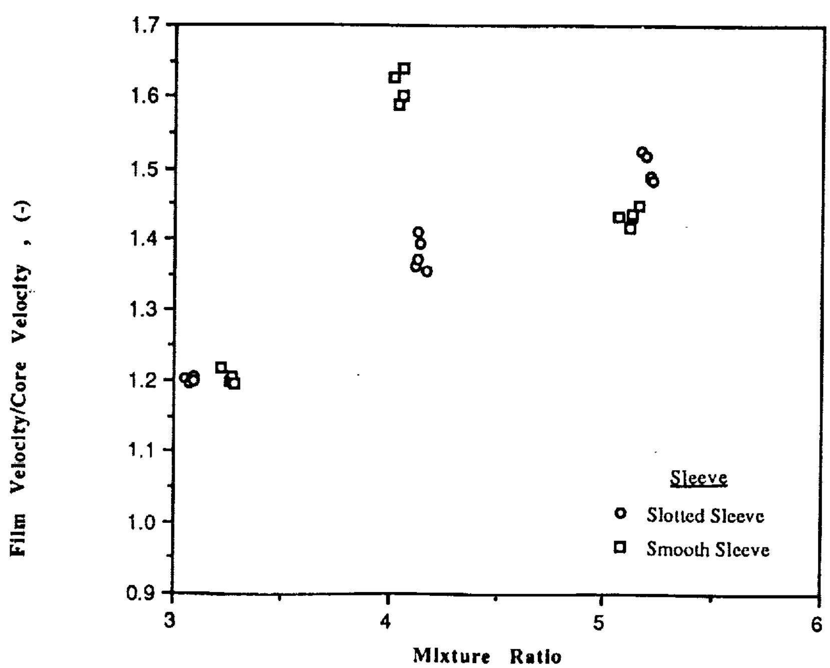

Figure 14. The Ratio of Film Velocity to Core Velocity versus the Mixture Ratio for 79.1% Film Cooling at a Chamber Pressure of 535 +/- 7 kpa (77 +/- 1 psia).

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13. ABSTRACT ***(Maximum 200 words)***

An experimental performance comparison of two geometrically different fuel film coolant injection sleeves was conducted on a 110N gaseous hydrogen/oxygen rocket. One sleeve had slots milled axially down the walls and the other had a smooth surface to give axisymmetric flow. The comparison was made to investigate a conclusion in an earlier study that attributed a performance underprediction to a symplifying modeling assumption of axisymmetric fuel film flow. The smooth sleeve had higher overall performance at one film coolant percentage and approximately the same or slightly better at another. The study showed that the lack of modeling of three-dimensional effects was not the cause of the performance underprediction as speculated in earlier analytical studies.

14. SUBJECT TERMS
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