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Experimental and Analytical Comparison of Flowfields in a 110 N (25 Lbf) H_2/O_2 Rocket

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EXPERIMENTAL AND ANALYTICAL COMPARISON OF FLOWFIELDS IN A 110 N (25 lbf) H_2/O_2 ROCKET

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Abstract

Analytical and experimental studies of a 110 N (25 lbf) gaseous hydrogen/gaseous oxygen rocket were conducted. The presence of thick, chemically reacting shear and boundary layers in these small rockets can be a source of performance losses and considerable difficulty in prediction of performance and thermal behavior. The RPLUS code, which has been developed to model supersonic combustion of hydrogen in air, was modified to model combustion in small rockets, and used to perform the parametric analyses. The code models the full Navier-Stokes equations and species transport equations in a coupled manner. Performance tests were conducted on the rocket in an altitude test facility. The parametric analyses, which were preliminary, were done for a range of mixture ratios and fuel film cooling percentages. The values of specific impulse and characteristic exhaust velocity computed by the code followed the trend of experimental data. However, the computed specific impulse and characteristic exhaust velocity values were consistently lower than the comparable test values by about two to three percent and three to four percent, respectively. Computed thrust coefficient values were within two percent of experimental data. The results of this preliminary study were of value in indicating the areas of the numerical modeling to be explored further.

Introduction

Low-thrust propulsion, in one form or another, is required on every launch vehicle, satellite, and spacecraft. Attitude control and orientation, stationkeeping, apogee insertion, rendezvous, docking, separation, planetary delta V, and planetary retro are functions utilizing low-thrust propulsion. The bulk of low-thrust propulsion has been carried out with small chemical rockets (or thrusters) with thrust levels ranging from 450 mN

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(0.1 lbf) to 4500 N (1000 lbf), depending on the application. Monopropellant hydrazine and Earth storable bipropellants have dominated the low-thrust propulsion field, but hydrogen/oxygen propellants are now receiving consideration for the space station[1], lunar/Mars spacecraft[2], and the auxiliary propulsion systems of the next generation of manned Earth-to-Orbit vehicles[3].

The low Reynolds number flowfields of small rockets differ from those of medium to launch class rockets, in that the flows are more strongly influenced by viscous effects. Compared to the thin boundary layers in medium to launch class rockets, the boundary layers in small rockets are relatively thick. Because of their relatively small size and a corresponding large surface-to-volume ratio, a substantial percentage of the fuel is usually required in small rockets for film cooling. The film reacts with the core flow (which is usually oxygen rich) through turbulent transport of gases across a shear layer, creating a secondary combustion zone. The presence of thick, chemically reacting shear layers can lead to significant performance losses and considerable difficulty in prediction of performance and thermal behavior. Modeling of small rockets for space station[4] and a 20 N (5 lbf) monopropellant hydrazine thruster[5] revealed this difficulty.

There have been efforts to model nozzle flows with thick boundary layers, using the Navier-Stokes equations. As an example, a recent study[6] used the parabolized Navier-Stokes equations with finite-rate chemistry to model the flowfields of the 20 N (5 lbf), monopropellant hydrazine thruster from reference 5, for estimated Reynolds numbers (based on throat radius) from 10,000 to 40,000.

This paper addresses preliminary efforts to use the full Navier-Stokes equations with finite-rate reaction kinetics to model the chemically reacting, viscous flow of a 110 N (25 lbf), gaseous hydrogen/gaseous oxygen thruster. This thruster has an estimated Reynolds number (based on throat radius) of about 30,000 at design conditions. The RPLUS code[7], originally developed to study supersonic combustion of hydrogen in air for ramjets and scramjets, is used for the present study. The RPLUS code numerically solves the coupled set of Navier-Stokes and species transport equations in axisymmetric coordinates, in the entire flowfield. The code is being developed as an analytic tool for small chemical rockets, with the eventual goal of serving as a design tool to reduce empiricism in the rocket design process.

The thruster analyzed in this study was specifically selected to simplify the modeling to that of an axisymmetric flowfield composed of a precombusted, oxidizer-rich core surrounded by an outer annular flow of gaseous hydrogen blanketing the wall. Performance testing of the rocket was also done over a range of mixture ratios and fuel film cooling percentages in an altitude facility. This paper discusses the preliminary parametric analyses from the RPLUS code and compares the numerical results with the test data.

Thruster Description

A cross-sectional view of the thruster, along with the contour coordinates is shown in Figure 1. A detailed description of the thruster is given in Reference 8. The thruster used in this study was designed and fabricated by Gencorp Aerojet Propulsion Division under contract to NASA Lewis Research Center[8]. The thruster was designed for space station propulsion. It operates on gaseous hydrogen and gaseous oxygen and is regeneratively cooled by the hydrogen. The thruster has a design point chamber pressure of 517 kPa (75 psia), an overall oxidizer-to-fuel mixture ratio of 8:1, a fuel film cooling percentage of 60, and a nominal thrust level of 110 N (25 lbf). The chamber liner is fabricated from a copper-zirconium alloy and the outer jacket of the thruster from electroformed nickel. The thruster has an overall length of 24.8 cm (9.75 in), a combustion chamber diameter of 2.54 cm (1.00 in) and a throat diameter of 1.27 cm (0.50 in). The nozzle is bell shaped with an area ratio of 33.4:1. Pressure in the combustion chamber was measured upstream of the chamber sleeve, through a port in the center of the platelet stack. The thruster is instrumented with thermocouples, both on the outer and inner combustion chamber wall.

The flow paths of oxygen and hydrogen in the thruster can be traced by referring to Figure 1. The oxygen flows through a platelet injector stack and is injected radially around the spark plug. The hydrogen first flows through cooling passages in the nozzle wall to a manifold, is then partially distributed through the platelet injector and radially injected just downstream of the spark plug tip. The remaining hydrogen flows through milled slots of the chamber sleeve to provide film cooling for the thruster wall (the percentage of film cooling is varied by changing the flow splitting washers). At the plane of the chamber sleeve exit (X = 0 in Figure 1), the flow is comprised of the combustion products of an oxygen-rich hydrogen/oxygen reaction in the core surrounded by a pure hydrogen cooling film. Figure 2 shows the geometry of the chamber sleeve. To simplify the modeling to axial symmetry, the channeled area was converted to an equivalent geometric annular area.

Test Facility

Testing of the Aerojet thruster was performed in the RL-11 test facility at NASA Lewis Research Center. The RL-11 facility has the capability to test low thrust, gaseous hydrogen/gaseous oxygen rockets with altitude simulation to 1.4 kPa (0.2 psia).

Altitude is simulated in a 0.9 m (3 ft) diameter and 1.8 m (6 ft) long tank using a two-stage, air-driven ejector system. During a test, the thruster fires into a water-cooled diffuser. The exhaust is cooled by a water spray further downstream. Test measurements are displayed in real time on a digital data acquisition system and recorded on floppy disks, stripcharts, and FM tape. Figure 3 shows a diagram of the test rig. A more detailed description of the RL-11 facility is given in Reference 9.

Test Data

Tests consisted of thirty-second and sixty-second duration runs over a range of mixture ratio from 4 to 8, at 49.1 %, 60.9 %, and 69.4 % fuel film cooling. Combustion chamber pressure was nominally at 517 kPa (75 psia), but ranged from 510 kPa (74 psia) to 560 kPa (81 psia).

Hydrogen and oxygen mass flowrates were measured using critical flow venturis. A zero drift in the thrust measurement load cell (probably due to distortion of the thruster flange under thermal loading) caused some distortion in thrust measurements. To compensate for this zero drift, the thrust was determined from the load cell reading at the last frame of data (just before shutdown) and adjusted using the posttest zero reading of the load cell (three and a half seconds after shutdown was initiated). Thrust calibrations were performed at altitude and with pressurized propellant lines.

The uncertainties in the vacuum thrust, specific impulse, characteristic exhaust velocity, and thrust coefficient were determined using standard JANNAF procedures[10]. The bias in the thrust measurement due to the zero drift was determined using the difference in the pretest and posttest zero readings of the load cell. Uncertainties in the vacuum thrust were typically 1 to 2 percent in the positive direction (the direction of the bias) and 0.6 percent in the negative direction. Uncertainties in the vacuum specific impulse were typically + 2.5 percent, - 1.2 percent. The uncertainties in the characteristic exhaust velocity were +/- 1.5 percent. Thrust coefficient uncertainties were + 2 percent, - 1.1 percent.

The RPLUS Code

The RPLUS code models the fully coupled Navier-Stokes and species transport equations using the lower-upper, symmetric successive over-relaxation (LU-SSOR) scheme[7,11,12,13]. The combustion process of hydrogen and oxygen is modeled by an 8-species, 18-step finite-rate reaction mechanism. Turbulence is simulated by the Baldwin-Lomax algebraic model for the wall boundary layer and by a modified Prandtl's mixing length model for the reacting shear layer between the film cooling flows and the pre-combusted, oxygen-rich core flow.

The plane of the chamber sleeve exit served as the inflow surface for the code. The calculation starts with an initial flowfield derived from one-dimensional, isentropic flow. A more detailed description of the RPLUS code can be found in Reference 7 and References 11-13.

The grid for this problem consisted of 202 axial and 60 radial lines and was clustered in the regions of high gradients in the flow. For each case in this study, the code was run for 27,000 iterations, reducing the residual of the density by three orders of magnitude. This required about 11 hours of CPU time on a Cray-YMP. The mass flowrate was generally conserved to within two percent.

RPLUS Input/Output

The input required for the RPLUS code includes the thruster geometry and specification of the Mach number, pressure, and temperature of each stream at the inflow surface, and the species mass fractions of the pre-combusted core stream. The input for the four cases used for this study are listed in Table I. The coordinates of the thruster are given in Figure 1.

The test values of chamber pressure and core mixture ratio were used to derive the temperature and mass fractions of the core flow, using the Chemical Equilibrium Composition (CEC) computer program[14]. The core mixture ratio is defined as

$$O/F_{core} = O/F_{overall}/(1 - FFC),$$

where $O/F_{overall}$ is the overall mixture ratio and FFC is the fraction of fuel film cooling. CEC computed the equilibrium composition of the hydrogen/oxygen combustion products of the core. The use of equilibrium composition implies an 100 percent core combustion efficiency. A combustion efficiency of 97 percent was estimated for this thruster[8]. The equilibrium composition assumption could add 3 percent to the inlet enthalpy and give higher computed performance values.

The input value of Mach number of the core flow was found from one-dimensional, isentropic relations, using the contraction ratio of the outer sleeve wall diameter (see Figure 2) to the throat diameter and the specific heat ratio determined by CEC. The Mach number of the cooling sleeve was set equal to the core Mach number to facilitate the calculation. This could have given an underestimation of the enthalpy in the sleeve by as much as 1 percent. Normally, the film Mach number would be determined from the temperature and flow area of the film.

The input Mach numbers of the core and the film flows at the inflow plane set the inflow total enthalpy. After each iteration, the inflow axial velocity is obtained by extrapolation from the interior, while total enthalpy, pressure, and mass fractions are held constant. Thus, the mass flowrates adjust to the choked condition at the throat. As a consequence, the overall mixture ratio and percentage of fuel film cooling are not known a priori but are an output. This necessitated using interpolated and extrapolated experimental data to make a direct comparison with RPLUS results.

The same nominal chamber pressure was used for both the film and core flows, a good approximation for subsonic flows. Film temperature was taken from measurements as the average of two inner wall thermocouples, located 180 degrees apart and extending into the flow, at the chamber sleeve exit plane. The average was felt to be an adequate approximation as there was no more than an 33 K (60 F) variation between the two thermocouple measurements and the results from the code are relatively insensitive to film temperature variations.

For this study, the walls were assumed to be adiabatic. Use of the measured film temperature accounts for the enthalpy that is added to the hydrogen from regenerative heating in the nozzle and combustion chamber. However, not accounted for are heat losses from the thruster. The adiabatic wall assumption would, in all likelihood, give higher computed performance values compared to a case using an actual wall temperature profile.

RPLUS Results and Discussion

The test data are presented in Figures 4 and 5, which shows specific impulse versus mixture ratio and characteristic exhaust velocity versus mixture ratio, respecitvely, for families of fuel film cooling percentage. A least squares linear fit of the test data (performance as a function of mixture ratio) was applied for each value of fuel film cooling.

The test data plots indicate that performance decreases with increasing mixture ratio fairly linearly. Performance differences between values of fuel film cooling are fairly constant, although test experience has indicated that performance degrades more rapidly at higher percentages of fuel film cooling. A thrust coefficient of about 1.76 was found over the range of fuel film cooling and mixture ratio values.

Three of the four RPLUS cases had mixture ratios and fuel film cooling percentages that fell in the range of test data. However, the output of the RPLUS cases did not directly correspond to the test data in either mixture ratio or fuel film cooling. Therefore, linear interpolation between the least squares curves in Figures 4 and 5 was used to derive the points for direct comparison with the RPLUS cases. For the fourth RPLUS case, outside of the mixture ratio and fuel film cooling range of the test data, the comparison was made to extrapolated data.

Figure 6 plots experimental and computed specific impulse versus mixture ratio at the output values of fuel film cooling percentage. Figure 7 shows a similar plot for characteristic exhaust velocity versus mixture ratio. The experimental uncertainties of the interpolated data are assumed to be the same as the measured data and are shown in Figures 6 and 7.

The comparison of RPLUS results with test data is also listed in Table II. The comparison shows that the computed results follow the same trend as the experimental data. The computed specific impulse values are lower than the nominal test data by two to three percent, while computed characteristic exhaust velocity values are lower than the nominal test data by three to four percent. Table II also shows that thrust coefficient was computed to within two percent of the experimental value.

Geometric factors in the RPLUS modeling may be contributing to the discrepancy between the experimental and analytical results. More accurate modeling of the fuel film injection channels at the chamber sleeve exit may be required to better simulate the mixing between the core and film flows. Furthermore, a mixing model that includes interaction between the turbulence model and chemical reactions along the shear layer may also better represent the flowfield.

Summary

A preliminary analysis of a gaseous hydrogen/gaseous oxygen 110 N (25 lbf) rocket using the RPLUS code and comparison with test data over a range of mixture ratios and fuel film cooling percentages was accomplished. The RPLUS code uses the full Navier-Stokes equations with finite-rate chemistry. Test data were generated from performance testing of the rocket in an altitude facility and data were interpolated for a direct comparison to the code output. The computed values of specific impulse and characteristic exhaust velocity correctly followed the trends of the experimental data. Specific impulse computed by the code was lower than the comparable test values by about two to three percent. The computed characteristic exhaust velocity values were lower than the comparable test values by three to four percent. Thrust coefficients computed by the code were found to be within two percent of the measured values. The discrepancy between computed and experimental performance values could not be attributed to experimental uncertainty. Ideal assumptions have been made, such as equilibrium composition in the core and adiabatic walls, in this study to simplify the modeling, which increase computed performance values. The discrepancy between computed and measured values, then, may be related to the modeling of the mixing between the core and film flows and to the lack of interaction between the turbulence model and chemical reactions along the shear layer.

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Table I: Input Cases for RPLUS

Case #	O/Fcore	Pc,kPa	Tfilm,K	Mfilm	Tcore,K	Mcore
1	10.77	524	448	.203	3246	.203
2	15.17	517	620	.203	3102	.203
3	16.00	517	650	.203	3074	.203
4	17.11	517	644	.203	3036	.203

Species Mass Fractions

Case #	H2	O2	OH	H2O	H	O	HO2	H2O2
1	.00802	.22982	.12043	.60377	.00208	.03473	.00044	.00002
2	.00284	.40608	.08969	.47204	.00078	.02762	.00051	.00002
3	.00239	.43302	.08389	.45335	.00065	.02577	.00050	.00002
4	.00191	.46626	.07652	.43061	.00051	.02331	.00049	.00002

O/Fcore = Core Mixture Ratio

Pc = Chamber Pressure

Tfilm = Film Flow Temperature

Mfilm = Film Flow Mach Number

Tcore = Core Flow Temperature

Mcore = Core Flow Mach Number

Table II: Comparison Between Test Data and RPLUS

			Test Data			RPLUS		
Case #	O/Foa	% FFC	Isp,sec	C*,m/s[a]	Cf	Isp,sec	C*,m/s[b]	Cf
1	4.80	55.4	396	2197	1.77	384	2131	1.77
2	7.30	51.9	365	2031	1.76	353	1963	1.76
3	8.05	49.7	358	1994	1.76	349	1913	1.79
4	9.07	47.0	344	1920	1.76	337	1866	1.77

Percent Difference Between Test Data and RPLUS

Case #	Isp	C*
1	3.0	3.1
2	3.3	3.3
3	2.5	4.1
4[c]	2.0	2.8

O/Foa = Overall Mixture Ratio

% FFC = Fuel Film Cooling Percentage

Isp = Specific Impulse

C* = Characteristic Exhaust Velocity

Cf = Thrust Coefficient

[a] Experimental C* was determined using the measured chamber pressure corrected by momentum pressure loss per standard JANNAF procedure (see CPIA Publication 245, April 1975, pp. 2.1.3A-2.1.3B)

[b] RPLUS C* was determined using stagnation pressure calculated from isentropic relations

[c] Comparison is being made to extrapolated test data

CHAMBER CONTOUR COORDINATES (INCHES)

X	R	X	R	X	R	X	R
0.0000	0.5000	2.5426	0.5601	3.6151	1.0205	4.6884	1.3042
1.3019	0.5000	2.6614	0.6240	3.7354	1.0591	4.8068	1.3284
1.5519	0.4330	2.7819	0.6849	3.8558	1.0957	4.9256	1.3516
1.7508	0.3182	2.9038	0.7430	3.9735	1.1299	5.0444	1.3736
2.0000	0.2514	3.0248	0.7971	4.0918	1.1624	5.1638	1.3946
2.0616	0.2678	3.1411	0.8461	4.2104	1.1935	5.2830	1.4146
2.1794	0.3424	3.2584	0.8928	4.3294	1.2232	5.4025	1.4330
2.3018	0.4188	3.3765	0.9375	4.4487	1.2513	5.5293	1.4541
2.4256	0.4932	3.4950	0.9800	4.5684	1.2784		

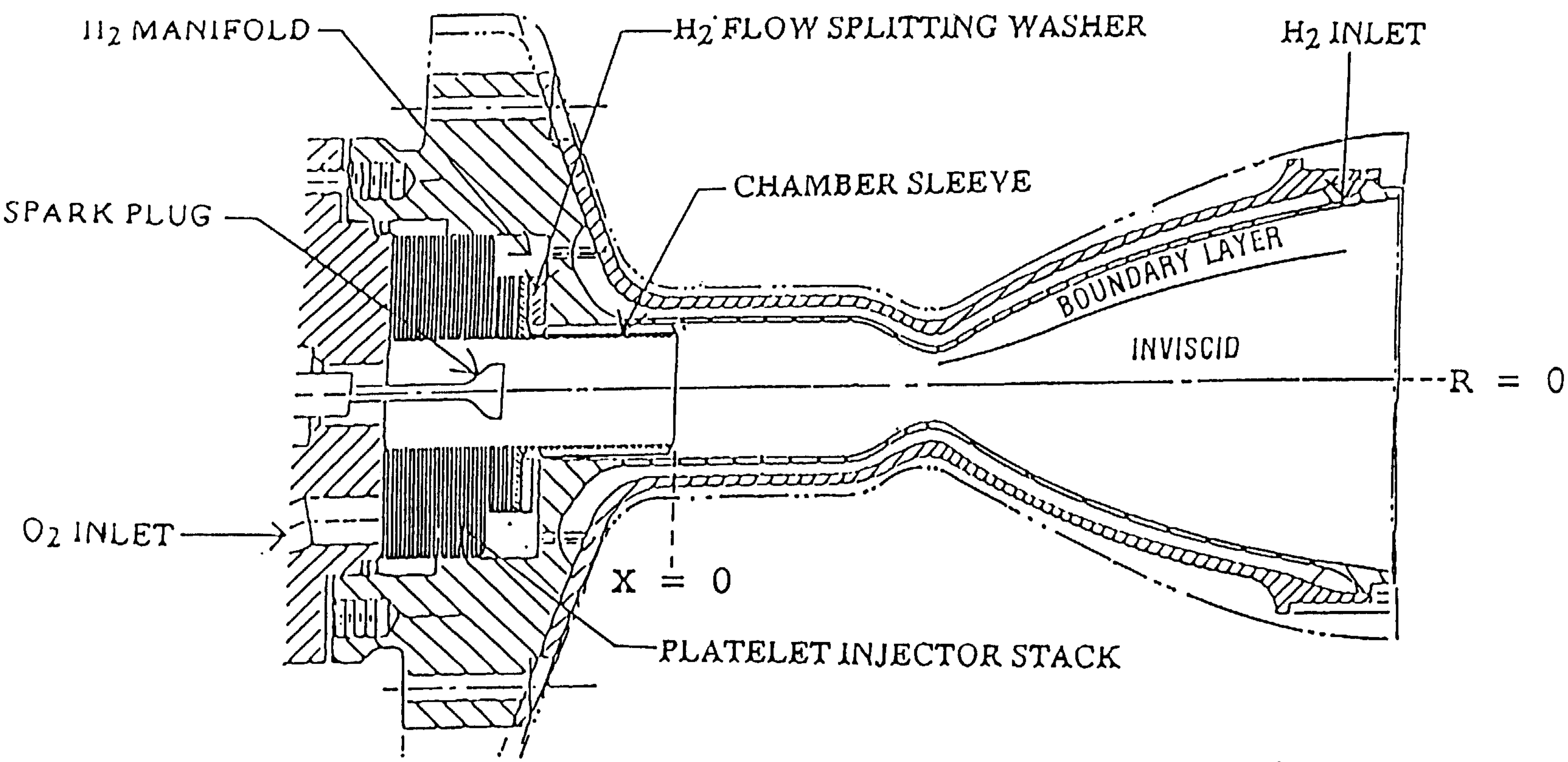
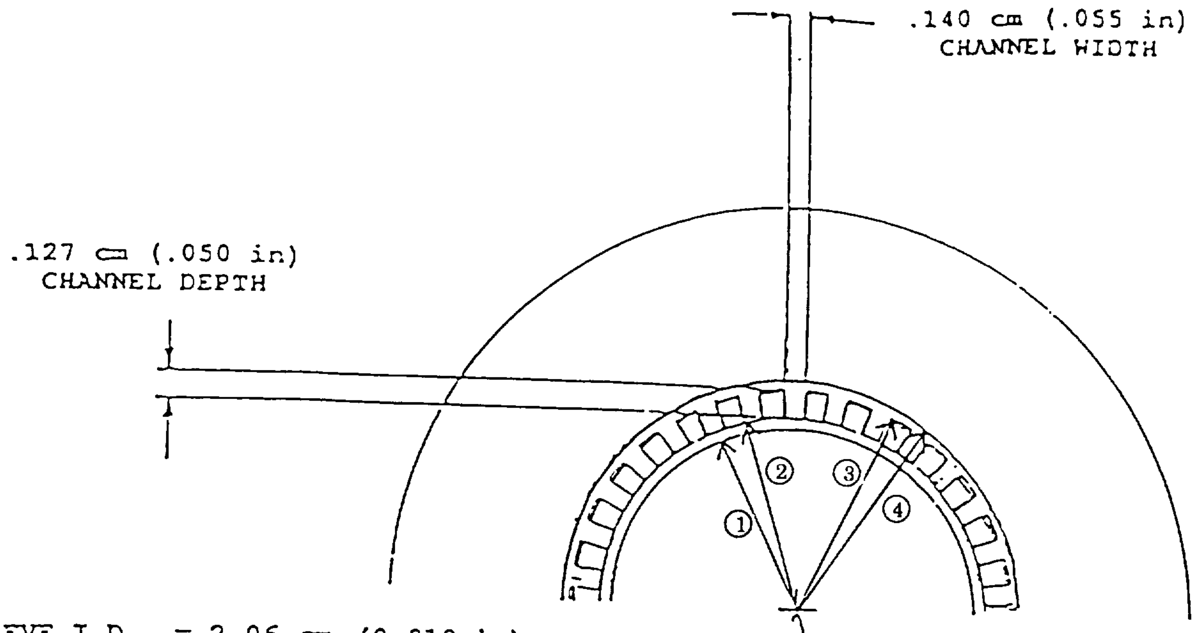


Figure 1: Aerojet Thruster Cutaway and Contour Coordinates



1 SLEEVE I.D. = 2.06 cm (0.810 in)
2 SLEEVE O.D. = 2.21 cm (0.870 in)
3 CHANNEL O.D. = 2.46 cm (0.970 in)
4 CHAMBER DIA. = 2.54 cm (1.000 in)

FILM FLOW AREA = 0.8316 cm^2 (0.1289 in^2)

Figure 2: Chamber Sleeve Exit Geometry

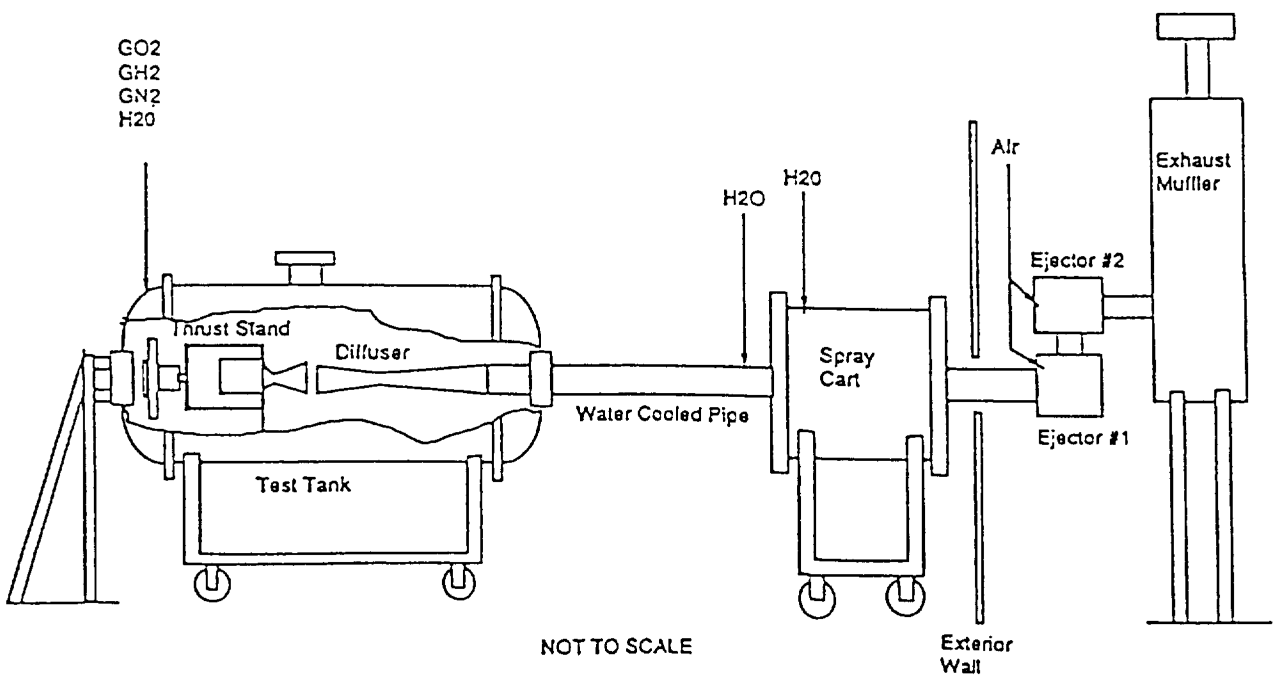


Figure 3: Diagram of RL-11 Test Rig

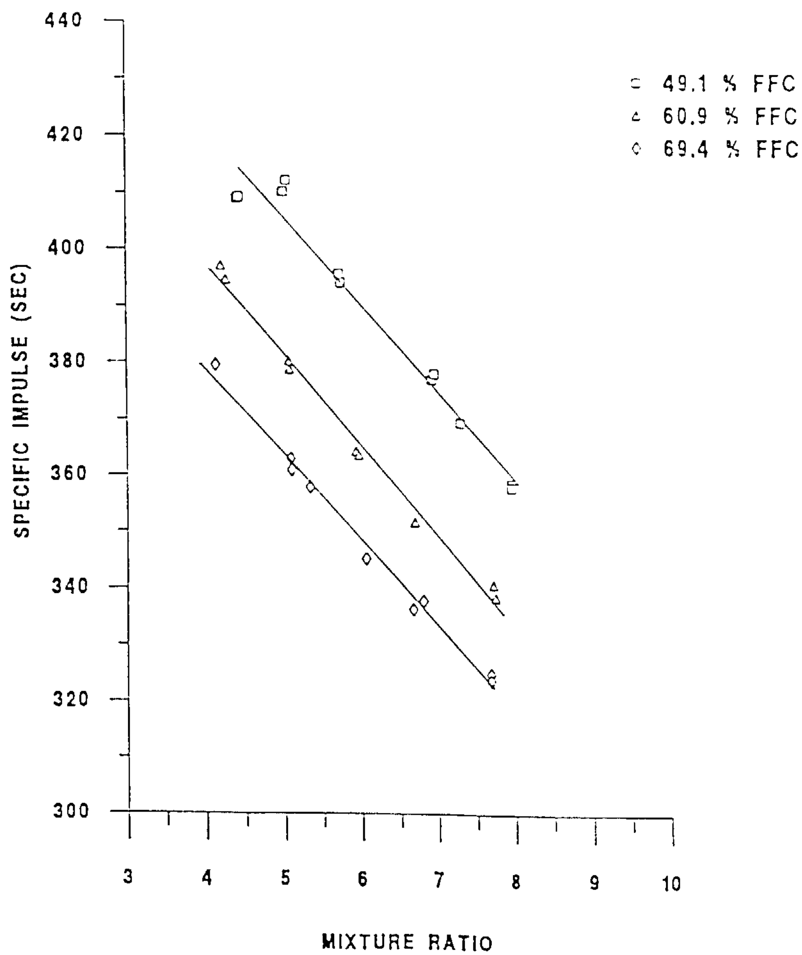


Figure 4: Specific Impulse vs. Mixture Ratio

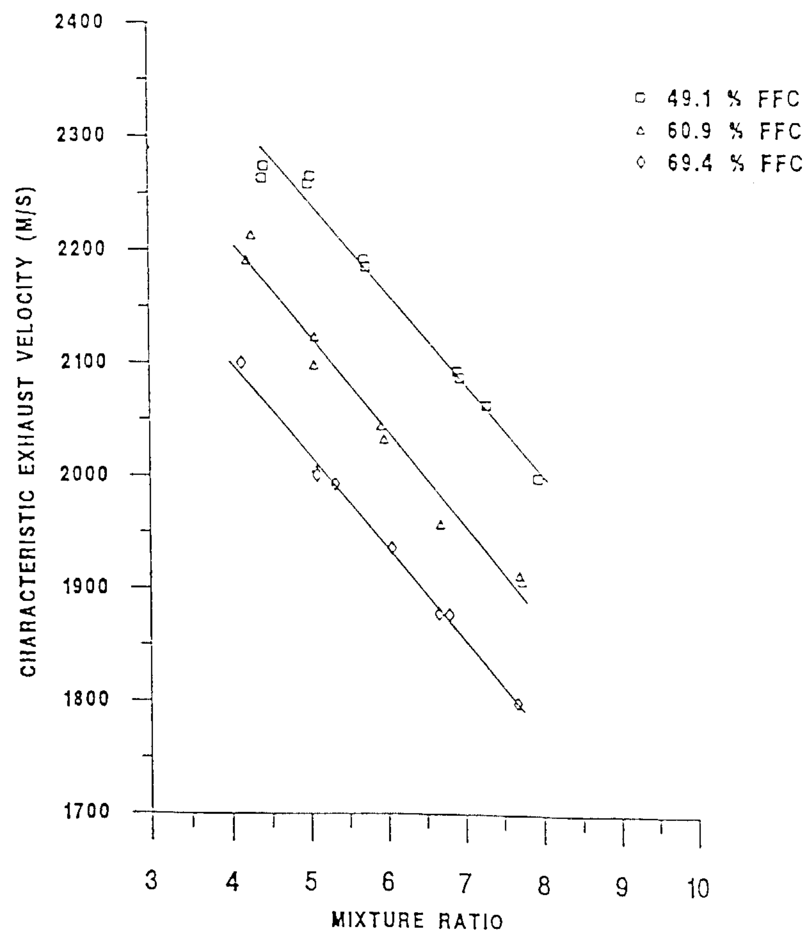


Figure 5: Characteristic Exhaust Velocity vs. Mixture Ratio

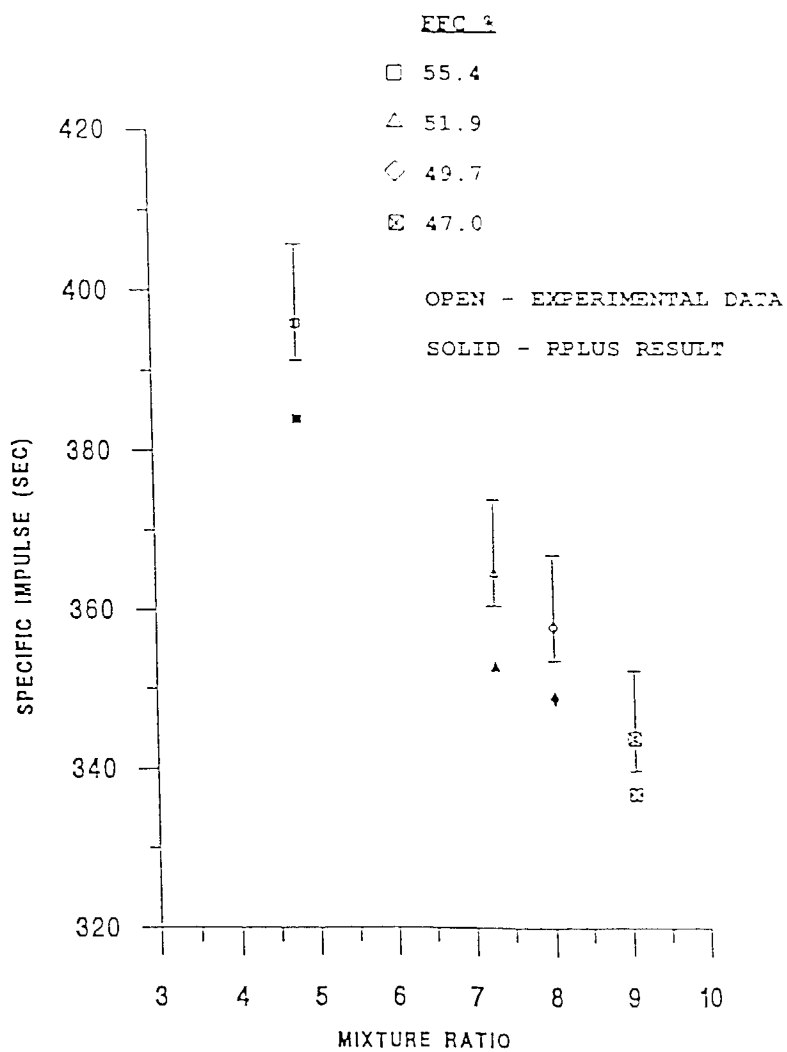


Figure 6: Specific Impulse vs. Mixture Ratio

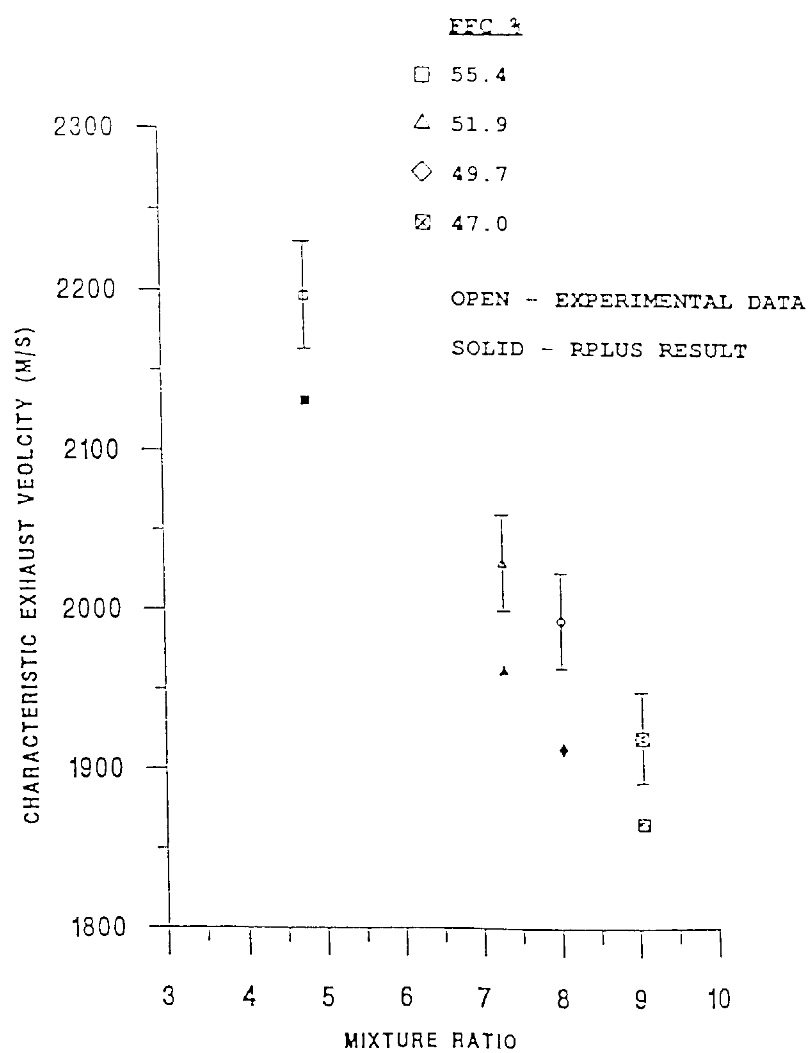


Figure 7: Charateristic Exhaust Velocity vs. Mixture Ratio

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

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