Research and Development of a 1-kw Plasmajet Thruster

ROBERT V. GRECO* AND WILLIS A. STONER†
Plasmadyne Corporation, Santa Ana, Calif.

Work is reported on the development of a small arc plasmajet thruster suitable for attitude control and orbit adjustment. The effort to date has resulted in a thruster capable of delivering the required thrust (0.01 lb) at a specific impulse in excess of 1100 sec. Kinetic efficiencies in excess of 35% have been obtained. This performance was obtained using hydrogen as a propellant. The specific design, along with other designs tested, is reported herein. Some design considerations pertinent to small plasmajet thrusters are presented.

Nomenclature

A = surface area
c_p = specific heat at constant pressure
D = diameter
f_s = surface friction coefficient
L = length
$\dot{m}$ = mass flow rate
M = Mach number of gas flow based on the average velocity V
P = static pressure of gas
P_r = Prandtl number
R = gas constant
R_e = Reynolds number based on the average velocity V
T = static temperature of gas
V = velocity averaged over the cross section
γ = ratio of specific heats
θ = nozzle half angle
λ = thermal conductivity
μ = viscosity
ρ = mass density of gas

1. Introduction

THIS paper describes the first phase of a program of development work on a plasmajet thruster suitable for the attitude control and orbit adjustment of a satellite. The thruster was designed to use a power of 1 kw and produce a minimum thrust of 0.01 lb.

The thruster development in this phase was involved primarily with overcoming problems that are unique to small size. These included the problem of reducing viscous and heat transfer losses to acceptable values at the low Reynolds numbers encountered (several hundred in some of the nozzle designs used). In addition, considerable effort was required in solving such mechanical problems as accurately maintaining critical arc chamber and nozzle geometries in configurations where the nozzle throat diameter is only 0.009 in. The accurate measurement of 0.01 lb of thrust required the development of a special thrust balance, and a special critical orifice flow meter was built and calibrated to measure the low propellant flows. Electrode and nozzle erosion rates are more critical in small size simply because there is closer tolerance on allowable dimensions of electrode and nozzle parts.

The program was largely experimental, with analytical work providing guidance. A number of designs were tried before obtaining a configuration with satisfactory performance. The latest design (the Cute VI) has achieved a specific impulse of 1100 sec with a kinetic efficiency of 35%. The operating life as demonstrated by the first two life tests is in excess of 25 hr. Refinement of this design for future tests should increase this duration.

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* Presently Senior Research Engineer, Space and Information Systems Division, North American Aviation. Member AIAA.
† Consultant. Member AIAA.

2. Design Considerations

Since difficulty was experienced in achieving good performance with small size thrusters, a large portion of the analytical effort has been concentrated on the evaluation of low Reynolds number designs. The performance of a small thruster, as compared to units of larger size, is adversely affected by percentage increases in the heat transfer losses that occur between the gas and the wall, in the wall friction losses, and in losses occurring in the electrode fall regions.

The strong influence of Reynolds number on heat transfer and viscous losses may be illustrated by a simple laminar flow model where the nozzle is sufficiently long that equilibrium temperature and velocity profiles may be assumed as in pipe flow; then

$$\frac{\text{viscous loss rate}}{\text{kinetic energy rate}} = \frac{[\mu(dV/dr)_{\text{wall}}A]}{\dot{m}(\bar{V}^2/2)}\,\bar{V} = \frac{\mu 8(\bar{V}/D)\pi DL\bar{V}}{\rho\bar{V}(\pi/4)D^2(\bar{V}^2/2)} = \frac{64\mu}{\rho\bar{V}D}\left(\frac{L}{D}\right) = \frac{64\left(\dfrac{L}{D}\right)}{R_e} \tag{1}$$

while

$$\frac{\text{heat transfer loss rate}}{\text{energy input rate}} = \frac{\lambda(dT/dr)_{\text{wall}}A}{\dot{m}\,c_p\bar{T}} = \frac{\lambda 4\frac{4}{11}[(\bar{T} - T_{\text{wall}})/D]\pi DL}{\rho\bar{V}(\pi/4)D^2c_p\bar{T}} = 17.5\left(\frac{L}{D}\right)\left(1 - \frac{T_{\text{wall}}}{\bar{T}}\right)\frac{\lambda}{\rho\bar{V}Dc_p}$$

since $P_r = \mu c_p/\lambda$ and $R_e = \rho\bar{V}D/\mu$, then

$$\frac{\text{heat transfer loss rate}}{\text{energy input rate}} = 17.5\left(\frac{L}{D}\right)\left(1 - \frac{T_{\text{wall}}}{T}\right)\frac{1}{R_eP_r} \tag{2}$$

These expressions apply for laminar flow only, and all designs considered in this program were in the laminar flow regime.

In both Eqs. (1) and (2) the percentage loss varies inversely as Reynolds number. In a short nozzle with accelerating flow, for which the flow is not developed fully, the dependency on Reynolds number will not be quite this strong because the flow profiles will show steeper gradients at the wall at higher Reynolds numbers. For very short nozzles a good model would include a thin boundary layer building up on a flat plate. In this case, the percentage losses vary inversely as the square root of Reynolds number.

Figure 1 shows an estimate of the variation of Reynolds number with the ideal specific impulse. The curves were obtained for various power levels and arc chamber pressures by using frozen flow conditions. It is noted that, at a power level of 1 kw and an arc chamber pressure of 1 atm, the Reynolds number at the throat is about 250 for an ideal specific impulse of 1300 sec. (For this case the actual specific impulse, taking losses into account, might be 1000 sec or less.) This is quite a low value for Reynolds number, and configurations of this type clearly would be subject to severe viscous and heat transfer losses. The early units built on this program used moderate values for arc chamber pressure in order to prevent extremely small nozzle throat diameters. However, it soon became evident from test results that the losses would be too high unless the Reynolds number could be increased by increasing arc chamber pressure.

For the scale of thermal losses shown in Fig. 1, it is assumed that the loss ratios vary as the $\frac{3}{4}$ power of Reynolds number. This variation is intermediate between the variation that would be expected for fully developed flow in a long nozzle and the flat plate case where the boundary layer is growing. The magnitude shown on the scale is selected to correspond approximately with test data for a water-cooled thruster.

In Fig. 1, the following assumptions are made: 1) Prandtl number is constant; 2) nozzle geometries are similar for all cases considered; 3) relative change in Reynolds number is the same through the nozzle for all cases considered, and 4) velocity profiles are midway between those for a flat plate and for pipe flow.

Figure 2 shows the variation of Prandtl number and molecular weight with temperature for hydrogen at 1 atm using gas properties of King.[1] The curve shows no severe changes in Prandtl number occurring as the gas dissociates.

It is interesting to note that, at high specific impulses, very low Reynolds numbers occur again even in the 30-kw power range. Many of the techniques which are found to be helpful in improving performance in the 1-kw range are expected to be useful in high specific impulse designs.

Following selection of a design with as high a Reynolds number as feasible, the next step is to choose a nozzle shape with good performance at this Reynolds number. Some insight into the choice of nozzle shapes for low Reynolds numbers may be obtained by determining the minimum nozzle angle that is useful for producing thrust. In order for an element of nozzle surface area to be acted on by a force in the positive thrust direction, the axial component of pressure force must exceed the axial component of drag force. This condition may be expressed as follows:

$$PdA \sin\theta > f_s dA(\rho V^2/2) \cos\theta \tag{3}$$

In an expanding laminar flow, the lowest value that the surface friction coefficient can approach is the coefficient for equilibrium laminar flow in a pipe is

$$f_s = 16/R_e \tag{4}$$

Substituting f_s from Eq. (4) together with the equation of state $\rho = P/RT$ in Eq. (3) yields

$$\tan\theta > (8/R_e)\cdot(V^2/RT) \tag{5‡}$$

Making use of the definition of Mach number $M = V/(\gamma RT)^{1/2}$ yields

$$\tan\theta > 8\gamma M^2/R_e \tag{6}$$

This relationship is plotted as solid lines in Fig. 3, using R_e/γ as a parameter. Notice that the true minimum useful angle is in all cases greater than the value shown because the true surface friction coefficient is greater than the equilibrium value assumed.

Equation 6 indicates that nozzles for low Reynolds numbers should be contoured in a horn shape rather than a bell shape to avoid excessive friction loss at the nozzle exit where the Mach number is high. The required divergence angles can be quite severe. For example, in one nozzle designed for frozen flow at 1 kw and an arc chamber pressure of $\frac{1}{10}$ atm, the Reynolds number was estimated to be between 40 and 50 in the nozzle with a γ of 1.585. In this case the nozzle half angle should substantially exceed 18° immediately following the throat, 36° at Mach 1.5, 52° at Mach 2, and 71° at Mach 3.

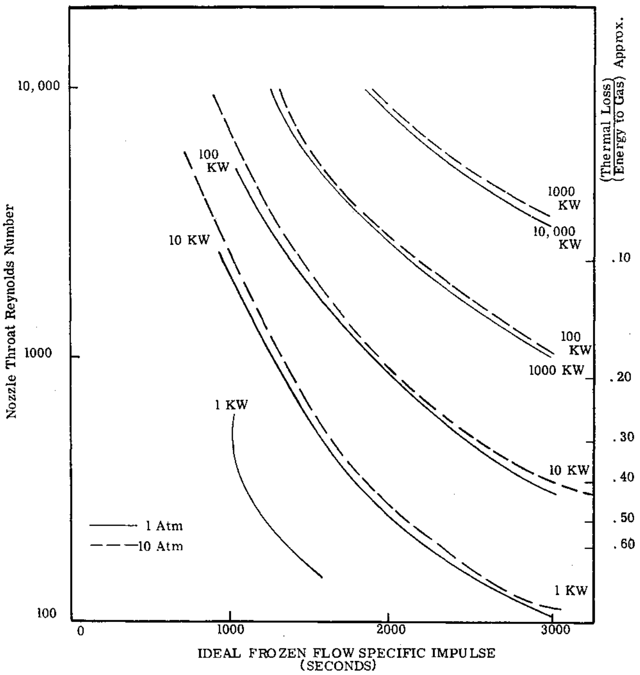

Fig. 1 Nozzle throat Reynolds number as a function of ideal frozen flow specific impulse and input power

The argument is sometimes posed that friction losses do not seriously affect the performance of nozzles that operate at high pressure ratios because the energy remains in the gas as heat and can be converted to kinetic energy further downstream. Although this argument generally is valid, it does not help in the limiting case where the nozzle angle is such that no contribution to thrust results. In this case, the Reynolds number is decreased in passing through the initial section of the nozzle, and, although the thermal energy still exists in the gas, it becomes more difficult than ever to convert to kinetic

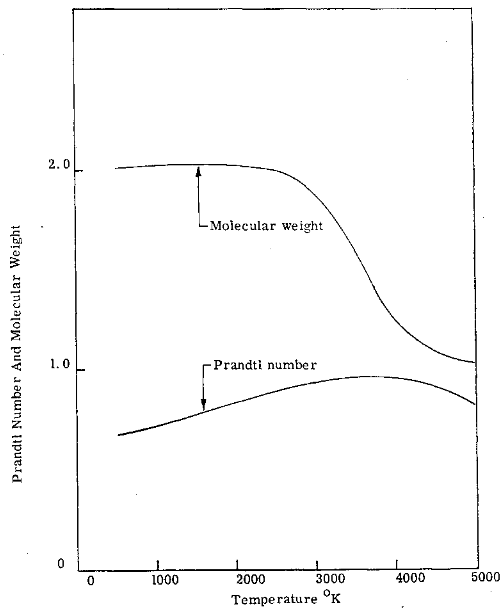

Fig. 2 Variation of Prandtl number and molecular weight with temperature for hydrogen at 1 atm (using gas properties given by King[1])

‡ In practice, a constant somewhat larger than 8 should be used to allow for the more severe velocity profile encountered in expanding flow.

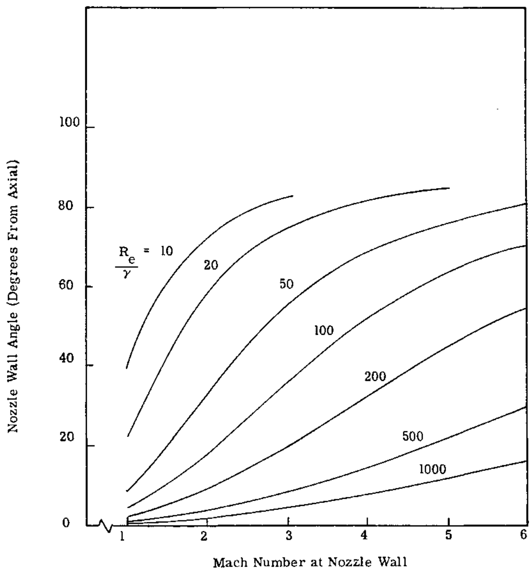

Fig. 3 Minimum useful nozzle angle for low Reynolds number applications

energy. In fact, the presence of a nozzle cone actually can reduce the overall thrust obtained.

When consideration is given to the fact that an increase in nozzle angle tends to increase the Mach number near the wall because of expansion waves generated by the turn, it becomes evident that in some cases it may be preferable to cut the nozzle off short. Fortunately, when the flow is frozen, the losses due to incomplete expansion are not too severe. Figure 4 shows nozzle efficiency (with the pressure-area correction term included) vs nozzle area ratio for incomplete isentropic expansion. With frozen flow, hydrogen will fall in between the two values of γ shown. For a γ of 1.667, the nozzle efficiency for a simple orifice is 0.64, which undoubtedly is higher than could be obtained with a nozzle of conventional geometry operating at the Reynolds numbers considered in this program.

3. Thruster Design and Performance

The preceding analyses prompted the modification of a water-cooled plasma head[2] to verify the conclusions reached. This modified thruster, although lacking the proper instrumentation, appeared to perform at a specific impulse of approximately 1000 sec at design mass flow and power when operated at high arc chamber pressures (approximately 5 atm). The results were encouraging enough to prompt the development of a high arc chamber pressure, radiation cooled thruster.

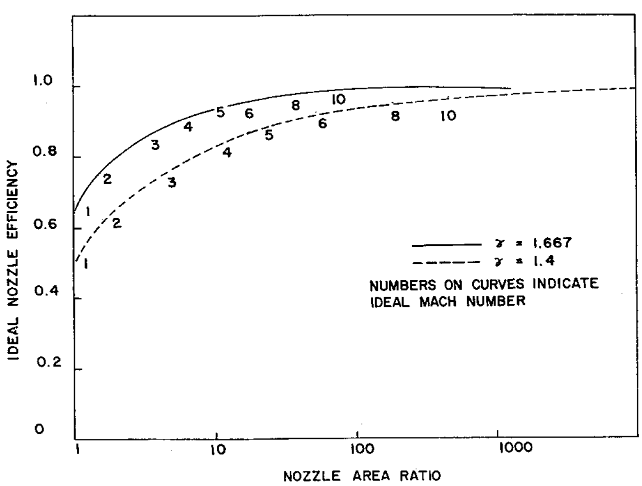

Fig. 4 Effect of area ratio on ideal nozzle efficiency for constant γ

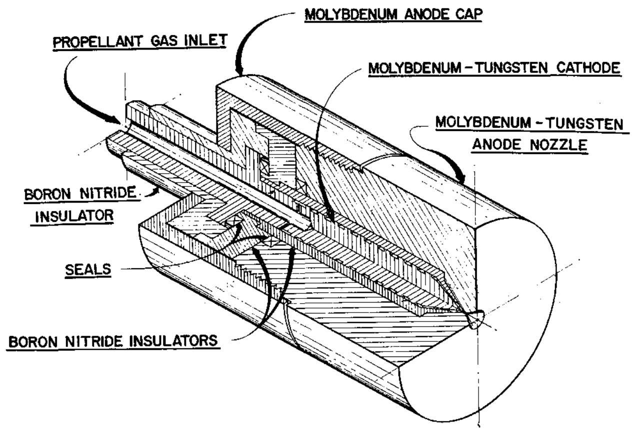

Fig. 5 Cute V 1-kw thruster

The series of thruster investigations resulted in a final design designated Cute V (see Fig. 5). The unit consists of anode and cathode housings of molybdenum and electrodes fabricated from 2% thoriated tungsten. Boron nitride was used for the insulators, and the seals were of inconel. The Cute V thruster was the first design that was sealed successfully and operated at a performance level in excess of contractual requirements.

The term "sealed" applies to the pressure checks conducted prior and after each test. There is presently no way to check for leakage during operation. The performance obtained with this design was in excess of 1000 sec with a kinetic efficiency of 25%, kinetic efficiency being the percentage of input power converted to useful directed energy. The life of the unit is limited by the erosion of the nozzle throat (anode) which reduces the arc chamber pressure and the performance. The life limit is presently 10 to 12 hr with this design. The foregoing performance was obtained using hydrogen as the propellant.

Normal operation of the Cute V thruster has yielded the following operating characteristics with a mass flow of 10^{-5}

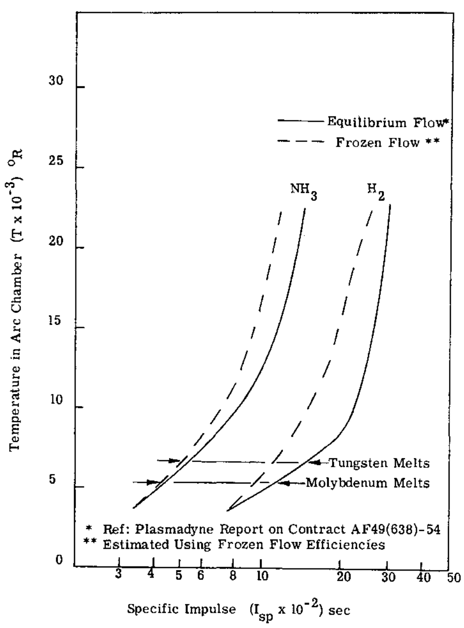

Fig. 6 Theoretical chamber temperature vs specific impulse

lb/sec and an input power (across thruster) of 1 kw: voltage ~ 80 to 90 v, current ~ 12 to 14 amp, arc chamber pressure ~ 5 atm, thrust ~ 0.01 lb, specific impulse ~ 1000 sec, and kinetic efficiency ~ 25%.

Under normal operation, the steady slow erosion of the nozzle throat decreases the arc chamber pressure, causing a decrease in voltage and a corresponding increase in current. Performance deteriorates as the voltage and arc chamber pressure decrease, raising the heat input to the thruster. Under normal operation, approximately 60% of the input energy is transferred to the gas, with the rest accounted for as thermal losses in the thruster.

Special tests were conducted with the Cute V thruster or minor modifications thereof. These special tests included 1) operation with reversed polarity, 2) operation with ammonia as the propellant, and 3) operation at conditions up to three times the design input power and mass flow. The results of these special tests are reported briefly in the following paragraphs.

The slow erosion or ablation of the nozzle throat (anode) limits the life of the thruster, whereas the upstream electrode (cathode) remains in near perfect condition. It was felt that reversal of polarity would transfer the erosion to the upstream electrode, resulting in longer life of the unit. However, the tests resulted in severe erosion of the upstream electrode, indicating that this part was not cooled adequately to operate as an anode. It might, however, be advantageous to try reversed polarity with a design using intensive cooling of the rear electrode and some method for keeping the arc foot in motion over the surface of this electrode.

Since hydrogen is a relatively difficult propellant to store under cryogenic conditions, an evaluation of ammonia was made. Figure 6 shows propellant temperature as a function of specific impulse for ideal frozen flow. As one can see from the molybdenum and tungsten melting points indicated, hydrogen presents less severe materials problems operating at the same specific impulse. Experimental tests proved that ammonia operation was severely detrimental to thruster life, even at low specific impulse levels of approximately 550 sec at a kinetic efficiency of 8%. The longest operation successfully sustained with ammonia was 83 min. Figure 7 shows the heat that can be absorbed by the incoming propellant as a function of the temperature to which the gas is heated. It is apparent that regenerative cooling offers more attractive gains with hydrogen than with ammonia. The analytical and experimental effort indicates that development of an ammonia thruster to operate radiation cooled in this small size would be a difficult undertaking and would involve rather severe compromises in performance.

Another approach that might increase life, at the expense of increased heat transfer and viscous losses, is the use of an enlarged arc chamber upstream of the nozzle throat. Several modified Cute V thrusters were tested using various geometries of this type. All were unsuccessful in that they either plugged on start or operated at lower performance levels. The use of enlarged arc chambers cannot be considered completely evaluated at this time. It does, however, appear that their use will result in a loss of performance due to an increase of wall surface area exposed to the hot gas. In general, the best performance has been obtained using the shortest possible passage for the hot gas (a short nozzle doubling as an anode).

Several tests were conducted to determine the range of operation of the Cute V thruster. To date, power levels to 2.85 kw have been operated successfully. Performance has been at the 1000-sec impulse level at efficiencies of 27%. Further extension of the mass flow and power levels could not be made because of test facility limitations. Operation with the Cute V at the high power levels has resulted in operation at arc chamber pressures of 14 atm. This operation was sustained at a mass flow of 3.12×10^{-5} lb/sec and an input power of 2.84 kw. The voltage was 148 v and the current 19.2 amp. Thruster operation appeared much cooler, indicating a higher percentage of input power being transferred to the propellant. No actual data were obtained at the maximum power, since the plume energy was in excess of the calorimeter range. However, measurements did show the energy in the gas to be more than 75% of the electrical power input.

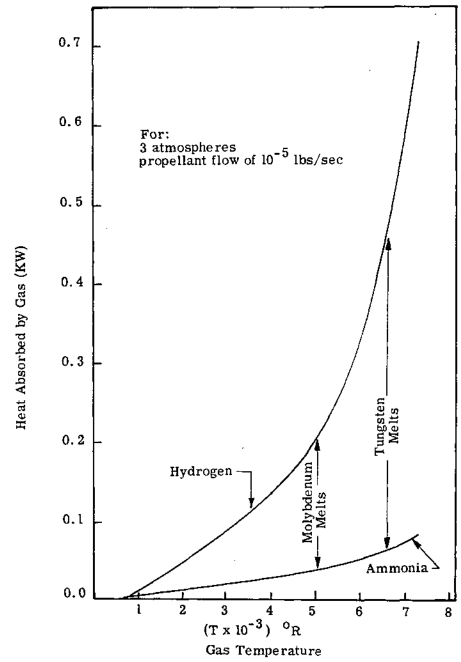

Fig. 7 Propellant heat absorption with regenerative cooling

A design designated Cute VI was fabricated and tested. The thruster is shown in Fig. 8. It is basically the Cute V thruster with internal flow passages that permit preheating of propellant and provide thermal distribution of heat in the thruster. The materials are the same as those used in the Cute V thruster. This design operated at a specific impulse in excess of 1100 sec and a kinetic efficiency of 35%. Life tests have been conducted in excess of 25 hr.

Typical operation of this unit at a mass flow of 10^{-5} lb/sec and an input power of 1 kw is similar to that of the Cute V thruster. The performance increase (kinetic efficiency) is probably due to the preheating of the propellant, which permits recovery of some of the power lost to the thruster walls.

The future effort of the program will be to extend the life and increase the performance of a 1-kw thruster. In addition, an investigation of a.c. operation will be conducted. To date, several tests with single-phase a.c. have been performed. It is, however, too early to evaluate the results obtained. The

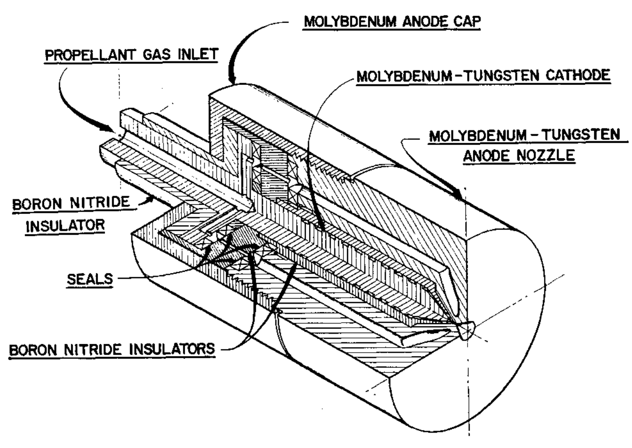

Fig. 8 Cute VI regeneratively cooled 1-kw thruster

effort also will be continued to determine the power and flow range over which the thruster design may be operated successfully.

Conclusions

Although there are a number of difficult problems associated with the development of a plasmajet thruster in the 1-kw size range, it has been demonstrated that satisfactory performance (1100 sec with 35% efficiency) can be attained. The problem of obtaining satisfactory life with the small physical dimensions involved is particularly difficult and will require continued development of the unit.

References

[1] King, C. R., "Compilation of thermodynamic properties, transport properties, and theoretical rocket performance of gaseous hydrogen," NASA TN D-275 (October 1960).

[2] Greco R. V. and Stoner, W. A., "Development of a plasmajet rocket engine for attitude control," Plasmadyne Rept. 1341-A (December 1961).

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Experiments on Recombination Effects in Rocket Nozzles

R. Hoglund,[1] D. Carlson,[2] and S. Byron[3]

Aeronutronic Division, Ford Motor Company, Newport Beach, Calif.

Rocket engine experiments with the RP-1/oxygen propellant system are described. Essentially, the experiments consist of measurement of flow rates, chamber pressure, nozzle pressure profile, and static gas temperature in a large (30:1) expansion ratio nozzle. The measured static pressures and temperatures are compared to theoretical predictions for frozen (chemical reaction inhibited) and equilibrium expansion processes. It is found that near-frozen conditions exist at low (below 1.6) oxidizer-to-fuel (O/F) ratios, and near-equilibrium conditions exist at higher (above 2.0) O/F ratios. The measurements at the higher O/F ratios agree with predictions based upon partial-equilibrium calculations using available reaction rate data from laminar flame and shock tube studies. The near-frozen experimental pressures at low O/F ratios indicate that solid carbon and methane (expected from equilibrium calculations) were not produced sufficiently fast to maintain chemical equilibrium.

ROCKET engine performance is usually quoted in terms of the two limiting cases of frozen or shifting equilibrium nozzle expansion processes. Frozen expansion results if all reaction rates are so slow that the combustion products are exhausted before chemical reaction can occur. Shifting equilibrium results if all reactions proceed fast enough in the nozzle that the local chemical composition everywhere is that associated with thermodynamic equilibrium. Although the difference between frozen and equilibrium expansion is frequently important, the number of possible reaction paths and the paucity of applicable rate data have discouraged analytical determination of the exact conditions to be expected with even the most common propellant systems. Definitive experiments also appear to be lacking, although some data on this subject have appeared recently and will be discussed later.

As part of a study of recombination and condensation effects in rocket nozzles,[4] experiments have been performed to provide quantitative data on recombination effects. The results of these experiments, along with techniques for predicting and detecting recombination effects, are described in this paper.

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[1] Manager, Fluid Mechanics Department. Member AIAA.

[2] Research Scientist, Fluid Mechanics Department. Member AIAA.

[3] Research Scientist, Fluid Mechanics Department. Member AIAA.

[4] This program was sponsored by the Advanced Research Projects Agency under Contract NOrd 17980.

Experimental Equipment

A rocket engine of nominal 1000-lb thrust level provides the desired stagnation pressure (usually 400 psia) and temperature (2000° to 3700°K) conditions. The only unusual feature of this engine is its relatively large combustion chamber, characterized by an L^* (ratio of chamber volume to throat area) of 125 in. This large chamber provides relatively complete combustion as evidenced by experimental characteristic velocities that average 98.3% of the theoretical equilibrium values for uncooled chamber operation (average deviation of 1.5%). Although the engine is equipped to handle a variety of liquid, gas, and slurry propellants, RP-1 fuel and gaseous oxygen were used for these experiments.

The products of combustion are expanded in a conical nozzle of 15° half angle, 1.25-in. throat diameter, and 30:1 exit area ratio. The nozzle is made of copper and is water-cooled at the throat. The nozzle exhaust is discharged at a pressure of about 0.1 atm into a supersonic diffuser section that exhausts into the atmosphere.

Information on conditions in the nozzle is obtained from both a series of static pressure measurements between the throat and the nozzle exit and static temperature measurements obtained at various stations. The pressure ports are connected to transducers through a commutating system that allows a single transducer to sense a number of pressures. The transducer outputs are recorded by an oscillograph.

Static temperature measurements have been obtained using both manual and automatic sodium-line reversal tech-