Simulation of Rocket-Nozzle Environments with an Arc-Plasma Generator

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I. Introduction

THIS note discusses the development and use of an arc-plasma generator as a rocket-motor simulator. A description of the arc-plasma generator, the requirements for accomplishing simulation of the chemically reactive rocket environment, and some typical test results are presented briefly.

II. Experimental Apparatus

A 1-Mw, magnetically stabilized, d.c. arc-plasma generator was used to provide the appropriate high-temperature, chemically reactive environment. The unit consists of two concentric, water-cooled copper electrodes surrounded by a 6-kgauss solenoid coil. The plasma generator has the capability of high pressure operation (1 to 30 atm) and multigas and metal oxide particle injection, making it particularly applicable to both liquid and solid rocket-motor simulation. The test nozzles of various ablative materials are attached to the plenum chamber of the plasma generator. The total enthalpy is determined by an energy balance on the system.

Nozzle materials that have been tested include graphites, graphite phenolics, and silica phenolics. A typical nozzle configuration for a materials test is shown in cross section in Fig. 1; throat diameters have ranged from 0.3 to 0.5 in. Instrumentation to measure internal temperature histories and heated-wall surface temperature was included in many of the firings. Also, the axial distribution of cold-wall heat-transfer coefficient for each nozzle configuration was determined with calorimetric nozzles formed by "stacking" individually water-cooled disks.

III. Simulation Requirements

The requirements for simulating the rocket-motor environment[1] are the duplication or close approximation of 1) the convective-transfer coefficients of energy, mass, and momentum, 2) the boundary-layer-edge properties of temperature, species concentration, and velocity, 3) the enthalpy-temperature relation of the combustion products, and 4) the radiative heat flux, if appropriate.

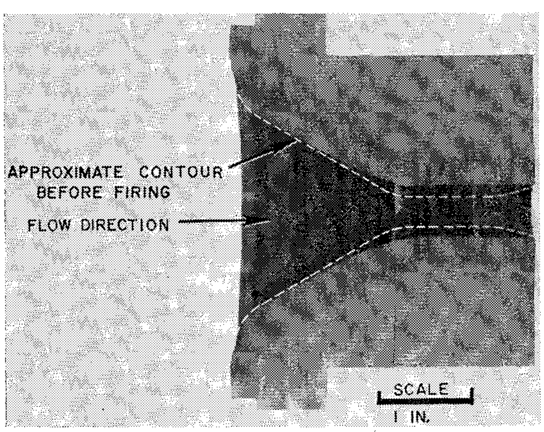

Fig. 1 Cutaway of a typical fired nozzle, graphite phenolic.

Failure to satisfy any one of these requirements could result in completely erroneous conclusions about a material. As an example, if shear stress were not duplicated, a weak char material or a melting material could appear as a good performer when, in the actual rocket environment, it would be quite the opposite.

Requirement 1 may be satisfied by duplicating, e.g., the convective heat-transfer coefficient. The mass-transfer coefficient and the friction coefficient are automatically duplicated closely through the Chilton-Colburn and Reynolds analogies, respectively. The relation between actual and simulated conditions that satisfies requirement 1 may be developed readily[1] and, with some appropriate simplifying assumptions, reduces to

$$\frac{p_{0_R}}{p_{0_A}}\left(\frac{D_{*A}}{D_{*R}}\right)^{0.222}\left(\frac{M_R}{M_A}\right)^{0.444}\times \left\{\frac{\left[\gamma\left(\dfrac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)}\right]_R}{\left[\gamma\left(\dfrac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)}\right]_A}\right\}^{0.444} = 1$$

for the nozzle throat, where the subscripts R and A refer to the rocket motor and arc-plasma generator, respectively, p_0 is the chamber pressure, D^* is the throat diameter, M is the molecular weight, and γ is the isentropic exponent.

Requirements 2 and 3 can be satisfied by the choice of a test-gas mixture that, when arc-heated to the proper temperature, yields the identical composition of the combustion products themselves. This, in fact, can be done for many propellants. Duplicating the molecular species concentration (requirement 2) may not be necessary nor even desirable in many cases, however. In the investigation of chemical reaction and mechanical shear effects, it often is advantageous to "isolate" the chemical reactant of interest by using a gas mixture, which contains inerts and the reactant only or, in the latter case, to eliminate entirely all of the chemically reactive species.

An example of two such gas mixtures is presented in Table 1. Both mixtures simulate the combustion products (less oxide particles) of a typical solid propellant in all respects but its chemical composition. Mixture 4 is appropriate for the study of oxidation effects on material performance and contains the mass fraction of oxygen available in the actual propellant for reaction with a wall material. All of the oxygen in the propellant except that tied up in CO and $\frac{1}{2}$ CO_2 is present in mixture 4 in the form of O_2, NO, and O. Mixture 5 is appropriate for the study of mechanical effects on material performance. It contains inerts only, and therefore chemical effects are eliminated.

The elemental compositions of the gas mixtures presented in Table 1 are dictated by requirement 3, the duplication or close approximation of the temperature-enthalpy relation.

Table 1 Comparison of typical gas mixtures for simulation of a solid propellant

	Mass fraction		
Element	Propellant	Mixture 4	Mixture 5
H	0.0538	...	...
He	...	0.2284	0.2237
C	0.1620	...	...
N	0.1259	0.6191	0.7764
O	0.3556	0.1525	...
Cl	0.3028	...	...

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Table 2 Typical total surface-recession test results

Material	Total temperature, °K	Initial plenum pressure, psia	Initial throat diameter, in.	Surface recession in throat, mils[a]	
				Mixture 4	Mixture 5
High-density graphite	3300	300	0.3	74 (30.7)	0 (28.6)
	2250	300	0.3	0 (30.7)	...[b]
Graphite cloth phenolic	3300	300	0.3	64 (30.3)	0 (26.2)[c]
	2200	255	0.3	43 (30.7)	0 (22.6)[c]

(a) Numbers in parenthesis indicate firing duration in seconds.
(b) No test since no surface recession occurred in the preceding more severe case.
(c) Significant weight loss caused by resin decomposition and out-gasing.

The comparison of the two gas mixtures with the actual propellant is presented in Fig. 2.

IV. Test Results

Both solid- and liquid-propellant rocket environment simulation have been accomplished using the arc-plasma generator. Testing of ablative nozzle material performance[2] and a study of the effects of specific chemical reactions on surface recession in ablative nozzles[1] have been performed. The test conditions and the results for total recession are shown in Table 2 for a number of test firings with the two gas mixtures presented previously. The primary variables are the nozzle material, the gas total temperature, and the chemical nature of the gas environment (mixture 4 is chemically reactive, and mixture 5 is inert). The test conditions of 3300°K total temperature and 300-psia chamber pressure using mixture 4 satisfy the simulation requirements for the nozzle throat of a solid-propellant motor with a 6-in.-diam throat at a chamber pressure of 500 psia. Note that, for these conditions, the surface recession was apparently caused solely by chemical reaction; no recession occurred in the chemically inert environment.

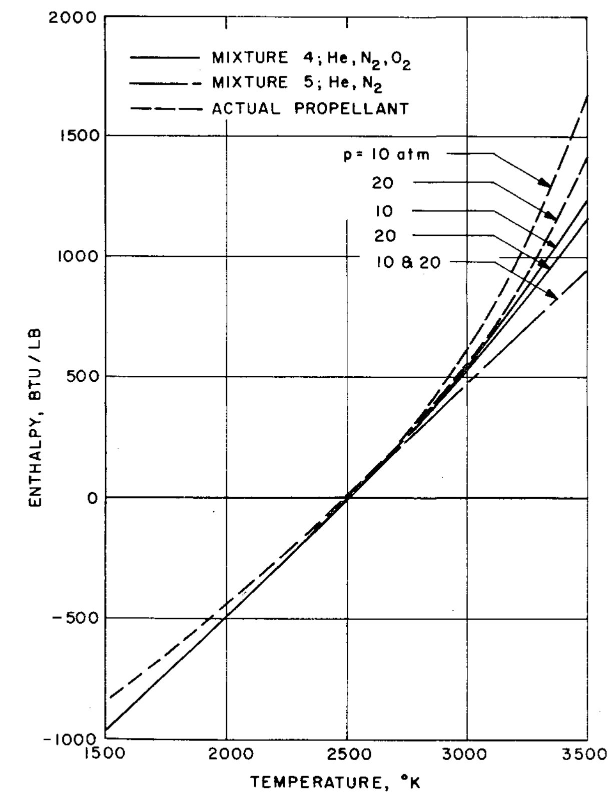

Fig. 2 Comparison of gas mixtures for simulation of a solid propellant (enthalpy base arbitrarily set at 2500°K and 20 atm).

The time history of surface recession and surface-recession rate as determined from the decay in plenum pressure (at constant mass-flow rate) are presented in Fig. 3 for a Graphitite GX graphite nozzle firing.[1]

Figure 1 shows an MX-4500 graphite cloth phenolic nozzle that has been cut for post-test inspection.[3] The dotted line indicates the pretest profile. This nozzle was fired at conditions that simulated the environment of the liquid propellant, nitrogen tetroxide-Aerozine.

V. Conclusion

The results presented in the foregoing indicate the capability and advantages of the arc-plasma generator for simulation of the rocket-nozzle environment. These results are typical and represent only a partial coverage of a few of the test firings to date. These tests have included a definitive investigation of the effects of chemical reaction and mechanical shear on material performance in addition to the investigation and development of the arc-plasma generator for rocket environment simulation. The anticipated future effort in the area of rocket-motor simulation includes the extension of the technique to still other propellant environments and the further investigation of chemical reactions with nozzle wall materials.

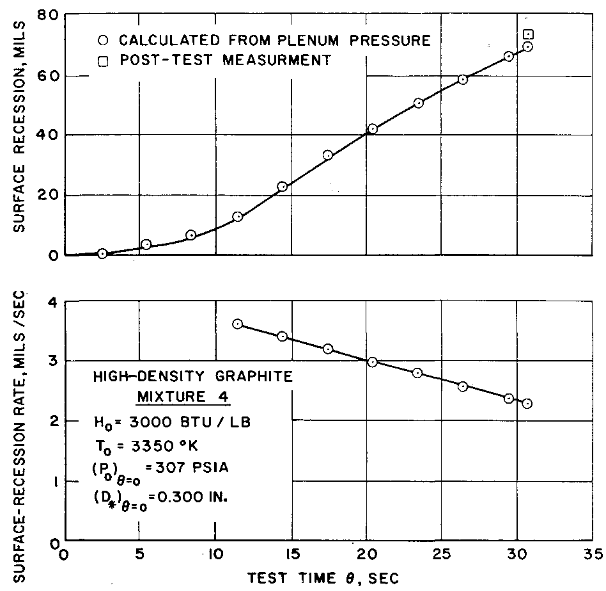

Fig. 3 Typical surface recession and surface-recession rate histories.

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[2] Flood, D. T., "A rocket-nozzle materials evaluation program using a 1-megawatt arc-plasma generator to simulate the throat environment of a typical solid-propellant rocket motor," Vidya Rept. 127 (February 27, 1964).

[3] Rindal, R. A., Flood, D. T., and Kendall, R. M., "Analytical and experimental study of ablation material for rocket engine application," Vidya Rept. 136, NASA Contract NAS7-218 (March 17, 1964).

Exploratory Hypersonic Boundary-Layer Transition Studies

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Nomenclature

h	=	distance normal to surface
M	=	Mach number
Pc	=	inviscid cone static pressure
P_t	=	pitot pressure
$P_{t,\infty}$	=	freestream stagnation pressure
R	=	Reynolds number
S_t	=	Stanton number
T	=	temperature
u	=	velocity
x	=	axial distance from cone tip
δ^*	=	displacement thickness
θ	=	momentum thickness

Subscripts

e	=	edge of boundary layer
w	=	wall
∞	=	freestream

An exploratory investigation to determine the feasibility of making boundary-layer transition studies at freestream Mach numbers on the order of 20 in the Langley 22-in. helium tunnel has indicated that 1) transition Reynolds numbers in the 30–40 $\times$ 10[6] range exist with local Mach numbers of about 15, 2) the transition Reynolds number at hypersonic speeds, at least for very slender axisymmetric configurations, is extremely sensitive to angle of attack, and 3) the thin-film resistance gage, which customarily is used to measure heat-transfer rates in impulse facilities such as the shock tunnel, is a useful device for measuring heat-transfer rates in relatively long running-time tunnels when heat-transfer rates are very low.

The model used in the exploratory investigations was a sharp-nosed (nose diameter = 0.003 in.) 5-ft-long cone with a base diameter of 6 in. (cone half-angle of 2.87°) and was constructed of Fiberglas. M_∞ varied with $P_{t,\infty}$ from a low of about 19 at 500 psi to a high of about 23 at 4000 psi. Model wall temperature was always about 540°R. Stagnation temperature was about 540°R during pitot-pressure measurements and about 820°R during heat-transfer measurements.

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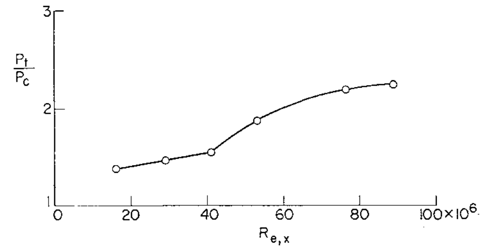

Fig. 1 Variation of surface pitot to wall static pressure ratio with Reynolds number at x = 58 in., $T_w/T_t \approx 1$.

The model initially was instrumented with surface thermocouples, as this was intended to be a temperature recovery model. However, the heat-transfer rates at M_∞ = 20 were so low that, for the run times available (about 40 sec), no useful data were obtained.

A surface pitot probe, 0.060 in. in diameter, was installed at x = 58 in., and data were taken at stagnation pressures ranging from 500 to 3600 psi. These pitot pressures have been ratioed to the inviscid cone static pressure and are shown in Fig. 1 plotted against $R_{e,x}$. Transition appears to initiate at a Reynolds number of approximately 40 $\times$ 10[6].

Additional information concerning the state of the boundary layer then was determined by a boundary-layer pitot-pressure survey at the same station. These data were reduced to velocity ratio, assuming constant total temperature and static pressure through the boundary layer, and they are presented in Fig. 2 along with the resultant δ^* and $R_{e,\theta}$.

The highest Reynolds number profile appears to be turbulent, though it is not certain that this represents a fully developed turbulent flow. The inner half of the lowest Reynolds number profile appears laminar; the outer half has the earmarks of turbulence. The variation in shape of the velocity profile with Reynolds number suggests that turbulence is initiated in the outer region of the boundary layer and moves in toward the wall as Reynolds number increases; that is, turbulence may exist in the boundary layer long before its effect is felt at the wall. This is in line with the observation of others with respect to the variation of the location of the critical layer with increasing Mach number (see, e.g., the discussion in Ref. 1, p. 42).

Prior to obtaining the data of Fig. 2, an attempt was made to obtain a pitot survey using eight tubes simultaneously during a run. The tubes were spaced 45° around the periphery of the model. There was a large amount of scatter in these data because of nonsymmetry of the boundary layer, although the model angle of attack was within ±0.2° of zero, as determined by 4 static pressure orifices in the model 90° apart. The nonsymmetry is attributed to the effects

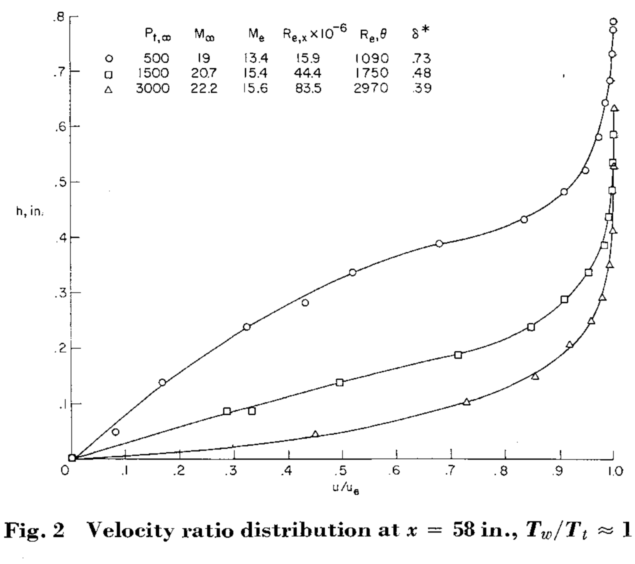

Fig. 2 Velocity ratio distribution at x = 58 in., $T_w/T_t \approx 1$.