Electron Bombardment Heating for Electrothermal Propulsion

K. E. STARNER* AND W. R. GRABOWSKY†
Aerospace Corporation, El Segundo, Calif.

The technique of high-voltage, low-current electron bombardment heating for the heat-transfer thrustor is investigated. The electron bombardment approach is shown to have inherent advantages for space application and to lack several restrictions found with resistance heating methods. A tantalum, electron bombardment-type gas heater was built to research thruster operation. Steady-state gas temperatures to 2700°K were produced with argon at a voltage-to-current ratio greater than 50. The device functioned successfully for a total operating time of approximately 50 hr. It was found that the electrical characteristics of the heater were adequately predicted by the Langmuir-Child law for space-charge limited emission. Possible causes of some electrical instabilities experienced during the tests are discussed.

I. Introduction

ELECTROTHERMAL propulsion engines produce thrust by the thermodynamic expansion of electrically heated gases through a supersonic nozzle. The heating may take place directly in the gas, as in an arc jet, or in an electrically heated material, over which the propellant is passed and heated by forced convection.

These engines cover the lower specific impulse range of the electric propulsion family, with the arc jets having a possible useful range with hydrogen propellant to 2500 sec and the heat-transfer types up to 1000 sec with present materials technology. Figure 1 gives possible frozen flow specific impulse and efficiency values for hydrogen as a function of stagnation temperature and pressure.[1, 2] Primary application for propulsors of this type will lie in satellite attitude control, orbit transfer, and station keeping.[3–5] Laboratory applications of these devices include high-temperature gas generators for materials testing and plasma sources for diagnostic studies and MHD experiments.

The arc jet engine, which utilizes ohmic heating directly in the propellant gas, has received considerable attention in the literature.[3] It has a potentially high specific impulse since the maximum enthalpy of the propellant is not necessarily limited by the structural temperature capability of the thruster housing. However, efficiency at high specific impulse is limited because some form of cooling is required to protect the device from the high-temperature flow.

In the heat-transfer type of propulsion device, maximum specific impulse is limited by the temperature capability of the heat exchanger. High thermal efficiencies with values approaching unity can be achieved by using regeneratively cooled thruster housings with adequate radiation shielding or insulation. Present designs of heat-transfer thrusters utilize resistance heating of thin-walled tungsten tubing through which the propellant is passed, or an insulating core wound with resistance wire over which the propellant flows.[6, 7] The resistance tube heater is a high-current, low-voltage device whereas the wire-coil design allows lower current operation at the same power level.

Presented as Preprint 64-673 at the AIAA Forth Electric Propulsion Conference, Philadelphia, Pa., August 31–September 2 1964; revision received August 7, 1964. This work was supported by the U. S. Air Force under Contract No. AF 04(695)-269. The authors wish to acknowledge the laboratory contributions of M. E. Gerard, who fabricated many of the components and assembled and operated the experimental equipment.

* Member of the Technical Staff, Aerodynamics and Propulsion Research Laboratory. Member AIAA.
† Head, Electrical Propulsion Section, Aerodynamics and Propulsion Research Laboratory. Member AIAA.

II. Electron Bombardment Heating

This paper introduces a different type of electrothermal propulsion engine having a high ratio of voltage to current ($E/I \gg 10$). The electrical heating process used is electron bombardment of a refractory metal heat exchanger by acceleration of electrons from an emitter to the collecting surface. Heating occurs at the heat exchanger surface by impingement of the electrons. Figure 2 shows the arrangement of emitter and heat exchanger of the experimental unit. This unit includes many of the characteristics of a possible space thruster to be described. Initially, the tungsten filament is heated resistively by current (I_F) produced by the filament power supply voltage (E_F). Thermionic emission of electrons occurs at elevated temperatures in accordance with Richardson's equation.[8] Voltage (E_B) is then applied between one end of the filament and the heat exchanger to accelerate the electrons to the positive collector. The current from the emitter to collector for space-charge limited emission (excess electrons available at emitter) is given by the Langmuir-Child law[8] ($I_B \propto V_B^{3/2}$). With the filament enclosed as shown in the figure, little or no I_F is required for emitter heating when the heat exchanger temperature approaches the required filament temperature, because thermal radiation provides adequate heating.

The electron bombardment heated type of heat-transfer thruster was first described in Ref. 9. This type of heating is well suited to space applications since the bombardment

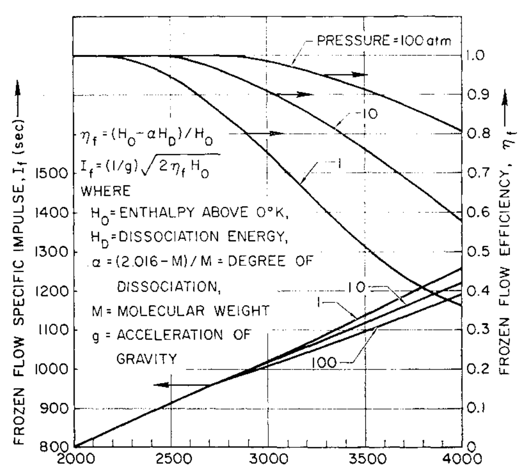

Fig. 1 Propellant characteristics of hydrogen.

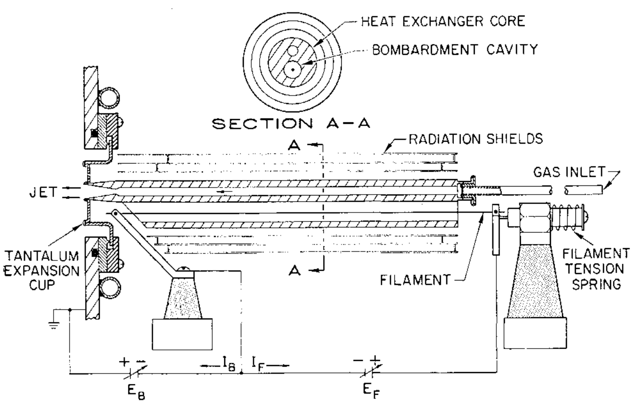

Fig. 2 Experimental heating unit.

chamber vacuum requirement is met inherently at altitudes of over 75 miles.

The electron bombardment approach permits control of axial heat flux distribution in the heat exchanger for minimum length through filament arrangement and bombardment cavity contouring. Since $E_B/I_B > 10$, small current leads can be employed to drain only minimal heat from the device. Also, the use of low-current values minimizes ohmic lead heating losses and eliminates high-current junctions. Because heating takes place only on the surface of the heat exchanger, the electrical characteristics will be independent of the electrical properties of the propellant.

Since neither the heating process nor the electrical characteristic is directly dependent on the wall thickness of the propellant flow channel, the device can be of rugged or "beefy" design. This feature permits a high pressure level that will aid in reducing frozen flow losses, allow compact heat exchanger design, and permit wide pressure variations for thrust control. With the thick wall design, heating will not be rendered critical by slight dimension changes due to erosion by the propellant, also axial heat conduction will reduce the length requirement. The heat-storage capacity can be used with stand-by power to provide short pulses of heated propellant for a control thruster application.

Numerous configurations of the heat exchanger and many types of emitters (indirectly heated emitters, emitting radiation shields, multiple filaments, etc.) can be envisioned within

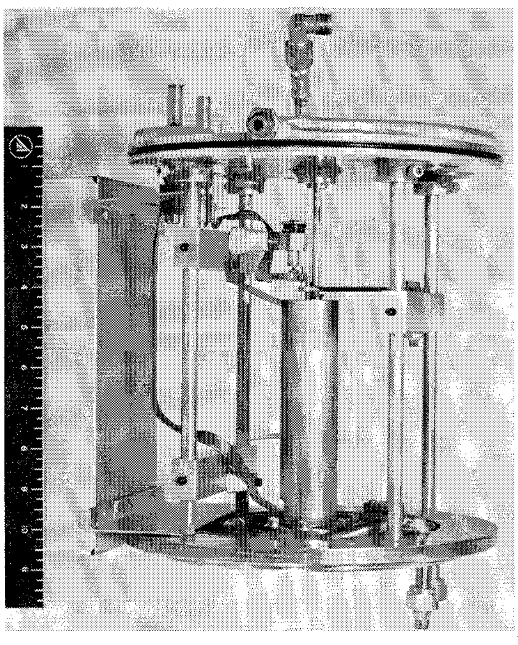

Fig. 3 Heater unit assembly.

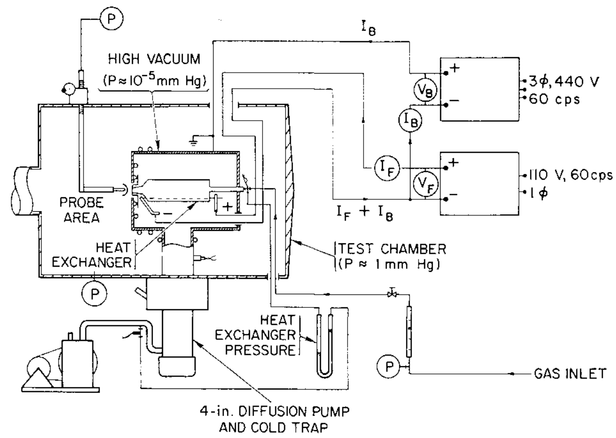

Fig. 4 Instrumentation schematic.

the electron bombardment concept. A quantitative discussion of possible thruster operating characteristic is contained in a later section of this paper. The following sections describe experimental studies with one basic thruster configuration suitable for laboratory experiments. The experiments were designed to yield information regarding possible electron bombardment heating limitations, prediction of electrical characteristics, and vacuum requirements for the bombarding process.

III. Equipment and Techniques

The experimental unit shown in Fig. 2 was a single flow channel model, employing a 0.156-in. diam, 5.5-in. long flow passage, and 0.125-in. radius (r_c) by 5.25-in. mean length (L_c) bombardment cavity. No effort was made to achieve high efficiency as no regenerative flow channels and only three radiation shields were employed. The filament was tungsten wire of 0.010-in. radius (r_f) and approximately 6-in. length. A throat diameter of 0.050 in. was used in the supersonic nozzle with an area ratio of 15. A photograph of the assembly prior to installation in the vacuum system is shown in Fig. 3, and the layout of the test facility is shown in Fig. 4.

Although tungsten is believed to be the best material for high-temperature heat exchangers, it was felt that much could be learned regarding the bombarding process, vacuum requirements, and emitter operation by constructing the laboratory unit of machinable tantalum (melting point 3269°K). In order to concentrate on the electron bombardment feature of the device, argon, known to be compatible with tantalum, was used as the test gas (standard grade argon was used). Filament and bombardment power was obtained from full-wave bridge rectifier power supplies.

Stagnation temperature of the argon at the exit of the heat exchanger was determined by use of the choked flow relation for constant mass flow through the nozzle throat:

$$T_{\text{hot}} = (P_{\text{hot}}/P_{\text{cold}})^2 T_{\text{cold}} \tag{1}$$

where P and T are the gas stagnation pressure and temperature, respectively, immediately upstream of the nozzle throat. As shown in Fig. 4, it was necessary to measure heat exchanger pressure in the flow channel at a distance of 10 in. from the nozzle location. Small errors introduced by the use of Eq. (1) as a result of pressure drop in the 10-in. length and the physical expansion of the nozzle throat at high temperatures are discussed in detail in Ref. 10. Mach number of the exit jet was determined by using a tantalum impact pressure probe.

Heat exchanger temperatures during the tests were monitored by use of an optical pyrometer focused on blackbody holes drilled on the core centerline. Holes were provided in

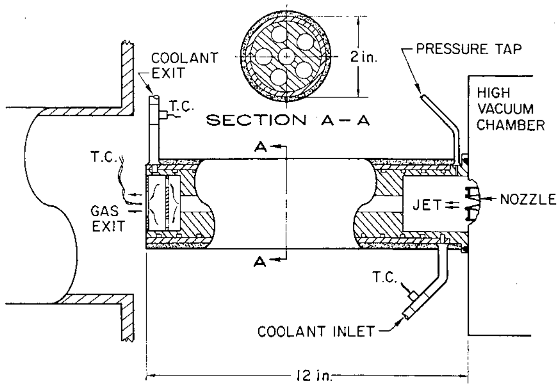

Fig. 5 Calorimeter installation.

the radiation shields to allow observation of the heat exchanger.

A steady-flow exhaust jet calorimeter was constructed to determine the energy remaining in the jet after expansion out the nozzle. Figure 5 shows the construction of this copper calorimeter.

IV. Experimental Results

Indicated argon temperatures upstream of the nozzle throat are plotted in Fig. 6 as a function of total power input (filament plus bombardment). Run 1 represents initial heating attempts that produced gas temperatures to 2000°K, at which point leaks in a heat exchanger weld developed, causing a poor bombarding vacuum. Normally, pressures in the bombardment chamber were kept below 10^{-5} mm Hg, but bombarding was possible at pressures as high as 10^{-4} mm.

Run 2 (Fig. 6) was made after leaks in the nozzle-cup joint were repaired. Temperatures to 2700°K at a flow rate of 4.85×10^{-2} g/sec were indicated by use of the choked throat equation. Heat exchanger pressures increased from 0.138 atm at room temperature to 0.412 atm at 2700°K. The difference in temperature levels at a given power input for runs 1 and 2 was caused by a reversal of polarity on the filament power supply between the runs. It was observed that the heat exchanger inlet was hotter than the exit for the first run. By reversing polarity to that shown in Fig. 2, the temperature gradient lessened and the nozzle end became the hottest, thus producing slightly higher temperature gas. During these runs, the heat exchanger temperature observed at the blackbody hole nearest the exit was a maximum of 300°K greater than the indicated exit gas temperature. Little systematic information was obtained with the pyrometer, as slight shifting of the heat exchanger and radiation shields often caused misalignment of the sighting holes.

Run 3 of Fig. 6 was made utilizing a.c. filament power with the bombardment current lead attached to a center tapped transformer. Maximum gas temperatures approached those of run 2, but slightly more power was required. It was observed that at high temperatures the heat exchanger temperature was symmetrical about the midpoint, but that the center was much hotter than the ends.

Only a small portion of the input power was actually absorbed in the argon so that most of it was lost by either radiation or conduction to the chamber walls. If most of the heat input loss were due to radiation (with the heat exchanger near exit gas temperature), then the curves of Fig. 6 would have a slope value near 4. If heat loss were by conduction through hardware alone, then the slope would be equal to about 1. The two high-temperature runs show slopes of 2.9 and 3.5, indicating a great percentage of heat loss by radiation.

Figure 7 compares the change of gas temperature with total power input for two flow rates of run 2. The result of more than doubling the flow rate is to decrease the exit gas temperature by less than 15%. Heat exchanger pressures ranged from 0.346 to 0.953 atm with the higher mass flow. Laminar flow prevailed for all tests as an Re_D (Reynolds' number) value of 1600 was the maximum attained in the heat exchanger. A heat-transfer analysis for the flow channel using the correlations of Ref. 11 indicates that the maximum gas temperatures attained are quite reasonable for a heat exchanger operating in the 3000°K temperature range.

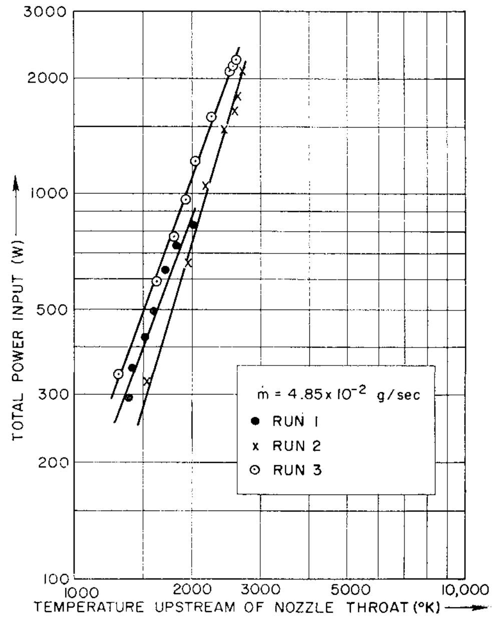

Fig. 6 Power input as a function of gas temperature.

Electrical characteristics for the three runs are compared in Fig. 8 with theoretical results. Most of the points fall near the $\frac{3}{2}$ power voltage line for theoretical space-charge limited emission. This equation strictly applies only for zero initial electron velocities and equipotential surfaces, but gives a good indication of expected results when the filament voltage drop is small compared to the bombarding potential. Reference 10 contains a discussion and an approximate analysis of first-order effects relating applied voltages E_B and E_F to bombardment current distribution and local filament current.

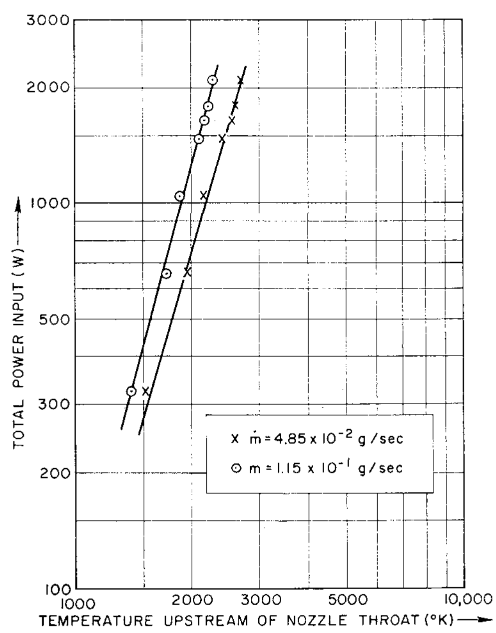

Fig. 7 Power input as a function of gas temperature and flow rate.

The filament current was maintained at the lowest value possible for apparent space-charge limited operation. This point was determined by noting at what value of filament current the bombarding current was influenced appreciably. As predicted, below this minimum value of filament current, the bombardment current would not remain constant at fixed voltage, but would slowly increase with time as the heat exchanger became hotter. The rise of bombarding current would cease when and if the filament became heated enough by radiation to provide sufficient electrons for space-charge limited emission. The previously mentioned axial temperature gradient that reversed with a change of polarity may have been caused by the fact that the ends of the filament were exposed and also that they carried unequal amounts of current (Fig. 2). Thus, if the negative end were just barely space-charge limited, the positive end might be slightly temperature limited due to radiation losses and lower resistive heating. This would lead to a deficiency of electrons in this area with a corresponding lower heating rate. In addition, the greatest potential difference between filament and heat exchanger occurs at the negative terminal.

Reference 10 contains details regarding filament power during the tests. In general, the required filament power decreased with total power input, and none was required at total power levels of about 2 kw. Above that point, the filament power supply was turned off completely and all control was maintained from the high-voltage supply. In this regime of operation, it was found necessary to adjust the supply voltage constantly to prevent excessive current flow. Current "run-away" time was on the order of several seconds, however, so that slight manual voltage adjustments (± 25 v) allowed determination of the steady-state gas temperature data shown for runs 2 and 3. As long as voltage adjustment was provided, the operating time or temperature level of the unit was not restricted by this instability. Several factors could have been combining to cause this phenomenon: As mentioned earlier, the filament was not of uniform temperature, which could have led to some portions of the filament experiencing a temperature-limited condition and others a space-charge limit. If such a condition existed, the temperature-limited portion of the emitter would tend to produce added bombardment current as it became hotter by radiation. Under investigation but not yet determined are the effects of the magnetic field (due to current in the filament) on the space-charge limited operation. Finally, any metal vaporization in the bombarding cavity would influence the electrical characteristics.

A power limit (increasing I_B and decreasing E_B) appears to exist above 2 kw, as shown by Fig. 8, probably as a result of the high-temperature vaporization. Appreciable metal vapor within the bombarding cavity would have the effect of lowering the effective cavity resistance at a given voltage. Increased heating would produce more vaporization with additional lowering of resistance. In this case, the onset of the power-limiting operation indicates impending heat exchanger burnout. The unit was disassembled after approximately 50 hr of operation, including numerous thermal cycles, and was found to be sagging by $\frac{3}{32}$ in. at the midpoint. There was also evidence of local heat exchanger melting on the roof of the bombardment cavity.

The tantalum impact probe indicated a Mach number of 5 for the initial nozzle with an area ratio of 15. Welding repairs to the nozzle lip following run 1 required reducing the exit area considerably so that an area ratio of about 6.5 remained. Impact probe readings yielded Mach numbers of 3.5 to 4 for this nozzle. At gas temperatures above 2200°K, the indicated impact pressure was quite erratic, and no reliable Mach number data were obtained in this region.

Calorimeter tests were run with and without gas flow, at constant power input, to isolate the radiation contribution to the calorimeter during gas flow. It had been observed previously that little change in visible radiation occurred from the nozzle and cup as a function of gas flow. As seen in Fig. 2, a heat conduction path exists from the nozzle through the cup to the cooled knife-edge and O-ring seal. Thus, because of the cool walls, some convection heat loss from the gas during expansion in the nozzle was expected. The calorimeter tests showed that, based on indicated temperatures upstream of the throat, the gas at 850°K cooled by 110°K, and at 1100°K the cooling increased to 240°K for a flow rate of 0.115 g/sec. Nozzle radiation amounted to 8 and 14% of the original gas energies, respectively, for the two tests. Although it was necessary to disassemble the unit before completing additional calorimeter tests, the limited data obtained indicate that severe nozzle heat losses must be avoided if high exhaust jet temperatures are to be attained in future units. Higher power levels would help to minimize relative nozzle losses in laboratory devices. Fortunately, highly cooled vacuum seals will not be required for space applications, thus alleviating the nozzle heat conduction problem.

V. Application to Thruster Design

The experimental results of this study indicate that the design concept shown in Fig. 9 is applicable to an electron bombardment heated thruster. This design features small-diameter, multiple-flow passages, allowing the highest possible heat flux to the propellant with a minimum of radiation surface area. The physical size of the thruster is determined by the heat-transfer area required to heat the desired mass flow of gas. For a channel of diameter D at constant temperature T_w,

$$(C_p/k)dT/(T_w - T) = Nu(\pi/\dot{m})dL \qquad (2)$$

where $\dot{m}$ is the mass flow rate, C_p the specific heat, Nu the local Nusselt number, k the gas thermal conductivity, and T the local bulk temperature of the gas.

It can be shown from Ref. 2 that, for hydrogen in the pressure range of 0.1 to 100 atm and temperature from 300° to

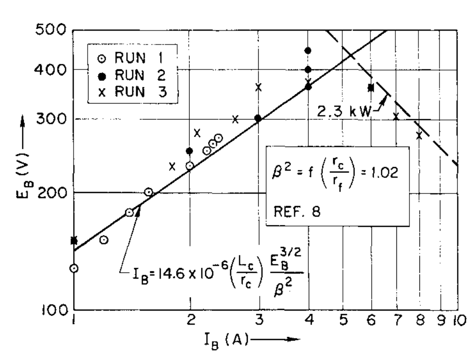

Fig. 8 Electrical characteristics.

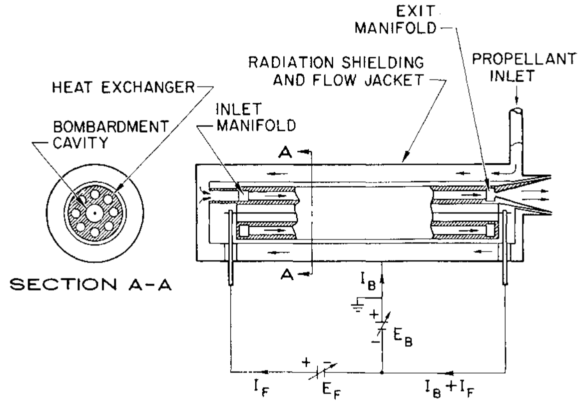

Fig. 9 Possible space engine concept.

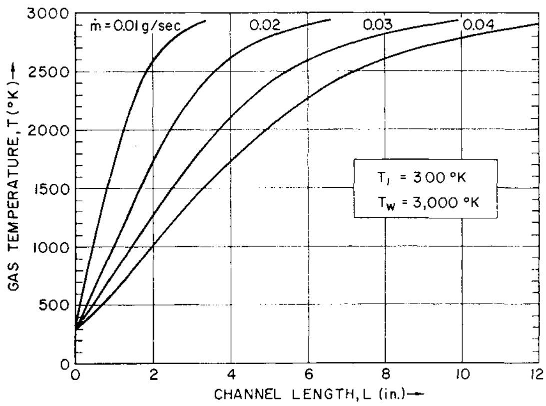

Fig. 10 Laminar heat transfer to hydrogen.

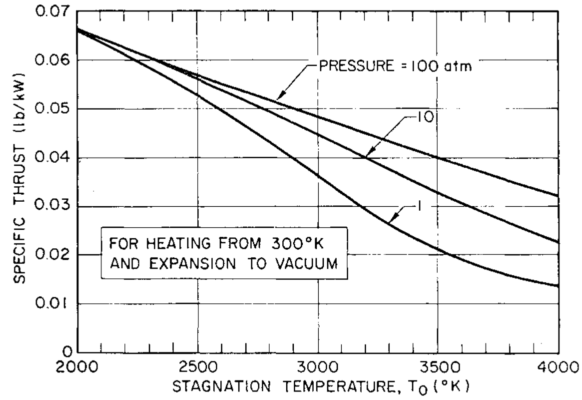

Fig. 11 Specific thrust of hydrogen.

3000°K, the C_p/k (sec-cm/g) can be approximated within $\pm 15\%$ by a function of T (°K):

$$C_p/k = 12 \times 10^4/T^{1/2} \tag{3}$$

For forced convection heat transfer in fully developed laminar flow with constant wall temperature, the Nusselt number is equal to 3.65 (Ref. 11). Conservative values of heat-transfer area required will be obtained by evaluating the fluid properties [Eq. (3)] at the local bulk temperature. By combining Eqs. (2) and (3) and integrating, an expression for required length L (cm) is found in terms of entrance and exit bulk temperatures T_1 and T (°K), wall temperature T_w (°K), and flow rate m (g/sec):

$$L = \frac{\dot{m}}{3.65\pi} \frac{12 \times 10^4}{T_w^{1/2}} \ln\left\{\frac{1 + (T/T_w)^{1/2}}{1 - (T/T_w)^{1/2}} \times \left[\frac{1 - (T_1/T_w)^{1/2}}{1 + (T_1/T_w)^{1/2}}\right]\right\} \tag{4}$$

Note that the length required to heat a given mass flow to temperature T is independent of channel diameter. This result is shown in Fig. 10 for a wall temperature of 3000°K and various flow rates. It can be seen that 0.02 g/sec of hydrogen can be heated to near 3000°K in a single, 6-in.-long heat exchanger channel. At a pressure of 10 atm, Fig. 1 indicates a thrust level ($I_s\dot{m}$) of 4.5×10^{-2} lb/channel. A 10-channel unit of the type shown in Fig. 9, 6 in. long, therefore would ideally produce a thrust of 0.45 lb. The net power to the propellant (at $T_1 = 300$°K) is found from Fig. 11 to equal 10 kw at this thrust level, stagnation temperature, and heat exchanger pressure. With individual flow channels of about 0.1-in. diam, the velocity and pressure drop will be low for this flow rate ($Re_D < 10^3$). Present experience indicates that a well-shielded heat exchanger core of these dimensions would experience total heat losses in the range of 1 to 2 kw. It should be possible to capture most of this heat loss (excluding nozzle radiation) in a relatively low-temperature (below 1000°K) regenerative flow jacket, giving a thermal efficiency η_t of over 0.9.

VI. Conclusions

1) Electron bombardment heating appears feasible as a high-voltage, low-current heating technique offering certain advantages over resistance heating methods for electrothermal propulsion and laboratory high-temperature gas-generating facilities.

2) Experimental studies have shown that electron bombardment heating is possible in a cavity with tantalum wall temperatures near 3000°K at environment pressures below 10^{-4} mm Hg.

3) Electrical characteristics of the experimental unit can be predicted adequately over most of the operating range by the Langmuir-Child law for space-charge limited emission. An upper limit on the controlled electrical power which can be transferred to the device may occur. This power limit is believed to represent the tantalum vaporization point of the bombardment cavity walls.

4) Tantalum appears to be satisfactory as a heat exchanger material for producing high-temperature argon in laboratory units.

5) Additional studies are desirable in the areas of optimum filament sizing and arrangement, influence of filament magnetic field on space-charge limited emission, possible input power limitations, and bombardment cavity contouring for optimum power distribution.

References

[1] Jack, J. R., "Theoretical performance of propellants suitable for electrothermal jet engines," AIAA J. **31**, 1685–1689 (1961).

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

[3] Page, J. R., "Current status and prospects of electrothermal propulsion," ARS Preprint 2649-62 (November 1962).

[4] Marchand, F. E., Arthur, P. D., and McCaughey, O. J., "Results of satellite raising and orbit transfer studies for low power electrothermal arcjet propulsion systems," AIAA Preprint 63010 (March 1963).

[5] Boucher, R., "Electric propulsion for control of stationary satellites," AIAA Preprint 63009 (March 1963).

[6] Howard, J. M., "The resistojet," ARS J. **32**, 961–962 (1962).

[7] Jack, J. R. and Spisz, E. W., "NASA research on resistance-heated hydrogen jets," AIAA Preprint 63023 (March 1963).

[8] Millman, J. and Seely, S., *Electronics* (McGraw-Hill Book Co., Inc., New York, 1951), pp. 106, 192–194.

[9] Grabowsky, W. R., Mirels, H., and Starner, K. E., "Electrothermal propulsor," Aerospace Corp. Invention Disclosure Docket 70 (October 1962).

[10] Starner, K. E. and Grabowsky, W. R., "Electron bombardment heating for electrothermal propulsion," Aerospace Corp. ATN-64(9227)-3 (February 1964).

[11] Eckert, E. R. G. and Drake, R. M., Jr., *Heat and Mass Transfer* (McGraw-Hill Book Co., Inc., New York, 1959), 2nd ed., pp. 190–198.