DIAGNOSTIC STUDIES OF A LOW DENSITY, ARC HEATED WIND TUNNEL STREAM

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ABSTRACT

Investigations were made of the local flow properties of an arc heated low density hypersonic argon jet by means of several different probe techniques. These included impact and static pressure probes, free molecular flow and stagnation point heat transfer probes, and a stagnation point Langmuir double probe. Results presented include axial and radial surveys of impact pressure and computed stagnation temperature, and radial surveys of ion density and electron temperature, and electrical conductivity computed from these latter quantities. It is pointed out that interpretation is difficult of measurements with continuum flow probes, such as the stagnation point heat transfer and static pressure probes, because the probes are subject to important viscous effects, and that some of these effects are associated with the relatively unexplored transition flow regime of rarefied gas dynamics. Further development of high temperature free molecular flow probes and of nonaerodynamic methods for flow measurements is desirable.

INTRODUCTION

Studies of hypersonic flow of arc heated argon from a simple conical nozzle have been made at the University of California, Berkeley, since 1959, under the sponsorship of the U. S. Air Force Office of Scientific Research. Some of the early results have been reported in Refs. 1, 2 and 3.

Presented at ARS International Hypersonics Conference, Cambridge, Massachusetts, August 16-18, 1961; this work was supported by the Air Force Office of Scientific Research under Contract AF49(638)502.

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The arc heater employed dissipates only about 12 kw, but since the flow rates of argon are small (typically 1 gm/sec), reasonably high stagnation temperatures in the range of 5000 to 9000 K are obtained. The arc chamber pressure is usually of the order of one-half an atmosphere.

The Mach number and Reynolds number ranges obtained are approximately $4.5 < M_\infty < 13$ and $200 < Re_\infty / cm < 650$ (or $10 < Re_2/cm < 100$), where Re_∞ is based on free stream conditions, and Re_2 on conditions behind a normal shock wave. The mean free path of neutral particles in the free stream varies from about 10^{-2} to 10^{-1} cm. These conditions place many of our investigations in the interesting and difficult transition-flow regime of rarefied gas dynamics.

Rapid expansion from the arc chamber leaves the stream ionized to a degree (about 0.3 to 0.4%) which provides a useful luminescence, the patterns of which can be strongly modified by a modest transverse magnetic field (300 to 2000 gauss). The fractional energy, mass and momentum fluxes associated with the charges particles, are small enough to play a negligible role in overall balances, at least in the absence of a magnetic field, so that it is permissible for many aerodynamic purposes to consider the argon simply as a high total enthalpy perfect monatomic gas.

The aim of the current experiments has been to establish methods for the local measurement of the dynamic, thermal and electrical state of the gas, and to apply these towards assessment and improvement of the spatial and temporal uniformity of the flow. To date the most useful data has been obtained from impact pressure probes, stagnation point heat transfer probes and stagnation point Langmuir probes. Some corroborating overall observations have been made with microwave interferometry and optical spectroscopy. Heat transfer and Langmuir probes which will operate in free molecule flow, and a time-of-flight anemometer are under development, but have to date have yielded only preliminary results.

Although the present instrumentation problems are not yet completely resolved, present estimates of Mach and Reynolds number ranges, stagnation temperature levels, electrical conductivity and flow uniformity indicate that the facility will soon be sufficiently well controlled and calibrated to be useful for research in hypersonic rarefied gas dynamics and a limited regime of magnetogasdynamics.

ARC HEATER AND WIND TUNNEL

The arc heater, which is shown schematically in Fig. 1, utilizes a pointed tungsten cathode and a copper anode in the shape of a convergent nozzle. Argon is introduced tangentially in the region around the cathode, is swept through the arc region in the throat of the anode, through a short plenum chamber, and finally through the converging-diverging nozzle, emerging as a free jet in the test chamber. A more detailed description of the arc heater is given in Refs. 1 and 2.

The test chamber is arranged to form a second leg of the low density wind tunnel facility at the University of California, and can frequently be operated simultaneously with the first leg, sufficient pumping capacity for both being provided by two five-stage steam ejector sets, arranged in parallel. In most major respects the test chamber and traverse mechanism for mounting and positioning the probes is similar in design to the low temperature leg of the Low Density Wind Tunnel. The latter has been described in detail in Ref. 4.

All parts of the equipment which are subject to high heating rates are water cooled, with particular attention paid to the anode and nozzle throat regions.

To counteract a natural tendency of the arc to run off-center in the anode channel, and to rotate continuously the point of attachment of the arc at the anode, a coil of about 1000 turns was wound in the anode cooling water passage. This coil provides an axial field of a few hundred gauss in the anode throat region when supplied with about 0.5 to 1 amp d-c. Magnetic fields of this magnitude are observed to have quite striking and varied effects on the flow achieved in the test chamber.

The supersonic nozzle has a copper throat section, which carries the expansion to a Mach number near 4.7 in a cone of 8 ½-deg half angle. This is followed by a brass extension cone frustrum of 13 ½-deg half angle, which continues the expansion to $M \approx 12.7$. The change of cone angles was chosen to offset the boundary layer growth in order to produce an effective nozzle shape which is approximately a simple 7 ½ -deg half angle. A second nozzle, which is a simple 7½ -deg half angle cone without boundary layer allowance, has been used, but provides almost no inviscid core of flow at its exit Mach number of 9.

PRESSURE PROBES

Sketches of the impact and static pressure probes employed are shown in Fig. 2. They are simply constructed from copper tubing and are water cooled. In all the probes the cooling is accomplished in essentially the same fashion. Into the outer copper tube are fitted two or more smaller copper tubes. One of these smaller inner tubes is open-ended for the entrance of the water which then returns in the spaces between the inner and outer tubes. The other inner tube or tubes are sealed from the water jacket at their ends, and thus provide a water-free passage from the probe tip to base for either pressure measurements or electrical leads, or both. The fabrication of such probes is not very difficult. The tip on the static probe is a 4 to 1 circular arc tangent ogive.

Four axial surveys of impact pressure P_i in the double-cone nozzle are shown in Fig. 3. Two of these surveys were taken in the free jet downstream of the throat section, with the extension removed for convenience. The other two were taken inside the extension. These surveys exhibit no indications of oblique shocks in the nozzle, over the range surveyed. Figure 4 shows radial surveys at the exits of the throat section and of the extension. The existence of an inviscid core of reasonably uniform impact pressure at both stations is indicated. The presence of a uniform core and the apparent absence of shock waves are two pieces of evidence which serve as preliminary justification for an isentropic flow assumption. Using this assumption the present first estimates of Mach number are obtained from the ratio of impact pressure to plenum chamber (stagnation) pressure. These Mach number estimates are those which have been cited in the previous section.

The profiles shown at the exit of the extension exhibit the flow asymmetries and lack of reproducibility with which one is bothered in the absence of a magnetic field in the anode. The magnetic field reduces these, but data is as yet insufficient to claim that this device is a complete success.

Impact probes can be subject to two systematic errors in low density, high temperature flow; one is a low Reynolds number effect and the other is a thermal creep effect due to the temperature gradient along the pressure line connecting orifice and manometer. The latter effect is negligible for the size tubing employed and the pressures measured with the impact probe.

Viscous effects in the present ranges of M and Re can presently only be guessed by extrapolation of data obtained at lower Mach numbers in air (Refs. 5, 6 and 7). The Reynolds number of the present probe, based on probe and conditions behind a normal shock, may range from 3 to 30 in the surveys shown. These Reynolds numbers are very low, but it may still be surmised that for the open-end probe geometry employed deviations between measured and inviscid impact pressures are probably less than 10%. At least a partial experimental investigation of these viscous effects is feasible in the arc heated wind tunnel, but it would not be possible for the investigators to employ probes large enough so that viscous effects are negligible.

The ogive "static pressure" probe has been used only for qualitative purposes, since it is subject to very large and unknown viscous interaction effects. This can readily be seen from the estimates of hypersonic interaction parameter $\chi = M^3/\sqrt{Re/cm}$ shown in Fig. 5. (The wiggly lines on Fig. 5 indicate exit Mach number conditions obtained for the four nozzle configurations tested so far.) Although measurements with the ogive probe are only qualitative, it has been shown with the use of this probe that the static pressure at the exit at Mach 12.7 almost certainly is much higher at the nozzle wall than on the centerline. Even the measured centerline ogive probe pressures were some 40% lower than nozzle wall pressures, under conditions when nozzle wall and test chamber pressures were carefully balanced, and the ogive probe pressures must be considerably higher than the true static pressure because of viscous interaction effects. Measurements with the ogive have also shown that radial asymmetries in impact pressure and stagnation point heat transfer distributions are accompanied by large asymmetries in static pressure.

In future attempts to determine static pressure consideration will be given to the use of a slender cone in the upstream regions of the flow where the weak-interaction regime is approached, taking account of viscous effects by weak-interaction theory. For measurements of the static pressure in the strong interaction region downstream it may be possible to use a "3/4-power" body, and apply the theory given by Yasuhara (Ref. 8) for interpretation of the viscous effects. The important advantage of the "3/4-power" geometry would be that exact similar solutions to the boundary layer equations may still be obtained with the transverse curvature effect included. For the high - χ flow conditions produced in the arc heated wind tunnel, the effect of transverse curvature on the induced pressure for a slender axisymmetric body shape appears to be quite important, and is outside the range of applicability of the Probstein-Elliott perturbation theory (Ref. 9).

HEAT TRANSFER PROBES

The stagnation point heat transfer and Langmuir probes employed in the present investigations are shown in Fig. 6. The metal parts are copper, the cross-hatched material is boron nitride. The boron nitride provides excellent electrical insulation even at elevated probe temperatures (1000 K at the most severe conditions encountered) but is nevertheless a fair thermal conductor, which is necessary since the probe end is conduction cooled. The heat received at the front face of the central copper rod in the heat transfer gage establishes a virtually constant temperature gradient along the rod. The gradient, which varies from about 3 to 60 deg C/cm, is measured by an array of five equally spaced no. 30 gage constantan wires, peened into the surface of the copper rod to constitute a sequence of thermocouple junctions. An attempt was made to proportion the probe so that the temperatures along the outer shell of the probe approximated those directly opposite on the rod, thus minimizing lateral heat losses. The readings of two thermocouples on the inner surface of the outer shell indicated that this design was quite successful when the probe was in a uniform flow region.

From the measured heat transfer rates q and stagnation point wall temperatures T_w (obtained by extrapolating the linear temperature gradient along the central rod to the stagnation point) initial estimates were obtained of the gas stagnation temperature T_o . This was done by working backwards through the boundary layer theory of Fay and Riddell (Ref. 10), as expressed in Fig. 7. The stagnation point velocity gradient was estimated from the formula given by Probstein (Ref. 11). According to boundary layer theory (as shown in Fig. 7) for a given probe diam D , the quotient of the stagnation point heat transfer q and the square root of the impact pressure p_i is only a weak function of the wall temperature T_w and depends almost entirely on the value of the stagnation temperature T_o , so that q/p_i should be essentially constant throughout the flow if the flow is adiabatic. It was observed, however, that this procedure when applied to measurements made along the centerline of the flow yielded stagnation temperatures which rose systematically from about 7000 K at the exit of the Mach 12.7 extension to about 10,000 K at the exit of the throat block, which at first glance would indicate that the flow was not adiabatic. However, radial heat transfer surveys taken with both stagnation point and free molecule heat transfer probes did not seem to suggest overlapping thermal boundary layers, as shown in Fig. 8. (The free molecule probe, which is being developed as a Masters thesis problem by R. Miura, is shown in Fig. 9.)

The possibility that there was significant radiant heat transfer to the probe from the arc region was considered, and it was concluded that it was negligible. However, it was observed that the use of boundary layer theory in the low Reynolds numbers range (about 7 to 70 based on conditions behind a normal shock and radius of the flat front face) was overly optimistic. Recent heat transfer data by Ferri and Zakkay (Ref. 12) covering an overlapping Reynolds number range for stagnation point flow about a hemispherical body in air and theoretical work such as that of Levinsky and Yoshihara (Ref. 13) confirm this suspicion, as does a simple log-log plot of measured heat transfer vs. impact pressure. This plot, shown in Fig. 10, exhibits a nearly constant slope of about 0.71 instead of the slope of 0.5 of boundary layer theory (assuming T_0 and T_w constant), and appears to be roughly consistent with the trends of the Ferri-Zakkay data and some unpublished low density wind tunnel data obtained by Hickman at the University of California, Berkeley, although the different specific heats ratios and body shapes prevent more direct comparison. A detailed investigation of heat transfer in the low Reynolds number range appears to be another fruitful area of future research in the arc heated wind tunnel facility.

All of the heat transfer data was obtained with the central copper rod electrically isolated from ground, so that the probe drew no net current from the ionized gas. If the potential of the probe is varied, particularly to positive values, the heat transfer can be strongly influenced, as shown in Fig. 11. The effect of variable negative probe potential is seen to be very small. In fig. 12, the heat transfer increase due to collection of electrons is correlated with the current drawn. This data was obtained in 1959, at Mach 6, using an uncooled probe in a transient heat-sink calorimeter experiment (Ref. 2).

LANGMUIR PROBE DATA

Characteristic curves of the stagnation point Langmuir double-probe (configuration B) obtained at $M \approx 12.7$ are shown in Fig. 13. Each set of points was obtained by discrete readings of an ammeter and voltmeter, a time-consuming procedure which introduced the apparent hysteresis effects. These effects have been subsequently eliminated by use of an X-Y plotting potentiometer.

The double-probe method, originally due to Johnson and Malter (Ref. 14) avoids a difficulty which has been encountered in the use of single-Langmuir probes, namely, in the determination of electron temperature the drawing from the plasma of electron currents was so large that they indicated a disturbance of the plasma which would invalidate the probe theory. With the double-probe, the net current from the plasma is always zero, and the consequent disturbance is localized in the immediate vicinity of the probe surfaces, as required by the theory.

From the double-probe characteristic curves have been deduced electron temperatures, ion densities and electrical conductivities by classical probe theory, as applied to a stagnation point flow by Talbot (Ref. 15). These values are shown in Fig. 14. The electron temperatures given are characteristic of the gas behind the normal shock, whereas the ion densities are free stream values, following from the probe data on the assumption of unchanging degree of ionization through the shock wave, shock layer and boundary layer. The electrical conductivities are based on the values of n_i and T_e shown, and are computed by adding the resistivity due to electron-neutral collisions to that due to electron-ion collisions in the usual way (see Ref. 16). Some of the scatter and apparent variation in the values of n_i and T_e is undoubtedly due to the errors introduced by the present point by point method of obtaining the probe characteristic curves, although the peak in ionization 0.2 in. off the nozzle centerline is real, and is associated with the lack of axial symmetry of the flow in these particular tests. It is expected that more significant data will be forthcoming from the X-Y plotter technique now being employed. Despite the scatter and hysteresis in the probe results it is encouraging to note that a well-defined ion saturation current limit was achieved, and that the initial slopes of the characteristic curves are very close to the 1:1 value on a log-log plot (Fig. 13), as required by the theory for a Boltzmann distribution of electron energies.

In earlier measurements at Mach 6, reported in Ref. 3, the free stream ion densities measured by a single stagnation point Langmuir probe were well corroborated by microwave interferometer measurements, when the conditions for applicability of the probe theory were met (see Ref. 14). These conditions were also reasonably well achieved in the present experiments, so that it seems fair to assume that the ion densities reported here are correct within experimental error. An independent measurement of the electron temperature has not as yet been obtained, but plans are under way to attempt measurements by optical spectrographic means. There is an interesting point concerning the value of the electron temperature. As has been noted, the electron temperature measured by the stagnation point Langmuir probe is characteristic of the gas behind the shock wave (in fact, somewhere within the stagnation point boundary layer). Now, the electrons, because of their very high random thermal speeds, comprise a gas in low subsonic flow so that they are not shock heated in passing through the bow shock wave. The electron gas is, however, compressed by the same amount that the ion density rises across the shock wave, because plasma neutrality is preserved. If thermal conduction in the electron gas is negligible, then the compression across the shock will be essentially isentropic, according to the law $n_e T_e^{3/2}$ const, and it can be shown that the electrical conductivity in the free stream would have approximately one-fourth the values plotted in Fig. 13. If, on the other hand, the effects of heat conduction in the electron gas were very large, then the electrons would be compressed in nearly an isothermal fashion, and the electrical conductivities computed would be appropriate to the free stream. The question of how the electrons are compressed in the shock wave is now being investigated theoretically. The use of a free molecule Langmuir double-probe would be one way to measure the local free stream electron temperature, but this requires an electrostatic probe theory which includes the effects of convection. A completely satisfactory theory of this kind is not yet available, although interesting work along these lines has been done by Clayden (Ref. 17).

CONCLUSIONS

From the data presented herein it can be concluded that:

1) The arc heated low density tunnel potentially provides a very interesting environment for testing in the transition flow regime of rarefied gas dynamics, although improvements in flow uniformity are still required.

2) The stream produced possesses a sufficient electrical conductivity to permit some experimentation in magnetogasdynamics.

3) Precise interpretation of many of the present probe measurements is hindered by the lack of a firm link to experimental results at higher densities, and by the paucity of well-established improvements on conventional boundary layer theory. This suggests the expanded use of probes which operate in free molecule flow, and of entirely nonaerodynamic techniques.

ACKNOWLEDGMENT

The authors are pleased to acknowledge the able assistance of C. L. Brundin and D. Otis in carrying out these experiments.

REFERENCES

1 Lai, W., Gustavson, J. and Talbot, L., "Design Considerations and initial evaluation of Model B Plasma Generator," Univ. of Calif. Engng. Project Rep. HE-150-161, Sept. 1958.

2 Brundin, C. L., Talbot, L. and Sherman, F. S., "Flow Studies in an Arc Heated Low Density Supersonic Wind Tunnel," Univ. of Calif. Engng. Project Rep. HE-150-181, April 1960.

3 Talbot, L., Katz, J. E. and Brundin, C. L., "A Comparison Between Langmuir Probe and Microwave Electron Density Measurements in an Arc Heated Low Density Supersonic Wind Tunnel," Univ. of Calif. Engng. Project Rep. HE-150-186, Jan. 1961.

4 Maslach, G. J. and Sherman, F. S., "Design & Testing of an Axi-Symmetric Hypersonic Nozzle for a Low Density Wind Tunnel," Univ. of Calif. Engng. Project Rep. HE-150-134, Feb. 1956.

5 Sherman, F. S., "New Experiments on Impact-Pressure Interpretation in Supersonic and Subsonic Rarefied Air Streams," NACA TN-2995, Sep. 1953.

6 Matthews, M. L., "An Experimental Investigation of Viscous Effects on Static and Impact Pressure Probes in Hypersonic Flow," GALCIT Memo no. 44, June 1958.

7 Enkenhus, K. R., "Pressure Probes at Very Low Density," Inst. of Aerophysics, Univ. of Toronto, UTIA Rep. no. 43, Jan. 1957.

8 Yasuhara, M., "An Exact Approach to the Hypersonic Viscous Flow Past a Slender Body of Revolution," Proc. 6th Japan Nat. Congress Appl. Mech., Kyoto, Japan, 1957, pp. 291-294; see also Hayes, W. D. and Probstein, R. F., "Hypersonic Flow Theory," Academic Press, N. Y., 1959.

9 Probstein R. F. and Elliott, D., "The Transverse Curvature Effect in Compressible Axially Symmetric Laminar Boundary-Layer Flow," J. Aero. Sci., vol. 23, 1956, pp. 208-224, 236.

10 Fay, J. A. and Riddell F. R., "Theory of Stagnation Point Heat Transfer in Dissociated Air," J. Aero. Sci., vol. 25, 1958, pp. 73-85, 121.

11 Probstein, R. F., "Inviscid Flow in the Stagnation Point Region of Very Blunt-Nosed Bodies at Hypersonic Flight Speeds," WADC Tech. Note no. 56-395, Brown Univ., Providence, R. I., 1956.

12 Ferri, A. and Zakkay, V., "Measurements of Stagnation Point Heat Transfer at Low Reynolds Numbers," PIBAL Aero. Research Lab. Rep. ARL 38, April 1961.

13 Levinsky, E. S. and Yoshihara, H., "Rarefied Hypersonic Flow Over a Sphere," published elsewhere in this volume.

14 Johnson, E. O., and Malter, L., "A Floating Double Probe Method for Measurements in Gas Discharges," Phys. Rev., vol. 80, Oct. 1950, pp. 1, 58-68.

15 Talbot, L., "Theory of the Stagnation-Point Langmuir Probe," Phys. Fluids, vol. 3, 1960, pp. 2, 289, 298.

16 Kunkel, W. B., "Some Considerations Concerning Magneto-hydrodynamic Exhause Control for Rocket Guidance," Space Technology Labs. Inc., Los Angeles, Rep. TR-59-0000-00608, Feb. 1959.

17 Clayden, W. A., "Langmuir Probe Measurements in the ARDE Plasma Jet," unpublished report.

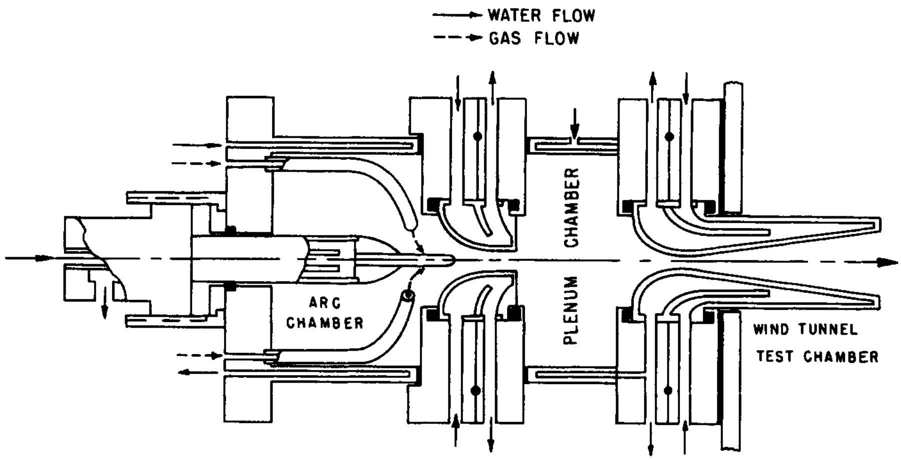


Fig. 1 Plasma generator.

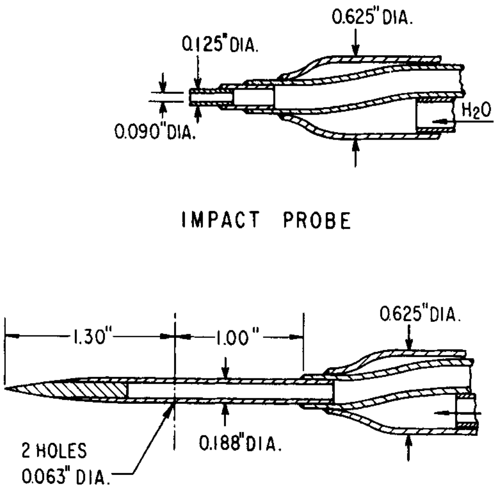


Fig. 2 Water cooled pressure probes.

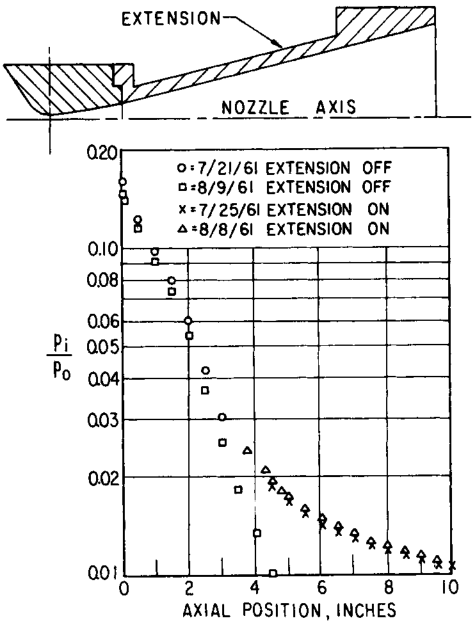


Fig. 3 Axial impact pressure surveys.

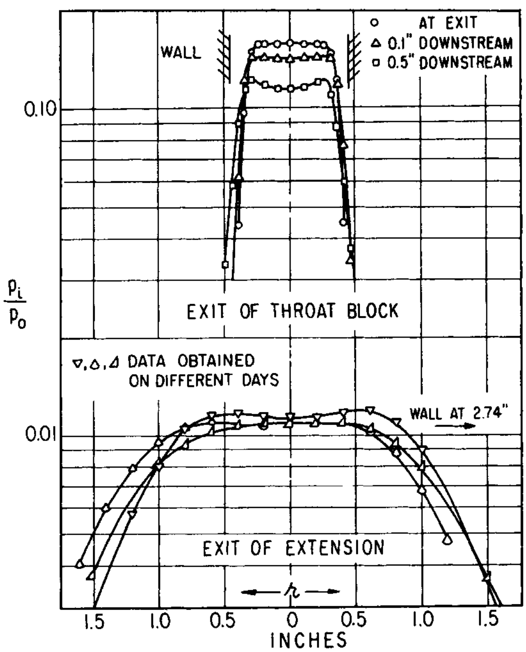


Fig. 4 Radial impact pressure surveys.

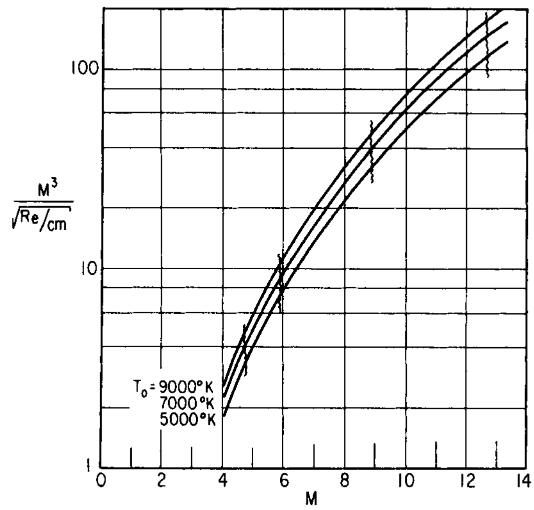
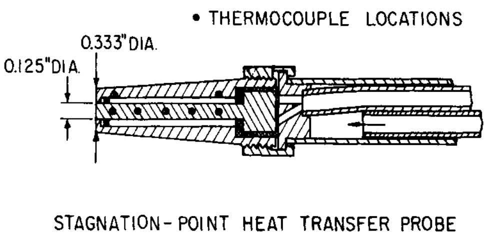


Fig. 5 Hypersonic interaction parameter (isentropic expansion of Argon from $p_0 = 350$ mm Hg).



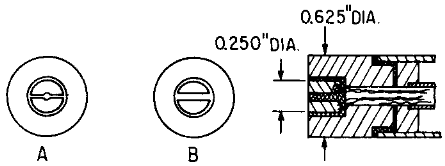


Fig. 6 Stagnation point probes.

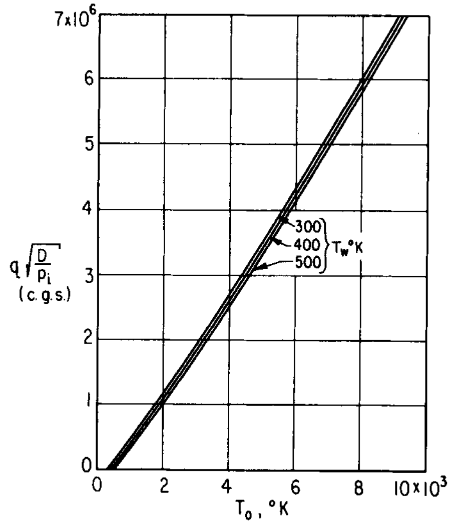


Fig. 7 Stagnation point heat transfer for a flat faced cylinder in Argon; $M \leq 4$

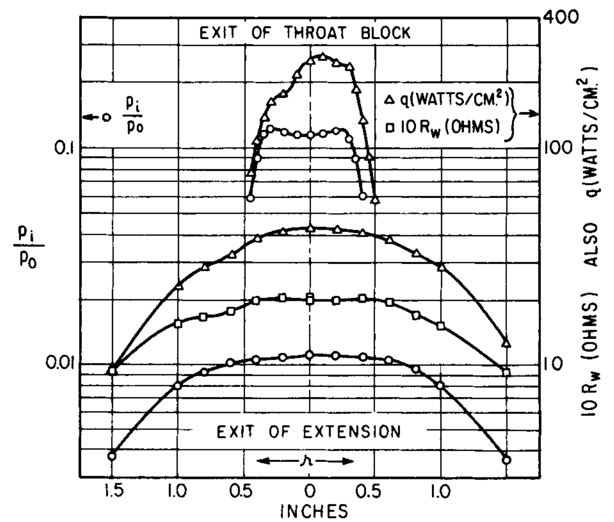


Fig. 8 Comparative heat transfer and impact pressure surveys.

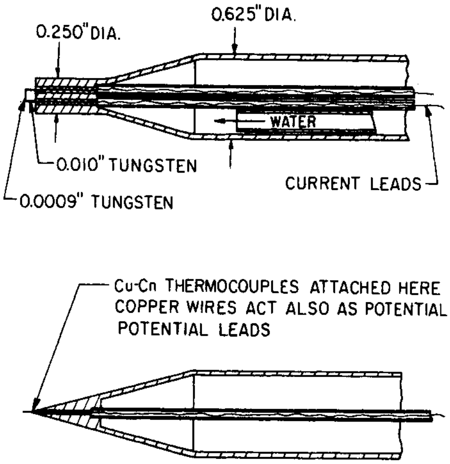


Fig. 9 Free-molecule flow heat transfer probe.

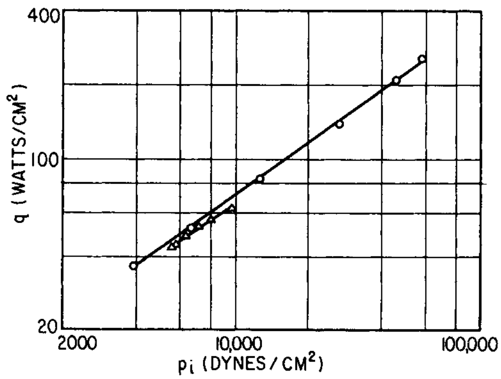


Fig. 10 Heat transfer vs. impact pressure on nozzle axis.

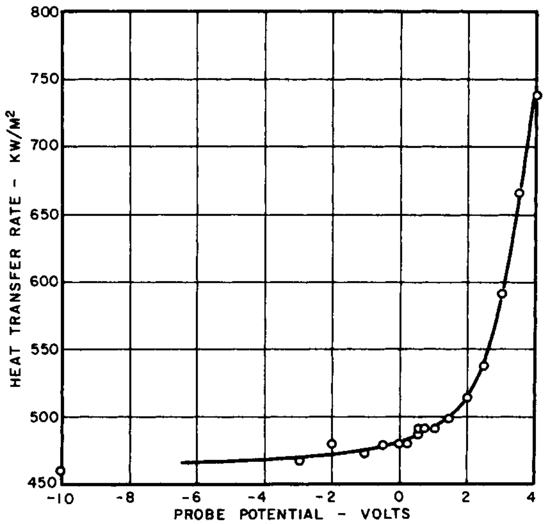


Fig. 11 Effect of prove voltage on stagnation point heat transfer.

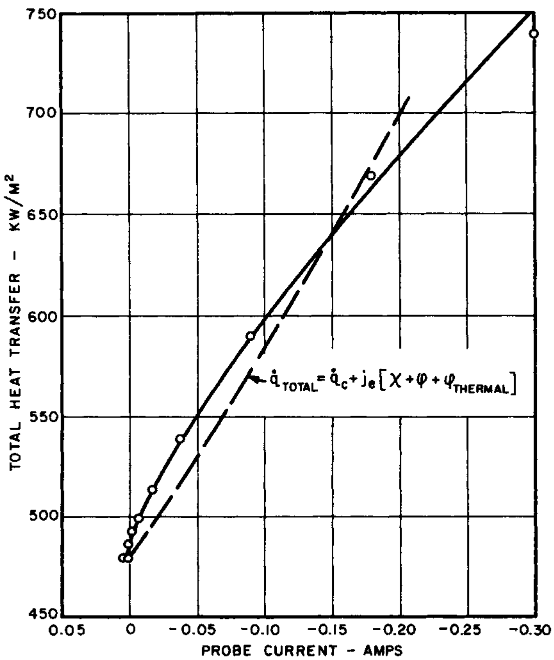


Fig. 12 Conduction plus electron heat transfer for stagnation point probe. (j_e = electron current density; χ = surface work function; ϕ = potential drop across the sheath; ϕ thermal = initial thermal energy of the electrons; all in mks units.)

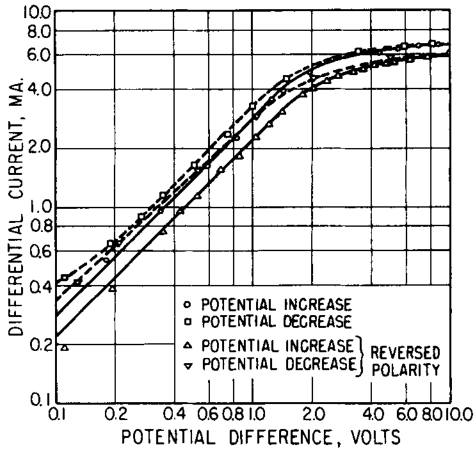


Fig. 13 Stagnation point langmuir probe (configuration B); characteristic curves (0.200 in. off center).

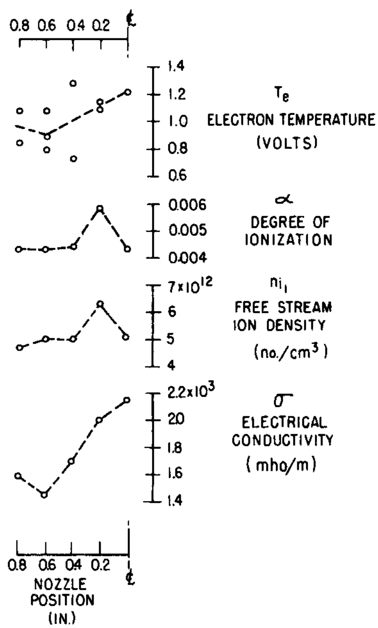


Fig. 14 Electrical properties of stream.