

Considerations in the Design and Performance of HYPERSONIC

The potential of air-breathing vehicles for low-altitude flight at velocities in excess of Mach 5 has led to an increasing need for information on high-speed aerodynamic phenomena. Presently existing theory cannot handle the complex interactions that occur between vehicle, shock waves, and boundary layer in a non-ideal atmosphere. It is, therefore, necessary that experiments be performed with test conditions simulating those of the actual flight environment. Complete simulation, however, becomes increasingly difficult as the Mach number increases and altitude decreases. For example, duplication of Mach 10 flight at 80,000 feet would require a wind tunnel with a supply pressure of 1500 atm, a supply temperature of 3500°K, and maintenance of thermodynamic equilibrium during the expansion of the gas in the nozzle. Such supply pressures and temperatures are beyond the range of a conventional wind tunnel. This is not to say that a conventional wind tunnel which provides simulation of Mach numbers and Reynolds numbers only is of no use in hypersonic testing. In fact,

the generally large size and long testing time of a conventional wind tunnel are ideal for making detailed configurational studies. The purpose of a facility capable of duplicating the flight environment is, then, to identify and study those areas beyond the capability of the conventional wind tunnel where such factors as non-ideal gas behavior, thermodynamic non-equilibrium, and high heat transfer rates may introduce problems.

In contrast to wind tunnels designed for continuous operation, there are the "pulse-type" facilities which are capable of providing high temperature but are limited in test time (approximately 2 msec or less) and in pressure. Pulse-type and plasma-arc tunnels are capable of duplicating the environment for hypervelocity flight at extremely high altitudes. Between these extremes, but more closely related to the "pulse-type" facilities, are the tailored shock tunnel and the hypersonic gun tunnel (free-piston compression tunnel). The test times available from the latter are an order of magnitude longer than those from the reflected

The performance of a hypersonic gun tunnel designed to operate at driving gas pressures up to 1000 atm is presented. An evacuated acceleration tube is shown to increase significantly the enthalpy of the gas. It is also shown that Mach 10 flight at 87,000 feet can be simulated in the tunnel. With a contoured nozzle designed for Mach 10 flow attached to a gun tunnel operating at a pressure of 1600 atm and a temperature of 3400° K, the flow in the test region is found to be vibrationally frozen.

R. A. Makofski and R. W. Henderson

GUN TUNNELS

shock tunnel while the enthalpy and pressure levels are considerably higher than those available in conventional wind tunnels.

The gun tunnel consists of three principal sub-assemblies: (a) the gun, which is a free-piston compressor, (b) the nozzle-test section assembly, and (c) a vacuum reservoir, referred to as the dump tank, for receiving the gas leaving the test section. Operation of the gun is described with the aid of Fig. 1, which shows a schematic of the compression unit and the nozzle. The compression unit is made up of a pressure vessel; the breech, which contains the high-pressure gas that drives the piston; and the barrel sections, an evacuated acceleration tube in tandem with a pump tube containing the test gas. The air in the pump tube is confined by thin plastic diaphragms located at the acceleration tube juncture and at the entrance to the nozzle. A scored metal diaphragm is used between the breech and the acceleration tube. At a given pressure in the breech, the metal diaphragm ruptures and the high pressure gas accelerates the

piston down the barrel. The piston enters the pump tube at supersonic velocity and the gas in the pump tube is compressed by the shock wave ahead of the piston. This shock wave is reflected from the end-wall of the barrel and then from the piston, resulting in further compression and heating of the gas. After about the third reflection the shock wave has been sufficiently weakened so that the remainder of

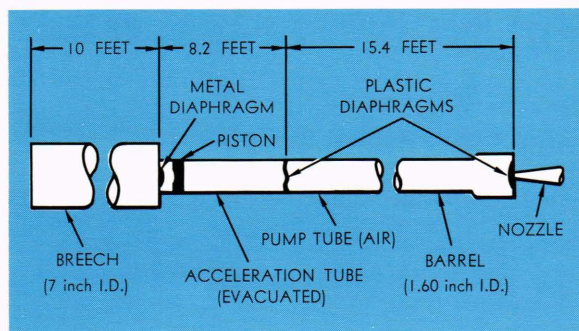


Fig. 1—Schematic of APL hypersonic gun tunnel.

the compression may be assumed to be isentropic. Behind the piston as it decelerates, a series of compression waves form and coalesce into a shock wave which travels back down the barrel to be reflected as an expansion wave from the breech-barrel interface.

The time available for testing is limited either by exhaustion of the gas in the pump tube or, if it occurs first, by the sudden drop in pressure that takes place when the reflected expansion wave strikes the back of the piston. This latter process is termed "wave quenching." It should also be mentioned that increasing the breech-to-barrel area ratio to values greater than one has the same effect on the piston velocity and final pressure as an increase in the breech pressure. The ratio of this apparent driving gas pressure (due to the increased area ratio) to the actual breech pressure, termed the overpressure ratio, can be as high as 1.89 for diatomic gases (air, N_2) and 2.04 for monatomic gases (helium). These values are for the condition of isentropic flow behind the piston and a breech-to-barrel area ratio of infinity.

History

The initial attempt at building a gun tunnel¹ was that of A. J. Eggers of NACA-Ames. The device used a heavy piston driven by the explosion of black powder. The high pressures and temperatures generated in the gas during piston overshoot resulted in considerable damage to the barrel end-wall and nozzle throat. To avoid this difficulty the piston was removed and the facility used as a reflected shock tunnel. Some time later two groups in England—one at the Armament Research and Development Establishment and the other at the University of Southampton—began the development of a gun tunnel using light pistons driven by compressed gas. These pistons were made of nylon and were limited to velocities of about 2000 ft/sec. Since this velocity severely limits the maximum temperature available, Lemcke² of the Aeronautical Research Institute of Sweden (FFA) devoted about two years to the development of a piston capable of extending this velocity range. His success in this effort provided the impetus to the building of the Applied Physics Laboratory facility which uses his piston. It is made of Makrolon, a polycarbon plastic manufactured by Farbefabriken

¹ A. R. Collar and J. Tinkler (eds.), "Hypersonic Flow," *Proc. 11th Symposium Colston Res. Soc.*, Apr. 6-8, 1959, 89-152.

² Bo Lemcke, "An Investigation of the Stagnation Conditions in the Shock-Compression Heater of a Gun Tunnel," *Flygtekniska Forsöksanstalten, FFA Report 90*, Stockholm, Sweden, Feb. 1962.

Bayer and has been used by APL at a driving gas pressure of 1000 atm which is about three times the pressure used by FFA. Under these conditions the piston withstands more than 10^6g acceleration and about 10^7g deceleration, a rather remarkable feat for a seemingly fragile bit of plastic.

APL Tunnel

Construction of the APL tunnel began in 1963. Operation with a 4-inch conical nozzle and a closed test section began in 1964. At present the tunnel is equipped with an 8-inch contoured nozzle and a free-jet test section, which were installed in November of 1965.

The overall length of the tunnel is approximately 50 feet. Nitrogen at pressures up to 15,000 psi is used to drive the gun. The 10-foot-long breech has 2½ inch thick walls and a volume of 3 cubic feet. The precision bore (1.6-inch diameter) barrel is 23½ feet long and is fitted with a heavy-walled end section capable of withstanding the very high peak pressures that occur at the end of the pump tube. The nozzle and test chamber are shown in Fig. 2. The nozzle is made in two sections—a steel section with a molybdenum insert in the throat region and an epoxy-fiber-glass section. The test section chamber is 17½ inches wide, 18 inches high, and 28 inches long with removable sides for easy access. Pressure-tight, thick-walled rubber expansion joints and sleeves are used at the juncture to the chamber, and test models are supported within the chamber from a massive floor-mounted pedestal. These items along with the epoxy-fiber-glass section of the nozzle serve to isolate the test instrumentation from the vibrations set up by the recoil of the gun. The test chamber is connected to a 75-cubic-foot cylindrical dump tank by means of a 3-foot-long diffuser duct.

The instrumentation used for shock tunnel pressure measurements works well for the gun tunnel.

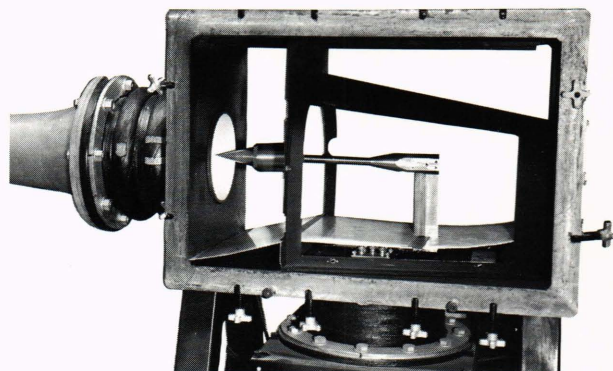


Fig. 2—Nozzle and test chamber of APL hypersonic gun tunnel.

Both pressure and force measurements are made using piezo-electric transducers in conjunction with charge amplifiers and an oscilloscope. The oscilloscope trace is triggered by a signal generated by the rupture of the metal diaphragm. Although piezo-electric transducers are quite rugged they are sensitive to temperature and acceleration and certain precautions must be taken in this respect. The transducers are particularly affected by the strong vibration set up when the gun is fired. It is, therefore, essential either to calibrate these effects or to isolate the instrumentation in a manner such as that just described. Photographs of the flow in the gun tunnel test section are made using a conventional double mirror schlieren system with a mercury arc light source. The light source is flashed by a condenser discharge triggered through a time delay circuit by a signal generated by the diaphragm rupture.

One of the most difficult measurements encountered in gun tunnels is that of determining total temperature, i.e., the temperature of the flow at zero flow velocity. The difficulty lies in the necessity for measuring high temperatures which change rapidly. One method of determining these temperatures employs a calorimeter with a platinum foil as the sensing element. In our applications the temperature of the foil is determined by measuring the change in electric resistance of a portion of the foil. The heat transfer rate is computed from the slope of the temperature-time curve and the calibrated properties of the foil.

The design of this facility has been governed to a considerable extent by the interest in flight velocities of the order of Mach 10 and such practical considerations as availability and cost of equipment. Aside from being the only tunnel of its kind presently operating in this country, it has a higher driving pressure than any known operating gun tunnel. To design the nozzle a thorough understanding of the working of the compression unit was needed. A discussion of this unit and of the performance of the 8-inch contoured nozzle and test section follow.

Characteristics of Free-Piston Compressor

The motion of a piston traveling at supersonic speeds involves many interesting and complex details of fluid mechanical phenomena. This discussion, however, will be confined to the pressure and temperature generated at the entrance to the wind-tunnel nozzle.

A piston accelerating into a gas will generate a shock wave which will increase in strength as the piston velocity increases. As a result, the temperature of the particles that passed through the early

stages of the shock wave will be lower than those passing through in the final stages of the piston motion. Subsequent mixing of the particles will reduce the overall temperature level. Also, the residence time for a particle heated in the early stages of the shock will be longer leading to higher relative heat losses. The above difficulty can be alleviated by the use of an evacuated acceleration tube in which the piston is accelerated to equilibrium velocity, i.e., the velocity in the pump tube at which the pressures on the front and rear surfaces of the piston are equal.

One means of measuring the performance of the gun compression unit is to compare the actual piston velocity with the computed equilibrium velocity of a frictionless piston with isentropic expansion of the driving gas. (Note that the equilibrium velocity and the velocity of a zero mass piston are identical.) The comparison is shown in Fig. 3. Note that for driving gas pressures above 400 atm, the data scatters about the value of one. This scatter can be attributed to reading and instrumentation error and such items as piston fit, driving gas temperature, and the mode of diaphragm rupture. The results do suggest that it is reasonable to assume that for driving gas pressures above 400 atm, the piston velocity may be computed using the equilibrium velocity techniques.

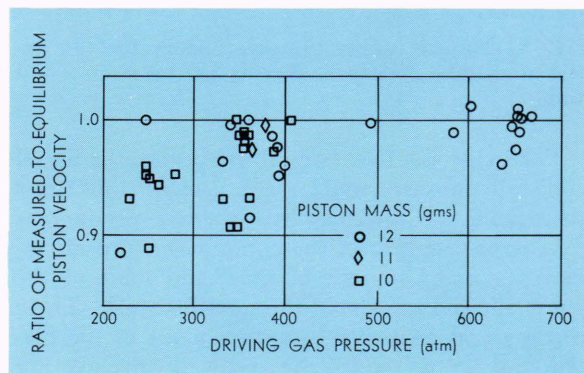


Fig. 3—Ratio of measured piston velocity to equilibrium piston velocity as a function of driving gas pressure.

The supply pressure to the nozzle, i.e., the pressure at the end-wall, as a function of time is shown in Fig. 4 for two acceleration tube pressures. The ratio of the supply pressure to the initial driving gas pressure is plotted as a function of the ratio of the elapsed time to the wave quench time, i.e., the time when the reflected expansion wave strikes the back of the piston. These curves show the relatively slow rise with time of the end-wall pressure. The termination of the run is indicated by the piston impact with the end-wall as the last bit of

gas is forced from the piston tube. The observation of piston impact permits a computation of the average temperature during a run, with the result that for these data the average temperature using the evacuated acceleration tube is about 35 percent greater than the temperature obtained with the acceleration tube at ambient conditions. Similar results were obtained for other driving gas pressures.

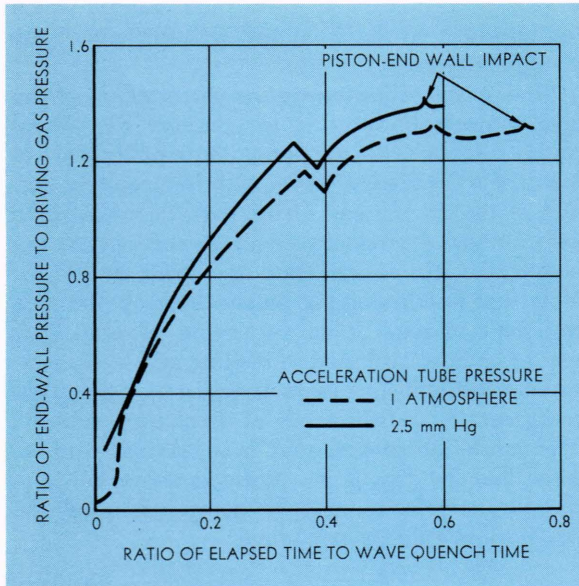


Fig. 4—Tunnel supply pressure history for two acceleration tube pressures. Driving gas pressure = 350 atm.

For isentropic flow behind the piston, the barrel end-wall pressure would reach a value 1.89 times the driving gas pressure. In reality the driving gas flow is not isentropic, the compression process requires a finite amount of time, and some of the gas is allowed to expand through the nozzle. These processes, as well as the rupture mode of the diaphragm, act to reduce the end-wall pressure. For the data of Fig. 4, the maximum overpressure ratio, i.e., the ratio of total pressure at the nozzle inlet to the initial driving gas pressure, is about 1.48. Other tests have shown that there is no noticeable effect of piston mass except for operation as a shock tunnel, i.e., with a piston mass of zero. For this case, the overpressure ratios are significantly lower than for the tests with piston masses of 7 to 12 grams. Similarly, no significant effect on the throat area is obtained. There is, however, an increase in overpressure ratio as the driving gas pressure increases. Values of 1.75 have been obtained at driving gas pressures of about 650 atm.

The relatively slow rise time of the pressure is of some concern. In an attempt to improve this a

teflon plug has recently been used in place of the plastic diaphragm at the interface of the pump tube and nozzle. The plug, which is made to have a light interference fit at the throat of the nozzle, offers more resistance to the compressed gas than the thin diaphragm. As a result the gas is confined longer and does not expand through the nozzle until a high pressure is reached. In this manner the tunnel-supply pressure during the test period is more nearly uniform.

The temperature of the gas supplied to the wind tunnel has been computed from the measurements obtained using the stagnation point calorimeter. These data are compared (Fig. 5) with the total temperature computed assuming shock-wave heating by the wave ahead of the piston, its first reflections from the end-wall and in turn from the piston face (three shocks), and isentropic compression thereafter. The agreement is quite good except for the final portion of the run which shows a sudden increase in temperature. This rise is believed to be combustion products from the front face of the piston.

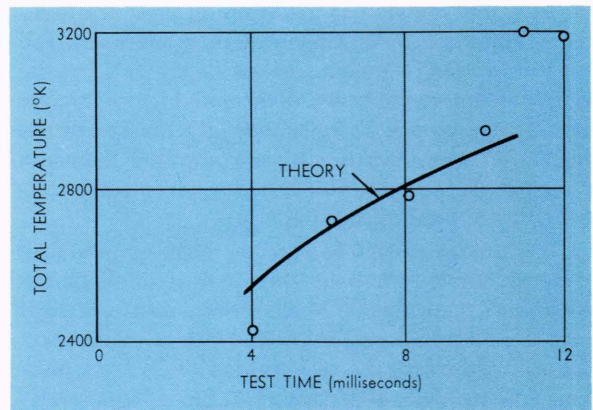


Fig. 5—Total temperature history based on stagnation point heat transfer measurements for a driving gas pressure of 650 atm.

Performance of Wind Tunnel

One of the most desirable features that a wind-tunnel designed for studies of boundary layer phenomena can have is constant flow throughout the test region. Such a region can be obtained by careful contouring of the nozzle. Unfortunately, this contouring is valid only for a given set of supply conditions. Studies have indicated that for this facility it was best to design the nozzle for the maximum supply pressure and temperature expected, i.e., 1600 atm and 3400°K. Operation at off-design conditions would then introduce small, isentropic gradients into the flow which are acceptable even if undesirable.

After the installation of the nozzle a series of tests was made to determine the flow properties. These tests consisted of measuring the pitot pressure, i.e., the total pressure behind a normal shock wave, at various radial and axial locations and of measurements of the pressure on the surface of a 10° cone along the wind-tunnel axis. One of the first questions to be studied was the behavior in time of those parameters describing the flow. The measured ratios of pitot to end-wall pressure as a function of the ratio of elapsed time to wave quench time, for a five-pitot rake assembly are shown in Fig. 6. The pressure ratio can be seen to be essentially constant for times greater than 0.2 of the wave quench time. The Mach number of the flow in the test region may be computed from the pitot pressure measurements. The results indicate that the average Mach number in this region is about 11.1 for the range of pressures tested and with the assumption of thermodynamic equilibrium of the expanding gas in the nozzle. For our conditions such an assumption is equivalent to stating that the time scale of the flow is considerably longer than the time scale for the transmission of energy from vibrational to rotational and translational states, and that the vibrational energy levels are in equilibrium at the static temperature of the gas. In general this is not the case and the test gas is not in vibrational equilibrium. To determine the deviation from equilibrium and to avoid extensive calculations that would be required to determine this value using non-equilibrium equations, the simplified concept of a freezing point is used. This concept assumes that the flow upstream of the freezing

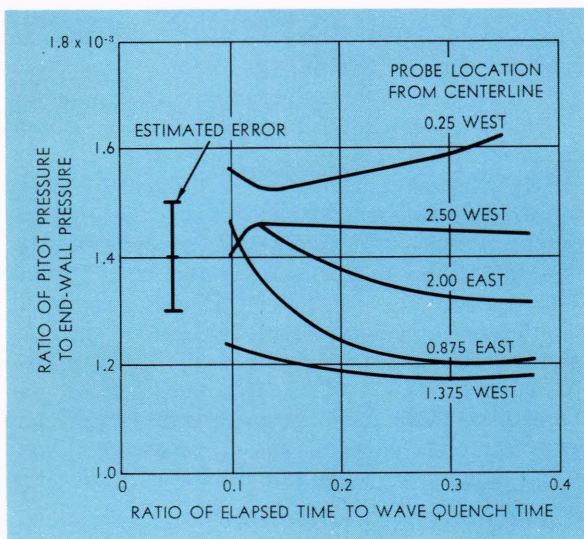


Fig. 6—Ratio of pitot-to-supply pressure as a function of time. Driving gas pressure = 241 atm at station 34. Rake horizontal.

point (computed using Phinney's criteria³) is in thermodynamic equilibrium and the flow downstream is frozen, i.e., the energy contained in the vibrational states at the freezing point can no longer be transferred to the other modes. A comparison of the calculated pressure on the surface of a 10° cone in equilibrium and in frozen flow is given in Fig. 7 along with the experimental data

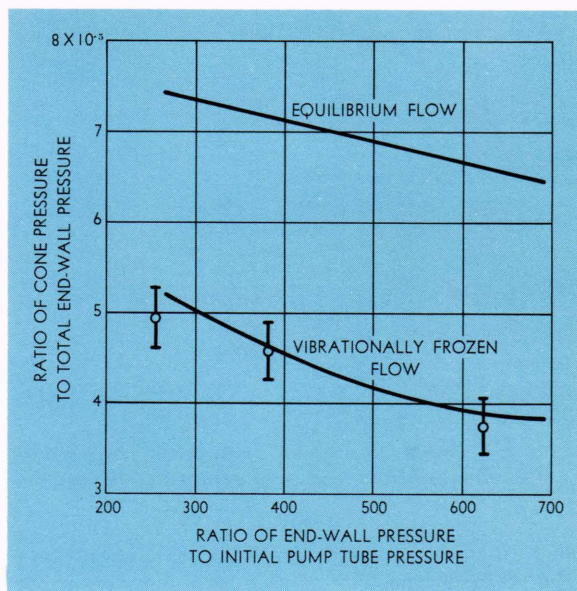


Fig. 7—Comparison of equilibrium and frozen flow calculations with measured pressures on a 10° half-angle cone.

for various wind tunnel supply pressures. As a result of this vibrational freezing, the Mach number of the flow is somewhat larger than the Mach numbers computed for equilibrium flow. The Mach number profiles for the frozen condition are shown in Fig. 8. The results are as expected with a disturbance occurring at the low driving gas pressures and constant Mach number profiles over the test region as the driving gas pressure increases.

Hypersonic Testing in the Gun Tunnel

The characteristics of the gun tunnel can provide an environment which is well-suited for the study of many of the problems associated with low-altitude hypersonic vehicle technology. For example, configurational studies of hypersonic ramjet inlets can be made and are, in fact, the main reason for constructing this facility. More basic phenomena can also be studied. For example, the interaction of a shock wave with the boundary

³ R. Phinney, "Non-dimensional Solutions of Flows with Vibrational Relaxation," *AIAA*, 2, 1964, 240.

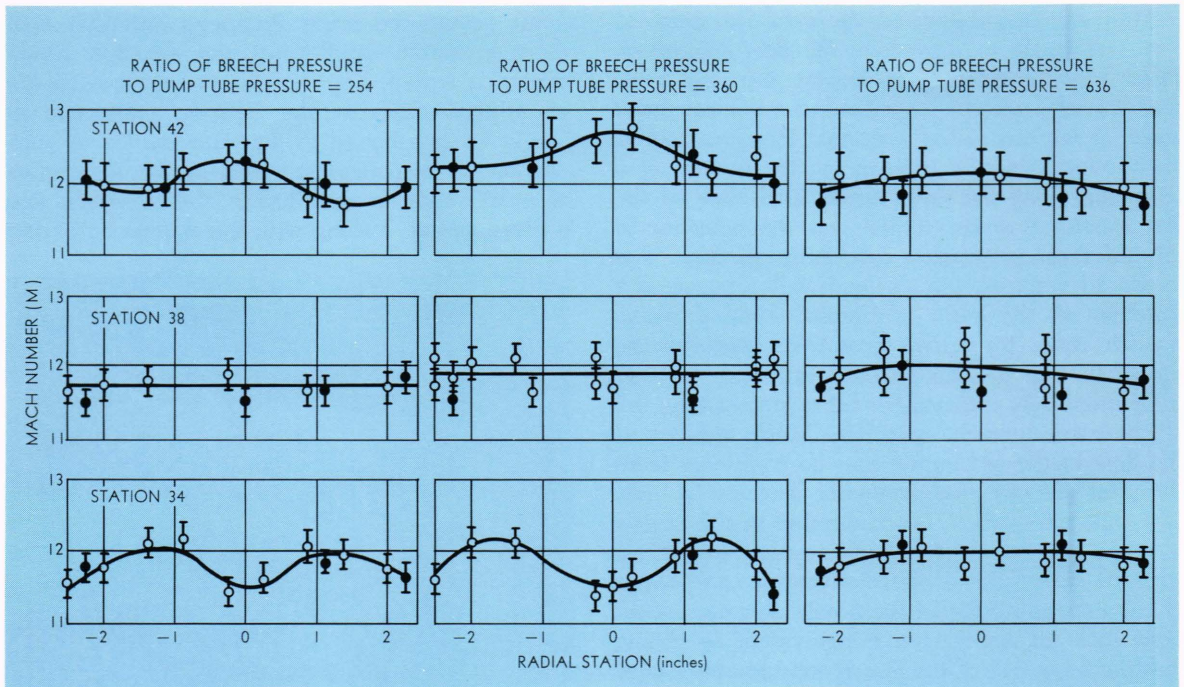


Fig. 8—Mach number profiles for frozen flow. Open symbols represent rake located horizontally. Solid symbols represent rake vertical. Negative numbers are distance below or west of centerline.

layer on a surface can result in flow separations from the surface with consequent changes in the flow configuration, pressure distribution, and heat transfer. A schlieren photograph of such an interaction is shown in Fig. 9. This photograph shows the interacting of the shock from the shock generator with the boundary layer of the plate, the thickening of the boundary layer, the turning of the separated flow back to the plate surface as witnessed by the expansion waves, and the reattach-

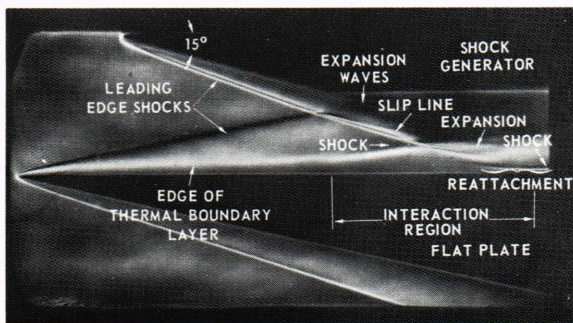


Fig. 9—Schlieren photograph of incident shock wave-laminar boundary layer interaction in hypersonic gun tunnel. Mach ≈ 12 ; leading edge thickness of flat plate ≈ 0.009 inch; leading edge thickness of shock generator ≈ 0.012 inch; free-stream Reynolds number $\approx 1.1 \times 10^6$ /ft; tunnel supply pressure = 480 atm.

ment of the boundary layer on the plate. Similar studies can be made on the transition from laminar to turbulent boundary layers and on the turbulent boundary layers themselves.

Conclusions

The results of the studies in the hypersonic gun tunnel have shown that the use of an evacuated acceleration tube can significantly increase the stagnation temperature. These results indicate that at least for this facility the stagnation temperature may be computed using the three shock hypothesis. Overpressure ratios as high as 1.75 have been obtained.

The calibration of the contoured nozzle designed to provide a uniform Mach 10 flow at supply conditions of 1600 atm and 3400°K has indicated the flow in the test region to be vibrationally frozen. The Mach number for these conditions is about 12, the difference being due to the freezing of the flow and an underestimate of the boundary layer displacement thickness. The data indicate the Mach number profiles to be uniform at the higher driving gas pressures whereas expansion waves and a pressure gradient are present at the lower driving gas pressures.

Finally, the data indicate that the hypersonic gun tunnel can provide flow simulation of a moderate Mach number ($M \sim 10$) flight environment at moderate altitudes (about 85,000 feet)