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THE PRINCETON HELIUM HYPERSONIC TUNNEL AND PRELIMINARY RESULTS ABOVE M = 11

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FOREWORD

The work reported herein was carried out at the Gas Dynamics Laboratory, James Forrestal Research Center, Princeton. It is part of a program of research on viscous effects at hypersonic speeds sponsored by the Aeronautical Research Laboratory, Wright Air Development Center, United Stated Air Force under Contract No. AF33(038)-250, Task 70114, "Viscous Effects in Hypersonic Wind Tunnels" with Kenneth F. Stetson as project engineer.

ABSTRACT

This report describes the development, construction and operation of a small research tunnel using helium as a working fluid. Mach numbers as high as 15 have been attained in a 3½ inch diameter test section and Mach numbers of the order of 20 seem practical with further development. Some preliminary results at $M = 12.7$ are presented in the form of pressure distributions on a flat plate and Schlieren photographs. Some results on the effect of leading edge radius are also included.

The experimental equipment is small, relatively simple and inexpensive, and seems to provide an excellent method for fluid dynamics studies at very high speeds.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

Leslie B. Williams

LESLIE B. WILLIAMS
Colonel, USAF
Chief, Aeronautical Research Laboratory
Directorate of Research

TABLE OF CONTENTS

LIST OF FIGURES

THE PRINCETON HELIUM HYPERSONIC TUNNEL AND PRELIMINARY RESULTS ABOVE M = 11

I INTRODUCTION

As the speed of aircraft and missiles has been constantly increased over the last few years, considerable interest has been aroused in the study of "hypersonic flows"--usually defined as flows many times the speed of sound. At these very high speeds, certain problems not encountered or negligible at lower speeds, may become very important. In supersonic studies, the flow may be considered, in the major part, as an inviscid compressible flow with the viscous effects limited to the boundary layers, i.e., the region close to the solid surface. The general treatment, therefore, is essentially that of computing a potential flow with a small perturbation to account for the effects of viscosity. There are only a few cases where the entropy gradients in the free stream are important and the gas characteristics are essentially constant throughout the flows. Since the viscous effects increase with increasing Mach numbers, at very high speeds they may become a primary rather than a secondary effect. The potential flow would then not be a good first approximation. Since the viscous flow may significantly alter the potential flow -- which in turn influences the viscous flow, and so on -- a sort of "feed back loop" would be established. The entropy gradients in the free stream may also become very important since at very high Mach numbers all shocks are strong shocks. Very high temperature rises can be caused by these strong shocks (very high flight stagnation temperature) and sufficiently high temperatures may be experienced to cause dissociation of diatomic molecules and ionization. These effects will cause important changes in the gas characteristics and dissociation and ionization may have a very strong effect on the viscous regions discussed above.

Very little is known of these effects of dissociation and ionization, and in particular, of the time lags associated with these phenomena. At the same time, there is no information available to determine whether the large body of boundary layer theory used in supersonic studies can be used where the Mach number is much greater than one. The boundary layer theory is not exact but includes many approximations which may or may not be valid at very high speeds. The only detailed boundary layer results available are limited to Mach numbers less than five and do not include any effect of dissociation and ionization.

In general, there are two basic techniques which are used for aerodynamics studies, (1) free flight studies and (2) wind tunnel investigations. The free flight studies can be made to exactly simulate the Mach number, Reynolds number, temperatures, dissociations, ionizations, etc. experienced in the actual flight. The technique is, however, quite expensive and it is extremely difficult to control the conditions of the test. It is almost impossible to test many configurations or to determine the effect of small changes, such as is usually done in the wind tunnel investigation of an airplane or missile. The free flight technique is particularly valuable for studying complete model dynamics and for studies of flight conditions unattainable in wind tunnels, for example, transient effects.

Most aerodynamic studies have been carried out in wind tunnels. A single basic model may be easily modified to study effects of many variables and the time available for testing is almost infinitely higher than provided by free flight techniques. Conditions in the tunnel can be kept constant for long periods while detailed studies of the forces and the flows are carried out. An error in the design simply means another test with a modified model, as constrasted to the loss of the entire model in the free flight work. For most supersonic speeds, one can simulate all of the steady state flight conditions with the exception of stagnation temperatures. This has, however, no significant effect at low Mach numbers since the temperature levels are below that at which the gas constants change significantly. The conventional wind tunnel has several important limitations with regards to hypersonic speeds (1) It is well known that expansion of air at room temperature will cause condensation of the components of air when Mach numbers in excess of about five are attained. For higher Mach numbers, the air must be heated, and just on the basis of preventing condensation, temperatures of the order of 1300° to 1500°F are needed to obtain Mach numbers of 10. (2) Although the wind tunnel may simulate the Mach and Reynolds numbers of flight, the stagnation temperature cannot be simulated. At a Mach number of 10, this temperature is several thousand degrees, far above the limits set by the metals of the settling chamber and nozzle. Therefore, the conventional wind tunnel may be used to reach Mach numbers of the order of 10 but cannot hope to simulate complete flight conditions. The temperatures which can now be attained will just about provide non-condensing air in which to make the test. These above characteristics apply to the conventional tunnel where the testing time may extend over many minutes.

Several new techniques have the possibility of providing full flight conditions in a wind tunnel at very high Mach numbers. A tunnel may be driven by air at very high pressures and temperatures generated by a combination of direct and compression heating (A. Ferri, Polytechnic Institute of Brooklyn) or by strong shocks developed in a shock tube driven by an explosion (Gas Dynamics Facility, ARO, Inc. AEDC, Tullahoma, Tennessee and Cornell Aeronautical Laboratory, Buffalo, New York). In both of these techniques, the testing time is extremely short but there is an excellent possibility of completely simulating flight conditions. With the combination of very high temperatures and short times, there is the obvious difficulty of making detailed measurements in such equipment.

The hypersonic problem may, in a crude way, be broken down into two parts: (1) A problem of fluid mechanics. A study of the applicability and approximations of present supersonic boundary layer theory, its effects on the "free stream", effects of strong entropy gradients, etc. (2) A problem of chemistry or physical chemistry. A study of dissociation and ionization, when theyoccur, how they effect the viscous phenomena, and to what degree.

The first of these problems can be studied in a wind tunnel under carefully controlled conditions. In References 1 - 8 Mach numbers of the order of 10 have been attained. There appears, however, to be little hope of going much higher with conventional air tunnels. At the same time, "hypersonic effects" may not be predominant or even large unless the Mach number is quite high. At low Mach numbers the tests must be of very high precision to obtain results of sufficient accuracy to verify or build a theory. If, however, very high Mach numbers could be obtained, the viscous effects may become predominant. Relatively simple tests with relatively crude instrumentation could yield major results and provide a much better basis of critical comparison with theory or for the contruction of a basic model. Several models have been used for the hypersonic viscous theory now in existence (Reference 9 - 19) but little data is available on which to base any critical evaluation of these efforts.

Since present wind tunnels at Mach numbers up to 10 do not simulate flight, but only provide a non-condensing medium, the obvious solution to get higher Mach numbers, and at the same time eliminate the mechanical problem of heating, is to use a gas which does not condense. Since viscous theory -- if correct -- would predict viscosity effects in any gas, the particular test medium is not vital as long as the gas does not change characteristics within the tunnel. A study of various gases has shown helium to be particularly suited for this application. This paper describes the development, construction and operations of a tunnel using helium as a working fluid and some preliminary results which have been obtained. Mach numbers as high as 15 have been attained and Mach numbers of the order of 20 seem practical with further development. Additional reports on the results of complete aerodynamics studies are in preparation.

II PRELIMINARY DESIGN CONSIDERATIONS

Before design of the hypersonic wind tunnel at Princeton University was begun, certain facilities were already available in the Gas Dynamics laboratory. There was in operation a high pressure air supply consisting of reciprocating compressors and an air storage system capable of operating up to 3000 psi. There was, however, no vacuum system suitable for receiving the discharge from a wind tunnel. It was found that helium could be obtained in semi-trailers at a pressure of 2500 psi. Initial considerations were given to the relative advantages and disadvantages of a closed system in which the helium was saved, or an open system, in which the helium was discharged to the atmosphere. The closed system would require either high capacity compressors or high pressure and low pressure helium storage facilities and smaller compressors. Since static pressures in the tunnel must be below atmospheric for any reasonable stagnation pressure, air would leak into the helium system, and contamination of the helium would result.

Therefore, a helium purification system would have to be incorporated in a closed system. For an open system, helium could be used directly from the tank trailer and passed through the tunnel to the atmosphere. The high pressure air available could be used to operate an ejector to provide a low back pressure for the discharge of such a tunnel. After studying these two systems, it was decided that the initial outlay for equipment needed in the closed system was not justified until the potentialities of the helium tunnel could be established, so the present wind tunnel was built using an open system.

A tunnel with a high stagnation pressure was considered desirable for several reasons: The higher the stagnation pressure, the higher the static pressure and density in the test section. These relatively higher pressures can be measured more easily and the relatively higher densities makes optical observation more sensitive. The high density gives higher Reynolds numbers which are in a range of greater practical interest. The high stagnation pressures aid in the generation of high pressure ratios across the tunnel and thus permit the attainment of high Mach numbers. The maximum Mach number is limited by the ratio of the maximum stagnation pressure to the minimum pressure which can be developed by the ejector. The maximum settling chamber pressure was set at 1500 psi. This value was chosen as one which could be reasonably handled in a standard settling chamber design and, at the same time, this pressure could be held constant for a period of at least several minutes compatible with the volume and pressure in the helium tank trailers.

III WORKING FLUID

If a hypersonic tunnel is to be operated without excessive heating, a gas must be used which has a low condensation temperature. The gases with sufficiently low condensation temperatures to permit the attainment of Mach numbers of the order of 15 without any heating at all are hydrogen and helium. Hydrogen has the advantage of a specific heat ratio about the same as air and would lend itself more to matching experiments in air. However, hydrogen is difficult and dangerous to handle and, for this reason, was not used in the hypersonic tunnel. Helium seemed to be the best choice for our purposes; it has a very low condensation point, an almost constant specific heat, and is not dangerous. Helium can be obtained in the quantities required to operate a small experimental wind tunnel with only 0.1% impurities, mostly hydrogen, which make no important change in the characteristics of the gas.

It has already been pointed out that a helium wind tunnel will not completely simulate flight conditions. It will provide hypersonic flows which contain the same viscous phenomena as that obtained in air tunnels with stagnation temperatures below the temperatures for internal changes of the gas molecules. Either gas can be used to study the fluid mechanical hypersonic flow phenomena. It is important to understand how the data found in helium would correspond to the data found in air, and how to compare the results of these two types of tunnels. From considerations of dimensional analysis of the flow of a compressible gas with heat transfer and viscosity at adiabatic conditions, it may be shown that the proper non-dimensional parameters are Mach number, Reynolds number, Prandtl number, geometry, force coefficient and specific heat ratio. Some of these parameters are functions only of the properties of the gas and others are functions of the flow and the configuration. Such quantities as specific heat ratio and Prandtl number depend on temperature and pressure but are not subject to large changes without going to impractical extremes of these quantities. Therefore, such quantities must be accepted as fixed by the gas. Other quantities such as Mach number, Reynolds number, geometry, and force coefficient are quantities which may be changed fairly easily by changes in pressures, temperatures or physical shapes and may be modified to fit desired conditions. From these considerations, it is obvious that, in general, a flow in one gas cannot be completely duplicated by a flow in another gas and that numerical and emperical data found for one gas may not be directly applied to another. For special cases, for example, a case in which heat transfer and compressibility are unimportant, in which the flow would not depend on Prandtl number or specific heat ratio, it might be possible to match the flow in two different gases. For hypersonic flow, compressibility is extremely important and, since the specific heat ratio is considerably different for air and helium, flows in helium will not directly match flows in air.

The relation of the various quantities to be compared, as a function of specific heat ratio, must be known in order to compare data from flows in helium and air. This relation may be obtained from either a theoretical understanding of the problem or by data gathered for different values of specific heat ratio. Without information of this type, the data for helium and air cannot be directly compared.

The properties of helium were obtained from References 20 and 21. From the results shown in Figure 1, it may be seen that the difference between actual helium and the perfect gas are never more than 1% through the range of interest. At zero pressures the specific heats and ratio of specific heats are constant and the dependence of the quantities on pressure is small.

The viscosity of helium as a function of temperature is shown in Figure 2. From the data in References 20 and 21, the Prandtl number has been calculated and is shown as a function of temperature in Figure 3.

The relation between temperature in the wind tunnel and condensation temperatures at the same pressure is shown in Figure 4 for the Mach number range used. This figure shows that the temperatures never approach close to the condensation temperatures.

In any consideration of high Mach number-low Reynolds number flow, we must consider the possibilities of slip flow and other deviations from the usual flow consideration. The slip flow conditions about the leading edge of bodies in hypersonic flows are discussed in Reference 9.

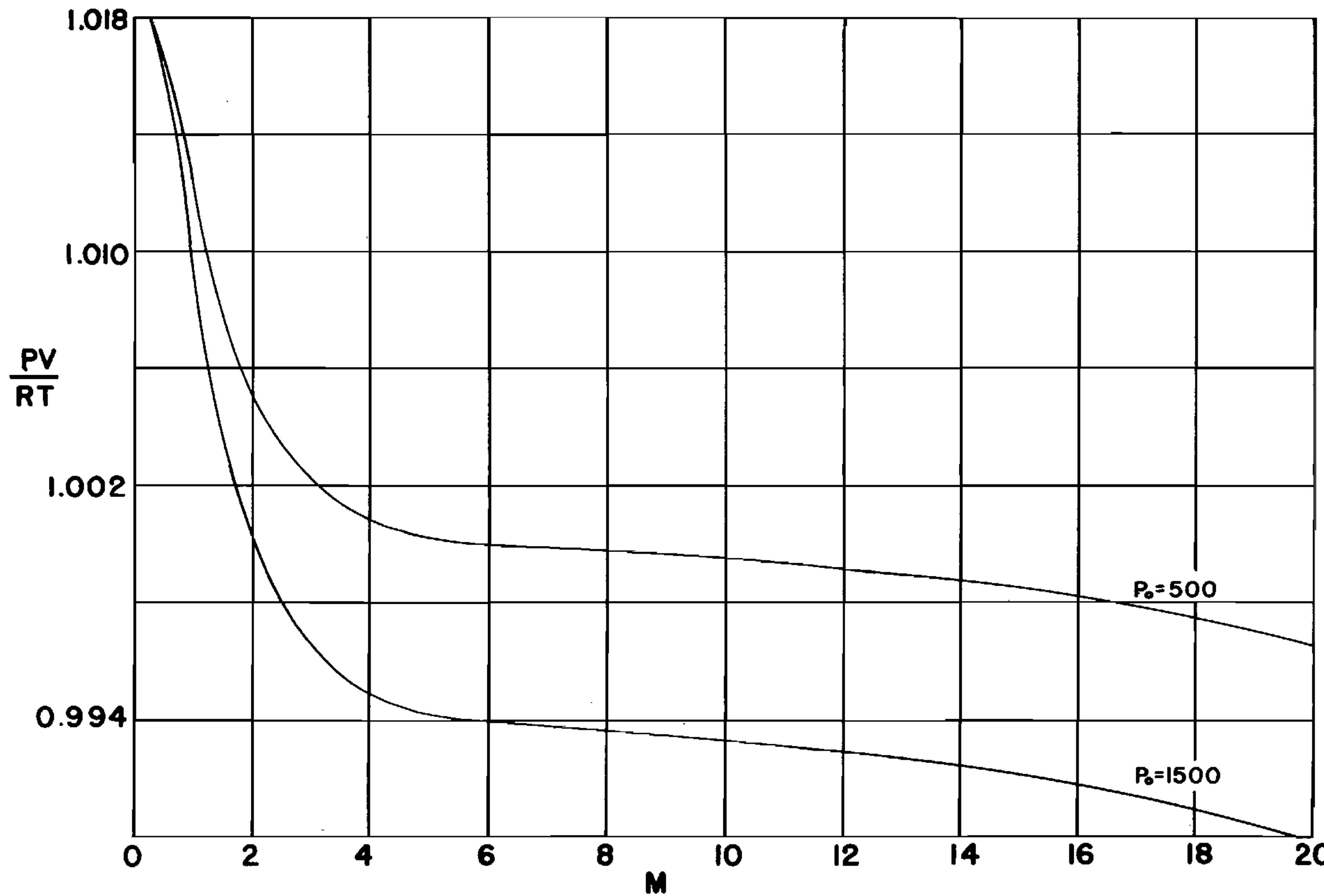


Figure 1 Plot of Perfect Gas Relation versus Mach Number for Helium at Two Stagnation Pressures, T_o = 530°F abs.

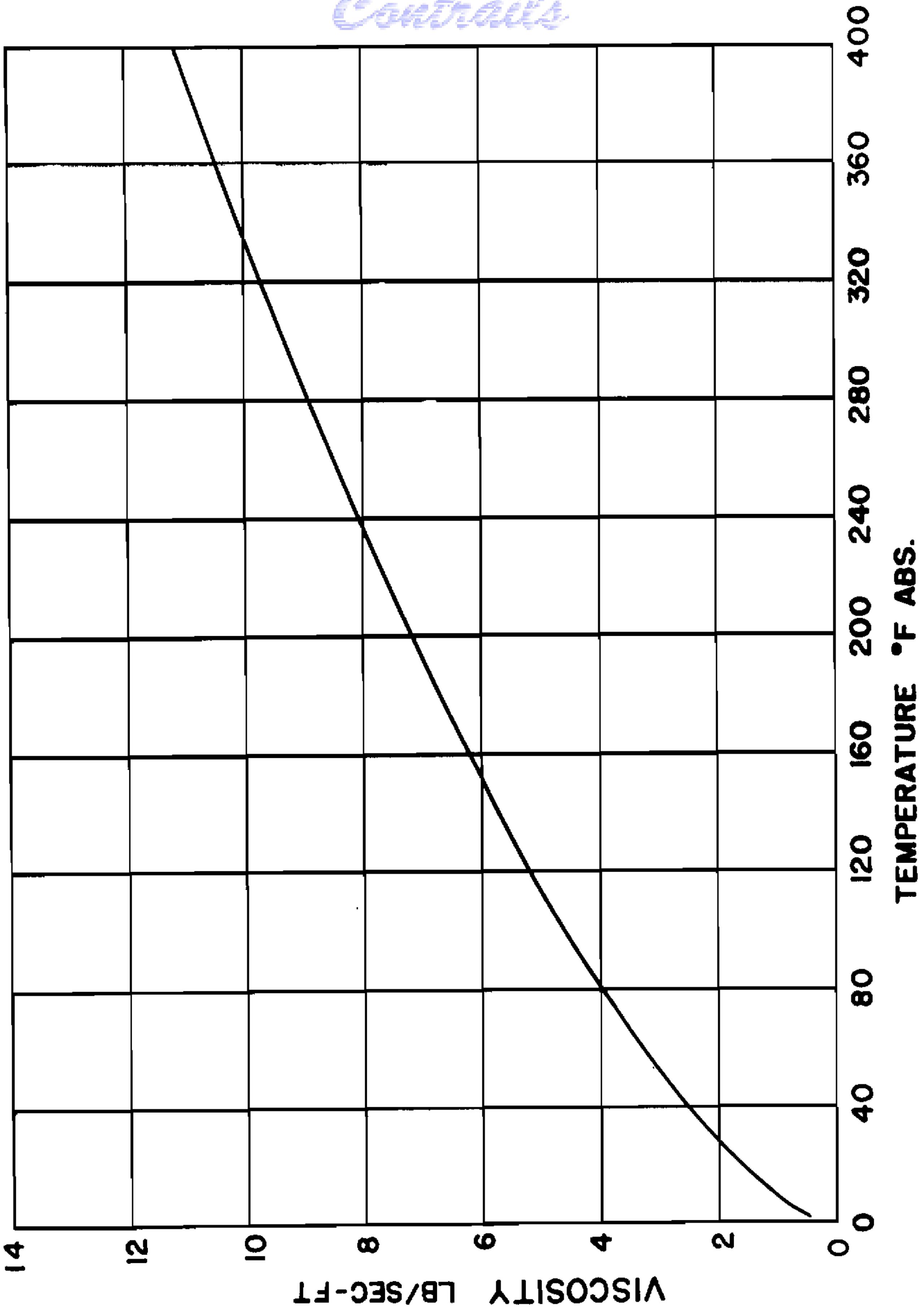


Figure 2 Temperature-Viscosity Relation for Helium

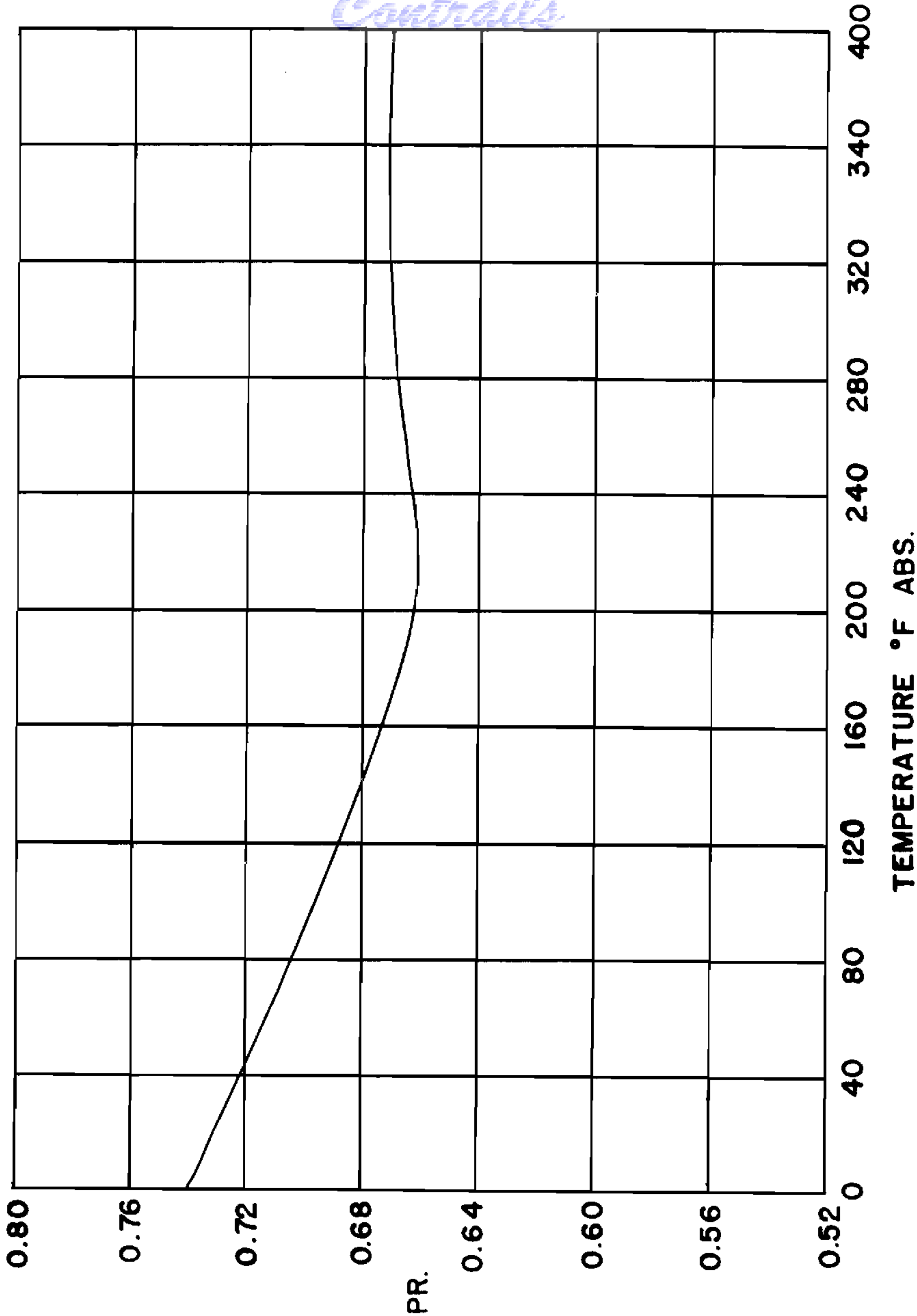


Figure 3 Prandtl Number-Temperature Relation for Helium

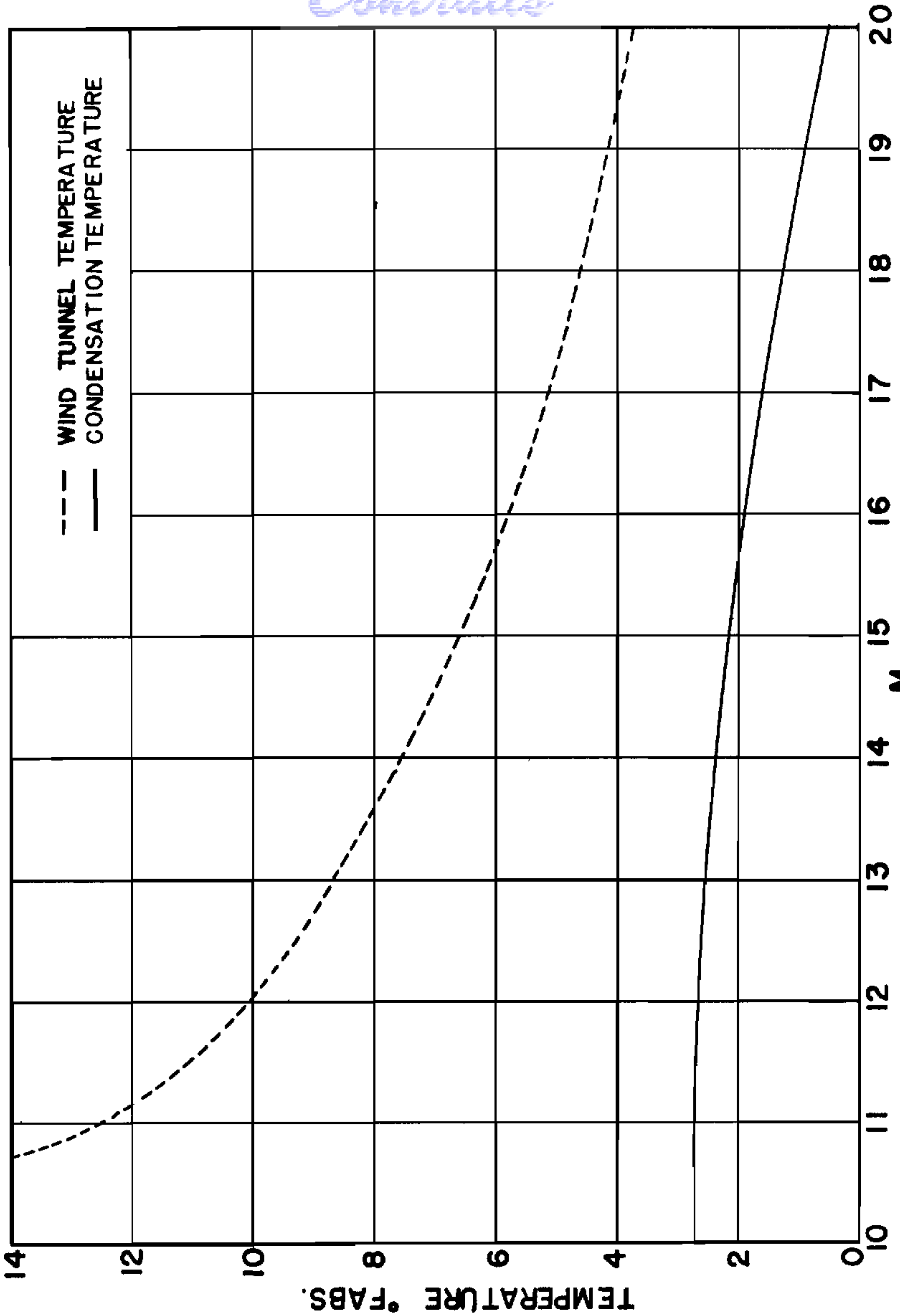


Figure 4 Comparison of Static Temperature in the Wind Tunnel and Condensation Temperatures for Helium

The range in which slip flow can be expected is for $M^2/Re > 0.1$. For $M = 15$ and $Re = 6(10^5)$ per inch, which are the most critical conditions which have been achieved in this tunnel, this criterion indicates that slip is unimportant for dimensions larger than .004 inches. Since the dimensions of most of the bodies tested will be much larger than this, slip flows will only occur in a very limited range and will be only important when this leading edge section has an important influence over the rest of the body. Another effect which must be considered at high M/Re ratios is the change in reading of total head tubes caused by the viscous effect at the front of the tube. The magnitude of this effect has been studied and is described in Reference 22. For $M/Re < .015$, a deviation of less than 1% from the high density case was observed on total head tubes. For $M = 15$ and Re of $.6(10^6)$ per inch, this restriction says that total head tubes of greater diameter than .002 inches will be free of this viscous effect.

IV DETAILED MECHANICAL DESIGN

A INTRODUCTION

The tunnel was designed completely as a research tool. Since there were no other tunnels operating in this Mach number range and little was known of the problems which would be encountered, the equipment was designed with a maximum amount of flexibility. The components are relatively simple, easily modified, and easily disassembled. The general arrangement chosen as most suitable was a bed plate with two horizontal rails on which the equipment could be mounted. The settling chamber was bolted to one end of the bed plate and connected to the high pressure piping. An air operated ejector system is mounted on wheels set on the rails at the other end of the bed plate. Between these two sections, the nozzle, diffuser, and survey sections are located and these parts may be changed as required. This general arrangement is shown in Figure 5.

B SETTLING CHAMBER

The construction of the settling chamber is shown in Figure 6 and 7. It is made out of a section of six inch pipe with slip-on flanges welded to each end. A blind flange is bolted to the back end and a flange modified to fit the nozzle section bolted to the front. The chamber is designed for a working pressure of 1500 psi and a maximum flow velocity of approximately five feet per second. No screens were considered necessary since a contraction ratio of approximately 500 to 1 is provided. The gas enters the chamber through a two inch pipe and discharges radially outward against the rear flange of the settling chamber. A pencil type thermocouple is mounted through the side wall to measure stagnation temperature and two pressure taps are provided at diametrically opposite locations to give an average stagnation pressure reading. (See Figure 6).

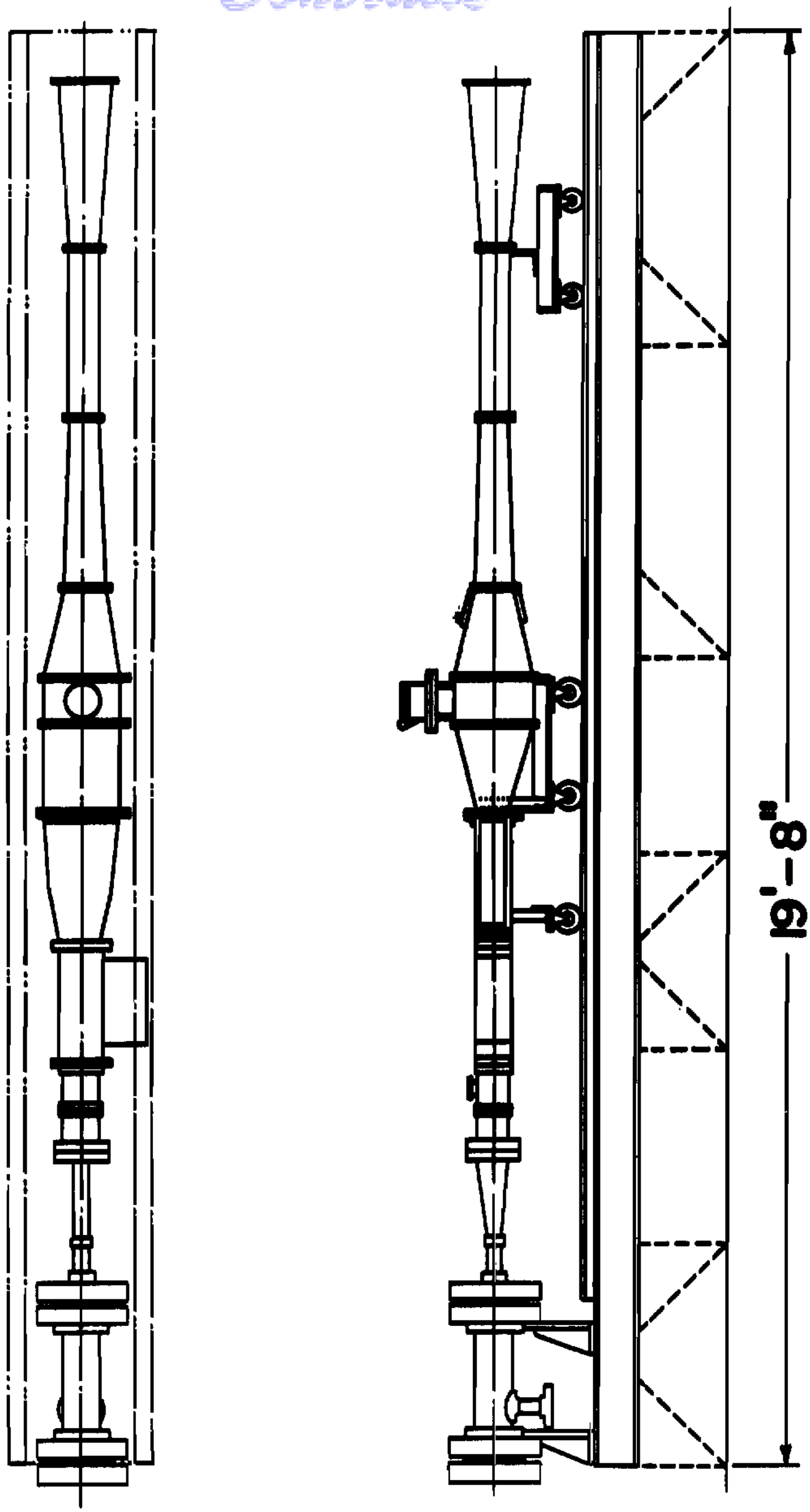


Figure 5 Assembly Lay-out of Princeton University Helium Hypersonic Wind Tunnel

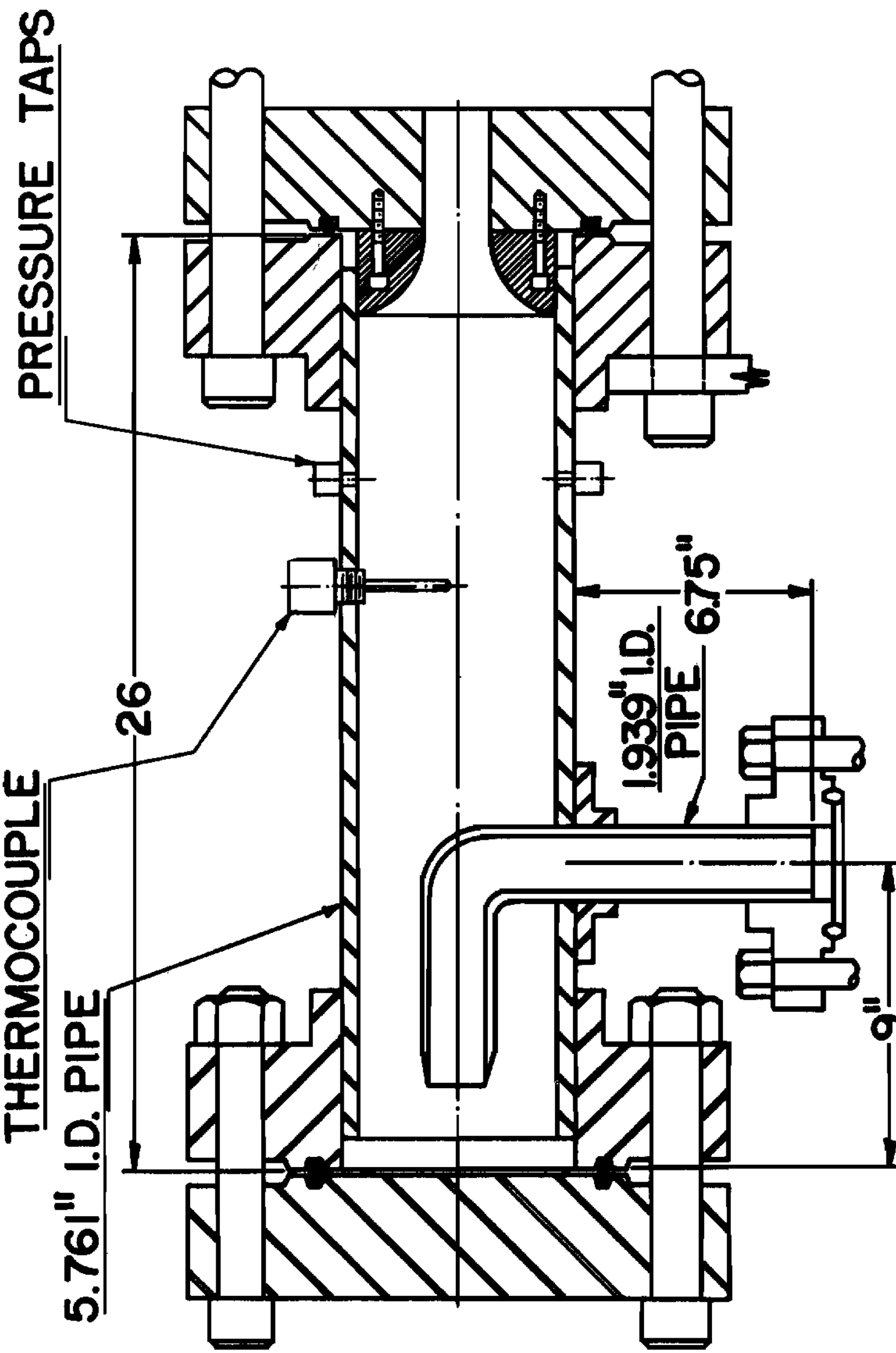


Figure 6 Cross Section of Settling Chamber for Helium Hypersonic Wind Tunnel

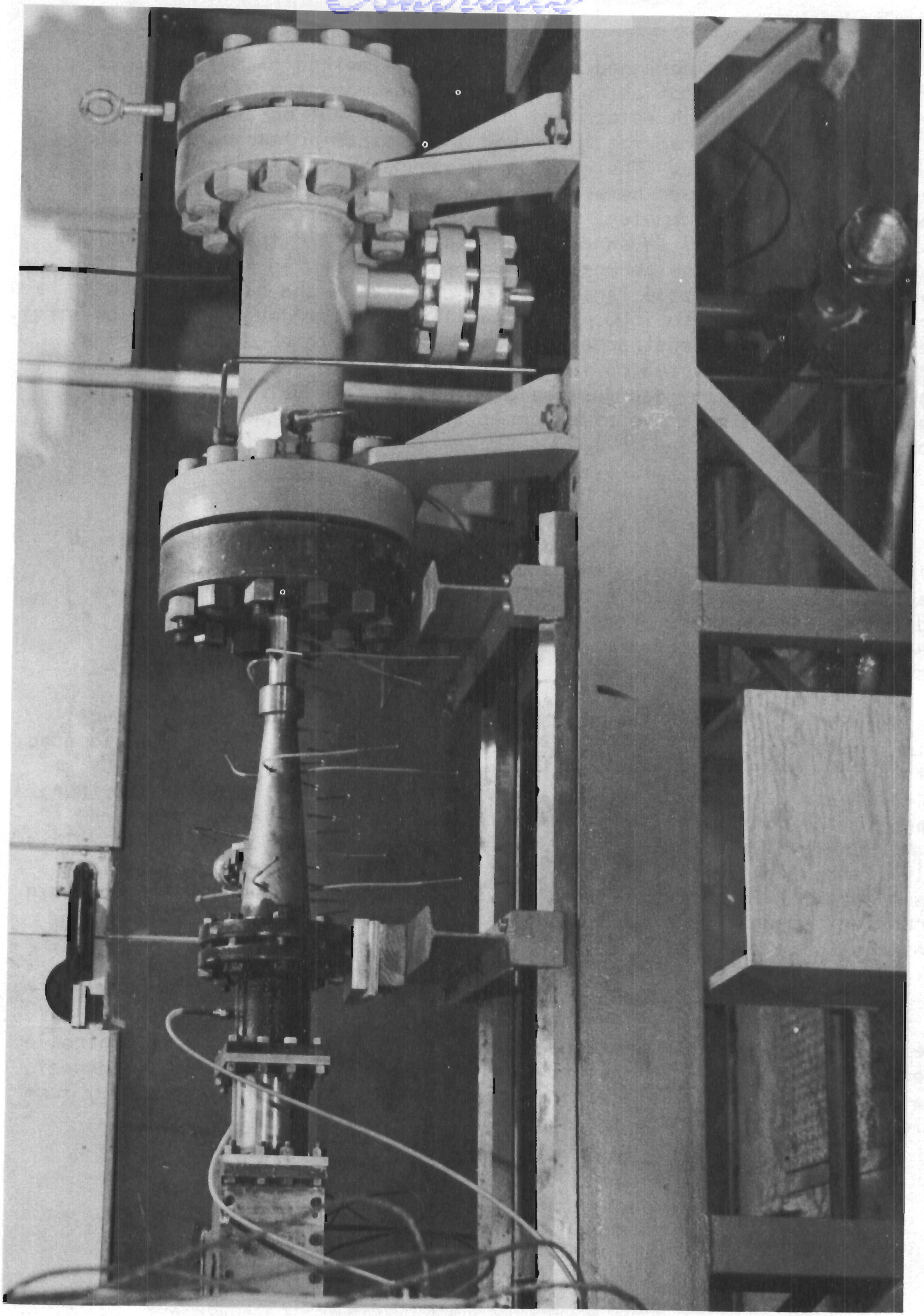


Figure 7. Settling Chamber of the Helium Hypersonic Wind Tunnel

C EJECTOR

The ejector was designed according to the method of Reference 23. This method consists of using one-dimensional momentum balance through the mixing region which is assumed to be at constant pressure. By this method the area required for the mixed and unmixed primary and secondary flow can be calculated. The constant pressure mixing section is then made as a straight taper between the two areas and about seven times as long as the minimum diameter. This section is then followed by a constant area section, seven diameters in length, and then an expanding subsonic diffuser. The general arrangement is shown in Figure 8. The supersonic primary nozzle is located on centerline and high pressure air at a pressure up to 1500 psi is supplied to it through the hollow strut. The outside duct is constructed of sheet steel in three sections. The whole assembly is mounted on two carriages which roll on the rails. A blow off valve is provided just before the ejector duct to prevent the pressure building up above atmospheric if the ejector should become blocked.

D PIPING AND CONTROLS

The piping system has been made of copper pipe and brass fittings silver soldered together. This follows standard high pressure piping practice. Controllers are provided for the settling chamber pressure and the ejector primary pressure. Pneumatic controllers operating pneumatic valves have been used in both cases. A diagram of this system is shown in Figure 9.

Helium is piped to the settling chamber from a semi-trailer truck, which contains about 35,000 cubic feet of free helium at 2500 psi, located outside the building. The truck is connected to the piping system by three one-half inch high pressure hoses in parallel. The helium is then piped to the control valve. The control valve is a three-eighth inch Hammel-Dahl valve operated by a diaphragm type pneumatic positioner. This valve is controlled by a Bristol Controller with a variable automatic reset rate. This controller and valve combination can be used to maintain a constant stagnation pressure and vary it at will from atmospheric pressure up to 1500 psi.

High pressure air (3000 psi) from the air storage system is piped to the ejector control valve which is similar to the settling chamber valve. The ejector primary air pressure is regulated by a Hanlon-Waters controller operating this valve. A one and one-half inch flexible metal hose connects the control valve to the ejector primary nozzle. The control console, from which one man can operate the tunnel, is shown in Figure 10.

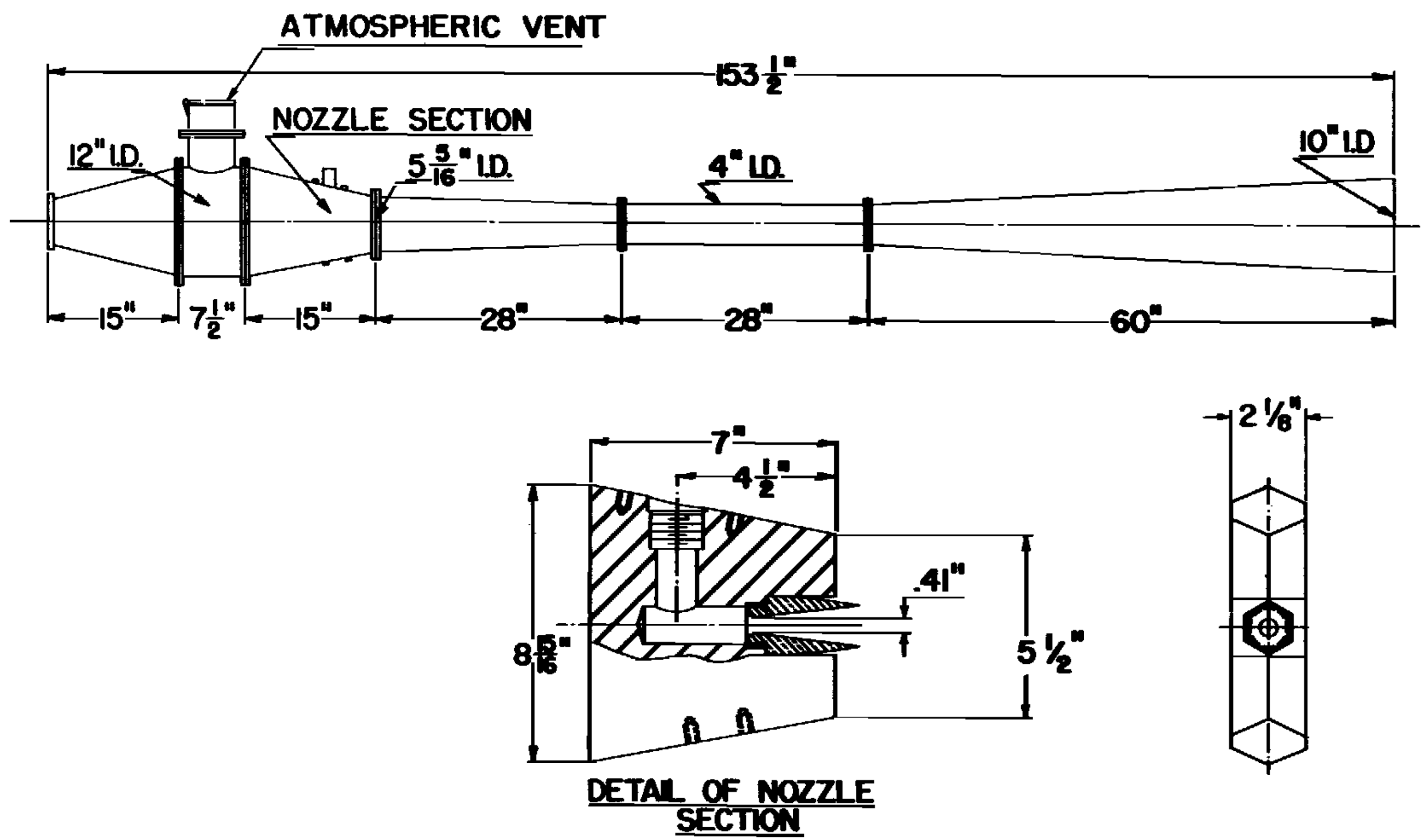


Figure 8 Lay-out and Details of Ejector System

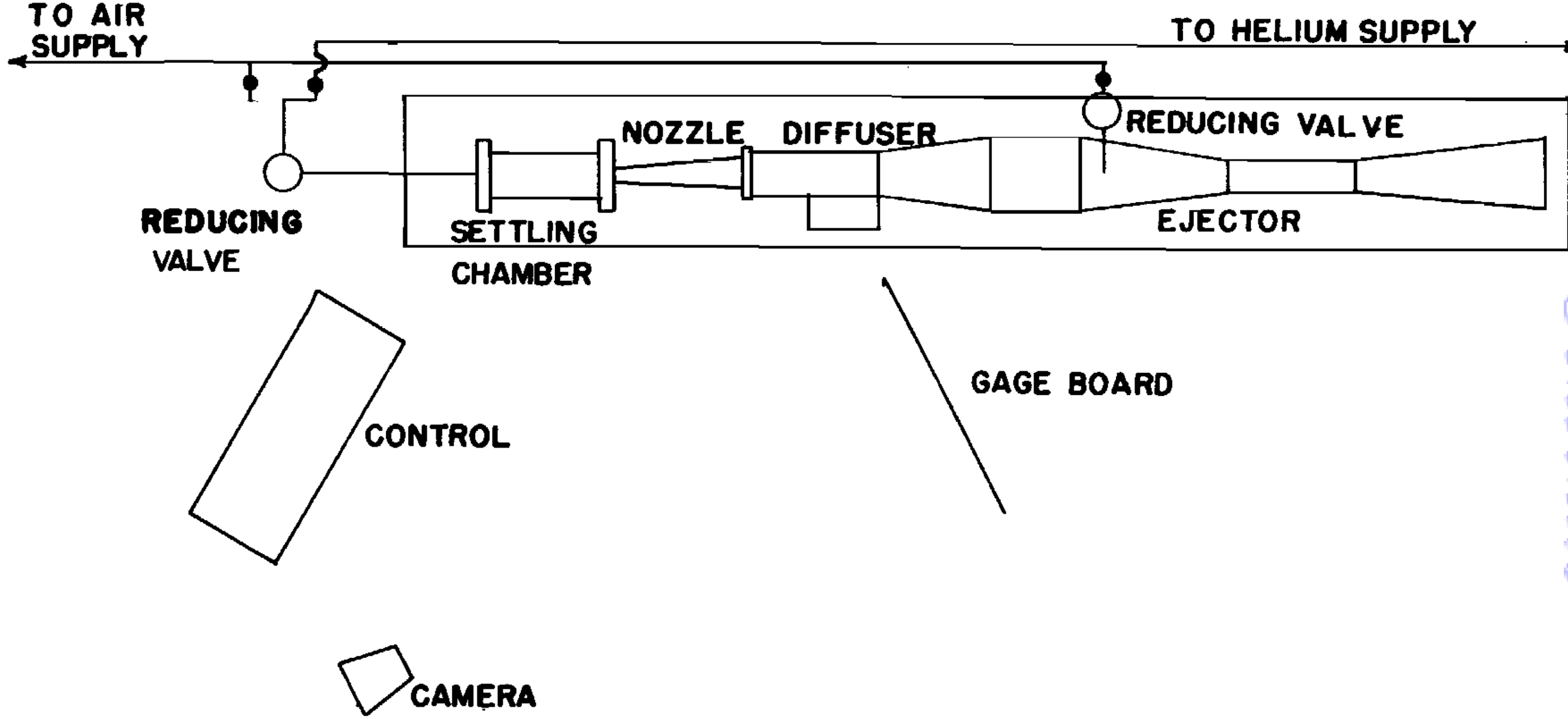


Figure 9 Piping and Control System for the Helium Hypersonic Wind Tunnel

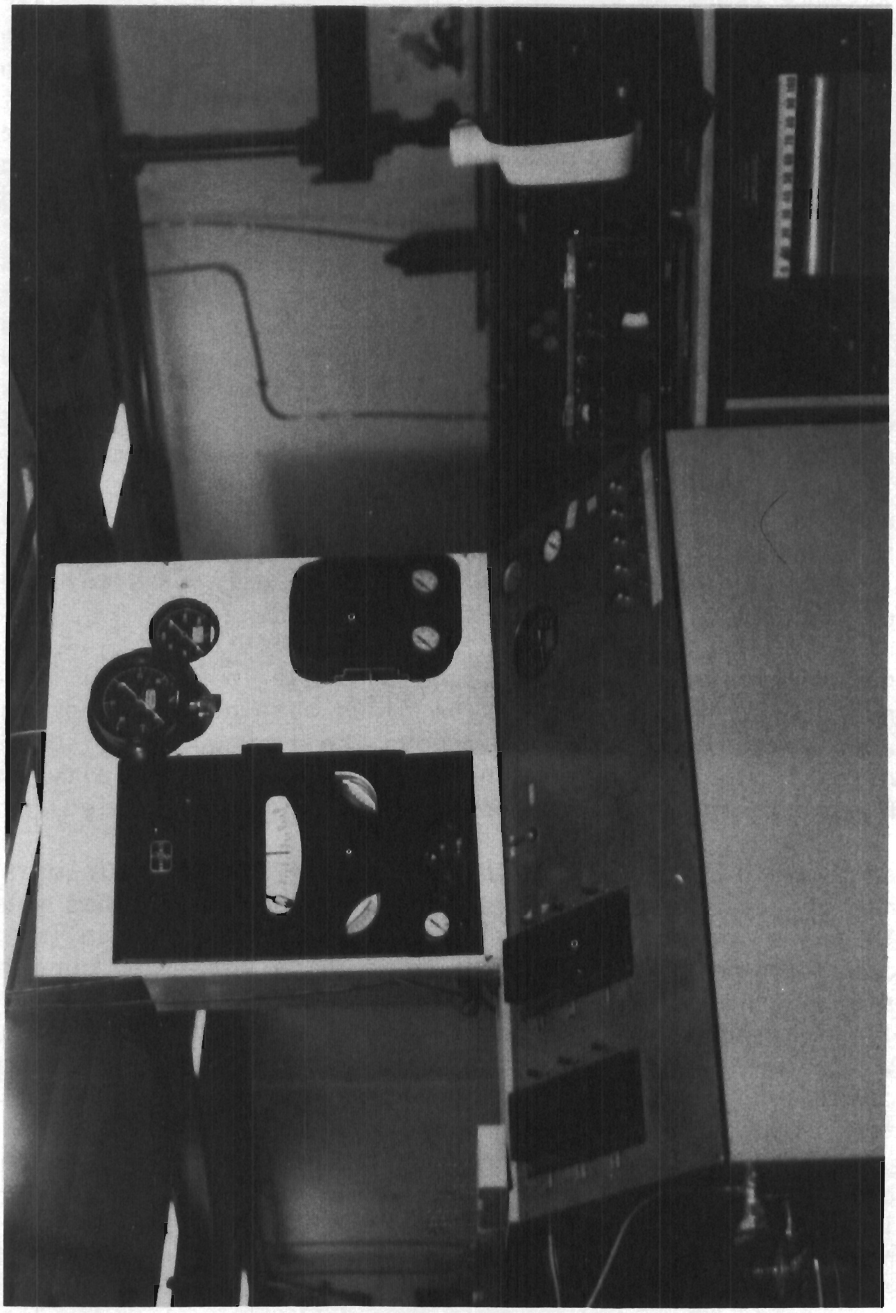


Figure 10. Control Panel for Helium Hypersonic Wind Tunnel

E NOZZLES, TEST SECTION, DIFFUSER AND SURVEY SECTION

It is standard practice to design wind tunnels with rectangular or square test sections. This is done primarily so that only two of the four surfaces involved must be contoured to obtain an essentially uniform flow in the test section. The design of such nozzles are relatively standard using two-dimensional characteristic methods or several available analytic methods. The contours developed must, of course, be corrected for boundary layers. The two flat walls also provide a simple window installation for visual and optical observation. As the Mach number is increased, however, considerable difficulties are experienced because of the sealing problem across the throat, the very narrow high aspect ratio throat, and subsequent "secondary" flows in the nozzle. These flows are caused by the high pressure gradients normal to the flow direction which occur at very high Mach numbers because of the very small angle between the Mach lines and the flow direction. These gradients cause a piling up of the boundary layer along the centerline of the two unprofiled tunnel walls. The use of helium instead of air alleviates the problem somewhat, since smaller expansion angles are needed to obtain a given Mach number and the throat size is considerably larger for a given test section size. Successful air tests have been made using a simple wedge nozzle at Mach numbers up to approximately 8 (Reference 1).

Preliminary tests were made with such a nozzle in a tunnel designed for a four-inch square test section. The flows obtained were not satisfactory and when several attempts to improve the flow were not successful, the two-dimensional nozzle was temporarily dropped and an axially symmetric nozzle, which had already been designed and built, was installed.

The axial symmetric nozzle eliminates the difficulties of seals around the throat area and of secondary flows. This type of nozzle has several drawbacks: It makes the installation of windows and flow visualization techniques more difficult. There is the difficulty that the round test section makes evaluation of optical studies - such as interferograms - more difficult. In general, such nozzles do not provide a uniform flow test section. The original axial symmetric nozzle is shown in Figure 11. This nozzle consists of a curved section, leading into the throat of 0.20 inches diameter, joined to a straight conical section which expands to 4.125 inches diameter. This expansion ratio would give a Mach number of 19 based on a one-dimensional area ratio calculation. The nozzle was constructed in two sections screwed together with an "O" ring seal. This two piece construction was used to facilitate the machining of the nozzle. The conical section has a half cone angle of 5° 30'. A 4.125 inch diameter straight diffuser section six inches long, is attached to the back of the nozzle. This round section is abruptly joined to the square survey section. No transition section was felt to be required since the shock structure was expected to be in the diffuser section and only low subsonic velocities experienced at the junction. The nozzle was run initially with no second throat restriction. Static pressures were obtained which indicated that the flow was separating from the nozzle after only a small expansion. Several conical bodies were made, Figure 12, and were supported in the diffuser section by a sting. By using the

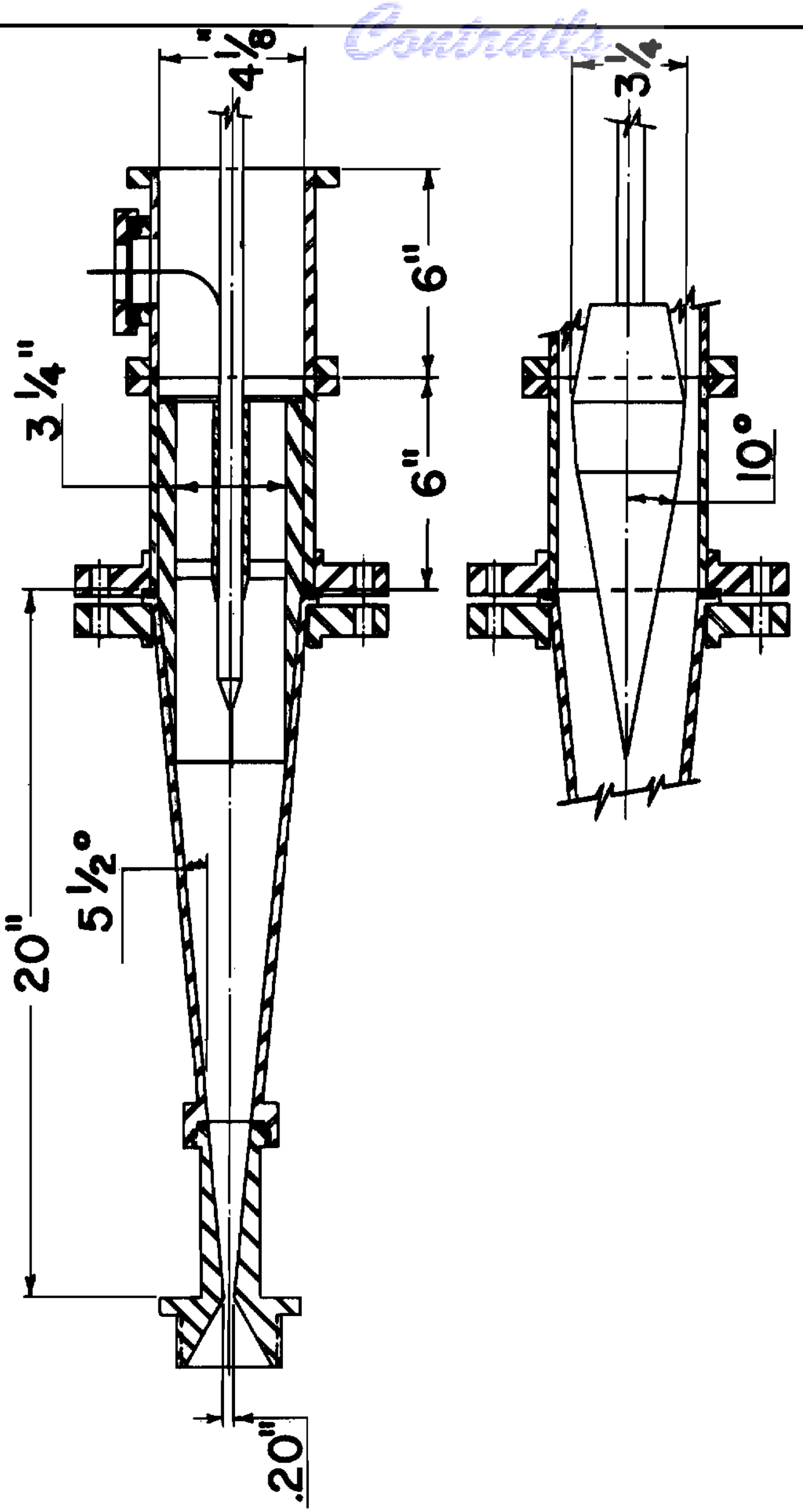


Figure 11 Cross Section of Axial Symmetric Nozzle and Diffuser

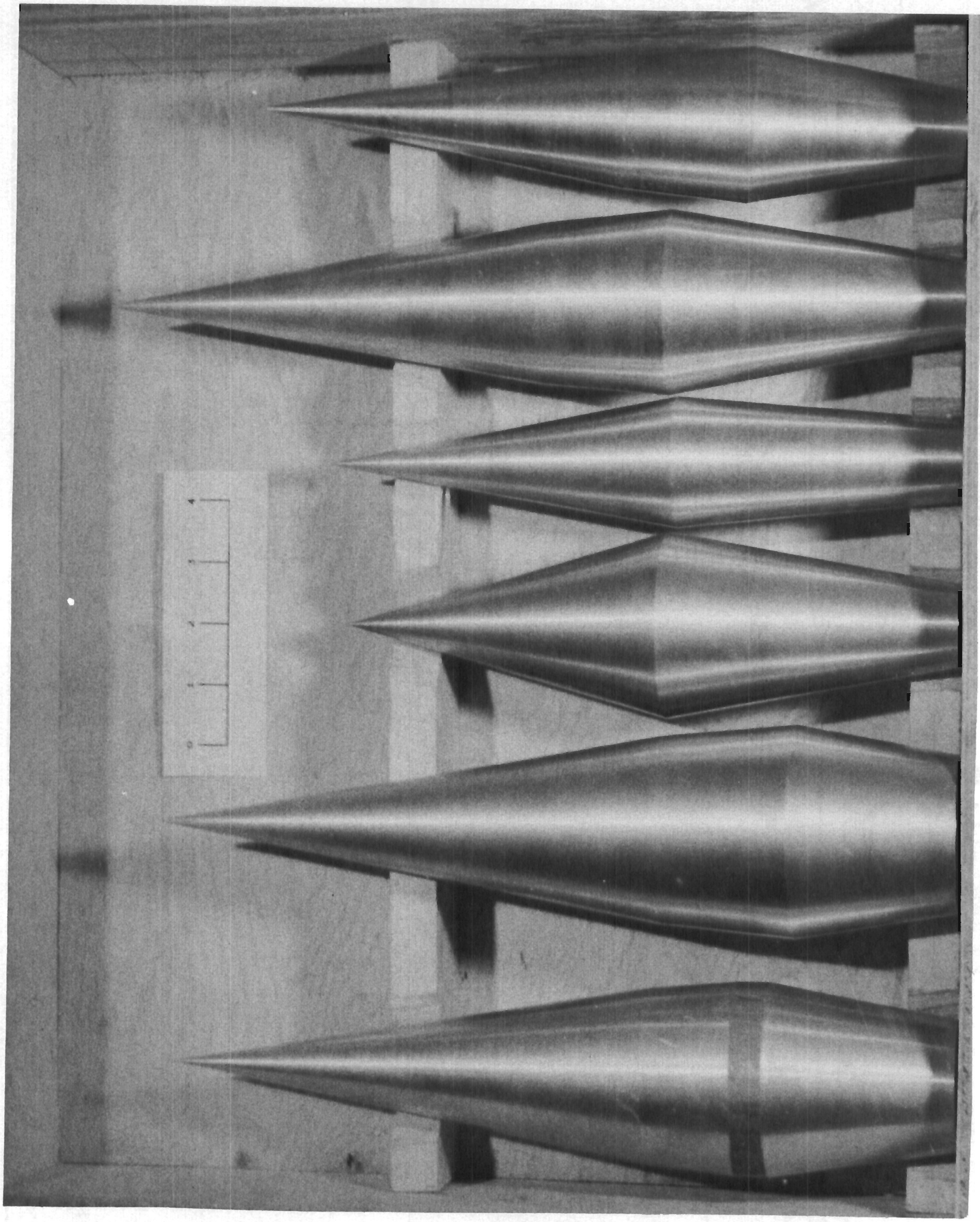


Figure 12. Central Body Diffuser Shapes Tested in Helium Hypersonic Wind Tunnel

cones shown in Figure 11, a wall static pressure distribution shown as crosses (+) in Figure 13 was obtained. A crude method of adjusting the second throat area was obtained by moving the cone into the narrower part of the tapered nozzle. This method of changing area has the disadvantage that the point of the cone enters so far into the nozzle that it is not practical to use when model tests are being made.

Separation was still indicated near the back of the nozzle and so this part was of little use for testing. Since the separation occurred at about a diameter of three inches, a round bushing, 3.25 inches inside diameter, was fitted to the conical nozzle to reduce the maximum diameter. In order to assure that the centerline sting was on axis, a hollow central body supported by three legs (usually called a spider) was built which was fitted into the bushing and through which the center-line sting could slide. This arrangement is shown in Figure 11. The sting was supported by the spider and driven by the screw in the survey section. Another round section was built and placed between the survey section and the diffuser section in which a fitting for removing pressure leads from the tunnel was incorporated. This provided a means of removing the pressure leads with much shorter lengths of tubing than in the original system. Tests of static pressures on the side wall yielded the distribution shown as circles (o) in Figure 13 which was the best distribution obtained up to this point.

The major part of the test program was carried out in the axial symmetric nozzle described. No optical studies could be made because no windows were installed. A second nozzle, similar in design to the first with the exception of a window installation was then contructed and tested. The windows were installed by cutting off the sides of the conical nozzle along two parallel planes and sealing glass plates to the opened sections (see Figure 14). The flow disturbances caused by the glass did not seem to interfere with the flow in the test area due probably to the very low Mach angles experienced at these speeds. Complete surveys of this nozzle have not been made and it has been used only for the optical observations.

The development of this nozzle and diffuser arrangement was not carried out with the idea of obtaining the best possible arrangement but only to develop a workable system in which the tests could be made. A better diffuser and, therefore, higher Mach numbers could undoubtedly be developed if a program were undertaken to do so.

The section following the nozzle was designed to hold models and survey equipment and to move this equipment in an axial direction. This section is shown in Figure 15. The section consists of a four inch square box with a slot in the middle of each side wall. One of these slots goes through the side wall into an air tight chamber mounted on the side wall. This chamber contains a lead screw and numerous tubes through the wall for taking pressure leads out of the tunnel. The screw passes through a pressure tight gland in the wall and is driven by an electric motor operated from the control panel. The position of the screw is indicated by a digital counter, geared to read thousandths

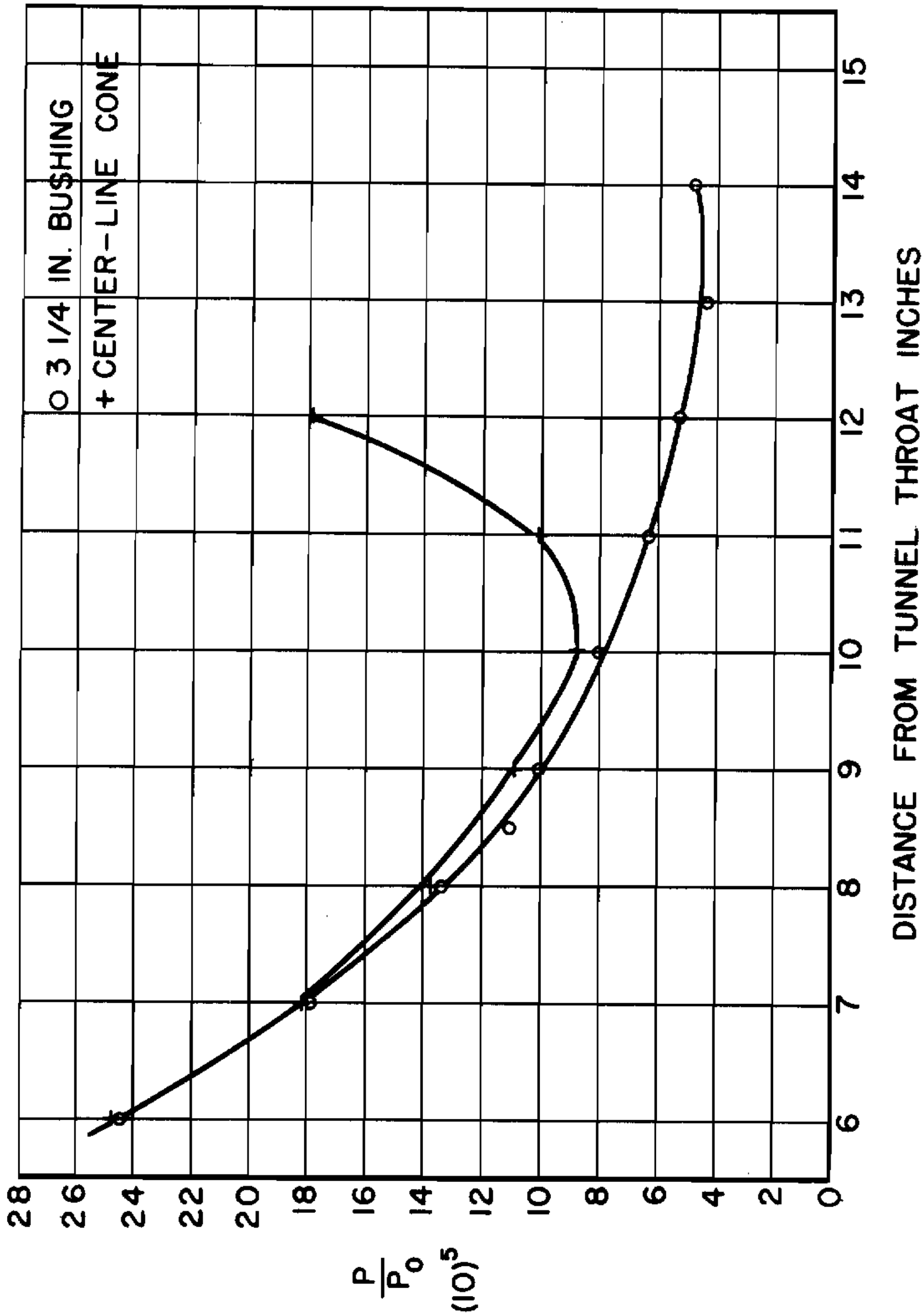


Figure 13 Wall Static Pressure Distribution Along the Axial Symmetric Nozzle

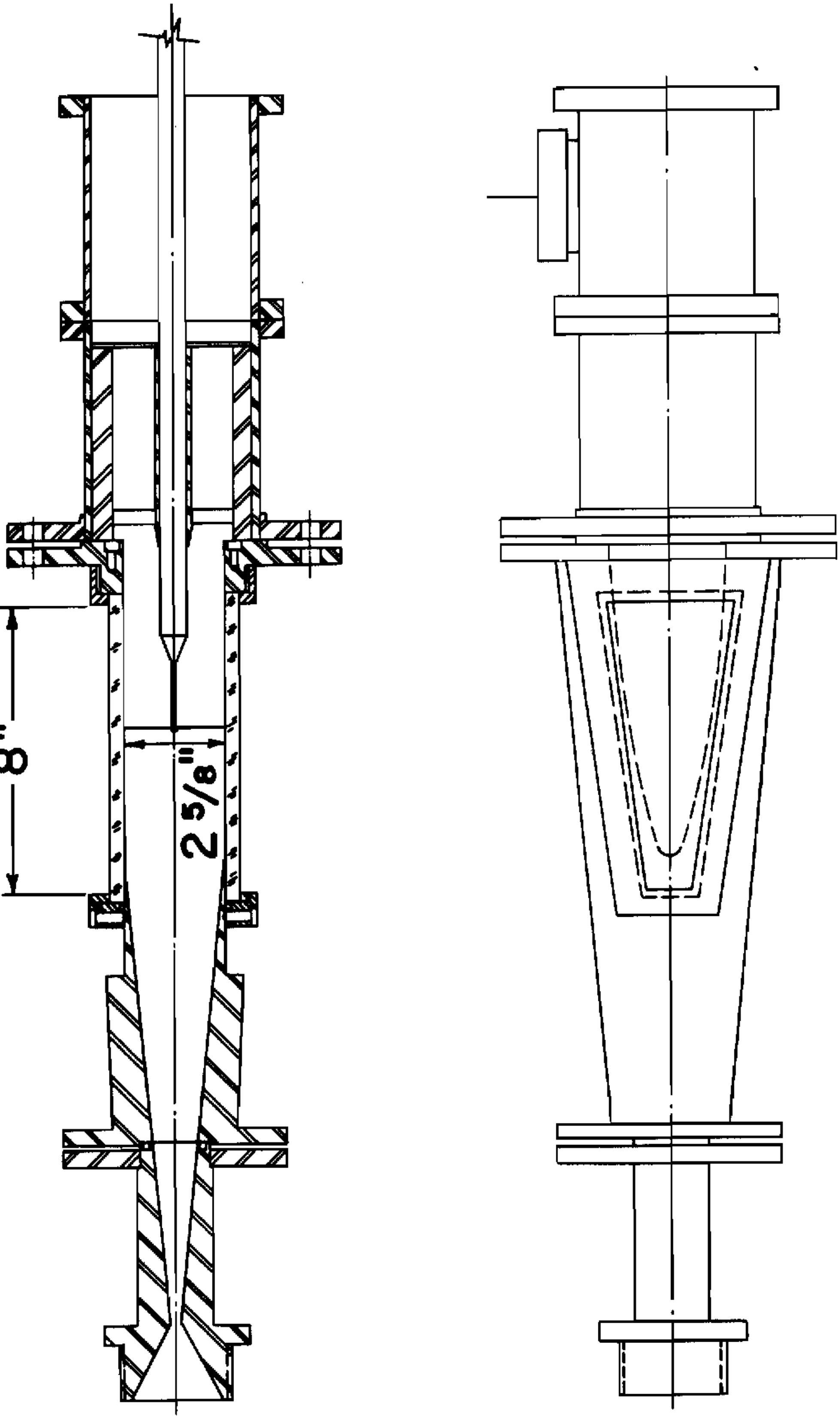


Figure 14 Window Installation in the Axial Symmetric Nozzle

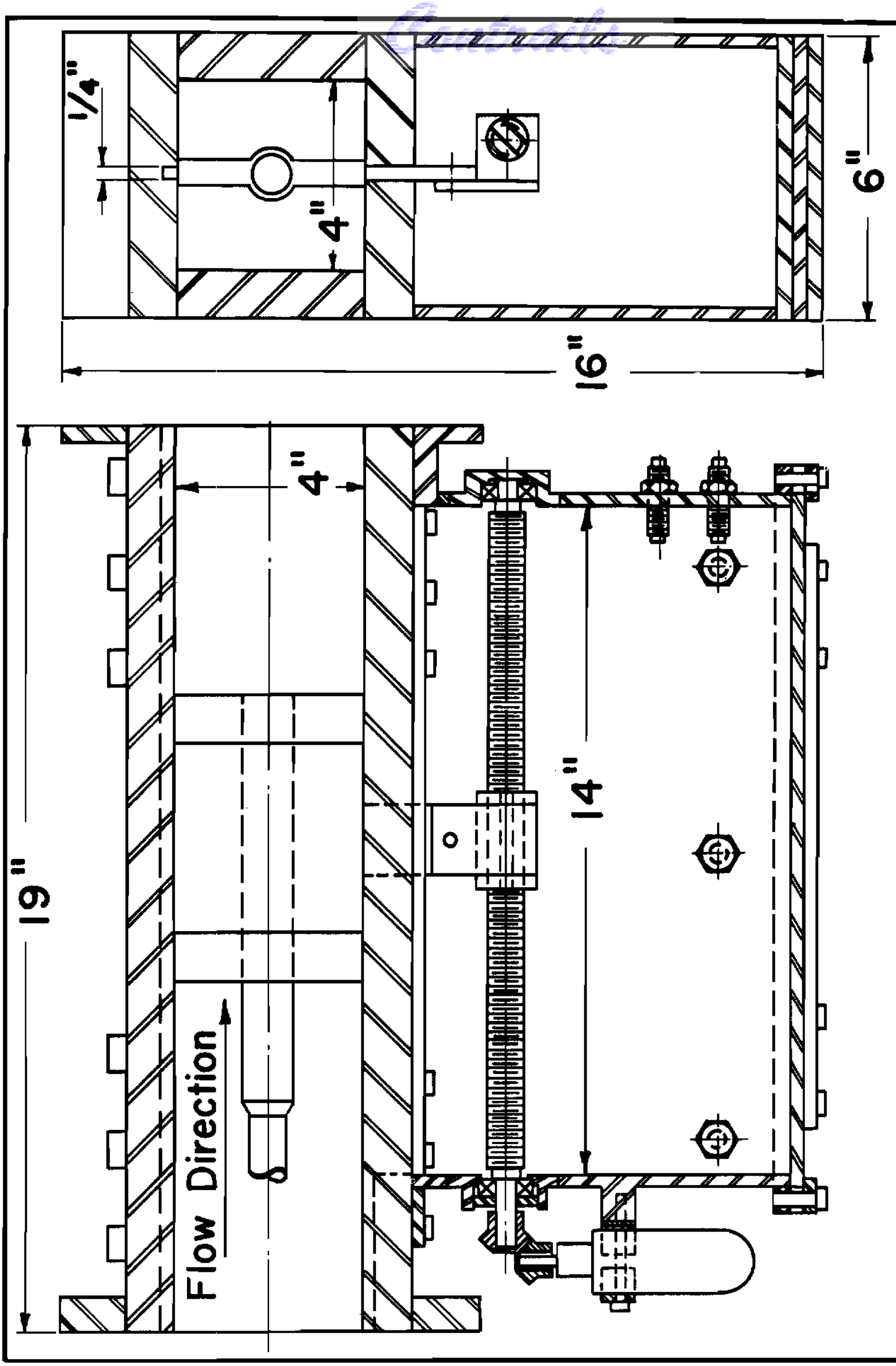


Figure 15 Axial Drive System for Models and Survey Equipment

of an inch. A similar counter is located on the control panel, and driven by a pair of selsyn motors. The system provides a nine inch travel in the axial direction.

V INSTRUMENTATION

The measurements which have been made in the hypersonic tunnel were chiefly pressure measurements. A few temperature measurements have been made and some preliminary work has been done with optical methods of flow visualization.

A PRESSURE

The pressure ranges in which instrumentation is required can be broken up into three main bands. For this tunnel, the ranges of these bands are:

Stagnation pressures	500	to	1500	psia
Total head	30	to	6	psia
Static pressure	2.0	to	.02	psia

The first two categories are in the ranges in which the instrumentation used on the other wind tunnels at Princeton is capable of operating. This instrumentation consists of dial type gages. The high pressure gage used is of the bourdon-tube type and is manufactured by Heise and the lower pressure gages are aneroid gages similar to aircraft instrumentation and manufactured by Kollsman. Two types of this latter gage are available. One is an absolute pressure gage which measures from 0 to 200 inches of Hg and the other a differential gage which has a range of 0 to 300 inches of water. These instruments are capable of measuring down to one psia with good accuracy.

The special instrumentation which had to be obtained for the hypersonic wind tunnel was for the range below one psia. Considering the length of running times available, on the order of several minutes, it was necessary to have some type of gage which could lend itself to rapid recording by a semi-automatic system. It was also highly desirable that the time constant for the tubing and gage combination be kept short. For this range, electrical pressure transducers made by the Statham laboratories and a multi-tube oil manometer, similar to designs furnished by the Naval Ordnance Laboratory, have been used.

The pressure transducers chosen were of the differential type with full-scale range of 0.2 psia. These gages are used by attaching one side to a reference vacuum and connecting the other to the desired pressure orifice. If the reference vacuum is maintained sufficiently low, it need not be known accurately. A McLeod gage was used to determine the value of this vacuum. The pressure transducers are not capable of withstanding a full atmosphere of differential pressure so some means must be provided to prevent this pressure from being applied during the starting-up and shut-down period of the tunnel operation. This is accomplished by the set-up shown in Figure 16. During starting and stopping, the reference vacuum side of the system is attached to the wind tunnel at some point which will have a pressure about the same as the pressure orifice being read. When the tunnel starts and the pressure in the tunnel has reached a sufficiently low value, the three way solenoid valve is tripped by the differential pressure switch connected to the reference vacuum. The output of the Statham gages is recorded on a self-balancing multi-channel potentiometer equipped with a digital read-out system.

Initial difficulty with these gages was found when the casing was subjected to high vacuum. The gages experienced a rather rapid and continuous zero shift at the high vacuum which they did not seem to have at a slightly lower vacuum. The zero reading depended, to a marked extent, on the pressure in the casing. It was found that this effect was caused by the poor heat conduction from the wires of the unbonded strain gage at a low pressure. To overcome this difficulty, the transducer casing was filled with low vapor pressure oil. The casing design was modified somewhat to make this practical. Special precautions are required to prevent the oil from bubbling out. With this modification, the gages behaved satisfactorily under high vacuum.

Some means was required to calibrate the transducers. An oil manometer was chosen as the most satisfactory means of doing so. With a special optical system for reading the manometer, repeatable readings could be made to 0.001 psi. By using a non-viscous oil (Dow Corning 200 fluid, 10 centistokes) the manometer did not have an excessive time lag and was quite satisfactory for our purposes. The calibrating set-up and the oil manometer are shown in Figure 17.

The multi-tube oil manometer used for reading static pressure is shown in Figure 18. This is a ten tube manometer with two reference tubes. A total differential pressure of about 1 psi may be read with accuracy of .0025 psi. The readings are recorded photographically. For low pressure readings, the manometer is read against a high reference vacuum, but it may be used in all ranges by varying the back pressure in the reference tank. A valve bank is provided so that each tube is attached to the reference vacuum in the tank on the back of the manometer in one position and may be switched to the desired pressure tube in the other position. The valves are of a sliding piston type and are sealed with "O" rings. They are all operated together from an eccentric on the top of the valve bank.

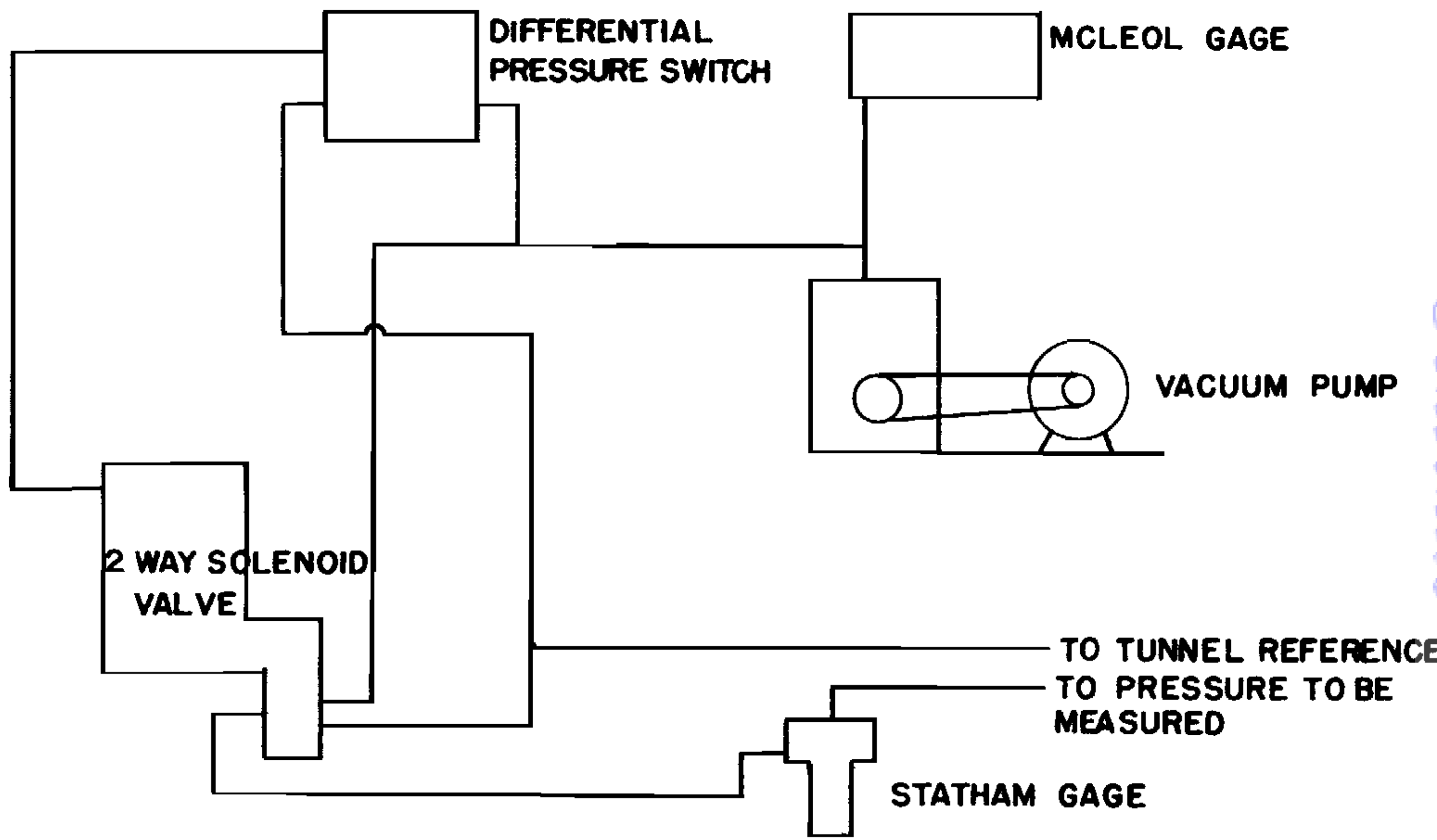


Figure 16 Piping Diagram for the Statham Gage System

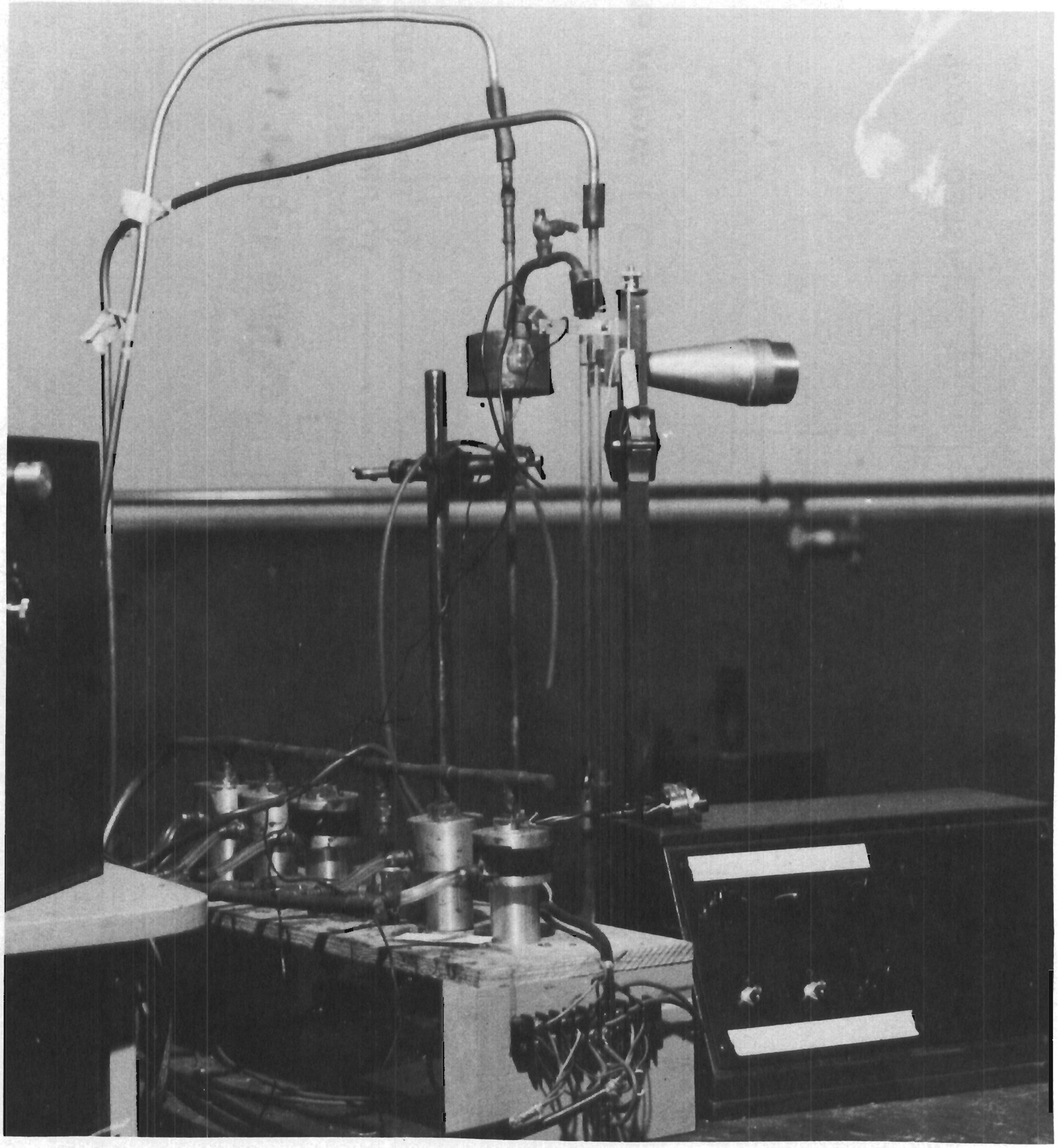


Figure 17. Statham Gage Calibrating Set-up

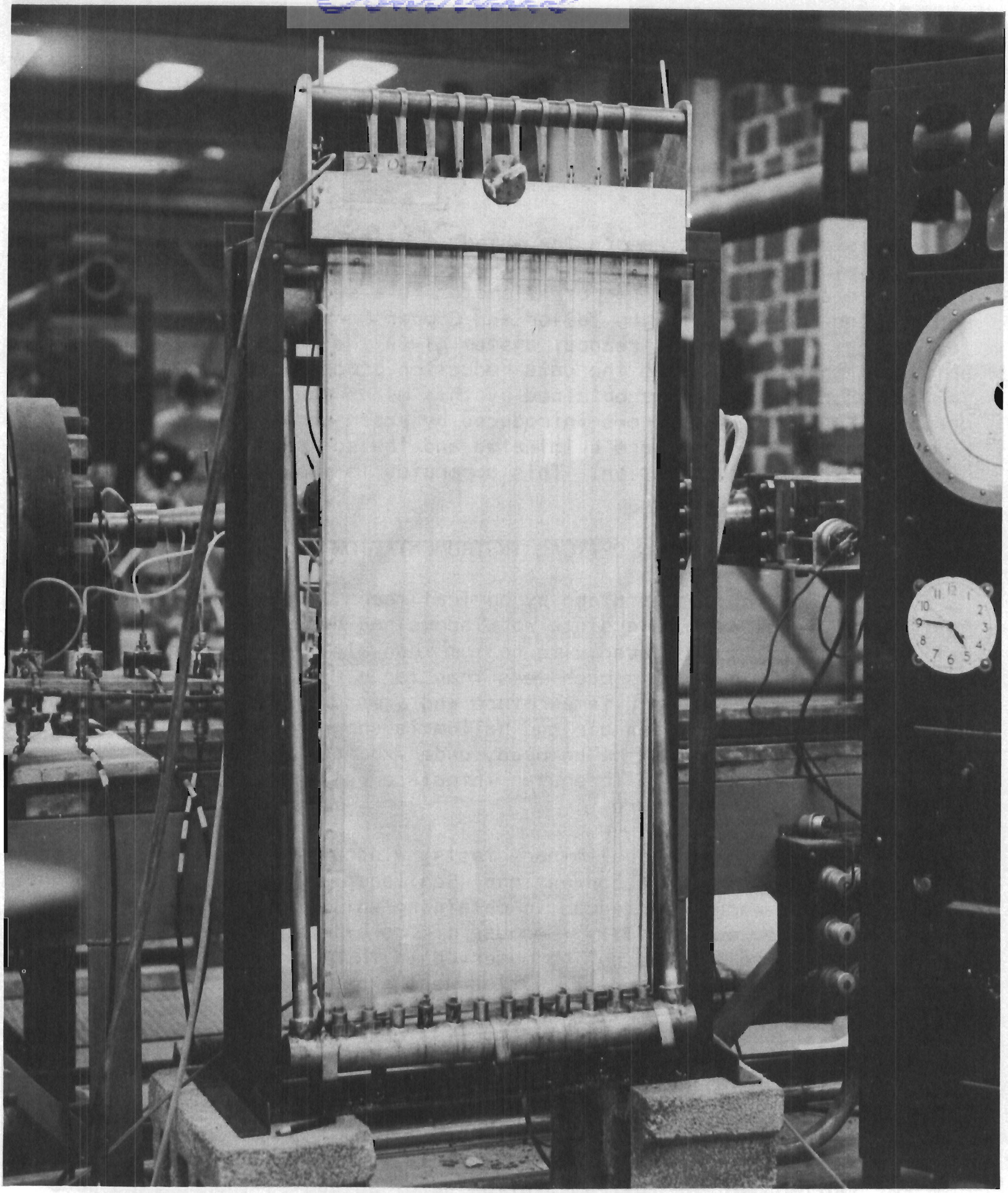


Figure 18. Ten Tube Absolute Pressure Oil Manometer

B TEMPERATURES AND RECORDERS

Stagnation temperature and nozzle wall temperatures are the only temperatures which have been measured. The stagnation temperature was determined by a pencil type thermocouple in the settling chamber. A thermocouple was also imbedded in the wall of the three dimensional nozzle. A complete program of surface temperatures and recovery temperatures has not yet been undertaken.

The system used to record the output of the pressure transducers and thermocouples is a multi-channel self-balancing type potentiometer with a digital readout system. The potentiometer is a standard Brown Electronic and has been modified by the Taller and Cooper Corporation to give a digital readout. The digital readout system gives the results in a very readable form which simplifies the data reduction process. Greater accuracy in the recording system is obtained by this means than with the usual graphical system. The errors introduced by paper shrinkage, which can be as much as 5 percent, are eliminated and the scale is read to 0.1 percent without interpolation. This apparatus is shown in Figure 19.

C OPTICAL INSTRUMENTATION

No information was obtained by optical means in the first series of tests in the conical nozzle since no windows had been installed. Higher optical sensitivity is needed with helium than air since the optical index of refraction for helium is much less than for air (1.000036 for helium and 1.00029 for air at standard temperature and pressure). An indication of the relative difference between air and helium is shown by Figure 20 which shows two Schlieren photos taken of an open, underexpanded jet using air and helium. In this case, the irregular interface between the helium and air further confuses the picture.

Near the end of the preliminary tests, windows were installed in the axial symmetric nozzle. A conventional Schlieren system was installed and no difficulties were experienced in obtaining satisfactory Schlieren photographs (see Section VII). Because of the extremely high gradients involved, fine detail could not be observed in these preliminary photographs. The fact that no difficulties were experienced is due primarily to the very low static temperature which results in a considerable density even though the static pressures are quite low. It is possible to see large density changes in the flow, but high sensitivity is not to be expected because of the low index of refraction of helium.

D OTHER INSTRUMENTATION

In addition to the instrumentation detailed above, the nozzles were provided with survey equipment for studying the boundary layer, both on the nozzle wall and on the models. One station along the nozzle was provided with a micrometer drive perpendicular to the axis which could

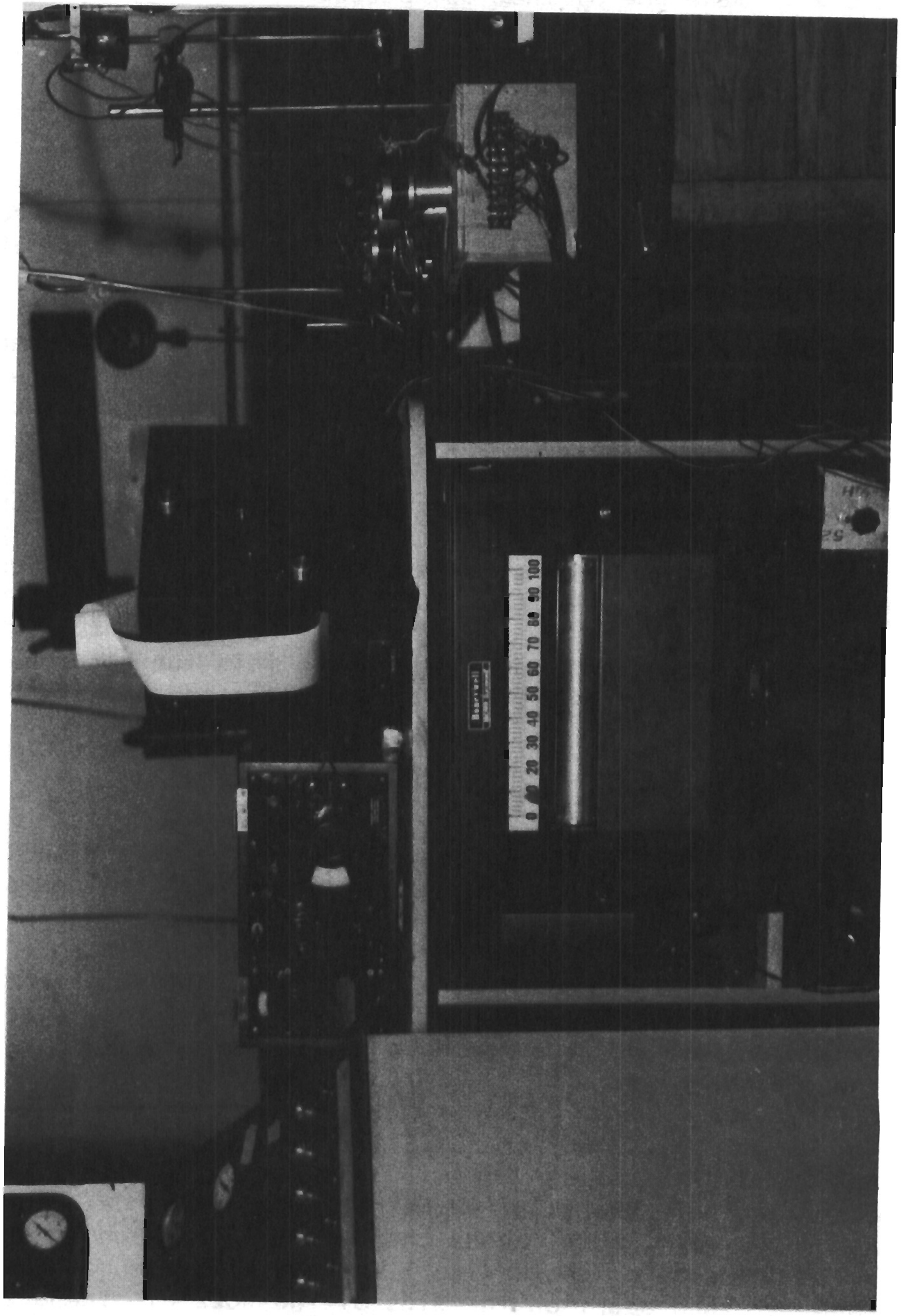


Figure 19. Multi Channel Potentiomer With Digital Read Out

be used to drive various length total head or static pressure survey tubes. An interferometer, with a five inch field, is also available as well as a one component strain gage balance for body drag determination. Preliminary work on all of these types of instruments has already been carried out and further programs will make use of them as needed.

VI TUNNEL PERFORMANCE AND OPERATION

A OPERATIONAL PROCEDURE

The operational procedure for making a test in the hypersonic tunnel is as follows: The ejector is started first and brought up to operating pressure of about 1200 to 1300 psi (primary jet stagnation pressure). This drops the pressure in the tunnel to about five inches of mercury. The ejector operates unsteadily and surges at primary pressures of around 1000 psi and no secondary flow and is brought through this pressure range quickly. After the ejector has stabilized, the helium valve is opened and the tunnel stagnation pressure is brought up to approximately 1000 psi. The tunnel starts by the time the helium pressure reaches 1000 psi and the various static pressure instrumentation may be turned on. The helium pressure may now be adjusted to the desired value. The time from the starting of the helium flow to stabilized conditions is about 20 seconds. To shut down the tunnel, the static pressure instrumentation is turned off. The helium is turned off and then the high pressure air to the ejector. Runs up to approximately ten minutes in length may be made in this tunnel. The helium capacity of the trailers is the limiting factor and not the high pressure air supply.

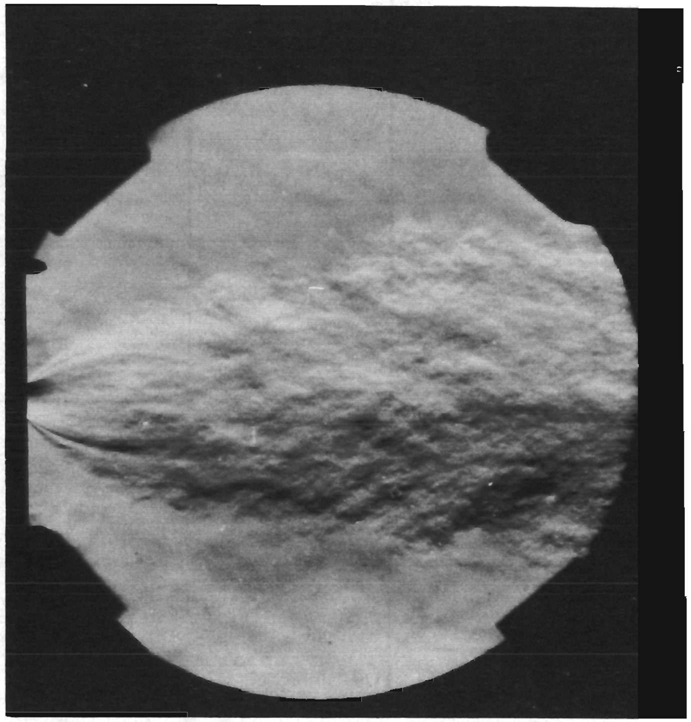
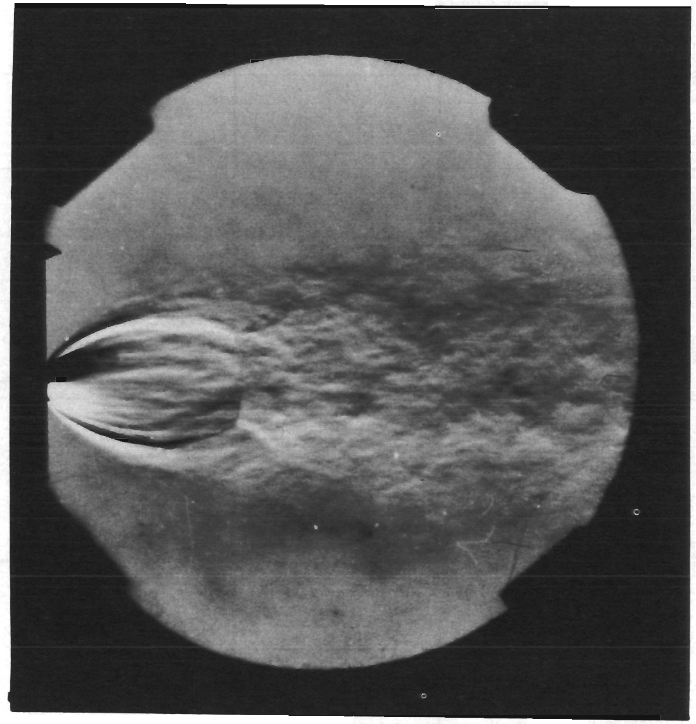
B EJECTOR AND TUNNEL PERFORMANCE

The performance of the ejector as operated with the wind tunnel is shown in Figure 21. The best overall pressure ratios are obtained around a helium pressure of 1000 psi and this is the reason for starting the tunnel at this condition. The Mach number-Reynolds number range at various stagnation pressure is shown in Figure 22 for the present diffuser-ejector arrangement. There is a fairly wide range of Reynolds number if tests are to be made at a low Mach number, around 11 to 12, but a very small range at Mach 15.

C NOZZLE SURVEYS

Survey measurements in the axial symmetric nozzle have been made of static pressure on the wall, total heads on centerline and across the tunnel, and static pressure on centerline. The static pressures were measured at stations one inch apart on the side of the nozzle. These are shown in Figure 13.

A survey was made with a centerline total head tube and a seven tube total head rake. The pressures as found by the rake are plotted in Figure 23. On the part of the survey near the throat, the diameter of the nozzle

Helium   Air

Figure 20. Schlieren Photographs of an Over-Pressured Sonic Jet Using Helium and Air, Stagnation Pressure = 1000 psi

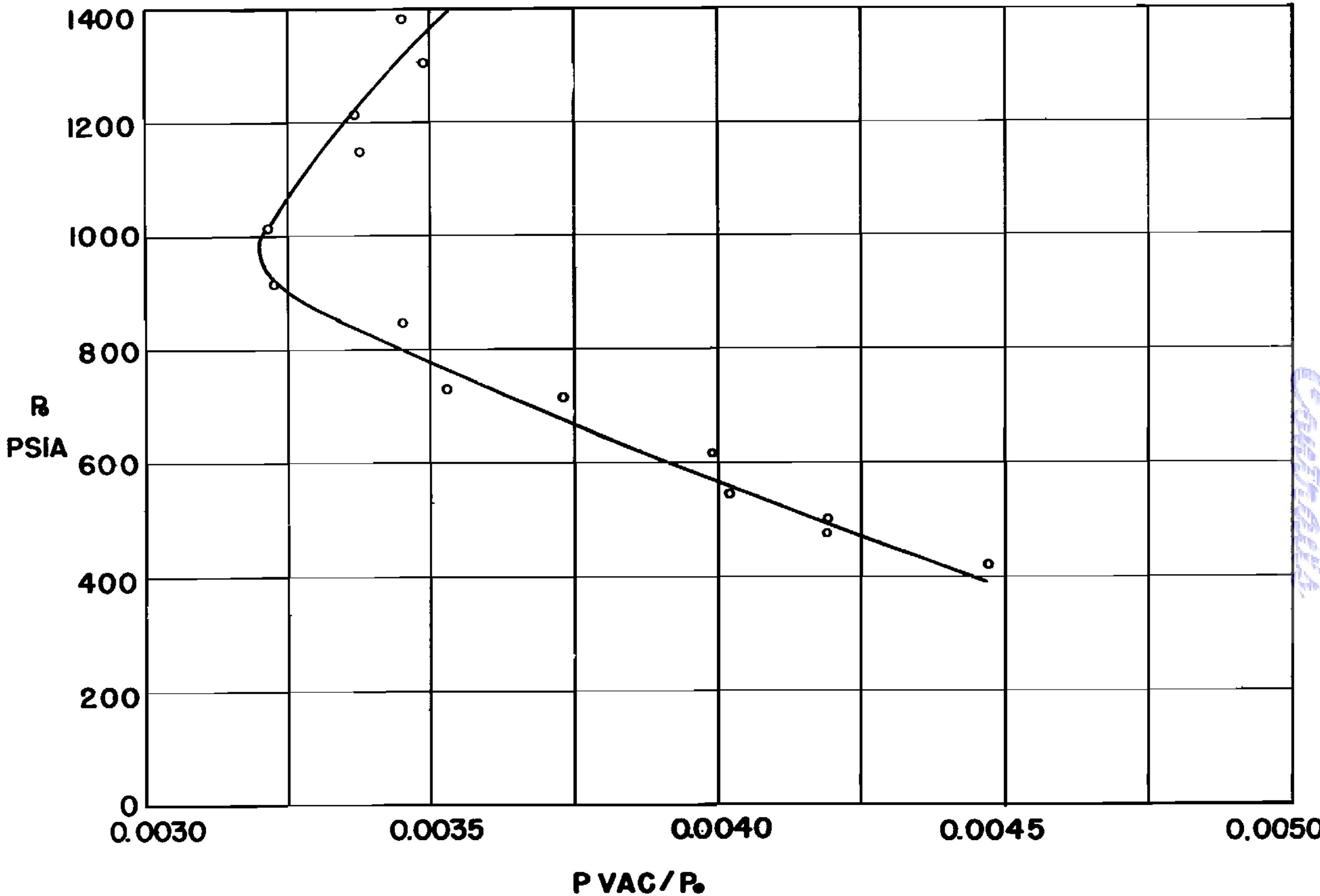


Figure 21 Pressure Ratio Characteristics of Ejector-Tunnel Combination

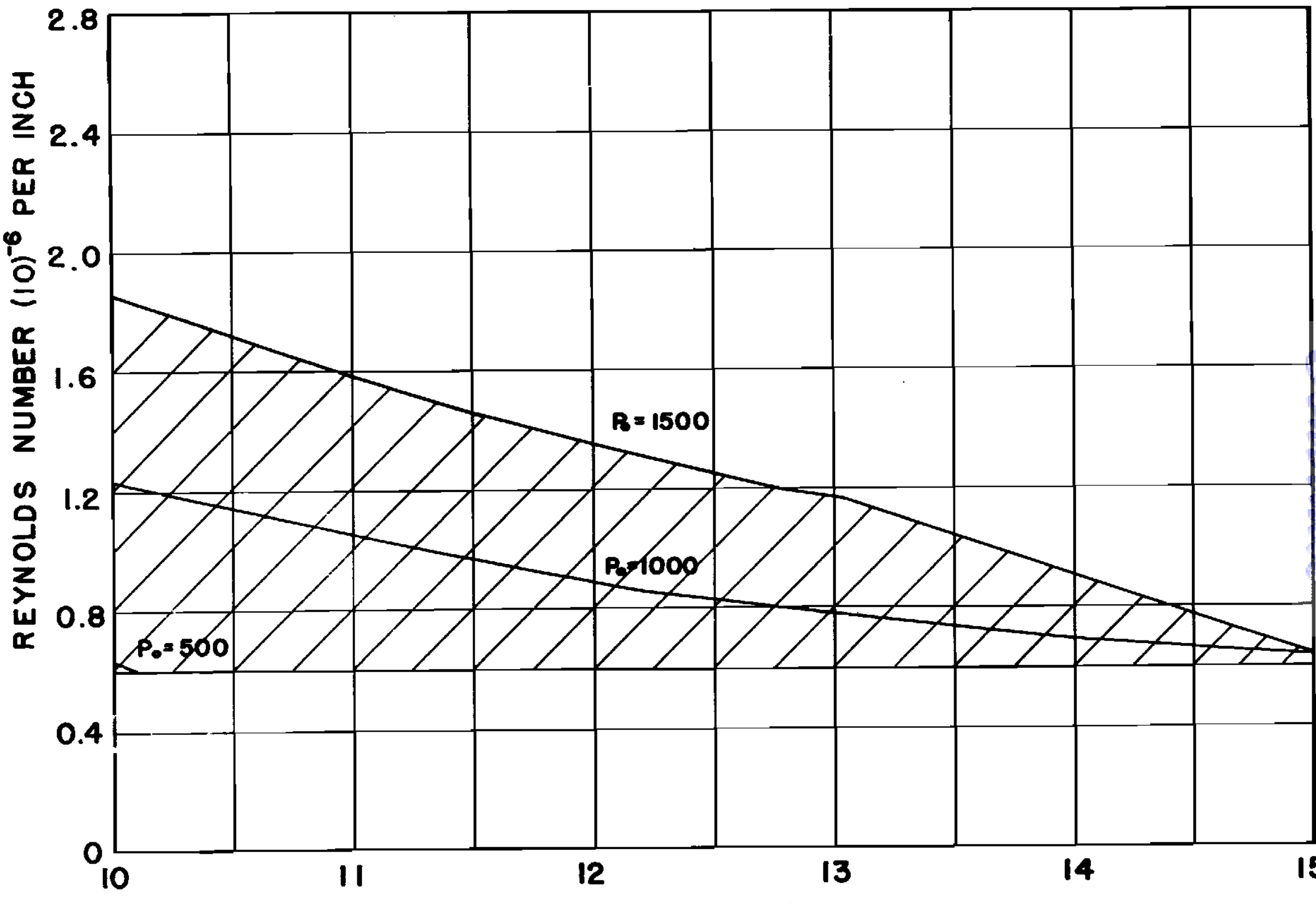


Figure 22 Reynolds Number-Mach Number Range for the Helium Hypersonic Wind Tunnel with Present Ejector

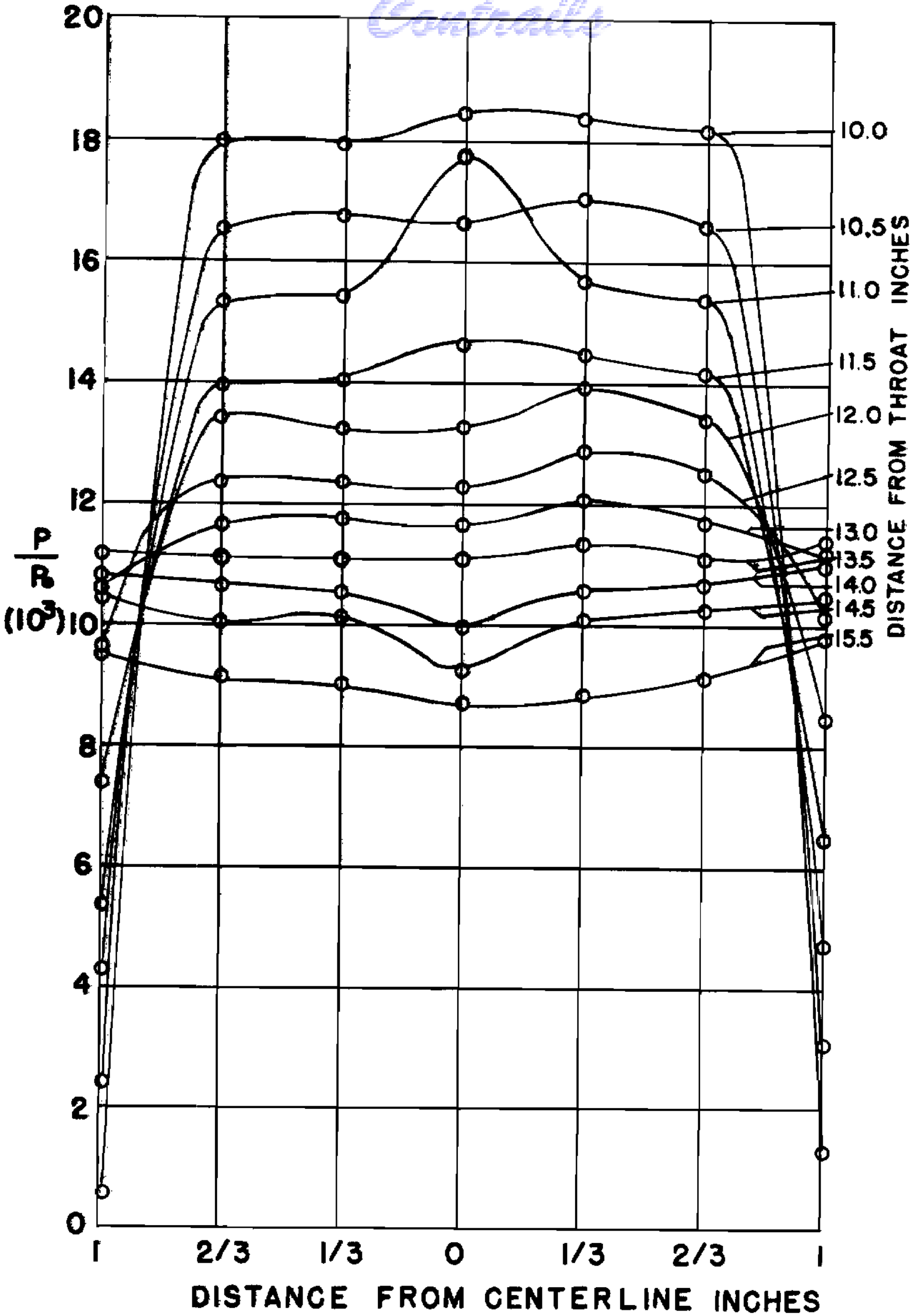


Figure 23 Cross Tunnel Total Head Surveys for Various Axial Positions Along the Symmetric Nozzle

is such that the outside tubes are in the boundary layer. As the rake is moved away from the throat, the diameter becomes larger and the side tubes move out into the free stream. At the station 11 inches from the throat, a "bump" exists which can be picked up on the model tests in this region. This "bump" does not seem to persist far downstream and is not detected at the next station. The cause for the disturbance is not known, but it seems to have little influence at other stations than this one.

A centerline survey was made with a static pressure tube on centerline. This tube had a 10° conical point and the hole located 20 diameters back from the shoulder. Various static pressure tubes of different length diameter ratios have been run in the tunnel in an attempt to determine what design was necessary for accurate measurement of static pressures. Conclusive results are not available on these tests at this time.

Mach numbers in the tunnel, calculated by use of total head on centerline and stagnation pressure, static pressure on centerline and stagnation pressure, and static pressure on the wall and stagnation pressure are shown in Figure 24. The Mach number, as measured by the two centerline surveys, agree fairly well, but the wall static pressures give a somewhat lower Mach number.

The higher pressures and lower Mach numbers along the wall would be expected since the flow in the center passes through all the families of the expansion waves before the flow on the walls. In comparing the conditions at the center and the walls, we are comparing across Mach lines and we should expect changes. If true source flow existed in the nozzle, conditions along arcs about the source should be the same, but this would require that effects of the throat area had disappeared and the boundary layer on the walls did not change the radial streamline direction.

The difference between the measured Mach numbers and Mach numbers calculated by one-dimensional area ratios is also shown in this figure. The difference between these curves is caused by the boundary layer filling up part of the nozzle. For nozzles of small divergence one might expect the rate of growth of the boundary layer to become equal to the expansion of the nozzle and an area of constant Mach number to be reached.

D STAGNATION TEMPERATURES AND HEAT TRANSFER

Stagnation temperatures have been measured with a pencil type thermocouple with a response time of about 10 seconds. The temperatures decrease slowly throughout the run varying from 10° to 15°F over a 10 minute test. The response time of the thermocouple is sufficiently fast to follow this variation with negligible lag. A check was made on whether heat transfer was playing an important role in the axial symmetric nozzle. This was done by comparing the static wall pressures when heat is flowing into the gas and when the nozzle was cooled with dry ice so that the heat was flowing out of the gas. A thermocouple in the nozzle wall was used to determine whether the wall temperature was rising or falling and indicated the

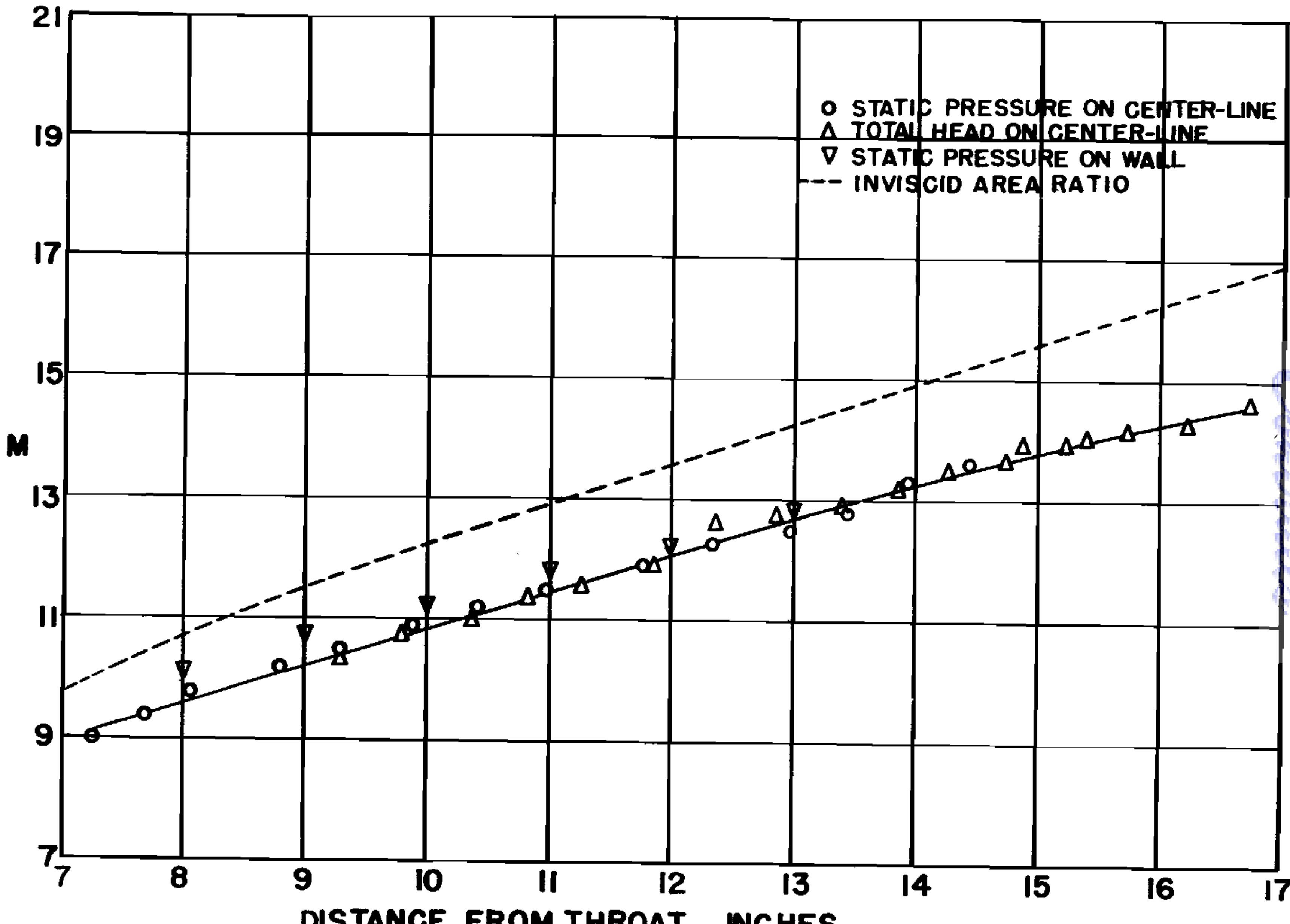


Figure 24 Mach Number Distribution Along the Centerline of the Axial Symmetric Nozzle

direction of heat flow. No change in wall pressures was detected, so it was concluded that heat transfer was not changing the flow in the tunnel to any measurable extent.

VII TESTING PROGRAM AND PRELIMINARY RESULTS

The several theories of hypersonic viscous flows which are currently available all agree on one general conclusion -- the pressure on the surface of a body at hypersonic speeds is made up of two parts: (1) The pressure rise due to the inviscid flow similar to that experienced in the supersonic region and (2) an additional pressure rise caused by effective thickening of the body (blunting) caused by the building up of the boundary layer. There is, however, a wide divergence among the theories as to the magnitude of this pressure rise and some question as to the parameters by which to describe the phenomena. In an experimental study, the two items which can be relatively easily studied are the pressure rise on a surface caused by the boundary layer growth and the growth of the boundary layer itself.

The first program to be undertaken in the hypersonic tunnel was the study of the pressure distribution and boundary layer growth on simple two and three dimensional bodies at Mach numbers significantly higher than previously investigated. The tests were made with two basic aims: (1) To try to determine which, if any, of the present theories was using the correct model of the flow and (2) to provide data over a wide enough range so that the check of a theory would be conclusive. If the theories were not correct, the tests would provide sufficient detail for the construction of a new model of the flow. The first phase of this study was the determination of pressure distributions, since this gives more results in a short time than a detailed boundary layer study. Several simple test models; a flat plate, 5° and 10° wedge, and a 10° cone, were constructed and are shown in Figure 25. All these bodies are provided with numerous static pressure orifices in the surface.

Subsequent reports will present the complete aerodynamic studies and conclusions. In this paper, only some typical results are given to show what sort of data can be obtained in such a system. Although Mach numbers from 11 to 15 have been obtained, the results presented herein are limited to some at 12.7 on one model, the flat plate. The general geometry of the model and the location of static pressure orifices are shown in Figure 26. It should also be noted that in the mechanical construction of this plate, the leading edge radius was set at .001 inches.

In Figure 27, the local pressure on the plate divided by the stagnation pressure is presented versus distance along the plate for three different stagnation pressures. The difference between the inviscid or free stream pressure and that actually obtained, divided by the inviscid pressures, is shown versus the Reynolds number (based on distance from the leading edge) in Figure 28 and the same pressure parameter is shown versus $1/\sqrt{Re}$ in Figure 29. A preliminary Schlieren photograph is shown in Figure 30.

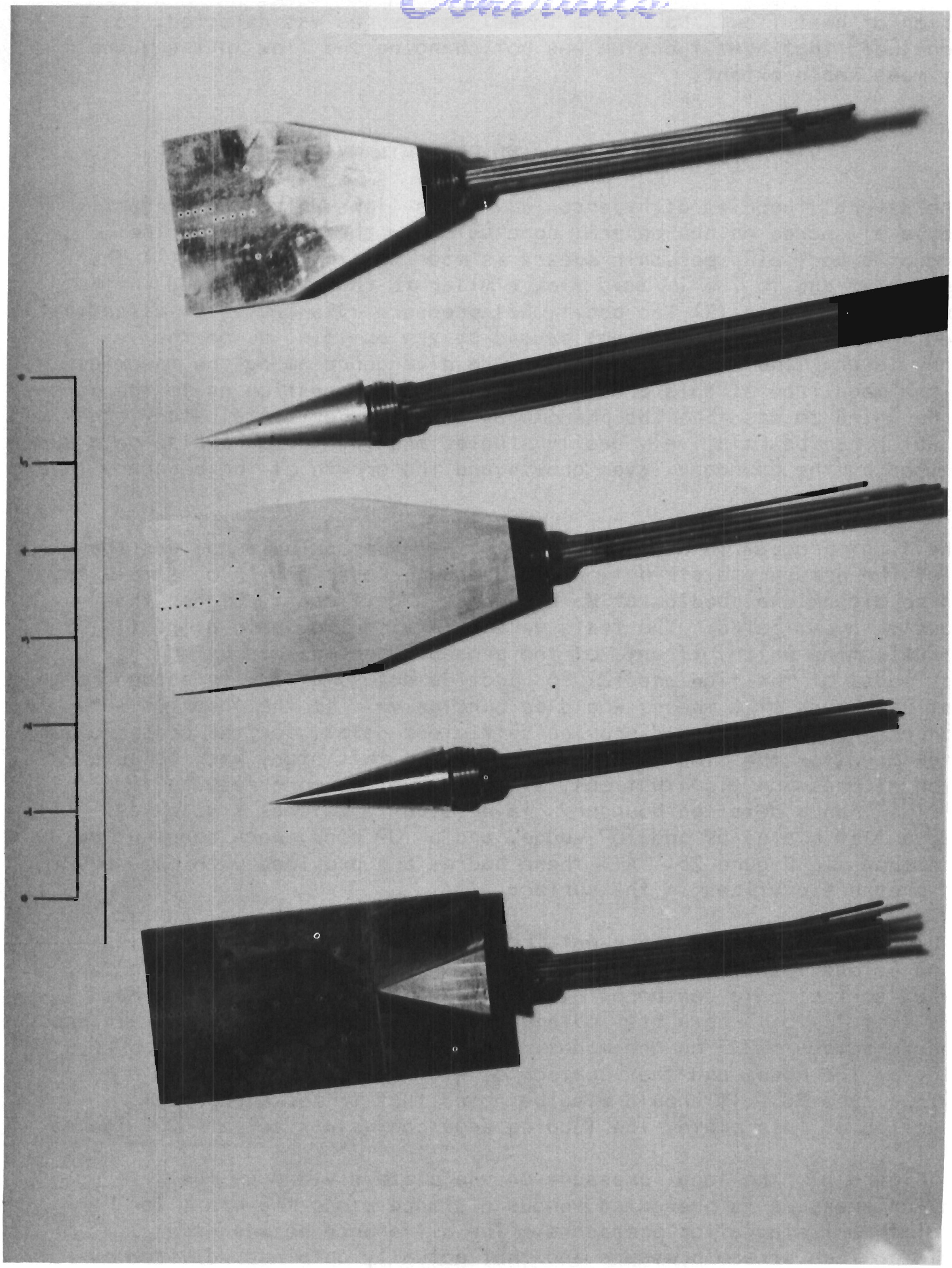


Figure 25. Test Models: Flat Plate, 5° Wedge, 10° Wedge, and Two 10° Cones

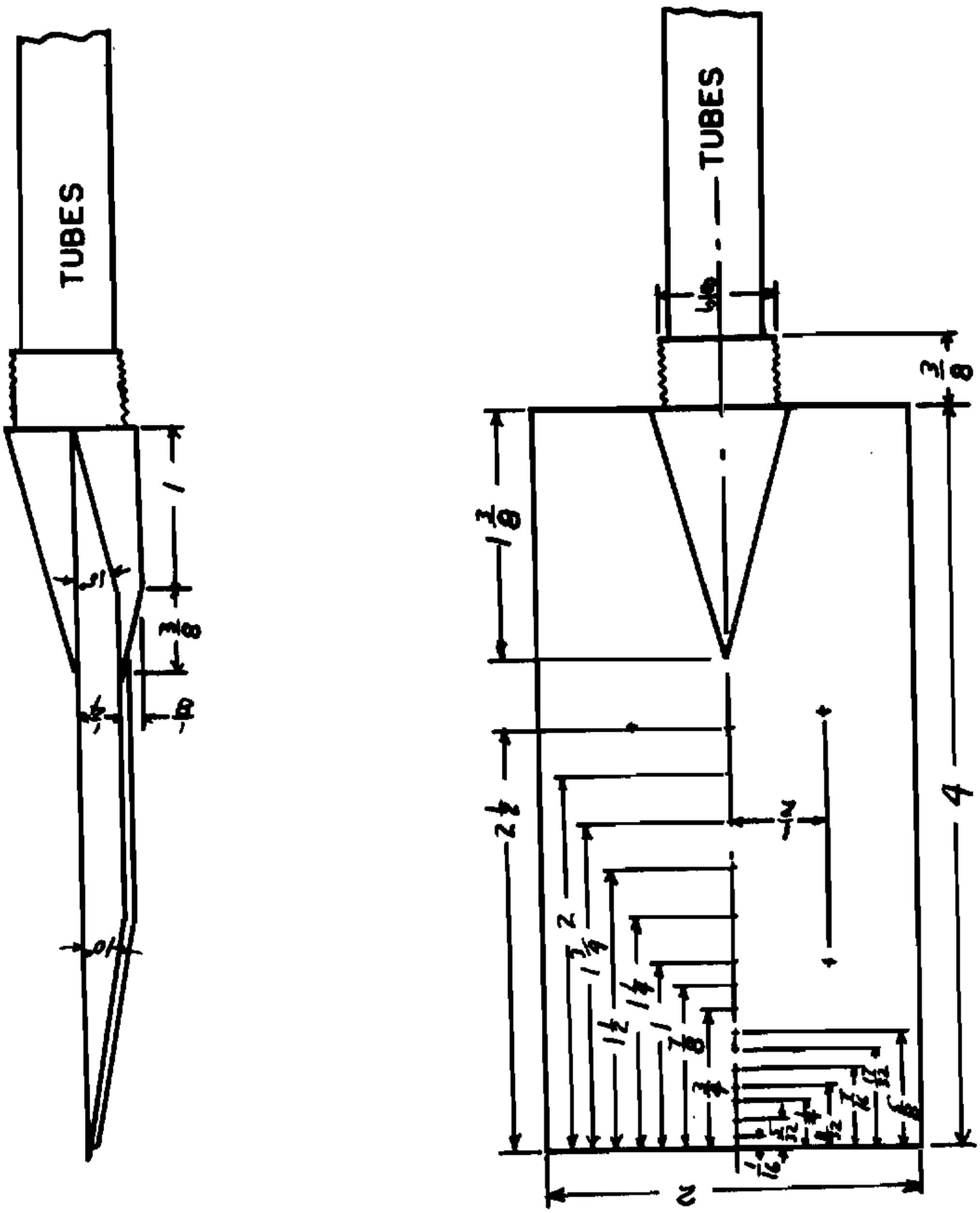


Figure 26 Details of the Flat Plate Model

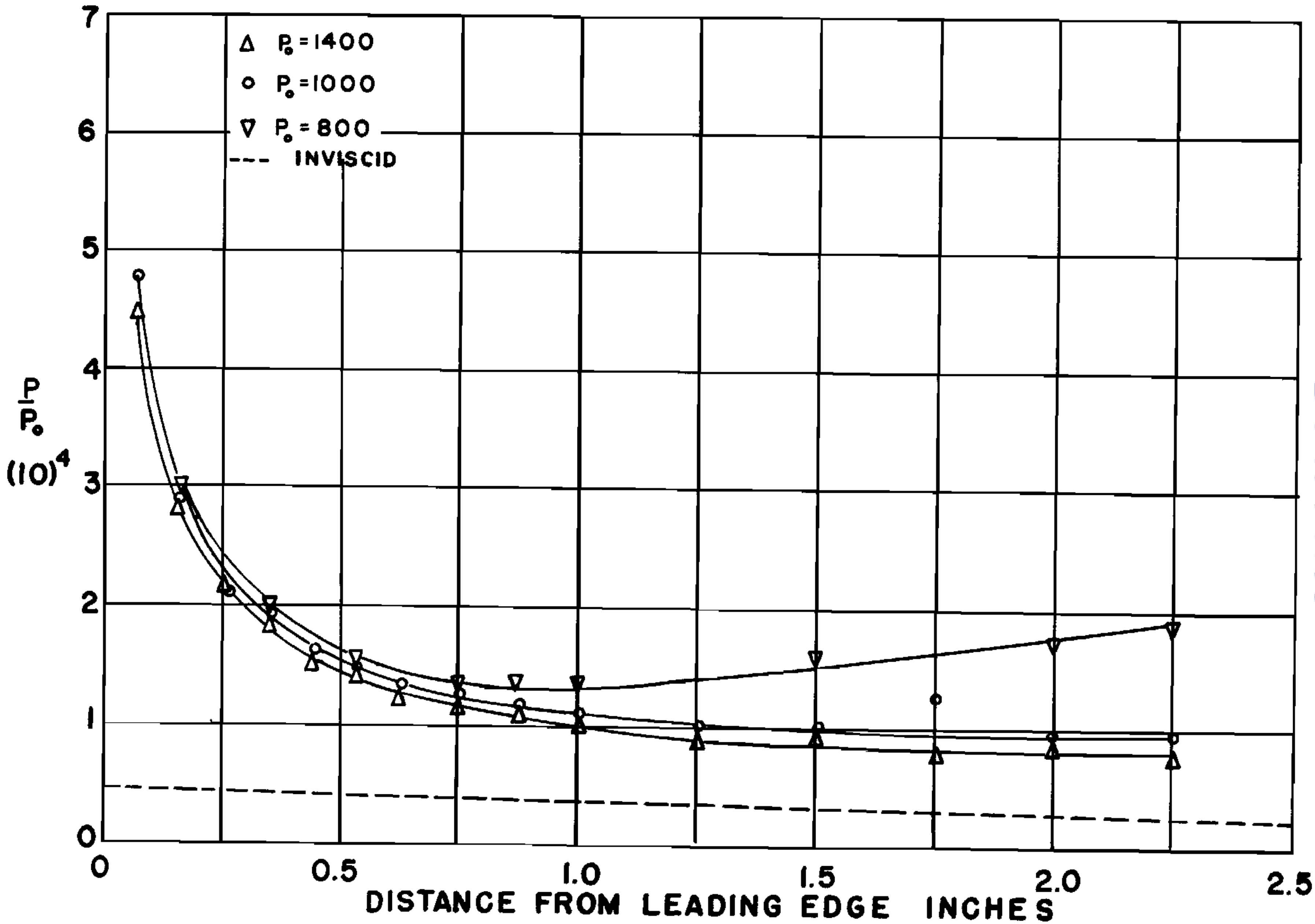


Figure 27 Static Pressure Ratio versus Distances Along the Flat Plate Model at M = 12.7

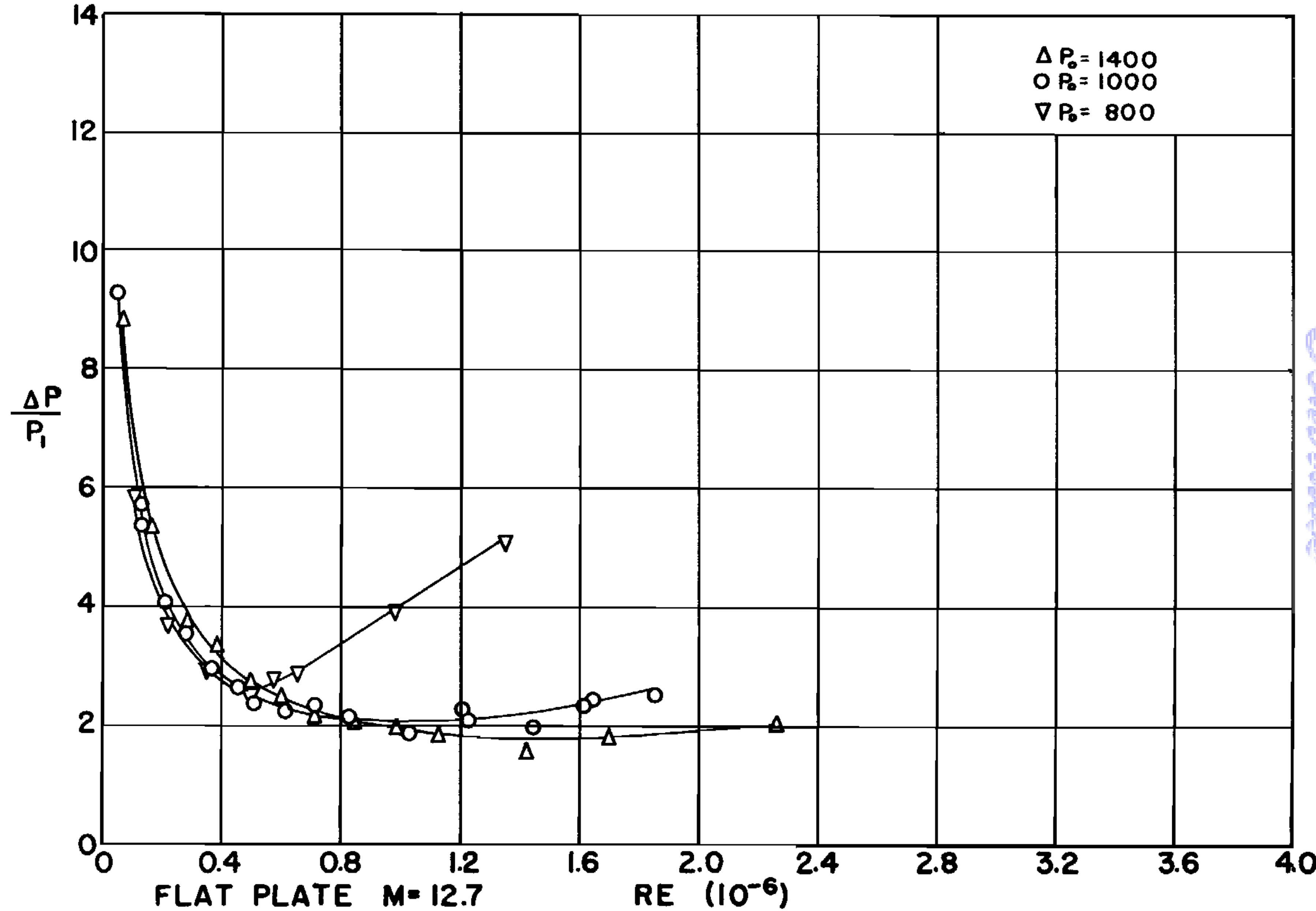


Figure 28 Static Pressure Ratio Increment versus Reynolds Number for the Flat Plate Model at M = 12.7

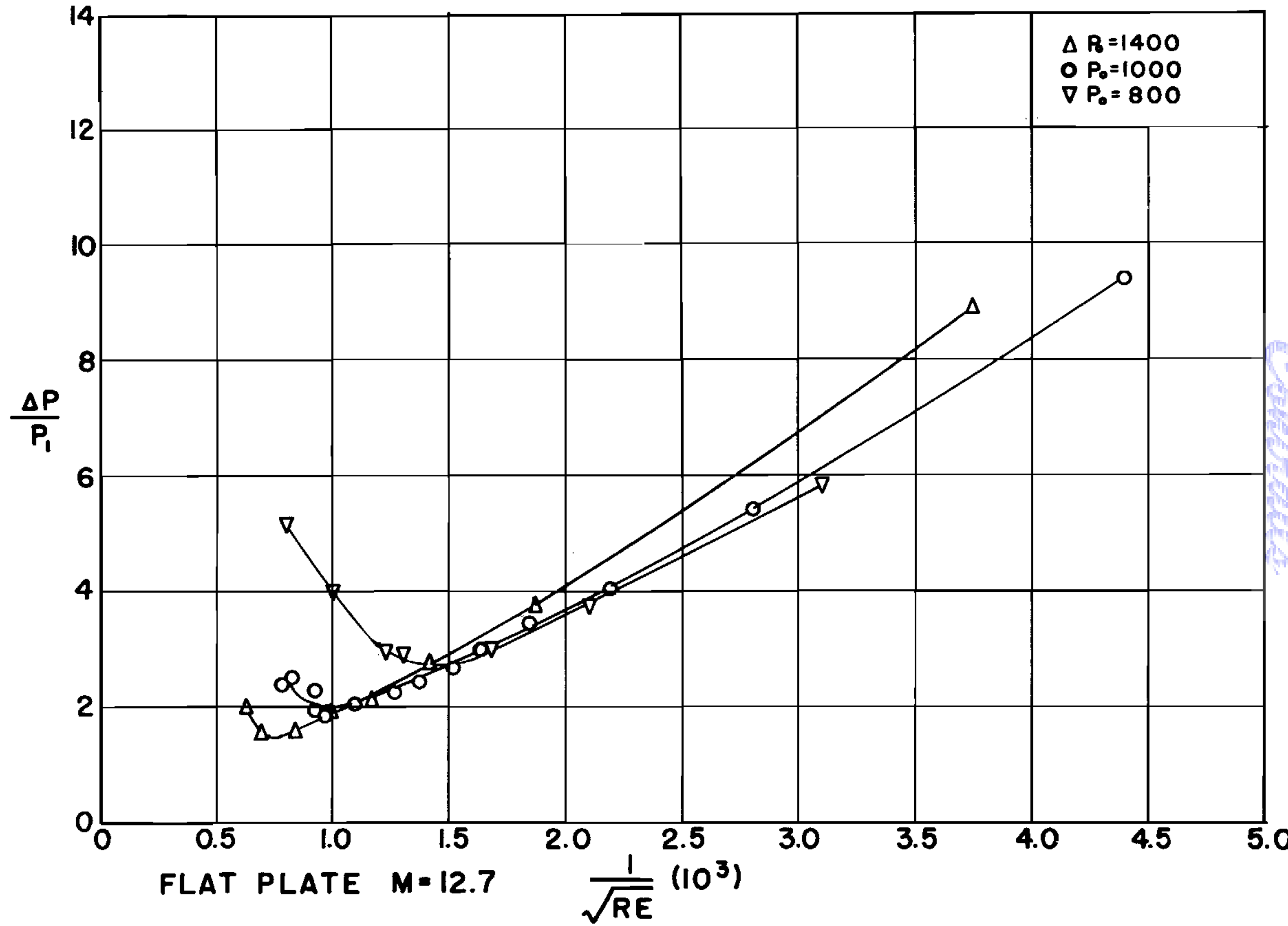


Figure 29 Static Pressure Ratio Increment versus $1/\sqrt{Re}$ for the Flat Plate Model at M = 12.7

There are two important points worthy of discussion but not concerned with comparison with theory.

(1) The effect of axial gradients in the tunnel. An examination of Figure 24 shows that there is no uniform Mach number region in the tunnel. There is a variation of approximately one-half M per inch so that all tests are conducted in this gradient. An examination of the results presented, Figure 27, show that the major pressure changes occur in about one inch. Tests at other Mach numbers show that this slight variation in Mach number over this range will have very small effects on the very high gradients near the leading edge. For a complete proof of this, tests will have to be conducted in a uniform flow nozzle.

(2) Pressure increase at the rear of the body. A pressure rise at the rear of the body as shown by the curling up of the curve (on the right end of Figure 27 and 28 and on the left end of Figure 29) is due to support interference. Further tests with longer plates have shown this disturbance to move rearward and has no effect on the forward part of the pressure curve.

It is quite obvious that very high pressures are experienced close to the leading edge of the flat plate. Almost ten times the inviscid pressure is experienced at the first orifice and this would lead to the conclusion that the shock at the leading edge may be a normal shock--detached-- or else a very strong oblique shock. In an attempt to partially explain this phenomena on the basis of a finite leading edge radius, tests were made with three different leading edge radii and the results are presented in Figure 31. A linear extrapolation of these three points to a zero thickness leading edge cannot explain any significant part of the very high pressure rise. It is reasonable to assume, therefore, that the major effect is the effective blunting of the body by the boundary layer growth. It should be pointed out that, in these tests, the smallest leading edge radius tested was an order of magnitude greater than the mean free path so that little or no slip effects are to be expected. This effect may change considerably when the mean free path becomes a larger fraction of the leading edge radius.

VIII CONCLUSIONS

On the basis of results presented herein, the following conclusions and results have been obtained.

1. Mach numbers of the order of 15 have been attained using helium as a working fluid in a small relatively simple and inexpensive wind tunnel.

2. This technique seems to provide an excellent method for studying fluid dynamic effects at very high speeds.

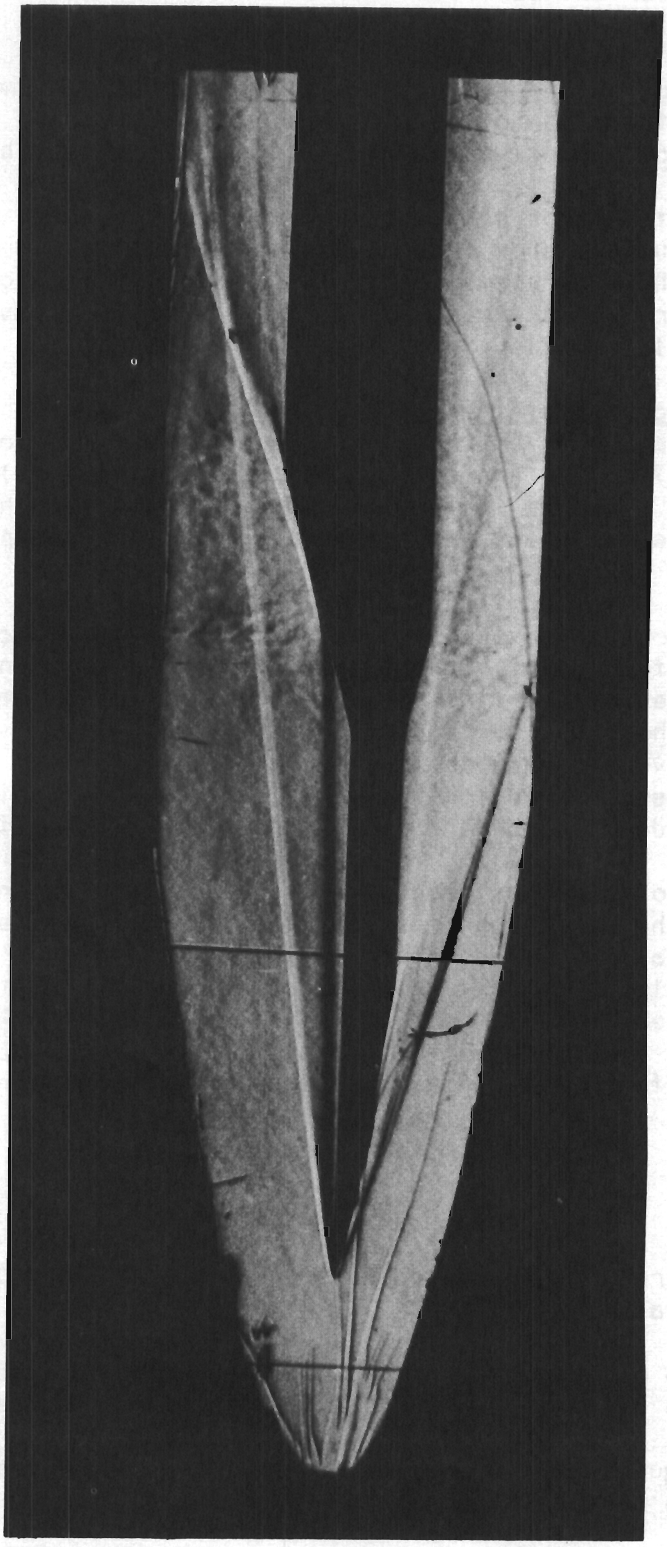


Figure 30. Schillieren Photograph, Horizontal Knife Edge of the Flat Plate Model at M 12.7

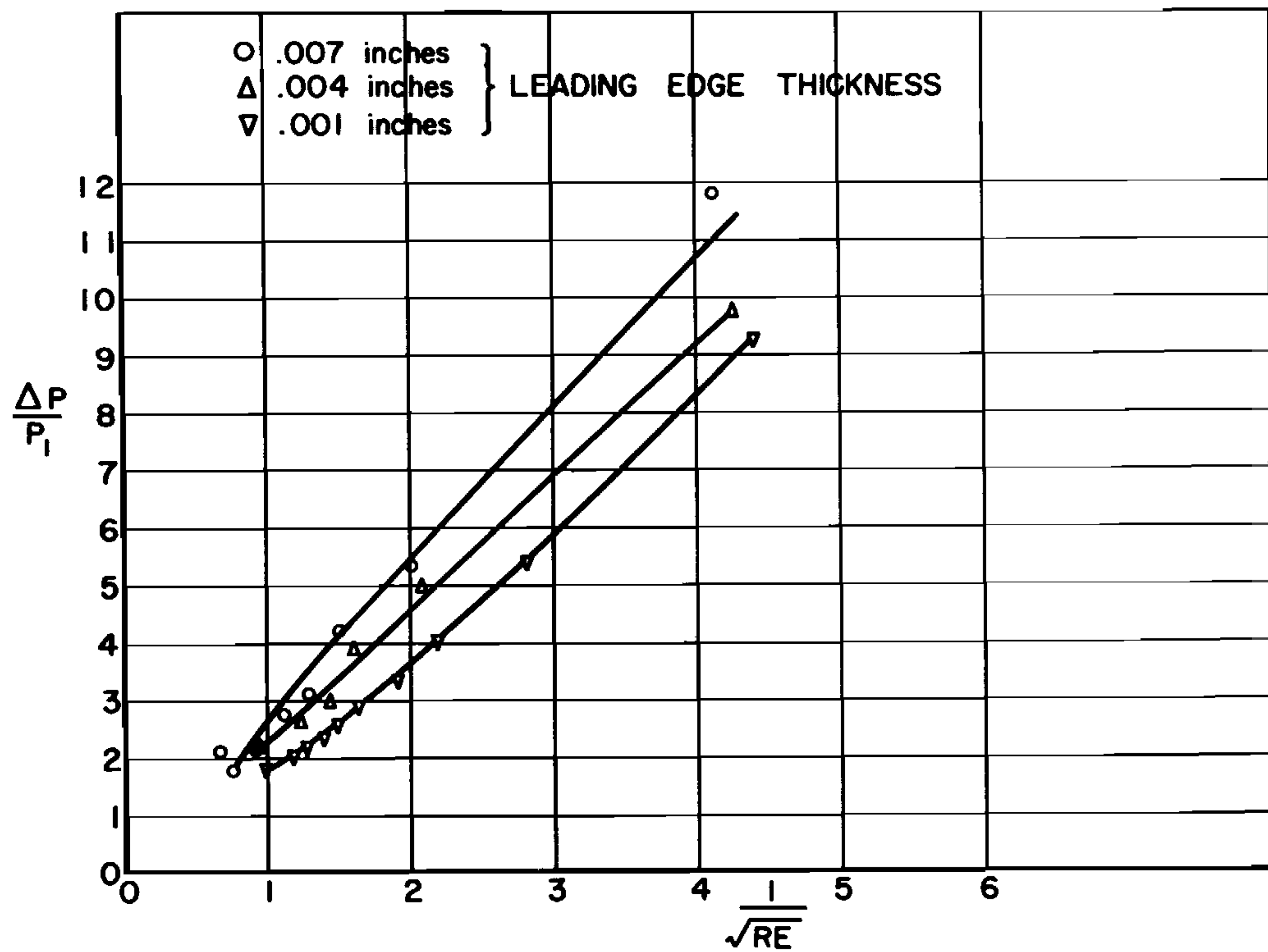


Figure 31 Static Pressure Ratio Increment versus $1/\sqrt{Re}$ for three Leading Edge Thicknesses: $M = 12.7$, $P_o = 1000$ psi

3. Preliminary results in the form of pressure distributions on a flat plate an Schlieren photographs at M = 12.7 have been obtained.

4. Preliminary studies of varying leading edge radii seem to indicate that the finite leading edge cannot be the cause of the very high pressures experienced over the front part of the plate.

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