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TECHNICAL NOTE

D-939

DESCRIPTION OF A 2-FOOT HYPERSONIC FACILITY AT THE LANGLEY RESEARCH CENTER

By George M. Stokes

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Langley Air Force Base, Va.

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TECHNICAL NOTE D-939

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By George M. Stokes

SUMMARY

This report describes the mechanical and aerodynamic features of a two-foot hypersonic facility at the Langley Research Center. The facility provides for the testing of aerodynamic models in the Mach number range between 3 and 7 at approximate Reynolds numbers between 0.5×10^6 and 1.0×10^6. The facility was designed to obtain the needed pressure ratio through the use of ejector nozzles. Compressors driving the ejectors operate continuously at a pressure ratio of 4 and thus give the facility a continuous running capability. Curves are presented to show the ranges of total temperature, total pressure, Reynolds number, dynamic pressure, and static pressure available in the tunnel.

The flow in the test section is suitable for model tests at all Mach numbers between 3 and 7, although the nozzle blocks were contoured for a Mach number of 6.

INTRODUCTION

Emphasis on higher flight speeds has brought about the need for hypersonic research facilities with various operating ranges. In order to help satisfy this need, a study was made to determine whether the large-capacity, low-pressure-ratio compressors used by the Langley 8-foot transonic pressure tunnel could be utilized for powering a high Mach number facility. Reference 1 had shown that supersonic Mach numbers up to about 5 could be obtained by using ejectors operating from low values of pressure ratio. Extrapolation of these data indicated the feasibility of obtaining Mach numbers up to about 7 through the use of the available compressor equipment.

A facility having a 2-foot-square test section has therefore been constructed which utilizes an ejector system to obtain a Mach number range from 3 to 7. The compressor system provides supply air for continuous running at very low test-section densities. Simulation of low density is important when testing configurations designed for high altitudes because of the rapid boundary-layer growth.

This report will describe both the mechanical and aerodynamic features of this tunnel and will show the ranges of test conditions.

SYMBOLS

p	pressure
q	dynamic pressure
T	temperature
M	Mach number
α	angle of attack
x	horizontal distance, measured downstream from nozzle minimum
y	vertical distance
Subscripts:	
t	total conditions
2	condition downstream of normal shock
ref	reference condition at sidewall station 80
ts	test section
ex	exit
en	entering

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TUNNEL DESCRIPTION

Figure 1 shows the general tunnel configuration with the relative positions of the sliding duct, contoured nozzle, test section, diffuser, ejector, and bypass. These components are all two dimensional and mounted between sidewalls 24 inches apart. The air normally enters through the sliding duct, is expanded into uniform flow by a nozzle, and passes through the relatively long test section into a hypersonic diffuser. The diffuser serves to lower the Mach number of the flow before it enters the pressure region provided by the ejectors. The ejectors of the facility serve as pumps to resupply a portion of the energy losses to the flow from the diffuser. After the supersonic streams from the ejector nozzles and diffuser mix, the flow decelerates through a shock system before moving into the subsonic diffuser. A description of the major components of this facility is given subsequently.

Nozzle and Test Section

The nozzle was designed to provide a uniform flow in the test section at a Mach number of 6. The theoretical calculations made to obtain the nozzle contours, without boundary layer, were obtained by expanding the flow by the method of characteristics into a radial flow (ref. 2) and then using the Foelsche equations for the transition from radial flow to the final uniform flow. Table I shows the nondimensional coordinates calculated for the nozzle. A nozzle length-to-height ratio of 10 was used. The minimum length possible aerodynamically is 7. The larger value was chosen because (1) local errors in nozzle construction would have less effect on the final flow; (2) the flow at off-design Mach numbers would have better flow uniformity; and (3) the possibility of flow separation in the nozzle would be reduced with the resulting lower turning angles.

An approximate boundary layer for the nozzle was calculated by using the method of reference 3. With a p_t of one atmosphere and T_t of 260° F at the design Mach number, the computed displacement thickness of the boundary layer increased throughout the nozzle with a final value of 1.78 inches at the nozzle exit. The calculated nozzle coordinates were adjusted to include the boundary-layer displacement thickness. The fixed-contour nozzle blocks shown in figure 1 (station -6.4 to 102.3) result from this aerodynamic design.

Changes of Mach number from the design condition are obtained by changing the positions of these nozzle blocks. The blocks are supported and moved vertically at the upstream end with screw jacks, while the downstream ends are attached to the fixed test section by flex plates. The nozzle minimum can be varied from 6.75 inches open to about 0.010 inch open. Two limit switches set on each side of the lower nozzle block, activated by a 0.010-inch-diameter wire protruding 0.010 inch above the minimum, prevent driving the blocks together. The mechanical design of the nozzle blocks provided for a temperature elongation with minimum change in shape for the temperature range between 60° F and 260° F. The nozzle surfaces were sprayed with zinc chromate for protection against corrosion.

The test section, noted on figure 1, has a cross section 2 feet square at station 102 and starts converging at station 126. The convergence angle for both top and bottom walls is 3°.

Two sets of windows are available for use in the diamond-shaped openings located in the two sidewalls of the test section. One set contains a diamond-shaped viewing area with the major and minor axes approximately 42 inches and 18 inches, respectively. The glass in these windows is ordinary plate glass, except chosen with the aid of a shadowgraph to be better than average optically. The other set of windows provides a 9-inch-diameter optically finished viewing port.

Diffuser and Ejector

No effort was made to design a specially contoured surface for the most efficient supersonic diffusion. Instead, two flat-surfaced blocks were used as shown between stations 155.1 and 264.4 of figure 1. The length of the blocks was a compromise between mechanical construction requirements and the need to keep the wall convergence angle relatively small at the design Mach number to minimize energy losses. Movement of the diffuser blocks is essential for starting the nozzle and for obtaining the most effective diffusion at various operating conditions. The blocks are supported and moved at the downstream end by screw jacks and are attached at the upstream end by flex plates similar to those on the nozzle blocks. The vertical opening at the diffuser exit may be controlled and set to any value between 3 inches and 20 inches.

In order to develop the low pressure required at the exit of the diffuser, the ejectors were designed to provide a Mach number of 3 in this region. The streams from the ejectors and diffuser were made to intersect at angles as low as practical to insure minimum energy losses from mixing conditions and shock waves. Theoretically, the flow remains above a Mach number of 2 in the ejector mixing region. The divergent section near the end of the ejector provides a place for stabilization of the normal shock system.

The ejector blocks were designed to be movable. The upstream ends are supported and moved by screw jacks and the downstream ends are hinged to blocks fastened to the subsonic diffuser. Although the servodrives for the diffuser and ejector blocks are independent of each other, the gear ratio of the ejector drive system is such that, when the diffuser blocks and the ejector blocks are opening or closing simultaneously, the minimum of the ejector nozzles will remain approximately constant.

A movable second minimum for the ejector mixing region was provided at station 310. Experiments showed that the ejector second minimum was needed only when running at the Mach numbers of 6 and 7 under low-density conditions. Therefore, to simplify the operational problem, the ejector second minimum was fixed in the full open position. The downstream end of the ejector mixing region exhausts into a subsonic diffuser, which connects with a 48-inch-diameter return line to the compressors. The nozzle, test section, diffuser, and ejector components are all fitted with sliding seals along the sidewalls. The seals are made of silicone rubber-type material (Viton A - type M-62) placed in a specially cut groove near the working surfaces. This material is compressed slightly so that it presses lightly against the sidewall. Lubrication with powdered graphite reduces friction and wear.

Compressors

A schematic diagram showing the general arrangement of the compressor piping and tunnel is presented in figure 2. Four compressors are normally used when operating the tunnel. Two 100,000-cubic-foot-per-minute compressors are connected in parallel and are used for the main drive. A third 10,000-cubic-foot-per-minute compressor furnishes air for the hypersonic nozzle and the fourth 3,000-cubic-foot-per-minute compressor is used to circulate air through the dryer. Figure 3 shows the variation of pressure ratio with volume characteristics for main drive compressors and the compressor feeding the hypersonic nozzle.

A portion of the discharge air from the main compressors is taken through the compressor for the nozzle. The output pressure of this compressor can be controlled and increased to four times the inlet pressure. The output from this compressor is fed through an 18-inch-diameter line to the electric heater before entering the tunnel. The fourth compressor is used to circulate air through the dryer. Figure 2 indicates the manner in which the drying system fits into the tunnel circuit. The dryers lower the dewpoint to about -20° F at one atmosphere pressure.

Model Support

A sting and model support strut is located in the downstream portion of the test section. (See fig. 1.) The strut rotates about center-line station 114.3 and provides an angle-of-attack range of ±15°. A servomechanism attached to the strut indicates the angle of the sting. A sting is attached to the center of the strut sector and extends 8 inches upstream of the strut leading edge. The sting is a thick-walled tube with an outside diameter of $1\frac{1}{4}$ inches. The 3/4-inch hole in the center of the sting connects with a specially cut raceway in the strut which is used to carry wiring and tubing from models to a pressure-sealed box on top of the tunnel.

Heater

The air is heated by a 1,000-horsepower nichrome-tube type heater. Figure 4 shows the tube bundle before installation into the case. Three-phase power (which is continuously controllable) is fed to the heater from a 1,000-horsepower motor-generator set. The heater design permits discharge air temperatures up to 1,500° F. Present tunnel operation is limited to much lower temperatures by construction materials.

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TUNNEL OPERATION

Because of the critical relationship of the aerodynamic operation of the facility to the geometry of the components, the positions of the nozzle blocks, diffuser blocks, and ejector blocks must be carefully controlled. A general starting and running procedure has been established whereby any Mach number can be obtained between 3 and 7. Briefly, the procedure calls for closing the minimum of the hypersonic nozzle blocks, closing the sliding duct, opening the diffuser block exit to about 64 percent of test-section area, and setting the ejector minimum area (about 3 inches open) to take the rated output of the main compressors. Valves isolating the compressors from the tunnel section are opened and supersonic flow (approximately $M = 3$) is established in the ejector nozzles. The hypersonic nozzle is then opened until starting occurs, that is, when the normal shock moves downstream of the diffusers into the ejector mixing region and supersonic flow is established in the test section and diffuser. Immediately after starting, the Mach number in the test section is approximately 5. The diffuser exit area is reduced and this reduction, in turn, lowers the pressure ratio needed for maintaining the flow. The ejector blocks are moved in unison with the diffuser blocks to maintain a nearly constant area at the ejector minimum. The Mach number in the test section is raised by closing the nozzle blocks slowly as the diffuser tips are closing. Normally, the diffuser exit is reduced to about 20 percent of the test-section area when running at $M = 6$. Efforts made to reduce this area to 10 percent caused an adverse pressure gradient in the boundary layer which initiated flow separation. This flow separation, in turn, resulted in losing the hypersonic flow. When the hypersonic flow is lost, the nozzle is closed and the diffuser is opened to the original starting position where restarting can be accomplished by again opening the nozzle.

The total pressure to the nozzle is controlled by changing the pressure level in the entire tunnel circuit or by changing the pressure ratio of the 10,000-cubic-foot-per-minute compressor.

The temperature of the air entering the nozzle is controllable between certain limits. The minimum temperature available (about 300° F) with sliding duct closed is that produced from the heat of compression from the 10,000-cubic-foot-per-minute compressor. When the sliding duct is open, the temperature is about 80° F. The electric heater is used to set any desired temperature between the temperature entering the heater and 500° F. The seal materials used and nozzle mechanical design restrict the maximum running temperature to 500° F.

INSTRUMENTS

The instruments for the facility are divided into two groups. One group (the tunnel operating group) gives information pertaining to the flow in the tunnel and operating equipment. The other group gives information relating to the model being tested.

The tunnel-operation instruments are built in and used for establishing and measuring the flow conditions during each run. Visual monitoring of these instruments is accomplished at the operator's control console. Measurements from the instruments giving p_t, T_t, p_{ref}, and α are taken during each run on automatic digital recording equipment.

Instruments for measuring model pressures, forces, and temperatures are available. The accurate measurement range of these instruments, however, is more limited than that required for covering the wide range of flow conditions available with the facility. Therefore, additional instruments covering special ranges will be needed for some tests. Automatic recording of forces and base pressures of the model has been provided. Because of the low-density range of the tunnel, lightweight models and sensitive balances will be needed. Also, since the static pressure is low, it will be impractical, because of the lag time in small tubing, to attempt model pressure surveys that require running tubing from the model to pickups outside the tunnel. An exception is made with respect to the base pressure measurements in that special space is made available in the raceway leading from the model to a point outside the tunnel for use of tubing large enough to reduce the lag time to within reasonable limits.

RANGE OF AERODYNAMIC CHARACTERISTICS

Data have been obtained to determine ranges of the various flow parameters and the uniformity of the flow over the test Mach number range.

Operational Curves

Figure 5 defines the ranges of operation of the quantities p_t, p_{ts}, q_{ts}, T_t, and Reynolds number. These data can be used to determine whether the operating envelope of the wind tunnel is appropriate to a particular investigation, and to aid in determining the ranges of instruments needed for a particular test.

Mach Number Distribution

The Mach numbers presented in figure 6 were determined by using the ratio of the static pressure (measured 2 inches laterally off the tunnel center line on a cylindrical probe extending through the nozzle minimum) and the total pressure entering the nozzle. An isentropic expansion has been assumed for Mach number calculations. The sidewalls were not diverged to account for the sidewall boundary layer. Therefore, the sidewall boundary layer may be responsible for part of the Mach number gradient ($\Delta M = 0.15$) in the test section at $M = 6$. Figure 6 shows that the recommended test section falls between tunnel stations 80 and 132. Variations in the flow of the test section are believed to be small enough to provide adequate uniformity for model tests for the entire Mach number range.

A static reference pressure measured on the side at station 80 is used to obtain the reference values of test-section Mach number. The reference Mach numbers for the distributions plotted on figure 6 are shown on the left side of the figure. In figure 7, these values of reference Mach number are plotted against a value of Mach number selected to be representative of that in the test section. It may be observed that the Mach number at station 114 is also representative of the Mach number in the test section.

Figure 8 is presented to show the effect of stagnation temperature on the Mach number distribution. It is seen that data obtained with $T_t = 490^\circ$ F causes the representative Mach number in the test section to be less than M_{ref}, and the Mach number gradient near $M = 7$ is much more severe than at the lower temperature of 355° F. Data are not available to show exactly what may have caused this change, but it is believed that nozzle warpage resulting from overheating and some air liquification (refs. 3 and 4) are primarily responsible for these changes. Measurements of temperature on a nozzle block, with $T_t = 490^\circ$ F, showed that the values ranged from 350° F at the nozzle minimum to 200° F near the downstream end.

Boundary-Layer Data

Surveys of impact pressure $p_{t,2}$ were made out from the center of the top wall at station 110. These surveys were accomplished by mounting a single total-pressure tube on a support strut and taking measurements at locations from the wall as shown in figure 8. The data of figure 9 show that the boundary layer extends to about $3\frac{1}{2}$ inches from the wall. Theoretical calculations (by using ref. 5) indicated that the boundary layer would be 3.02 inches thick at this point for the design conditions. Increasing the Mach number to 7 caused the boundary-layer thickness to increase to about 4 inches. An increase of boundary layer with Mach number is in agreement with theoretical predictions.

The Mach number distribution shown at the bottom of figure 9 was obtained using the ratio $p_{t,2}/p_t$ for points outside the boundary layer beginning at 4 inches from the wall. These vertical Mach number distributions show, as expected, about the same order of variations as the longitudinal distributions.

Model Size

Some experiments were made to determine the effects of blockage in the test section. With a 5-inch-diameter flat disk mounted normal to the flow on the end of the model support sting it was difficult to maintain the hypersonic flow for long run periods, although tunnel starting could be accomplished by using a slightly larger than normal diffuser opening. The tunnel could not be started when the 5-inch disk was replaced with a 6-inch disk. The tunnel ran normally with a 4-inch disk mounted in the test section.

When the blockage effects and the extent of uniform flow region of the test section are considered, it appears that no difficulty would be experienced in testing slender-bodied configurations 2 to 3 feet long with wing spans of about 1 foot.

CONCLUDING REMARKS

The two-foot hypersonic facility at the Langley Research Center is an ejector-type tunnel which provides continuous flow at high Mach number and low density. The flow uniformity in the test section is adequate to permit investigation of slender configurations of at least 2 feet to

3 feet in length and having 1-foot spans. Data have been presented to show the aerodynamic characteristics in the Mach number range of 3 to 7.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., June 9, 1961.

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2. Beckwith, Ivan E., and Moore, John A.: An Accurate and Rapid Method for the Design of Supersonic Nozzles. NACA TN 3322, 1955.

3. Stever, H. Guyford, and Rathbun, Kenneth C.: Theoretical and Experimental Investigation of Condensation of Air in Hypersonic Wind Tunnels. NACA TN 2559, 1951.

4. Pope, Alan: Aerodynamics of Supersonic Flight. Pitman Pub. Corp. (New York), 1950.

5. Tucker, Maurice: Approximate Turbulent Boundary-Layer Development in Plane Compressible Flow Along Thermally Insulated Surfaces With Application to Supersonic-Tunnel Contour Correction. NACA TN 2045, 1950.

TABLE I.- NONDIMENSIONAL COORDINATES FOR HYPERSONIC NOZZLE DOWNSTREAM OF MINIMUM FOR $M = 6$

x	y	x	y	x	y
0	0.28872	1.375	0.3973	12.00	2.999
.10130	.28874	1.561	.4310	13.00	3.249
.12174	.28877	1.771	.4722	14.00	3.499
.14972	.28882	2.058	.5336	15.00	3.750
.18755	.28896	2.267	.5801	16.00	4.000
.21571	.28911	2.494	.6319	17.70	4.426
.23906	.28926	2.742	.6906	20.75	5.158
.27775	.28963	3.014	.7559	23.97	5.873
.3100	.2901	3.315	.8288	28.03	6.700
.3387	.2905	3.649	.9105	32.60	7.545
.3647	.2911	4.023	1.002	37.92	8.435
.4115	.2923	4.441	1.106	44.15	9.364
.4539	.2937	4.913	1.223	51.43	10.32
.4935	.2953	5.447	1.357	59.96	11.29
.5313	.2970	6.057	1.511	69.97	12.25
.6372	.3037	7.000	1.747	81.73	13.18
.7731	.3146	8.000	1.997	95.60	14.02
.9100	.3290	9.000	2.247	111.9	14.73
1.053	.3472	10.00	2.498	131.3	15.23
1.207	.3697	11.00	2.748	154.3	15.43

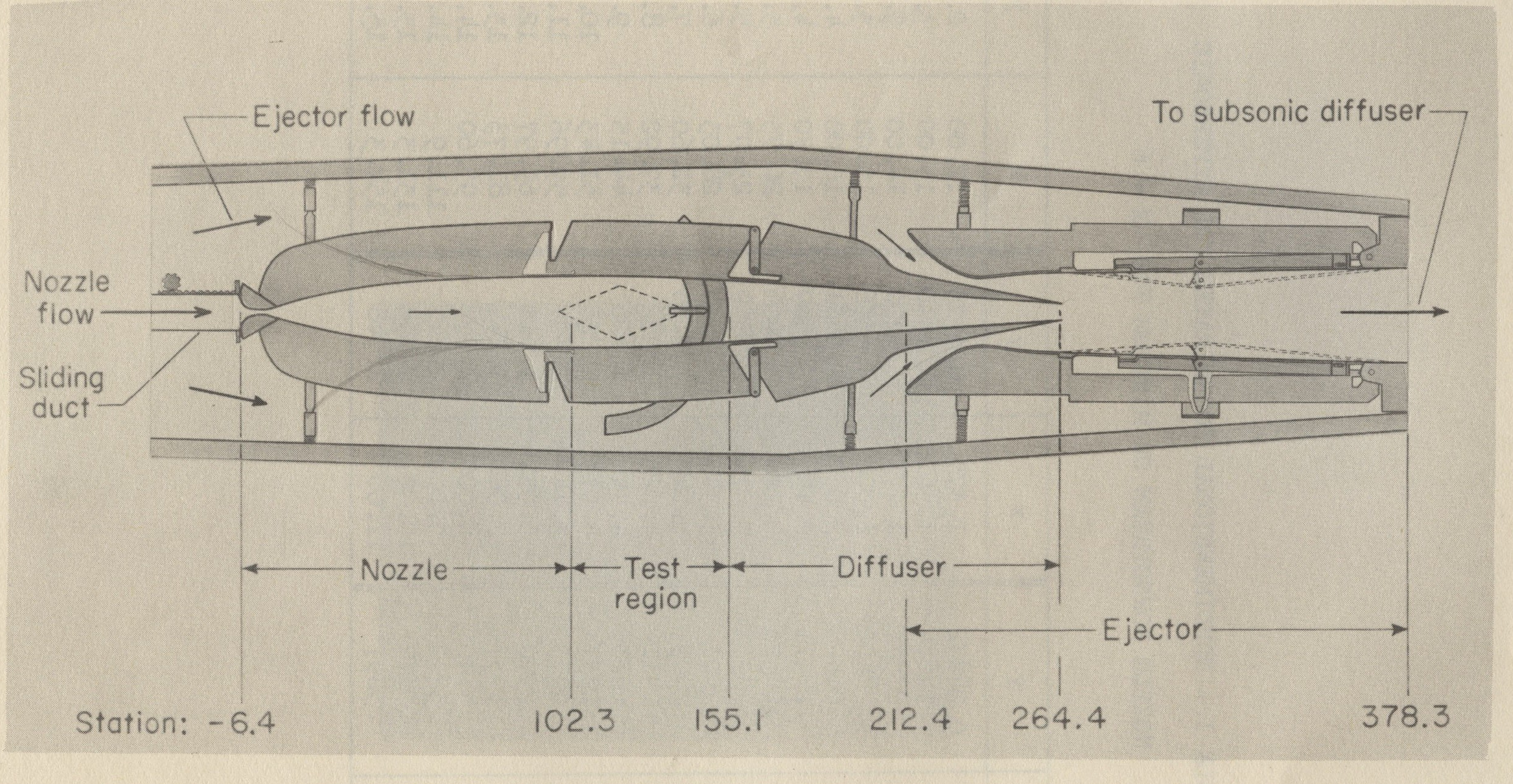


Figure 1.- Side view of nozzle, diffuser, and ejectors.

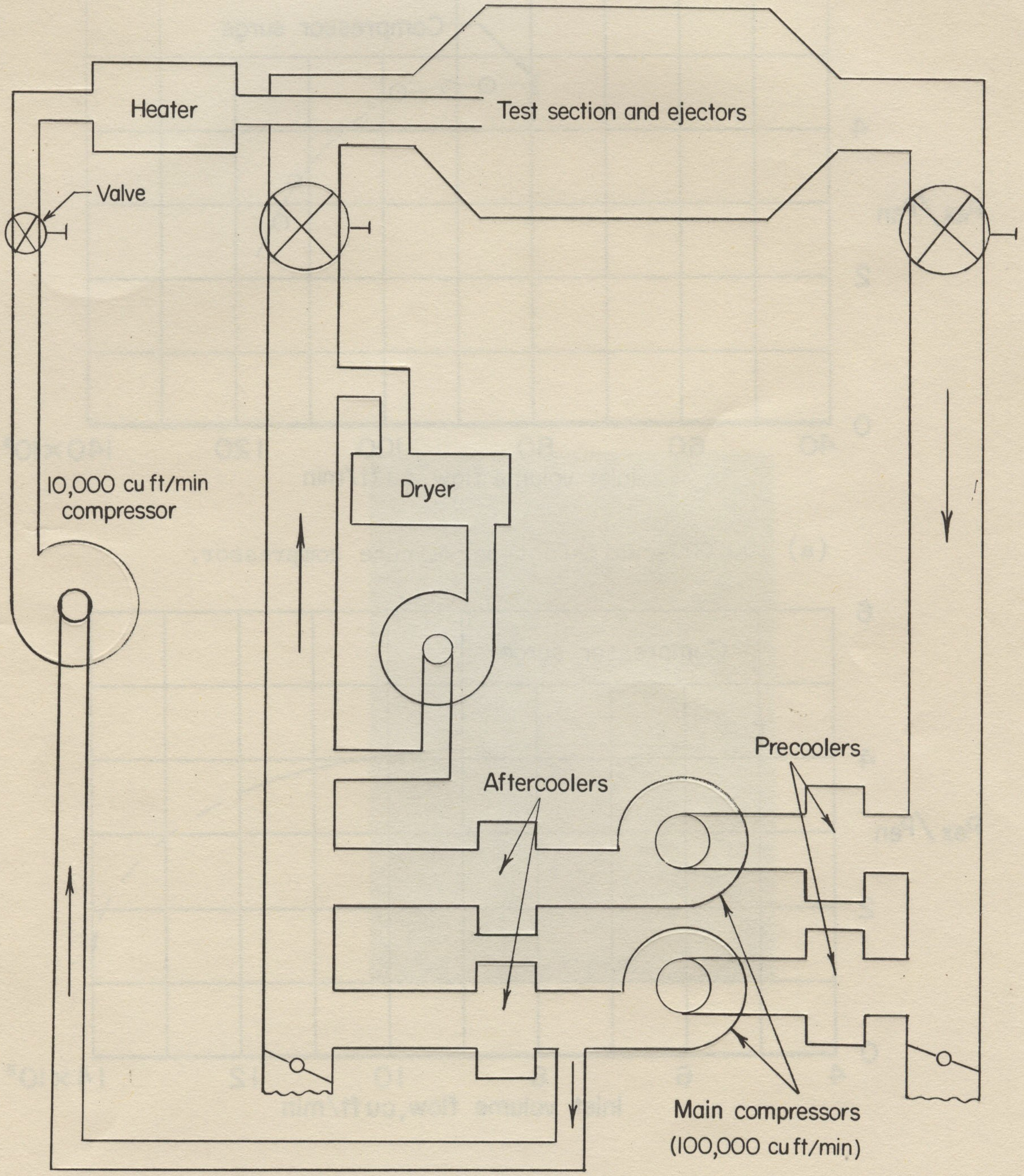
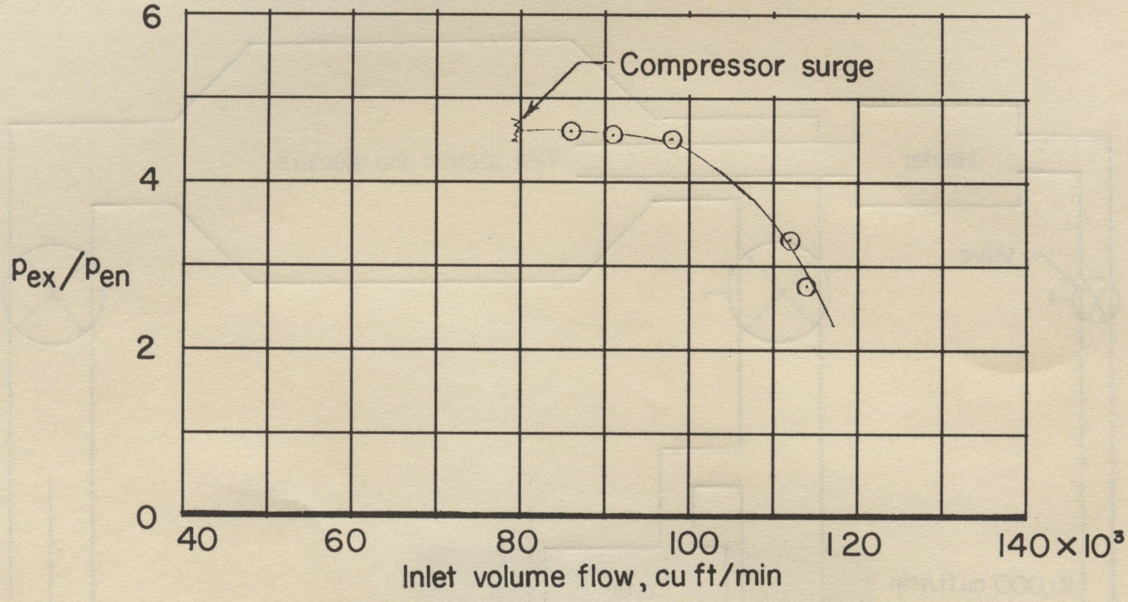
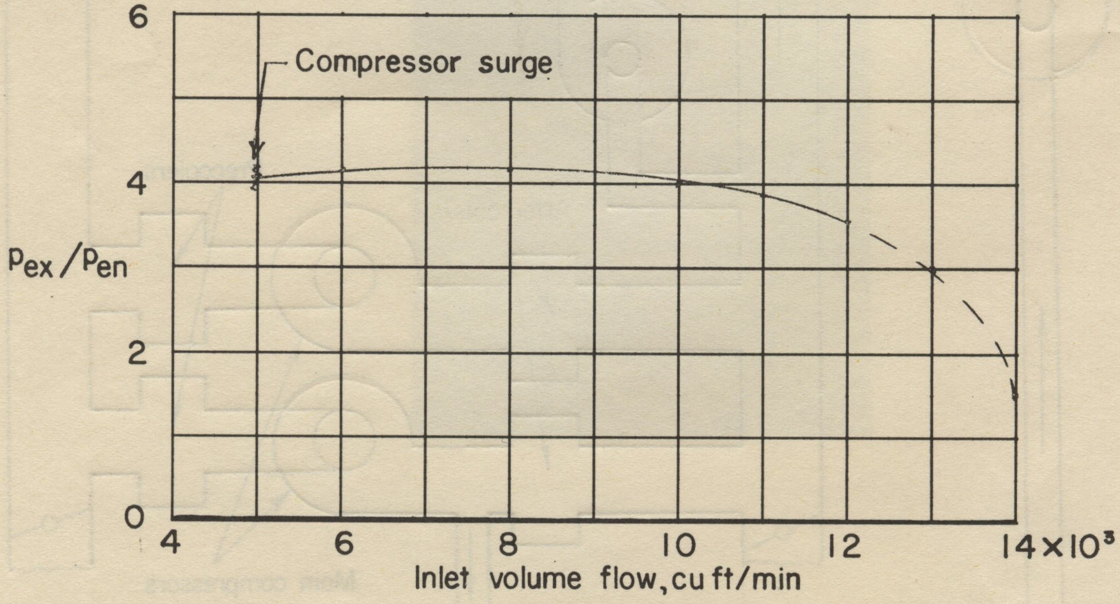


Figure 2.- Block diagram showing how major components are connected in tunnel circuit.

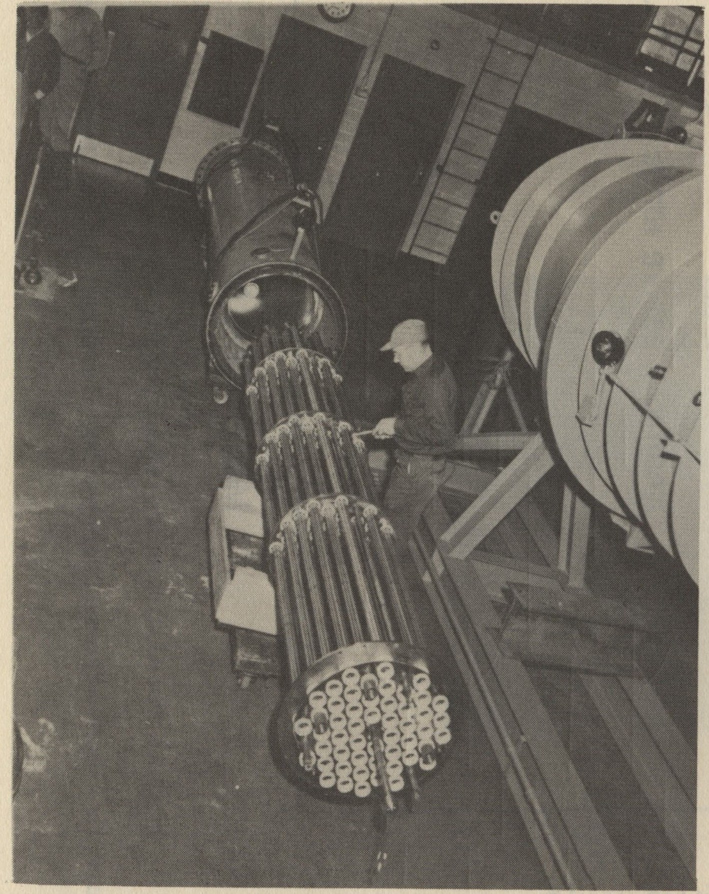


(a) 100,000-cubic-foot-per-minute compressor.



(b) 10,000-cubic-foot-per-minute compressor.

Figure 3.- Operating pressure-ratio characteristics for the main compressors and the nozzle feed compressor. Guide vanes set for maximum pressure ratio.



Heating unit before installation in case

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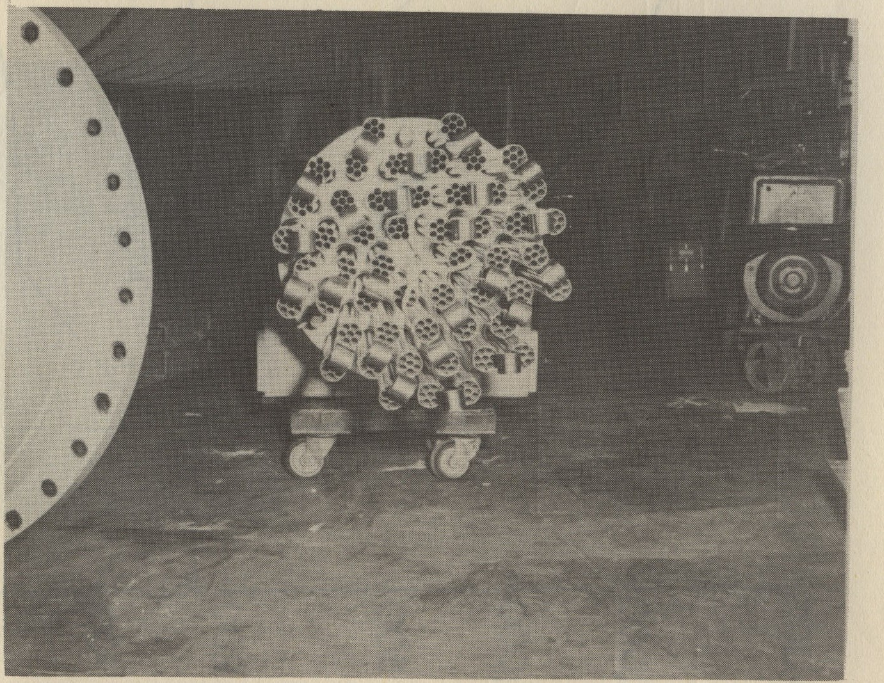

End view of nichrome heating tubes

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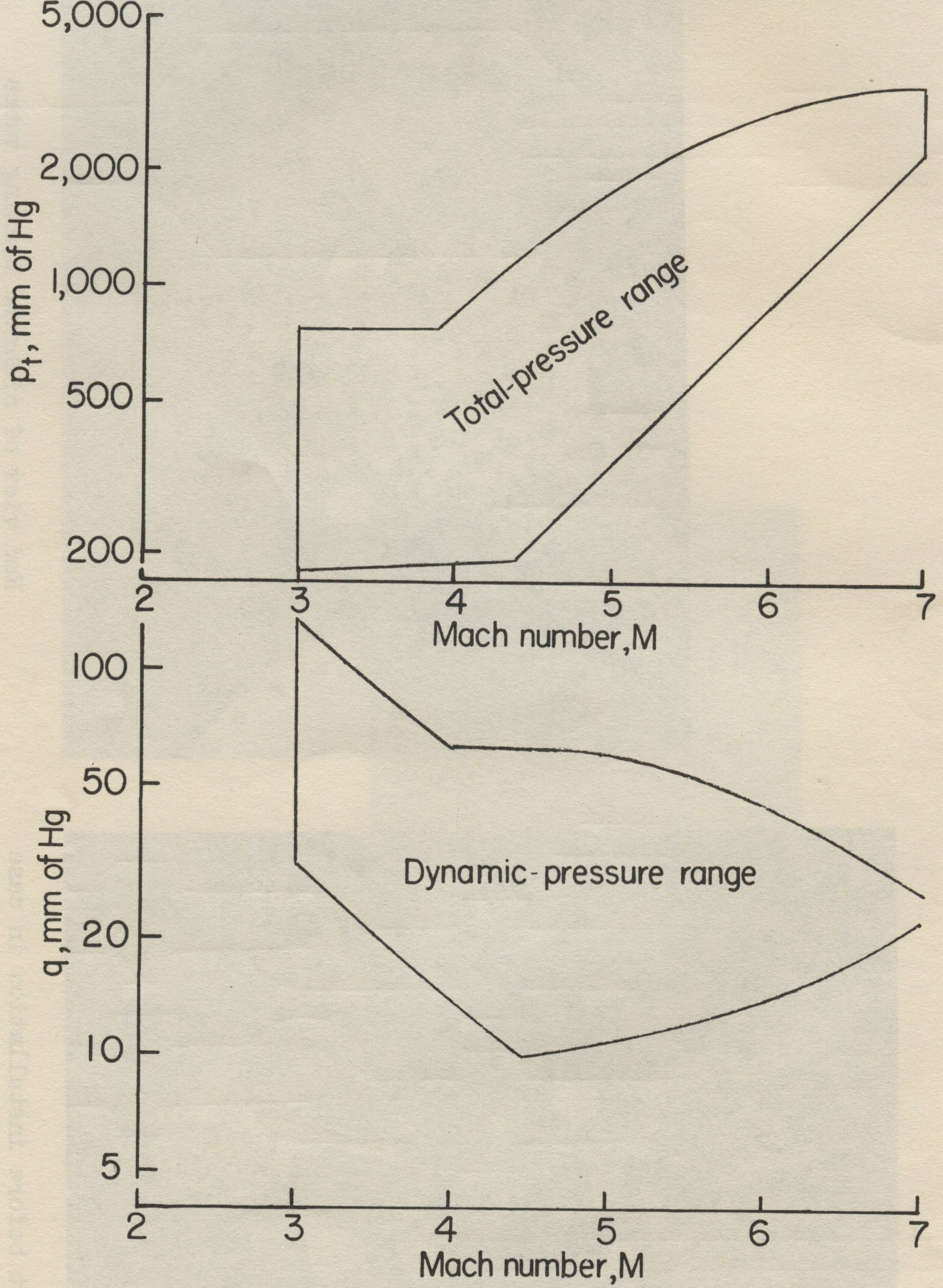
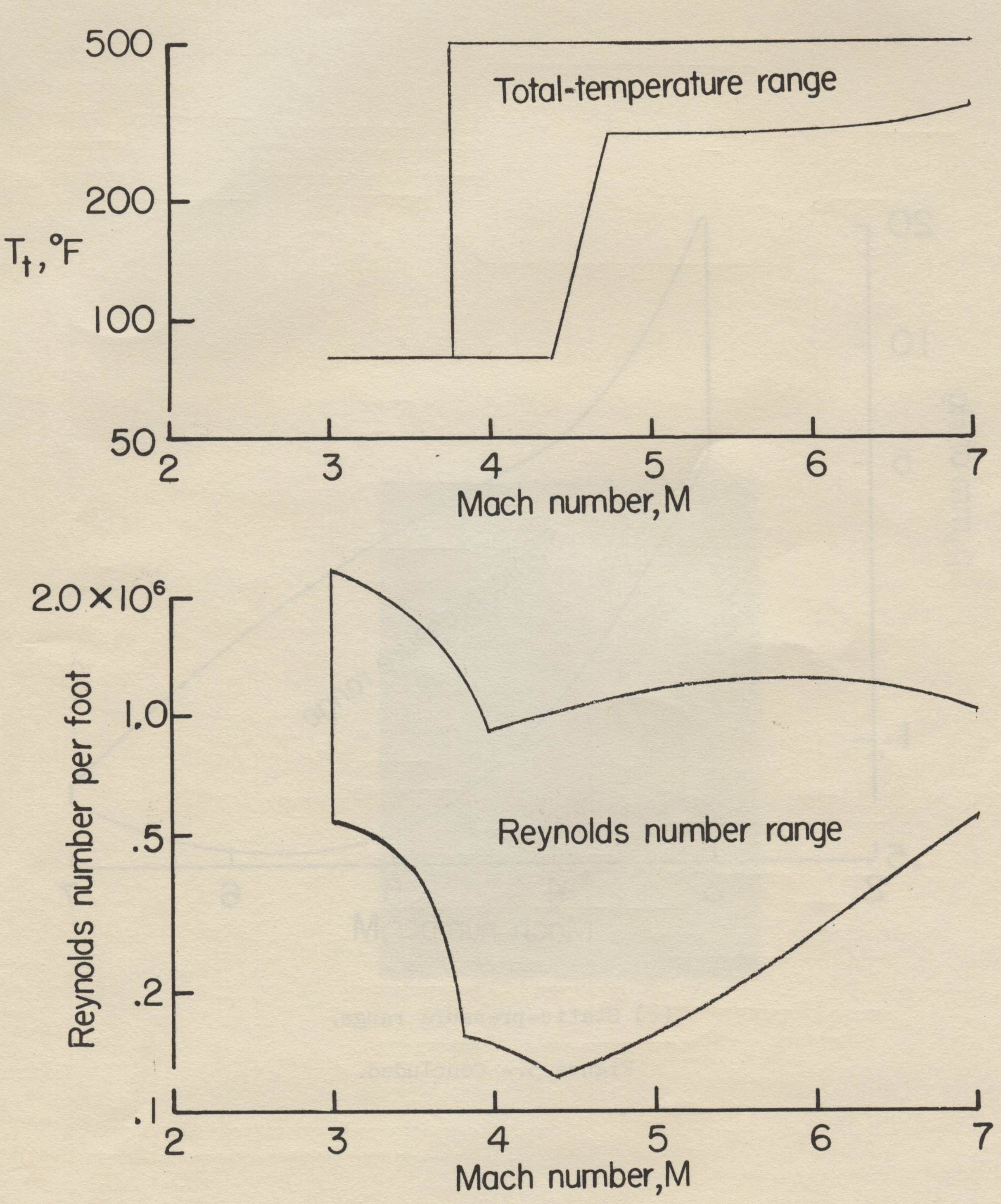
Figure 4.- Photographs of heater.



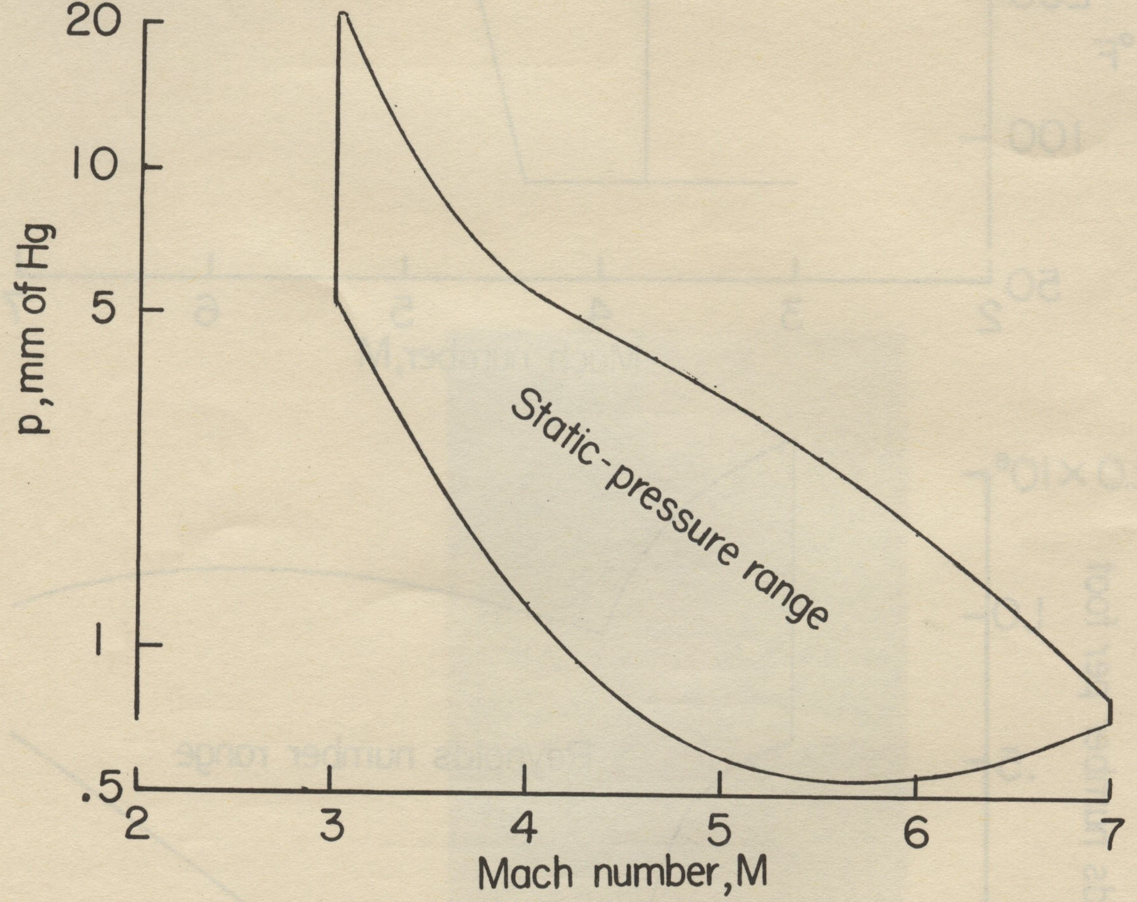
(a) Total-pressure and dynamic-pressure range.

Figure 5.- Operational ranges for various quantities in the tunnel.



(b) Total-temperature and Reynolds number range.

Figure 5.- Continued.



(c) Static-pressure range.

Figure 5.- Concluded.

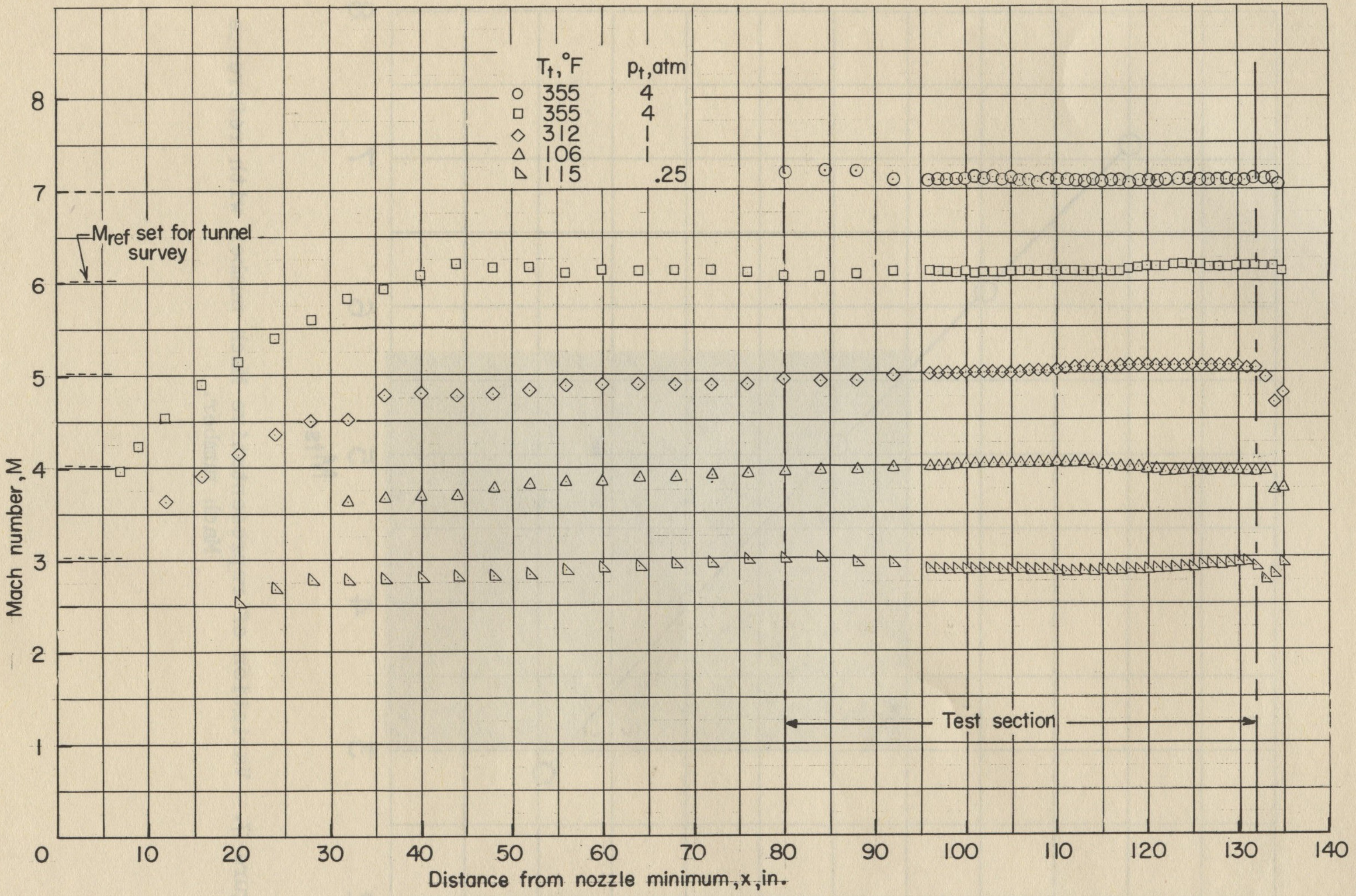


Figure 6.- Center-line Mach number distributions in two-foot hypersonic tunnel.

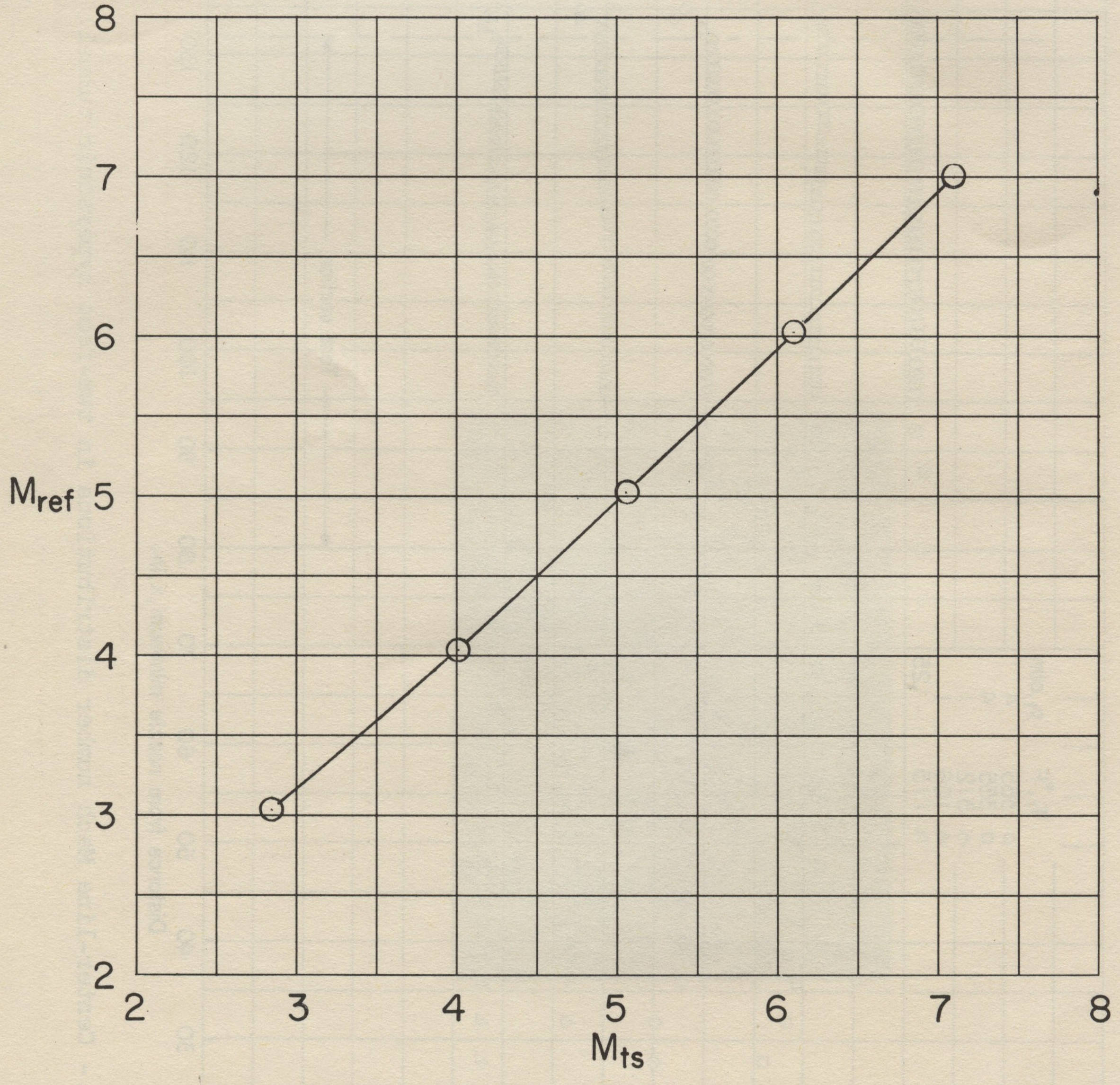


Figure 7.- Variation of representative Mach number with reference Mach number.

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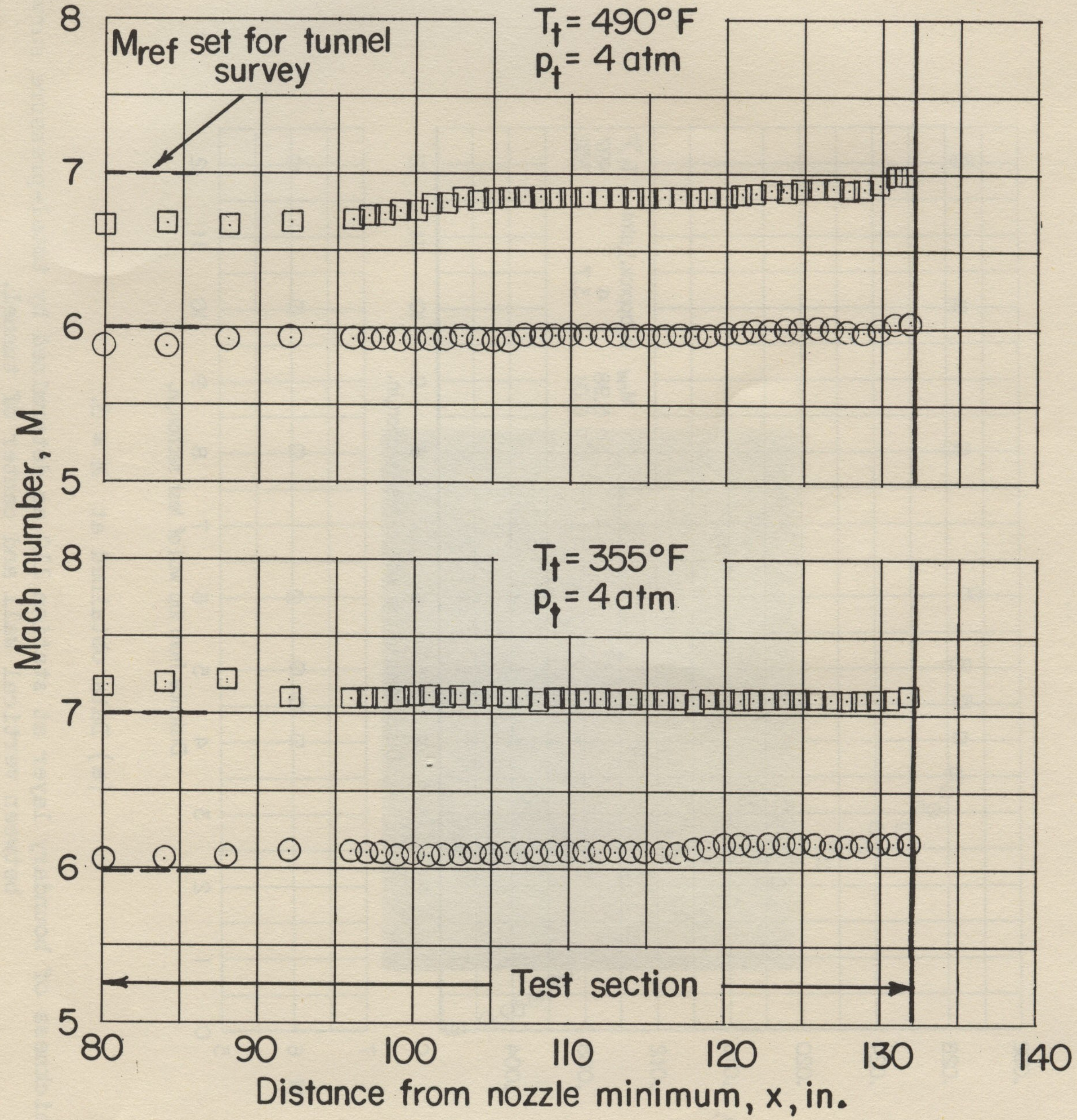
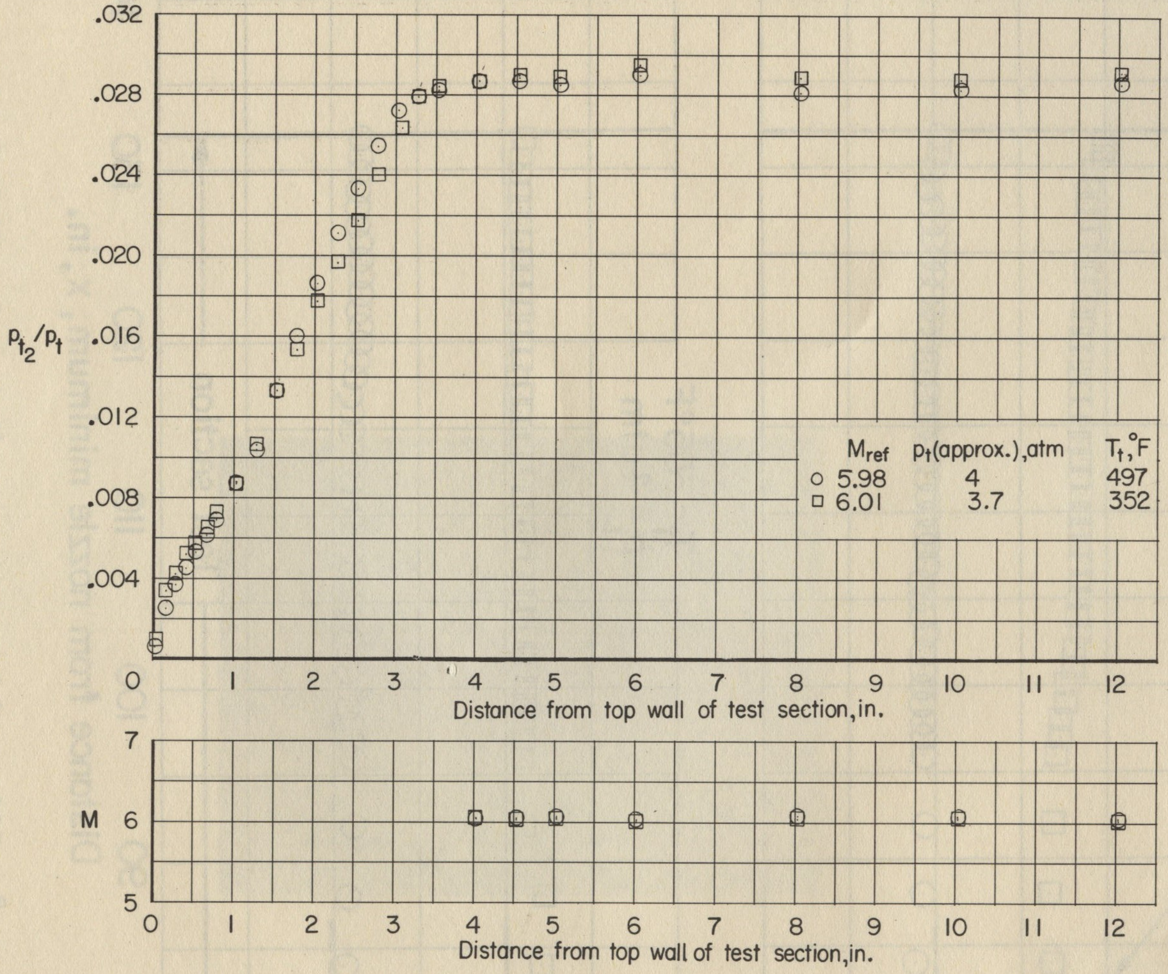


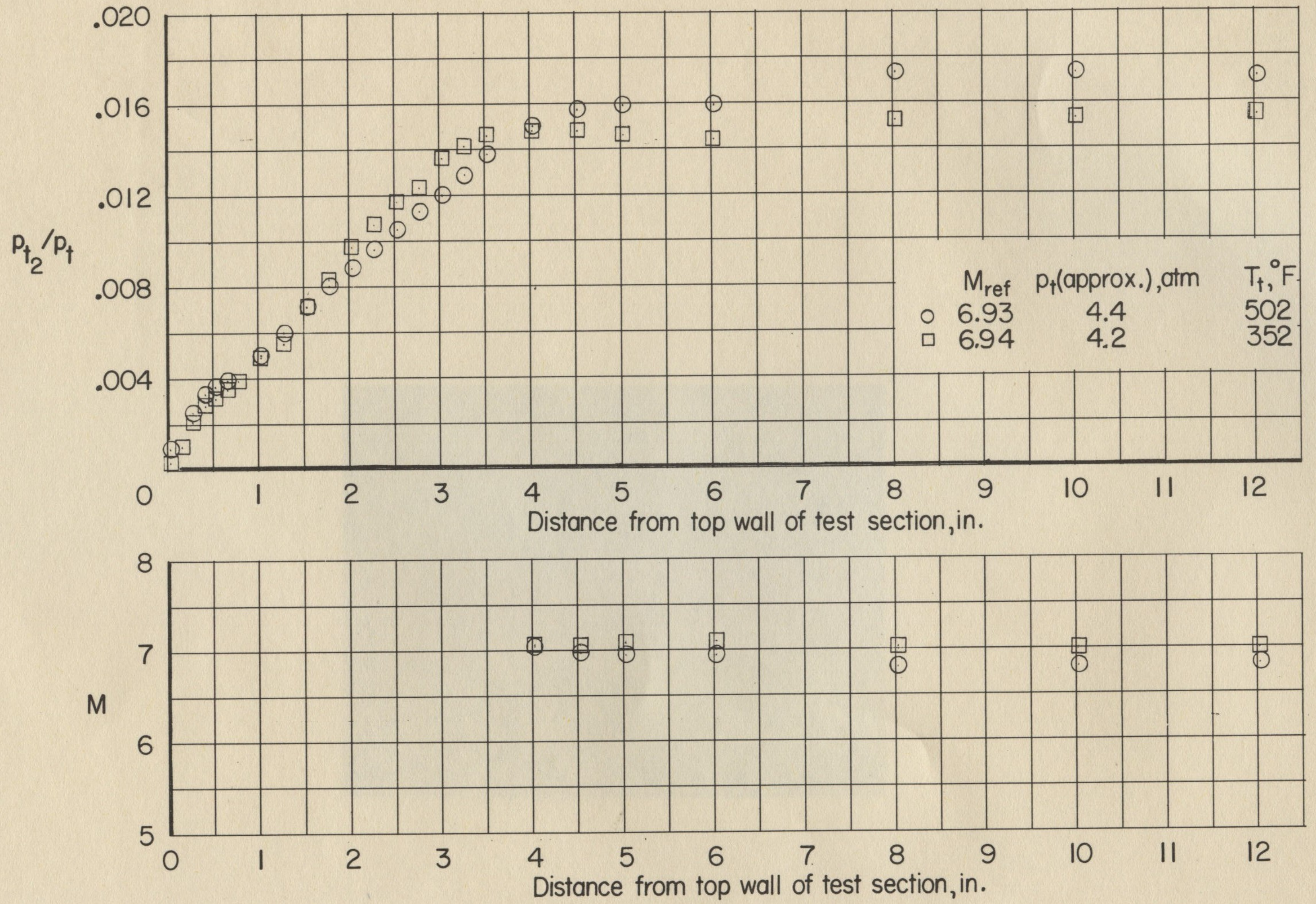
Figure 8.- Effect of temperature on Mach number distribution.



(a) Data obtained at M = 6.

Figure 9.- Thickness of boundary layer at station 110 as determined by total-pressure survey between vertical wall and center of tunnel.

NASA-Langley, 1961 L-1390



(b) Data obtained at M = 7.

Figure 9.- Concluded.

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George M. Stokes. September 1961. 23p. OTS price, $0.75. (NASA TECHNICAL NOTE D-939)

This test facility provides for aerodynamic model test in the Mach number range from 3 to 7 at relatively low densities. The facility can operate continuously by the use of an ejector system driven from compressors having a pressure ratio of 4. Curves are presented to show the ranges of total temperature, total pressure, Reynolds number, dynamic pressure, and static pressure.

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20, Fluid mechanics;
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This test facility provides for aerodynamic model test in the Mach number range from 3 to 7 at relatively low densities. The facility can operate continuously by the use of an ejector system driven from compressors having a pressure ratio of 4. Curves are presented to show the ranges of total temperature, total pressure, Reynolds number, dynamic pressure, and static pressure.

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