# BRL

REPORT NO. 1076
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# BALLISTIC RESEARCH LABORATORIES' NEW HYPERSONIC TUNNEL

J. Sternberg



Department of the Army Project No. 5B03-03-001
Ordnance Management Structure Code No. 5010.11.814

BALLISTIC RESEARCH LABORATORIES

ABERDEEN PROVING GROUND, MARYLAND

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A B E R D E E N   P R O V I N G   G R O U N D,   M A R Y L A N D

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SUMMARY

The B.R.L. hypersonic wind tunnel, which is now under construction, is designed to cover the range M = 5 to 10, and will be capable of continuous flow, variable density operation. Full use will be made of the compressor plant for our existing supersonic wind tunnels. Three new centrifugal compressors have been added to reach a maximum supply pressure of 2200 psi. The tunnel air will be heated to a maximum temperature of 2000°R, using a combustion heater and an electric heater arranged in series.

The effect of air condensation in the flow about models is considered for the simple case of a two dimensional flat plate. In the absence of local supersaturation, the errors in force measurements depend importantly on how close the tunnel air upstream of the model is to the condensation point. When the test section air is at the saturation point, large force errors may result. A large decrease in the force error follows from a relatively small increase in the tunnel supply temperature. If it should prove necessary, over most of the Mach number range, the tunnel can be operated at supply temperatures well above the temperature levels required to just bring the test section flow up to the saturation point.

With the range of supply pressures available, the peak tunnel Reynolds numbers will vary from a maximum of $1 \times 10^6$ per inch at M = 5, to a maximum of $3 \times 10^5$ per inch at M = 10. Except possibly for Mach numbers very close to M = 10, we believe that it should be possible to produce turbulent boundary layers on the wind tunnel models at these Reynolds numbers. The conventional two dimensional type nozzle has certain drawbacks at hypersonic speeds. An axisymmetric nozzle has been chosen for the upper end of the speed range, with the possibility that we may use several such nozzles to cover the range M = 5 to 10. The advantages and disadvantages of this type of nozzle are discussed. The nozzle will be water cooled, and will be made by electroforming to obtain high precision and the desired surface finish and continuity.

Some of the complications in hypersonic flight at high altitudes are discussed. Departures from a continuum, low Reynolds number effects, and oxygen dissociation in the atmosphere are all important above 250,000 ft.

*This paper also appears in Proceedings of Office of Ordnance Research Conference on Fluid Mechanics and Aerodynamics, Fort Monroe, Va., March 1958.

## INTRODUCTION

For the purpose of this paper hypersonic means Mach numbers from 5 to 10. At the present time it is not possible to solve all of the flight problems that are encountered in this speed range in a single type experimental facility. Instead it is necessary to patch together results from many different types of facilities in order to obtain the information that is desired. One of the useful tools for this speed range is a conventional wind tunnel, such as the hypersonic wind tunnel that is now being built at the Ballistic Research Laboratories.

First, we will present a brief account of how the pressure and temperature levels were selected for the wind tunnel and how we expect to achieve these operating conditions. Second, we will consider the design of a nozzle for the upper end of this speed range. Our investigation on how to build such a nozzle has led us to a somewhat unconventional design. Finally, we will make a few remarks on the limitations and uses for a conventional hypersonic wind tunnel. In particular, the complicating factors associated with high altitude flight will be discussed.

### Hypersonic Wind Tunnel Pressure and Temperature Ranges

Mach number 5 is a natural dividing line separating supersonic from hypersonic wind tunnels since above Mach number 5 it is necessary to heat the supply air to keep the air from condensing in the nozzle. The expansion to hypersonic speeds in the test section is accompanied by an extreme decrease in the air temperature. To keep the air in the test section just at the air saturation point at a Mach number of 10, the supply section temperature must be of the order of 2000°R. At first it was hoped that these high stagnation temperatures might not actually be necessary. It is theoretically possible for the air in the test section to be in a supersaturated condition without any significant air condensation occurring. Such supersaturation is known to occur in the case of water vapor condensation in supersonic nozzles. Numerous investigations such as those reported in references 1 and 2, have been conducted in existing hypersonic wind tunnels to determine whether or not supersaturation of the air occurred. Although there was much disagreement in the data obtained in these different tunnels, there seems to be general agreement on the following. Small amounts of water vapor and carbon dioxide normally present in the wind tunnel air themselves condense in the supersonic portion of the wind tunnel nozzle and serve as nuclei for condensation of the air components. It takes time for this air condensation to develop so that in a small wind tunnel the air is probably supersaturated to some degree. On the other hand, it appears that in a large wind tunnel the amount of supersaturation possible in the test section flow may be much smaller and that the supply temperature should at least be high enough to keep the air in the test section at the saturation point. In fact, it may be necessary to keep the air in the test section above the saturation point to avoid errors in flow measurement.

Generally, at some regions in the air flow about a model, the local Mach number will be greater than the test section Mach number and it is important to consider the possible effects of local condensation in the air flow about the model. In Figure 1, the results of some numerical calculations are presented for a simple two dimensional flat plate at a Mach number of 10. Since the Mach number on the top surface exceeds the free stream Mach number, local air condensation may occur in the neighborhood of the model when it has not occurred upstream of the model. The calculations have been carried out for two different cases. First, it is assumed that the air in the test section is just at the saturation point which corresponds to a supply temperature $T_o$ of 1960°R and a supply pressure $P_o$ of 2200 psi, and that in the expansion at the top surface sufficient condensation occurs to just keep the air saturated. The effect of this air condensation is to release heat and thus raise the pressure on the top surface changing the lift. The curve labeled $T_o$ = 1960°R presents this error in lift as a function of the angle of attack of the flat plate. The percentage error in lift reduces as the angle of attack increases because at fairly large angles of attack most of the lift comes from the increase in pressure on the bottom surface of the wing with a relatively small contribution from the decrease in pressure on the top surface. Such large lift errors at small angles of attack would be undesirable.

On the other hand as we have previously noted, there should be some local supersaturation. Applying the data given in Reference 2 to the conditions represented in Figure 1 indicates that the flat plate might reach an angle of attack of 7° before significant condensation occurred on the top surface. The lift would then be error free below $\alpha = 7^\circ$. At still larger angles of attack, the lift error should approach the curve marked $T_o$ = 1960°R. It is clear that at small angles of attack, at a Mach number of 10, the degree of local supersaturation that may be realized is of major importance in determining the accuracy of force measurements.

In principle, condensation could be avoided by increasing the supply temperature. At $T_o$ =3150°R the air would be above the saturation point locally, at all angles of attack less than 7°. But a marked decrease in the lift error can be achieved by a relatively small increase of supply temperature. The results of the calculations for $T_o$ = 2273°R are shown in Figure 1. The air on the upper surface of the flat plate does not reach the saturation point below $\alpha = 2.5^{\circ}$. When condensation does occur, the error in lift is smaller than for the lower supply temperature. As can be seen from Figure 1, by increasing the supply temperature about $300^{\circ}R$, the maximum percent error in lift over a large angle of attack range can be kept below the most optimistic expectations for a lower supply temperature allowing for local supersaturation.

So far there appears to be a limited amount of information on what actually can be expected in the way of errors due to condensation in model testing. We expect to have several nozzles to cover the range from Mach 5 to 10. In order to make it possible to carefully determine the importance of condensation effects at the highest Mach numbers, we have chosen a Mach number of 9.2 for the high end of the Mach number range instead of M = 10. At M = 9.2, supply temperatures several hundred degrees above the saturation temperature are available. Depending upon the results obtained with this nozzle or results obtained elsewhere, the Mach number range may be extended up to M = 10. For Mach numbers below 9, we are not concerned about condensation effects since it will be possible to operate at supply temperatures well above saturation temperatures.

The choice of the supply pressure for the wind tunnel is dictated by our desire to have high Reynolds numbers in the test section, that is high enough Reynolds numbers to enable us to establish turbulent boundary layers on the wind tunnel models. In our experience, at supersonic speeds a Reynolds number greater than $4 \times 10^5$ per inch in the test section is desirable. In order to reach these Reynolds number levels in the hypersonic region, very high supply pressures are necessary. The Reynolds ranges for our existing supersonic wind tunnels are shown on the left hand side of Figure 2. The maximum Reynolds numbers that we expect to reach with our hypersonic wind tunnel are also shown on the figure and except for Mach numbers very close to 10 they are greater than $4 \times 10^5$ per inch. To obtain these Reynolds number values, the supply pressure increases from about 300 psi at a Mach number of 5 to approximately 2200 psi at a Mach number of 7.7. In order to achieve similar Reynolds numbers between 7.7 and 10, it would have been necessary to increase the supply pressure still further. We lost our courage and instead set the maximum supply pressure at 2200 psi above Mach number 7.7. Consequently, the maximum Reynolds number decreases rapidly with Mach number between M = 7.7 and 10.

## Hypersonic Air Handling Plant

The new B.R.L. hypersonic wind tunnel will make use of the existing compressor plant for our supersonic wind tunnels which automatically determines certain features of the wind tunnel. The cross-sectional area of the nozzle exits will be comparable to the test section sizes in our supersonic wind tunnels and will run from about 225 sq. inches at a Mach number of 5 to about 340 sq. inches at a Mach number of 9.2. The hypersonic tunnel, as our supersonic tunnels, will be continuous flow and with variable density. The performance characteristics of the first compressor stage sets a limit on how low a pressure can be used in the wind tunnel. Reynolds number effects at low pressure markedly reduce the performance of these centrifugal compressors. Consequently, there is a minimum tunnel Reynolds number as shown on Figure 2.

The five centrifugal compressors of the supersonic facility will be placed in a three stage arrangement as the first stages of compression for the hypersonic wind tunnel. These first three stages will develop a compression ratio from the high to the low side of 12, and can deliver air at a pressure level of 100 psi. In order to reach the much higher compression ratios and pressure levels needed for the hypersonic tunnel, we have added three centrifugal compressors arranged in series. The new compressors are shown in Figure 3. Originally some thought was given to using a positive displacement, carbon ringed compressor for the last stage. The last machine produces a compression ratio of 3.1 at an intake volume of about 500 cubic feet per minute. Small high pressure centrifugal compressors that would meet these performance requirements were not available. On the other hand, there were some obvious disadvantages to the positive displacement compressor. It probably would be necessary to add a pulsation damper to the system and there was also considerable doubt as to whether the carbon rings would stand up in service in view of the fact that the tunnel air is extremely dry. We compromised by using a somewhat oversize centrifugal compressor and by-passing some of the air around the compressor. The resultant increase in power consumption is not significant.

The first machine is of the split casing type with interstage diaphram cooling and consumes approximately 7000 horsepower. The last two machines are barrel type compressors and are rated at 5000 horsepower and 3500 horsepower respectively. Each compressor is driven by a separate synchronous motor through speed increaser gears.

At maximum pressure and temperature, air is delivered from the last compressor at 2250 psi at a temperature of about 950°R. In order to reach the design temperature of 2000°R, two different types of air heaters are used. From an economical standpoint, it would have been preferable to attain the design temperature with a combustion heater; however this possibility was not within the state of the art and the maximum air temperature from the 8000 KW combustion heater was set at 1800°R. A 1000 KW electric heater is used to raise the air temperature from 1800°R to 2000°R. The heat exchange in the combustion heater takes place in the rectangular section at the top of the cylindrical base.

(See Fig. 3) The tunnel air circulates through stainless steel thick walled finned tubes running back and forth across the opening normal to the axis of the heater. Hot combustion gases from an oil burner at the bottom of the combustion heater then pass, with the help of an air blower, over the finned tubing and out of the top of the stack. The upper limit on the temperature that can be obtained in a heater of this type is set by the design of the high pressure fin tubing.

The air is then brought from the outlet manifold of the heat transfer section of the combustion heater to the inlet manifold of the electric heater in six 3" O.D., 2" I.D. stainless steel tubes. These tubes are relatively flexible and permit differential motion of the electric heater which rests on a roller support and the combustion heater which is rigidly attached to the ground. Some relative motion between the heaters will occur due to thermal expansion of the tunnel components during operation. The air then passes down through the vertical electric heater finally following a $90^{\circ}$ turn into the supply section. The electric heater consists of a steel shell,water cooled on the outside and insulated on the inside, and it is packed full of hollow inconel tubes which serve as resistance heating elements. The tunnel air flows both inside and outside of these tubes. Power is supplied to the electric heater through a saturable reactor which makes it possible to have continuous voltage control from 25% to 100% of full power. The ability to get a smooth, fine control on the electric heater power input should be of considerable importance in maintaining constant supply conditions during operation of the wind tunnel. As can be seen from Figure 3, the air always passes through the electric heater but may be by-passed about the combustion heater which is not needed at low Mach numbers. By either having all the tunnel air go through the combustion heater or by-pass the combustion heater, we are able to avoid the problem of mixing air streams at two widely different temperatures between the combustion heater and the electric heater. By always having the electric heater in the circuit section, we can take advantage of the superior temperature control of the electric heater as compared with the combustion heater.

In Figure 3, a temporary circuit is shown in place of the nozzle, test section, and diffuser. This temporary circuit consists of a water cooled axisymmetric orifice to control the mass flow, followed by a low pressure pipe. We expect to use this temporary circuit to operate the air handling plant prior to the installation of the tunnel itself. Downstream of the low pressure pipe is the model catcher and the tunnel aftercooler. The air leaves the aftercooler at about $560^{\circ}R$ and returns to the intake side of the supersonic compressor plant.

## Nozzle Design

The favored type of supersonic wind tunnel nozzle is a two dimensional design with two flat walls and two contoured walls. There are certain disadvantages to a two dimensional nozzle design at Mach numbers over 8. First of all, the throat becomes a very narrow slit. For instance, if the test section is 15" square then at a Mach number of 9.2 the throat would be a section .041" high and 15" wide. At M = 9.2, a 1% change in throat height corresponds to a 1% change in dynamic pressure in the test section which in the case of a force measurement corresponds to a 1% change in force. Therefore, variations in throat height of the order of .0004" can be significant. The heat transfer to the nozzle walls reaches a maximum very close to the throat and at $2000^{\circ}R$ and 2200 psi supply pressure, the heat transfer to the tunnel wall at the throat would be of the order of 8 B.t.u./$in^2$/sec. It is difficult to design a throat section for these heat transfer rates which will not distort due to thermal expansion more than .0004". We can make a one dimensional estimate of the effect of throat distortion as follows. Consider the channel to be divided by a series of planes parallel to the side walls. Suppose that in each section formed in this way, the flow is independent of the flow in neighboring sections. Then the Mach number at the test section in each section is given by the ratio of the height of the throat for that section to the test section height, which is the same for all sections. On this basis, a variation in throat height from one side of the tunnel to the other of the order of .0004" would produce a 1% non-uniformity of dynamic pressure in the test section. Of course, the flow in each of these longitudinal sections cannot be independent of each other. Possibly lateral relief of pressure differences, as they develop downstream of the throat, might greatly reduce the importance of a given throat distortion on the Mach number distribution in the test section.

At our request, an investigation of the effect of a simple throat distortion on the downstream flow in a nozzle was carried out at GALCIT. We chose a throat section with a linear variation in height across the tunnel width. The results of this investigation have been reported by Oliver and Cummings[3]. They found that, as suspected, the simple one dimensional model is incorrect; however the results were not of any comfort to wind tunnel designers. Although the distribution of Mach number was not that predicted by the simple one dimensional model, the magnitude of the Mach number variation was about the same as the one dimensional prediction. A second disadvantage of two dimensional nozzles for high Mach numbers arises from the nature of the boundary layer development on the flat side walls. At a given axial position in the nozzle, the boundary layer on the side wall may be non-uniform in thickness. The result will be flow non-uniformity in the test section. A third difficulty is in adequately sealing the wind tunnel channel in the vicinity of the nozzle throat. To overcome these disadvantages, we have been considering an axisymmetric nozzle for the high end of the Mach number range. At a Mach number of 9.2 for the same test section area as before, the throat diameter is now .88". The symmetry of the design reduces the likelihood of throat distortion and in any case a given absolute distortion should have a less severe effect on the downstream flow. Also because of the symmetry, the boundary layer will be of uniform thickness at each axial position and the sealing problem at the throat, as well as along the rest of the nozzle, is eliminated.

There are, as might be expected, certain disadvantages to this scheme. An axisymmetric nozzle is subject to focussing effects whereby disturbances at the nozzle walls may concentrate and produce unacceptable disturbances near the axis. Thus, it is necessary to avoid small concentric irregularities in the construction of the nozzle. At hypersonic speeds, the tunnel boundary layer is much thicker than at low supersonic speeds and may mask,to some degree, what surface irregularities are present. On the other hand, the nozzle contour must be shaped to allow for the effect of the boundary layer displacement thickness on the flow in the nozzle. Since this is a large correction, if uniform flow is to be achieved in the test section, it is important to be able to make an accurate calculation of the boundary layer growth in the nozzle. Unfortunately, a relatively limited amount of information is available on high Mach number turbulent boundary layers with high heat transfer[4] and it is not clear that the desired accuracy can be achieved on the first attempt. Secondly, when the tunnel Reynolds number is changed, the boundary layer thickness will also change and therefore change the flow in the wind tunnel. If the change in boundary layer only results in a change in the Mach number level in the test section, no harm will be done since this can be taken care of by calibration. The significant question is whether the Mach number distribution in the test section will still be uniform at other than the design pressure. Finally, the observation of the model in the test section is more limited than with a two dimensional nozzle, where there are no restrictions on the use of flat windows.

There are then two main problems in the design and construction of an axisymmetric nozzle. Although the axisymmetric shape avoids the problem of throat distortion, the structural design of the throat section is still formidable. The other essential problem is how to fabricate the nozzle to the required tolerances on shape and surface continuity. First we will consider the throat design.

To restrict the temperature rise in the nozzle at the throat, we plan to use a high pressure cooling water system. The water will flow axially downstream along the nozzle in a passage formed by the outer surface of the nozzle liner and the inner surface of a concentric section called the water director. By using water pressures of 600 psi, water velocities of about 100 ft. per sec. can be used at the throat region. Another possibility would have been to use some form of boundary layer cooling. This might have consisted of admitting cold air at the boundary of the tunnel upstream of the throat to reduce the heat transfer at the throat section. We have chosen the water cooling system because of our uncertainty as to what influence a boundary layer cooling system might have on the flow distribution in the tunnel. It turns out that with our supply pressure and temperature, we are just about at the limit for the cooled wall solution for the throat design. For structural reasons, it is undesirable to make the throat section much thinner than 1/4". The maximum thermal stresses at the throat essentially depend on the parameter $\alpha E/\lambda$ where $\alpha$ is the coefficient of thermal expansion, E is the modulus of elasticity, and $\lambda$ is thermal conductivity for the material. In Table I, the values of this parameter, and the resulting thermal stresses for a 1/4" thick water cooled throat section are given for nickel and beryllium copper. The beryllium copper is a copper alloy which contains a small amount of beryllium and cobalt. Its heat transfer coefficient is almost 60% that of copper, whereas it has much better strength than copper at elevated temperatures. Even with this material, the temperature difference across the liner is of the order of $400^{\circ}R$. With a nickel throat, which is a reasonably good heat conductor, the temperature difference is about $900^{\circ}R$. The thermal stress of 55,000 psi with beryllium copper is tolerable, whereas the much higher stress with a nickel liner would be unacceptable. Therefore, we plan to make the throat section out of beryllium copper.

We believe the best technique for fabrication of the nozzle is to make it by electroforming. A mandrel for the electroforming can be machined without too much difficulty and can be readily hand polished in a random way to produce the desired surface finish and continuity. Nickel is then electrodeposited to the design thickness on the mandrel. After any necessary machine work on the outside of the liner, the mandrel is removed. A very accurate reproduction of the shape and surface of the mandrel can be obtained in this way. Using a suitable end design for the throat block, the nickel section of the nozzle can be mechanically joined to the beryllium copper throat block as a byproduct of the electrodeposition. The nickel can be deposited on the downstream end of the throat block which serves as the upstream end of the mandrel.

In order to permit a careful evaluation of the aerodynamic performance of an axisymmetric hypersonic nozzle design, and to get some experience with electroforming, a small scale model of our planned axisymmetric nozzle has been constructed and is presently undergoing tests in the hypersonic facility at GALCIT. A sketch of this model tunnel is shown in Figure 4. Because the available temperatures in the GALCIT facility are lower than those we will have available in the B.R.L. tunnel, the nozzle has been designed for a Mach number of 8.8. The lower supply pressure of 450 psi and supply temperature of 1500°R greatly reduces the heat transfer at the throat so that this is not a problem in the model tunnel. Accordingly, the liner for the model tunnel was made completely out of electroformed nickel roughly 1/4" thick. A photograph of the nozzle liner is shown in Figure 5. As shown in Figure 4, the nozzle ends inside a sealed chamber and there is a "free jet" in the test section. The air stream is captured by a scoop and further downstream passes through the supersonic diffuser not shown in the figure. Models in the "free jet" section of the tunnel flow can be viewed through flat windows in the side of the plenum chamber. We believe this arrangement has some advantages. More flexibility is provided for installations around the test section and the heating of the windows is much less than if the windows were placed in contact with the high speed air stream. The flow disturbance from the end of the nozzle propagates towards the axis at a very shallow angle and does not appreciably reduce the available model testing region. If necessary, the region of uniform flow inside the nozzle can be used, though in that case only part of the model can be viewed optically. We hope to have the aerodynamic results from these tests within a few months.

The B.R.L. Mach number 9.2 hypersonic wind tunnel nozzle, as presently designed, is shown in Figure 6. The horizontal cylindrical section of pipe on the left is available for use as a thermal equalizer if we are dissatisfied with the air temperature distributions from the electric heater. The nozzle assembly is mounted on wheels and can be removed as a unit leaving the test section chamber and the thermal equalizer section in place. This would allow us to interchange nozzles for different Mach numbers. The section immediately downstream of the test section chamber is the supersonic diffuser. Further downstream is the model catcher section which is designed to prevent high speed fragments from the test section from damaging the tunnel aftercooler. We are not willing to count on all our model designs working. Since the test section chamber and the downstream sections of the tunnel are designed for low pressures, a safety valve is located in the test section. The duct sitting on top of the test section chamber is there to direct the high temperature air out through the roof of the tunnel room if for some reason air pressure builds up in the test section region.

## Role of the Conventional Hypersonic Wind Tunnel

It is clear that a conventional hypersonic wind tunnel of the type we are constructing at the B.R.L. cannot be used to solve all of the flight problems between M = 5 and 10. In flight air temperatures near the model will be large enough to cause partial or complete dissociation of the air. In the wind tunnel, the air temperature will not exceed the supply temperature which is well below the temperature level at which dissociation begins. Thus, those problems where the thermodynamic and transport properties of air at temperatures between $3000^{o}R$ and $10,000^{o}R$ are important cannot be directly solved in a conventional wind tunnel. A second group of flight problems in the Mach number range of 5 to 10 that are inaccessible to a conventional hypersonic tunnel arise from flight at altitudes above 250,000 ft. These upper atmosphere effects are represented on Figure 7. It seems likely that if the ratio of the mean free path to the boundary layer thickness is greater than .1, then some deviation from continuum flow will occur. As we see from Figure 7, the mean free path in the atmosphere becomes appreciable above 250,000 ft. and by 350,000 ft. is of the order of 2.5 ft. At a foot from the leading edge of a flat plate at a Mach number of 10 with a surface to free stream ratio temperature of about 6, the ratio of the mean free path to the boundary layer thickness[5] would be greater than .1 above 320,000 ft. Also as shown on Figure 7, the Reynolds number for a length of one foot becomes less than $1 \times 10^2$ above 280,000 ft. This roughly constitutes the lower Reynolds number boundary for using ordinary boundary layer theory and we enter a new regime at higher altitudes. A third complication follows from the fact that above 250,000 ft., a varying percentage of the oxygen in the air is dissociated. By 370,000 ft., the oxygen is almost entirely dissociated. This dissociation is not due to high air temperatures but is produced by radiation from the sun which is absorbed in these layers. The recombination of the dissociated atoms depends on three body collisions, which at the low densities at these altitudes proceeds very slowly. Furthermore, the amount of oxygen dissociation in the atmosphere is not a fixed function of the altitude but instead varies with the time of day and from day to day. One of the reasons for this variability is that vertical air mixing plays a very important role in determining the distribution of atomic oxygen in the atmosphere, as has been shown by Nicolet and Mange[6]. Thus, it would appear that in the range of altitudes from 250,000 ft. to perhaps 370,000 ft., hypersonic aerodynamics will be complicated by departures from continuum flow, low Reynolds number effects, and also by the dissociated state of the oxygen in the air. Free molecule flow will be reached by 400,000 ft.

A conventional hypersonic wind tunnel is useful for stability and control investigations and for fundamental investigations of hypersonic flow, such as are described elsewhere in these Proceedings. Its value in studies of boundary layer and mixing phenomena should be specially mentioned. The excessive heat transfer rates encountered at hypersonic speeds have turned the attention of aerodynamicists towards novel and varied methods of boundary layer cooling or boundary layer control. While real gas effects will have to be accounted for eventually, it may be a considerable advantage to be able to study the fundamental fluid mechanics aspects of these problems without having to worry about such factors as the state of dissociation of the gas, appropriate values for the transport properties, and possible surface reactions.

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TABLE I

For 1/4" Thick Water Cooled
Throat Section

| | $\alpha$ | K | E | $\frac{\alpha E}{K}$ | $\Delta T$ | Maximum Thermal Stress |
|---|---|---|---|---|---|---|
| Berylco 10 | $9.8 \times 10^{-6}$ | 1600 | $19 \times 10^{6}$ | .12 | $400^{o}R$ | 55,000 psi |
| Nickel | $8.6 \times 10^{-6}$ | 380 | $30 \times 10^{6}$ | 0.67 | $900^{o}R$ | 140,000 psi |

$\alpha$ = coefficient of thermal expansion

K = thermal conductivity BTU's/ft$^2$/in/hr/$^o$F

E = Youngs Modulus

$\Delta T$ = Temperature difference across liner thickness

Composition of Berylco 10

| | |
|---|---|
| .4 to .7 | beryllium |
| 2.3 to 2.7 | cobalt |
| Balance | copper |

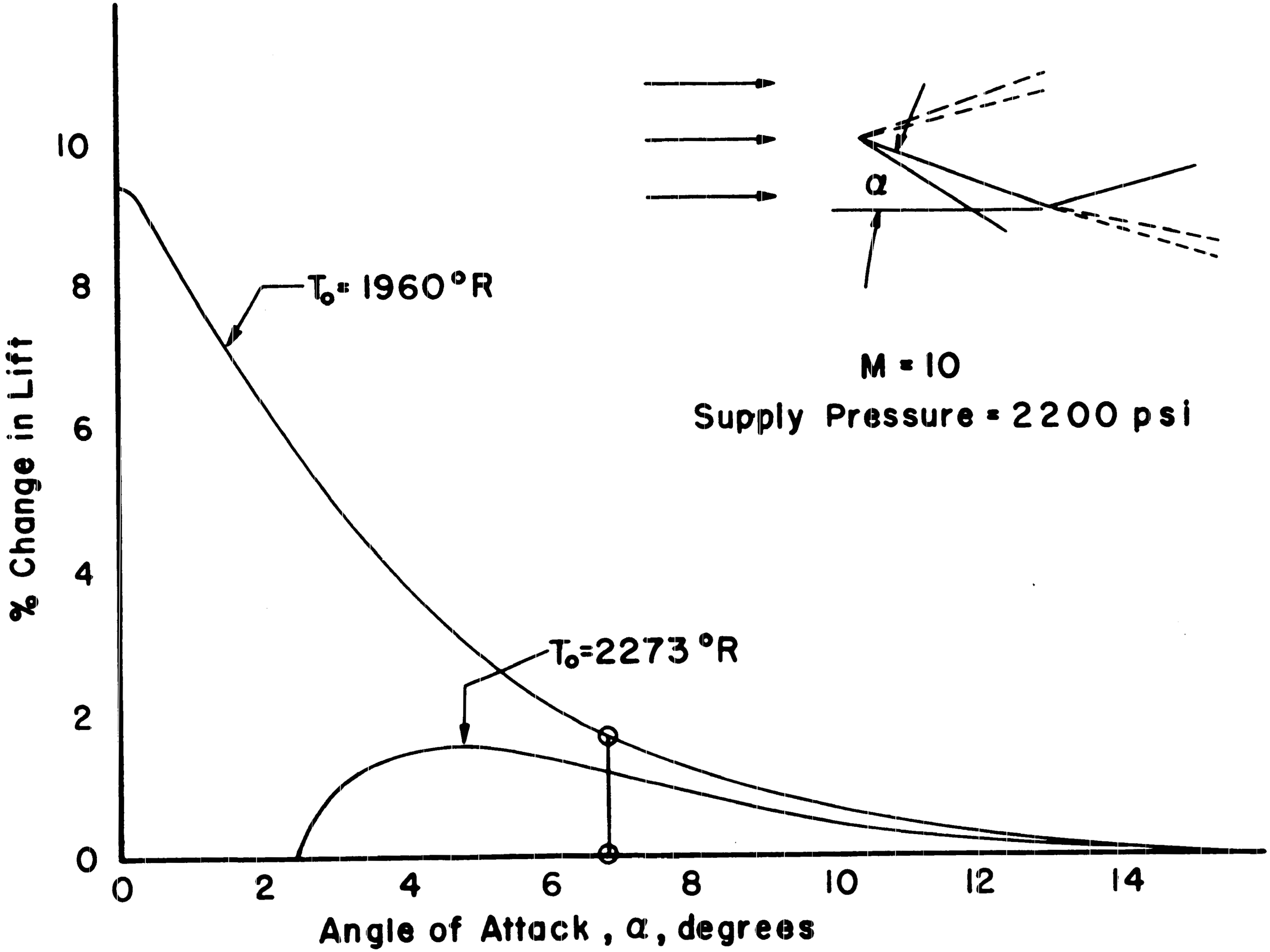
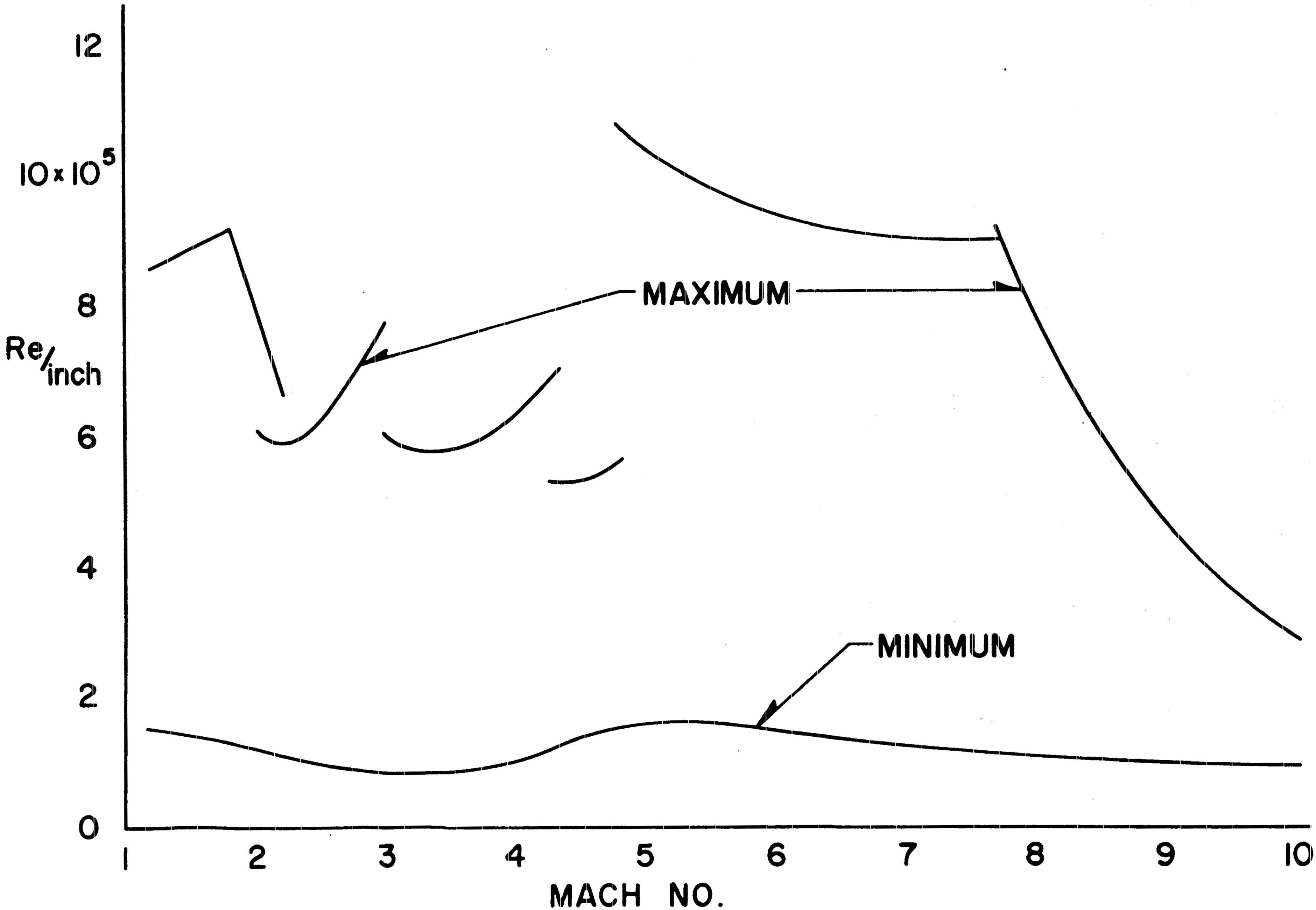


Figure 1. Lift Error due to Air Condensation




Fiure 2. Tunnel Reynolds Number Range

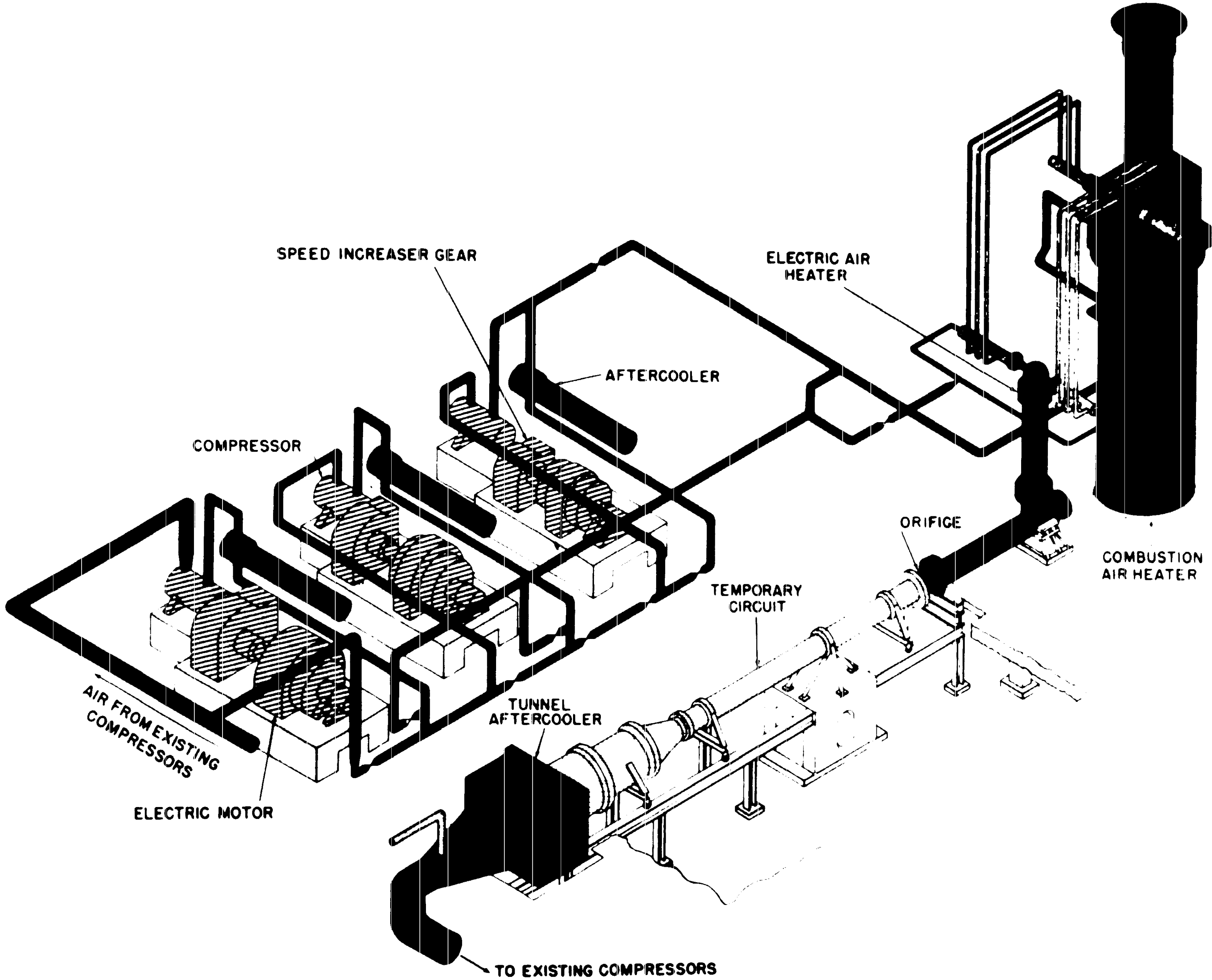



Figure 3. Air Handling Plant

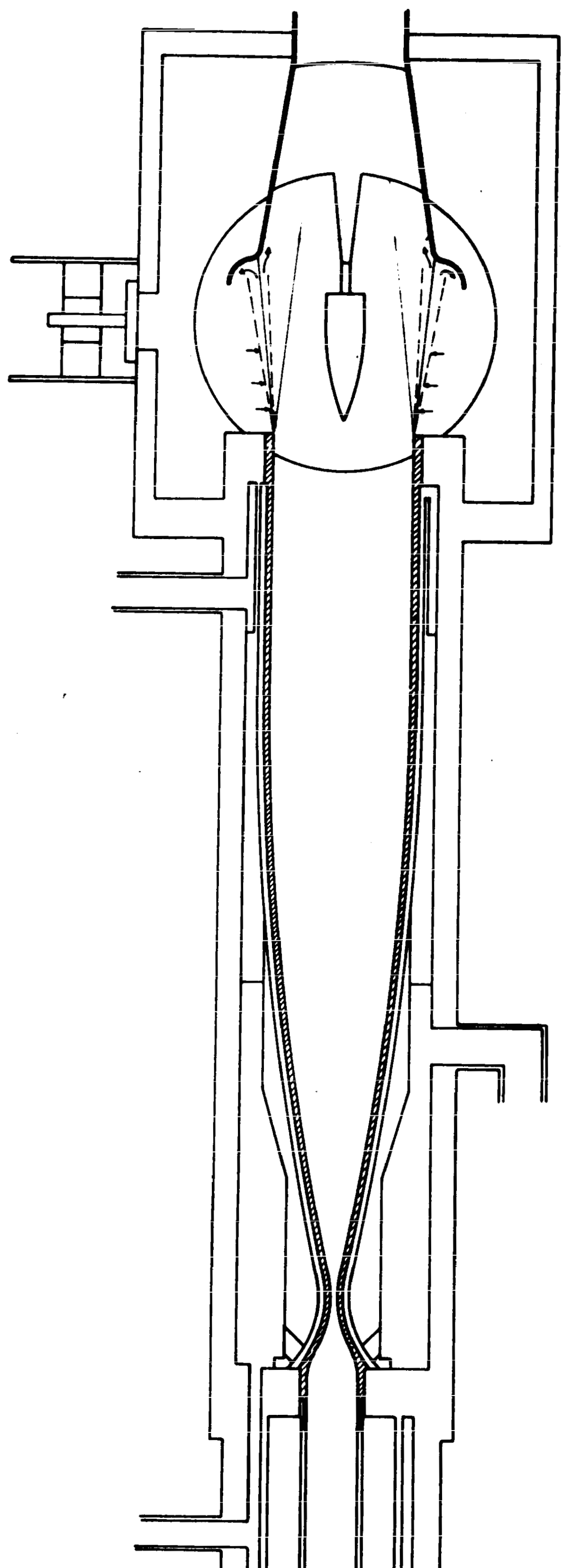


Figure 4. Axi-Symmetric Model Tunnel

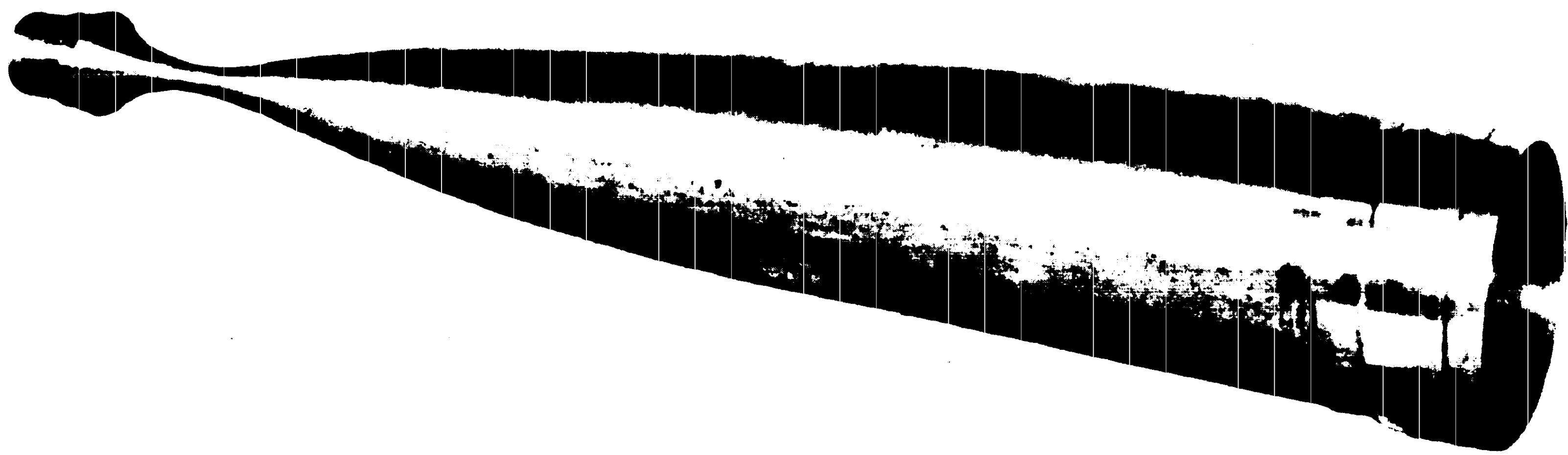


Figure 5. Electroformed Nickel Nozzle

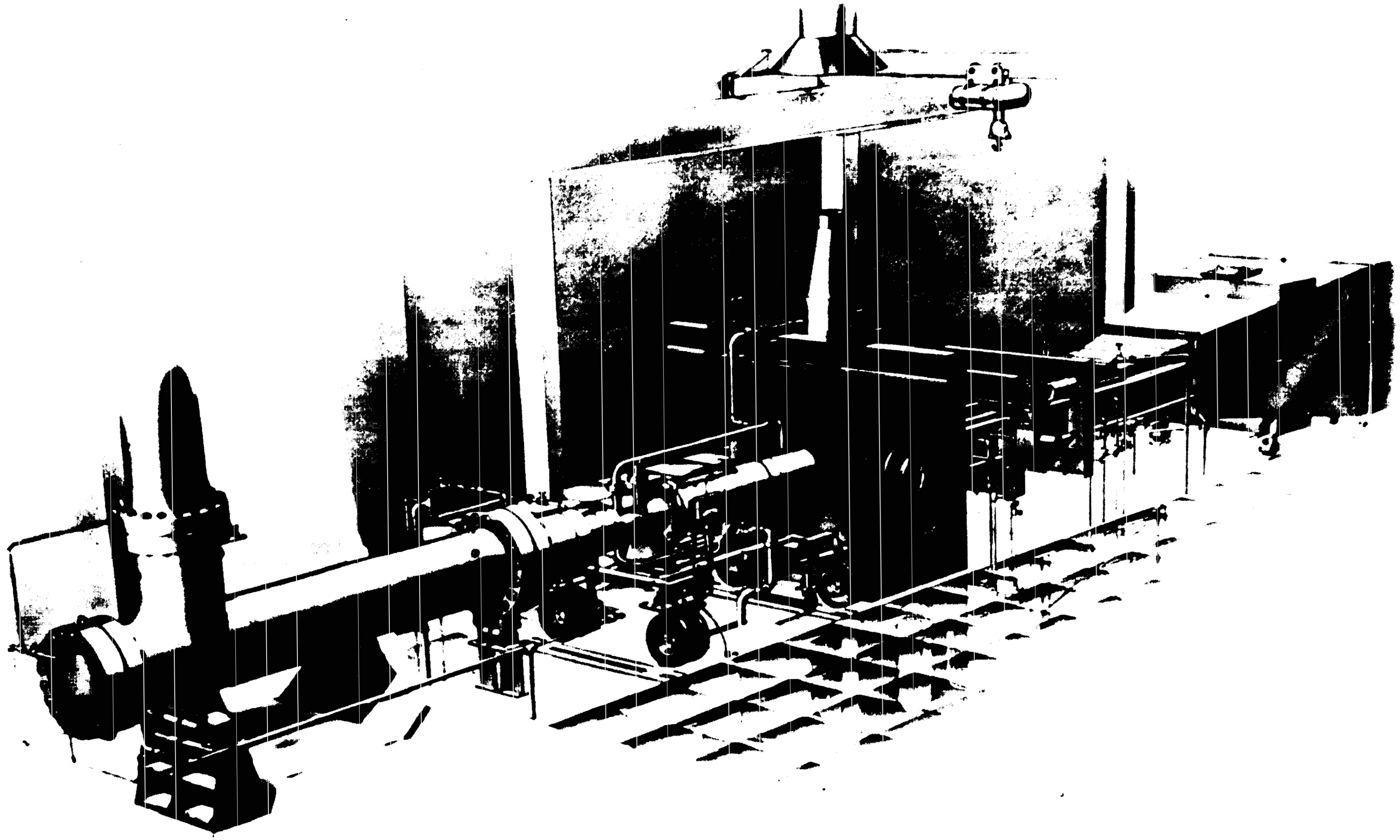


Figure 6. M=9.2. Nozzle, Test Section, and Diffuser

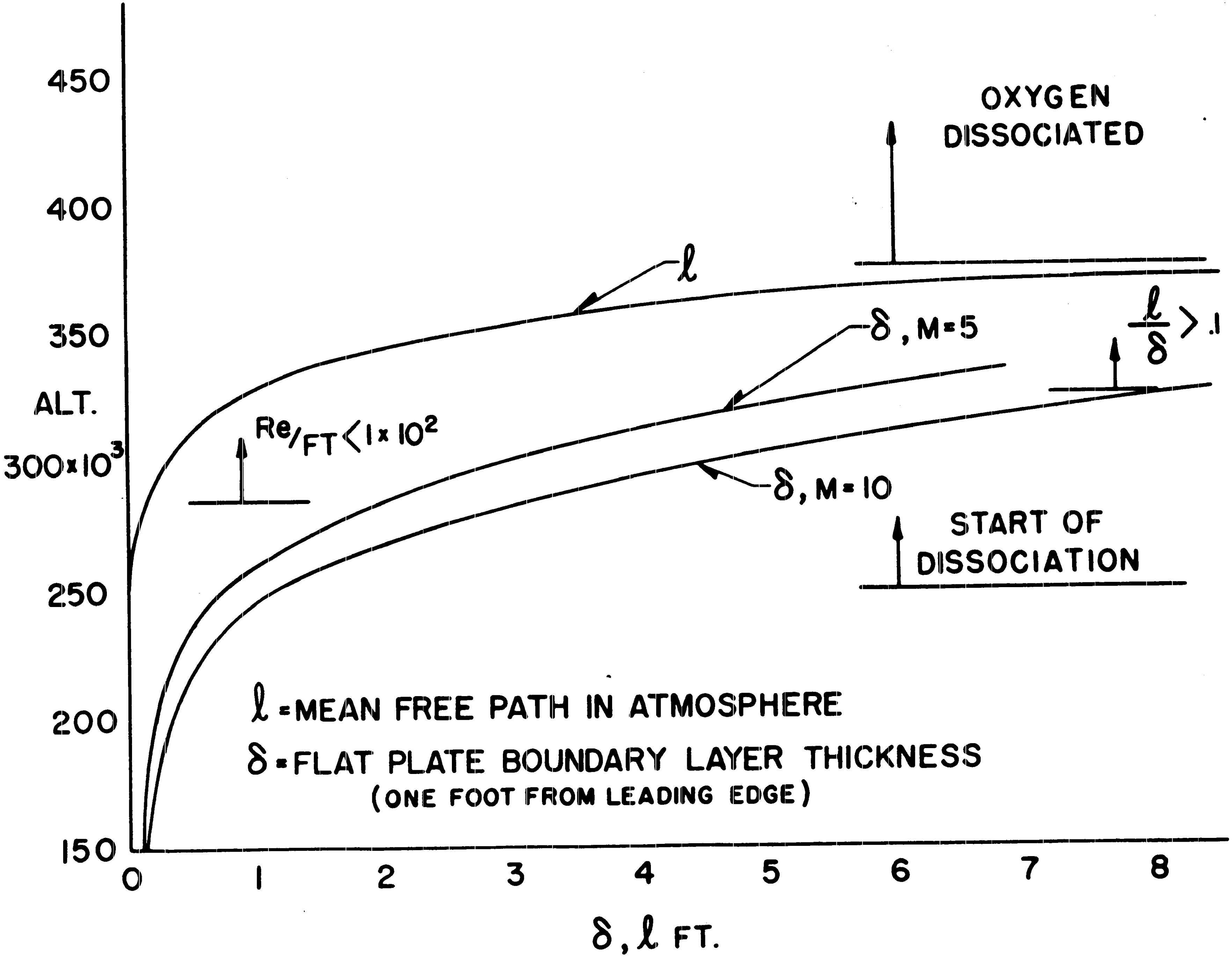



Figure 7. Hypersonic Flight at High Altitudes

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