The helium hypersonic tunnel at APL is described, and the raison d'être for such research equipment is discussed. In a test of a shock-duct diffuser in this tunnel, it was found that such a diffuser is capable of providing normal shock recovery. Results of the interaction between an incident shock wave and a laminar boundary layer indicate that the fluid model developed from lower Mach-number studies is essentially valid at hypersonic speeds if certain modifications are taken into account.

THE HELIUM

Even before the flight of the Wright brothers, the aerodynamic wind tunnel was used to test components of proposed "flying machines." The Wrights themselves used a wind tunnel to test the lift and drag characteristics of their vehicle and its components. As the design of aircraft became more sophisticated, the use of the wind tunnel to provide design data and research information increased.

For low-speed aircraft, the duplication of flight conditions in a wind tunnel was a relatively simple matter. One had only to duplicate the velocity, Reynolds number, and, in some cases, the turbulence level encountered by the vehicle in flight. As aircraft speeds increased, the effects of compressibility became significant and required that the Mach number as well as the Reynolds number be duplicated if simulation was to be achieved. As flight velocities increased even further, the simulation requirements became even more difficult to attain. This led to the introduction of wind tunnels providing only partial simulation. However, even with these tunnels, much useful data can be obtained on the full-scale requirements of flight vehicles.

The difficulties in achieving complete Mach-Reynolds number simulation between a model in a wind tunnel and a chosen flight condition can best be illustrated by an example. The Reynolds number, $Re = UL\rho/\mu$, is related to the Mach number M by $Re = M^2(\gamma pL/\mu U)$, where γ is the ratio of specific heats, p is free-stream static pressure, μ is the coefficient of viscosity, U is the velocity, and L is a length representative of the model. This equation indicates that if full M-Re simulation is to be obtained, the ratio $\gamma pL/\mu U$ in the tunnel must be equal to the flight value. If a wind tunnel operating at a stagnation temperature* of 60°F were used to test a $\frac{1}{10}$-scale model at Mach 1, the required ratio of tunnel stagnation pressure to the ambient pressure at the altitude under consideration, p_t/p_a, is 23.7, while for Mach 3, $p_t/p_a = 1360$. If the tunnel gas is heated so that the stream temperature in the tunnel is equal to the ambient temperature in flight, the required ratios at Mach 1 are $p_t/p_a = 18.9$ and $T_t/T_a = 1.2$. For Mach 3, these ratios are 370.4 and 2.8, respectively.

The heating of the gas has other favorable effects in addition to the reduction in tunnel pressure ratio. For example, in a tunnel operating at a stagnation temperature of 60°F and a stagnation pressure of 60 atm, the oxygen in the air would condense at about Mach 3.3. Since this condensation would cause drastic changes in the downstream flow, the air must be heated to delay the onset of this phenomenon. Also, at high Mach numbers, heat transfer becomes important, and the ratio of the model wall temperature to free-stream temperature, as well as the Mach and Reynolds numbers, must be duplicated. Further increases in flight Mach number, and thereby temperature, lead to the introduction of real-gas effects for the simulation problem.

For a vehicle flying at Mach 10 at 30,000 ft, the stagnation temperature is of the order of 5000°F. At this temperature, some oxygen dissociation has already taken place. Since for these high Mach numbers the temperature varies approximately as the square of the Mach number, it can be seen that further increases in M would result in more dissociation of the air, and even ionization. Therefore, if complete simulation is to be obtained, one must duplicate, in addition

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* Stagnation conditions are those that would exist if the flow were isentropically decelerated to zero velocity.

R. A. Makofski

HYPERSONIC WIND TUNNEL

to Mach number, Reynolds number, and temperature ratio, the correct absolute temperature. The conventional wind tunnel cannot hope to satisfy these conditions because the temperature requirements are far above the limits set by the materials used in the settling chamber and nozzle. New techniques using high temperatures generated by either adiabatic compression, strong shock waves, or arc heating have the possibility of achieving full simulation. These facilities, however, permit such short testing periods that, combined with the high temperatures that are present, there is great difficulty in obtaining detailed measurements.

The hypersonic flow problem mentioned above may, in general, be divided into two categories. First are the fluid-mechanical aspects of hypersonic flow, namely, the study of high Mach number flow about bodies, the effects of the interaction between the external flow and the boundary layer, and the effects of strong entropy gradients, high heat-transfer rates, and low densities. Second are the chemical or physico-chemical aspects of hypersonic flow, including a study of dissociation and ionization, molecular and chemical effects and their reaction rates, and the effects of these phenomena on viscous flow.

The first of these categories can be studied in a wind tunnel under controlled conditions. Such tests could then be used either to verify a theory or to construct a fluid model upon which a theory could be built. However, the conventional air tunnel is now limited to Mach numbers of about 10, at which speed some of the "hypersonic effects" are of only limited importance; as vehicles of higher speed are designed, however, these effects will become significant and will require development of tunnels of higher Mach numbers.

Since a fluid-mechanical theory, if correct, would predict the hypersonic effects on any gas, the possibility of using a medium in which significantly higher Mach numbers may be attained without condensation and without the added complexity of heating the gas is worth investigating. A study of various gases has shown helium to be particularly suited for this application. Helium is a safe, nearly perfect gas with an extremely low condensation temperature. In addition, it is readily available in quantities suitable for use in a small research facility.

The main disadvantage in using helium is that the data obtained are not directly comparable with air since certain quantities, such as specific heat ratio and Prandtl number, differ from those of air. (Prandtl number is given by $P_r = \mu c_p / K$, where c_p is specific heat at constant pressure and K is thermal conductivity of the gas.) For such a comparison to be made, the relation between the parameters being studied must be known. This relation may be obtained either from a theoretical analysis of the problem or from data collected for different values of the specific heat ratio.

The areas of interest in wind tunnel work that will now be discussed can be placed in two divisions: development of the wind tunnel and its instrumentation; and studies of fluid flow and its interaction with solid surfaces. Examples of the work performed in the helium hypersonic tunnel in each of these two areas are given.

The APL Helium Hypersonic Wind Tunnel

A schematic view of the APL helium hypersonic tunnel, which was developed by the Research Center under the direction of F. K. Hill,
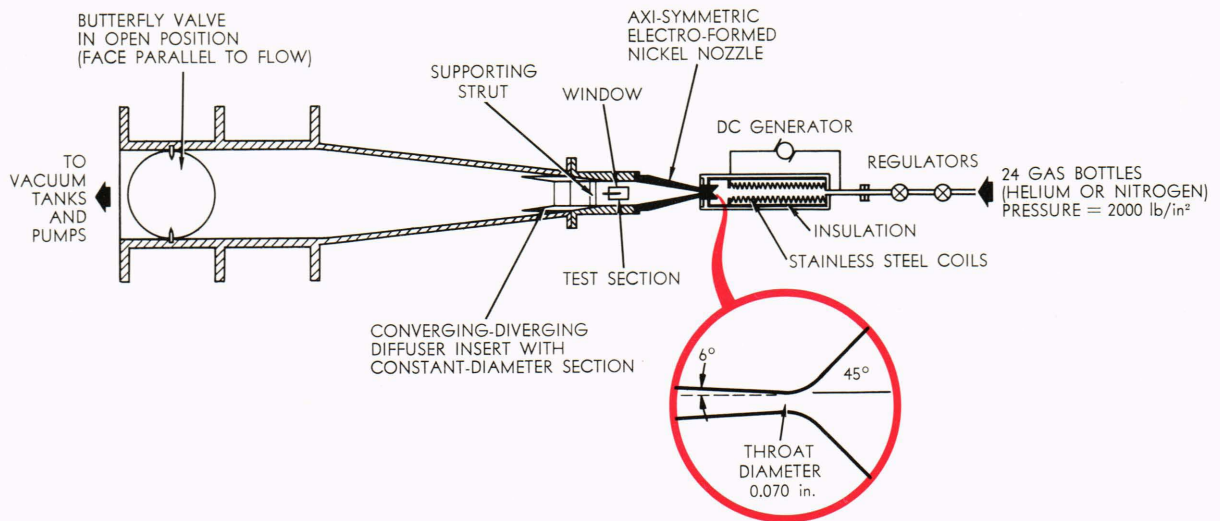
Fig. 1—Schematic of the helium hypersonic wind tunnel.

is shown in Fig. 1. Helium is supplied at 2000 psia through the control panel regulators to the supply chamber of the wind tunnel. This chamber is equipped with a heating unit. The nozzle, of electroplated nickel, consists of a 12° total-angle cone with a 0.070-in. throat and a 2.0-in. exit diameter. After expansion in the nozzle, the gas flows through the test section and diffuser and into 650-ft³-volume vacuum tanks, which are evacuated by vacuum pumps with a capacity of 1900 ft³/min.

The Mach number distribution along the centerline of the present tunnel is shown in Fig. 2. The drop in Mach number as supply pressure p_o decreases is a result of the increase in the boundary layer thickness.† The shock waves occurring at the lower values of p_o are caused by disturbances in the diffuser, test section, and nozzle exit feeding upstream through the subsonic portion of the boundary layer. This causes flow separation, and thereby shock waves, in the nozzle.

Studies of a Shock-Duct Diffuser

The primary use of diffusers in supersonic wind tunnels is to reduce the pressure ratio necessary to maintain supersonic flow. This decrease in pressure ratio permits an increase in the operating time of blow-down tunnels, i.e., tunnels using a limited amount of high-pressure gas discharging through a nozzle. Also, the higher pressures available downstream from the diffuser allow more efficient and less expensive methods of recompressing the gas. For this reason, a great deal

of effort has been given to determining the optimum configuration for a diffuser as a function of the wind-tunnel geometry and Mach number.

At the lower supersonic Mach numbers, the variable-geometry, convergent-divergent diffuser has proved successful, providing pressure recoveries well in excess of the recovery from a normal shock wave at the test section Mach number. (A variable-geometry diffuser is one in which the minimum cross-sectional area of the diffuser may be varied while the tunnel is in operation.) However, as the Mach number increases, viscous losses become severe, and normal shock recovery is difficult to attain. Also, the pressure recovery obtainable from multiple oblique shock reflections is reduced because of the shallow shock angles associated with higher Mach numbers. The latter problem may be alleviated by
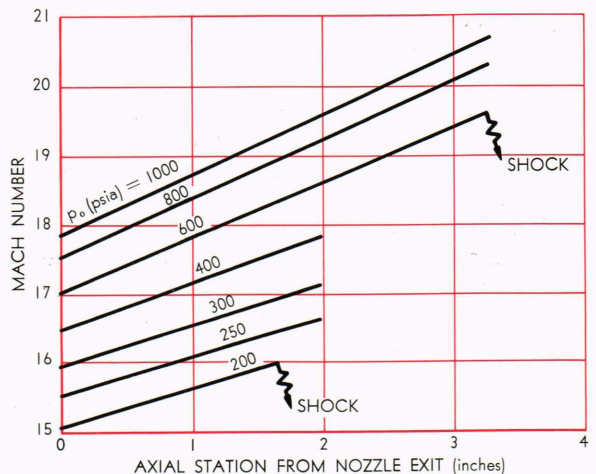

Fig. 2—Mach number distribution in the helium hypersonic wind tunnel with a 12° conical nozzle.

† For a given temperature and Mach number, Re is directly proportional to the supply pressure. The boundary layer thicknesses in these cases vary as approximately $1/Re$.

using a constant, or nearly constant, area duct between the convergent and divergent sections of the diffuser. With this arrangement, the shock waves formed upstream of the diffuser are reflected from the walls of the diffuser. This multiple reflection process leads to increased pressure recovery but must be balanced against the viscous losses incurred by having the longer duct lengths.

Experimental investigations at APL of diffusers at hypersonic speeds in air have been limited to Mach numbers of less than 10. Some of these tests have shown that the use of a shock duct can increase the recovery, especially if the original recovery is less than that obtained through a normal shock. The only available data at Mach numbers greater than 10 are those of Johnston and Witcofski[1] at Mach 20 in helium. These tests were conducted in a 3.0-in.-diameter tunnel having a conical nozzle and equipped with a two-dimensional, variable-geometry, convergent-divergent diffuser. The optimum recovery reported was 0.6 of the recovery possible from a normal shock at the nominal test section Mach number. This result, in conjunction with those noted at APL, suggested that a shock duct may result in significantly better recovery. For this reason, an investigation was undertaken in the APL helium tunnel to determine the effect of a shock duct on the pressure recovery available at high Mach numbers.

A schematic of the diffuser is shown in Fig. 3. Tests were made with diffusers having diffuser-to-test-section area ratios from 0.712 (the lowest value at which the tunnel would operate) to 1.00. The pitot pressure was measured at various radial positions at the nozzle exit and at the exit of the diffuser. Supply pressures ranged from 400 to 1000 psia, and the corresponding Mach number range at the nozzle exit was 16.4 to 17.8. Pitot pressures are measured by use of a tube with its orifice normal to the flow. In supersonic flow, the pressure measured is equivalent to the stag-
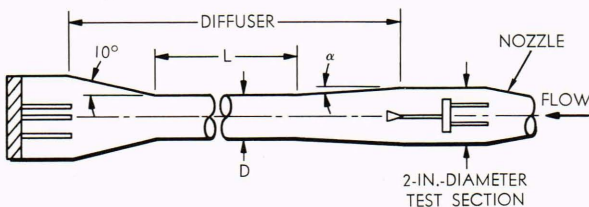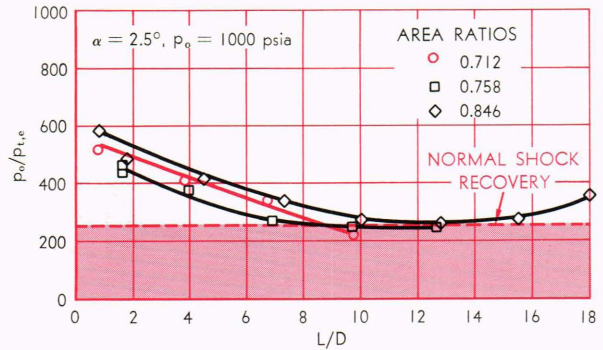

Fig. 4—Pressure recovery as a function of shock-duct length.

nation pressure behind a normal shock wave. In subsonic flow, the actual stagnation pressure is measured.

Measurements of the pitot pressure at the diffuser exit were made for various values of supply pressure p_o, duct length L, and duct diameter D. The data indicate that for $L/D < 9$, the flow in the diffuser is completely supersonic, whereas for $L/D > 9$, the shock compression is complete and a subsonic flow appears.

The effect of the diffuser duct length on the operating pressure ratio, $p_o/p_{t,e}$, where $p_{t,e}$ is a mean value of the measured pitot pressures at the diffuser exit, is shown in Fig. 4 for one of the values of diffuser inlet angle α and p_o studied. As L/D increases, the pressure ratio decreases to a minimum; this minimum value occurs for values of L/D between 11 and 15. Figure 5 presents the values of the minimum pressure ratio as a function of p_o, α, and area ratio. Also included in the figure is a curve representing the pressure recovery across a normal shock. These curves indicate that the optimum value of α occurs somewhere near 2.5° and that for the larger values of p_o, normal shock recovery was obtained. Also, it can be seen that the minimum value of the pressure ratio is only weakly dependent upon the area ratio and that the dependence is in the expected manner— better recovery at smaller area ratios. It should be remembered that the data shown in this figure are not for constant Mach number (see Fig. 2). Had the Mach number been constant during these tests, the increase in operating pressure ratio $(p_o/p_{t,e})_{min}$ with a decrease in supply pressure would be expected to be greater than that shown in Fig. 5.

In general, these tests show that a fixed-geometry shock-duct diffuser is capable of obtaining a pressure recovery equivalent to that available from a normal shock at the nominal test section Mach number, whereas the optimum value obtained by Johnson and Witcofski, using a variable-
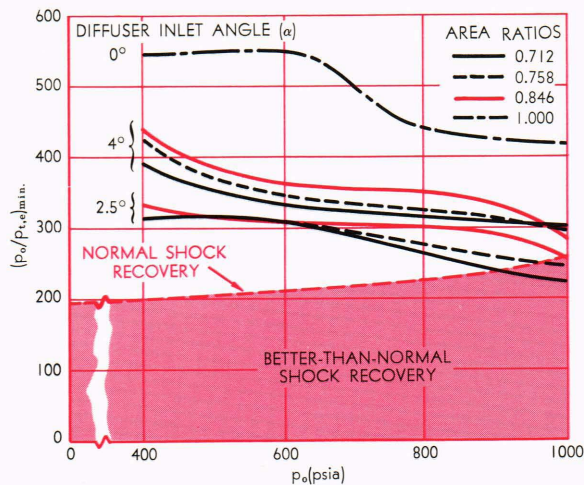

Fig. 3—Sketch of interior dimensions of the shock-duct diffuser, showing probe locations.

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[1] P. J. Johnston and R. D. Witcofski, "Effect of a Variable Geometry Diffuser on the Operating Characteristics of a Helium Tunnel Designed for a Mach Number in Excess of 20," *NASA* Report TN D-237, Feb. 1960.

Fig. 5—Effect of supply pressure on the minimum tunnel-pressure ratio, showing effect of diffuser area ratio and inlet angle.

geometry, convergent-divergent diffuser, is approximately 0.6 of normal shock recovery.

Incident Shock Wave—Laminar Boundary Layer Interaction

One of the most common phenomena occurring in fluid mechanics is separation of a flow from a surface. Although the causes and some of the effects of separation are relatively well understood, the study of this phenomenon is still largely empirical because of the many factors that enter into determining the separation point and the resultant downstream flow. It appears that just about every fluid mechanical parameter contributes to the problem, from the geometry and type of boundary layer flow to the usual factors such as Mach number, Reynolds number, and amount of heat transfer.

In subsonic flow, the occurrence of separation can produce radical changes in the entire flow field, which cannot be handled reasonably by the usual perturbation techniques. Fortunately, in supersonic flow, the problem is eased because the separation effect is localized. In addition, simple relations exist to account for the effect on the external flow of the thickening of the viscous layer. For these reasons the study of supersonic separation is more tractable, both experimentally and theoretically, and much effort has been expended along both these lines. Experimental research,[2,3]

[2] D. R. Chapman, D. M. Kuehn, and H. K. Larson, "Investigation of Separated Flows in Supersonic and Subsonic Streams with Emphasis on the Effect of Transition," *NASA* Report TN 3869, Mar. 1957.

[3] R. J. Hakkinen, I. Greber, L. Trilling, and S. S. Abarbanel, "The Interaction of an Oblique Shock Wave with a Laminar Boundary Layer," *NASA* Memo 2-18-59W, Mar. 1959.

which has generally been limited to Mach numbers lower than 4, has shown that the type of interaction that occurs between the viscous layer and the external stream is dependent, primarily, upon the condition of the boundary layer within this region, i.e., whether the boundary layer is laminar, turbulent, or transitional. A turbulent boundary layer has been found to be difficult to separate, and even then the separated region is of only limited length (of the order of 10 boundary layer thicknesses or less). On the other hand, a laminar layer separates quite readily, and the separated region can be extensive. If transition from laminar to turbulent flow occurs between separation and reattachment, the pressure distribution can be markedly affected.

Furthermore, these experiments have suggested the following flow model (Fig. 6) for the interaction of a shock wave with a laminar boundary layer. Because the subsonic portion of the boundary layer is unable to support a pressure discontinuity, it propagates the disturbance upstream and downstream of the shock wave. The resulting pressure increase distorts the velocity profile and thickens the boundary layer, resulting in a series of compression waves in the external flow. If the shock is sufficiently strong, separation will occur and the pressure will rise to a plateau region. Finally, the incident shock will be reflected as an expansion wave, the flow will turn into the wall and reattach to it, and the pressure will rise to its final, inviscid flow value. Experiments have also shown that the pressures at separation and at the plateau are dependent primarily upon the boundary layer characteristics of the undisturbed flow and are independent of the method used to produce the separation, if the length of the separated region $(x_{sh} - x_{sep})$ is sufficiently long. The data
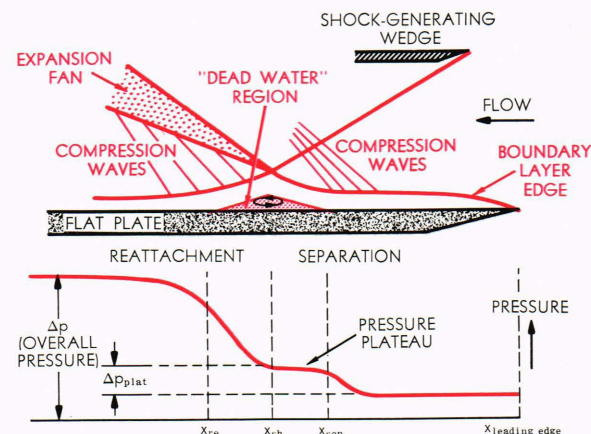

Fig. 6—Schematic of a flow model for an incident shock wave—laminar boundary layer interaction, showing pressure distributions.

of Hakkinen, *et al.*,[3] indicate that this length should be at least 0.20 x_{sh}.

Although pure laminar separation may be only of academic interest at the high Reynolds numbers and low supersonic Mach numbers of present-day flight, the advent of low-Reynolds-number—high-Mach-number vehicles can make this problem of practical importance. For this reason it was necessary to determine if the above flow model must be modified at the higher free-stream Mach numbers. Such a test, conducted in the helium tunnel at APL, used a series of flat plates having orifices distributed along the centerline from 0.7 to 3.2 in. downstream of the leading edge of the plate. The plates were essentially identical except for orifice location. An incident shock wave was generated by either a 5° or 15° wedge mounted on a movable sting above the flat plate. (Measurements have shown that the finite width had a negligible effect upon the two-dimensionality of the results.)

Data have been obtained at leading-edge Mach numbers from 17.1 to 17.8 and for the range of Reynolds numbers (3.3×10^5 to 12×10^5, based on free-stream values and shock impingement distance) to which the present experiment was limited. The generated shock waves have pressure ratios ranging from 6.7 to 55. The experiment covered separated and unseparated interactions. In Fig. 7 are shown the unseparated interactions where Δp is the pressure rise over the
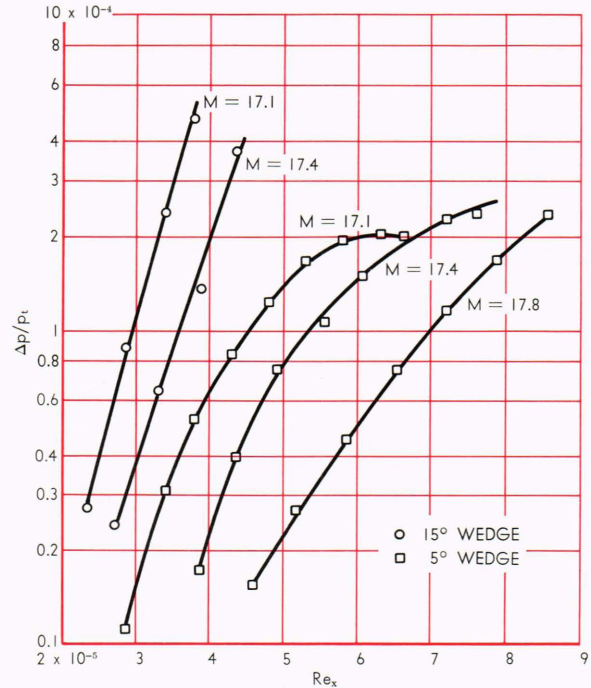

Fig. 7—Pressure increment in unseparated interactions.

non-interacting pressure, and p_t is the stagnation pressure. For the 15° wedge, the pressures are seen to increase exponentially as the shock impingement point is approached. The 5° wedge data are shown to become increasingly exponen-
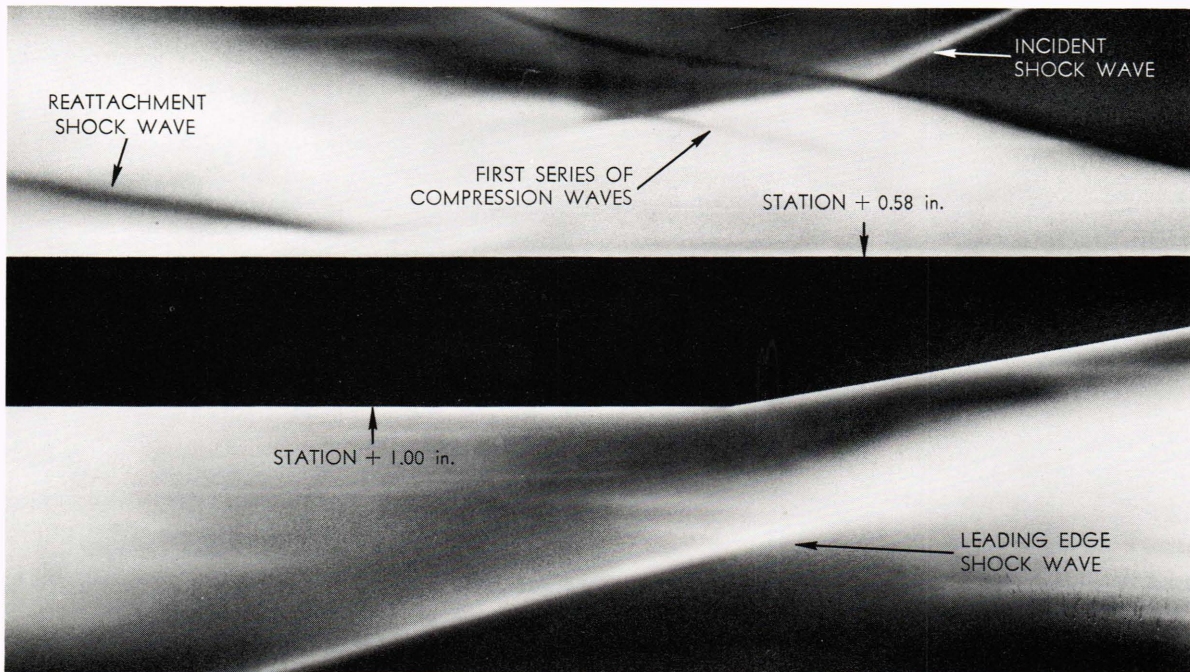

Fig. 8—Schlieren photograph of a separated interaction.

tial as the separation distribution is approached. This exponential distribution was predicted[4] for weak shock waves, but it is somewhat surprising for the present studies in view of the extremely strong shocks involved (pressure ratios up to 55). The fact that the weaker shock is not exponential creates some interesting speculation with regard to the interaction between the viscous layer and the entropy gradients that exist in the external stream.

Figure 8 is a schlieren photograph of a separating interaction. Since the background pressure is low, it is not possible to see the boundary layer development, but the shock pattern is quite well defined and closely resembles the flow model shown previously. A representative pressure distribution is shown in Fig. 9. This distribution has the characteristic pressure rise to a plateau and then the additional rise at reattachment. Again, this resembles the pressure distribution for the flow model except for the extremely large gradients at reattachment.

As a final check on the flow model, we can determine if the separation and plateau pressures are independent of the strength of the shock wave causing the separation. The theories based upon this assumption indicate that the plateau and separation-pressure coefficients are proportional to the square root of the unperturbed skin-friction coefficient C_f. Of these, the simplest to use is that
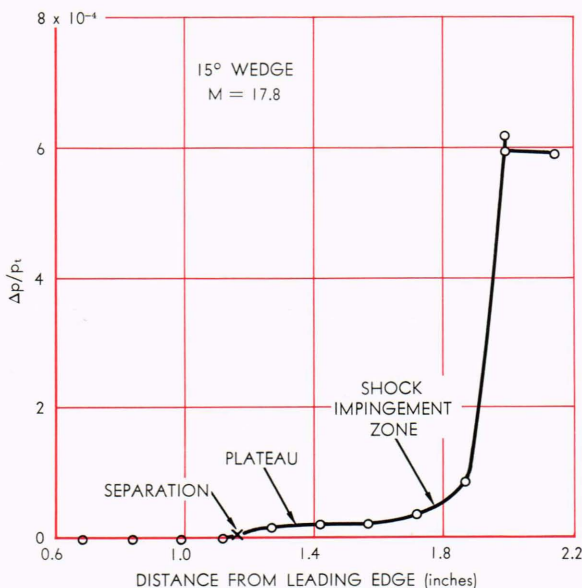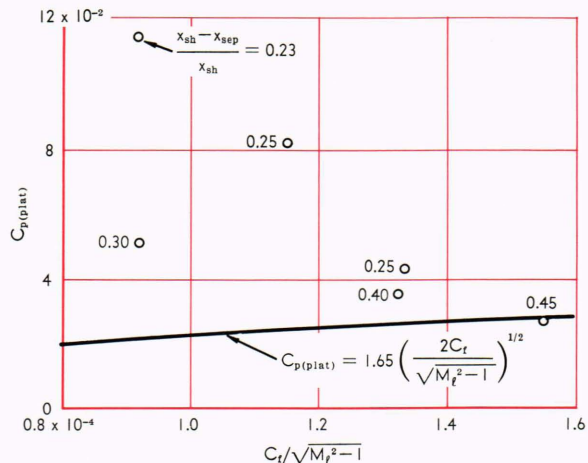

Fig. 10—Plateau pressure coefficient as a function of local skin friction coefficient (15° wedge).

of Hakkinen,[3] which yields the relation for the plateau pressure coefficient $C_{p(plat)}$

$$C_{p(plat)} = 1.65 \left(\frac{2C_f}{\sqrt{M_t^2 - 1}} \right)^{1/2},$$

where the coefficients are based on the local Mach number. Since the plateau pressure is the easiest to measure, it will be used to compare the present data with the above equation, as shown in Fig. 10. Beside each data point is the ratio of the length of the separated region to shock-impingement distance. As this ratio and C_f increase, the data approach the "theoretical" curve, suggesting that for the separation flow parameters to be independent of the shock wave, the ratio $(x_{sh}-x_{sep})/x_{sh} \geq 0.45$, as opposed to the 0.2 obtained by Hakkinen at Mach 2.

It appears, therefore, that the low-speed model of shock wave—laminar boundary layer interaction is applicable at hypersonic speeds, except that the length of the separated region must be greater than given by previous data. This also means that the region in which the separation parameters are dependent upon shock strength can no longer be neglected, and that considerable experimental work will be necessary in this region to define the important parameters and their effects.

The information obtained from the above study is indicative of the type of information that may be obtained in a helium hypersonic wind tunnel. Using the fluid mechanical models provided by these data, one can then proceed to estimate by theoretical or experimental methods the effects of the physical and chemical phenomena associated with air at the high velocity and temperatures of hypersonic flight.

Fig. 9—Pressure increment for interaction with separation.

[4] M. J. Lighthill, "On Boundary Layers and Upstream Influence. II. Supersonic Flows without Separation," *Proc. Roy. Soc.*, London, **217**, 1953, 478–507.