AN INVESTIGATION OF THE LAMINAR SEPARATION BUBBLE
IN THE TRANSONIC VELOCITY REGION

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LIST OF SYMBOLS

\( a \) = Speed of sound

\( C_p \) = Pressure coefficient; \( C_p = \frac{p_s - p}{p_0} \)

\( C_D \) = Screen drag coefficient

\( d \) = Diameter of rods comprising the screen

\( L \) = Length

\( M \) = Mach number, periodic spacing of turbulence producing screen

\( p \) = Static pressure

\( p_0 \) = Total head pressure

\( p_s \) = Local static pressure

\( R/e \) = Reynolds number; \( Re = \frac{vL}{\nu} \)

\( t \) = Time

\( U \) = Velocity of stream relative to screen

\( u \) = Local velocity in the boundary layer

\( V \) = Local free stream velocity

\( x \) = Distance along surface; distance from screen

\( y \) = Distance perpendicular to surface

\( \gamma \) = Ratio of the specific heats

\( \nu \) = Kinematic coefficient of viscosity

\( \Omega \) = Vorticity
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SUMMARY

This thesis was undertaken in an effort to determine the existence of the laminar separation bubble in a transonic flow field. The particular form of separation, referred to, was that imposed by forcing the boundary layer to turn a sharp corner. In this case a sharp pointed wedge was placed in the Georgia Institute of Technology, 2" x 4" supersonic wind tunnel at an angle of attack of 5.3°. The resulting stagnation point on the lower surface caused a separated flow to initiate at the point of the wedge.

The separation phenomena was investigated for characteristics observable by a schlieren system. Some investigations of the pressure distributions over the model were also conducted to determine the extent of separation.

The tunnel arrangement was such that the angle of attack of the model could be varied without moving the model itself. A plate was added to the exit of the test section which could block off any portion of the exit area desired. By blocking the exit area on one side of the model, the flow could be changed in such a manner as to cause a change in the angle of attack of the model. The angle of attack change produced in this manner could not be calculated or accurately measured; however, its direction could be ascertained.

It was found that the separation bubble did exist at the Mach numbers considered. At a free stream Mach number, M = 0.93, a lambda shock was found to exist on the upper surface of the model. Increas-
the angle of attack of the model increased the size and strength of the lambda shock. At a free stream Mach number, $M = 0.73$, and the same angle of attack, there was no lambda shock but a bubble did exist. An increase in the angle of attack at $M = 0.73$, caused complete separation of the flow from the surface.

The mechanism of reattachment of the boundary layer, in the case of subsonic flow, was found to be similar to that described by Bursnall and Loftin (1). It seemed that the flow within the separated boundary layer went through a transition from laminar to turbulent while separated. The turbulent boundary layer, formed after transition, spread much like a turbulent jet. Bursnall and Loftin suggested that only, if the spreading of the turbulent jet were sufficient to reach the surface, would there be reattachment of the boundary layer to the wall. It would seem from these experiments that this was the case.

Investigations into the Reynolds number of the subsonic separation bubble led to the conclusion, that this criterion for bubble size is valid to high subsonic Mach numbers.
CHAPTER I

INTRODUCTION

Explanation of the Problem. — In the field of transonic wind tunnels, much attention has been paid to an adequate means for the control of the flow. At transonic Mach numbers the area ratio is very critical. This fact, combined with the necessity for continuous curvature of the nozzle, make the design and construction of transonic wind tunnels an extremely difficult task.

Investigations of certain peculiarities of the lift curves of various airfoils in the vicinity of the stall have led investigators to examine a small region of separated flow near the leading edge of these airfoils. Particular attention has been paid to the laminar flow airfoils, where the stall is accompanied by a sudden decrease in the lift coefficient. Investigations have included an extensive examination of the separated boundary layer affecting this stall characteristic. Further investigations were conducted on separation at the leading edge of a double wedge airfoil at low subsonic speeds.

The characteristics of the separated region, examined in these investigations, led to the suggestion of their possible application as a means of controlling the flow in a transonic stream.

Separation. — The separation point on a surface is normally defined as the point at which the boundary layer profile at the wall becomes normal to the plate. The boundary layer profile is a plot of the velocity distribution in the boundary layer, along a line normal to the surface.
When the velocity at the surface is zero, and remains zero some distance away from the surface, the velocity profile will be normal to the wall. Further away the velocity will increase till it becomes equal to the free stream velocity. The slope of the boundary layer profile is given as \( \frac{\partial u}{\partial y} \), where \( u \) is the local velocity and \( y \) is the reference axis normal to the surface. At the surface, \( y = 0 \), and separation is defined by

\[
\left( \frac{\partial u}{\partial y} \right)_{y=0} = 0
\]  

(1)

Physically, the viscous effects of the fluid require the fluid adjacent to the body to have zero velocity relative to the body. However, the fluid away from the wall is moving with some kinetic energy; the inertia force represented by this kinetic energy is bucked by the viscous stresses. The pressure gradient along the surface will have more effect on the low velocity region near the surface than it will on the faster portion further out. With an increasing pressure in the direction of flow (adverse pressure gradient) the effect is to retard the flow. If this gradient has enough effect the flow will actually reverse direction near the wall, thus causing separation of the boundary layer from the surface.

When the \( \frac{\partial P}{\partial x} = 0 \) the flow will eventually become turbulent as the Reynolds number is increased. There is much more effective mixing of the fluid in a turbulent boundary layer. This has the effect of increasing the velocities near the wall over what they would be in the laminar case. This increased velocity near the wall permits the turbulent boundary layer to more successfully resist an adverse pressure gradient. Thus separation of the flow from the wall is greatly retarded.
Transonic Flow. — A basic parameter of any flow can be shown to be the Mach number. This number is defined as

\[ M = \frac{V}{a} \]  \hspace{1cm} (2)

where \( V \) = the local velocity
\( a \) = the local speed of sound.

The region where \( V < a \) (i.e., \( M < 1 \)) is defined as subsonic flow. In this region any pressure disturbances will not affect all parts of the flow field, since these disturbances move at a velocity equal to that of sound. The characteristics of the two flows are as entirely different as are the mathematical methods of attack.

There exists a flow region in the fluid, which has properties of both subsonic and supersonic flows. In this range there exists a mixed flow which is composed of both supersonic and subsonic velocities. It has been found that this region usually extends from the approximate limits, \( 0.8 \leq M \leq 1.2 \). In this paper, transonic flow will be defined to include this range of Mach number.

Lambda Shock. — A transonic flow over a body with a laminar boundary layer is usually characterized by the existence of a lambda shock. This shock derives its name from its similarity to the Greek letter \( \Lambda \).

The velocities over a body in a transonic stream will reach the sonic velocity at some point on the body. At this point a normal shock of infinitesimal strength is formed. As the Mach number is increased, the shock will become much stronger till it forms a normal shock with a considerable pressure rise through it. This normal shock forms the downstream leg of the lambda shock.
If the entire flow field were supersonic there could be no evidence of the pressure rise caused by the shock, ahead of the shock itself. However, the viscous effects of the fluid produce a boundary layer of which a portion is subsonic. The increase in pressure then has an opportunity to make itself felt ahead of the shock, by communicating through the subsonic portion of the boundary layer. This greatly increases the adverse pressure gradient ahead of the shock and causes the laminar boundary layer to separate. Such separation causes a change in the direction of the flow. In order for a supersonic flow to change its direction an infinitesimal amount, it must pass through a Mach line. Naturally to change its direction through a finite angle a succession of Mach lines must exist. It is this succession of Mach lines which coalesce to form the up-stream leg of the lambda shock.

The action of a turbulent boundary layer is still in doubt and few conclusions have been reached as to reasons for its behavior. It is known, however, that the turbulent boundary layer will produce only the normal shock mentioned above. It would seem that the adverse pressure gradient produced by the shock is not sufficient to cause separation. A possibly important consideration here is that the subsonic portion of the boundary layer is thinner than that encountered with a laminar boundary layer.

**Sharp Nosed Airfoil.**— The existence of a sharp corner deserves some comment in light of the preceding discussion. If a sharp nosed airfoil is placed at some angle of attack, there will be a stagnation point on
the lower surface. The flow in the boundary layer in this case would leave the stagnation point and turn around the sharp edge of the airfoil. If the edge radius were infinitesimal the velocities then become large but not infinite. In any case the viscous effects of the fluid would prohibit an infinite velocity.

The large velocities around the leading edge produce a very low pressure peak. The viscous forces quickly reduce the velocities to a considerably lower (although still high) velocity. The result is a correspondingly high adverse pressure gradient. The high adverse pressure gradient causes the boundary layer to separate at the leading edge.

Early Investigations. — Early investigations by von Doehoff (2) led to an approximate criterion for the extent of the separated region. His experiments showed that \( R_e = 70,000 \) was a characteristic of the separated region. He was dealing with very low Mach numbers and laminar flow along a flat plate in an adverse pressure gradient. He defined the separation bubble to exist over the distance from the point of laminar separation to the position of the first fully developed turbulent boundary layer profile. The Reynolds number was based on this distance, and on the local velocity, temperature, and pressure just outside the boundary layer at the point of separation.

In an investigation conducted by Gault (3) on a thin laminar flow airfoil at low Mach numbers it was discovered that the region occupied by the separation bubble was in a state of vortex motion. It had previously been considered that the laminar separation bubble was an area of stagnant air. He also showed that the separated boundary layer passes through a transition to turbulence while separated. It is this turbulent boundary
layer which returns to the surface.

Lamb (4) has shown that the equations of motion may be reduced to

$$\frac{D\Omega}{Dt} = \nu^2 \nabla^2 \Omega$$

where

- $\Omega$ = vorticity of a fluid element
- $t$ = time
- $\nu$ = kinematic viscosity coefficient.

This is the equation for the diffusion of vorticity. The vorticity formed at the interface of the separated boundary layer and the bubble will diffuse causing the interface to thicken and produce an angle of spread. When the flow changes from laminar to turbulent this angle of spread is greatly increased. If the spreading is sufficient to reach the surface the flow reattaches.

Bursnall and Loftin (5) conducted experiments which seem to bear out this conclusion. Their investigations with a hot wire anemometer showed the transition to turbulence in the separated boundary layer, and the spreading of the turbulent "jet".

Rose and Altman (6) followed a similar investigation, except that they were concerned with a sharp nosed, double wedge airfoil. The phenomena investigated by them was similar to that which is discussed in this paper.

For a perfect fluid (i.e., one which has no viscosity or compressibility) the velocity becomes zero when the flow is turned $90^\circ$. The same is true if the angle is less than $130^\circ$. If the angle is greater than $130^\circ$ the local velocity of the flow at the corner becomes infinite. Viscous
effects will, of course, modify these conditions. The action of a fluid in passing a corner greater than $180^\circ$ is similar to the case of a wedge at some angle of attack. When the wedge is at zero angle of attack the properties of the fluid are similar to the case of flow through an angle less than $180^\circ$.

Rose and Altman showed that small changes in the angle of attack produced large changes in the size of the separation bubble. They also found, by tuft, pressure, and smoke studies, that the laminar separation bubble was probably in a state of vortex motion. Their results also indicated that the bubble was much larger than that found on a round nosed airfoil.

The boundary layer has been shown to be of great importance in the transonic flow region. Ackeret, Feldman, and Rott (7) conducted an extensive investigation of the properties of the boundary layer in this region of flow. The boundary layer was formed over a surface, designed to simulate the surface of an airfoil in a transonic flow. They found that the flow phenomena depended on whether the boundary layer ahead of the shock is laminar or turbulent. The lambda shock was found to occur only in the case of the laminar boundary layer. They also found that the boundary layer was changed to turbulent immediately after the shock.

The analysis of Nitzberg and Crandall (8) followed the investigations of Ackeret, Feldmann, and Rott in an attempt to find some similarities between boundary layer flow at transonic and low speeds. Their principal effort was designed to make use of the vast amounts of boundary layer literature available in the low speed case as a means of analyzing
transonic flow flow. They found that the flow deceleration in the boundary layer preceding the lambda shock was the same as for low speed laminar separation. They also found a similar pressure distribution over the separation bubble as existed through a lambda shock. The literature currently available led to the conclusion that an examination of the laminar separation bubble would prove fruitful, at least to the extent of conclusively determining its existence in a transonic flow. In this paper separation on a wedge at an angle of attack is investigated, and some of its characteristics are determined. Some comparison is made of separation in the case of subsonic flow at high Mach numbers, and in the case of subsonic flow. Comparisons are then drawn on the basis of experimental results, and an analysis of previous investigations.
CHAPTER II

INSTRUMENTATION AND EQUIPMENT

The Wind Tunnel. — The basic piece of equipment used in these experiments was the Georgia Institute of Technology, two inch by four inch, two-dimensional, blow-down, supersonic wind tunnel.

The Pressure System. — Air was supplied to the reservoir by a ten horsepower "Worthington" air compressor, which was connected to the reservoir by approximately 150 feet of one inch pipe. The reservoir itself consisted of one 176 cubic foot tank and one 4.2 cubic foot tank. Air entered the small tank through a porous stone filter, which removed most of the oil introduced into the air by the compressor. The air then entered the large tank through a line which incorporated a safety valve.

Air passed from the large tank to a pressure regulator which was used to control the static pressure of the air entering the test section. This regulator was accurate to ± 2 psig, in the range from 10 to 100 psig.

Immediately following the regulator was a quick opening gate valve, so that the stored air might be fully utilized for the maximum amount of time. This valve permitted instantaneous stopping of the air supply when the desired data were recorded, so that it was not necessary to fill the reservoir entirely before running another test.

The Schlieren System. — This system has been used to provide a means of flow visualization. The light source was a 400 watt, General Electric BH-4, mercury vapor lamp. The power source was 208 volt, three phase, alterna-
tung current, which was passed through a rectifier to obtain 233 volt direct current for the lamp. The lamp was mounted horizontally with a parabolic reflector behind it to converge its rays for maximum intensity. This light passed through a Eastman Anastigmat 13.5 inch, f/3.5 camera lens to a narrow slit located at the focal point of the lens. This slit was one and one-half inches long, and mounted horizontally. It was designed to pass only the most intense and uniform portion of the light. At one focal length from the slit was a second lens similar to the one described above. This lens formed the light into parallel rays which went through the plexiglass test section walls. A third lens, similar to the first two was used to converge the light after it left the test section. The distances involved in the placement of these lenses, and the test section, is extremal critical. The second and third lens must be two focal lengths apart, while the test section should be at some distance greater than one focal length from the third lens (9).

Since it was desired to view the vertical density gradients, a knife edge was mounted horizontally at the focal point of the third lens. The height of this knife edge was adjusted so that maximum sensitivity was obtained. The light passing this knife edge focused on the focal plane of a "Graflex" camera without a lens. This camera was desirable since it provided both a means to observe the flow and take a picture when a steady flow condition was reached. Kodak Panchro-Press, type B film was used, at an exposure time of 1/30 second.

The Test Section.— For these tests, a two inch by four inch test section (Fig.1) was designed. This section was constructed of laminated wood and was provided with plexiglass walls in order to make use of the schlieren
system. All walls were parallel and no correction was made for the boundary layer, since the boundary layer (at these Mach numbers) is small and would not seriously affect the particular experiments being conducted. A two-dimensional model was mounted at the center line of the exit, parallel to the direction of the flow. Two screens were incorporated in the design to reduce the turbulence. One was located at the test section entrance, and a second was six inches downstream. Total and static head tubes were placed 3.6 inches downstream of the second screen. The screens were selected and located on the basis of two equations suggested by Batchelor and Townsend (10). The first was defined to determine the screen drag coefficient, which was given as

$$C_D = \frac{d}{M}(2 - \frac{d}{M})$$

The second equation gives the percent turbulence as a function of distance downstream of the screen.

$$\frac{U^2}{u^2} = 106 \left( \frac{x}{M^2} - \frac{x^2}{M^4} \right)$$

In this case, $x_0 = 0$, $M = 0.03$, $d = 0.002$. Based on the criterion of 0.5% turbulence at the model, $x$, was found to be 3.6 inches.

It was necessary in these tests to provide some means of altering the angle of attack of the model. This was accomplished by placing a plate over the exit of the test section. This plate covered half of the exit, so that no air would flow by the model on the side on which this plate was mounted; thus inducing the air to approach the leading edge of the model at some angle. The plate was fastened in such a way that the area it covered could be varied. Thus it was possible to change the in-
duced angle of the flow, or the angle of attack of the model.

The Model. — There were two models used in these experiments. One was a brass model (Fig. 2a), which was used for schlieren observation and photos. A second model made of plexiglass (Fig. 2b) was used to take the pressure distributions over the model. It was constructed of plexiglass because of the ease with which it could be machined. The use of a metal was prohibited for the pressure model since it was almost impossible to obtain small enough metal tubing so that its presence would not affect the flow. There was no practical way of fastening any plastic tubing to the metal.

Both models were three inches long and 3/16 inches thick, and completely spanned the test section. Both were tapered to a sharp point with the smallest angle possible, consistent with the need for alignment after installation and the desire to measure the pressure distributions over the model.

The pressure model was built by gluing strips of Polyethylene Medical tubing (0.034" I.D. x 0.050" O.D.) to the lower surface and drilling holes through the model and into the tubing. The ends of the tubes subject to a dynamic head were sealed with glue. The other ends were conducted outside of the tunnel to hyperdermic needles, which acted as pressure pickups for the manometer board.

The Manometer Board. — The manometer board used in these tests was a group of fourteen tubes thirty-three inches in length, connected to a common reservoir at the bottom. Connected to this "feeder" reservoir was a reservoir bottle containing the extra mercury and supplying a pressure balance with the atmosphere. Fluorescent tubes were mounted behind a glass plate in back of the mercury tubes. Tracing paper, graduated in
inches and tenths of inches, was placed on the glass plate so that mer-
cury column heights could be measured directly.

All readings of the manometer board were taken by a Kodak "Tourist"
camera, using Kodak plus X film. The exposure used was f/3.3 at 1/25 sec.
CHAPTER III

PROCEDURE

There were two general types of tests conducted. The first type made use of the brass model, and was designed primarily for visual flow observations. The other tests were conducted with the plastic pressure model to determine model pressure distribution and to obtain some correlation with the brass model tests.

A first set of preliminary runs were made to determine the manometer and pressure regulator characteristics. At the same time these tests served to provide general preliminary information so that the test program could be more directly channeled to an investigation of the bubble than was previously possible. These runs also served to examine the schlieren system and determine the best camera shutter speeds.

The first series of tests, included in the test program, were made at high subsonic Mach numbers. These tests included both pressure and schlieren investigations of the lambda shock encountered at these Mach numbers. In addition to the total and static head tubes up-stream of the model a small tapered probe was placed in the stream just ahead of the model. A total head tube was used to investigate the conditions in the immediate vicinity of the lambda shock.

The next series of tests were conducted to determine the free-stream Mach number at which the lambda shock disappeared. The static pressures at the pressure regulator were lowered in 5 p.s.i.g. increments till the shock disappeared. Runs were then made at increments of
2 p.s.i.g. to more closely determine the free stream Mach number at which the entire flow field became subsonic. The 2 p.s.i.g. increments were the smallest taken since they were considered the smallest possible within the accuracy of the gage.

A further series of tests were made to find the effect of a change of angle of attack. These tests were conducted under conditions which spanned the region where the lambda shock disappeared.

In general, the same data were taken in all cases. Schlieren photographs, Mach number and pressure distributions were taken at all times. Other specialized data was taken at certain times depending on the peculiar circumstances involved in the test being conducted. These specialized measurements included the total head measurements in the vicinity of the lambda shock, measurements of bubble length, and angle of spread of the separated boundary layer. The amount of exit area closed by the backplate was also noted whenever the back plate was used.
CHAPTER IV

ANALYSIS AND DISCUSSION

These experiments involved two distinct methods of analyzing the flow in the vicinity of the wedge. In the following paragraphs these two methods, schlieren photographs and pressure distributions, will be discussed separately.

Schlieren Photographs. — The schlieren photograph (Fig. 5) clearly shows the existence of the lambda shock. The free stream Mach number in this case was subsonic ($M = 0.93$). This value of the Mach number was found by the isentropic flow formula (12)

$$\frac{p}{p_0} = \left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma}{\gamma - 1}}$$

where $p$ = static pressure at the tunnel wall
$p_0$ = the stagnation pressure measured by the total head tube shown in Fig. 1.

The use of $\gamma = 1.4$ for air reduces this relation to

$$\frac{p}{p_0} = \left[1 + 1.2 M^2\right]^{-2.5}$$

The possibility existed that there was supersonic flow up-stream of the model (where the readings were taken) in spite of the readings. This would have been the condition if a detached shock existed in front of the total head tube. The pressure readings in that case would merely be measuring the subsonic Mach number just down-stream of the shock. In order to better determine the regime up-stream of the model, a tapered conical probe was placed in the field of view of the schlieren system just ahead of the model. It could be assumed that the free stream Mach
number given by the pressure measurements was valid, since no shock existed on the probe.

The schlieren system used in these experiments was of the horizontal slit type, which is sensitive to vertical density gradients. In (Fig.5) the grey tone up-stream of the model indicates a zero vertical gradient. The lighter areas indicate a negative gradient while the darker areas a positive. No absolute values of the density may be obtained by a schlieren system. The schlieren system is also marked by an axis of asymmetry which is defined by the knife edge and the slit for the light source. For these experiments the axis was approximately 0.04 inches above the upper surface of the model. The streaks parallel to the flow are believed to be streamlines but might possibly be involved with the boundary layer at the wall.

In the figure the model itself can be distinguished by a dark tapered area. It should be noted that the pictures are images of the actual mounting system as shown in (Fig.3). Just down-stream of the leading on the lower surface is a sharp line defined by a dark area below it and a light one above. The intersection of this line and the lower surface is believed to be the stagnation point. The dark area represents a density gradient which is increasing toward the model surface. This would be expected in the case of a stagnation point. The light area in turn represents a decreasing density gradient in the same direction. Again, for a stagnation point, the flow is away from the point and so the density is decreasing. Complete density plots of this type of phenomena may be found in a paper by Griffith (11). These density plots were determined by a Mach-Zehnder interferometer. This
equipment was not available for this problem.

Consider the model as a wedge with a wedge angle of 10.6°. Placing the upper surface parallel to the stream effectively places the model at an angle of attack of 5.3°. It is this effective angle of attack which produces the stagnation point on the lower surface.

The expansion of the flow away from the stagnation point is continued around the leading edge to a line almost normal to the model surface. On the upper surface, however, the asymmetrical axis has been crossed, and increasing density toward the model surface shows as a light area. The expansion around the leading edge now represents an area of decreasing density toward the surface, and still shows as a light area.

The dark area immediately following, in a counter-clock-wise direction, would have to represent a density gradient that decreases away from the model surface. The existence of a separation bubble at this point would demand that there be a Prandtl-Meyer expansion around it. It can be seen that this type of expansion would be expected to appear in just the manner observed.

The lambda shock would, in consideration of the foregoing analysis, be expected to show as a white area. The greatly increased density gradients toward the model surface make the shock appear much lighter than the adjacent areas.

As the boundary layer reattaches itself after separating at the leading edge, the flow is turned by the formation of Mach lines. These Mach lines coalesce to form the up-stream leg of the lambda shock. This leg intersects the main shock forming the full lambda shock.
Unfortunately some of the lines on the photographs are on the plastic walls of the test section itself. They can be discerned from the flow phenomena if (Fig. 4) is consulted. This photo was taken with no flow in the tunnel. Observation of the schlieren system without the test section showed a uniform light field to exist. Considerable efforts were expended, on the plexiglass, to polish it, and so realign the non-uniformity. Since this proved to no avail, it was assumed that the non-uniformity was a result of stresses in the plexiglass walls.

At lower Mach number (M = 0.73) (Figs. 9 and 10), the schlieren photos show no lambda shock, although there is separation over the leading edge as shown by a careful study of the original negative. Fig. 9 is a photograph of the model at M = 0.73 with the back plate open. The back plate being open infers an effective angle of attack of 5.3°. There is a stagnation point just below the leading edge. It should be noted that the stagnation point in the case of the higher velocity flows was somewhat further down-stream on the lower surface.

The boundary layer itself can be seen as the dark area (which represents a low density) leaving the model surface at the leading edge. The boundary layer in this case very nearly takes the shape found by Bursnall and Loftin (13). The boundary layer leaves the surface and thickens slightly. This thickening continues till the boundary layer reaches the point of maximum thickness of the separation bubble (the light region between the boundary layer and the model surface). After reaching this point, the boundary layer thickens considerably and becomes "fuzzy". The upper limit of the boundary layer appears to re-
main a continuous line. The lower limit appears to be smooth until the point of maximum bubble thickness is reached. Here the lower limit seems to change its direction, and approach the plate. The angle enclosed by the upper and lower surfaces of the boundary layer was found to be approximately $13.0^\circ$. Bursnall and Loftin gave no quantitative measurements of the angle of spread in their low speed investigations.

Figure 10 shows the same sort of phenomena at $M = 0.73$ and a higher angle of attack. The boundary layer leaves the surface until it reaches a point of maximum separation. The upper limit of the boundary layer remains continuous while the lower limit changes direction suddenly. The angle of "spread" in this case was approximately $13.0^\circ$. In this case the back plate was closed $1/4$ inches, thus producing a higher effective angle of attack.

Bursnall and Loftin (13) contended that the laminar boundary layer went through transition after separation and formed a turbulent boundary layer. This turbulent boundary layer then spread as if it were a turbulent jet. They contended that only, if the spreading of this jet were sufficient to approach the plate, would there be reattachment of the flow.

It would seem that these experiments bear out their contention. In the case where there was flow reattachment (Fig.9) it appeared that the turbulent spreading did reach the plate. In the case of complete separation, at a higher angle of attack (Fig.10), the spread of the turbulent "jet" was not sufficient to cause reattachment of the flow.

Separation Reynolds Number. — Von Doenhoff (14) found that the extent
of the laminar separation bubble could be expressed as a Reynolds num-
ber of 70,000. Gault, while conducting an investigation of the bubble
on an airfoil surface, discovered that the Reynolds number that charac-
terized the bubble was about 60,000. He also found that the Reynolds
number of the region of laminar separation was not a constant. The re-
gion of laminar separation is defined as that distance extending from
the point of laminar separation to the point of transition. He found
this Reynolds number to vary from 60,000 to approximately 30,000 near
complete laminar separation.

Reynolds number calculations based on the bubble length as shown
in (Fig. 9), and the density, viscosity and velocity at the separation
point, gave a Reynolds number of 68,000 for the full extent of the bubble.
For the region of laminar separation alone, \( R_e = 33,000 \) was found. Thus
the results found in this case would appear to be similar to those by
previous investigators at very low Mach numbers. It should be remem-
bered that the Reynolds numbers found from these experiments are the re-
sult of a single test, and so should be considered with caution.

Unfortunately it was difficult to obtain Mach numbers inside the
lambda shock, since the equipment available did not permit accurately
locating a total head tube in this region. Total head measurements ob-
tained here were unsteady and it was difficult to make any conclusive
interpretation. Because of this no conclusions could be reached concern-
ing the Reynolds number of separation in the case of shock flow.

Results from Pressure Distributions. — The pressure distributions with
a lambda shock proved to be very much unlike those considered by Ackert, Feldmann, and Rott (15). These investigators found an almost constant, though slightly increasing, pressure through the lambda shock. Nitzberg and Crandall (16) have described the pressure distribution over an airfoil in a subsonic flow, and have shown a constant pressure region, similar to those of Ackert, Feldmann and Rott, to exist over the laminar separation bubble.

Rose and Altman (17) conducted tests for laminar separation on a double wedged airfoil in a subsonic flow. Their results indicated a pressure peak at approximately the point of maximum separation thickness. The pressure then fell off sharply through the separation bubble, and became approximately constant after the bubble.

The pressure distributions found in these experiments correspond to the general form found by Rose and Altman. The pressure peak, as shown in (Fig. 11), was evident, as was the rapid pressure rise through the lambda shock. The almost constant, though slightly decreasing pressure following the shock also appears.

Thus it can be seen that the phenomena dealt with was essentially similar to the case of a wedged airfoil in a subsonic flow, rather than the case of a lambda shock normally encountered. It was felt that the results that were obtained by Ackert, Feldmann, and Rott were not applicable in this case, since they were dealing with shocks spontaneously formed by laminar separation. These experiments demanded the formation of the lambda shock by forcing separation at the point of a wedge.

Ackert, Feldmann, and Rott demonstrated that the lambda shock is a product of laminar separation. Their results show the similarity of
the shock phenomena to separation phenomena in subsonic flow. The re-
sults obtained in this experiment further carry the similarity to sub-
sonic flow. The comparison may easily be made by considering the results
obtained by Rose and Altman at low Mach numbers, and those herein ob-
tained.

**Angle of Attack.** — The position of the stagnation point should be ex-
pected to have some effect on the separation phenomena investigated.
The location of the stagnation point should, in turn, be a function of
the angle of attack. Rose and Altman (13) have shown that the effect of
angle of attack change is quite considerable in the subsonic case. They
found the bubble size to be dependent on angle of attack to the extent
that a one degree change could double the size of the bubble. They also
found that complete separation occurred at approximately seven degrees.

In these investigations the angle of attack was taken as 5.3 de-
grees. (Fig.5) shows the lambda shock at this angle, and (Fig.9) shows
the separation phenomena at subsonic speeds. (Figs.6 and 10), respec-
tively, show the same phenomena at some higher angle of attack. The
higher angle of attack was obtained by closing the back plate. This
would undoubtedly induce some higher angle of attack, but there was no
way of calculating or measuring this increase. It will be noted by com-
parison of (Figs.5 and 6) that increasing the angle of attack increased
the size of the lambda shock. The main shock was obviously stronger at
the higher angle of attack. Probably, the size of the bubble was in-
creased and a larger area ratio was produced. Thus the Mach number of
the shock was higher.

It will be noted, that from (Figs. 9 and 10), that the increased angle of attack increased the size of the separation bubble, in subsonic flow. It would seem that (Fig. 10) shows complete separation. The fact that this should occur with a small change of angle of attack is not inconsistent. The fact that these experiments were conducted at a higher Mach number (though on a larger wedge angle) than those conducted by Rose and Altman, would contribute to the existence of separation at a smaller angle of attack. It is impossible to reach any conclusions regarding this variation with Mach number since the angle of attack in this experiment is not known.
CHAPTER V

CONCLUSIONS

1. High subsonic free stream Mach numbers produced supersonic flow over the model. This supersonic flow was evident as a lambda shock.

2. Separation exists over the leading edge of a sharp wedge at a moderate angle of attack of 5.3°. At angles of attack slightly over 5.3°, the flow reattached itself after separation in the case of supersonic flow. The boundary layer did not return to the surface under the same conditions in a subsonic flow.

3. The mechanism of reattachment in subsonic flow is most probably dependent on the spreading of the turbulent boundary layer after transition.

4. A similarity between transonic and subsonic flows exists in that the pressure distributions through a lambda shock compares with that of a subsonic separation bubble.

5. A $Re = 60,000$, based on the total length of the bubble was found. This bears out a basic characteristic of the separation bubble first mentioned by von Doenhoff. The fact that this Reynolds number was calculated at $M = 0.73$ would indicate that this characteristic of laminar separation is independent of Mach number, up to the critical Mach number.

6. A $Re = 33,000$, based on the length of the laminar portion of the separated boundary layer, was found. This would tend to bear out the low Mach number investigations of Gault (19), thus extending the flow similarity further.
CHAPTER VI

RECOMMENDATIONS

On the basis of the results found in this paper, some further investigations should prove fruitful. Some recommendations are given which might aid future investigators of the separation phenomena discussed here.

1. The present wind tunnel should be altered to include a larger reservoir and a surge tank. These alterations would eliminate many of the difficulties presently encountered, and would permit more accurate pressure measurements. Such accuracy is necessary for future quantitative analysis.

2. The present schlieren system should be more rigidly mounted, while certain components (such as the knife edge) should be made more easily adjustable. Such an arrangement would permit more sensitive adjustments, and would allow the investigator to examine both the vertical and horizontal density gradients.

3. Provision should be made to use optical glass for the test section walls instead of the present plexiglass. Stresses in the plexiglass produced a non-uniform light field.

4. An extension of this investigation should be carried out, using a double wedge with a smaller included angle than was used in this investigation. This would permit more accurate angle of attack measurements.

5. Total head measurements of the boundary layer should be con-
ducted so that some of the phenomena discussed in this paper might be more fully investigated.

6. The Mach number in the vicinity of the lambda shock should be investigated thoroughly. Such an investigation would permit the determination of the Reynolds number of the separation phenomena involved in the lambda shock. This type of investigation would lead to more information on transonic and subsonic flow similarities.
APPENDIX
FIGURE 1
TEST SECTION
MOUNTING HOLES

FIGURE 2A. BRASS MODEL

MOUNTING HOLES

FIGURE 2B. PLASTIC PRESSURE MODEL
Figure 3
General View of Wind Tunnel

Figure 4
Schlieren Photograph; No Flow
Figure 5
Schlieren Photograph; \( M = 0.93 \)

Figure 6
Schlieren Photograph; Increased Angle of Attack
Figure 7
Schlieren Photograph; $M = 0.78$

Figure 8
Schlieren Photograph; $M = 0.78$ (Plastic Model)
Figure 9
Schlieren Photograph; $M = 0.73$

Figure 10
Schlieren Photograph; Increased Angle of Attack
Figure 11. Pressure Distribution
BIBLIOGRAPHY


15. Ackeret, Feldmann, and Rott, op. cit. Fig.5.


17. Rose and Altman, op. cit. p.16.
