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EFFECTS OF INITIAL TURBULENT BOUNDARY LAYER ON SHOCK-INDUCED SEPARATION IN TRANSONIC FLOW

by

B.H. Little, Jr.

RHODE-SAINT-GENESE, BELGIUM

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FOREWORD

The work described herein was done by Mr. B.H. LITTLE Jr. under the supervision of Professor J. GINOUX, in partial fulfillment of the requirements for receiving the Diploma of the von Karman Institute for Fluid Dynamics. Mr. Little, an American student, obtained a grade of Distinction for the academic year 1966-1967.
ABSTRACT

The effect of varying initial turbulent boundary layer conditions on shock-induced separation at transonic speeds was investigated experimentally. The experiments were performed in a solid wall axisymmetric nozzle at a unit Reynolds number of $0.2 \times 10^6$/cm. Initial turbulent boundary layer conditions were found to have a significant influence on the extent of separation within a narrow Mach number range. The results fit logically into the framework of earlier work and afford a better understanding of the basic interaction phenomena.
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SYMBOLES

C    -    airfoil chord
Cₚ    -    pressure coefficient,
L    -    length of boundary layer development pipe
M    -    Mach number
P    -    pressure
q    -    dynamic pressure
Re    -    Reynolds number
u    -    velocity
x    -    distance in streamwise direction
y    -    distance normal to surface
α    -    angle of attack
δ    -    y at \( \frac{u}{u_e} = .995 \), boundary layer thickness
δ*    -    boundary layer displacement thickness
θ    -    boundary layer momentum thickness

Subscripts

1    -    conditions at start of interaction
2    -    conditions just behind shock
∞    -    freestream condition
s    -    separation point
K    -    kink in pressure distribution
e    -    edge of the boundary layer
SUMMARY

Tests were performed in an axisymmetric solid wall nozzle to determine the effects of changing initial turbulent boundary layer conditions on normal-shock-induced flow separation. Initial boundary layer conditions were changed by varying the length of a constant diameter section of approach pipe upstream of the nozzle. This configuration change resulted primarily in a thickening of the initial boundary layer with little change in profile shape. Unit Reynolds numbers were approximately $0.2 \times 10^6$ per centimeter. All of the initial boundary layers were turbulent and no artificial tripping devices were used.

In the nozzle the expansion near the exit was great enough that separation existed in that region even at subsonic Mach numbers. When separation first occurred at the shock (some distance upstream of the exit separation) it was confined to a relatively small region near the shock, and some attached flow existed between this bubble type separation and the downstream separation. At higher Mach numbers, with stronger shocks, separation induced by the shock extended throughout the length of the nozzle.

It was found that the Mach number for significant shock-induced separation (separation starting at the shock and merging with the downstream separated flow) was that Mach number at which insufficient compression was obtained through the shock at the edge of the boundary layer to obtain a wholly subsonic flow behind the shock. This is in agreement with results from two-dimensional airfoil investigations presented in Reference 1.
Finally it was found that the Mach number for significant shock-induced separation was reduced by thickening the initial boundary layer. This reduction resulted from two causes. Thickening the initial boundary layer reduced the static pressure rise to separation. It also reduced the rate of shock compression by spreading it over a longer distance. This latter effect, though small, would tend to diminish total pressure losses through the inviscid region at the edge of the boundary layer. Both actions combined to yield higher Mach numbers behind the shock at the edge of the boundary layer when the initial boundary layer was thickened. This, in turn, lowered the initial Mach number at which supersonic flow was first obtained behind the shock.
I. INTRODUCTION

In Reference 2 a description is given of problems encountered in scaling transonic wind tunnel data to full scale flight conditions. The tests reported in Reference 2 were concerned with the determination of forces and moments on wing sections in transonic flows. In low Reynolds number wind tunnel tests, boundary layer transition was fixed near the leading edge of the wing model. The flight tests were conducted at Reynolds numbers an order of magnitude higher and the boundary layer was naturally turbulent over most of the wing section chord. Comparisons of data from the two sets of tests revealed some rather serious discrepancies in wing loads. These discrepancies were traced to differences in the location of shock waves on the upper surface of the wing. These differences, as shown in Figure 1, were sufficiently great to cause the measured discrepancies in wing load data.

In subsequent wind tunnel tests, transition was fixed at several locations on the wing surface. In every case, it was stated, transition was complete upstream of the shock interaction region. The resulting data showed the location of the shock interaction region to be quite sensitive to the location of the transition strip. The most obvious factor controlled by location of the transition trip was the thickness of the boundary layer upstream of the interaction region. In general however this sensitivity of shock location to boundary layer trip existed only at positive angles of attack, and the degree of sensitivity increased with increasing angle of attack. Thus the sensitivity appeared to be a function of the severity of the pressure gradient to which the boundary layer was exposed downstream of
the shock interaction region. This implies that the interaction was com-
plicated by the existence of separation or incipient separation downstream
of the shock location.

The author of Reference 2 concluded that the problem was one of Reynolds
number effect on the shock-induced separation - associated with differences
between the relative thickness of the boundary layer on models and full-
scale airplanes.

Most of the research conducted prior to that of Reference 2 seems to indi-
cate that shock-induced separation in the transonic range is relatively
insensitive to boundary layer thickness or to unit Reynolds number. Refer-
ence 3 presents results from several different investigations and a number
of different models which show, for the pressure rise to incipient separa-
tion and for the peak pressure rise of shock interactions, practically no
sensitivity to Reynolds number in the low supersonic Mach number range.

On the specific subject of scaling effects in transonic wing flows, the
most complete and authoritative study is the work of Haines, Holder, and
Pearcey in Reference 4. In that reference a large quantity of wind tunnel
and flight test data was analyzed. One of the conclusions reached was
that generally accurate representation of full scale two-dimensional flows
can be obtained so long as care is taken to insure that the correct state
(laminar or turbulent) of the boundary layer is duplicated in the region
immediately upstream of the shock wave. This conclusion was qualified
slightly by a brief statement to the effect that boundary layer thickness
may be important for cases where the boundary layer must sustain not only
the shock interaction pressure rise but also a strong pressure rise down-
stream of the interaction region. The authors of Reference 4 did not elaborate on this point.

There has been at least one investigation (Reference 5) which showed an initial boundary layer effect on the extent of separated flow at Mach 2.9. There are however enough differences between transonic and supersonic shock-boundary layer interactions so that the application of those results to the present problem is obscure.

While no investigations were found in the literature that applied directly to the problem of initial boundary layer effect, there were several which were of considerable help in understanding the fundamental flow mechanisms. Foremost among these was the work of Pearcey in Reference 1 which describes in considerable detail the development of shock induced separation on two-dimensional airfoils in transonic flow.

The purpose of this investigation was to provide a study of the effects of varying initial turbulent boundary layer on the interaction of this boundary layer with a normal shock wave in a pressure field similar to that of an airfoil flow. An axisymmetric nozzle was chosen as the test configuration in order to eliminate, insofar as possible, any extraneous effects from side wall boundary layers. Normal shock waves of different strengths (Mach numbers from 1.1 to 1.5) were imposed on the turbulent boundary layers, and the initial boundary layer was changed by using different lengths of constant diameter tube ahead of the test nozzle. The unit Reynolds number was approximately $0.2 \times 10^6$/centimeter.
II. APPARATUS AND METHODS

The apparatus used in these tests is shown in Figure 2. Sections of constant diameter tube were used to change the length ahead of the test nozzle in which the initial boundary layer developed. In the discussion of results the configuration is identified by the length, in millimeters, of the boundary layer development tube. Thus the \( L = 150 \) configuration was that arrangement in which a section of 150mm length was installed between the inlet bell-mouth and the test nozzle.

The test nozzle is shown in Figure 3. It has an entrance diameter of 52mm, a short section in which the diameter is constant at this value, and then a gradual expansion along walls which turn at constant radius. The exit-to-inlet area ratio of the nozzle is almost 3.9 to 1, and the included angle of an equivalent conical geometry would be about 12 degrees. This configuration was deliberately designed to provide a region of severe adverse pressure gradients downstream of the shock location since it was in this type of flow that the difficulties described in Reference 2 were experienced.

Initial boundary layer thickness was measured at one station in the test nozzle - the station at the end of the constant area portion of the test nozzle - or the station at which the test nozzle walls first began to diverge. The boundary layer probe is shown in Figure 4. With this probe it was possible to measure pressures as close as 0.1mm from the surface. During the tests an electrical circuit signalled the point at which the probe first left the wall and the point at which the probe first touched the wall as it was traversed back.
The independent variable, shock strength, or Mach number immediately upstream of the shock, was varied by controlling pressure ratio across the test system.

Static pressures along the wall of the test nozzle were measured generally at axial spacings of 4mm. These orifices were arranged in three rows which were staggered 22.5 degrees apart. Additional statics were later added to reduce the axial spacing to 2mm in the region near the shock location. The orifice diameter was 0.5mm. All pressures were recorded on chart type recorders through a transducer-Scanivalve hookup.

Flow visualization was obtained using the method described in Reference 6. Lampblack was added to a medium grade oil to form a mixture which was applied to the surface of the test nozzle in minute dots. For a given mixture the size of the dots determined the length of time required to completely streak the dot. By trial and error it was found that the best results were generally obtained when the size of the dot was large enough that some of the oil was still slightly moist at the end of the run. During the runs these dots of oil flowed in response to local wall shear forces—leaving a streak which showed the direction of flow adjacent to the wall.
III. RESULTS

Boundary Layer Measurements

Boundary layer velocity profiles at the nozzle inlet are plotted in Figure 5. It can be seen that the profiles have characteristic turbulent boundary layer shapes - even for the L = 0 configuration. This is also shown in Figure 6, where curves fitting the profiles are plotted on logarithmic scales. The curve for L = 0 has a slope which very closely matches that for 1/7th power profiles. A line for a 1/9th power slope falls between lines for the L = 150 and L = 450 profiles. The profiles thus became increasingly full as the approach pipe was lengthened. This effect is also shown in Figure 7 where δ* , θ , and δ*/θ are plotted as functions of approach pipe length. The shape parameter δ*/θ decreases from about 1.36 at L = 0 to 1.21 at L = 450.

Since the boundary layer profiles were measured at a fixed station in the nozzle - the point where the diverging nozzle walls were tangent to the approach pipe - there is some pressure gradient effect in these data. For the thinnest initial boundary layer, L = 0, the survey station was quite close to the sonic point in the flow. The ratio of static-to-total pressure, P/P_t, was 0.519. For L = 150, P/P_t was 0.504; and for L = 450, P/P_t was 0.490. Thus the survey station was farther downstream of the sonic point for successively thicker initial boundary layers, or, in other words, the effective throat in the flow moved upstream with increasing boundary layer thickness. If the data of Figure 7 were corrected for this effect, the shape parameter values for L = 150 and L = 450 would be increased, and so
would be the values for $\hat{\delta}$ and $\theta$.

It has not been shown in Figure 7, but there was some asymmetry in the inlet flow. Boundary layer profiles measured along the top of the nozzle were about 10 percent thicker than those along the bottom. This asymmetry was aligned vertically and apparently resulted from the way that the flow was brought into the plenum chamber from the bottom (Figure 2). This was ascertained by rotating all components of the test configuration. The thickest boundary layer was always along the top of the nozzle regardless of the position of any component.

Since this "built-in" flow asymmetry was later found to fix the alignment of separation regions in the nozzle, it was recognized to have some beneficial effects, and no attempts were made to correct it. All measurements of boundary layer are those made along the top wall.

Flow visualization

Some typical results of the flow visualization data are shown in Figures 8-10. The direction of flow near the surface can be determined from the shape of the oil dots. Generally the head, or thickest portion of the dot, and tail, thinnest portion, are reasonably well defined, and the direction of flow is from head to tail. In those cases where there is not well-defined tail, the flow was apparently at low velocity and fluctuating in direction. There are some regions of obviously reversed flow and there are others in which the exact direction of flow is difficult to define. Generally it was assumed, for the purposes of data interpretation, that flow was separated from those regions of the surface on which there were no clearcut downstream velocity components.
Figures 8-10 show quite clearly the strongly three-dimensional nature of the nozzle flow. Separation in such a nozzle is obviously characterized by complex primary and secondary vortex flows. And, as is evidenced by these photographs, the flow is strongly asymmetric. The flow in such a system requires separation in just one sector to relieve the harsh adverse pressure gradients imposed by the wall geometry. And when separation thus relieves these pressure gradients, the flow in other sectors can remain attached to the wall.

Generally, one might expect difficulty in predicting which portions of the nozzle flow would separate if the geometry and the inlet flow were perfectly symmetric. Separation in such a perfectly symmetric inlet system would depend on a scratch in the surface, or the existence of dirt particles, or some other such random factor.

In these tests, however, asymmetric separation always occurred in line with that region where the initial boundary layer was thickest - along the top of the nozzle. This can be seen by noting the numerals on the nozzle surface in Figures 8-10. These numerals are arranged in clock-face fashion - looking upstream into the nozzle. The numeral 12 indicates the top, 6 indicates the bottom, and so on. Thus a slight asymmetry of the inlet flow fixed the asymmetry of the separation pattern.

Since the nature of this investigation was not strictly quantitative, but rather more exploratory - seeking to determine if a significant effect of initial boundary layer existed, and, if so, how that effect occurred, it
was decided that the asymmetry of the initial flow was more helpful than harmful. The region of separation was fixed with respect to time and was independent of random surface disturbances. And one could still determine the effects of changing initial boundary layer thickness on the overall nozzle flow distribution.

It can be seen from examination of the photographs of Figures 8-10, that several different patterns of separated flow existed. These are classified according to the sketches of Figure 11. Even before shock-induced separation occurred, there was a region of separated flow near the downstream end of the nozzle. This can be seen in Figure 8 at $M = 1.18$ and is depicted in Figure 11(a). At higher Mach numbers there existed, in addition to this downstream region of separation, a small region of separation at the shock. Between these two regions of separated flow there existed, however, a region of attached flow. This can be seen in Figure 9(b) and is depicted in Figure 11(b). At slightly higher Mach numbers, it is likely that there existed larger separation bubbles at the shock followed by reattached flow. This, however, is difficult to discern from the photographs and the next clearly defined pattern is as shown in Figure 11(c) where flow separated at the shock and, on one side at least, did not reattach in the nozzle. Examples of this type can be seen at the highest Mach numbers in Figures 8-10.
Static Pressure Measurements

Distributions of wall static pressure in the nozzle are shown in Figures 12-14 for the three different approach lengths. These data show the same general pattern for all three configurations. Flow expanded to low pressures in the upstream region of the nozzle, underwent a rapid but finite rate of compression through the shock (as opposed to the ideal discontinuous shock compression), and then a lower rate of compression behind the shock as the flow progressed through the diverging-wall channel.

In Figures 12-14, one can see some places where the inordinate flattening of a curve in a particular region might be interpreted as an indication of separation. But, in general, this is a difficult and somewhat uncertain technique. Only when these curves were combined with the flow visualization data was it possible to draw firm conclusions about the extent of separation.
IV. DISCUSSION OF RESULTS

Significant Separation

Pearcey, in Reference 1, introduced a definition for significant separation on airfoil models. This concept arose from observations that shock-induced separation could occur without causing any serious disturbance to the overall flow as long as the extent of separation was not large. More specifically he noted that separation did not exert a dominating influence on the flow until the separation bubble began to "expand rapidly toward the trailing edge and beyond." In later work he refers to the Mach number at which this rapid bubble expansion starts as the "Mach number for significant separation."

The pattern of flow development depicted in Figure 11 is basically the same as that which Pearcey observed on airfoils and reported in Reference 1. This similarity suggests that the "Mach number for significant separation" would be a good parameter to examine in searching for effects of the initial boundary layer. Attention was therefore concentrated on the Mach number at which transition from small bubble type separated (Figure 11(b)) to large bubble separation (Figure 11(c)) occurred.

Pearcey made extensive use of wall static pressure measurements in his studies of shock-induced separation. From these he was able to deduce a rather interesting result. He found that rapid separation bubble expansion started when the shock failed to reestablish subsonic flow downstream of the shock. In order to understand how this can happen, one must visualize a flow model such as that shown in Figure 15. In this flow model the normal shock does not extend to the boundary layer, but because the boundary layer
can anticipate the shock, there is a region of flow adjacent to the boundary layer in which several weak shocks may exist. In this region the full normal shock total pressure loss will not be experienced — in fact, an isentropic compression may be approached. Seddon in Reference 7 made detailed flow measurements in such a shock interaction region. He showed a static pressure gradient in the direction normal to the wall and very clearly demonstrated the possible existence of this "supersonic tongue" of flow behind the shock wave.

Pearcey then established a strong correlation between the existence of a supersonic tongue and rapid separation bubble growth. Or, in terms of wall static pressures, when the pressure $P_2$ behind the shock was such that $P_2/P_{t_1} < 0.528$, he found rapid bubble growth. He further conjectured a physical explanation for this growth as follows. The supersonic tongue, when it exists, exists in a region of rising static pressures in the downstream direction. Thus the streamlines in the supersonic tongue will be contracting. The contraction, in turn, will tend to retard any tendency of the separated flow to turn back towards the wall. A subsonic flow on the other hand will have expanding streamlines and tend to aid reattachment.

**Correlation of Pressure and Flow Visualization Data**

The pressure distribution curves of Figures 12-14 bear strong resemblance to pressure curves for airfoils and one may identify a $P_2$ on these curves. This pressure $P_2$ is generally called the kink pressure in the work of Pearcey and other airfoil researchers. It is the point on the pressure curve where the slope changes from that value through the shock to a lesser
value immediately behind the shock. Portions of the pressure distributions from Figures 12 and 14 have been expanded and reproduced in Figure 16 to show this kink pressure (or $P_2$) more clearly.

In the literature a relationship between kink pressure and separation pressure has generally been observed. Some authors make a distinction between the two and some do not. Using the flow visualization data to establish the location of the separation line and the kink pressures from the data of Figures 12-14, an attempt was made to establish the degree of correlation between these points. The results are shown in Figure 17. It can be seen that there was a high degree of correlation between the location of the kink pressure and the location of the separation line. If there existed a distinction between kink pressure and separation pressure, the accuracy of these data did not permit its identification.

Effect of Initial Boundary Layer on Kink Pressure ($P_2/P_{t_1}$)

Having established that the kink pressure or $P_2$ is a reasonably good indicator of separation, an attempt was made to determine the effects, if any, of the initial boundary layer on $P_2$. The results are shown in Figure 18 where $P_2/P_{t_1}$ is plotted against $P_1/P_{t_1}$. The value for $P_1/P_{t_1}$ was read as the minimum value upstream of the interaction. It is, of course, a measure of the Mach number just upstream of the shock. The ratio $P_2/P_{t_1}$ is reasonably close to $P_2/P_{t_2}$, since the normal shock losses at $M = 1.3$, for instance, are only about 2%. Furthermore, the flow at the edge of the boundary layer does not experience the full normal shock loss, so $P_2/P_{t_1}$ can be interpreted
as a reasonable indicator of $M_2$ — the Mach number behind the shock at the edge of the boundary layer.

The theoretical curve for normal shock recovery in Figure 18 shows that one would expect $P_2/P_{t_1}$ to increase with decreasing $P_1/P_{t_1}$ in an ideal flow. The data appear to follow this trend only for values of $P_1/P_{t_1}$ greater than about 0.45. The theoretical pressure recovery curve is intersected at this point by another curve representing the separation pressure rise. This curve (from Reference 8) is empirically based so that it is not surprising that the data of this investigation follow the general trend of this curve quite well. An initial boundary layer effect can be seen however, because the data for $L = 0$, the thinnest boundary layer configuration, are reasonably close to the separation pressure rise curve, while those values of $P_2/P_{t_1}$ for the thicker boundary layer configurations fall somewhat lower.

Curves are faired through these data points only in the vicinity of $P_2/P_{t_1} = 0.528$ in order to emphasize the boundary layer effect in this region. This, of course, is the region which Pearcey noted as critical insofar as rapid separation bubble growth is concerned. If these data confirmed Pearcey's observation, one would expect small bubble separation when $P_2/P_{t_1} > 0.528$ and much larger regions of separation when $P_2/P_{t_1} < 0.528$. This was indeed borne out by the flow visualization data.

To illustrate the criticality of $P_2/P_{t_1}$ three points in the vicinity of $P_2/P_{t_1}$ were isolated for closer study. They are the points labeled A, B, and C in Figure 18. Pressure distribution curves for these points are
shown in Figure 19. Points A and B, for the L = 0 and L = 150 configurations respectively have values of $\frac{P_2}{P_{t_1}}$ almost exactly equal to 0.528. Point C has the value $\frac{P_{2}}{P_{t_1}} = 0.520$. The pressure distributions in Figure 19 show a much reduced slope for curve C in the region immediately behind the shock - indicating that separation for point C was more severe than for A or B.

Flow visualization data were also obtained for points A, B, and C. These data showed regions of attached flow behind the shock for A and B, but none for C.

Thus there is strong evidence in Figures 18 and 19 that:

1. thickening the initial boundary layer at a given initial Mach number reduces the compression through the interaction region, so that one may change the flow behind the shock from wholly subsonic to partially supersonic, and

2. the extent of separation behind the shock is strongly dependent on the compression through the shock when $\frac{P_{2}}{P_{t_1}}$ is about 0.528.

These data then lend strong support to the criterion for significant shock-induced separation set forth by Pearcey. They also confirm the conjecture of Reference 2 that boundary layer thickness can play a significant role in determining the flow pattern around airfoils in transonic flows. Furthermore they provide a basis for an understanding of how initial boundary layer can produce this effect.

It is significant at this point to note that the data from Reference 2 which are reproduced in Figure 1 are in good agreement with the conclu-
sions of this report. In Figure 1 there is a relatively well defined kink in the curve from the wind tunnel tests. That curve from the flight tests shows no such well defined kink, but if a $P_2$ value were selected it would be in the same region as $P_2$ from the wind tunnel tests. This region would be that at which $C_p$ is about -.30 to -.35. The critical value of $C_p$ at the Mach number of Figure 1 is 0.31. So the $P_2/P_{t_1}$ values at which trouble occurred in Reference 2 are in the vicinity where $P_2/P_{t_1} \approx 0.528$.

The Nature of the Initial Boundary Layer Effect

In order to examine more closely the nature of the influence of initial boundary layer on the interaction region, let us examine again the curves of Figure 16. These curves show pressure distributions in the range near $P_2/P_{t_1} = 0.528$ for the thinnest ($L = 0$) and thickest ($L = 450$) initial boundary layers. Two features may be noted: (1) the rate of pressure rise in the shock region is diminished by thickening the initial boundary layer; and (2) the pressure rise to separation as indicated by $P_2$ is diminished by thickening the boundary layer. The lower rate of pressure rise implies a trend away from normal shock discontinuity to multiple shock compression in which total pressure losses are diminished. A given pressure rise at lower rate would then result in a slightly higher Mach number. But perhaps more significant is the fact that the static pressure rise to separation is actually diminished by thickening the boundary layer. At any rate it can be seen that the combined effect of initial boundary layer thickening is to lower $P_2/P_{t_1}$. 
V. CONCLUSIONS

Thickening the turbulent boundary layer upstream of a normal shock in a transonic flow reduced both the rate at which static pressure increased through the interaction and the final pressure attained at separation. Thus, for a given initial Mach number, thickening the boundary layer increased the Mach number behind the shock at the edge of the boundary layer.

When the compression through the shock was insufficient to establish subsonic flow behind the shock, the extent of shock-induced separation was much greater than for the cases where subsonic flow was established. This finding is in agreement with results from two-dimensional tests reported in the literature.

These two findings form the basis for the conclusion that the thickness of the initial boundary layer can have a significant effect on the extent of shock-induced separation. This effect will be most pronounced when a change in thickness is sufficient to change the flow behind the shock from wholly subsonic to transonic or vice versa.
LIST OF REFERENCES


FIGURE 1 - DISCREPANCIES BETWEEN WIND TUNNEL AND FLIGHT TEST RESULTS. REFERENCE 2.
BOUNDARY LAYER MEASUREMENT STATION AND POINT OF TANGENCY

R = 1 m

52 mm

40 mm

210 mm

96 mm

FIGURE 3 - TEST NOZZLE
FIGURE 4 - BOUNDARY LAYER PROBE

SIDE VIEW

END VIEW OF PROBE

(DIMENSIONS IN MILLIMETERS)
FIGURE 5 - BOUNDARY LAYER VELOCITY PROFILES

(a) $L = 0$ CONFIGURATION
FIGURE 5 - BOUNDARY LAYER VELOCITY PROFILES

(b) L = 150 CONFIGURATION
FIGURE 5 - BOUNDARY LAYER VELOCITY PROFILES

(C) L = 450 CONFIGURATION
FIGURE 6 - LOGARITHMIC SLOPES OF BOUNDARY LAYER PROFILES
\[ \frac{\delta^*}{\theta} \]

\[ \delta^* = \left( \frac{450 + x}{x} \right)^{\frac{4}{5}} \]

**Figure 7 - Initial Boundary Layer Conditions**
Figure 8. - Flow Visualization.

$L = 0$

$M = 1.20$

$M = 1.34$

$M = 1.39$
Figure 9. - Flow Visualization.

L = 150
(a) M = 1.18
Figure 9. - Flow Visualization.
L = 150
(b) M = 1,26
Figure 9. - Flow Visualization.

L = 150
(c) M = 1.35
Figure 9. - Flow Visualization.

$L = 150$

$(d) M = 1.43$
Figure 10. - Flow Visualization.

L = 450

(a) M = 1.16
Figure 10. - Flow Visualization.

L = 450

(b) M = 1.21
Figure 10.- Flow Visualization.

L = 450

(c) M = 1.35
Figure 10. - Flow Visualization.

L = 450
(d) M = 1.42
Figure 11 - Development of separated flow in nozzle

(a) Downstream separation only

\[ M_1 < 1 \]

(b) Bubble separation at shock and downstream separation

\[ M_1 \approx 1.2-1.3 \]

(c) Shock-induced and downstream separation regions merge

\[ M_1 > 1.3 \]
FIGURE 12 - WALL STATIC PRESSURE DISTRIBUTIONS, L=0

\[ \frac{p}{p^*} \]

X-MILLIMETERS
FIGURE 13 - WALL STATIC PRESSURE DISTRIBUTION, L = 150
FIGURE 14—WALL STATIC PRESSURE DISTRIBUTIONS, L = 450
FIGURE 15 - MODEL OF TRANSONIC SHOCK INTERACTION REGION
FIGURE 16 - PRESSURE DISTRIBUTIONS IN INTERACTION REGION
FIGURE 17 - RELATIONSHIP BETWEEN SEPARATION POINT OBSERVED IN FLOW VISUALIZATION AND POSITION OF KINK IN PRESSURE DISTRIBUTION
FIGURE 18 - EFFECT OF INITIAL BOUNDARY LAYER ON PRESSURE RISE ACROSS SHOCK
FIGURE 19 - CURVES SHOWING SIGNIFICANCE OF $\frac{p}{p_1}$