THE EFFECTS OF NOSE AND CORNER RADII ON THE FLOW
ABOUT BLUNT BODIES AT ANGLE OF ATTACK

By
James C. Ellison

Thesis submitted to the Graduate Faculty of the
Virginia Polytechnic Institute
in candidacy for the degree of
MASTER OF SCIENCE
in
AEROSPACE ENGINEERING

August 1967
THE EFFECTS OF NOSE AND CORNER RADII ON THE FLOW
ABOUT BLUNT BODIES AT ANGLE OF ATTACK

By

James C. Ellison

Thesis submitted to the Graduate Faculty of the
Virginia Polytechnic Institute
in candidacy for the degree of
MASTER OF SCIENCE
in
AEROSPACE ENGINEERING

APPROVED:

J. B. Eades, Jr.

Fred R. DeJarnette

F. J. Maher

August 1967

Blacksburg, Virginia
# TABLE OF CONTENTS

<table>
<thead>
<tr>
<th>Section</th>
<th>Title</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>I.</td>
<td>TABLE OF CONTENTS</td>
<td>2</td>
</tr>
<tr>
<td>II.</td>
<td>LIST OF FIGURES</td>
<td>3</td>
</tr>
<tr>
<td>III.</td>
<td>INTRODUCTION</td>
<td>6</td>
</tr>
<tr>
<td>IV.</td>
<td>LIST OF SYMBOLS</td>
<td>9</td>
</tr>
<tr>
<td>V.</td>
<td>APPARATUS AND TESTS</td>
<td>12</td>
</tr>
<tr>
<td></td>
<td>Test Facility</td>
<td>12</td>
</tr>
<tr>
<td></td>
<td>Operating Procedure</td>
<td>12</td>
</tr>
<tr>
<td></td>
<td>Test Conditions</td>
<td>13</td>
</tr>
<tr>
<td></td>
<td>Models</td>
<td>13</td>
</tr>
<tr>
<td></td>
<td>Instrumentation</td>
<td>14</td>
</tr>
<tr>
<td>VI.</td>
<td>DATA REDUCTION</td>
<td>15</td>
</tr>
<tr>
<td></td>
<td>Reduction of Pressure Data</td>
<td>15</td>
</tr>
<tr>
<td></td>
<td>Reduction of Shock Data</td>
<td>16</td>
</tr>
<tr>
<td>VII.</td>
<td>THEORETICAL TECHNIQUES</td>
<td>17</td>
</tr>
<tr>
<td></td>
<td>Calculation of the Pressure Distribution</td>
<td>17</td>
</tr>
<tr>
<td></td>
<td>Calculation of the Velocity Distribution</td>
<td>21</td>
</tr>
<tr>
<td></td>
<td>Calculation of the Velocity Gradient</td>
<td>22</td>
</tr>
<tr>
<td></td>
<td>Calculation of the Shock Detachment Distance</td>
<td>23</td>
</tr>
<tr>
<td>VIII.</td>
<td>RESULTS AND DISCUSSION</td>
<td>27</td>
</tr>
<tr>
<td></td>
<td>Comparison of Experimental and Theoretical Results</td>
<td>27</td>
</tr>
<tr>
<td></td>
<td>Effects of Model Geometry on Pressure and Velocity</td>
<td>29</td>
</tr>
<tr>
<td></td>
<td>Effects of Model Geometry on Velocity Gradient</td>
<td>30</td>
</tr>
<tr>
<td></td>
<td>Effects of Angle of Attack</td>
<td>31</td>
</tr>
<tr>
<td>IX.</td>
<td>CONCLUDING REMARKS</td>
<td>34</td>
</tr>
<tr>
<td>X.</td>
<td>ACKNOWLEDGMENTS</td>
<td>36</td>
</tr>
<tr>
<td>XI.</td>
<td>REFERENCES</td>
<td>37</td>
</tr>
<tr>
<td>XII.</td>
<td>VITA</td>
<td>41</td>
</tr>
</tbody>
</table>
II. LIST OF FIGURES

<table>
<thead>
<tr>
<th>FIGURE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Schematic of the Langley Mach 8 variable density tunnel</td>
<td>42</td>
</tr>
<tr>
<td>2. Model and flow geometry</td>
<td>43</td>
</tr>
<tr>
<td>(a) Model geometry</td>
<td>43</td>
</tr>
<tr>
<td>(b) Coordinates and flow geometry</td>
<td>43</td>
</tr>
<tr>
<td>3. Sketch and dimensions of models. Dimensions are in inches</td>
<td>44</td>
</tr>
<tr>
<td>(a) K = 0.707</td>
<td>44</td>
</tr>
<tr>
<td>(b) K = 0.417</td>
<td>45</td>
</tr>
<tr>
<td>(c) K = 0</td>
<td>46</td>
</tr>
<tr>
<td>4. Comparison of experimental and theoretical pressure and velocity distributions</td>
<td>47</td>
</tr>
<tr>
<td>(a) K = 0.707</td>
<td>47</td>
</tr>
<tr>
<td>(b) K = 0.417</td>
<td>48</td>
</tr>
<tr>
<td>(c) K = 0</td>
<td>49</td>
</tr>
<tr>
<td>5. Effects of model geometry on pressure</td>
<td>50</td>
</tr>
<tr>
<td>(a) $R_c = 0.8$</td>
<td>50</td>
</tr>
<tr>
<td>(b) $R_c = 0.4$</td>
<td>50</td>
</tr>
<tr>
<td>(c) $R_c = 0$</td>
<td>50</td>
</tr>
<tr>
<td>6. Effects of model geometry on velocity</td>
<td>51</td>
</tr>
<tr>
<td>(a) $R_c = 0.8$</td>
<td>51</td>
</tr>
<tr>
<td>(b) $R_c = 0.4$</td>
<td>51</td>
</tr>
<tr>
<td>(c) $R_c = 0$</td>
<td>51</td>
</tr>
<tr>
<td>7. Effects of bluntness parameter on velocity gradient</td>
<td>52</td>
</tr>
<tr>
<td>8. Effects of corner radius on velocity gradient</td>
<td>53</td>
</tr>
</tbody>
</table>
FIGURE

9. Effect of angle of attack on pressure .......................... 54
   (a) Model 2-0 ........................................... 54
   (b) Model 2-4 ........................................... 54
   (c) Model 2-8 ........................................... 55
   (d) Model 4-0 ........................................... 55
   (e) Model 4-4 ........................................... 56
   (f) Model 4-8 ........................................... 56
   (g) Model F-0 ........................................... 57
   (h) Model F-4 ........................................... 57
   (i) Model F-8 ........................................... 58

10. Effects of nose bluntness and corner radius at angles of
     attack on pressure ........................................... 59
    (a) Effect of nose bluntness, $R_c = 0$ .......................... 59
    (b) Effect of corner radius, $K = 0.707$ ........................ 59

11. Location of stagnation point at angles of attack ............. 60
    (a) $K = 0.707$ and $K = 0.417$ (flagged symbols) .......... 60
    (b) $K = 0$ ............................................. 60

12. Detachment distance measured in vertical plane of
     symmetry ...................................................... 61
    (a) Model 2-0 ........................................... 61
    (b) Model 2-4 ........................................... 61
    (c) Model 2-8 ........................................... 62
    (d) Model 4-0 ........................................... 62
    (e) Model 4-4 ........................................... 63
    (f) Model 4-8 ........................................... 63
<table>
<thead>
<tr>
<th>FIGURE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>(g) Model F-0</td>
<td>64</td>
</tr>
<tr>
<td>(h) Model F-4</td>
<td>64</td>
</tr>
<tr>
<td>(i) Model F-8</td>
<td>65</td>
</tr>
</tbody>
</table>

13. Typical schlieren photographs

<table>
<thead>
<tr>
<th>FIGURE</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>(a) Model 2-0</td>
<td>66</td>
</tr>
<tr>
<td>(b) Model 4-0</td>
<td>67</td>
</tr>
<tr>
<td>(c) Model F-0</td>
<td>68</td>
</tr>
<tr>
<td>(d) Model 2-4</td>
<td>69</td>
</tr>
<tr>
<td>(e) Model 4-4</td>
<td>69</td>
</tr>
<tr>
<td>(f) Model F-4</td>
<td>70</td>
</tr>
<tr>
<td>(g) Model 2-8</td>
<td>71</td>
</tr>
<tr>
<td>(h) Model 4-8</td>
<td>72</td>
</tr>
<tr>
<td>(i) Model F-8</td>
<td>73</td>
</tr>
</tbody>
</table>
III. INTRODUCTION

At present, vehicles which are capable of performing missions requiring reentry into the atmosphere at hypersonic speeds are of primary interest in aerospace research. Various configurations of the lifting capsule type are being considered as reentry vehicles; however, due to their bluntness, problems related to the large aerodynamic forces and extreme heating rates acting on the vehicle's surface must be evaluated, in addition to payload and mission requirements, in order to establish design criteria. As an aid to the designer, it becomes most desirable, if not necessary, to determine the effects of vehicle geometry on these forces and heating rates.

During the last decade, considerable interest has been focused on the study of flow over blunt bodies in an attempt to alleviate the problems met with in flight by reentry and hypersonic cruise vehicles. A selected list of such studies is presented here to amplify the importance of this investigation.

The shock detachment distance and shock shape, which can be employed to calculate the flow field behind the shock, have been investigated by Ambrosio and Wortman (ref. 1) who presented an expression for the stagnation point shock detachment distance and compared it with the result of Serbin (ref. 2). Kaattari (ref. 3) has devised a technique to calculate the shock shape for very blunt bodies, having various corner radii, at angles of attack. Boison and Curtiss (ref. 4) experimentally investigated the stagnation point velocity gradient on blunt bodies in supersonic flow, and Zoby and Sullivan (ref. 5) more recently presented
the results of a study of the effects of corner radius on the
stagnation point velocity gradient. In addition, Griffith and Lewis
(ref. 6) investigated laminar heat transfer on blunted cones and
hemisphere cylinders, while Cleary (ref. 7) investigated the pressure
distribution and flow field on blunted cones. Other studies on blunt
bodies include those of Reshotko and Cohen (ref. 8) on stagnation
point heat transfer and Cooper and Mayo (ref. 9) on local pressure and
heat transfer. Although these, and other investigations (refs. 10-20),
have analyzed the flow over blunt bodies, there is insufficient data on
very blunt (blunter than a hemisphere) axisymmetric bodies - especially
at angles of attack - to provide correlations between body geometry
and flow parameters or to allow for comparisons with theoretical
results.

The present investigation incorporates a compilation of theoreti-
cal results and experimental data for a family of blunt, axisymmetric
bodies which included a range of ratios of body radius to nose radius
from 0 (flat-face) to 0.707; and ratios of corner radius to body radius
from 0 (sharp corner) to 0.4. The tests were made at a free stream
Mach number and unit Reynolds number (per foot) of approximately 8.0
and 4.10 \times 10^6, respectively, for angles of attack varied from 0^\circ to
29^\circ. Also, included in this investigation are theoretical predictions,
experimental pressure and velocity distributions, and the shock-
detachment distances for blunter-than-hemisphere and flat faced bodies.
The experimental pressure and velocity distributions are compared with
theoretical predictions, at zero angle of attack; in addition,
pressures, velocity, and shock-detachment distance data obtained at
angles of attack are presented. Where possible, correlations between model geometry and flow properties are presented.
IV. LIST OF SYMBOLS

A constant in equation (39)

a defined in Figure 2

b defined in Figure 2

C velocity gradient \((du_e/ds)\), sec\(^{-1}\)

\(C_p\) pressure coefficient, \((p - p_0)/q_\infty\)

d diameter of model

H total enthalpy, ft\(^2\)/sec\(^2\)

h static enthalpy, ft\(^2\)/sec\(^2\)

\(h_f\) incremental height of manometer fluid, in.

K bluntness parameter, \(R_B/R_N\)

k free stream density divided by density behind bow shock at body axis

M Mach number

n exponent in equation (39)

P manometer reference pressure, psia

p pressure (local surface pressure when used without a subscript)

\(q_\infty\) free stream dynamic pressure, \(\frac{1}{2} \rho_\infty V_\infty^2\)

\(R_B\) body radius

\(R_c\) corner radius, in.

\(R_N\) nose radius

\(R_S\) shock radius

r radial coordinate

\(r_b\) model base radius
\( r_{\text{max}} \) equivalent to body radius
\( r^* \) radial distance to sonic point
\( s \) coordinate along surface, measured from stagnation point
\( T \) temperature, \(^{{0}\text{R}}\)
\( TP \) distance along surface from x-axis to the point of tangency
\( U' \) velocity gradient defined by equation (27)
\( u \) velocity component in the s-direction
\( V_{\infty} \) free-stream velocity
\( v \) velocity component in the y-direction
\( x \) coordinate along body axis of symmetry
\( y \) coordinate normal to surface
\( \alpha \) angle of attack
\( \gamma \) ratio of specific heats
\( \Delta \) shock detachment distance, incremental change
\( \varphi \) angle between the x-axis and a vector normal to the surface at the stagnation point
\( \rho \) density
\( \rho_f \) density of manometer fluid
\( \theta \) angle between the free stream velocity vector and a vector normal to the surface
\( \Psi \) defined in Figure 2

Subscripts
\( e \) at the edge of the boundary layer
\( \text{stag} \) stagnation point on bow shock
\( t \) stagnation point value
\( 2 \) conditions behind normal shock
free-stream conditions

The symbols 2, 4, and F have been used to denote the models having a 2.83 inch nose radius, a 4.80 inch nose radius, and a flat face, respectively; the numbers 0, 4, and 8 have been used to denote models having a sharp corner, a 0.4 inch corner radius, and an 0.8 inch corner radius, respectively. Models are generally designated by a hyphenated combination of these symbols; for example, 2-8 designates a model having a 2.83 inch nose radius and an 0.8 inch corner radius.
V. APPARATUS AND TESTS

Test Facility

The experimental program was conducted in the Langley Mach 8 variable-density hypersonic tunnel at the Langley Research Center (see Fig. 1). This is an intermittent, blowdown tunnel which is equipped with a quick injection mechanism and schlieren windows. Basically, the tunnel consists of a heat exchanger, a contoured, axisymmetric nozzle, an 18 inch diameter test section, a diffuser section, and a vacuum system. The facility is equipped to accommodate pressure, force, heat transfer and oil flow tests.

Operating Procedure

The following procedure is generally used in conducting tests in the above mentioned facility: Initially, the tunnel pressure is reduced below that required for starting the tunnel by the vacuum system. Next, air from a high pressure storage field is heated to the desired stagnation temperature. Once the desired stagnation temperature and pressure are obtained a quick opening valve allows the air to expand down the nozzle into the test section. After passing through the test section, the flow is decelerated through a diffuser and exhausted into the vacuum system.

An injection mechanism is used to place the experimental model in the free stream once the desired stream conditions are established. This mechanism is located in a chamber beneath the test section. All models are strut mounted to a plate which is fastened to the injection mechanism. The models are injected into the flow after the desired
flow has been established and retracted after obtaining the data. This is accomplished by means of a pneumatic piston which operates the injection mechanism.

The models are positioned in the test section by use of a coordinate cathetometer; a graduated inclinometer can be used to set the angle of attack. The accuracy for measuring angle of attack is ±0.05°.

For the pressure tests, at angles of attack, a test run is made and the manometer levels observed. When subsequent runs are made the angle of attack is adjusted until the pressures at the orifices - immediately adjacent to (approximately 0.150 inch) the desired stagnation point - are equal. When this balanced condition has been attained the procedure is repeated in order to obtain additional stagnation point locations. On the average, four runs are required to define the correct angle of attack for a desired stagnation point.

Test Conditions

All tests for this investigation were conducted in air at an average stagnation temperature of 1460° R, and for a stagnation pressure of 1000 psig. The corresponding free stream Mach number and unit Reynolds number (per foot) were 8.0 and $4.10 \times 10^6$, respectively. Run times of approximately 90 seconds were obtained in the tunnel; this allowed the columns of manometer fluid to reach equilibrium.

Models

During the investigation a family of blunt, axisymmetric bodies having arbitrary nose and corner radii, any one of these might be
considered as a reentry vehicle, were studied. Shown in Figure 2 is the model geometry, and in Figure 3 sketches and dimensions of the models are shown. In this program a total of nine models were tested, representing nine configurations. All models were 4.0 inches in diameter.

The models were constructed from 347 stainless steel and had 0.100-inch walls. Pressure orifices were formed as the inside diameter (0.040 inch) of a stainless steel tube which had been soldered into holes in the model and cut off flush with the model's outer surface. Orifices were located along the axes of symmetry, approximately 0.150 inch apart.

Instrumentation

A dial gage and a chromel alumel total temperature probe were used to measure the stagnation pressure and temperature, respectively. The surface pressures on the models were measured on one mercury and two butyl phthalate manometers. Two Wallace and Tiernan (dial) gages were used with a vacuum pump to regulate the reference pressures on the butyl phthalate manometers; a barometer recorded the atmospheric pressure to which the mercury manometer was referenced. From photographs of the manometer the pressure levels were read to within ±0.05 inch of fluid.

Photographs of the flow field about each model were taken during each run using a single pass schlieren system. A wire was located across one of the schlieren windows, parallel to the tunnel centerline, for reference purposes.
VI. DATA REDUCTION

Reduction of Pressure Data

From photographs of the manometer tubes the fluid levels were read to an accuracy of ±0.05 inch. All columns on the mercury manometer were referenced to the atmosphere; and, the absolute pressures were obtained by the simple expression

\[ p = h_f \rho_f \]  \hspace{1cm} (1)

where \( p \) is the pressure, \( h_f \) is the difference in height between a given column and the reference column, and \( \rho_f \) is the density of mercury. Each butyl phthalate manometer had two reference columns; these pressures were measured on the mercury manometer. To obtain the pressure levels recorded on the butyl phthalate manometers an expression similar to equation (1) was used; precisely,

\[ p = P + h_f \rho_f \]  \hspace{1cm} (2)

where \( p \) and \( h_f \) have the same definitions as before; \( P \) is the manometer reference pressure, and \( \rho_f \) is the density of butyl phthalate.

The pressure ratio \( p/p_t \) (local pressure divided by the pressure at the stagnation point) has been used to present the pressure distributions. From the pressure distributions the velocity distributions were calculated using the isentropic relation
\[
\frac{u_e}{\sqrt{2H_e}} = \sqrt{\frac{\gamma - 1}{\gamma}} \left( \frac{p_r}{p_t} \right),
\]

which is later derived as equation (25).

Reduction of Shock Detachment Distance

Schlieren photographs were taken during each test run after flow had been established over the model. From the finished prints (Fig. 13) values of \( r/r_{\text{max}} \) were determined; at these points, the distance from the body surface to the shock, along a line parallel to the x-axis, was measured. Appropriate scale factors, based on model diameter, were used to determine the actual values of the shock detachment distance. Using this technique the complete shock shapes, about each model at angles of attack, were obtained. These data are presented as plots of the dimensionless quantity \( \frac{\Delta}{d} \) versus \( \frac{r}{r_{\text{max}}} \).
VII. THEORETICAL TECHNIQUES

Calculation of the Pressure Distribution

Three methods for predicting the pressure distribution over a blunt body are presented for comparison with the experimental results: two of these methods are for bodies with a finite nose radius, and one method is for bodies with an infinite nose radius (flat face).

Newtonian impact theory expresses the local pressure coefficient in terms of the angle between the free stream direction and a vector normal to the local surface of the body. In equation form, the pressure coefficient is given by (ref. 21)

\[ C_p = 2 \cos^2 \theta. \]  

(4)

In general, the definition of the pressure coefficient is

\[ C_p = \frac{p - p_\infty}{q_\infty}. \]  

(5)

where

\[ q_\infty = \frac{1}{2} \rho_\infty V_\infty^2. \]  

(6)

Equation (4) overpredicts the pressure coefficient at the stagnation point but can be modified to yield the correct value. This is accomplished, as suggested by Lees, by replacing the constant in equation (4) with the maximum (stagnation) value of \( C_p \), defined as

\[ C_{p,t} = \frac{p_t - p_\infty}{q_\infty}; \]  

(7)
thus,

\[ \frac{C_p}{C_{p,t}} = \cos^2 \theta . \]  

(8)

From these previous expressions the ratio of the local to maximum pressure coefficient is related as

\[ \frac{C_p}{C_{p,t}} = \frac{P - P_\infty}{P_t - P_\infty} . \]  

(9)

Equating the last two equations, and manipulating the terms, yields

\[ \frac{P}{P_\infty} = \frac{P_t}{P_\infty} \cos^2 \theta + (1 - \cos^2 \theta) , \]  

(10)

which may be written as

\[ \frac{P}{P_t} = \cos^2 \theta + \frac{P_\infty}{P_t} \sin^2 \theta . \]  

(11)

The modified Newtonian pressure distribution over part of the exposed surface of a blunt body is given by equation (11); however, it is not applicable to a body of infinite nose radius (flat face) since it would suggest that the pressure is constant on the face.

Another modification of Newtonian impact theory is concerned with the correction for centrifugal force effects. In reference 21 the Newtonian-plus-centrifugal-force relation predicts the pressure coefficient on a sphere by means of the following equation:

\[ C_p = 2 \cos^2 \theta - \frac{2}{3} \sin^2 \theta . \]  

(12)

As before, it is desirable to obtain the correct value at the stagnation point; therefore, equation (12) is modified to yield
Equating equations (9) and (13) and rearranging terms will give an expression for the pressure distribution as

\[
\frac{P}{P_t} = \frac{P_{\infty}}{P_t} + \left(1 - \frac{P_{\infty}}{P_t}\right) \left(\cos^2 \theta - \frac{2}{3} \sin^2 \theta\right).
\] (14)

By manipulating the terms in the last equation, one finds that

\[
\frac{P}{P_t} = \left[\cos^2 \theta + \frac{P_{\infty}}{P_t} \sin^2 \theta\right] + \left[\frac{P_{\infty}}{P_t} \left(1 - \cos^2 \theta - \frac{2}{3} \sin^2 \theta\right) - \frac{1}{3} \sin^2 \theta\right].
\] (15)

The first quantity in brackets is identical to the right hand side of equation (11); consequently, the second quantity in brackets can be attributed to centrifugal force effects. While equation (11) predicts a pressure equal to the free stream pressure at \( \theta = 90^\circ \), it can be shown that equation (15) predicts the free stream pressure occurring at \( \theta = 60^\circ \) for a sphere. Even though equation (15) was derived on the assumption of a sphere, it is valid (in this investigation) since a majority of the models tested had spherically blunted forward surfaces.

Another technique used to predict the pressure distribution over a blunt body has been developed by Li and Geiger. The analysis begins with the equations of motion and conservation of mass for an inviscid fluid at hypersonic free stream Mach numbers. Several assumptions are made to simplify the governing differential equations (see ref. 22).
which permit their solution. These assumptions include: (1) \( r \approx s \); (2) \( y/R_B = 0 \) \( \Delta/R_B \) and can be neglected since \( \Delta/R_B \ll 1 \) for the hypersonic case; (3) the flow near the stagnation point is considered incompressible, therefore, \( \rho \approx \rho_2 = \rho_\infty/k \); (4) \( k \ll 1 \), and \( u_e/V_\infty \) and \( v/V_\infty = 0[k] \) so that higher order terms involving these terms can be neglected; (5) the analysis is applicable to the nose region (on the basis of the first assumption); and (6) \( R_N \approx R_s \). Application of these assumptions to the governing equations results in the following expression for the pressure coefficient:

\[
C_p = (2 - k) \left[ 1 - \left( s/R_N \right)^2 \right].
\] (16)

By rearranging terms and making substitutions, as in the previous analysis, the last expression becomes

\[
\frac{p}{p_t} = \frac{p_\infty}{p_t} \left[ 1 + \frac{\gamma M_\infty^2}{2} \left( 2 - k \right) \left[ 1 - \left( s/R_N \right)^2 \right] \right].
\] (17)

Equation (17) is applicable to either a two dimensional or an axisymmetric flow; however, it is limited to the stagnation region of a blunt body.

The last prediction of pressure distributions considered here is for a body with an infinite nose radius (flat face), developed by Probstein (ref. 23). This solution has been obtained by assuming incompressible irrotational flow over a circular disk, and by matching the results to a curved shock wave displaced from the body. It is also assumed that the afterbody has no effect on the flow in the stagnation region. The resulting equation for the pressure coefficient is
where \( r^* \) is the radial distance to the sonic point. Although this equation was developed for a disk with a cylindrical afterbody it may be applicable to a flat-faced body with curved or sharp corners by using the appropriate value of \( r^* \). Once again making substitutions and rearranging terms, equation (18) may be written to express the pressure distribution as

\[
\frac{p}{p_t} = \frac{p_\infty}{p_t} + \left( \frac{p_t - p_\infty}{p_t} \right) \left[ 1 - \frac{16}{9\pi^2} \left( \frac{r}{r^*} \right)^2 \right].
\]

Calculation of the Velocity Distribution

For a body in hypersonic flow the adiabatic energy equation is

\[
h_e + \frac{u_e^2}{2} = H_e.
\]

Solving for the velocity and writing the equation in dimensionless form, one finds that

\[
\frac{\frac{u_e}{\sqrt{2H_e}}}{\sqrt{2H_e}} = \sqrt{1 - \frac{h_e}{H_e}}.
\]

For a perfect gas

\[
\frac{h_e}{H_e} = \frac{T_e}{T_t};
\]

and, for a perfect gas and isentropic flow,
Substituting the last expressions into equation (21) results in the following:

$$\frac{u_e}{\sqrt{2H_e}} = \sqrt{1 - \left(\frac{p_e}{p_t}\right)^{\gamma - 1}}.$$ 

(24)

It is generally assumed that the pressure through the boundary layer along a normal to the surface is constant; thus \( p_e = p \). Therefore, the velocity distribution over the body surface may be expressed in terms of the local surface pressure by the expression

$$u_e = \sqrt{\frac{\gamma - 1}{1 - \left(\frac{p_e}{p_t}\right)^{\gamma - 1}}}.$$ 

(25)

Equation (25) is dependent on the body geometry, since the surface pressure depends on body geometry, and is used exclusively in this investigation to calculate the velocity distribution over each model tested. This is accomplished by utilizing either the experimental or a theoretical pressure distribution.

Calculation of the Velocity Gradient

The experimental velocity gradient (rate of change in velocity with distance) was approximated from the expression

$$c \approx \frac{U' \sqrt{2H_e}}{R_B}.$$ 

(26)
where,

\[
U' = \frac{\Delta(u_e \sqrt{2H_e})}{\Delta(s/R_B)},
\]

and

\[
C = \frac{du_e}{ds}
\]

In the absence of an experimental velocity distribution, an estimate of the velocity gradient may be obtained from several expressions. For instance, Newtonian flow gives (ref. 9)

\[
C = \frac{1}{R_N} \sqrt{\frac{2(P_t - P_\infty)}{\rho_t}};
\]

and, according to reference 5, the velocity gradient for a blunt body is given by

\[
C \approx \frac{1}{R_{eff}} \sqrt{\frac{2P_t}{\rho_t}}.
\]

The quantity \( R_{eff} \) is an effective nose radius defined (ref. 5) as the radius of a hemisphere which will produce the same velocity gradient as that computed for the blunt body.

**Calculation of the Shock-Detachment Distance**

Several equations are presented as a means of predicting the stagnation point shock detachment distance; all of these express the distance in terms of the density ratio across the shock. In their analysis of flow over a blunt body, Li and Geiger obtained an equation
which predicts the stagnation point shock detachment distance. The analysis is for an inviscid fluid at hypersonic Mach numbers and begins with the equations of motion and conservation of mass. Again, simplifications are made which limit the analysis to the stagnation region (see page 20) and the following equation results:

\[
\frac{\Delta_{\text{stag}}}{R_N} = k \left\{ 1 - \sqrt{1 - (1 - k)^2} \right\} \frac{1}{(1 - k)^2}^{-1}.
\]  

(31)

In addition to the assumptions of the previous analysis, that \( \Delta / R_S \ll 1 \) and \( R_N \approx R_S \), Reshotko (ref. 24) assumed: a Newtonian pressure distribution; the shock layer was of uniform (constant) thickness near the stagnation point; and, the velocity gradient was linear. After applying the conservation of mass for axisymmetric flow, one obtains

\[
\rho_\infty V_\infty \pi s^2 = \rho_e u_e (2 \pi s) \Delta,
\]  

(32)

where the velocity is given by

\[
u_e = C s;
\]  

(33)

and, the velocity gradient, \( C \), can be defined as

\[
C = \frac{V_\infty}{R_S} \sqrt{\frac{2k}{1 + k}}.
\]  

(34)

Combining the last two equations with equation (32), Reshotko obtained the following expression to predict the stagnation point shock detachment distance:
In an article by Ambrosio and Wortman (ref. 1), two expressions are presented for the prediction of the stagnation point shock detachment distance for a sphere. The first expression,

\[
\frac{\Delta_{stag}}{R_N} = \frac{k}{2} \sqrt{\frac{1 + k}{2k}}
\]  

(35)

was taken from reference 2; when this is compared to experimental data at low Mach numbers \(M < 7.0\), it is found to be less accurate than the expression,

\[
\frac{\Delta_{stag}}{R_N} = \frac{2k}{3(1 - k)}
\]  

(36)

which was derived by Ambrosio and Wortman in reference 25.

In Probststein's analysis (ref. 23) of the flow over a flat faced body the stagnation point shock detachment distance was predicted by

\[
\frac{\Delta_{stag}}{x^*} = 0.52 \left( \frac{k}{1 - k} \right)^{0.861}
\]  

(37)

which is valid for small values of \(k\), that is, hypersonic Mach numbers.

The relation

\[
\frac{R_s}{d} = A \left( \frac{x}{d} \right)^n
\]  

(38)

which is valid for small values of \(k\), that is, hypersonic Mach numbers.
has been presented in references 26-28 for predicting the shock wave shape. Values of the coefficient A, and the exponent n, have been obtained from experimental data for several shapes with various nose radii by plotting the experimental shock wave coordinates on logarithmic paper (see ref. 26). A more thorough method of predicting the shock wave shape and other pertinent properties of the shock wave has been developed by Kaattari in references 3, 29, and 30.
Comparison of Experimental and Theoretical Results

The experimental pressure and velocity distributions, at zero angle of attack, are presented as a function of \( s/R_B \) and compared to theoretical results in Figure 4. The data are presented as the pressure ratio \( p/p_t \) and the velocity parameter \( u_e/\sqrt{2H_e} \), which was calculated from the corresponding pressure distribution by using equation (25), plotted as a function of the parameter \( s/R_B \).

In Figure 4(a) the experimental distributions for the models with \( K = 0.707 \) are presented and compared to distributions calculated by Newtonian impact theory and the theory of Li and Geiger for a sharp cornered body. It is apparent that the modified Newtonian theory, given by equation (11), closely predicts the pressure in a region near the stagnation point while slightly overpredicting the pressure toward the corner of the model; consequently, this method underpredicts the velocity distribution. On the other side of the data curve is the modified Newtonian-plus-centrifugal-force theory, given by equation (14), which underpredicts the pressure and overpredicts the velocity by an amount comparable to that of the modified Newtonian theory. The Li and Geiger theory, given by equation (17), predicted the pressure and velocity distributions best; in fact, equation (17) only varied slightly from the data near the stagnation point and predicted the magnitude of the pressure ratio at various locations along the periphery of the body. However, the Li and Geiger theory does not
give \( p/p_t = 1.0 \) and \( u_e/\sqrt{2H_e} = 0 \) at \( s/R_B = 0 \) as Newtonian theories do; this slight deviation is due to the quantity \( (2 - k) \), in equation (16), which is not identically equal to \( C_{p,t} \).

Experimental data for the models having \( K = 0.417 \) are presented in Figure 4(b) in a form identical to Figure 4(a); however, the results are not the same. As can be seen from the figure, none of the theories provided very accurate predictions. It should be noted that the modified Newtonian-plus-centrifugal-force theory most closely predicts the distributions away from the stagnation point; this should be expected since this theory employs a correction for increasing curvature (decreasing values of \( K \)), even though it assumes the shock is on the body.

The experimental data for the flat faced models are compared to the theory of Probstein, given by equation (19), in Figure 4(c). The theory inaccurately predicted the distributions calculated for the flat faced, sharp cornered body; instead, it very closely predicted the distributions for the bodies having a corner radius. This result can most likely be attributed to the dependence of equation (19) on \( r^* \).

From the comparisons shown on Figure 4, it appears that the modified Newtonian theory may be applied with accuracy to bodies having a value of \( K > 0.7 \) (where the body approaches the shape of a sphere); and the modified Newtonian-plus-centrifugal-force theory is reasonably accurate for the range of values \( 0.40 < K < 0.7 \). The Li and Geiger theory appears to be applicable for a range of \( K \) which overlaps both of the Newtonian theories considered, and the range \( 0.5 < K < 0.75 \) seems a reasonable approximation. In applying these theories to bodies
having a corner radius, proper geometric relations beyond the point of tangency must be used; also, these predicted limits on $K$ may be extended easily, provided the application is limited to a region near the stagnation point. For a prediction of the pressure and velocity distributions over flat-faced bodies, the theory of Probstein, at first glance, appears inaccurate; however, if one realizes that a sharp corner forces the sonic point to occur at a point having nearly the same value of $r^*$ as a body with a small corner radius, the theory has merit.

Effects of Model Geometry on Pressure and Velocity

To show the influence of model geometry on the pressure distribution, at zero angle of attack, the pressure ratio $p/p_t$ has been plotted against $s/R_B$ in Figure 5. Considering bodies with the same corner radius (Figs. 5(a), 5(b), and 5(c)), it is evident that the nose radius significantly influences the pressure distribution, especially outside the stagnation point region. It is also apparent from the figure that the nose radius has the greatest effect when the corner of the body is sharpest ($R_C$ is smallest). The effects of corner radius on the pressure distributions can be seen from Figures 4 and 5. First, for a given nose radius, the pressure gradient (the change in $p/p_t$ with $s/R_B$) decreases as the corner radius decreases; secondly, the effect of increasing the corner radius is more pronounced on the blunter bodies.

In Figure 6, the dimensionless velocity parameter $u_e \sqrt{2H_e}$ (calculated from the pressure distributions of Figure 5) has been
plotted against \( s/R_B \). Figure 6 shows that at a given location - measured from the stagnation point - the magnitude of the velocity increases if: (1) the corner radius is increased for a given nose radius, and (2) the bluntness parameter is increased for a given corner radius. It is important to note the region where the velocity changes linearly with distance from the stagnation point, since it has been suggested (ref. 21) that the magnitude of the inviscid velocity gradient determines the local heating rate. The figure indicates that the region of a linear change in velocity with distance from the stagnation point decreases if the corner radius or the nose radius (a decrease in \( K \)) is increased.

Effects of Model Geometry on Velocity Gradient

The slopes, at the stagnation point, of the curves in Figure 6 have been determined and the velocity gradients calculated using equation (26). These velocity gradients have been plotted against the bluntness parameter, \( K \), and the dimensionless quantity \( R_c/R_B \) in Figures 7 and 8, respectively.

The results in Figure 7 imply that the stagnation-point velocity gradient varies linearly (except when \( R_c/R_B \) approaches zero on a flat-faced body) with the bluntness parameter and increases as \( K \) increases. Also, the value of \( C \) for a sphere is predicted if the curves are extrapolated to a value of \( K \) equal to unity. Furthermore, the value of \( C \) obtained by this procedure differs from the value predicted by simple Newtonian theory, equation (29), by less than 7 percent.
As shown in Figure 8, an increase in corner radius causes an increase in the value of $C$. The rate of change in $C$ with respect to $R_c/R_b$ is linear; however, this rate of change is considerably less than the change in $C$ with respect to $K$. Figure 8 also shows that the corner radius affects the velocity gradient more as the bluntness parameter decreases.

Effects of Angle of Attack

Pressure distributions.- Shown in Figure 9 are pressure distributions which have been measured on the windward side ($\psi = 180^\circ$) of each model at several angles of attack. The distributions for $\alpha = 0^\circ$ are shown in Figures 4 and 5, and have been discussed previously. As Figure 9 shows, angle of attack significantly affects the pressure distribution and consequently the velocity distribution. (See equation (25).) It is quite apparent that as the nose or corner radius increases, the pressure gradient (the change in $p/p_\infty$ with distance along the surface) increases with increasing angle of attack. Several additional observations may be made by recalling the effect of decreasing pressure on the velocity and heating rates. Realizing that as the stagnation point approaches the corner of a body (increasing $\alpha$) the pressure gradient becomes much larger; thus, there is an increase in the velocity and the heating rate. A second interesting observation, which can be concluded from Figure 9, is that in many cases the flow over certain portions of the body is no
longer subsonic; and, consequently, many of the assumptions made for
the analysis of the flow over the body at \( \alpha = 0^\circ \) can no longer be
applied.

Presented in Figure 10 are several of the pressure distributions
taken from Figure 9 to show more vividly the effects of nose and
corner radii on the flow when the body is at angle of attack. Figure
10(a) shows the trend in pressure distributions for a given corner
radius (sharp corner) as the bluntness of the nose varies. It is
evident that the nose radius (or bluntness parameter) greatly
influences the pressure distribution; and, as shown in the figure,
an increase in nose radius (a decrease in \( K \)) tends to yield a
smaller varying pressure. In the case of a flat faced body a nearly
constant pressure exists, even at fairly large angles of attack. From
Figure 10(b) it may be concluded that the effect on pressure due to
corner radius is less significant than the effect due to nose radius.
However, here an unusual trend is indicated. By comparing
Figures 9(a) - 9(c), it can be seen that at \( \alpha = 0^\circ \) the magnitude
of the pressure decreases over the surface if the corner radius is
increased; while Figure 10(b) shows that the magnitude of the pressure,
at a given location from the stagnation point, increases if the corner
radius is increased (from 0.4 inch to 0.8 inch). Before attempting to
draw any definite conclusions, it must be pointed out that the angle
of attack is not identical in both tests.

**Stagnation point location.** On Figure 11 the stagnation point
location, in terms of the angle \( \theta \) (or \( s/R_B \)) measured from the
stagnation point to the x-axis, has been plotted against the angle of
attack. Due to the manner in which this data were obtained the accuracy may not be precise; however, the trend is indicated. Figure 11 shows that the stagnation point location varies linearly with angle of attack, over most of the range of $\alpha$. In general, the change in stagnation point location with angle of attack was less on models having the larger nose and corner radii.

Shock detachment distance.- Shown in Figure 12 are the shock shapes for all of the models at angle of attack, presented as the nondimensionalized shock detachment distance ($\Delta/d$) plotted against $r/r_{\text{max}}$. The figure indicates that the detachment distance is generally smaller on the windward side and larger on the leeward side when the body is at angle of attack. At zero angle of attack, Figures 12(a) - 12(f) show that the value of $\Delta/d$ (at $r/r_{\text{max}} = 0$) for the models with a finite nose radius decreases only slightly if the corner radius is increased, while Figures 12(g) - 12(i) show that the value decreases significantly if the corner radius is increased on a flat faced model. A significant change in $\Delta/d$ occurs when the nose radius is increased; as shown in Figure 12, the value of $\Delta/d$ (at $r/r_{\text{max}} = 0$) changes from 0.10 to 0.25, approximately, as the bluntness parameter decreases from 0.707 to 0.
IX. CONCLUDING REMARKS

An investigation was conducted to determine the effects of the nose and corner radii on the flow about blunt bodies at angle of attack. Experimental pressure distributions were measured on a family of blunt, axisymmetric bodies at angles of attack from $0^\circ$ to $29^\circ$. The experimental data were obtained at a free stream Mach number and unit Reynolds number (per foot) of approximately $8.0$ and $4.10 \times 10^6$, respectively.

The experimental pressure distributions (at $\alpha = 0^\circ$), and velocity distributions calculated from them, were compared to several existing theories. The Li and Geiger theory and Newtonian theories used predicted the distributions very closely on the models with $K = 0.707$; however, these theories were less accurate for the models with $K = 0.417$. Probstein's theory, used for predicting the distributions on the flat faced, sharp cornered model, was found to be inadequate; although, it is expected that the theory would be applicable to flat faced models with a corner radius.

Results showed that the pressure and velocity distributions were significantly affected by the nose bluntness; and, while there was an effect due to corner radius, it was less noticeable than the effect of nose radius. Also, a decrease in the bluntness parameter ($K$) or the corner radius increased the magnitude of the pressure ratio $p/p_t$ over the surface of the model.

Stagnation point velocity gradients obtained from the experimental data showed a dependence on nose and corner radii; in fact, the value
of C (the stagnation point velocity gradient) on the model with a sharp corner and $K = 0.707$ was about two and one-half times the value for the flat faced sharp cornered model. Results further indicated that for a given corner radius, the value of C decreased when the bluntness parameter (K) decreased; and, for a given nose radius, the value of C increased when the corner radius increased.

Data obtained at angles of attack showed that the nose and corner radii affected the pressure distribution and stagnation point location. The magnitude of $p/p_t$ increased on all models tested as the angle of attack was increased, and this increase generally became more pronounced when the nose or corner radius was increased. Also, the change in stagnation point location with angle of attack ($d\varphi/d\alpha$) increased, and approached a value of unity, as the bluntness parameter (K) increased.

Results obtained from the schlieren photographs revealed that the stagnation point shock detachment distance increased with decreasing corner radius or increasing nose radius.
X. ACKNOWLEDGMENTS

The author wishes to express his appreciation to the National Aeronautics and Space Administration for the permission to use the data and material contained in this thesis which was obtained from a research project conducted at the Langley Research Center.

For their assistance and advice during the experimental phase of this thesis, the author wishes to thank and

The author wishes to express his thanks to Dr. Fred DeJarnette for his assistance in preparing this thesis.
XI. REFERENCES


9. Cooper, Morton; and Mayo, Edward E.: Measurements of Local Heat Transfer and Pressure on Six 2-Inch-Diameter Blunt Bodies at a Mach Number of 4.95 and at Reynolds Numbers Per Foot up to $81 \times 10^6$. NASA MEMO 1-3-59L, 1959.


17. Katzen, Elliott D.; and Kaattari, George E.: Flow Around Blunt Bodies Including Effects of High Angles of Attack,


The vita has been removed from the scanned document
Figure 1. Schematic of the Langley Mach 8 variable density tunnel.
(a) Model geometry.

(b) Coordinates and flow geometry.

Figure 2.- Model and flow geometry.
Figure 3.- Sketch and dimensions of models.
Dimensions are in inches.

(a) K = 0.707.
Model 4-0

\[
\begin{align*}
R_N &= 4.80 \\
R_c &= 0 \\
b &= 1.312 \\
r_b &= 1.597
\end{align*}
\]

Model 4-4

\[
\begin{align*}
R_N &= 4.80 \\
R_c &= 0.4 \\
a &= 0.701 \\
b &= 1.558 \\
r_b &= 1.642 \\
TP &= 1.775
\end{align*}
\]

Model 4-8

\[
\begin{align*}
R_N &= 4.80 \\
R_c &= 0.8 \\
a &= 0.824 \\
b &= 1.858 \\
r_b &= 1.680 \\
TP &= 1.454
\end{align*}
\]

(b) \( K = 0.417 \).

Figure 3.- Continued.
Model F-0
Flat face
$R_c = 0$
$b = 1.263$
$r_b = 1.496$

Model F-4
Flat face
$R_c = 0.40$
$a = 0.40$
$b = 1.344$
$r_b = 1.600$
$TP = 1.600$

Model F-8
Flat face
$R_c = 0.80$
$a = 0.80$
$b = 1.673$
$r_b = 1.676$
$TP = 1.200$

(c) $K = 0$.

Figure 3.- Concluded.
Figure 4.- Comparison of experimental and theoretical pressure and velocity distributions.

(a) $K = 0.707$. 
$\frac{u_e}{\sqrt{2H_e}}$

$b) \ K = 0.417.$

Figure 4.- Continued.
Figure 4. - Concluded.
Figure 5. - Effects of model geometry on pressure.
Figure 6.- Effects of model geometry on velocity.
Figure 7.- Effects of bluntness parameter on velocity gradient.
Figure 8.- Effects of corner radius on velocity gradient.
Figure 9. - Effect of angle of attack on pressure.
Figure 9.- Continued.

(c) Model 2-8.

(d) Model 4-0.
(e) Model 4-4.

(f) Model 4-8.

Figure 9.- Continued.
(g) Model F-0.

(h) Model F-4.

Figure 9.- Continued.
(1) Model F-8.

Figure 9.- Concluded.
Figure 10.- Effects of nose bluntness and corner radius at angles of attack on pressure.
Figure 11.- Location of stagnation point at angles of attack.
Figure 12.- Detachment distance measured in vertical plane of symmetry.
Figure 12.- Continued.
Figure 12.- Continued.
(g) Model F-0.

(h) Model F-4.

Figure 12.- Continued.
(1) Model F-8.

Figure 12.- Concluded.
Figure 13.- Typical schlieren photographs.

(a) Model 2-0.
(b) Model 4-0.

Figure 13.- Continued.
(c) Model F-0.

Figure 13.- Continued.
Figure 13.—Continued.
(f) Model F-4.

Figure 13.- Continued.
(g) Model 2-8.

Figure 13.- Continued.
\( \alpha = 0^\circ \)

\( \alpha = 9^\circ \)

\( \alpha = 16^\circ \)

\( \alpha = 25^\circ \)

(h) Model 4-8.

Figure 13.- Continued.
Figure 13.- Concluded.

(i) Model F-8.
THE EFFECTS OF NOSE AND CORNER RADII ON THE FLOW
ABOUT BLUNT BODIES AT ANGLE OF ATTACK

By
James C. Ellison

ABSTRACT

An investigation has been conducted to determine the effects of nose and corner radii on the pressure distribution and shock geometry for blunt bodies at angle of attack. Experimental pressure distributions were measured on a family of blunt, axisymmetric bodies which included a range of ratios of body radius to nose radius from 0 (flat face) to 0.707 and ratios of corner radius to body radius from 0 (sharp corner) to 0.4. The tests were made at a free stream Mach number and a unit Reynolds number (per foot) of approximately 8.0 and $4.10 \times 10^6$, respectively. The range of angle of attack was $0^\circ$ to $29^\circ$.

Included in this investigation are pressure and velocity distributions, shock detachment distances measured from schlieren photographs, and a comparison of the experimental pressure and velocity distributions (at zero angle of attack) with theoretical predictions.